VOLUME I

MARS OBSERVER Mission Failure Investigation Board Report

A report to the Administrator, National Aeronautics and Space Administration on the investigation of the August 1993 mission failure of the MARS Observer spacecraft.

Submitted by the
MARS Observer Mission Failure
Investigation Board

31 DECEMBER 1993

TABLE OF CONTENTS
VOLUME I

PART A

TRANSMITTAL LETTER

SIGNATURE PAGE

LIST OF MEMBERS, ADVISORS, OBSERVERS AND OTHERS

DEPARTMENT OF THE NAVY
NAVAL RESEARCH LABORATORY
WASHINGTON. D.C. 20375

IN REPLY REFER TO:
1001/002
5 Jan 1994

Mr. Daniel S. Goldin
Administrator
National Aeronautics and Space Administration
400 Maryland Avenue, S.W.
Washington, D.C. 20546

Dear Mr. Goldin,

In response to your letter of 10 September 1993, a Mars Observer Mission Failure Investigation Board was established to review the circumstances that may have contributed to the loss of communications with the Mars Observer spacecraft in August 1993. The names and signatures of Board members are shown on the following pages.

Enclosed is the report of the Mars Observer Mission Failure Investigation Board, which consists of three volumes:

Volume I:	Report text
Volume II:	Appendices (2 books)
Volume III:	Witness Statements and Presentations

The Board wishes to extend its appreciation for the cooperation that was extended to it by the principal Mars Observer contractors, the Jet Propulsion Laboratory of the California Institute of Technology and Martin Marietta Astro Space. Without their assistance and suggestions, it would not have been possible to conduct this investigation in a timely and thorough manner.

Timothy Coffey
TIMOTHY COFFEY
Director of Research

Enclosure

Sianatures of Board Members

This report discusses the activities, investigations, findings and recommendations of the Mars Observer Mission Failure Investigation Board.

T. P. Coffey (chairman)

T. C. Betterton

M. D. Griffin

J. F. Janni

K. D. Sullivan

P. G. Wilhelm

L. B. Gray (ex-officio)

List of Members, Advisors, Observers and Others

Members

Timothy Coffey, Director of Research, Naval Research Laboratory (Chairman)

Thomas C. Better-ton, Rear Admiral, USN

Michael D. Griffin, Chief Engineer, National Aeronautics and Space Administration (NASA)

Joseph F. Janni, Chief Scientist, Air Force Phillips Laboratory (AFPL)

Kathryn D. Sullivan, Chief Scientist, National Oceanic and Atmospheric Administration (NOAA)

Peter G. Wilhelm, Director, Naval Center for Space Technology, Naval Research Laboratory (NRL)

Ex-officio Members

Leven Gray, Office of Safety and Mission Assurance, NASA (ex officio)

Gerald Hoskins, NRL (executive secretary)

Advisors. Technical Team Members. and Observers

William Panter, Program Manager - Mars Observer, NASA

Paula Cleggett-Haleim, Public Affairs Office, NASA

Christine Anderson, AFPL

William Baker, NRL

Daniel Barney, NASA

Robert Beal, NRL

Jay P. Boris, NRL

Michael Brown, NRL

Larry Burman, NRL

Robert Eisenhauer, NRL

Joe Foreman, NRL

Andrew Fox, NRL

Henry Hoffman, NASA Goddard Space Flight Center

Samuel Hollander, NRL

Nelson Hyman, NRL

Marvin Levenson, NRL

Christopher Lichtenberg, NRL

Porter Lyon, NRL

Joseph Mattaino, NRL
Wilfred Mazur, NOAA
William McIntyre, NRL
James Nichols, AFPL
William Purdy, NRL
James Rochier, Captain, USAF, Defense Meteorological Satellite Program (DMSP) Office
David Schriftman, NRL
D. Jeffrey Shortt, NRL
Robert Skalitzky, NRL
Gregg Snyder, NRL
Kurt Weiler, NRL

Report Editor
Kenneth Lackie, NRL

PART B

EXECUTIVE SUMMARY

The **Mars** Observer Mission Failure Investigation Board was established by Mr. Daniel S. Goldin, Administrator, National Aeronautics and Space Administration. The Board was charged to review, analyze, and evaluate the facts and circumstances regarding the loss of spacecraft communications and the failure of the Mars Observer mission; determine the cause of the failure; and report the results to the Administrator.

The Mars Observer program, originally named Mars Geoscience Climatology Orbiter program, was recommended and developed by the Solar System Exploration Committee of the NASA Advisory Council during the period 1981-1983. The spacecraft, orbit, and instruments were to be designed to maximize the scientific return within a modest cost framework. Given approval for a program start in fiscal year 1985, the Jet Propulsion Laboratory (JPL), acting as the implementing Field Center for NASA, was assigned responsibility for managing the program, including contracting with industry for the build and test of the spacecraft bus, acquiring the science instruments, and conducting the flight operations.

Mars Observer was launched from the Cape Canaveral Air Force Station on 25 September 1992. Both the Titan III and the Transfer Orbit Stage vehicle worked well, although the first stage of the launch vehicle suffered-a fuel-depletion shutdown during launch. The cruise phase from Earth to Mars was relatively trouble-free, with only a few anomalies noted. The first of a series of maneuvers designed to insert the spacecraft into an orbit around Mars had been planned to take place on 24 August 1993. The sequence of events leading to the first maneuver began as scheduled on 21 August. The first action in this sequence involved pressurization of the propulsion system, initiated and controlled by a sequence of software commands previously stored in the spacecraft computers.

In accordance with the mission's published flight rules, the transmitter on the spacecraft had been turned off during the propellant-tank Pressurization Sequence on 21 August; as a result, there was no telemetry during this event. No data from the spacecraft have been received since that time. This lack of telemetry has seriously hampered an unambiguous determination of the cause of the mishap. The Failure Investigation Board therefore adopted an approach that first identified technically possible failure scenarios, eliminated those deemed implausible, and then categorized the remaining scenarios as either "possible" or "most probable." These scenarios were developed for each spacecraft system. They are included in Part F of this report.

To carry out the investigation, the Board established technical teams corresponding to the major subsystems of the spacecraft. The specific technical teams established were:

- Electrical Power
- Attitude and Articulation Control
- Command and Data Handling
- Telecommunications
- Mechanical, including Propulsion
- Software.

The teams included representation from NASA, NOAA, AFPL, NRL, and the DMSP Program Office. Each technical team member was required to have significant hands-on experience in areas related to the team's assigned system.

The investigation process involved briefings to the Board and the technical teams by JPL and Martin Marietta Astro Space (MMAS) to establish a baseline understanding of the Mars Observer spacecraft and the ground system supporting its mission. Team visits were made to MMAS and JPL for detailed subsystem reviews and for the development of failure scenarios. The Board was also briefed on a range of related topics, including the NOAA-13 spacecraft failure investigation; the LANDSAT-6 satellite failure; and lessons learned and observations from the LANDSAT-7 program and from the Global Geosciences (GGS) program. Additionally, the Board and the teams closely monitored the progress of independent JPL and MMAS investigation boards.

The Board and the technical teams began by identifying credible failure modes and design weaknesses in the spacecraft. Approximately 60 scenarios were developed and assessed in terms Of:

- Consistency with observables;
- Probability of occurrence; and
- Correlation with Pressurization Sequence events.

Four stages of filtering were involved in this process. The first stage identified those failures that could lead to the immediate loss of telecommunications downlink. The second stage eliminated all random failures, since the circumstances of this mishap required a failure to have occurred during the specific 14-minute period (ten minutes during which the transmitters were off, plus four minutes of tube warm-up time) without telemetry. The third stage consisted of identifying the subset of single failures from the second stage that could lead to extended loss (hours to days) of downlink. The fourth stage was focused on the subset of the third-stage failures that could be correlated with the Pressurization Sequence, which included several commands and activities that were being executed for the first time during the mission. Those failures that survived through the fourth stage were then examined with respect to supporting test data, analyses and failure history. Specific tests and analyses were identified and performed to validate or invalidate postulated scenarios. This process permitted the Board to classify the failures as to the most probable cause and potential causes.

As a result of these studies, analyses and tests, the Board was led to three principal conclusions:

- *First Principal Conclusion*

Despite extensive analysis of the circumstances surrounding the mission failure of the Mars Observer spacecraft, the Board was unable to find clear and conclusive evidence pointing to a particular scenario as the "smoking gun." Most of the failure scenarios were determined to be implausible or extremely unlikely. The Board was, however, unable to eliminate several failure scenarios. From these remaining scenarios, the Board concluded through a process of elimination that the most probable cause of the loss of downlink from the Mars Observer was a massive failure of the pressurization side of the propulsion system. The Board also concluded that the most probable cause of that failure was the unintended mixing of nitrogen tetroxide (NTO) and monomethyl hydrazine (MMH) in the titanium tubing on the pressurization side of the propulsion system. This mixing was believed by the Board to have been enabled by significant NTO migration through check valves during the eleven-month cruise phase from Earth to Mars. This conclusion is supported (but not proven) by NTO transport-rate data acquired by JPL, by NTO/MMH reaction simulations performed by NRL, and by NTO/MMH mixing tests performed by AFPL.

- *Second Principal Conclusion*

The Board concluded that the Mars Observer spacecraft design is generally sound. The investigation did, however, identify issues (some unrelated to this failure) that should be addressed and corrected prior to any flight of the same or derivative-design spacecraft.

- *Third Principal Conclusion*

The Board concluded that, although the result was a very capable spacecraft, the organization and procedural "system" that developed Mars Observer failed in several areas. In particular, the system failed to react properly to a program that had changed radically from the program that was originally envisioned. Too much reliance was placed on the heritage of spacecraft hardware, software, and procedures, especially since the Mars Observer mission was fundamentally different from the missions of the satellites from which the heritage was derived. The complementary strengths of JPL and Martin Marietta Astro Space (formerly RCA Astro-Electronics and General Electric Astro-Space Division) were not used by NASA as effectively as they should have been.

Secondary Conclusions

In addition to its assessment of the most probable failure presented earlier, the Board found that the following failures must also be considered as potential causes of the loss of downlink:

- Electrical Power System failure resulting from a regulated power bus short circuit.

- Regulator failure resulting in NTO and/or MMH tank over-pressurization and rupture

- Ejection of a NASA Standard Initiator at high velocity from a pyro valve, puncturing the MMH tank or causing severe damage to some other spacecraft system.

The Board was generally impressed with the spacecraft that was developed for the Mars Observer mission. However, considering the potential for reflight of an identical spacecraft, or the' use of derivative designs or hardware in spacecraft currently in development or planned for future similar mission requirements, a number of specific concerns were noted:

- Propulsion System

- Inappropriate isolation mechanisms between fuel and oxidizer for an interplanetary mission.

- Lack of post-assembly procedures for verifying cleanliness and proper functioning of the propellant pressurization system
- Current lack of understanding of the differences in pyro-initiator characteristics between European Space Agency initiators and NASA Standard Initiators.
- Inadequate thermal instrumentation, control, and modeling for the mission profile.

• Electrical Power System

- Potential power bus short circuit susceptibility, due to improper assembly, single component failure, or insulation failure.

• Command and Data Handling Svstem

- Critical redundancy control functions can be disabled by a single part failure or logic upset.
- Redundant crystal oscillator (RXO) can lose one of its two outputs without remedy of fault protection.
- The actual state of the backup oscillator in the RXO is not available in telemetry.

• Software/Fault Protection

- A top-down audit of fault protection requirements, implementation, and validation is needed.

• Svstems Engineering/Flight Rules

- The flight system should be qualified and capable of providing insight into critical mission events. An example of this would be the availability of telemetry during critical events.
- The flight system should be allowed to maintain attitude control during critical operations.
- If any rebuild or modification of the spacecraft is anticipated, the documentation should be updated to reflect the as-built/as-flown configuration.

The Board noted that the Mars Observer that was built departed significantly from the guiding principles originally established for the program, yet the acquisition and management strategy remained unchanged. The role of JPL in this fixed-price procurement was, at best, cumbersome, and did not appear to make the most effective use of the unique resource represented by JPL. In any event, the use of a firm, fixed-price contract was inappropriate to the effort as it finally evolved. The original philosophy of minor modifications to a commercial production-line spacecraft was retained throughout the program. The result was reliance on design and component heritage qualification that was inappropriate for the mission. Examples of this reliance were the failure to qualify the traveling wave tube amplifiers for pyro firing shock; the design of the propulsion system; and the use of a fault-management software package that was not fully understood, The Board also noted that the discipline and documentation culture associated with, and appropriate for, commercial production-line spacecraft is basically incompatible with the discipline and documentation required for a one-of-a-kind spacecraft designed for a complex mission. Mars Observer was not a production-line spacecraft.

While the Board can find no direct linkage between the mishap and these systemic weaknesses observed in the Mars Observer program as it evolved over the years, these weaknesses, nevertheless, remain a significant concern for future programs.

The Board would like to express its appreciation for the support provided to the investigation by the six technical teams, the other NRL and AFPL personnel who supported it, the NASA representatives, the JPL Project Team and Investigation Board, and the MMAS Technical Teams.

PART C

BOARD ORGANIZATION AND METHOD OF INVESTIGATION

PART C

BOARD ORGANIZATION AND METHOD OF INVESTIGATION

Chapter Cl
Background Chronology

A detailed narrative description of the circumstances and events leading up to the loss-of downlink from the Mars Observer spacecraft is provided in Part E of this report. The following brief description is provided as background for the chronology of Investigation Board activities that will follow.

On 25 September 1992, the Mars Observer spacecraft was launched from the Cape Canaveral Air Force Station, Florida. As discussed in Part E, the 337-day cruise phase to the vicinity of Mars was relatively uneventful, with only a small number of anomalies noted in spacecraft operation. It does not appear that any of these anomalies could have contributed to the later mission failure.

On 4 August 1993, the Jet Propulsion Laboratory of the California Institute of Technology began loading the Mars Observer spacecraft controls processor (SCP) with a series of commands that the spacecraft would be called upon to execute later in August, in order to modify its trajectory for insertion into orbit around Mars. On 20 August, the last of these commands and final maneuver parameters were inserted into the SCP.

The Mars Orbit Insertion (MOI) maneuver sequence was scheduled to begin on 22 August (spacecraft time) with a series of actions associated with the pressurization of the bi-propellant fuel tanks aboard Mars Observer. To protect the spacecraft radio frequency transmitter from damage during the Pressurization Sequence (albeit a very low probability), the software included a command to turn off the Mars Observer transponder and radio frequency (RF) telemetry power amplifier for a period of ten minutes, beginning at 234:00:21 UTC (spacecraft time, equivalent to 00:21 Greenwich Mean Time [GMT] on 22 August; UTC is Universal Time Coordinated - see footnote in Chapter D1 below). This was a standard procedure that had been implemented several times earlier during the mission. Since the RF power amplifiers required about four minutes to warm up completely, a 14-minute gap in downlink telemetry was expected.

Mars Observer telemetry was observed to cease on schedule at 234:00:40 UTC, but did not reappear as scheduled at 234:00:54 UTC. No Mars Observer downlink has been observed since that time.

Chapter C2
Board Organization

In the week following the 21/22 September loss of downlink (LOD) from the Mars Observer spacecraft, when it became clear that it was not a temporary problem, Mr. Daniel S. Goldin, Administrator of NASA, contacted Dr. Timothy P. Coffey, the Director of Research at the Naval Research Laboratory, and requested that he serve as chairman of a Mars Observer Mission Failure Investigation Board. The Board was officially appointed by Mr. Goldin on 10 September 1993. The appointment letter and Board charter are attached as Appendix B of Volume II of this report.

The Board immediately requested briefings on the Mars Observer spacecraft and the events preceding the loss of downlink signal. On 8, 9, and 10 September, the Board was given a series of overview presentations by personnel from JPL, MMAS, Martin Marietta Denver, and Allied Signal Corporation. These briefings described-the spacecraft; its command and data handling, flight software, attitude control, telecommunications, electrical power, and propulsion subsystems; the sequence of steps associated with pressurizing its fuel. tanks; and a preliminary analysis of potential failure modes.

Due to the nature of the mishap (i.e., no physical evidence and no telemetry data during the failure itself), the Board focused the investigation on those spacecraft systems and events that could have caused the observed downlink failure. In order to investigate as many potential problem areas as possible in the shortest possible time, the Board decided to form six technical teams, each of which was responsible for evaluating a particular spacecraft subsystem and its possible contribution to the loss of the Mars Observer downlink. One Board member was assigned to each technical team. The teams were:

- Electrical Power System
- Attitude and Articulation Control System
- Command and Data Handling System
- Mechanical Systems (including Propulsion)
- Telecommunications Systems
- Software.

The teams included representation from NASA, NOAA, AFPL, NRL, and the DMSP Program Office. Team members were required to have significant hands-on experience in the area of their technical team assignments. The membership of each team is provided in Appendix E of Volume II of this report.

A seventh team, composed of Board members, investigated other possible LOD scenarios that were not the result of a spacecraft system failure (i.e., those caused by factors external to the spacecraft).

Part F of this report is organized in accordance with this taxonomy.

To ensure that no conflict of interest existed for any of the Government personnel assigned to the investigation, all personnel granted access to Mars Observer investigation material were required to sign a Participation Agreement provided by NASA Legal Counsel, and to have an up-to-date financial disclosure report on file.. Copies of these financial disclosure reports have been provided to NASA Legal Counsel.

The Naval Research Laboratory provided logistical and administrative support to the Board, Technical Teams, and Advisors. A conference room and associated administrative area were set aside for Mars Observer Board use. These facilities, protected by a cipher lock, were administratively staffed by NRL clerical personnel full time during normal working hours. Access to Mars Observer Board spaces was strictly controlled by color-coded badges issued specifically for the purpose. Non-Government personnel making presentations to, or meeting with, the Board were issued special visitor badges and were logged in and out at each visit.

A Mars Observer Board Archive was established to archive and control access to all documentation generated by, or provided to, the Board. Identification and tracking of archive material was accomplished by bar code. This archive will be turned over to NASA for permanent retention after completion of Board efforts. News releases were handled by NASA Public Affairs after approval by the Board Chairman.

NASA provided funding for travel by non-NASA Board members. Salary and other support for Board and technical team members was provided by their respective Government agencies. Administrative support to the Board and the production of this report were provided by the Naval Research Laboratory. A cost-accounting system was established by NRL to track and monitor all costs associated with NRL support to Board operations. All travel or visits to other activities by technical team members to obtain Mars Observer information were reported to the Executive Secretary in the form of a trip report. In addition, several interim reports were required of each technical team to keep the Board abreast of developments in between Board meetings.

Inquiries by technical teams to Mars Observer contractors were documented on an inquiry request form and submitted via specifically identified points of contact at each contractor organization.

All official Board correspondence, test results, inquiry request forms, and hard copies of material presented to the Board are bound separately as appendices to this report.

A copy of all trip reports, interim reports, Board meeting minutes, and other materials reviewed by the Board during this investigation will be turned over to NASA for archiving.

CHAPTER C3
Method of Investigation

The Mars Observer Mission Failure Investigation Board adopted a method of investigation driven by the available data. The spacecraft downlink telemetry failed to return on schedule after the MOI Pressurization Sequence. The lack of telemetry during the period of the failure left the scope of the investigation unconstrained by hard evidence. As a result, a wide net had to be flung from the outset, at first entertaining all possible scenarios that could have caused the loss of downlink. From that point, a series of assumptions were used to filter and separate the implausible, the unlikely, the potential, and finally the most probable failure scenarios. This process was applied by the technical teams and the results presented to the Board in plenary session.

As detailed in Chapters D2 and D3, random failures, i.e., those not associable with the Pressurization Sequence events were deemed implausible, due to the extremely low probability that such a failure would coincide with the 14-minute Pressurization Sequence after 11 months en route. Failure scenarios also had to be capable of causing the loss of downlink within the 14-minute period during which the telemetry was off. Scenarios associable with the Pressurization Sequence but likely to take longer than 14 minutes to cause loss of downlinkwere deemed implausible.

Next, scenarios capable of resulting in loss of downlink, but not capable of explaining the persistent inability to reestablish downlink, were eliminated. The Board felt that the persistence of the loss of downlink must be interpreted as the result of either a catastrophic spacecraft failure or a spacecraft attitude that rendered its downlink unreceivable on Earth.

The results of the technical teams' analyses of possible loss-of-downlink scenarios are provided in Part F of this Report. Laboratory tests in support of these analyses are briefly described in Part F, and are included in their entirety in Appendix Q of this report.

Chapter D3 summarizes the Board's categorization of the one most probable and three potential causes of the persistent loss of downlink from Mars Observer. Part F of this Report addresses this categorization in detail.

PART D

OVERVIEW

PART D

OVERVIEW

Chapter D1
Background

The Mars Observer program, originally named Mars Geoscience Climatology Orbiter program, was recommended by the Solar System Exploration Committee of the NASA Advisory Council during the period 1981-1983. The original spacecraft, orbit, and instruments were to be designed to maximize the scientific return within a fixed cost framework. This was to be the first spacecraft in a series with the same development philosophy. Approval was granted for a program start in fiscal year 1985, and the Jet Propulsion Laboratory, acting as the implementing Field Center for NASA, was assigned responsibility for managing the project, acquiring the science instruments, conducting the flight operations, and contracting with industry for the build and test of the spacecraft bus. JPL selected a-bus design based on that used for the Defense Meteorological Satellite Program (DMSP), manufactured by what was then RCA Astro-Electronics. This RCA component became General Electric Astro-Space Division shortly after the selection, and it was GE that built and tested the Mars Observer spacecraft, The company is today owned by Martin Marietta, and is called Martin Marietta Astro Space (MMAS).

Envisioned as a low-risk, well-bounded, first-of-a-series project for focused science, the Mars Observer mission underwent a number of significant changes during its eight-plus year development period. The majority of these changes were driven by events that were external to the project, and included funding reductions, launch vehicle uncertainty, redirection in the number and complexity of science experiments, and elimination of follow-on missions. The net result of these changes was to stretch the schedule by two years, change the launch vehicle from the Space Shuttle to a Titan III, and increase the cost by a factor of two.

These changes also had a more subtle, but possibly more serious effect. They led to frequent violations of one of the basic tenets of the program: namely that Mars Observer was simply a slightly modified for Mars Observer version of a well-proven, reliable, high-heritage-design spacecraft that would undertake a different mission. In fact, many of the spacecraft systems had been so extensively modified for Mars Observer that their heritage had been lost; others, whose heritage remained intact, should have been requalified to verify that they would function properly on an interplanetary mission of three years duration (an environment for which they were not designed). Part E of this report includes a review of program changes and their effects.

Mars Observer was launched from Cape Canaveral on 25 September 1992. Both the Titan III and the Transfer Orbit Stage (TOS) vehicles worked well (although there was an unexpected transient due to a fuel depletion shutdown of the first stage). The cruise phase from Earth to Mars was relatively trouble-free, with only a few anomalies noted. The trajectory was so close to the plan that only three of the four trajectory correction maneuvers were required to hit the Mars Orbit Insertion (MOI) aim point. An artist's concept of the planned trajectory is shown in Figure D-l. The point where telemetry was lost is indicated. In order to assist the reader, an artist's concept of the spacecraft showing the location of various antennas, the pyrotechnic valves, and the MMH tank is shown in Figure D-2.

No significant problems were encountered with any of the spacecraft hardware systems, and all scientific instrumentation had been exercised and calibrated as necessary. Problems had been encountered with the inertial reference flight software on 11 separate occasions, five of which were serious enough to put the spacecraft into Contingency Mode. However, corrections made in July 1993 to the star identification and processing software were expected to solve the problem. In any. case, the spacecraft's Redundancy Management software responded properly to the most serious of these anomalies, placing Mars Observer into Contingency Mode, with the-solar array oriented toward the Sun and telemetry switched to the low-gain antenna (LGA). It is clear that this portion of the software operated appropriately and correctly.

The series of MOI maneuvers was the single most important dynamic event of the Mars Observer mission. The first of seven planned orbit insertion maneuvers had to occur at a precise time, and had been planned to take place on 24 August 1993. The sequence of events leading to the first of these was to begin at 234:00:21 UTC*, with the execution of the Propulsion System Pressurization Block of software commands. This sequence included the firing of two normally. closed pyrotechnic valves (one at 234:00:45:04, and the second at 234:00:50:04 UTC), that would allow high-pressure gaseous helium to pressurize the nitrogen tetroxide oxidizer tank and the monomethyl hydrazine fuel tank. The engine firing would occur 68 hours later. Concern existed in the Mars Observer project team that the pyro-firing event might damage the traveling wave tube

* Universal Time Coordinated. UTC uses Julian day and Greenwich Mean Time. 234:00:21 UTC was the actual time the event was to occur, and equates to 00:21 GMT on 22 August (17:21 on 21 August in Pasadena). However, the spacecraft was so far from Earth that it took the telemetry signal that confirmed execution of the event 19 minutes to reach the NASA Deep Space Network. For clarity, therefore, ground receipt time (234:00:40 UTC in this case) will be used in this report except when otherwise specified.

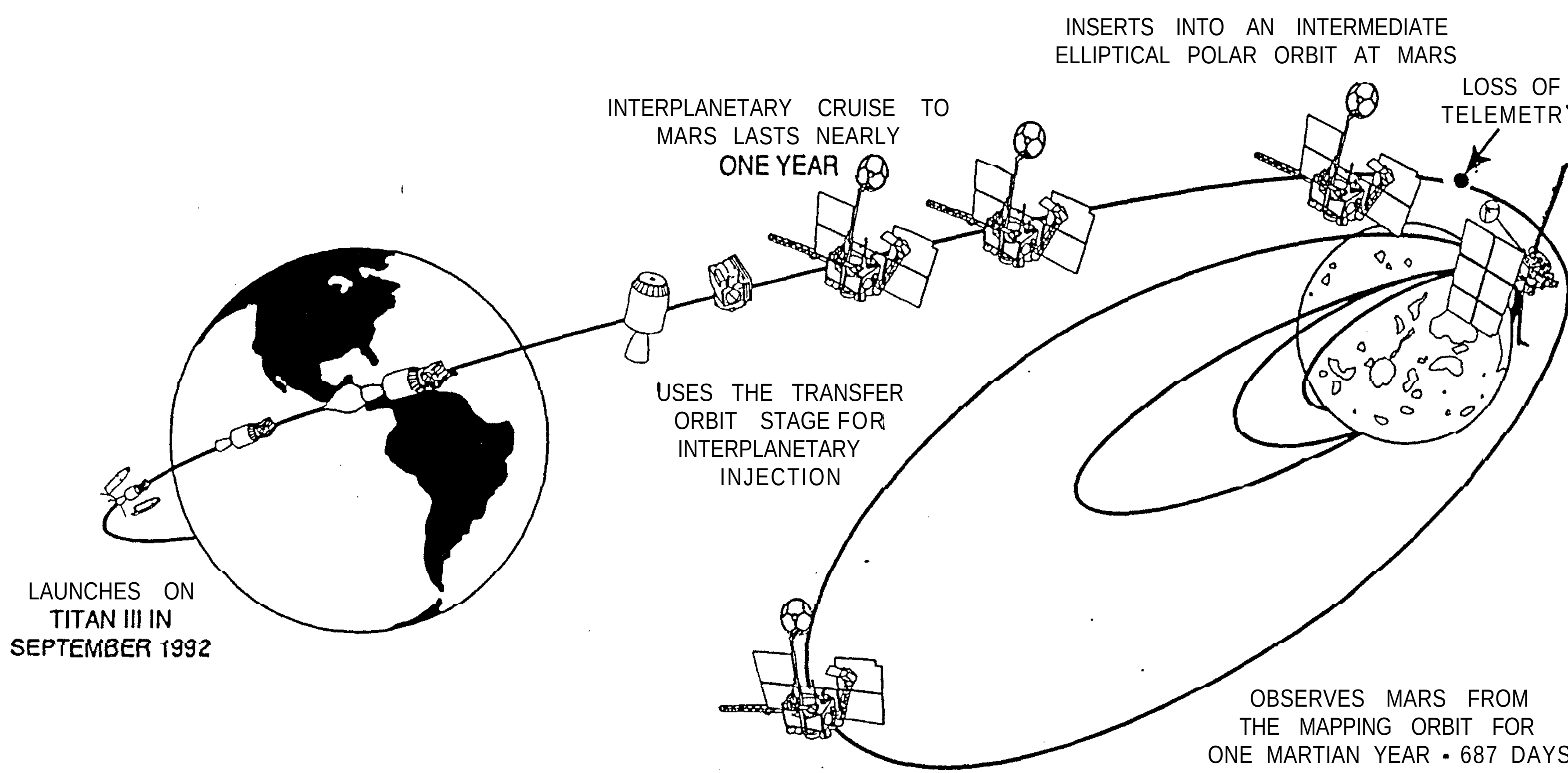


Figure D-l. Mars Observer Mission Outline

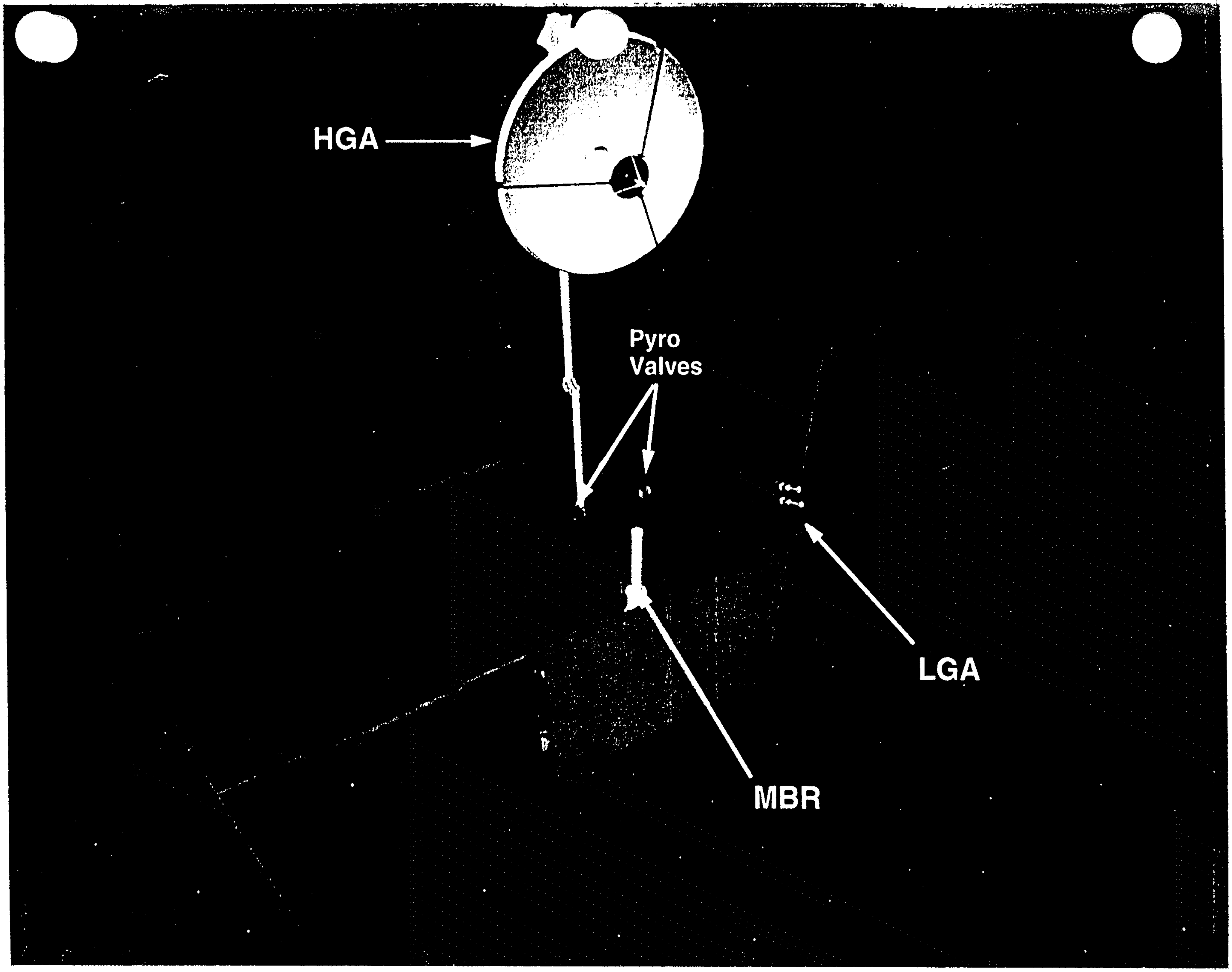


Figure D–2. Mars Observer Spacecraft

amplifiers in the spacecraft telecommunications system if the amplifiers were left on. Therefore, the spacecraft's transmitter was deliberately turned off prior to the pyro valve firing for what was to be a period of ten minutes (plus an additional four minutes of warm-up time, for a total silent period of 14 minutes). However, communications with the spacecraft were not reestablished at the expected time, nor in response to any of the numerous ground commands sent after the anomaly.

Within minutes of the failure to acquire downlink, the NASA Deep Space Network (DSN) of receivers began reconfiguring in an attempt to improve reception. At about 234:01:10 UTC (records are unclear as to the exact time), Spectral Signal Indicator (SSI) processing of the data received by the 34-m antenna at Goldstone, California began. At about 234:02:00 UTC, the larger 70-meter antenna at Canberra, Australia was brought on line; and at about 234:02:15, SSI processing began at Canberra. Nevertheless, efforts by the Mars Observer Flight Team at JPL over the following two months were unsuccessful in restoring or detecting any communications from the spacecraft on either the high-gain or low-gain antennas. The initial assumption was that an anomaly had prevented the primary spacecraft transmitter from switching on, and recovery activities were directed to that scenario.

It was also initially assumed that the MOI Sequence (already loaded into the spacecraft computers) would be executed by the spacecraft, putting it into orbit around Mars. However, it was not known whether Mars Observer's failure to communicate was due strictly to a telecommunications problem, or to a failure in other spacecraft systems. Therefore, recovery attempts (commands) were directed at both the predicted capture orbit (MOI assumed) and the fly-by point (no MOI execution).

Later in September 1993, at the request of the Failure Investigation Board, several attempts were made to turn on the small beacon transmitter in the Mars Balloon Relay (MBR) system (which is entirely separate from the spacecraft transmitter/antenna equipment) and acquire that signal through three radio telescopes. Figure D-2 shows the location of the MBR antenna. Reception of the MBR signal would have proven that the spacecraft was intact, but unable to communicate on its normal telecommunications system. These activities were without success. Unfortunately, it was not understood until this report was being written that the MBR could not be activated if the spacecraft were in the "Safe" Mode (see Chapter F5, Section a(3)). Since commands had been sent to place the spacecraft in Safe Mode, the MBR experiments that were conducted in September were not conclusive. At the time of this writing, a second attempt was being made to activate the MBR prior to conjunction of Mars Observer with the Sun. If this attempt fails (and preliminary results from this second attempt have also been negative), a third attempt should be made after solar conjunction,

This incident is significant in another way, however. The attempt to detect the MBR signals was a fairly significant effort, and involved the voluntary cooperation of a number of radio astronomy groups in the U.S. and U.K. that were not otherwise involved in Mars Observer. Yet these initial attempts (September 1993) were doomed to failure because the spacecraft almost certainly could not have responded to the commands sent by JPL to turn on the MBR transmitter. This contributed to the Boards impression that JPL did not have as deep an understanding of the spacecraft as the Board would have expected. The Board attributes this to the particular contract vehicle used to procure the spacecraft.

The lack of success of all of these recovery activities led the Board to conclude that the spacecraft most probably met with a catastrophic event on 21/22 August 1993 that terminated its mission.

CHAPTER D2
Board Analysis

The nature of this mishap is such that there is neither physical evidence nor telemetry data upon which to base an analysis of the failure. The challenge, therefore, was to find a probable cause without any direct evidence. To respond to this challenge, the Board used a step-wise, process-of-elimination approach to the problem The first step in the investigation was to focus on those systems and events that could have caused loss of the downlink signal. The Board studied all spacecraft systems and identified the failure modes that could have resulted in immediate loss of downlink. Many of these were random failures that occasionally occur even in well-designed systems - a short circuit in a high-heritage, space-qualified capacitor, for example. Though theoretically possible, the Board considered these to be of extremely low probability, since the circumstances of this mishap required the random failure to have fortuitously occurred during the specific 14-minute period (ten minutes during which the transmitters were off, plus four minutes of tube warm-up time) without telemetry. As a second step, therefore, such failures were eliminated.

Next, the Board devoted considerable attention to the antenna patterns and signal strengths in both the spacecraft and ground-based telecommunications systems in order to determine whether:

- There were spacecraft attitudes and configurations in which the downlink signal could not be received by the NASA Deep Space Network.
- A spacecraft without attitude control (but still otherwise functional) could receive commands.
- The downlink carrier from a spacecraft rotating about various axes could be detected by the NASA Deep Space Network.
- There was evidence that the Deep Space Network could have missed an intermittent, possibly weakened downlink signal radiated by a dying spacecraft in the hours immediately following the failure.

A summary of the Deep Space Network response to the Mars Observer emergency is shown in Figure D-3. In this figure, the detection limit for the high gain antenna (HGA)-to-Earth pointing angle is plotted as a function of time for ten hours after firing the first pyro valve during the Pressurization Sequence. The third step in the investigation was to eliminate those failure scenarios that were not compatible with Figure D-3.

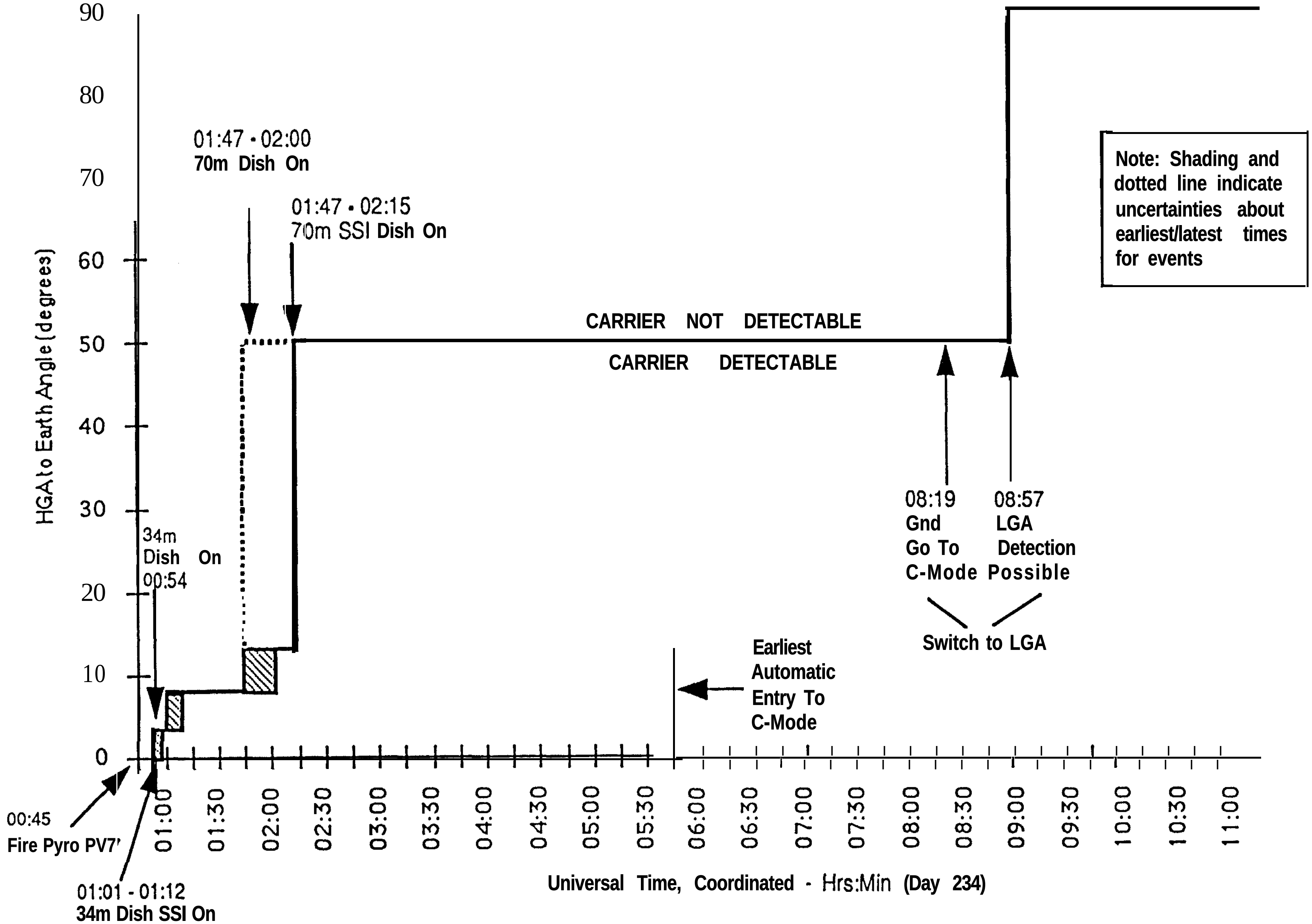


Figure D-3. High Gain Antenna Boresight-To-Earth Pointing Requirements For Carrier Signal Detection (positive angles only)

The fourth step of the Board investigation focussed more closely on those commands, actions and software that were associated with the Pressurization Sequence itself, assuming that something in that sequence triggered the failure.

The Board also attempted to identify all factors that could have contributed to such a failure, whether they were likely to be the cause of this particular failure or not.

In order to analyze the Mars Observer spacecraft in a tractable manner, the Board studied its performance characteristics on a system-by-system basis. This taxonomy, which is used in Part F of this report, was as follows:

a. Electrical Power System
b. Attitude and Articulation Control System
c. Command and Data Handling System
d. Telecommunications System
e. Mechanical Systems, including Propulsion
f. Software.

The Board then integrated the results of these analyses into a spacecraft-wide evaluation.

Despite lengthy and extensive analyses of the circumstances surrounding the mission failure of the Mars Observer spacecraft, the Board was unable to find clear and conclusive evidence pointing to a particular scenario as the "smoking gun." Most of the failure scenarios were determined to be implausible or extremely unlikely. This is based on the spacecraft design, the circumstances of the mishap (all spacecraft systems functioning nominally before the Pressurization Sequence) and the requirement that the causative mechanism had to occur and cause the loss of downlink in a particular 14-minute period after 11 months of relatively uneventful flight. All failure modes that were random and could not be related in some manner to the Pressurization Sequence fell into this category, and were eliminated from further consideration.

Several failure scenarios, however, were deemed worthy of further scrutiny. These appeared sufficiently promising to trigger further analysis, computer simulation, or testing of individual spacecraft components that were identical to those employed aboard Mars Observer. The results of these simulations and tests eliminated another group of potential causes for the loss of Mars Observer. Although all would have been damaging (and some also would have ultimately been fatal to the spacecraft), none could completely account for the immediate and persistent loss of downlink. Some trace of the downlink carrier should have been detectable, especially after the Deep Space Network had been reconfigured to improve the signal link margin, about an hour after the initial loss of downlink (see Figure D-3).

Those that remained as plausible explanations for the mission failure of Mars Observer, after the elimination of all those that could not account for all of the observables, are briefly discussed in Chapter D3 below, and in more detail in Part F. Before presenting the likely candidates, the Board feels very strongly that the following points should be made:

a. Mars Observer was healthy, with all systems operating well, until the Pressurization Sequence began executing on 21/22 August 1993.

b. The Mars Observer spacecraft was generally well designed. It did, however, contain several design errors that, even if not responsible for the loss of the spacecraft, should certainly be corrected before any consideration is given to a reflight of this mission. For example, there were several significant operational and environmental attributes of the spacecraft that were either not known or not well understood by project personnel prior to this investigation. Some of these (e.g., failure to utilize fully the redundancy in the redundant crystal oscillator) represent serious threats to spacecraft health.

c. The Pressurization Sequence somehow triggered a single, fatal malfunction in the spacecraft (either hardware or software) that very rapidly became catastrophic (e.g., loss of power, explosion, or rapid, uncontrollable spin).

e. Although a number of random potential failure modes were identified, all were dismissed as being of low probability, since the random failure would have to occur fortuitously during the particular 14 minutes without downlink.

f. Several plausible failure modes were identified that could conceivably have been triggered by the pyro shock or other events taking place during the Pressurization Sequence; however, only a small number of these could have caused the observed initial and persistent loss of downlink signal.

Chapter D3
Likely Causes of Mars Observer Mission Failure

No conclusive evidence could be found that pointed at a particular failure or series of failures as the cause for the loss of Mars Observer. A number of random, single-point failures were identified, but all were assessed by the Board as being extremely unlikely to have randomly occurred during the specific 14-minute period when telemetry was off.

Therefore, all failure scenarios selected as probable or potential meet the following criteria:

- Not random; caused or triggered by a Pressurization Sequence event or activity;
- Had to be able to cause the loss of downlink in 14 minutes or less; and
- Had to explain the continued lack of downlink - i.e., were either catastrophic to the spacecraft (e.g., explosion, loss of electrical power, rapid spin, failure of transmitter tubes, etc.), or could put the spacecraft in an attitude where the downlink signal could not be received.

Table D-l provides a summary of the -evaluation of the 59 scenarios that were examined. Only a subset of those connected with the Propulsion System and the Electrical Power System survived through the final stage.

The elimination of a failure of the redundant crystal oscillator as a potential cause of the loss of downlink is worthy of a brief discussion in order to illustrate the elimination process, and because of the early visibility that this potential failure was given. The concern for the RXO revolved around the possible failure of its Unitrode transistors, such as those that failed during the NOAA-13 satellite ground test. The Mars Observer RXO used four transistors from the same lot as those that failed on NOAA-13. Two were used to control oscillator heaters; the failure of these could not have caused a loss of downlink. The other two were used in the power supplies for the two oscillators in the RXO unit. However, telemetry indicated that both of these critical transistors were operating at the time the telemetry was turned off as part of the Pressurization Sequence. It was asserted that the primary oscillator could have failed during the Pressurization Sequence as a result of shock waves generated by firing the pyro valves. The Board noted that although the backup oscillator in the RXO was known to be operating prior to launch, its operation was never verified thereafter during the 1 l-month cruise phase from Earth to Mars. However, the Board discounted a scenario in which both RXOs failed.

When a test was run on the Verification Test Laboratory (VTL) at JPL to simulate the failure of the primary oscillator, it was discovered that the fault management software did not transfer all

TABLE D-i. **FAILURE SCENARIO** ASSESSMENT SUMMARY

SUBSYSTEM/SCENARIO	BOARD ASSESSMENT	CONDUCTED TESTING/ANALYSES
• ELECTRICAL POWER SYSTEM		
- Insulation failure (4)	N	NO
- Open wire/circuit (5)	N	NO
- Bus capacitor failure (2)	N	NO
- PSE power diode/chassis short	B	YES
• ATTITUDE & ARTICULATION CONTROL SYSTEM		
- Sun sensor # 4 failure	N	NO
- Reaction wheel failure (2)	N	NO
- IMU failure	N	YES
- Loss of primary clock reference - see RXO		
• COMMAND & DATA HANDLING SYSTEM		
- RXO failure/primary clock reference	N	YES
- Loss of computational/CIU functions (3)	N	YES
- Command/Control Logic failure (6)	N	NO
• TELECOMMUNICATIONS SYSTEM		
- Component failures (6)	N	YES
- RPA and SCU interlock failure	N	NO
- Propagation/operational problems (5)	N	NO
• PROPULSION SYSTEM		
- Regulator failed open/tank overpressure (7)	B	NO
- Pyro shock failed critical component	N	YES
- Pyro valve failure/NSI expelled (2)	B	YES
- Unintended mixing of NTO/MMH (4)	A	YES
• SOFTWARE		
- Fatal error in execution	N	NO
- Memory corruption	N	NO
Simultaneous SCP failures	N	NO
- Failure in fault management software	N	NO
• EXTERNAL CAUSES		
- Micrometeoroid impact	N	NO
- Solar effects	N	NO

A - MOST PROBABLE CAUSE   B - POTENTIAL CAUSE   N- UNLIKELY

Note: Numbers in parentheses indicate the number of failures if more than 1.

important functions to the backup system For example, the spacecraft controls processor and the inertial measurement unit (IMU) were switched to the backup oscillator, but the clock divider circuit was not automatically switched. Thus, in the event of failure of the primary oscillator, attitude control would be lost, but the RF power amplifier would still be turned on and would begin transmitting telemetry on schedule. Numerous VTL simulation runs were conducted to examine the spacecraft response to this particular failure. One of the predictions made by these simulations was of the high-gain antenna-to-Earth pointing angle. The worst case that was found is shown in Figure D-4, where the VTL simulation predictions for HGA angle as a function of time are overlaid on the HGA-boresight-to-Earth pointing requirements for signal detection, originally presented in Figure D-3. It is clearly evident from Figure D-4 that while this failed oscillator scenario could explain the early loss of telemetry, it does not explain the loss of downlink signal after approximately 1.5 hours. The Board, therefore, found that to meet the observables, both oscillators in the RXO would have to fail. Such a double failure was considered highly unlikely, Therefore, the RXO failure was eliminated as a potential cause for the loss of downlink.

a. MOST PROBABLE CAUSE: LEAKAGE OF NTO THROUGH CHECK VALVES

Fourteen scenarios were examined with respect to the Propulsion System. A simplified schematic of the pressurization side of the Propulsion System is shown in Figure D-5. In examining the Propulsion System, it was found that the NTO oxidizer tank was separated from the rest of the pressurization side of the system by two check valves; one manufactured by Futurecraft Corporation, and the other manufactured by VACCO Corporation. These valves were in series for redundancy. Since for much of the cruise the pressurization plumbing was cold, the Board proposed a scenario in which NTO migrated either in liquid or gaseous form through the check valves and condensedon the cold tubing beyond (upstream of) the check valves. This would then theoretically create a situation in which liquid NTO could mix rapidly with MMH in the pressurization lines when the Pressurization Sequence was executed. The Board requested that tests be conducted by JPL to examine the leakage of NTO through check valves identical to those used aboard Mars Observer.

The results of these tests are summarized in Figure D-6. The tests showed that a rather surprising amount of leakage of NTO could occur. An extrapolation of these test results to an 1 1-month cruise period indicates that even without any valve failure, one to two grams of NTO could have migrated through the check valves. The results also indicate that had a single failure occurred

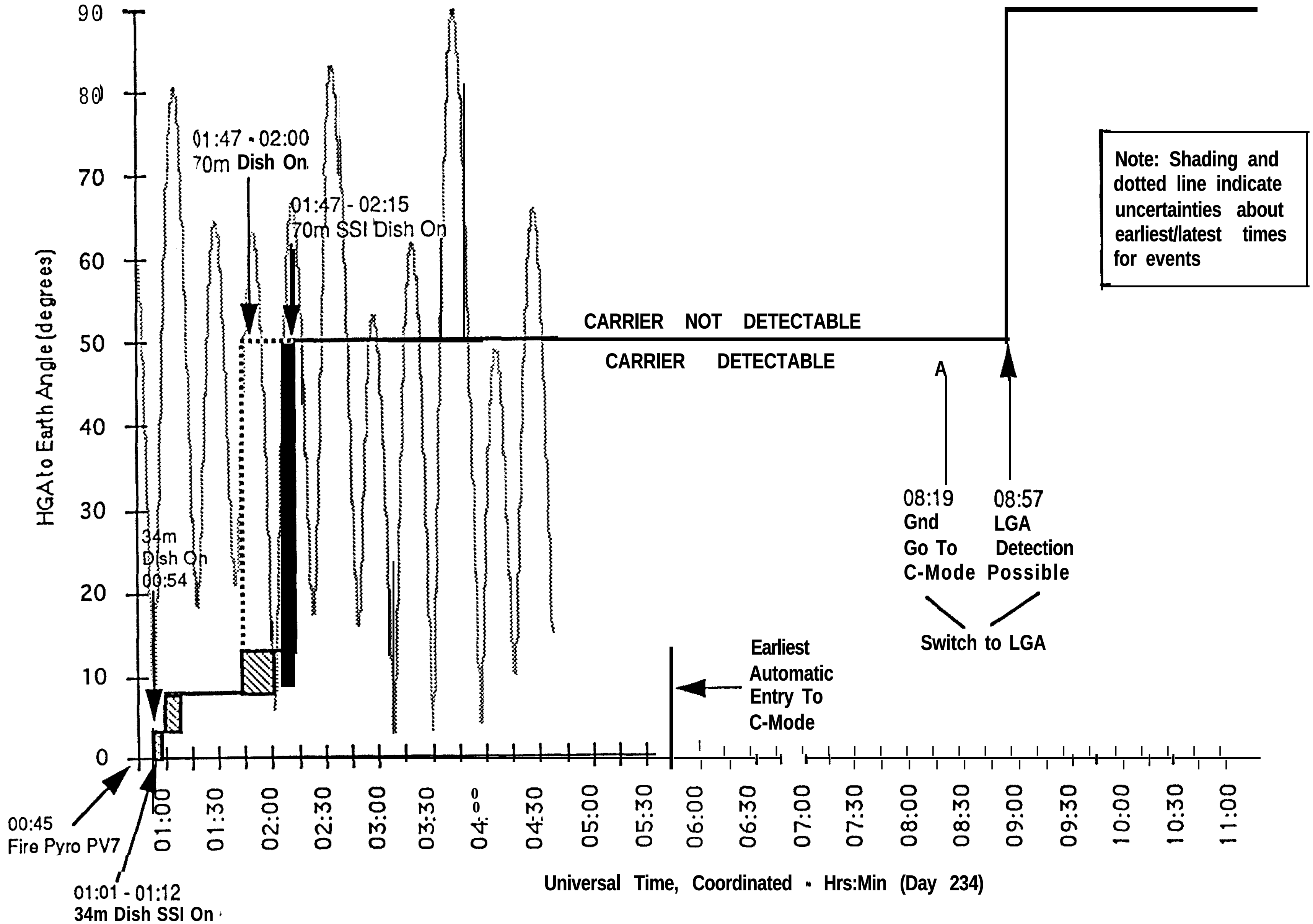


Figure D-4. Overlay of Simulated High Gain Antenna Pointing Angle and High Gain Antenna Boresight-To-Earth Pointing Requirements For Carrier Signal Detection (positive angles only)

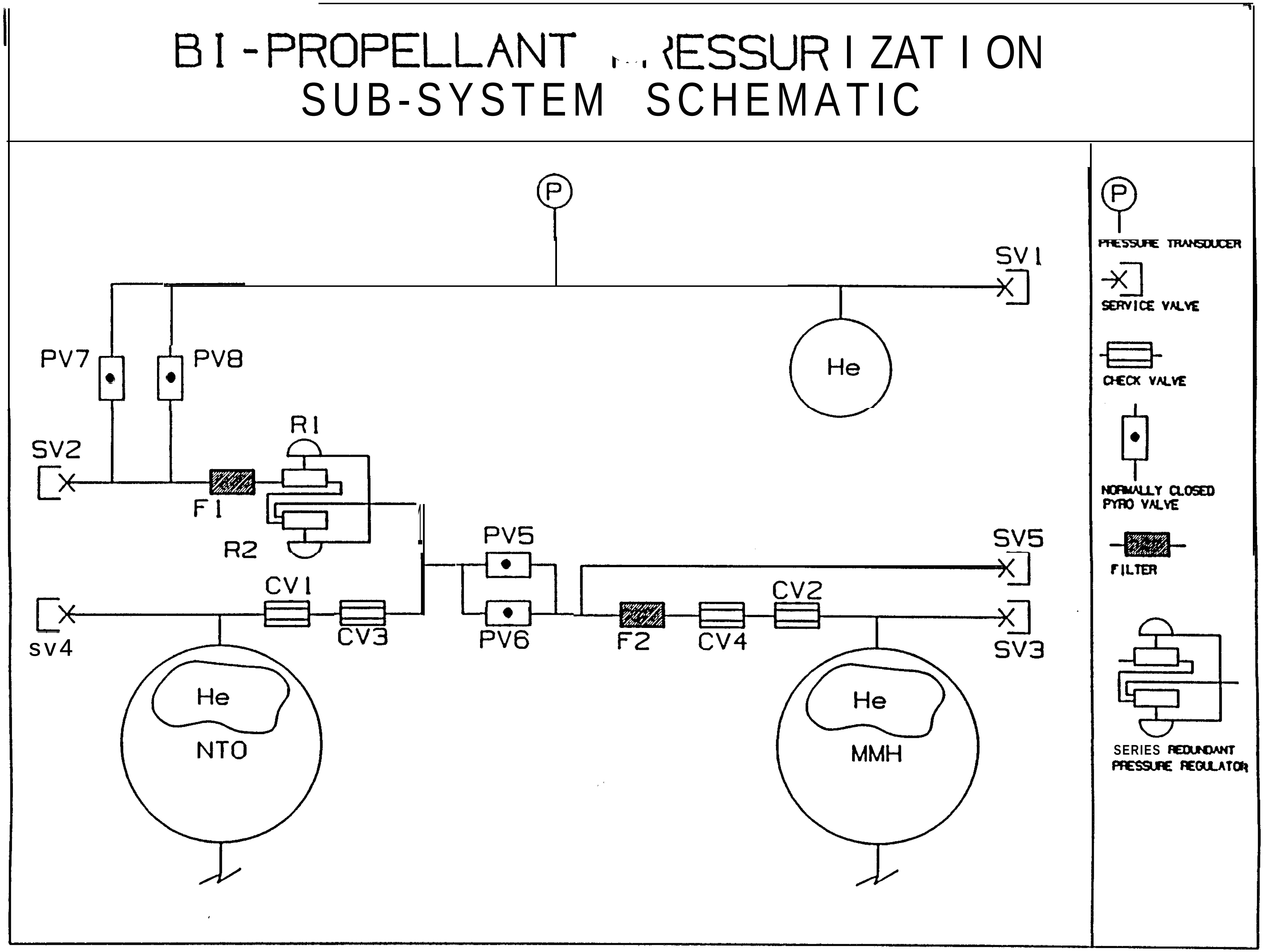


Figure D-5. Bi-Propellant Pressurization Sub-System Schematic

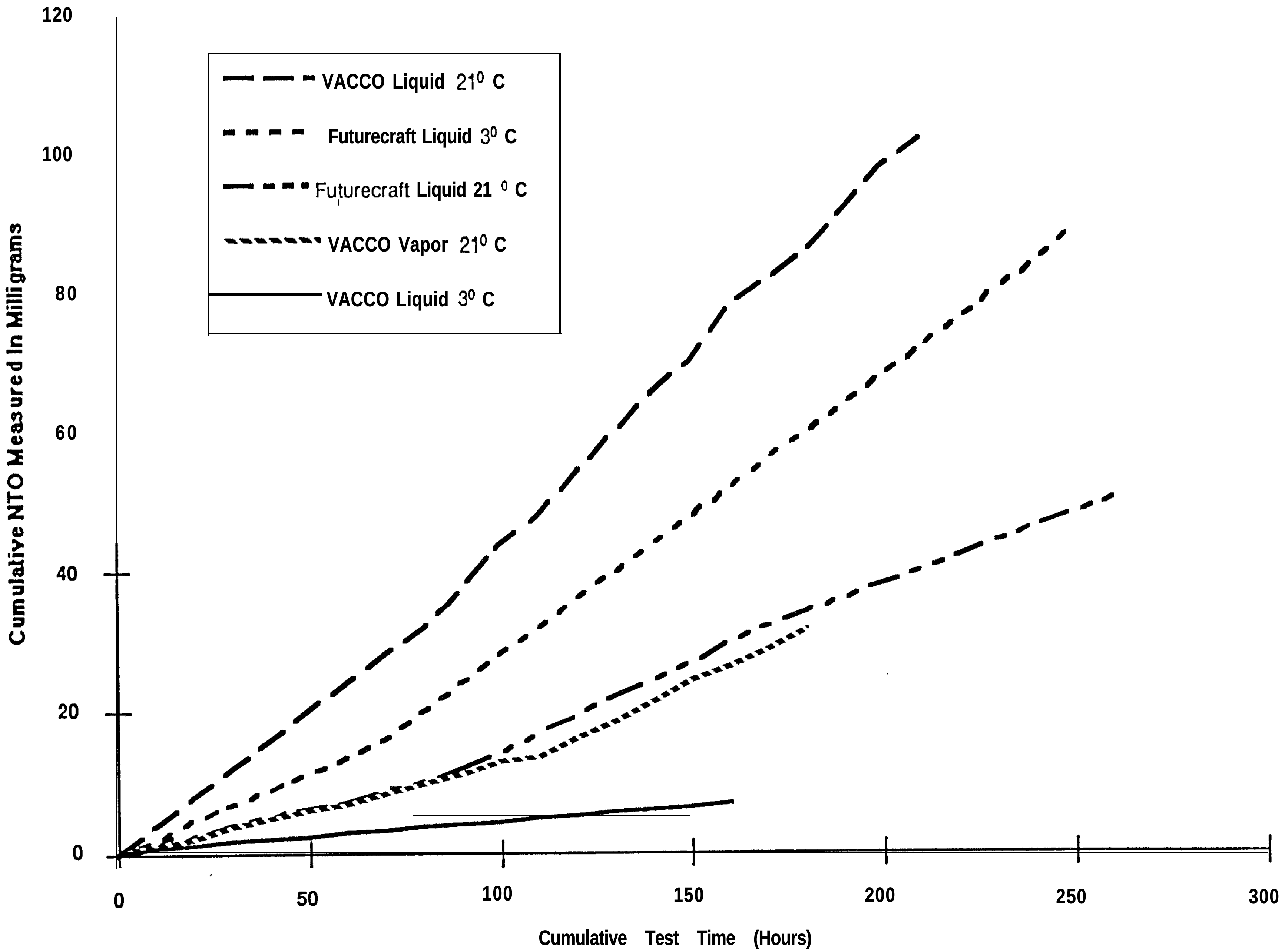


Figure D-6. Check Valve Test Results Summary

in the VACCO valve, then several grams of NTO would have leaked through the valves and condensed in the upstream plumbing. Thermal analysis of the spacecraft indicates that the **vicinity** of PV5 and PV6 would be the coldest part of the pressurization system. One would expect that the NTO would migrate to the coldest part of the system. If some of this condensed NTO were swept into the MMH lines and mixed with MMH during the Pressurization Sequence, a hypergolic reaction could occur, releasing on the order of 100 kilocalories per mole of mixed NTO and MMH.

To address the movement of NTO through the system, the sequence of events during pressurization must be considered. This discussion will refer to Figure D-5. The first pyro valve fired was PV7, which pressurized the NTO tank and in the process, would clear any NTO from the upstream line going from the helium tank to the NTO tank. It would also force additional NTO into the lines going to PV5 and PV6. The next pyro valve fired was PV5 (there remains some ambiguity as to whether it was PV5 or PV6 that was fired). This would force the NTO that was in the line upstream of PV5 through the filter F2, through check valves CV4 and CV2, and into the line between CV2 and the MMH tank. This line would be expected to be filled with MMH or at least to be wetted by MMH. The filter F2 has a 20-micron pore size and a large effective open area. Hence it would not greatly impede the flow of liquid NTO, but would atomize it into 20-micron droplets.

The issue then becomes: how much NTO would have to mix with MMH to create a serious problem? The tubing involved is 3/8-inch diameter, .015-inch thickness titanium alloy (Ti-3AL-2.5V). It requires 106 calories to raise a one-centimeter length of this tubing from 0°C to its melting temperature of 1668°C. It would take an additional 53 calories to melt the tubing. The combustion temperature of NTO/MMH is about 3000°C. NTO uniformly mixed with MMH at liquid densities is theoretically able to release about a thousand calories per centimeter length of this tubing (0.5 grams NTO, 0.3 grams MMH). The 159 calories required to melt a one-centimeter length of the tubing corresponds to burning less than one-tenth of a gram of NTO. The static melting of the tube would not actually occur like this, since the rapid generation of pressure by the combustion process would quickly force fluid dynamic motion. Also, the rate of energy release would depend on how the NTO and MMH are mixed. Nevertheless, this simple calculation illustrates that a few tenths of a gram of NTO moving into the MMH line is a matter of serious concern.

The actual situation would be much more complex than that described above. It would be a dynamic situation involving mixing, heat generation, thermal conduction, and fluid flow. . Any self-consistent solution requires numerical simulation. However, some additional insight can be gained analytically. For example, the characteristic time required to raise the temperature of the thin-walled titanium tube completely (i.e., the outer surface of the tube is the same temperature as the inner surface) is 14 milliseconds. Since thermal diffusion varies as the square root of time, the temperature at the outer surface of the tube at, for example, 1.4 ms would be about one-third of the temperature of the combusting fluid at the inside surface of the tube. (These times are referenced to the time when burning began.)

Another parameter that must be examined is the yield stress of titanium as a function of temperature. This is shown in Figure D-7 for the titanium alloy used in this application. (For ease of discussion, the yield stress has been replaced by the pressure in the tube that would produce the yield stress.) It should be noted that the yield stress of titanium declines very rapidly with temperature, losing essentially all of its strength above 500°C. It would require about 32 calories to change the temperature of a one-centimeter length of this tubing by 500°C. This much heat could theoretically be produced by burning MMH with about 20 milligrams of NTO.

One must now examine the pressure-time histories that might develop if NTO were suddenly mixed with MMH inside the titanium tubing. For the sake of calculation, consider the situation where a quantity of NTO moves through filter F2, through check valve CV2, and into the MMH line, where it rapidly and completely mixes over a distance of 5 cm and reacts with the MMH. Since the state and quantity of the MMH in this line is not known, two different conditions to represent different extremes will be assumed. In one case, it is assumed that the line contains only 10 percent MMH; in the other case, it is assumed that the line is filled with 90 percent MMH. Numerical simulations of the chemically reactive flow that would develop have been performed by NRL for each case. These simulations modeled the chemical reactions between NTO and MMH vapors; the conversion of liquid to vapor; the decomposition of MMH for temperatures above 600°K; and the compressible hydrodynamic response of the material inside the tubing. It must be emphasized that these calculations assume thorough mixing of NTO and MMH, conditions that may or may not have prevailed in the actual situation.

The predicted pressure-time history for the 10 percent MMH case on the MMH tank side of the check valve is shown in Figure D-8, for the case where two grams of NTO are injected. One observes that the pressure rises rapidly to 30,000 psi, then settles down to 12,000 psi for several milliseconds. Figure D-9 shows the corresponding temperature-vs.-time history for several milliseconds. Both figures also show the effects of reducing the NTO to 0.2 grams and 0.02 grams.

Figure D-10 shows the pressure-time profile just behind the check valve (CV2) for the case where two grams of NTO are injected into a tube that is 90 percent filled with MMH. Here, because pressure relief is tamped by the MMH, pressure rises rapidly to 70,000 psi and then falls

YIELD PRESSURE VS TEMPERATURE FOR TITANIUM & TITANIUM ALLOYS

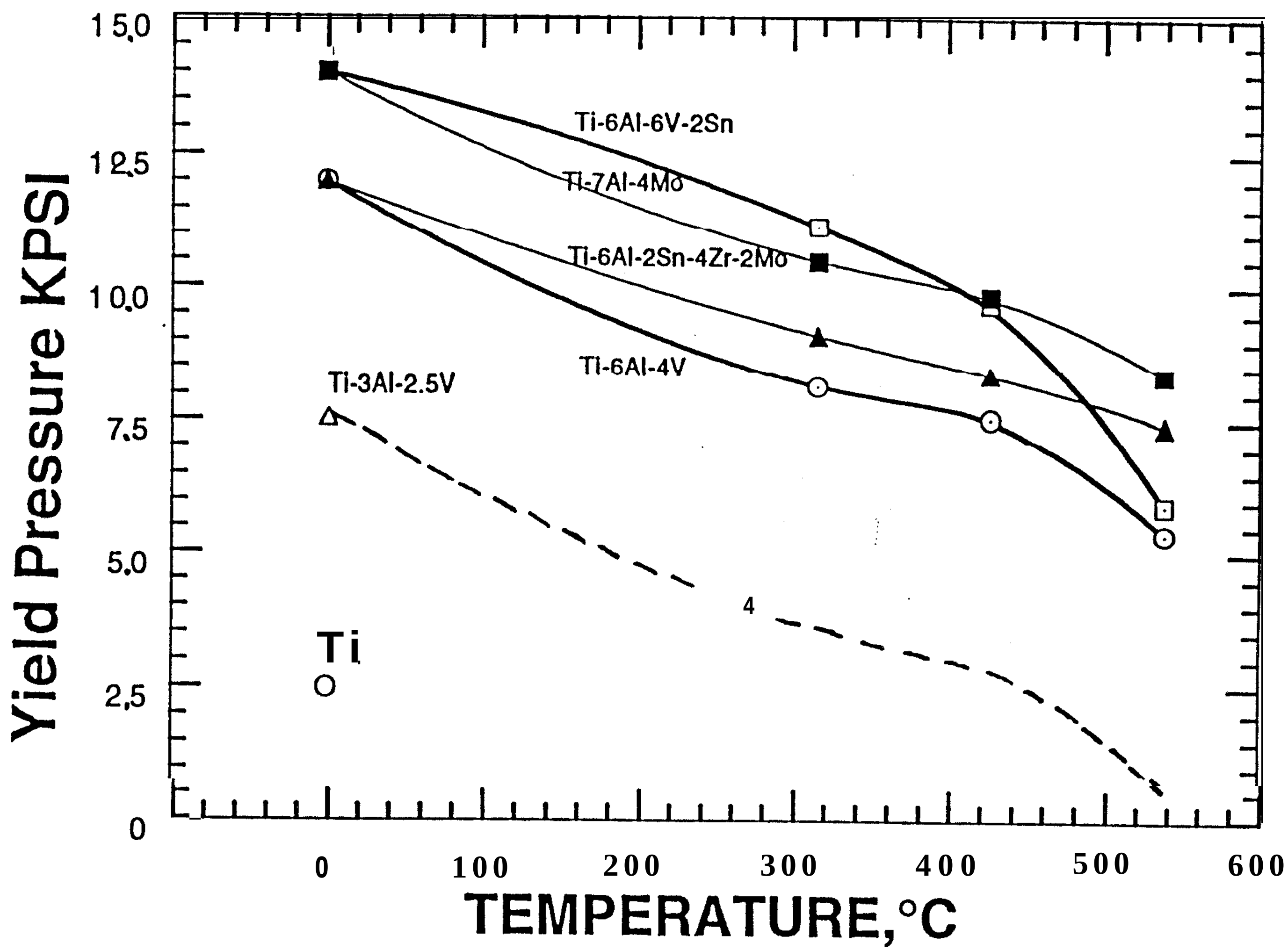


Figure D-7. Yield Pressure for Titanium and Titanium Alloys as a Function of Temperature

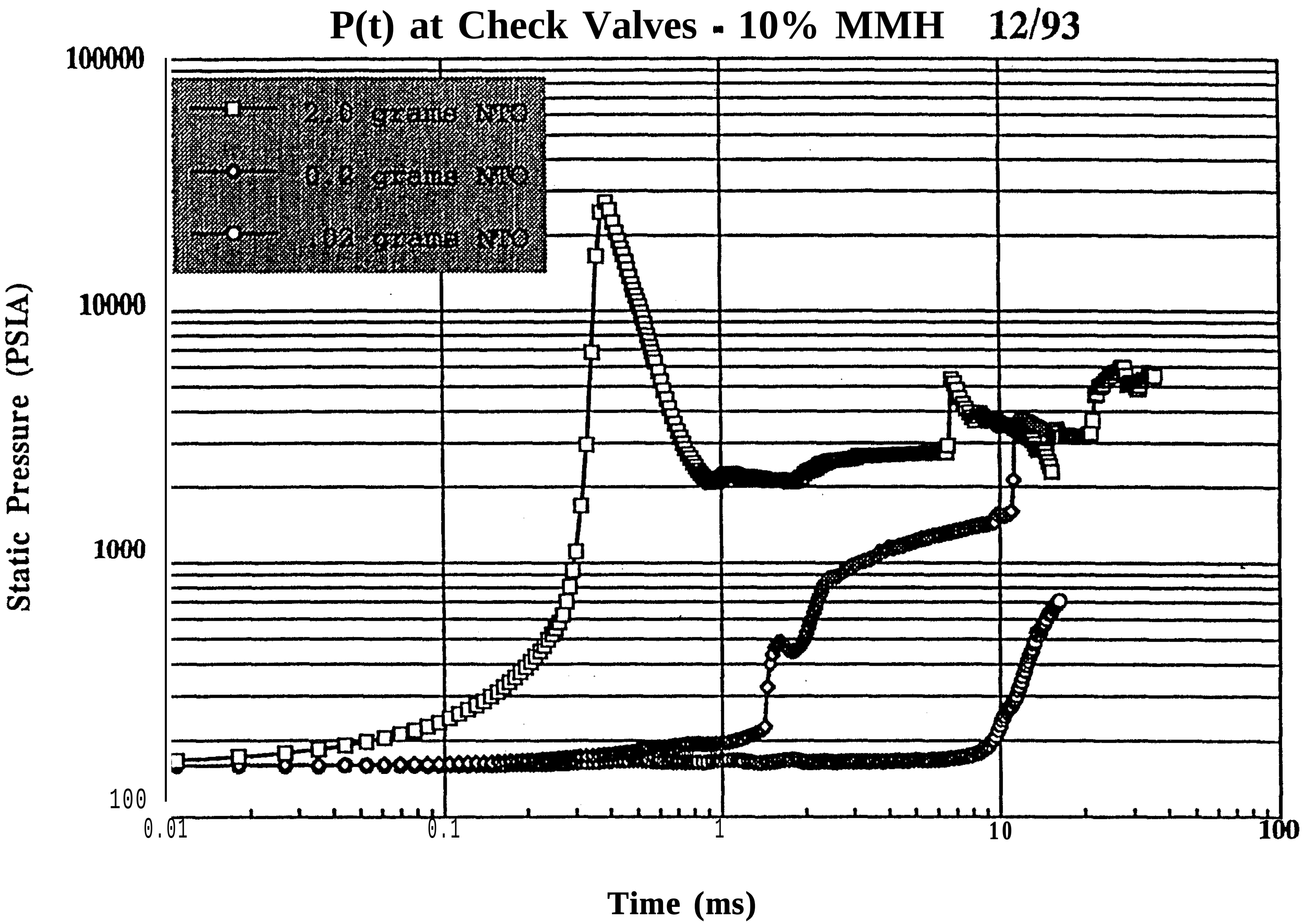


Figure D-8. Pressure Time History, Mixture of 2.0, 0.2, and 0.02 Grams of NTO in a Tube containing 10% MMH

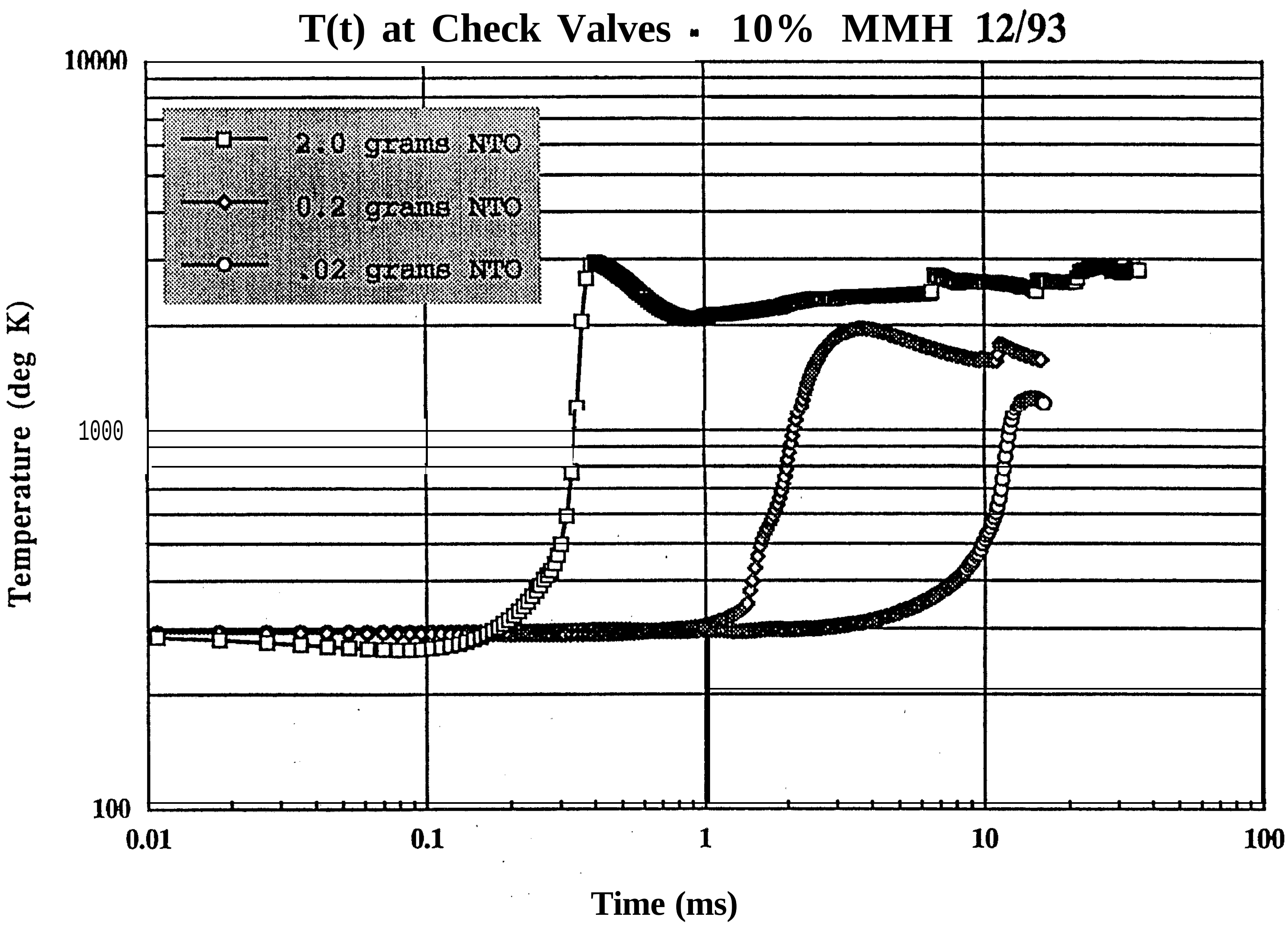


Figure D-9. Temperature Time History, Mixture of 2.0, 0.2, and 0.02 Grams of NTO in a Tube Containing 10% MMH

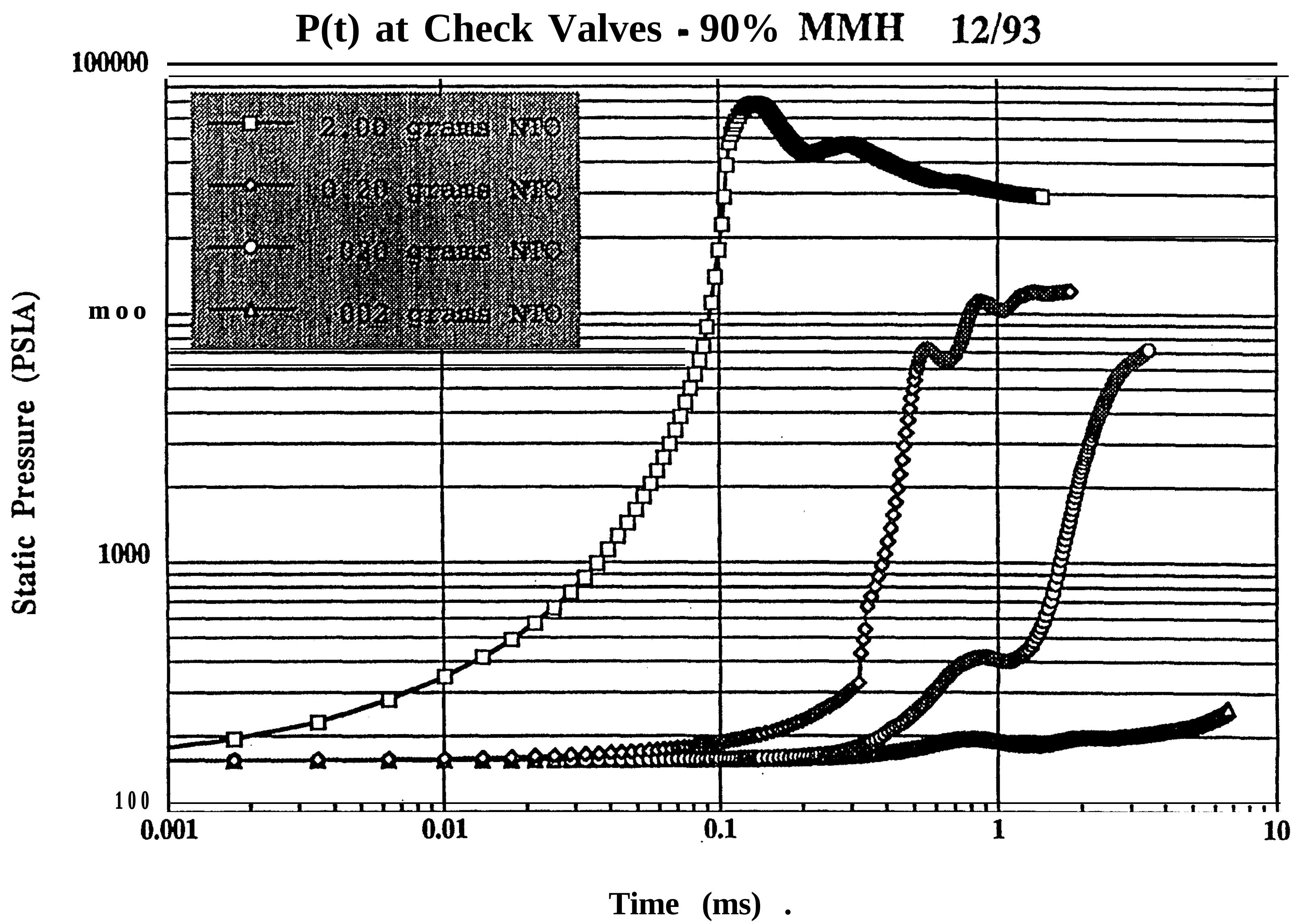


Figure D-10. Pressure Time History, Mixture of **2.0, 0.2,** and 0.02 Grams of **NTO** in a Tube Containing 90% MMH

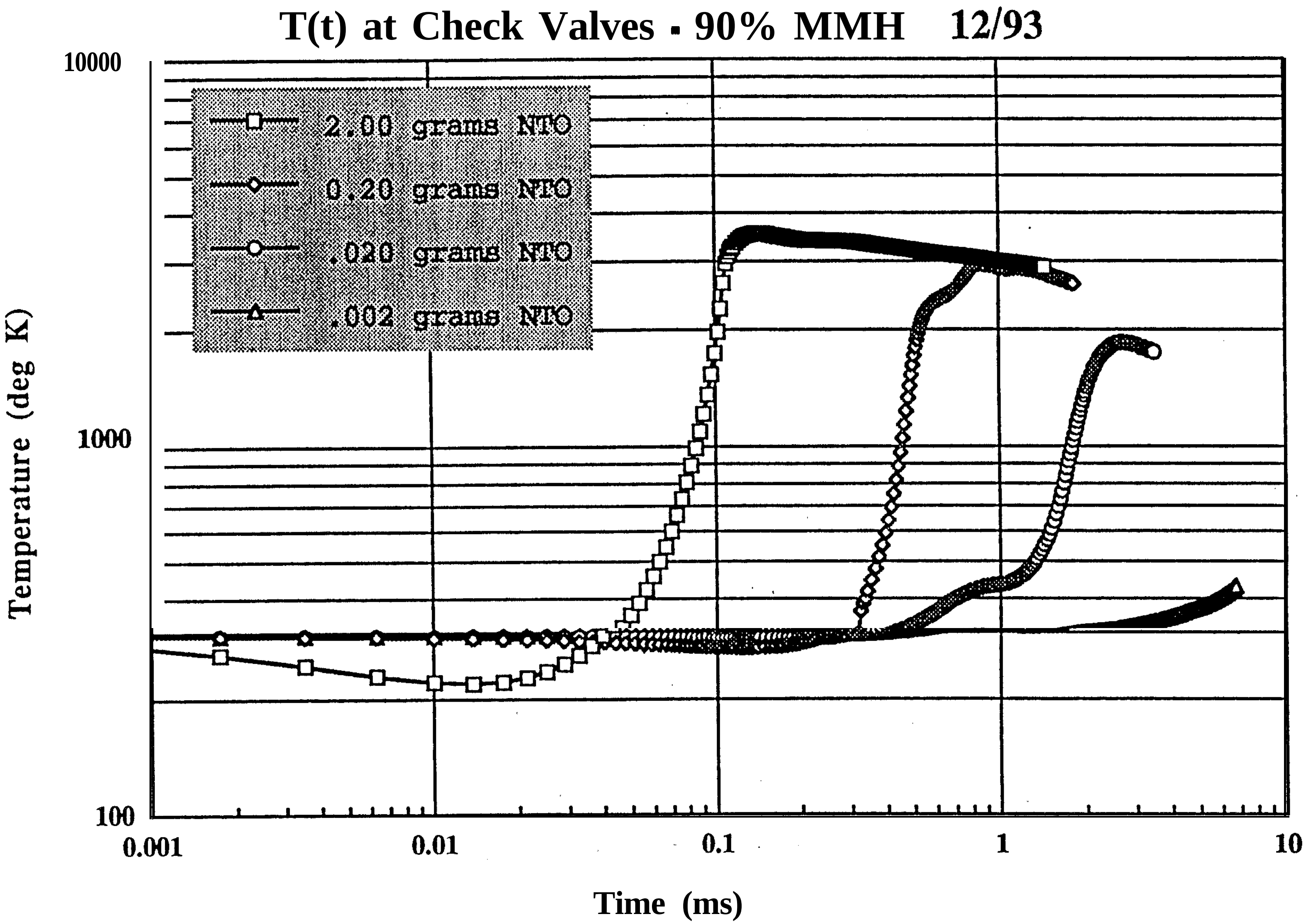


Figure D-11. Temperature Time History, Mixture of 2.0, 0.2, and 0.02 Grams of NTO in a Tube Containing 90% MMH to 30,000 psi by one millisecond Figure D- 11 shows the temperature history for the 90 percent MMH-filled tube case. Both figures also show the effects of reducing the NTO to 0.2 grams, 0.02 grams and 0.002 grams.

In both of the cases simulated (10 percent MMH and 90 percent MMH), when 2 grams or 0.2 grams of NTO were introduced, the tubing would have reached a temperature above 500°C within one millisecond. Hence, the tubing would have lost its strength (see figure D-7). The pressures on the walls of the tubing would far exceed the yield stress. The question then becomes: will the transient pressure last long enough to disrupt the tube? In the case where the pressure far exceeds the yield pressure, one can estimate the acceleration of the tubing by ignoring the tensile strength and treating the tubing as a fluid shell. Furthermore, one could expect that the tubing will rupture if the tubing shell is accelerated, say, ten times its thickness. Under these assumptions, the acceleration of the tubing can be estimated in a planar approximation. For the purpose of discussion, assume the pressure in the tube reaches 10,000 psi and the temperature is above 5OO'C. A simple calculation shows that the time required to displace the tubing shell by one shell thickness is about 4 microseconds. Hence at 10,000 psi, the tubing shell will be displaced by ten times its own thickness in ten microseconds (or one hundred times its own thickness in 40 microseconds); These times are much less than the duration of the pressure and thermal pulses. In all likelihood, the tubing would rupture before it could cool down and regain its strength. Indeed, if the tubing yield strength were low enough, the accelerated tubing wall would be Rayleigh-Taylor unstable. The growth time for a mode whose wavelength equals the shell thickness would be a few microseconds. Hence the tube would be expected to rupture in a few tens of microseconds.

The results presented above are indicative of the problems that could be encountered if a few grams (or even fractions of a gram) of NTO were rapidly injected into the tubing leading to the MMH tank and thoroughly mixed with the MMH in it. If the tubing ruptured, then the helium pressure tank would vent through the ruptured tubing, possibly spinning the spacecraft up to rates so high as to render it useless, and possibly tearing loose extended segments, such as the'high-gain antenna. Because these calculations assumed complete mixing, they probably represent the most stressful situation, and should be viewed as an upper bound

The NRL simulations also indicated that MMH had begun to selfdecompose due to the high temperature. If this decomposition were able to propagate through the MMH tubing and into the MMH tank, then the spacecraft would literally blow up. However, the simulations that have been performed were not able to predict whether the MMH decomposition wave would propagate into the MMH tank. Experimental tests also showed no evidence of MMH self-decomposition. Hence no conclusions can be drawn in this regard.

Because of the potential impact of the JPL check valve test results shown in Figure D-6, the Board requested that AFPL undertake a series of NTO/MMH mixing tests in a configuration similar to the pressurization plumbing on Mars Observer. Tests conducted up to the time of this writing have shown variable results. In most cases, no pressure pulses were observed. However, in one experiment (involving 4 grams of NTO), pressures of 2000 psi and 4500 psi were measured in different parts of the tubing. There were also indications (from bulges produced in the stainless steel tubing of the test rig) that even higher localized pressures (11 ,000 psi) were produced in areas of the test configuration that were not monitored with pressure sensors. The duration of these measured pressure pulses was several milliseconds. Another test (involving 2 grams of NTO) showed a pressure pulse of 8300 psi with several milliseconds duration. There is some question, however, whether the instrumentation was working properly on this test. While these tests are not true simulations of NTO-MMH mixing aboard the Mars Observer spacecraft (e.g., they do not simulate the probably significant effect of zero gravity), they do indicate that gram quantities of NTO in the MMH lines can produce significant pressure loading of these lines.

If the thermal and pressure loading of the Mars Observer titanium tubing occurred and caused a rupture of the tubing, then the helium pressurization gas would be expelled, causing the spacecraft to spin during the 10 minute period between firing pyrovalve PV5 and reactivating the X, Y, and Z-axis reaction wheels, Analysis indicates that the maximum spin rate that could be achieved would be about 90°/sec This spin rate represents the maximum rate achievable, given a unidirectional gas-expulsion stream with no obstructions in its path. This situation is unlikely to occur in practice, as the pressurization lines are well-covered with thermal insulation blankets. The initial rupture could have blown off the insulation blankets in the vicinity of the leak, or left the blanket partially attached, splaying the exhaust plume. Exhaust gases that hit the insulation blanket, the upper bulkhead of the spacecraft, or any other obstacles or appendages in their path would exert forces tending to cancel the spin that was induced from the initial thrust at the break. Since the gas is not likely to be released in a directed beam, but in a widening plume with a high likelihood of hitting obstacles in its path, it is reasonable to conclude that only a fraction of the energy stored in the GHe would be converted to spacecraft angular momentum. A spin rate of between 30°/sec and 50°/sec is reasonable to assume. In addition to venting gaseous helium the rupture should also cause venting of liquid MMH which would spray across the spacecraft damaging cabling and exposed electronics..

It would take approximately 14 minutes to expel the entire 10.7 lbs of gaseous helium. During the Pressurization Sequence, the attitude control system was deliberately disabled. When the attitude control system was reactivated, the spacecraft would likely be spinning at a rate (greater than 9°/sec) that would saturate both the digital and analog electronics of the IMU.

The momentum-unloading logic in the AACS would be unable to use thrusters to absorb some of the momentum, since the gyros would be saturated on all axes. For spin rates above about 30°/sec, the gyros would remain saturated on all axes indefinitely.

Saturation of two gyros would trigger entry into Contingency Mode. This would switch the downlink from the high-gain to the low-gain antenna. As discussed in Chapter F5, Mars Observer was close to maximum range from Earth at the time of the mishap. As a result, the Deep Space Network required 100-second integration times to process the signal from the Mars Observer's low-gain antenna. It is therefore very unlikely that the DSN receivers would be able to detect the Mars Observer's LGA downlink if the spacecraft were spinning at high rates. If Contingency Mode were entered within about four minutes of firing pyro valve PV5, then the HGA downlink would not be turned on, nor would the switch to the LGA downlink occur. This situation (i.e., no down link) would remain for as long as the spacecraft remained in Contingency Mode. There is sufficient energy available from the gaseous helium to force entry into Contingency Mode in less than four minutes from the time pyro valve PV5 was fired.

A rapid rotation of the spacecraft would also make it impossible to uplink groundcommands into the Mars Observer's computers. In addition, such a rotation rate would also prevent the solar array from receiving enough solar energy to keep the batteries charged. Depending on the rotation axes, they would discharge within a period of a few hours to a few days. The net result of the above events would be to render the spacecraft useless and probably unable to communicate.

The above calculations, simulations, and postulations'do not prove that the rapid mixing of MMH and NTO in the pressure manifold either took place or caused the failure. They do, however, show that if NTO in the quantities predicted from the JPL tests were to migrate through check valves CV1 and CV3, it would be a matter of grave concern, and must be considered to be a possible cause of the loss of downlink.

The design of the pressurization side of the propulsion system, while appropriate for situations where the pressure lines are warm and purged regularly (i.e., Earth-orbiting satellites), was not appropriate for the Mars Observer mission. This design permitted the possible accumulation (over an 11-month period) of significant quantities (grams) of NTO in the pressurization manifold tubing, where (during the Pressurization Sequence) it might be rapidly mixed with MMH. Such mixing would have the potential to release enough energy to rupture this tubing, or possibly even cause (via a decomposition wave) the MMH tank to burst. Either result would cause loss of the spacecraft. The Board cannot prove that sufficient NTO was forced into the MMH line and properly mixed with enough MMH to cause failure of either the line or the MMH tank. Nevertheless, the presence of significant NTO in regions of the pressurization manifold where it did not belong, the potential threat to the spacecraft that this represents, the clear correlation of such a failure with the Pressurization Sequence and the lack of other, more compelling scenarios leads the Board to consider this particular failure as a probable cause of the loss of downlink.

In any case, this design weakness must be rectified in any reflight of the Mars Observer or derivative spacecraft on an interplanetary mission. NASA should establish standards for the quantity of NTO (also MMH) that will be allowed to accumulate upstream in the pressurization manifold for bipropellant systems using common pressurization lines.

b. POTENTIAL CAUSE: PRESSURE REGULATOR FAILURE

Failure (in the open position) of the pressure regulator (see Figures D-S and D-12) between the gaseous helium tank and the NTO and MMH tanks would cause a rapid over-pressure and rupture of the NTO tank shortly after the firing of pyro valve PV7. This would destroy the spacecraft. Failure of this regulator could have been caused by:

- NTO frozen in the regulator balance orifice;
- Contamination blockage in the regulator balance orifice; or
- Contamination in regulator seat, causing leakage.

The frozen NTO mentioned above would have the same source as that discussed in Section a. above, namely migration through check valves CV1 and CV3. Unlike the chemical reaction scenario, however, only small quantities of frozen NTO would be required to fail the regulator. What is required for this to happen is for the regulator temperature to be less than the freezing temperature of NTO (11.8°F, -11.2°C). The Board observed that the temperature on the outside of the spacecraft where the regulator and pressurization manifold were located was not well known or well modeled. However, there was a temperature sensor on the inside of the bulkhead to which the regulator was mounted. This sensor indicated that the temperature on the inside of the bulkhead was about 1.5°C, which was about 22.9°F (12.7°C) above the freezing point of NTO. If the thermal conductivity between the bulkhead and the regulator was good, then the regulator would have been warm enough to prevent the NTO from freezing. If, however, the thermal conductivity between the regulator and the bulkhead were poor, then the regulator may have been cold enough to freeze the NTO and thereby fail the regulator. The Board considers this failure to be unlikely. However, since: (1) the thermal environment of the regulator was not known; (2) the failure scenario correlates with known NTO migration; and (3) the failure of the regulator provides a simple explanation for the loss of downlink; the Board decided to retain this postulated failure as a possible cause of the loss of downlink from Mars Observer.

Pressure regulators have been known to fail in the open position due to particulate contamination. The Board could find no documentation that indicated that the proper functioning of the pressurization system had been verified after assembly. Components were certified to be clean by their manufacturers; however, no post-assembly tests were performed to actually verify that the entire system was free of contamination and functioning properly. (While not related to this scenario, the Board also noted that the fuel and oxidizer were verified to be clean prior to "fueling," but were not tested afterward to verify that the fueling process had not stirred up contaminants.) The Board concluded that it could not be assured that the pressurization system was truly clean. The Board therefore concluded that the possibility of a pressure regulator failing open and causing failure of the NTO or MMH tanks, while unlikely, cannot be ruled out. For the above reasons and because this failure provides a simple explanation of the mission failure, the Board decided that this failure must be listed as a possible cause of the loss of downlink.

c. POTENTIAL CAUSE: FAILURE OF A PYRO VALVE CHARGE INITIATOR

Tests made by British Aerospace Corporation on European Space Agency (ESA) pyro valves and initiators that were designed to the same specifications (but were not identical) to the pyro valves and NASA Standard Initiators used aboard Mars Observer have shown that some firings result in the electro-explosive initiator being ejected from the valve body at speeds of approximately 200 m/s. Severe damage to the spacecraft (wiring, propellant tanks) could result. For example, one of the Mars Observer initiators (see Figure D-12) was located such that it would impact the MMH tank if it were ejected. This initiator (in PV6, Figure D-5) should not have fired, though there remains some ambiguity on this point. No NASA Standard Initiator has ever been known to exhibit this problem, but such an event is considered possible until the exact causes of the British Aerospace experience have been determined.

It should be noted that the initiators are ejected as a result of "erosion" of the threads in the titanium body of the pyro valve itself. The Board requested that the acceptance test lot of the Mars Observer pyro valves be examined for thread erosion. It was found that these valves had suffered erosion of about 50 percent of their threads, though none had failed by ejecting their initiators. Nevertheless, since the valves used by ESA have failed and are very similar in design to those used on Mars Observer, and since the Mars Observer test lot were found to suffer thread erosion similar to that found on ESA valves, the Board believes that a failed pyro valve charge initiator must be considered a potential cause of the loss of downlink, pending further studies on these valves. The Board believes that the differences between the ESA valves and the Mars Observer valves need to be understood.

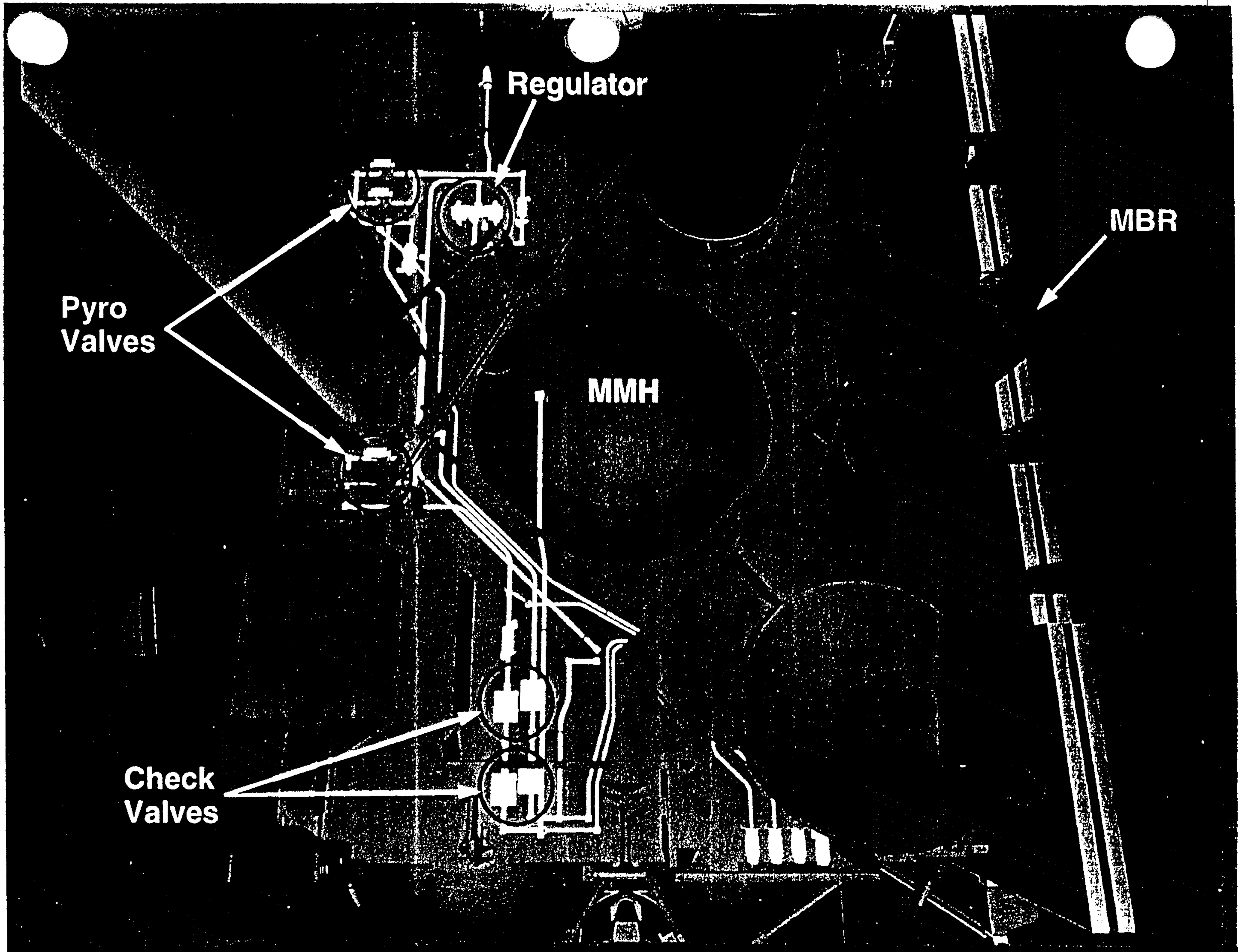


Figure D-12. Location of Mars Observer Pyro Valves, Regulator, Check Valves, MMH Tank, and MBR

d. POTENTIAL CAUSE: POWER SUPPLY ELECTRONICS POWER DIODE INSULATION FAILURE

The Mars Observer power supply electronics (PSE) module contains several power diodes with cathodes connected to the power bus. A permanent short in any one of these diodes would render the spacecraft useless. The diode cathodes am insulated from the chassis by a 0.006-inch flexible insulator, a fiberglass washer, and conformal coating. The insulation material is susceptible to cuts and tears, especially if there are burrs or irregularities in the materials separated by the insulation. The shock from firing the pyro valve(s) could have provoked a final breakthrough of the insulation, causing a short-circuit between the power bus and chassis ground, and a complete failure of the Electrical Power System. The Board therefore requested that the spare Mars Observer PSE box be opened and inspected to determine the tolerances between components and the workmanship employed. The inspection revealed several discrepancies. The most serious discrepancy was a misalignment of three out of ten stud-mounted power diodes in the boost voltage regulator (BVR), such that the diode stud was in, or very close to, direct contact with the chassis. The misalignment was due to improper installation of an isolating shoulder washer. There was no quality control inspection of this installation. Five of these ten diodes have the potential to short the power bus directly to ground. The Sil Pads (insulator/thermal heat sink) showed no breaks in the Kapton layer and passed an isolation test. However, the pads contained embedded metal particles and scratches in the thermal material on both sides of the Kapton.

Another major discrepancy was the incorrect and incomplete application of thermal solithane to 16 PSE stud-mounted power diodes. This was required by the plan drawings. Six of the diodes had no thermal solithane at all. These were “stamped” as approved by both MMAS quality control and Defense Plant Representative Office @PRO) inspectors.

Based upon this inspection of the spare PSE (see Appendix Q for a more complete report), the Board believes that this failure scenario, while only weakly correlated with the Pressurization Sequence, must be retained as a possible cause for the loss of Mars Observer downlink.

Chapter D4
Board Observations and Concerns

In its investigation of the Mars Observer mission failure, the Board was obligated to review in detail the history of the spacecraft development, the management and contractual procedures used to develop and build the bus and its scientific instrument suite, as well as the characteristics of the actual hardware and software launched into space. The Board developed a number of concerns relative to the program that, although they could not be directly related to the Mars Observer mishap, may have contributed in an indirect way to the failure. In addition, many of these concerns should be carefully considered by NASA management, since they have the potential to affect future spacecraft developments and operations.

The rationale and background information supporting these concerns are provided in Part F of this report

a. GENERAL OBSERVATIONS AND CONCERNS

(1) The telemetry should have been left on during critical events.

(2) The top-level systems engineering (i.e., integration of spacecraft sys terns under realistic, mission-driven environmental conditions) was inadequate.

(3) The contract philosophy (firm, fixed-price contract), while appropriate when the program was formulated, turned out to be inappropriate for the Mars Observer mission after 1987. In addition, this contract philosophy limited the utilization of Jet Propulsion Laboratory expertise and oversight in spacecraft development.

(4) Overall software development did not follow sound practices; e.g., inadequate configuration control and no independent verification and validation.

(5) There was far too much reliance on heritage for spacecraft hardware, software and procedures, especially given that the Mars Observer mission was fundamentally different from the mission of the satellites from which the heritage was derived.

(6) Several inappropriate trade-offs were made between redundancy and weight. For example, the redundancy and reliability of the propulsion system was reduced to save a few pounds.

(7) Spacecraft autonomy was overly relied upon and its execution was neither well understood nor adequately tested.

(8) There was inadequate testing of some spacecraft systems (e.g., Command and Data Handling), and the spacecraft as a whole.

(9) The program was buffeted by many externally driven changes throughout its history in the 1980s, but did not have the flexibility (mission, budget, contract type) to accommodate them.

(10) There was no risk-management plan to make system-wide trade-offs between competing requirements; caused partly by changing requirements and partly by organizational structure/contract philosophy.

(11) The decision not to qualify the traveling wave tubes in the transmitter power amplifiers for the shock induced by the firing of the pyro valves was an error. It is also likely that the Pressurization Sequence would have been designed differently had the decision been to leave the telemetry on. A more step-by-step sequence, designed to take advantage of telemetry at each step, could well have allowed a potentially catastrophic failure to be detected and corrected. At the very least, had telemetry remained on, it is very likely that the cause of the failure would be known.

(12) There was too much reliance on pre-loaded software scripts for long command sequences when and where it was neither necessary nor appropriate.

b. SYSTEM-SPECIFIC OBSERVATIONS AND CONCERNS

(1) Electrical Power System

(a) There were several single-point failure modes that were of concern:

1 There were 33 unfused capacitors, each of which could cause a failure of the Electrical Power System.

2 Thin insulation between seven power diodes and the chassis could be subject to break-through, causing a short circuit of the Electrical Power System

3 Single insulation on unfused wiring from the power supply electronics to the fuse boards and battery chargers could be damaged and short-circuit the Electrical Power System

4 Both primary and backup RF telemetry systems were fed from the same fuse board.

(b) The use of copper-clad aluminum wire to save a small amount of weight is not good practice.

(c) Quality control inspection of fabrication by both contractors and DPRO was inadequate.

(2) Attitude and Articulation Control System

No problems were noted, except for the loss of inertial reference on several occasions (primarily a software problem).

(3) Command and Data Handling System

(a) Redundant crystal oscillators were not employed in a truly redundant manner. The clock dividers did not switch to the redundant unit in some failure modes; this situation was not recognized until the post-incident investigation.

(b) There was no method of determining the health (i.e.,oscillator output) of the backup redundant crystal oscillator.

(c) There was inadequate testing of the complete Command and Data Handling System; too much testing was done in non-flight-like modes.

(d) The possibility existed that the complete Command and Data Handling System could be hung up and unable to receive commands; could be caused by discrete device failures, voltage transients or improper software commands.

(e) Three single-point failure modes were identified and appropriately waived (i.e., risk accepted), but a fourth was not discovered until the post-anomaly investigations.

(f) The Verification Test Laboratory was not available in time to support spacecraft system design, nor to test completely the response of the spacecraft to various failure scenarios. It also was not of sufficient fidelity to rigorously test the Command and Data Handling System in any case.

(g) The command script for the Pressurization Sequence called for the skewed reaction wheel to be powered on (for first time in 11 months) immediately after the transmitters were turned off - it could have been tested first and/or powered on before the transmitter shut down.

(4) Telecommunications System

No problems noted except for turning off transmitters, discussed above.

(5) Mechanical Systems

(a) There was unjustified reliance on check valves with only Earth-orbital heritage (where leak-proof operation would not be required for extended periods) for an application requiring flawless performance for 11 months at cold temperatures and zero-G conditions (heritage trap), risking the possibility of the hypergolic bipropellants mixing in the pressurization system, with potentially catastrophic results.

(b) Propulsion system redundancy was inappropriately removed after the change from the Space Shuttle to a Titan III.

(c) There was a lack of ability to measure temperature on the pressurization side of the propulsion system.

(d) There is some evidence that the NASA Standard Initiator (used in Mars Observer pyro valves) could cause the threads in the valve body to fail and eject a projectile when fired, as has been observed by British Aerospace Corporation on similar units; a complete analysis of the failure mechanism is required to ensure that this threat does not exist for U.S. spacecraft using the NASA Standard Initiator and pyro valve.

(6) Software

(a) The software did not always fulfill system requirements (e.g., three software builds were made after launch).

(b) The program had a software development plan, but it was not followed.

(c) There was no independent verification and validation process, violating good design practice.

(d) The software development effort was understaffed, when compared to the magnitude and importance of this task to the mission and the spacecraft.

(e) There was a lack of adequate software configuration control.

(f) The Redundancy Management software was not adequately understood.

(g) Software developed for Earth-orbital missions does not straightforwardly transfer to an interplanetary mission (heritage trap).

(h) The software development process was the.problem, not the software-developing personnel.

Chapter D5
Concluding Remarks

Unambiguous determination of the cause of the loss of downlink from the Mars Observer spacecraft was hampered by a lack of telemetry relating to the Pressurization Sequence. This forced the Board to approach the problem by eliminating the implausible in a step-wise fashion. Those failure scenarios that remained were then considered to be possible, with some being more probable than others.

During the final days of its deliberations, the Board learned that the first stage of the launch vehicle had suffered a fuel-depletion shutdown during launch. This could have caused a significant transient shock to the spacecraft and its systems. The Board did not have time to investigate whether or not this transient might have had a bearing on the loss of downlink. The Board recommends that NASA undertake to investigate the potential effect of this transient on the pressurization manifold. Of particular interest would be the pressure lines leading from service valves SV1 and SV2 (Figure D-5). Failure of either of these lines would not have had any effect on spacecraft operations (and could not have been detected from the ground) until the Pressurization Sequence was initiated. However, pressurization of the bipropellant system would have permitted gaseous helium to be vented to space through the broken line, producing a spin-up of the spacecraft similar to the failure mode described with the first principal finding of the Board-

As discussed in this report, the Board observed a number of systemic weaknesses in the Mars Observer program as it evolved over the years. While no direct linkage can be made between these weaknesses and the mishap, they do remain a significant concern for future programs.

The Board would like to express its appreciation for the support provided to it by the six Technical Teams; the other NRL and AFPL personnel who supported it; the NASA representatives; the JPL Project Team and the JPL investigation team; and the various Martin Marietta Astro Space teams that supported the Boards deliberations.

Chapter F6
Mechanical and 'Propulsion System

Chapter F1 of this report provides an overview of Mars Observer Mechanical Systems. The only part of the Mechanical System that could have contributed to the loss of Mars Observer is the Propulsion System Therefore, this chapter will address only Propulsion System operation and failure modes.

a. PROPULSION SYSTEM DESCRIPTION

(1) General

The Mars Observer Spacecraft Propulsion System consists of a hydrazine monopropellant section for attitude control, and a monomethyl hydrazine (MMH) and nitrogen tetroxide bipropellant (NTO) section for velocity addition and correction maneuvers. Both of these systems are pressurized with gaseous helium and are shown schematically in Figures F6-1 and F6-2, respectively. Because it is important to understand the relative locations of Propulsion System components, an isometric view of the bipropellant pressurization system is included as Figure F6-3.

The monopropellant section is of a conventional blow-down design used extensively by the Naval Research Laboratory, Martin Marietta Astro-Space Corporation, and other spacecraft manufacturers for the past 25 years. In this system, two hydrazine tanks can supply propellant to any of 12 catalytic hydrazine thrusters. The thrusters are arranged in two redundant branches, each of which contain four 4.45-N thrust units and two 0.9-N thrust units. Since this system worked properly throughout the Mars Observer mission and was deactivated during the failure period, it is mentioned only for completeness, and will not be discussed further.

The bipropellant section is a pressure-regulated propulsion system using four 490-N thrust main engines to provide delta-V and four 22-N thrust engines to provide thrust vector control (TVC). In normal operation, only two of the 490-N engines are operated at a time; the second pair provides redundancy. The pressurant supply consists of a carbon-filament-wound stainless steel tank with a maximum operating pressure of 4,500 psia. Pressurant flow to the two titanium propellant tanks is controlled by a single body, series-redundant, hard seat regulator. The pressurant tank is isolated from the regulator by two normally closed pyro-valves, PV7 and PV8 and a filter, FG1. The function of these components was to preclude overpressurization of the Mars Observer propulsion system due to any regulator seat leakage that might occur during the 1 1-month cruise phase to Mars. In addition, the MMH tank is positively isolated from both the NTO tank and pressurization system by normally closed pyro valves PV5 and PV6. The function of these valves was to eliminate the risk of bipropellant reaction-product salts forming during the

MONOPROPELLANT PROPULSION SUBSYSTEM SCHEMATIC

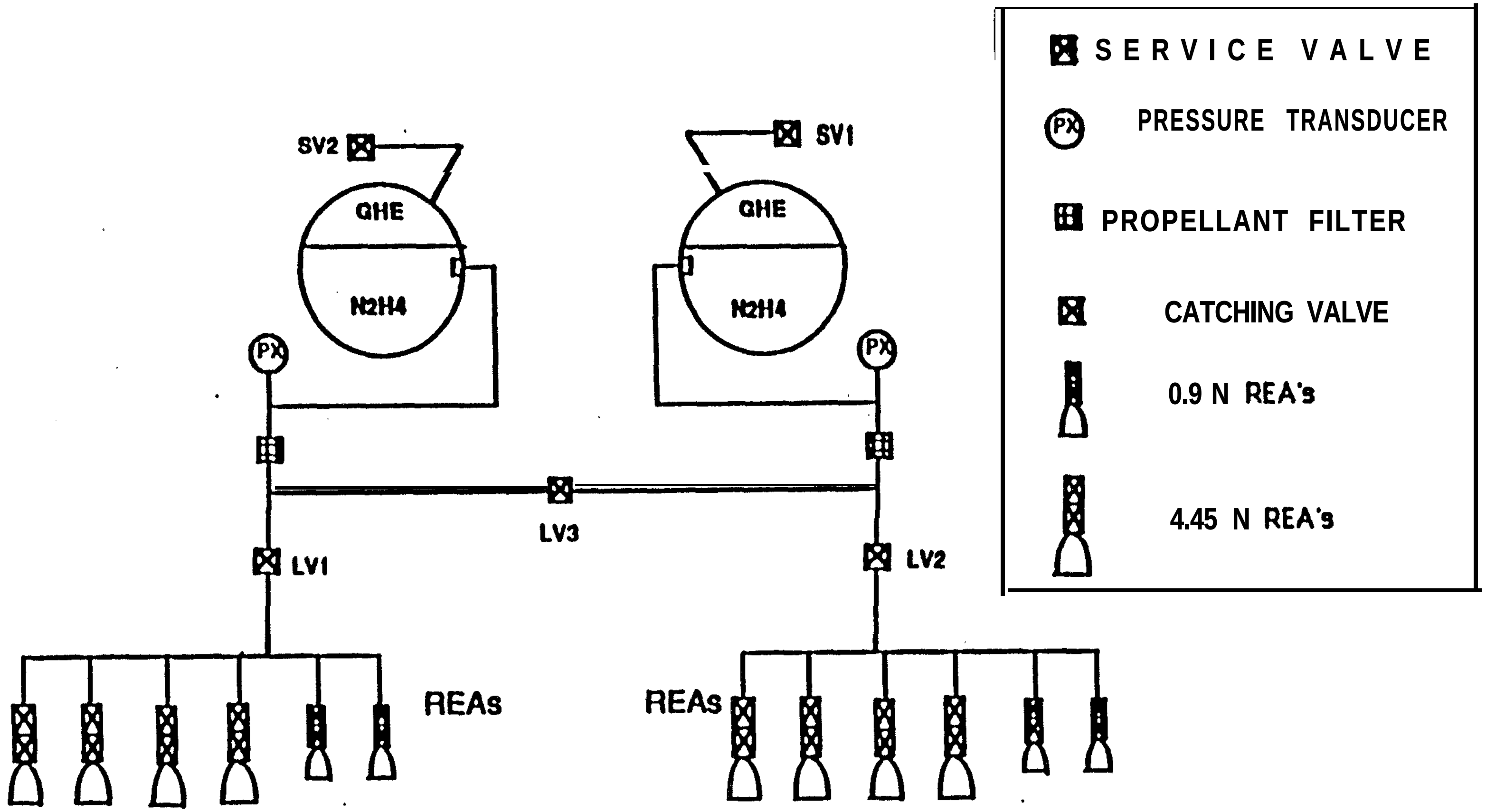


Figure F6-1. Monopropellant Propulsion Subsystem Schematic Program

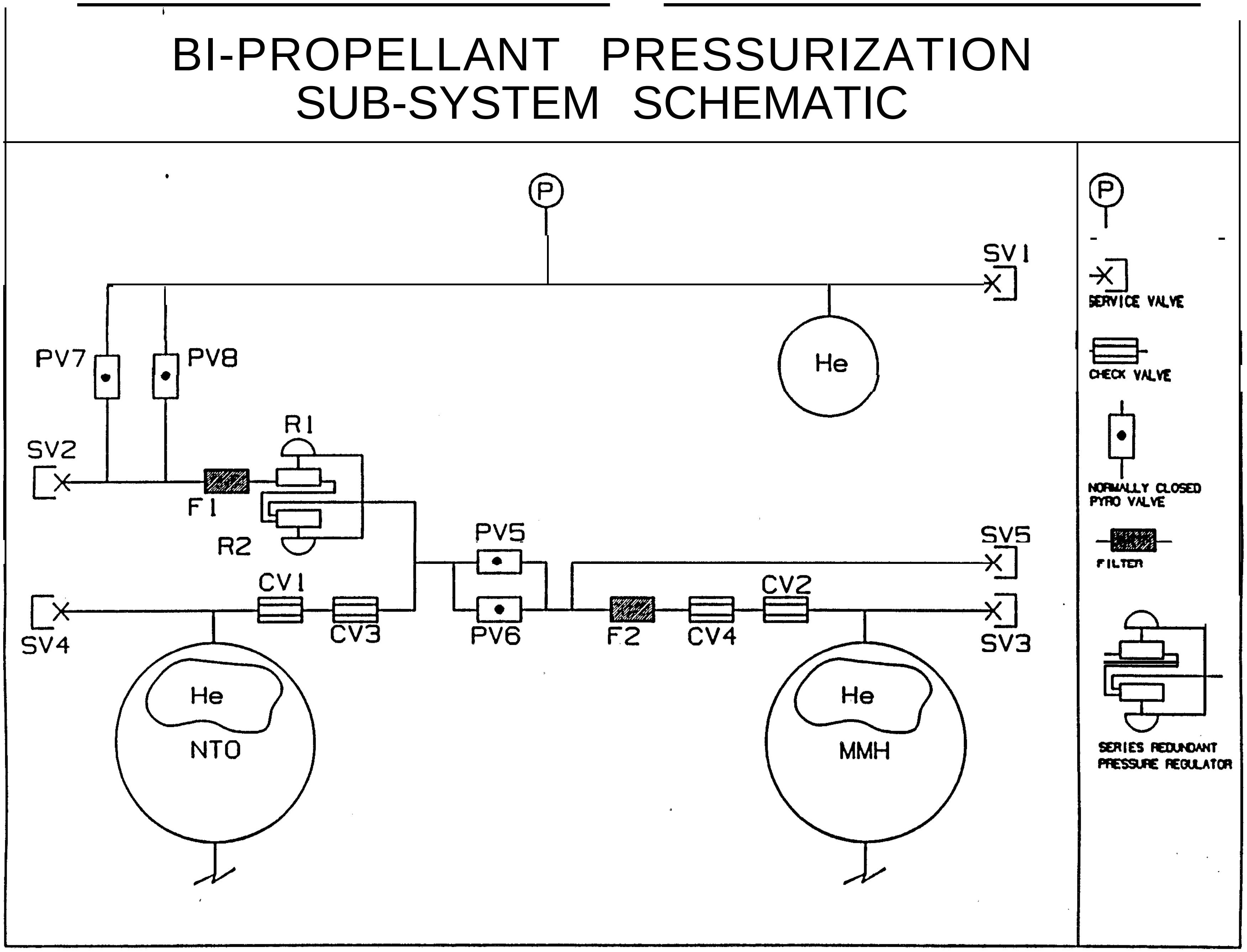


Figure **F6-2.** Ripropellant Propulsion Subsystem Schematic Program

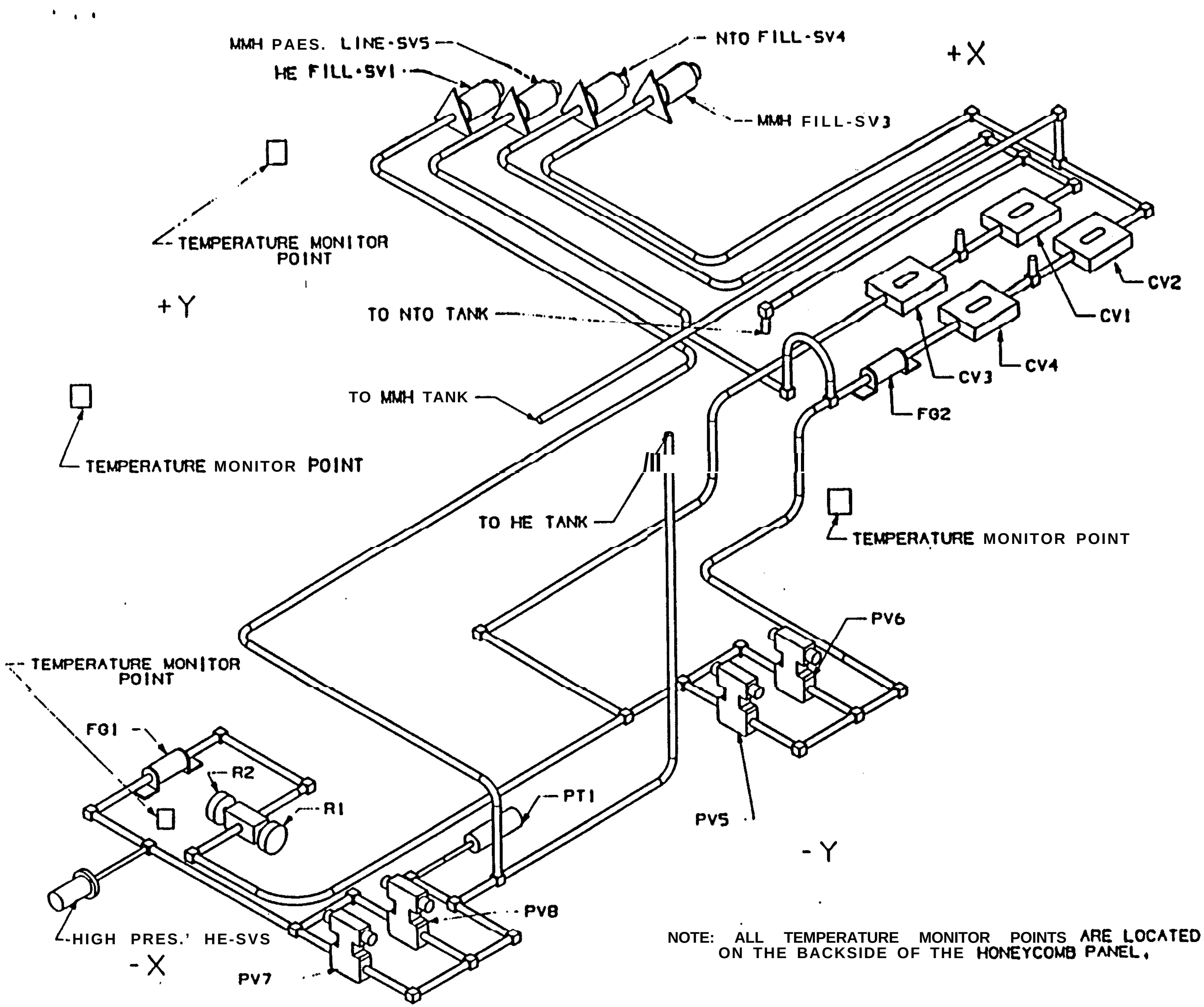


Figure F6-3. Bipropellant Pressurization Subsystem (Isometric View) cruise phase. These salts have been observed to lead to regulator leakage and check valve sticking on other spacecraft. The low-pressure gaseous helium (GHe) manifold is protected from MMH and NTO back-steaming by two pairs of check valves (CV1/CV3 and CV2/CV4) mounted in series at the inlets to the MMH and NTO tanks. Each pair consists of two check valves that are parallel-redundant, soft-seated and made by different manufacturers.

Prior to a delta-V maneuver such as Mars Orbit Insertion, the propellant tanks are pressurized by GHe to a nominal 255 psia. The Mars Observer spacecraft carried 1170 Ibs of MMH fuel and 1865 lbs of NTO oxidizer, each contained in its own spherical tank mounted along the Z axis in the spacecraft center cylinder. The bipropellant system uses filters, latch valves and thrust valves to isolate NTO and MMH from the thrust combustion chambers. Service valves in the high- and low-pressure manifolds permit filling, testing, and venting of the GHe system, Service valves in the inlet and outlet line of the MMH and NTO tanks permit filling, testing and emergency offloading of the propellants.

Electrical power, command and control functions, pressure/fault monitoring, and thermal regulation are provided to the Propulsion System by the C&DH System. Propulsion System components for Mars Observer were selected based on MSFC-SPEC-522B for material resistance to stress corrosion cracking and MSFL-HOBK-527E for material compatibility with MMH and NTO. System component maximum-expected-operating-pressure (MEOP), proof pressure and burst pressure are summarized in Table F6-1. Low-power heaters maintain proper system temperature. Only a few system component temperatures are monitored by the spacecraft, but heater circuits contain protective shut-down circuitry. For ground operation, the spacecraft propulsion system interfaces with four propellant carts for propellant loading, pressurant loading, and emergency offloading.

(2) Propulsion System Components, Heritage and Redundancy

(a) GHe Pressurant Tank

1 Description

The GHe pressurant tank (Figure F6-4) is a .66-m (26-inch) outside diameter cryoformed 301 stainless steel shell tank with a graphite/epoxy fiber overwrap. This tank is designed and tested to MIL-STD-1522A using fracture mechanics analysis techniques. The tank MEOP is 4,500 psia. It has been proof pressure tested to 5,625 psia and burst rated at 6,750 psia (actual burst at 7,180 psig). This results in a design burst safety factor of 1.5 and a demonstrated burst safety factor of 1.6. Stress analysis indicates a positive safety margin at burst pressure based on the ultimate strength of materials. Stress analysis and environmental testing have demonstrated

TABLE F6-1. PROPULSION SYSTEM COMPONENT PROOF AND BURST PRESSURES

Component	Launch (psia)	MEOP (psia)	Proof (psia)	Burst (psia)
Bipropellant Subsystem				
MMH, NTO Tanks	250	300	375	450
GHe Pressurant Tank	4200	4500	5625	6750
GHe High Pressure Transducer	4200	4500	6750	11250
High Pressure Service Valve (0.25 in.)	4200	4500	6750	11250
GHe Pyrotechnic Valve	4200	4500	6750	11250
GHe High Pressure Filter	4200	4500	675	10000
GHe Low Pressure Filter	250	300	450	750
GHe Pressure Regulator	250	4500	6750	11250
Low Pressure Service Valve (0.25 in.)	80	300	450	750
GHe Check Valve	250	300	450	20000
GHe Manifold	4200	4500	6750	20000
Bipropellant Service Valves (0.5 in.)	250	400	600	1000
Bipropellant Pressure Transducer	250	400	600	1200
Bipropellant Latch Valves	250	300	900	1500
Bipropellant Filter	250	300	450	750
490-N Thruster	80	400	600	1000
22-N Thruster	80	400	600	1000
Bipropellant Manifold	80	400	600	8850

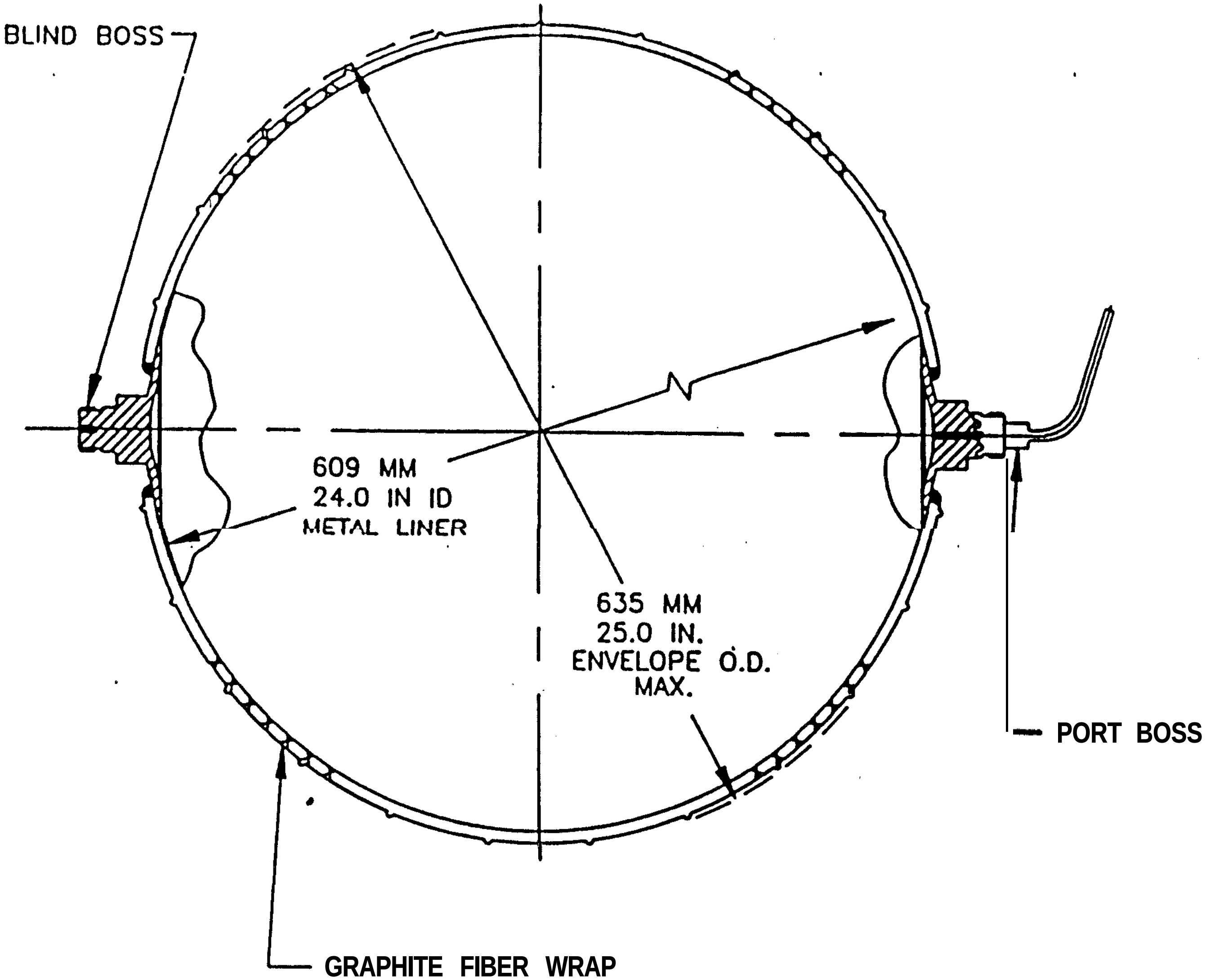


Figure F6-4. Helium Pressurant Tank

a design margin based on worst-case expected Titan-III launch loads, which exceed loading experienced during ground handling and transportation. The fracture mechanics analysis indicates that hazardous flow growth does not occur under pressure cycling or sustained loads, and that a leak-before-burst (LBB) failure mode is indicated. Stress analysis on the two tank bosses indicate significant safety margins for both yield and ultimate strength under worst-case combined Ioads (vibration and pressure).

Tank lines and fittings are proof tested at 9,000 psia. Tank tube stubs are designed with a 4-to-1 safety factor. Prior to closure welding, the finished inside and outside surfaces of the helium tank liner were penetrant-inspected in accordance with MIL-STD-6866B, Type 1, Method A. The penetrant used for the inspection was free of chlorides. All helium-tank butt welds were subjected to radiographic inspection in accordance with MIL-STD-453B, and fluorescent penetrant inspected in accordance with MIL-STD-6866B. The GHe pressurant tank uses a two-point mount. The tank liner is compatible with NTO vapor, MMH vapor, nitrogen, isopropyl alcohol, water, and GHe. No other fluids were used during tank processing or integration and test. CRES 301 is compatible with MMH and NTO per MSFC-HDBK-527E and is highly resistant to stress corrosion cracking per MSFC-SPEC-522B.

2 Heritage

The GHe pressurant tank, MMAS P/N 2631013, is manufactured by ARDE, Norwood, New Jersey; P/N E4299. This tank, with a different outlet tube configuration, was used previously by MMAC on the series 5000 spacecraft at 4200 psi MEOP. The identical tank with boss modification is used on the Atlas missile at 4000 psi MEOP.

3 Redundancy

The GHe pressurant tank is a non-redundant unit.

(b) Manifolds

1 Description

Manifolds are all-welded construction using 3A1-2.5V titanium tubing. Titanium-to-stainless-steel transition tubes are used to install stainless steel components. Proof pressure and leak tests were performed on the assembled subsystem to verify the assembly integrity. Redundant nichrome wire heater elements are routed along all liquid tubing up to the check valves and all tubing is over-wrapped with goldized Kapton tape for thermal control. The 3A1-2.5V titanium alloy is compatible with MMH and NTO per MSFC-HDBK-527E and is highly resistant to stress corrosion cracking per MSFC-SPEC-522B.

2 Heritage

All tubing used for the propulsion system is supplied by Nikko Wolverine. MMAS uses this tubing for all 3000, 5000, and 7000 series spacecraft. The Mars Observer tubing assemblies incorporated an acid etch of the weld area prior to the welding of each tube end. This process enhances the weld quality and is now used on all MMAC Programs.

(c) Service Valves

1 Description

Three types of service valves, MMAS P/Ns 2631025, 3264496, and 326551, are used on the Mars Observer spacecraft: 0.25-inch high-pressure service valves; 0.25-inch, low-pressure service valves; and 0.5-inch, liquid service valves. Figure F6-5 shows a typical Mars Observer spacecraft service valve. All three types of service valves are manually operated and constructed of 6A1-4V titanium. Each service valve provides three mechanical seals to prevent propellant or pressurant leakage: the valve seat when closed; the installation of an internal cap; and the external cap. The 0.25-inch, high-pressure service valves are used to pressurize the helium pressurant tank and to perform pressure tests on the subsystem. The 0.25-inch, low-pressure valves are used for venting the MMH and NTO tanks during loading and for conducting pressurization tests of the subsystem. The 0.25-inch, low-pressure service valves used in the MMH manifolds, NTO manifolds, and GHe manifolds (as well as the hydrazine monopropellant service valves) have unique inlet fittings to prevent mismating ground support equipment couplings, The 0.5-inch service valves are used to load and offload the MMH and NTO tanks. Each has a unique inlet fitting to prevent mismating. All service valves arc compatible with MMH, NTO, water, isopropyl alcohol, GHe, and gaseous nitrogen.

2 Heritage

All service valves were manufactured by Pyronetics, Denver, Colorado; P/Ns 1846-9, 1846-126 and 1845-10. All three valves have identical operating and seat designs, which are scaled for each size. This valve design has been used on spacecraft and launch vehicles for 25 years- All MMAS spacecraft incorporate this type service valve. Qualification was by similarity.

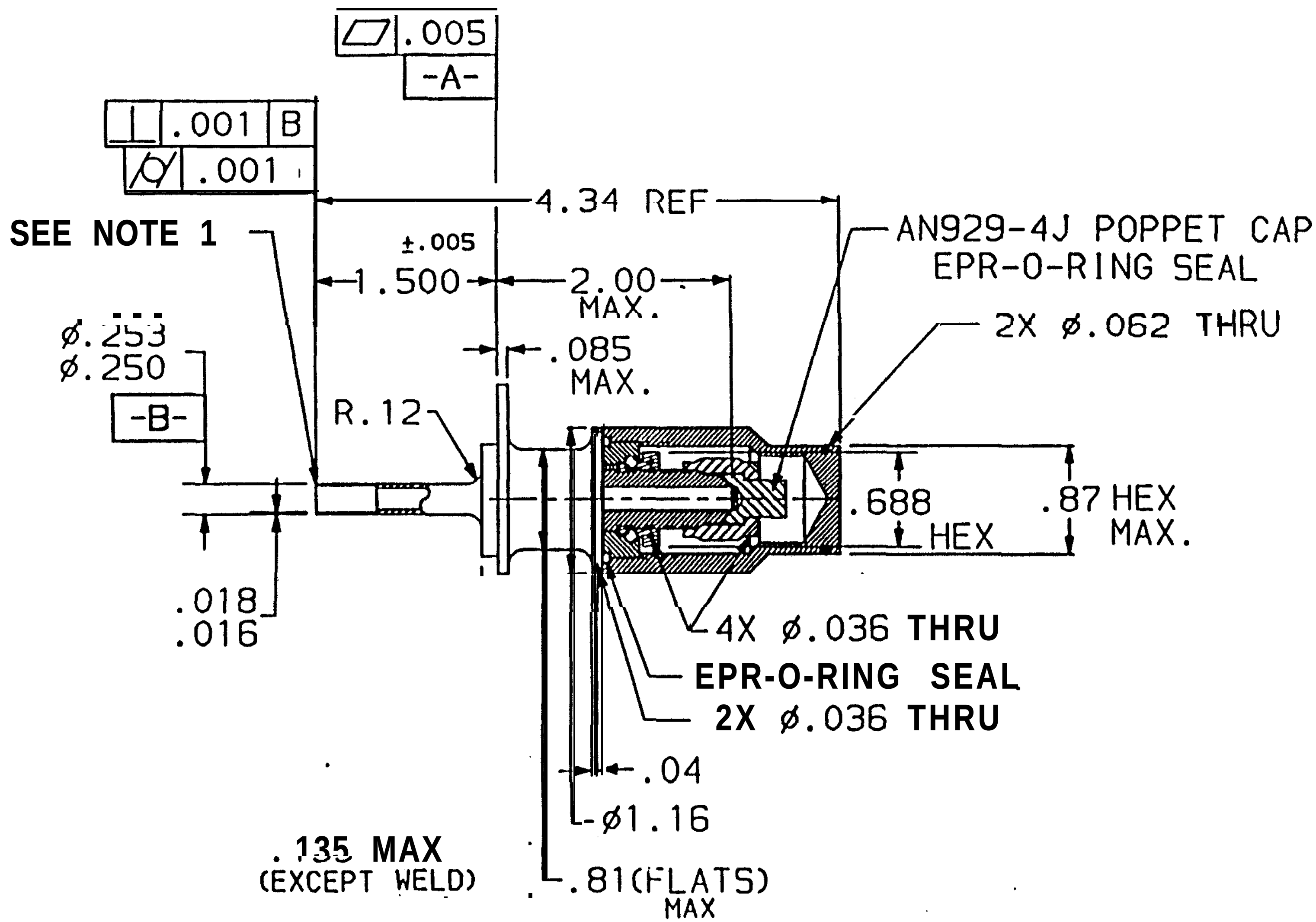


Figure F6-5. Mars Observer Service Valve (Typical)

3 Redundancy

Each valve is internally redundant.

(d) Pressure Transducers

1 Description

Pressure transducers are used as part of the bipropellant subsystem to measure tank pressures of the MMH, NTO, and GHe pressurant. The high-range pressure transducer uses a strain gauge sensor to produce a O-5 VDC signal over the pressure range of O-4500 psia. The low-range pressure transducer uses a strain gauge sensor to produce **a O-5 VDC** signal over the pressure range of O-450 psia. The high-range pressure transducer is compatible with GHe, GN2, isopropyl alcohol, water, and Freon 113, while the low-range pressure transducer is compatible with MMH and NTO in addition to the above-listed fluids.

2 Heritage

The pressure transducers are manufactured by Statham, Oxnard, California; P/N PA-489-4.5M (4500 psia) and PA-489450 (450 psia). Qualification for the Mars Observer was by similarity. The transducer design is identical for all applications (except for the strain gauge diaphragm thickness). This design is used on all MMAS 3000, 5000, and 7000 series spacecraft, and has been used extensively throughout the spacecraft industry.

3 Redundancy

The pressure transducers are non-redundant units.

(e) Pyrotechnic Valves (Normally Closed)

1 Description

Two parallel, normally closed pyro valves isolate the high-pressure GHe tank from the regulator inlet. Similarly, two identical valves isolate the MMH from the NTO section of the GHe delivery manifold to prevent mixing of MMH and NTO vapors. Figure F6-6 shows a cross-sectional view of one of the pyrotechnic valves, which were opened in preparation for Mars Orbit Insertion. Dual O-ring seals stop external leaks, and individually sealed inlet and outlet lines preclude internal leaks. The valves are actuated by a NASA Standard Initiator (NSI). The valve is considered dual fault tolerant against mechanical failure. The valve-firing circuit provides dual fault tolerance against inadvertent initiation. The valve MEOP is 4,500 psia, proof pressure is 6,750 psia, and burst pressure is 11,250 psia.

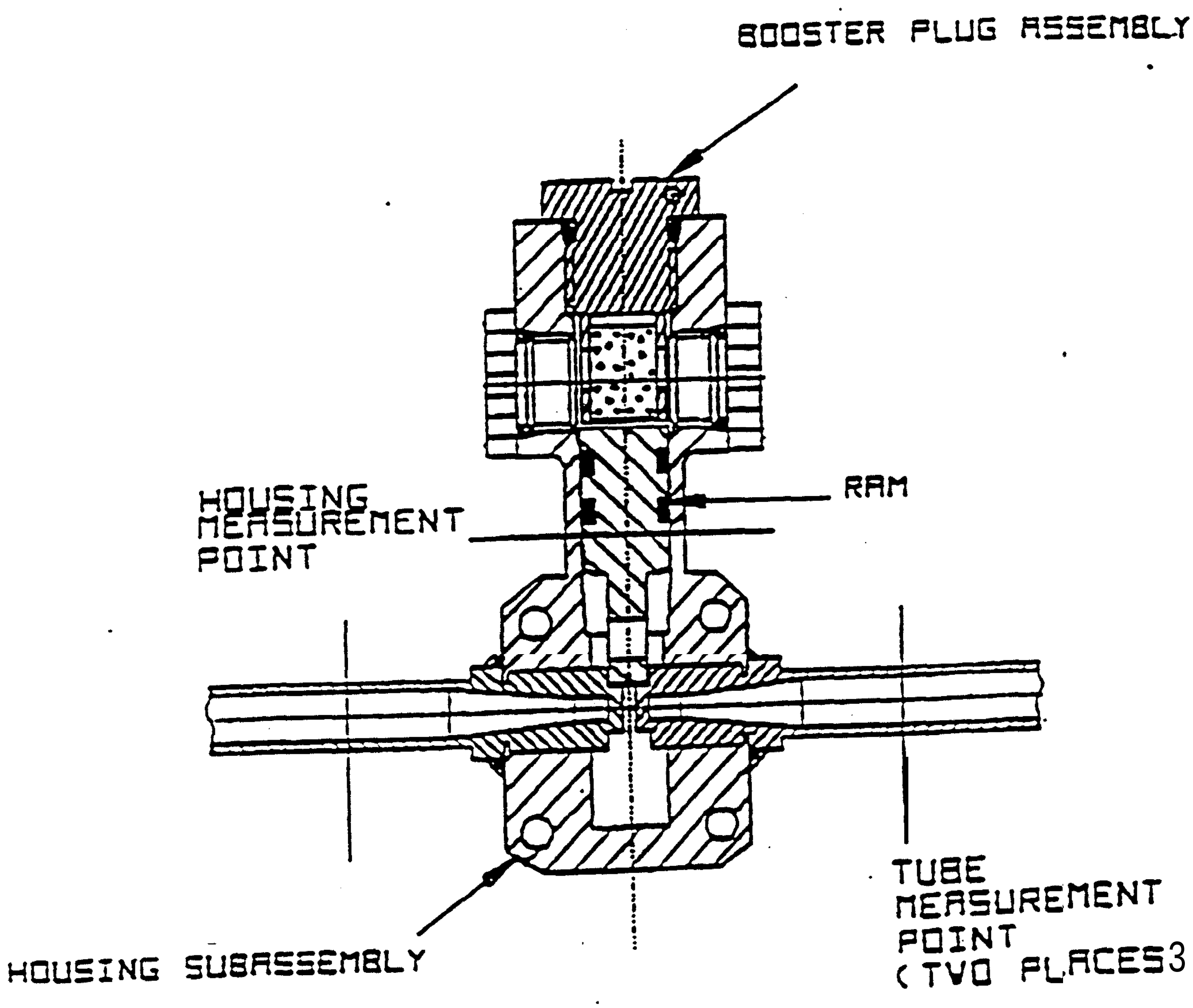


Figure F6-6. . Mars Observer Normally Closed Pyro Valve (Cross Section)

2 Heritage

Two pyro valve designs are installed in the Mars Observer bipropellant propulsion system. Both are manufactured by OEA/Pyronetics, Denver, Colorado; P/Ns 1467-19 (high pressure) and 1467-20 (low pressure). The valves are identical except for inlet/outlet tubing wall thickness for the high- and low-pressure applications. They are used on the MMAS Series 5000 and IABS Programs. When used with an OEA-produced initiator, this model valve has experienced structural failures of the threads in the valve body into which the initiator screws. This failure expels the initiator as a projectile at high velocity. However, no known failures have occurred using a NASA Standard Initiator.

3 Redundancy

Pyro valves are mounted in parallel, providing 2-for-l redundancy.

(f) Filters

1 Description

Bipropellant, low-pressure GHe, and high-pressure GHe filters are installed to prevent contamination of the bipropellant subsystem components. The GHe filters are rated at 10 mms and the bipropellant filters are rated at 15 mms. The filters are constructed of titanium and are designed and sized to trap all system and pyro valve contaminants. GHe filters are located downstream of the pyro valves to prevent contamination of the pressure regulator and check valves. The bipropellant filters are located downstream of the MMH and NTO tanks to prevent contamination of the latch valves and thrusters. The filters are compatible with MMH, NTO, GHe, gaseous nitrogen, water, isopropyl alcohol, and Freon 113. The high-pressure GHe filters have an MEOP of 4,500 psia, a proof pressure of 6,750 psia, and a burst pressure of 10,000 psia. The low-pressure GHe filters and the bipropellant filters have an MEOP of 300 psia, a proof pressure of 450 psia, and a burst pressure of 750 psia.

2 Heritage

All filters in the Mars Observer Propulsion System are manufactured by VACCO Inc., Elmonte, California; P/Ns 2631014-1, -2, -3 and 2631030-l. All filters are manufactured using photo-etched discs stacked to produce the filter element. These filter P/Ns are used on the MMAS IABS and Series 5000 spacecraft. VACCO states that this filter design has over 30 years of flight time.

3 Redundancy

Propulsion *system* filters are non-redundant.

(g) GHe Pressure Regulators

1 Description

The Propulsion System pressure regulators are **a** series-redundant design. Figure F6-7 provides a sectional view. Each regulator stage operating mechanism is enclosed in a plenum that senses the downstream operating pressure through a surge control orifice at the outlet of the regulator body. The control orifice is a Lee Jet design, as shown in Figure F6-8. Each regulator stage is capable of maintaining low system pressure within MEOP. The worst-case regulator lockup pressure, assuming a failed primary regulator stage? is 276 psia (1.903 MPa). The regulator is not in use until the high-pressure pyrotechnic valves are initiated. The maximum specified regulator leak rate is 30 cc/hr. The regulator is designed to maintain the MMH tank and NTO tank delivery pressure to the 490-N engines at 255 psia during engine firings. The regulator MEOP is 4,500 psia, proof pressure is 6,750 psia, and burst pressure is 11,250 psia.

2 Heritage

Pressure regulators are manufactured by Fairchild Controls, Germantown, . Maryland; P/N 88356001. They are packaged into several different housing combinations, but all use identical critical-flow control parts. This unit is used on LABS, Eurostar and LSAT. The Space Shuttle Reaction Control System (RCS) regulator incorporates this design as the pilot valve for a larger, two-stage regulator.

3 Redundancy

Pressure regulators are internally redundant 2-for-l units.

(h) Check Valves

1 Description

Two groups of two series-connected check valve assemblies preclude mixing of MMH and NTO after the pyro valves are fired. Each check valve assembly is internally configured to be parallel redundant. The check valve assembly (Figure F6-9) closest to the propellant tank is manufactured by Futurecraft and has two Kalrez seats. The check valve assembly (Figure F6-10)

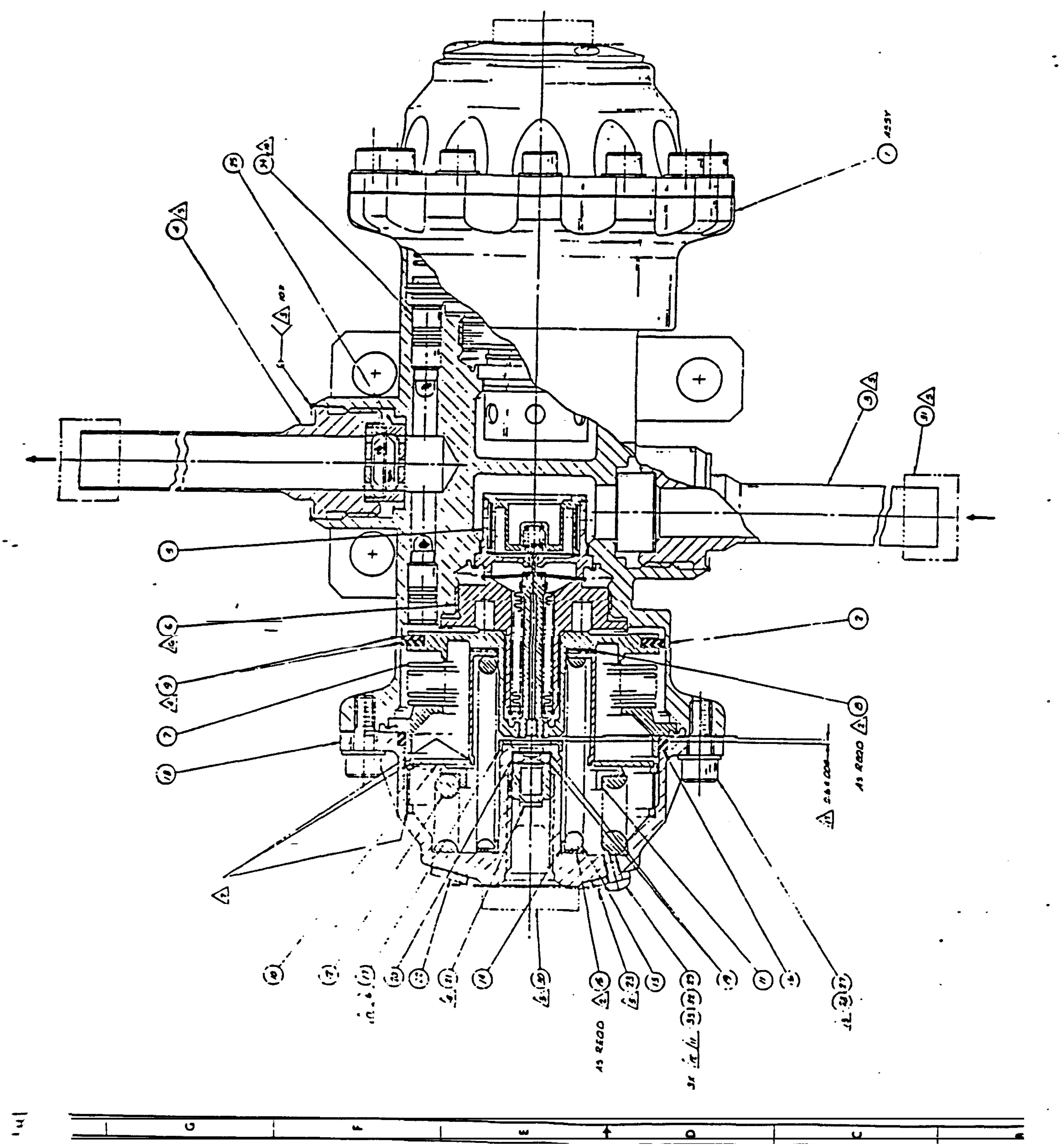
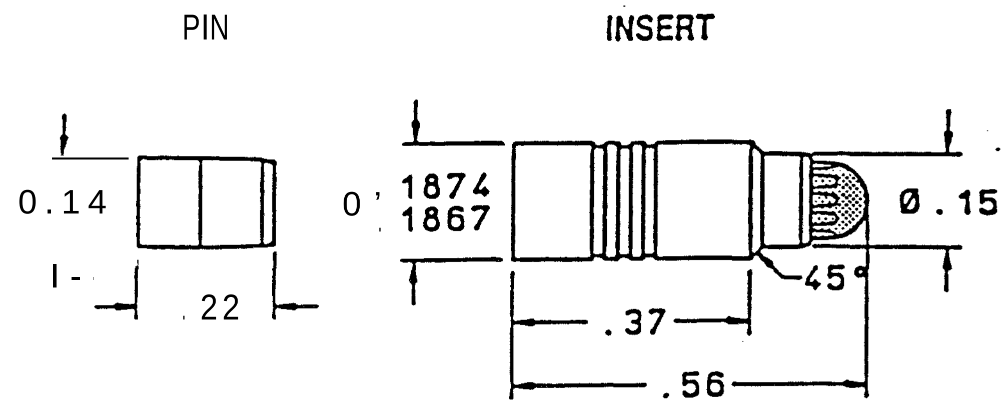


Figure F6-7. Mars Observer Pressure Regulator (Cross Section)

16 Lee Jet

CAD 300



MATERIALS			
PART	MATERIAL	SPECIFICATION.	PASSIVATE
Body	303 Cres	QQ-S-763C	MIL-S-5.002
Pin	303 Cres	QQ-S-763C	MIL-S-5002
Screens	304L Cres	QQ-S-766	'MIL-S-5002
Base Washer	304L Cres	QQ-S-766	MIL-S-5002
Braze		AMS-4774	

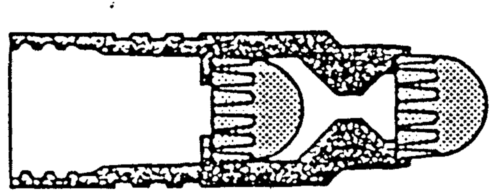
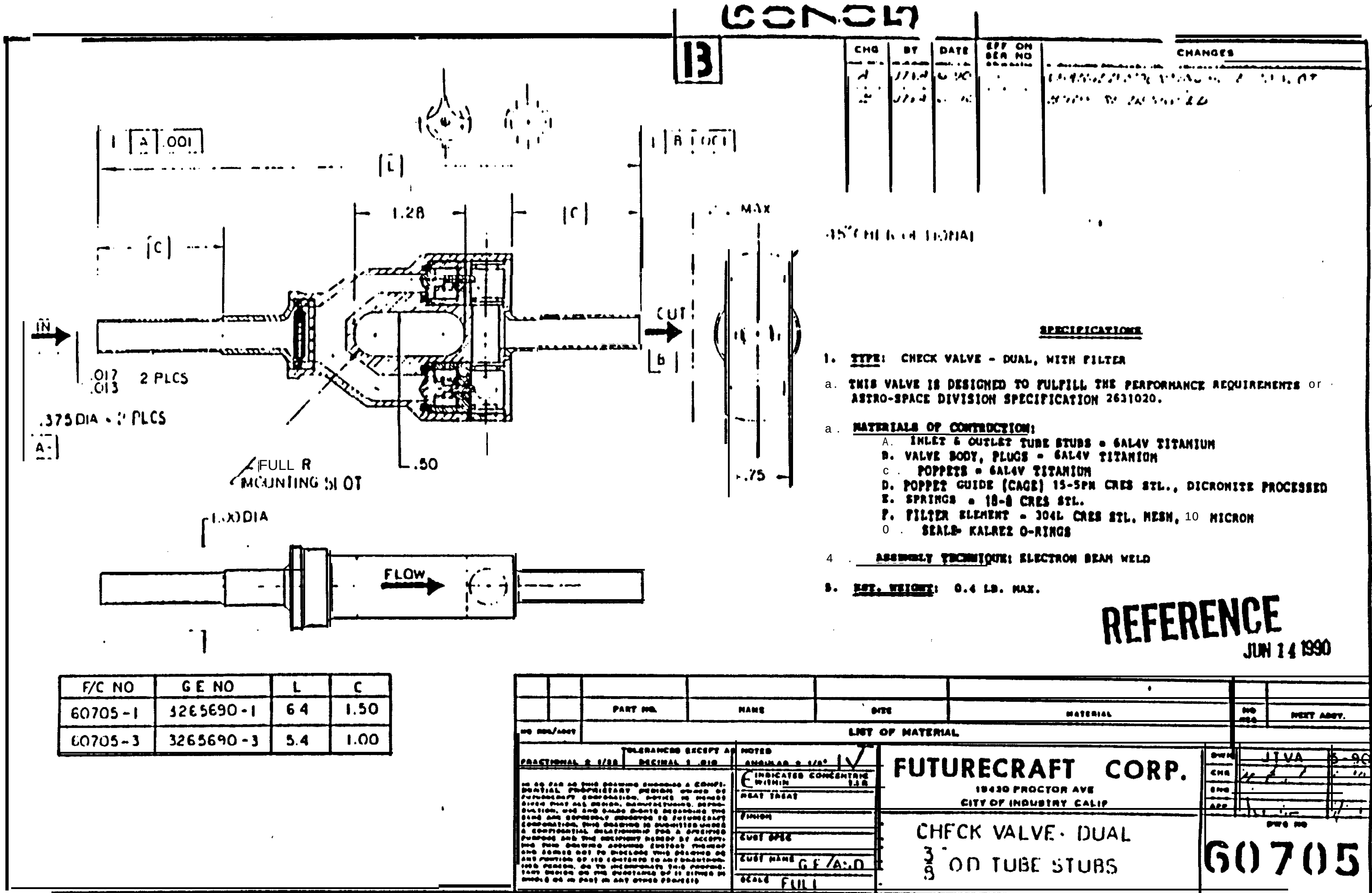


Figure F6-8. Mars Observer Pressure Regulator Control Orifice



F/C NO	G E NO	L	C
60705-1	3265690-1	6.4	1.50
60705-3	3265690-3	5.4	1.00

Figure **F6-9.** Futurecraft Check Valve Assembly

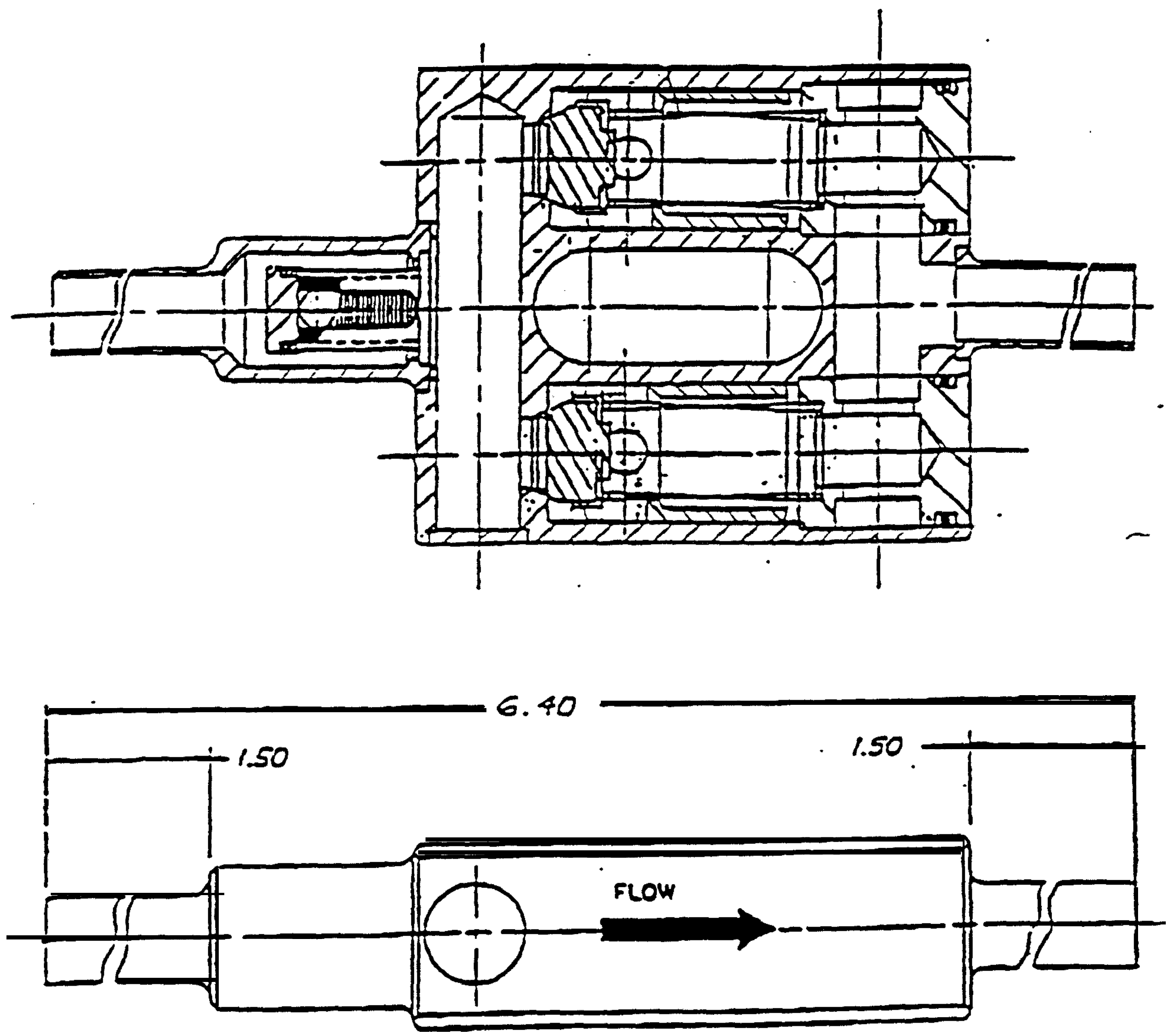


Figure **F6-10.** VACCO Check Valve Assembly on the pressure regulator side is manufactured by VACCO and has two Teflon seats. Check valve materials are compatible with MMH, NTO, water, alcohol, gaseous nitrogen, and GHe. Check valve MEOP is 300 psia, proof pressure is 450 psia, and burst pressure is 20,000 psia.

2 Heritage

Mars Observer check valves are manufactured by two companies, VACCO, El Monte, California; P/N VID10782 and Futurecraft, City of Industry, California; P/N 61705. Although of different design, both valves are designed, manufactured, and tested to the same performance specification. Both were accepted for the Mars Observer program through qualification by similarity. The valves were designed for MMAS, Previous flight experience has been on their IABS and Series 5000 programs.

3 Redundancy

Check valves are mounted in series and provide 2-for-l redundancy.

(i) MMH and NTO Tanks

1 Description

The MMH and NTO tanks (Figure F6-11) are 1.07-m (3.5-ft)-diameter, spherical, .6428-cubic meter capacity, 6A1-4V titanium tanks. The tanks are designed and tested to ML-STD-1522A using fracture mechanics analysis techniques. The tanks are mounted on flange-type mounts inside the central cylinder of the spacecraft structure. The MMH tank mounts to a central cylinder ring and the NTO rank mounts to the central cylinder lower separation ring. The maximum expected tank temperature is 50°C (122°F). A stress analysis shows that tank and mounting hardware have positive safety margins based on worst-case Titan III vibration and acceleration loads. The tank MEOP is 300 psia, the proof pressure is 375 psia, and the burst pressure is 450 psia (actual burst was demonstrated at 663 psia). This results in an in-flight burst-safety factor of 1.5. Stress analysis indicates a positive safety margin at the burst pressure based on the ultimate strength of materials. Stress analysis and environmental testing have demonstrated a design margin based on worst-case expected Titan III launch loads that exceed loading experienced during ground handling and transportation.

Tank lines and fittings are proof-pressure tested to 600 psia. The MMH tank has a maximum capacity of 532 kg (1170 lb) and the NTO tank a capacity of 832 kg (1830 lb). Each tank forging was ultrasonically inspected per MIL-STD-2154 Class AA, and, prior to closure welding, all finished surfaces were penetrant inspected per MIL-STD-6866, Type I, Method A. All shell welds were radiographically inspected per MIL-STD-453 with acceptance criteria per

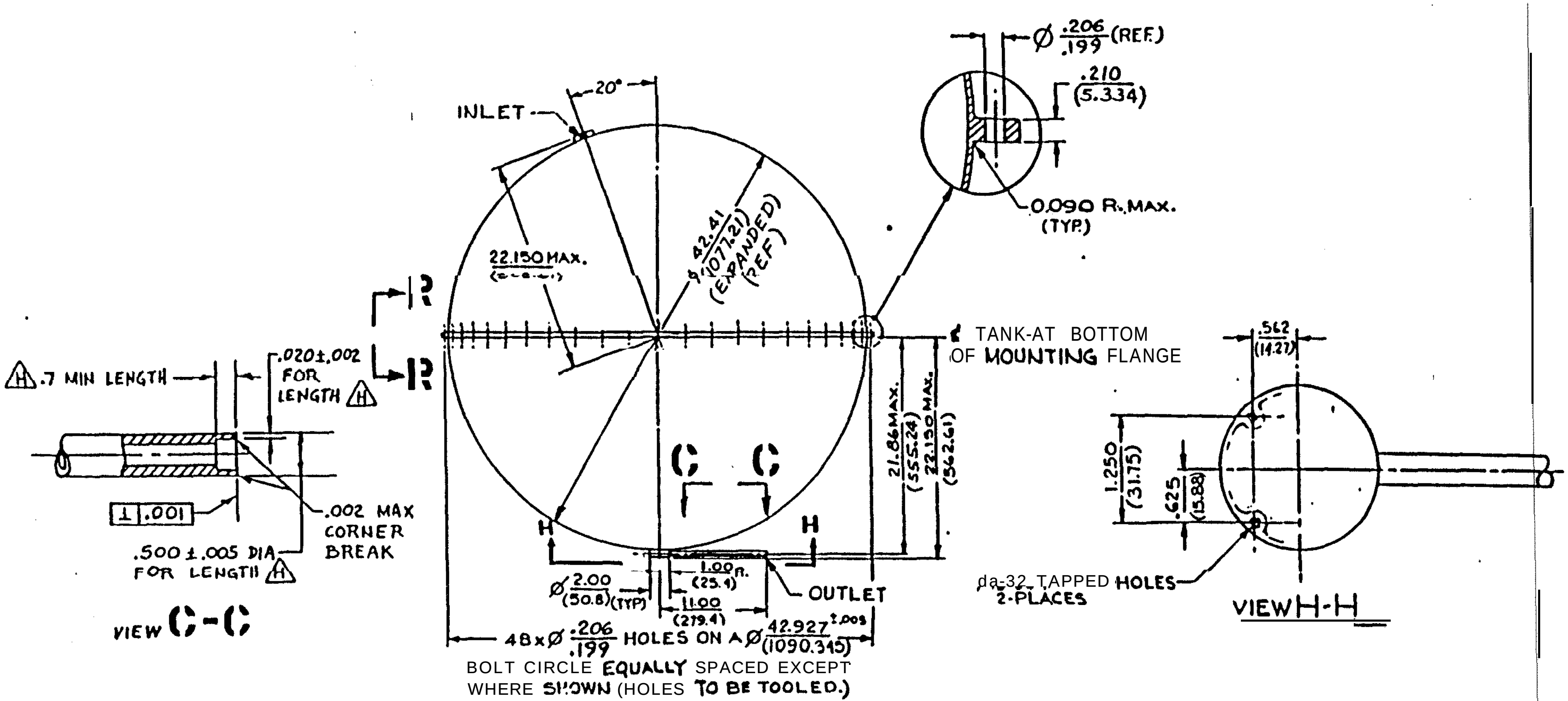


Figure **F6-11.** MMH and **NTO** Tanks

NAS 1514, Class I. All tank welds and heat-affected areas were given a penetrant inspection per MIL-STD-6866, Type I, Method A, following final welding and heat treatment. The penetrants used for the tank inspections were free of chlorides and halogenated compounds. The tank is compatible with MMH, NTO, Freon 113, helium, nitrogen, water, and isopropyl alcohol. No other fluids were used during processing or integration and test. The 6A1-4V titanium alloy is compatible with MMH and NTO per MSFC-HDBK-527E, and is highly resistant to stress corrosion cracking *per* MSFC-SPEC-522B. For in-flight temperature control, 24 flexible laminar strip heaters were bonded to the MMH tank and 30 to the NTO tank. Each tank and its heaters were overwrapped with thermal blankets consisting of two layers of goldized Kapton film separated by a polyester mesh.

2 Heritage

The bipropellant tanks are identical, manufactured by Pressure System Inc., Los Angeles, California. This is a new design, qualified to MIL-STD-1522A for this program.

3 Redundancy

The propellant tanks are non-redundant units.

(j) Latch Valves

Four single-seat, torque-motor-actuated latch valves (Figure F6-12) isolate MMH and NTO from the four 490-N thrusters and from the four 22-N thrusters. The position of each valve is sensed through a microswitch position indicator, and becomes part of the spacecraft telemetry stream. Valve actuation time is 50 milliseconds. The valve MEOP is 300 psia (inlet) and 600 psia (outlet). The proof pressure is 900 psia, and the burst pressure is 1,500 psia. The 600-psia outlet pressure takes water-hammer-effect spikes into account. The valves provide back pressure relief capability. The valve is compatible with NTO, MMH, GHe, gaseous nitrogen, Freon 113, and isopropyl alcohol.

2 Heritage

The latch valves were manufactured by Eaton Consolidated Controls, El Segundo, California; P/Ns 48006010, 48005020-101, and 48005020-102. This valve design was previously used on the TIROS program, and MMAS Series 3000 and 4000 spacecraft. The design was accepted for qualification by similarity.

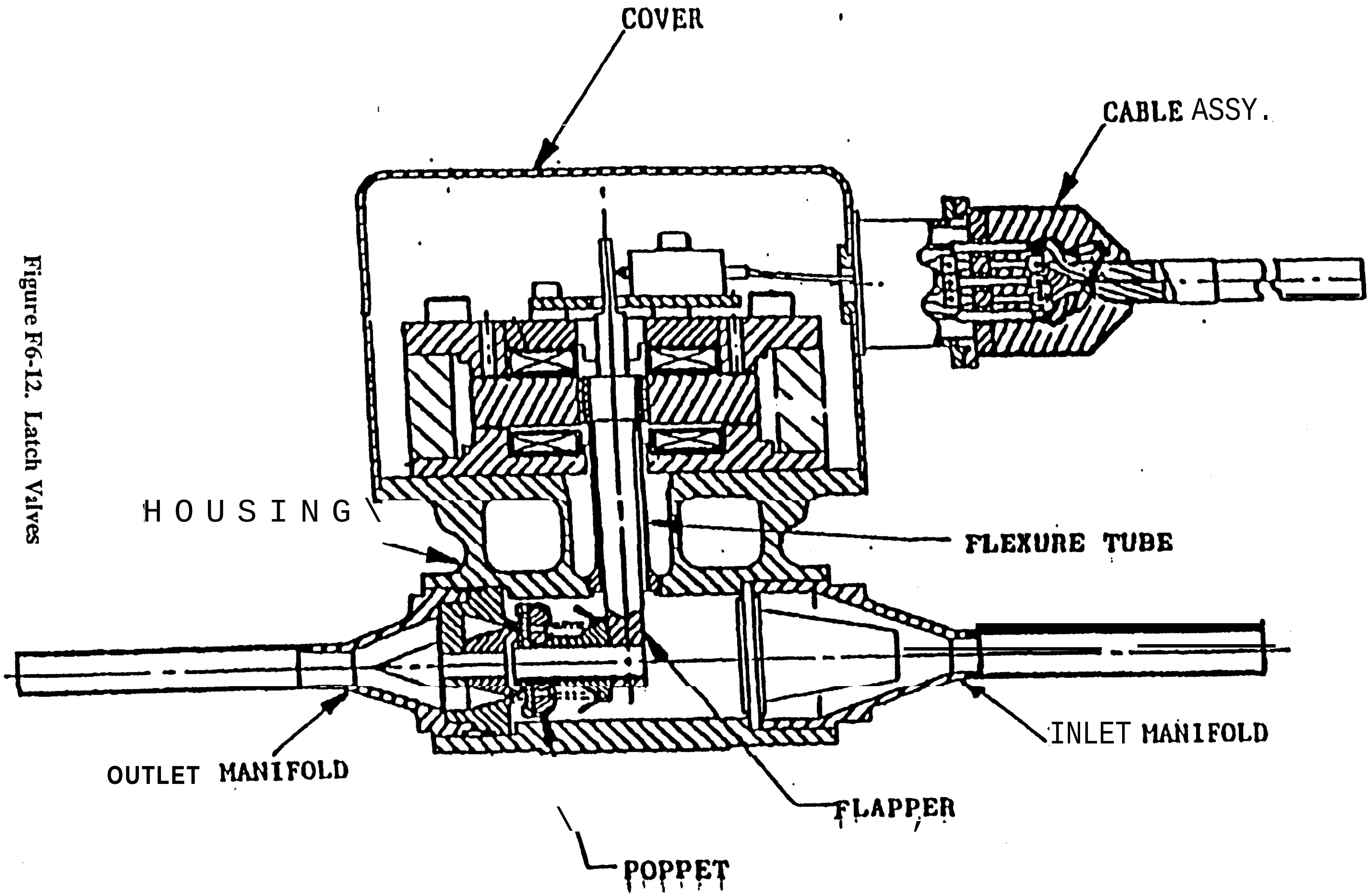


Figure F6-12. Latch Valves

3 Redundancy

Latch valves are non-redundant units; however, ample redundancy is built into the bipropellant system to overcome any single latch valve failures.

(k) 490-N Thruster Bipropellant

1 Description

Mars Observer was equipped with four 490-N thrusters. The type of thruster used has two valves that control the flow of NTO and MMH. Each valve has an armature that causes a poppet to open and close the ports, allowing oxidizer or fuel to flow from the valve, through an injector, and into the thrust chamber where combustion takes place. The valves, which were EB-welded and hydrostatically tested to ensure leak-free performance, are designed to fail safe (fail closed) when control signal is lost. The engine electrical components are explosion-proof. The thruster is compatible with MMH, NTO, GHe, gaseous nitrogen, isopropyl alcohol, and Freon 113. The thruster MEOP is 400 psia. It is proof-tested to 600 psia and has a burst pressure greater than 1,000 psi.

2 Heritage

The 490-N thrusters were manufactured by Raiser Marquardt, Los Angeles, California; P/N R4D. This thruster was originally designed for the Lunar Orbiter and Apollo programs. To date, over 600 units have been produced for use aboard spacecraft. This thruster has no record of flight failures.

3 Redundancy

Four thrusters, arranged in two-set pairs, were used aboard Mars Observer. This provides 2-for- 1 redundancy.

(I) 22-N Thruster (Bipropellant)

1 Description

Mars Observer was equipped with four 22-N thrusters. The type of thruster used has a torque-operated valve that controls the flow of NTO and MMH through separate ports. A torque motor armature uses a button assembly to open and close the ports, causing oxidizer and fuel to flow from the valve through an injector and into the thrust chamber where combustion takes place- The valves are designed to fail safe (fail closed) when control signal is lost. The valves am EB-welded and hydrostatically tested at the factory to ensure leak free performance. The thruster electrical components are explosion-proof. The thruster is compatible with MMH, NTO, GHe, gaseous nitrogen, isopropyl alcohol, and Freon 113. The thruster MEOP is 400 psia, proof pressure is 600 psia, and burst pressure is equal or greater than 1,000 psi.

2 Heritage

The Mars Observer 22-N thrusters were manufactured by Atlantic Research Corp., Buffalo, New York. This thruster design is traceable to the Minuteman Attitude Control System. More recently, it has been used on both Hughes and Loral spacecraft. As noted above, the Mars Observer version is equipped with a torque motor bipropellant valve, whereas most recent applications have used single in-line redundant valves.

3 **Redundancy**

Four thrusters, arranged in two-set pairs, were used aboard Mars Observer. This provides 2-for-1 redundancy.

(m) Contamination Control

1 Description

The propulsion system is a sealed, welded system that is kept under an inert gas blanket during shipping and storage to preclude entry of moisture and contaminates. All seal materials are compatible with hydrazine, NTO, MMH, and cleaning/referee fluids. A review of all cleaning process/requirements (GE specification 2280784) and materials listed on Propulsion System drawings was performed by MMAS to identify all cleaning solvents and verify their compatibility with Propulsion System materials and service fluids. All Propulsion System servicing equipment that interfaces with propulsion system plumbing is equipped with filters to preclude entry of physical contaminants. All Propulsion System fluids were tested for proper chemical characteristics prior to loading into the Propulsion System. The NTO loading cart is equipped with a molecular sieve to reduce the NTO iron content. Service valves are capped when not connected to the loading cart to preclude contamination. Contamination control procedures were enforced during all assembly, test, and servicing operations performed on the Propulsion system.

All inlet fittings were sized/keyed and uniquely threaded to preclude improper connection to the service cart or other flexible lines at the launch site. Written procedures governed the connection of all lines with the spacecraft. During cleaning, propellant loading, and Propulsion System testing, GE Quality Control personnel inspected and verified all fluid connections between the spacecraft and service equipment. Servicing operations were also closely supervised All propellant and high-pressure tanks were designed using fracture mechanics techniques as per GE specification 2624847, "Fracture Control Plan for Mars Observer." Fracture mechanics analyses were performed to evaluate GHe gas, and mono and bipropellant tank fracture criticality. Nondestructive evaluation and proof-tests ensured that flaw sizes were within the acceptable limits established by the fracture analysis. The tanks were stored and integrated into the Propulsion System under controlled conditions to prevent scratching or damaging exposed tank surfaces. Tank mounts were designed to preclude hazardous stress concentration points by using bearing sockets (for the MMH and NTO tanks) or spherical bearings and gimbal mechanisms (for the helium pressurant tank). The Propulsion System was made of materials with high resistance to stress-corrosion cracking per MSFC-SPEC-522B.

Components were cleaned and bagged at the manufacturer. Assembly, including all tubing welds, was performed in a clean room (class 1000). Clean gas was flowed through the system during assembly. No system cleaning was performed

2 Heritage

The cleaning and assembly approach used on Mars Observer is typical of that used on all MMAC spacecraft.

b. PROPULSION SYSTEM CONFIGURATION AND FLIGHT STATUS

The original flight plan for Mars Observer called for the propellant tanks to be pressurized five days after launch. This assumed that several large delta-v maneuvers would be needed to correct the spacecraft's trajectory to arrive at the MOI aim point. When it became clear that these large maneuvers would probably not be required (if the launch vehicle and TOS performed well, which they did), JPL decided to postpone the Pressurization Sequence until 68 hours before MOI. This decision was prompted by the desire to minimize the time during which salts might form in the low-pressure manifold if check valve leakage allowed small amounts of the propellants to mix and react there. These salts could contaminate the regulator seats, causing leakage which could over-pressurize the propellant tanks during the 1 l-month cruise phase. This condition had been observed on the Viking spacecraft and remains an ever-present concern on long-duration missions with a common propellant tank pressurization source.

Thus, for 11 months, the Mars Observer bipropellant system operated in a blow-down mode, in which the propellants were pressurized by the nominal 250-psia GHe tank ullage present at launch The low-pressure manifolds downstream of the regulators had dropped to approximately

160 psia (Figure F6-13) when the bipropellant system was commanded to be pressurized in preparation for MOL Fuel and oxidizer temperatures (Figure F6-14) were closely in accord with predictions.

The Pressurization Sequence included a number of separate commands affecting all spacecraft systems. Appendix L contains a complete list of the individual commands in the sequence. These directly affecting the Propulsion System were as follows:

- Items 585-588 (234:00:44:54), Enable/Arm Pyro Buses: Applies power to pyro valve power buses A and B.

- Item 589 (234:00:45:04), Fire Pyro Valve 7: Fires the initiator in normally closed pyro valve PW, allowing gaseous helium at 3,744 psia into the line to the regulators (R1 and R2). The regulators allow the GHe to pressurize the line through the cheek valves (CV1 and CV3) to the NTO tank, but at a reduced pressure of approximately 260 psia.

- Item 591 (234:00:50:04), Fire Pyro Valve 5: Fires the initiator in normally closed pyro valve PV5, allowing GHe to pressurize the low-pressure lines of the MMH system; regulators reduce pressure to 260 psia.

- Items 593-596 (234:00:50:14), Disarm/Disable Pyro Buses: Removes power from pyro valve pyro buses A and B.

After pressurization, a total of seven large delta-V maneuvers would be required for the MOI sequence, which was planned to begin on 24 August 1993 and would require 118 days to complete. The bipropellant system was to remain permanently pressurized to provide the large quantities of fuel and oxidizer for these burns.

C. PROPULSION SYSTEM SCENARIOS THAT COULD CAUSE LOSS OF DOWNLINK

The Propulsion System Technical Team conducted an analysis of all events that took place during the bipropellant system Pressurization Sequence to determine which items might have contributed to the loss of downlii and the apparent subsequent loss of the spacecraft Particular attention was paid to those actions that had not previously been performed during the mission (see Appendix L).

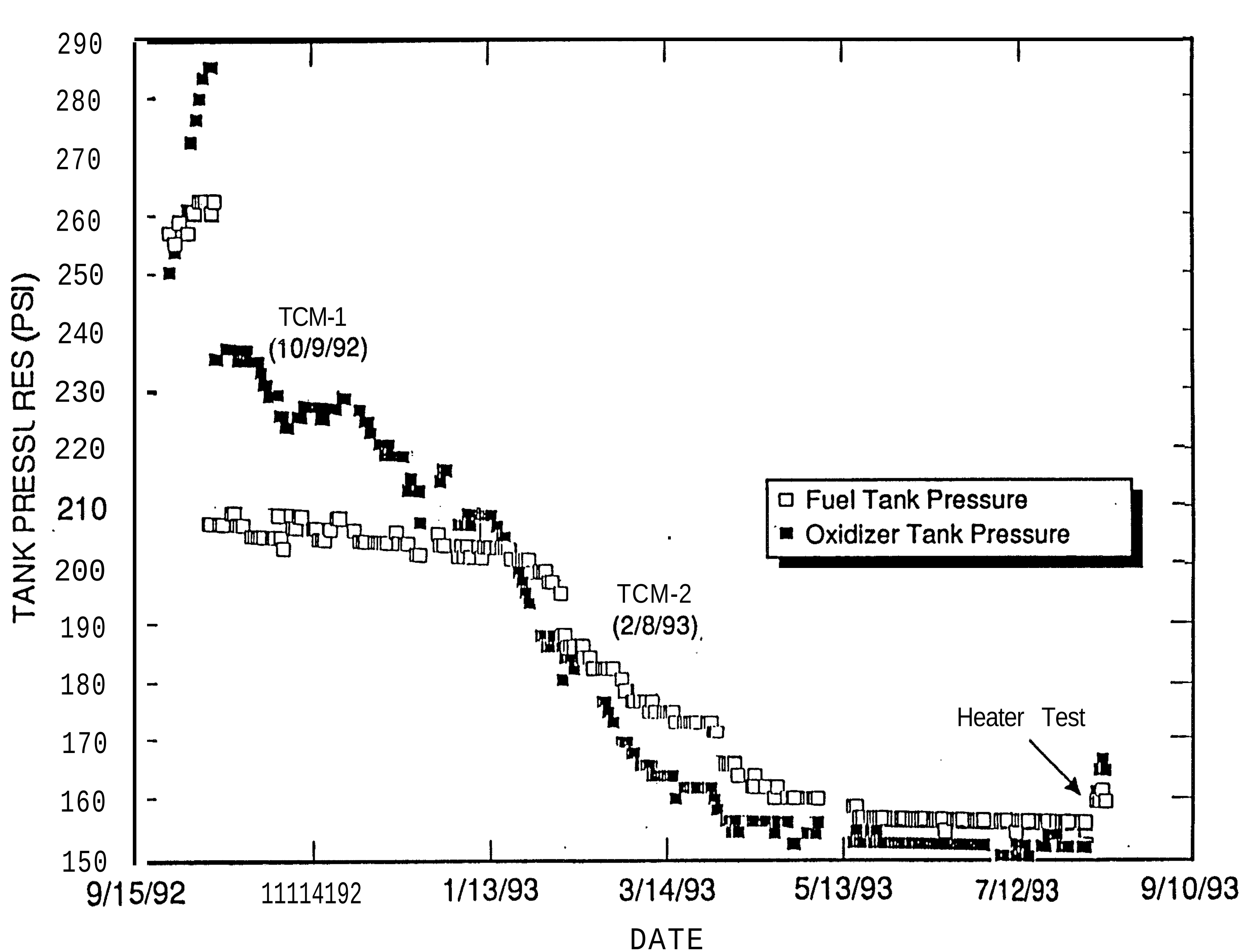


Figure F6-13. Fuel and Oxidizer Tank Pressures

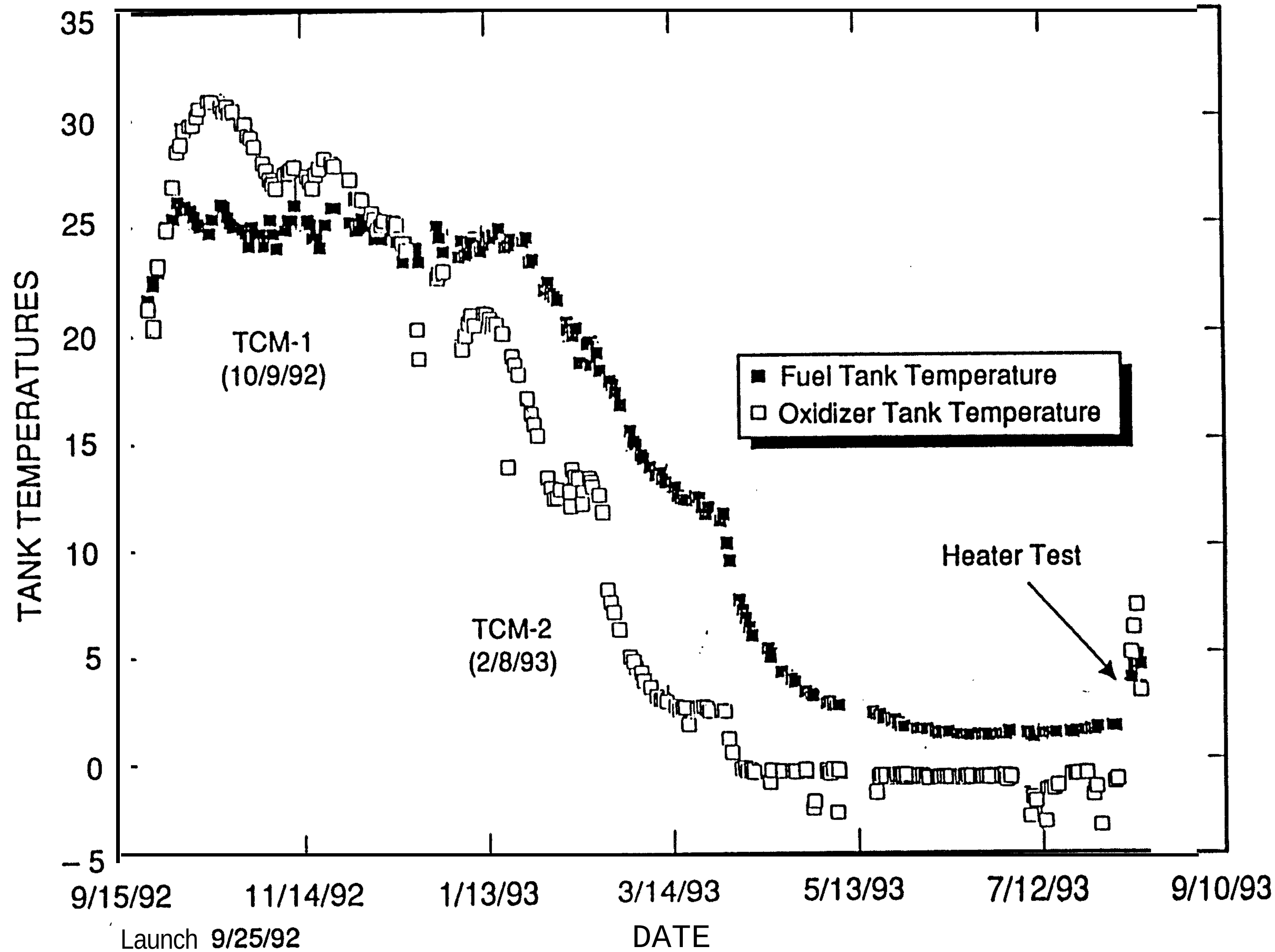


Figure F6-14. Fuel and Oxidizer Tank Temperatures

The technical team then developed a series of failure scenarios that could conceivably have been triggered by Pressurization Sequence events. Some caused a high impulse (angular momentum change or shock) that could have either damaged other spacecraft components or spun the spacecraft up sufficiently to cause signal loss or structural damage. Obviously, any kind of explosion would result in the destruction of many critical EPS, AACS, C&DHS, or Telecom System components, and would be fatal to the spacecraft.

Multiple component failures that were triggered by a single problem (material incompatibility, contamination, freezing, systemic weakness, etc.) were included in this analysis. Double failures of components operating within specified limits and structural failures were not considered credible and were dismissed. No single point failures that could lead to bipropellant system destruction were discovered.

The failure of pressurization subsystem of its components due to over-pressure is of primary concern Over-pressurization could result from the failure of a pressure regulator, or from the reaction of propellants if they somehow came into contact with each other in the pressurization subsystem Temperatures resulting from the chemical reaction of MMH and NT.0 could cause spontaneous MMH decomposition if the mixing occurred in or near the MMH tank. The combined effects of over-pressurization and chemical activity in the pyro valves was also examined

The following paragraphs discuss all of the failure scenarios postulated by the Propulsion Technical Team.

(1) Propulsion System Failure #1: Regulator Failure Caused by N TO Incompatibility

Chemical incompatibility between NTO and materials used in the construction of the regulator could cause both stages of the regulator to jam or operate improperly, allowing high-pressure GHe to over-pressurize the low-pressure side of the bipropellant system Rupture would result

(2) Propulsion System Failure #2: NTO Frozen in the Regulator Balance Orifice

If NTO migrated upstream of the check valves, a drop of it might be frozen in. each of the regulator balance orifices. This would prevent both regulator stages from sensing over-pressure on the low pressure side of the bipropellant system, rupturing the system

(3) *Propulsio System Failure #3: Contamination Blockage in the Regulator Balance Orifice*

Any particulate contamination could plug both stages of the regulator balance orifices. This could prevent both regulator stages from sensing over-pressure on the low-pressure side of the bipropellant system, rupturing the system.

(4) *Propulsion System Failure #4: Shock or Vibration Damage to Regulator Seats*

Shock induced by pyro events or launch-phase vibration could cause damage to the ruby balls that seal on hard seats in both regulator stages. The resulting leakage into the low-pressure side of the bipropellant system would cause over-pressure and rupture.

(5) *Propulsion System Failure #5: Regulator Seat Leakage Due to Contamination*

Particulate contamination lodging between the ruby ball and hard seat would cause very high leakage. Leakage through both regulator stages would over-pressurize the low-pressure side of the bipropellant system, resulting in rupture.

(6) *Propulsion System Failure #6: Locked Regulator Balance Mechanism Due to Frozen NTO*

NTO migration through the check valves could condense in a cold regulator body and lock the regulating mechanisms of both stages in the open position. This would result in over-pressurization of the low-pressure side of the bipropellant system, and subsequent rupture.

(7) *Propulsion System Failure #7: NTO in Regulator Balance Section*

A large amount of NTO leaking past the check valves might condense in the balance section of both regulator stages behind the Teflon seal ring. Since this ring seals well and liquid NTO is incompressible, the regulator would be locked open. This would result in flow into the low-pressure side of the bipropellant system and subsequent rupture.

(8) *Prop&ion System Failure #8: Component Fazlure Caused by Pyro Shock*

Acceleration forces caused by the ignition of the pyro valves could cause damage or failure of sensitive piece parts and components. This potential failure mode was also postulated by the AACS, C&DH, and Telcom Technical Teams.

(9) *Propulsion System Failure #9: Critical Spacecraft Component Damaged by Ejected NSI*

A failure in the thread in the valve body of the pyro valve would allow the NSI to be ejected as a high-velocity projectile, with potentially sufficient energy to damage critical components.

(10) *Propulsion System Failure #10: High-Pressure Gas is Expelled When Pyro Valve Case Ruptures*

When the tapered ram in the pyro valve is wedged into the valve body by the firing of the NSI, the case could be ruptured or split, allowing high-pressure GHe to escape.

(11) *Propulsion System Failure #11: NTO and MMH Migrate Through Check Valves and Mix in the MMH Pressurization Manifold*

This failure postulates that in response to the relatively cold temperature of the plumbing, NTO has migrated upstream of the check valves and is condensed in the line just upstream of the low-pressure pyro valves (PV5 and PV6). When the high-pressure pyro valve PV7 fires, some NTO remains in a "dead-ended" line and will not be pushed back into the NTO tank. In this scenario, MMH has similarly migrated upstream through the check valves and was partly filled the line immediately downstream of PV5 and PV6. When PV5 is fired, NTO is rapidly mixed with MMH, resulting in pressures and temperatures high enough to rupture the MMH line.

(12) *Propulsion System Failure #12: NTO is Injected into the MMR Pressurization Line andlor Tank and Reacts*

As in postulated Propulsion System Failure #11 above, NTO migration through the check valves would allow liquid to be trapped in front of PV5. In this scenario, however, no MMH has leaked into the manifold upstream of the MMH check valves. When PV5 fires, the NTO is pushed through the MMH lines and possibly even into the MMH tank. This would cause a violent reaction. The pressure of the chemical reaction might be high enough to rupture a line or the MMH tank itself. As a minimum, this would vent the GHe at 260 psia and the full load of MMH.

(13) Propulsion System Failure #13: NTO is Injected Into The MMH System and Reacts, Causing Decomposition of MMH

As in Propulsion System Failure #12 above, NTO could have been injected into the MMH tank after firing PV5. However, the amount of NTO injected might not be sufficient to create enough pressure to cause a rupture of the line or tank. Nevertheless, it still might generate enough heat to initiate the exothermic decomposition of MMH. The pressure thus generated would exceed the burst pressure of the MMH tank, causing rupture.

(14) Propulsion System Failure #14: Low-Pressure Pyro Valve is Fired Before . High-Pressure Pyro Valve

This failure postulates that out-of-sequence firing commands or a wiring harness error causes PV5 to be fired before PV7. If the check valves had permitted any NTO or MMH migration into the manifold, this material would not have been blown out of the lines by the GHe. Any differential pressure between the tanks and the manifold could allow either propellant to migrate in the "wrong" direction. When PV7 was fired, the migrated propellant could be pushed into the "wrong" tank, causing a rupture like that &scribed in Propulsion System Failures #12 and 13 above.

d. PROPULSION SYSTEM SCENARIOS ELIMINATED AND RATIONALE

The following 12 candidate failure scenarios have been dismissed by the Board as an explanation or contributing factor in the loss of the Mars Observer downlink.

(1) _Propulsion System Failure #1: Regulator Failure Caused by NTO Incompatibility_

The Mars Observer GHe pressure regulator has a very long history of usage (see F6a.(2)(g)) in an NTO environment. An in-depth discussion with the engineers that designed and have built these regulators for over 25 years revealed no instances of compatibility problems. The most convincing argument to support the regulator compatibility comes from its usage on the Shuttle Orbital Maneuvering System Regulators of identical design (but not internally redundant) and material are used on the Orbiter. The Orbiter NTO system is not drained or cleaned between flights. These regulators have been in Orbiter NTO systems for several years without experiencing any compatibility problems. Therefore, this postulated failure scenario was considered to be extremely unlikely.

(2) Propulsion System Failure #2: NTO Frozen in the Regulator Balance fice

The orifice between the flow-control mechanism plenum and the output side of the regulator (see Figure F6-7) is a .0054-inch diameter Lee Jet. The small orifice diameter and a number of .004-inch diameter holes, which form a filter on either side of the orifice, make an ideal spot for a drop of NTO to be captured by surface tension forces. This drop of NTO would normally be blown clear of the Lee Jet as the pressure changed at the regulator outlet and gas flowed into or out of the plenum. Temperatures cold enough to freeze NTO (-11°C) would be required to block the orifice and hold the regulator valve open. Figure F6-15 is a plot of the temperatures measured by the sensors closest to the check valves and regulator in the NTO pressurization line. It should be noted that these sensors were mounted to the inside of the Z panel of the spacecraft; the check valves and regulator were mounted to the outside of this panel (see Figure D-12). There were no direct measurements of the temperatures of the check valves and regulator. If there was good heat conductivity between the spacecraft Z panel and the pressurization plumbing, then it appears very unlikely that NTO could have frozen in the regulator.

A review of predicted and measured temperatures and an analysis (see Section e.(2) of this Chapter below) based on thermal conditions at the start of the Pressurization Sequence shows that no part of the regulator body could be colder than -4°C. This temperature is about 7°C above the freezing point of NTO. Therefore, the regulator control mechanism should respond to outlet pressure changes and control properly. However, since there were no actual temperature measurements on the regulator itself, the Board decided to leave this scenario on the list of potential failures, even though it is considered unlikely.

(3) Propulsion System Failure #3: Contamination Blockage in the Regulator Balance Orifice

The regulator inlet is protected by a filter. For contamination to be able to block the balance orifice, it must either have been built into the system or pass through the filter, pass through both regulator stage seats, travel out of the flow path and into "dead ended" balance ports, and clog both Lee Jet filters. A very large amount of contaminant would be required The regulator passed system performance and leak tests during factory acceptance testing at both the component and system level. Launch site procedures for propellant loading and pressurization require adequate filtering and sampling to ensure that the system stays clean. This failure is very unlikely because of the procedural controls, extensive filtration and the redundancy inherent to this regulator. Even though unlikely, the Board decided to retain this scenario as a potential failure, since the Board could not verify that the regulator was functioning properly after installation in the pressurization manifold (no system-level tests were performed).

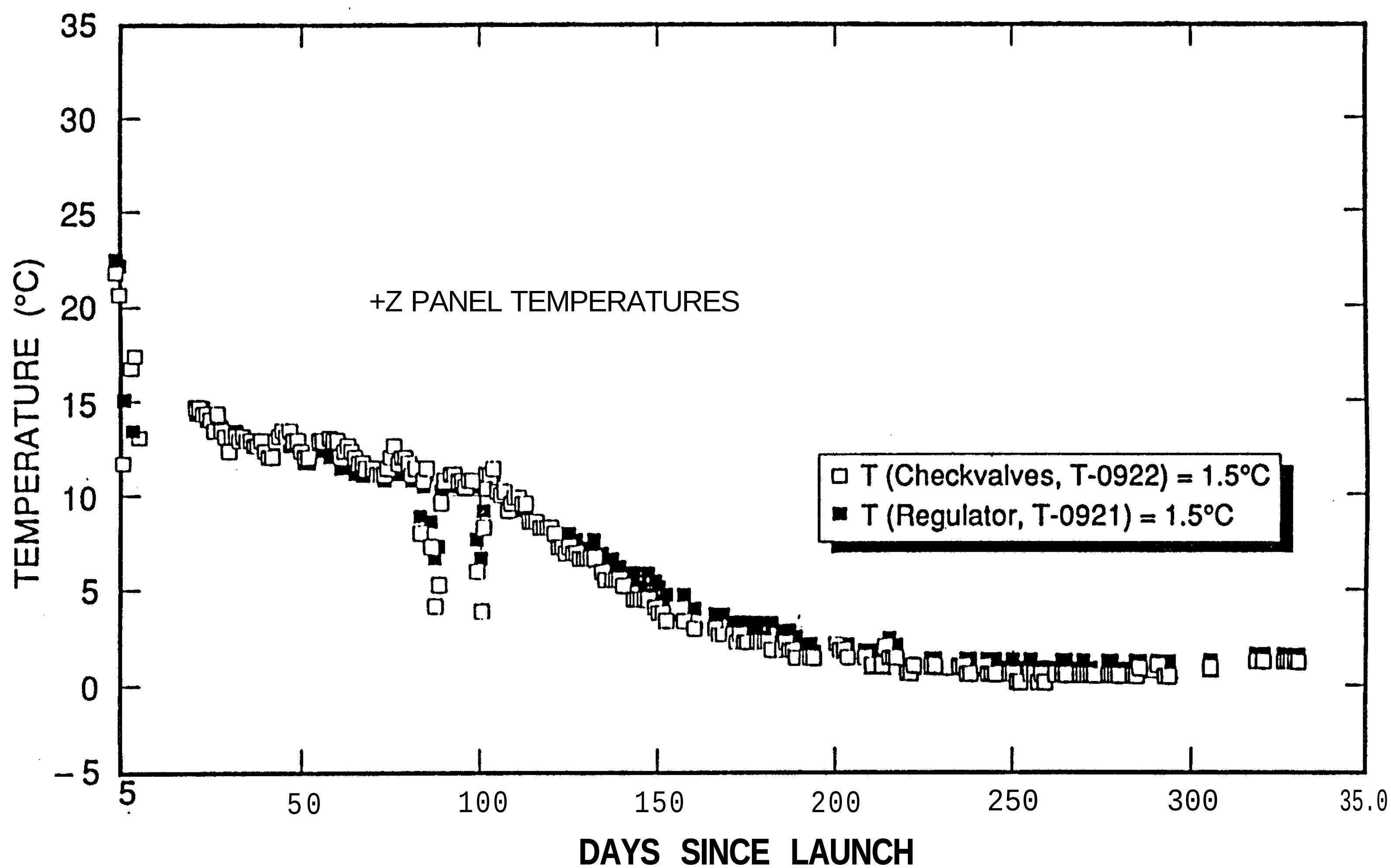


Figure F6-15. Temperatures on +Z Panel, Mars Observer

(4) Propulsion System Failure #4: Shock or Vibration Damage to Regulator Seats

The Mars Observer regulator design has been qualified to substantially higher mechanical environmental levels than were predicted for the spacecraft. During the approximately 30 years' usage of this seat design, no mechanical failures that resulted in seat leakage due to ball or seat damage have been known to have occurred. No information was discovered during the course of this investigation that would point to gross seat leakage as a potential concern.

(5) Propulsion System Failure #5: Regulator Seat Leakage Due to Contamination

Although the hard valve seat used in the Mars Observer regulator is very susceptible to contamination-caused leakage, several precautions were taken to mitigate this possibility. The regulator was operated during system pressurization without gross leakage occurring. The regulator input tubing is protected from contamination by a filter. Pyrotechnic isolation valves between the propellant tanks ensure that no bipropellant salts could form in the seat area. These factors, considered along with the series-redundant seat design, make a leak that could rupture a tank during the 14-minute period without downlink very unlikely.

(6) Propulsion System Failure #6: Locked Regulator Balance Mechanism due to Frozen NTO

This failure is not considered credible because it would require a much larger transfer of NTO across the check valves than appears possible based on the JPL leak tests (see Figure D-6 and discussion in Section e.(2) below). In addition, as discussed in Propulsion System Failure #2 above, the regulator temperatures appear to have been too high for NTO to freeze.

(7) Propulsion System Failure #7: NTO in Regulator Balance Section

As stated in-the previous failure scenario, check valve leak tests indicate that not enough NTO would migrate through the valve to prevent the proper operation of the regulator. In addition, any NTO migrating through the check valves would also have to leak past the seal ring protecting the balance section. This scenario is considered implausible.

(8) Propulsion System Failure #8: Component Failure Caused by Pyro Shock

Based upon component heritage and placement, the shock levels anticipated from pyro valve firing were predicted to be low. Preflight testing of one set of valves produced no failures. At the time of this writing, a series of post-flight pyro-firing shock tests requested by the Board bad just been completed at MMAS. Although complete data analyses were not yet available, pyro shock-induced accelerations at the locations of the RXO, IMU, and other critical components appeared (as expected) to be quite low. These tests are discussed further in Section f. below.

Although the shock from pyro-valve firing might have triggered some potential failure modes (e.g., the final breakthrough of electrical insulation, causing a short circuit (discussed in Chapter F2), it does not appear that it could have caused the failure of a spacecraft component, such as the RKO, IMU, or TWT amplifier. As a result, this failure scenario was dismissed as being implausible. See also the discussion of related postulated failure scenarios in Chapters F2, F3, F4, and F5.

(9) Propulsion System Failure #IO: High-Pressure Gas is Expelled When Pyro Valve Case Ruptures

ESA pyro valves have been sectioned for post-firing inspections, and small cracks were found in the valve body. In the housing subassembly (Figure F6-6), the body of the ESA valve is much thinner than the valves used on Mars Observer. Although no Mars Observer valves have been sectioned a visual inspection of fired valves identical to those used on Mars Observer has disclosed no cracks. In addition, no failures of the valve design used on Mars Observer have been experienced, either in ground tests or on spacecraft with the same heritage. This failure is considered extremely unlikely.

(10) Propulsion System Failure #11: NTO and MMH Migrate Throueh Check Valves and Mix in the MMH Pressurization Manifold

In this scenario, during the 1 l-month cruise phase, small amounts of both MMH and NTO migrate upstream past check valves and accumulate on both sides of PV5 and PV6. Firing PV5 would force the NTO into the MMH manifold, where they would react.

Test conducted at JPL at Board request (see discussion of Propulsion System Failure #12 in Section e. below) have shown that NTO will migrate through check valves identical to those employed aboard Mars Observer. No testing of MMH leakage through check valves has taken place to date; however, extrapolation of check valve performance with helium and NTO suggest that, due to its low vapor pressure, MMH would have a very low transport rate across the type of valve used on board Mars Observer. If a back-streaming, diffusion-type leak is assumed, the quantity on MMH in the manifold would be too small to rupture the tubing or components.

(11) Propulsion System Failure #13: NTO is Injected Into the MMH System and Reacts, Causing Decomposition of MMH

Since the &composition of MMH is an exothermic reaction, it was postulated that a small NTO/MMH reaction (hot spot) could start a self-sustaining decomposition reaction inside the MMH manifold and tank. Propulsion chemistry experts consulted by the Propulsion System Technical Team theorized that the reaction would quench; however, no test data could be found to substantiate this theory. As of this writing, a series of tests were being completed at AFPL to investigate NTO/MMH reaction temperatures, pressures, and times, and associated MMH decomposition. These tests are discussed in Section e.(2) below.

In addition, NRL has simulated the reaction of NTO and MMH in the MMH pressurization tubing (see Section e.(2) below). These simulations indicated that such reactions inside the tubing could generate temperatures high enough to initiate decomposition of the MMH. However, the simulations performed to date were not able to predict whether the decomposition wave would propagate into the MMH tank.

As discussed under Propulsion System Failure #12 in Section e. below, a more likely outcome of the injection of NTO into the MMH pressurization subsystem would be rupture of the titanium tubing at the point of reaction; Therefore, this failure scenario was dismissed as remotely possible, but unlikely to be the primary cause of the Mars Observer mission failure.

(12) Propulsion System Failure #14: Low-Pressure Pyro Valve is Fired Before High-Pressure Pyro Valve

The proper pyro valve firing sequence was verified during Mars Observer electrical functional tests, and during pyro valve shock testing performed with the spacecraft flight harness and command software during system testing. This scenario was dismissed as being implausible.

e. CREDIBLE PROPULSION SYSTEM FAILURE SCENARIOS

(1) Propulsion System Failure #9: Critical Spacecraft Component Damaged by Ejected NSI

European Space Agency test firings of pyro valves in support of the Cluster satellite have experienced initiator ejection on three out of four pyro valves fired. The fourth initiator would have ejected, but was mechanically restrained. The initiator ejection velocity was approximately 200 meters per second. The upper portion of the Mars Observer (Figure F6-6) and ESA pyro valves are very similar except for initiators. Mars Observer used an NSI, and ESA uses an initiator that is manufactured by OEA/Pyronetics to specifications that are identical to those used for the NSI; however, the ESA initiator is not identical in every way to the NSL Two NSIs and two OEA initiators were test-fired to compare their performance. The OEA initiators produced much faster pressure rise times and higher pressure pulses.

Examination of the two NSI-equipped pyro valves fired during these tests and the ten Mars Observer pyro valves that were fired during lot acceptance testing revealed that all had suffered a similar level of "erosion" of the threads in the titanium body of the valve.

About four threads are engaged to hold the initiator into the pyro valve body. Virtually all four threads erode away on valves fitted with the OEA initiator, permitting the initiator to be ejected All 12 Mars Observer-type valves showed erosion of two threads, with little damage to the other two. The damage is reasonably uniform from valve to valve. It appears to be unaffected by booster charge (80, 100, and 120%), electrically fired initiator, or sympathetically fired initiator. No damage was observed on any inconel initiator body, and none of the NSIs had been ejected.

The yield strength of titanium decreases rapidly as temperature increases. At room temperature, the valve body would have sufficient margin to prevent initiator ejection. If chemical action or detonation occurred along the thread interface, and hot gases removed parts of the titanium threads, the margin would decrease. Some margins exist with a typical chamber pressure of 30,000 psi, and two threads engaged up to about 500°F. All of the Mars Observer pyro valve firing data would indicate that the valves used with NSIs have some margin, and thus are not considered the most probable cause of the Mars Observer mishap. However, since the margin of safety for the threads is unknown, this failure mode cannot be eliminated as the cause of the Mars Observer Mission Failure.

The Board has recommended to NASA that the differences between the NSI and OEA initiator be studied to determine the cause of the ESA failures and eliminate any possibility of a similar failure occurring on a U.S. spacecraft.

(2) Propulsion System Failure #12: NTO is Injected Into the MMH Pressurization Line andlor Tank and Reacts

A simplified schematic of the pressurization side of the Propulsion System is shown in Figure F6-2. The NTO oxidizer tank was separated from the rest of the pressurization side of the Propulsion System by two check valves; one manufactured by Futurecraft Corporation, and the other manufactured by VACCO Corporation. These valves were in series for redundancy. Since the pressurization plumbing was cold for much of the cruise, the Board proposed a scenario in which NTO migrated either in liquid or gaseous form through the check valves and condensed on the cold tubing beyond (upstream of) the check valves. This would then theoretically create a situation in which liquid NTO could mix rapidly with MMH in the pressurization lines when the pressurization sequence was executed. The Board requested that tests be conducted by JPL to examine the leakage of NTO through check valves identical to those used aboard Mars Observer.

The JPL test geometry is shown in Figure F6-16 and the results of these tests are summarized in Figure D-6. The tests showed that a rather surprising amount of migration of NTO could occur. An extrapolation of these test results to the situation for the 1 l-month cruise indicates that even without a valve failure, one gram or more of NTO could have migrated through the check valves. The results also indicate that had a single failure occurred in the VACCO valve, several grams of NTO would have leaked through the valves and condensed in the upstream plumbing. A thermal prediction of the temperature of the pressurization system was performed by JPL and is shown in Figure F6-17. The locations of the NTO tank inlet, the NTO check valves (CV1, CV3), the "T" in the tubing to pyro valve PV5 and the regulator am indicated

The prediction shows that the coldest parts of the system were the "T" to PV5 and the regulator, with the regulator being slightly colder than PV5. However, PV5 and the tubing were mounted above the bulkhead on plastic standoffs, while the regulator was bolted directly to the bulkhead. There was a temperature sensor mounted on the inside of the bulkhead just below the regulator that was reading 1.5°C prior to the loss of downlink (Figure F6-15). It has been argued that since the regulator was bolted directly to the bulkhead, it would have had approximately the temperature of the bulkhead. This argument was used in the discussion of Propulsion System Failure #2 above, in which frozen NTO was proposed to cause the failure by sealing the sensor ports in the regulator. If one accepts the warm regulator argument and accepts the Figure F6-17 data as showing that the regulator temperature was 1.5°C, then the "T" to PV5 becomes the coldest part of the pressurization manifold In this case, the NTO would be expected to migrate to the vicinity of PV5 and condense. This situation is actually intuitively obvious, since the NTO check valves were deliberately heated to 4°C and the temperature on the other side of the bulkhead from the regulator was measured at 1.5°C. If some of this condensed NTO were swept into the MMH lines and mixed with MMH during the Pressurization Sequence, a hypergolic reaction could occur, releasing on the order of 100 kilocalories per mole of mixed NTO and MMH.

A hypergolic reaction caused by MMH and NTO mixing in the pressurization system is a potential problem for any spacecraft that uses a common pressurization source such as employed by Mars Observer. Tests conducted by JPL (see Section f.(l) below) to quantify the NTO that could backstream through the check valves into the pressurization system have shown that sufficient NTO could have migrated through the check valves during the 1 l-month cruise phase to cause concern The probability of a reaction occurring that could damage the titanium tubing in the pressurization system is dependent upon several factors:

CHECK VALVE DIFFUSION/PERMEATION TEST SCHEMATIC

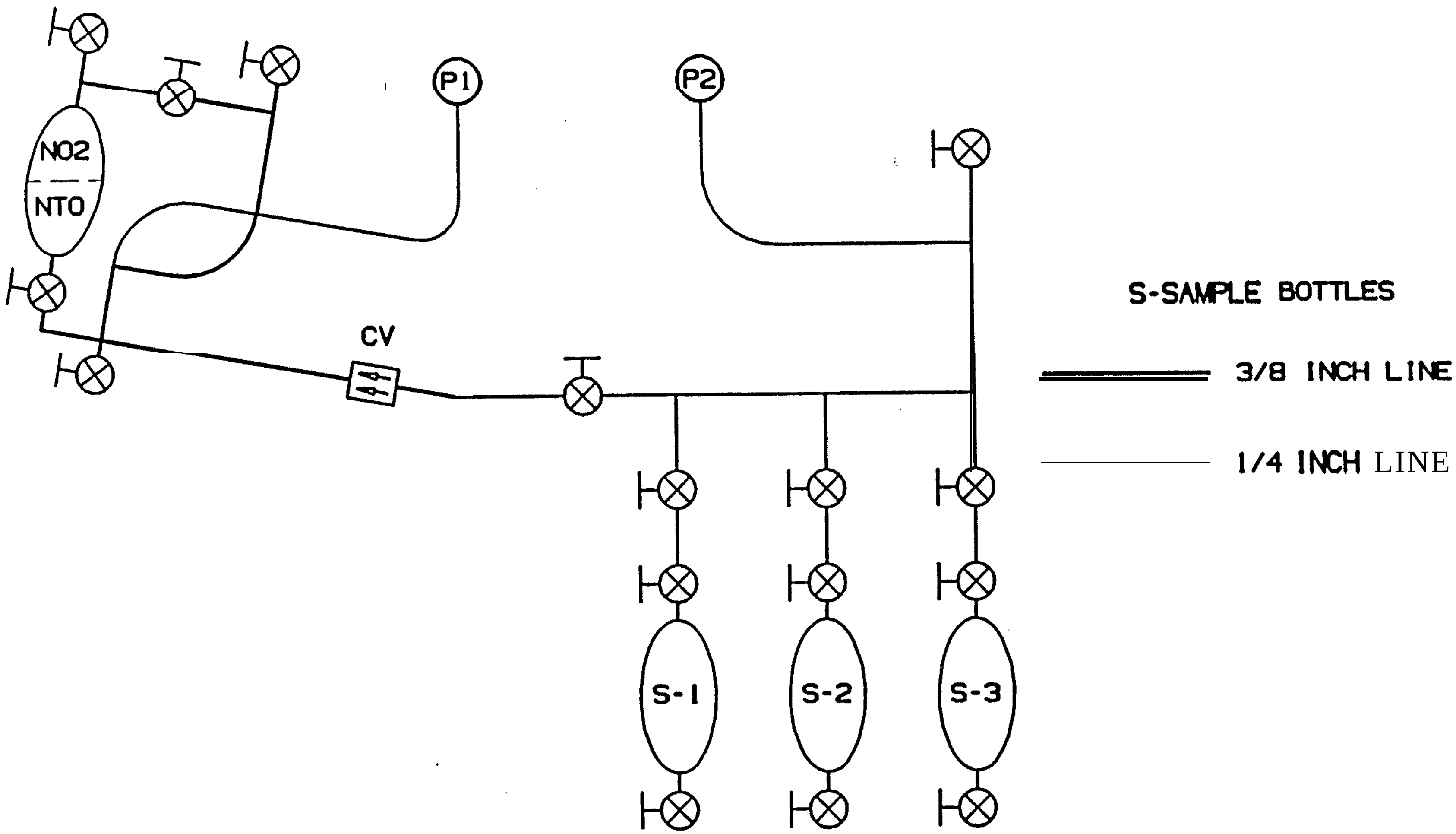


Figure F6-16. NTO Check Valve Diffusion/Permeation Test Schematic

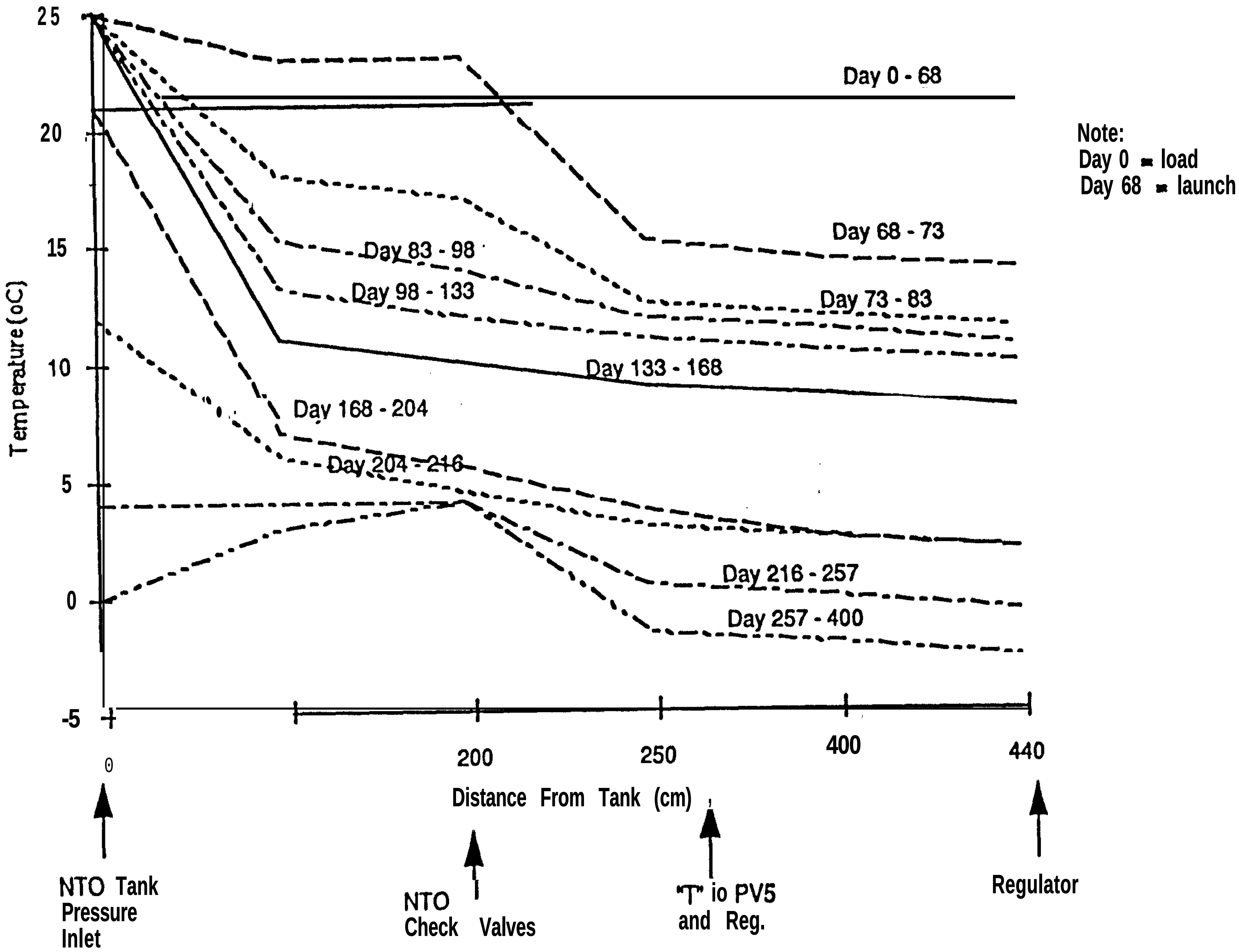


Figure F6-17. Temperature Predictions For Mars Observer Propellant Pressurization Subsystem

- Quantity of NTO and MMH present
- Free volume to dissipate the pressure of reaction products
- Vapor, liquid, or atomized phase of propellants
- Temperature of propellants and components
- ○♏♋♍⧫♓□■ ⧫♓❍♏⬧
- MMH decomposition associated with reaction
- Heat dissipation into propellants and plumbing
- Temperature/strength relationship of titanium tubing

A series of NTO/MMH propellant interaction tests were undertaken by AFPL. Figure F6-18 is a schematic of the AFPL test rig. As of this writing, 12 separate tests had been conducted, with highly variable results. Ten of these tests produced reactions resulting in slight temperature increases and no significant pressure pulses. One test produced an 8000 psi spike on one transducer channel that cannot be verified by other system instrumentation (possible recording problems). The remaining test verified pressure pulses throughout the system The stainless steel tube bulged ,008 to .010 inch, and distorted a flare tube ferrule. This damage indicates an internal pressure of 11,000 to 12,000 psi in the stainless steel tubing (0.375-inch diameter, 0.035-inch wall thickness). Static weld-verification tests on the Mars Observer titanium tubing (0.0375-inch diameter, 0.015-inch wall thickness) has shown burst pressures (at room temperature) of between 10,500 and 12,500 psi. When one considers the reinforcing effect of the ferrule sleeve on the stainless steel tubing, and the greater strength of stainless steel compared to titanium (especially at higher temperatures), it is reasonable to assume that if a titanium tube had been used in the AFPL test rig, there is a high probability that it would have burst.

The last set of studies ordered by the Board to evaluate the likelihood of this postulated failure scenario were undertaken by NRL, and were a series of calculations and numerical simulations of the potential effects of NTO mixing with MMH in the MMH pressurization manifold It was intended to answer the question: How much NTO would have to mix with MMH to create a serious problem?

The tubing used in the Mars Observer propellant pressurization plumbing is 3/8-inch diameter, 0.015-inch thickness titanium alloy (Ti-3AL-2.5V). For titanium:

$$\text{Density} = \rho = 4.507\ \text{gm/cm}^3$$

$$\text{Specific heat} = c_p = .124\ \text{cal/(gm)(°C)}$$

$$\text{Heat of fusion} = 104\ \text{cal/gm.}$$

PHILLIPS LAB NTO/MMH INTERACTION TEST

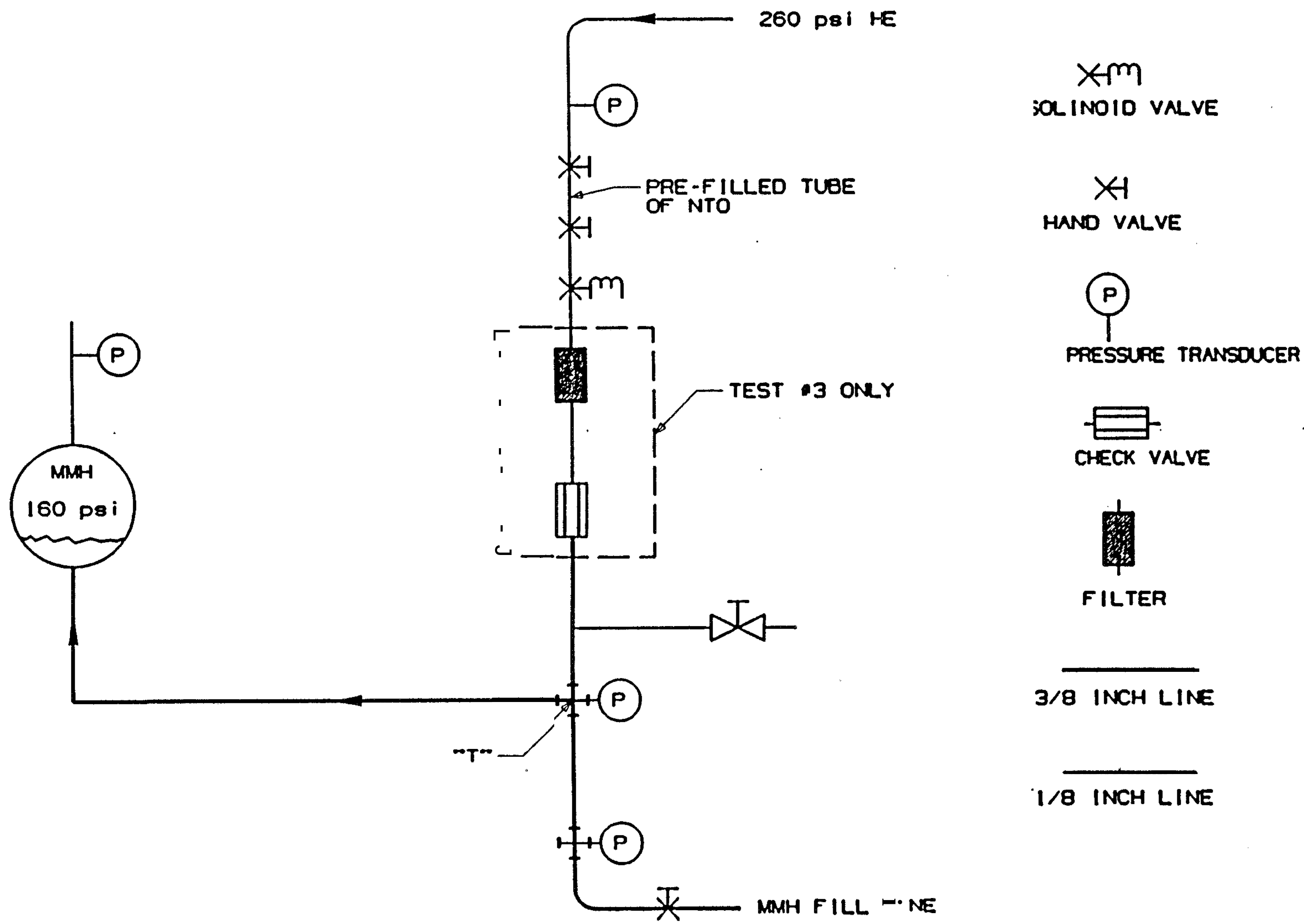


Figure F6-18. NTO/MMH Interaction Test Geometry

From these values, one calculates that it would require 106 calories to raise a one-centimeter length of this tubing from 0°C to its melting temperature of 1668°C. It would take an additional 53 calories to melt the tubing (i.e., 159 calories to melt a one-centimeter length of the tubing, neglecting a small amount of heat due to phase transformation). The combustion temperature of NTO/MMH is about 3000°C. NTO uniformly mixed with MMH at liquid densities is theoretically able to release about a thousand calories per centimeter length of this tubing (0.5 gms NTO, 0.3 gms MMH). This static melting of the tube would not actually develop, since the rapid generation of pressure by the combustion process would quickly force fluid dynamic motion. Nevertheless, this simple calculation illustrates that a few tenths of grams of NTO moving into the MMH line is a matter of serious concern.

The actual situation is much more complex than that described above. It is a dynamic situation involving mixing, heat generation, thermal conduction and fluid flow. Any self-consistent solution requires numerical simulation. Some simplified numerical solutions will be presented below. However, some additional insight can be gained analytically. For example, the characteristic time τ required to raise the temperature of a thin-walled titanium tube of thickness ℓ is:

$$\tau = \frac{\ell^2 \rho c_p}{\kappa}$$

where K is the thermal conductivity. For titanium,

$$K \sim 4 \times 10\text{-}2 \ \mathrm{cal/(cm)(sec)(°C)}$$

Hence:

$$\tau \sim 14 \text{ ms}$$

Since thermal diffusion varies as the square root of time, the temperature at the outer surface of the tube after 1.4 ms would be about one-thii of the temperature of the combusting fluid at the inside surface of the tube. (These times are referenced to the time at which combustion began.)

Another parameter that must be examined is the yield stress of titanium as a function of temperature. This is shown in Figure F6-19 for the titanium alloy (Ti - 3Al - 2.5V) used in this application. The yield stress of titanium declines very rapidly with temperature, losing essentially all of its strength above 500°C. For our application, we must replot Figure F6-19, replacing yield stress with the pressure in the tube that would produce the yield stress. This is shown in Figure

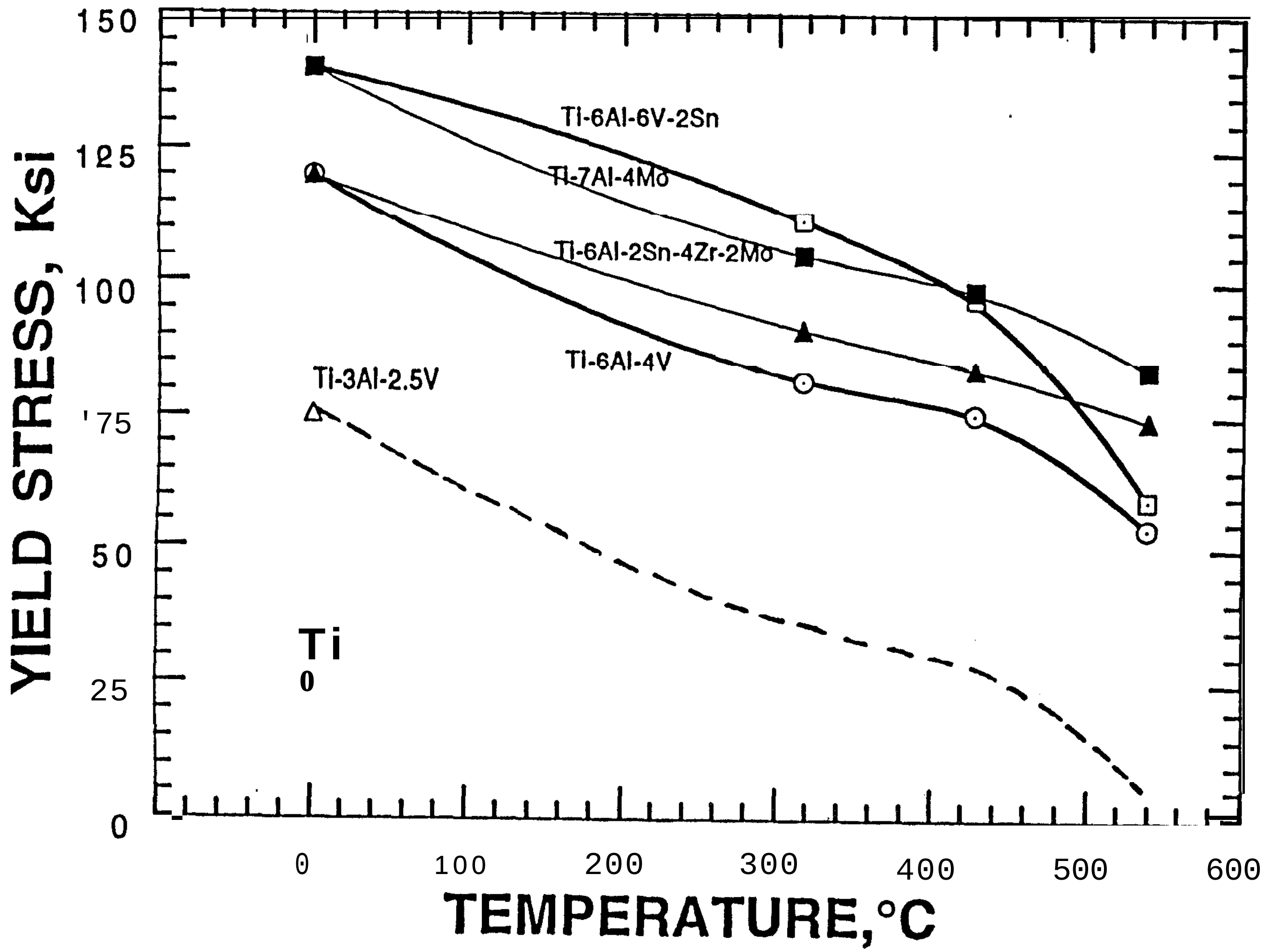


Figure F6-19. Yield Stress of Titanium and Titanium Alloys as a Function of Temperature

F6-20. As can be seen from this figure, the static pressure that the titanium line can contain drops from about 104 psi at 0°C to about 10^3 psi at 500°C. Beyond 500°C, the tubing has essentially no strength. From the numbers presented earlier, one can determine that 32 calories would be required to raise a one-centimeter length of the titanium tubing from 0°C to 500°C. The amount of NTO required to produce 32 calories through a reaction with MMH is about 20 milligrams.

One must now examine the pressure time histories that might develop if NTO were mixed with MMH inside the titanium tubing. For the sake of calculation, consider the situation where a given amount of NTO moves through filter F2 (Figure F6-2) and then through check valve CV2 and into the MMH line, where it rapidly and completely mixes over a distance of 5 cm and reacts with the MMH. Since the state of the MMH in this line is not known, two different conditions to represent different extremes will be examined. One case assumes that the line contains only ten percent MMH, while the other case assumes that the line *contains* ninety percent MMH. Numerical simulations of the chemically reactive flow that would develop have been performed for each case.

A simplified chemically reactive flow model was developed by NRL to provide some insight into the dynamics that might evolve. The gas-phase chemical reaction of NTO and MMH is approximated by the equations,

$$\frac{\partial N^g_{NTO}}{\partial t} = -3\, C_R\, \alpha(T)\, N^g_{NTO} \left[\frac{N^g_{MMH}}{10^{22}}\right],$$

$$\frac{\partial N^g_{MMH}}{\partial t} = \frac{2}{3} \frac{\partial N^g_{NTO}}{\partial T},$$

$$\frac{\partial N^g_{prod}}{\partial t} = -\frac{5}{3} \frac{\partial N^g_{NTO}}{\partial T}, \text{ and}$$

$$\frac{\partial E}{\partial t} = -\Delta E \frac{\partial N^g_{NTO}}{\partial T}.$$

The nondimensional factor a(T) varies from 1 at low temperature, to 100 as the temperature approaches infinity to accelerate the reaction. Energy release is governed by an input parameter AE that can be varied, generally in the range 60 - 120 kilocalories per mole of reactants, and nominally specified as 100 kcal/mole. Superscripts g and d label 'gas' and 'droplet' phase components of the two fluids (NTO and MMH) respectively. A constant factor, nominally 0.5,

YIELD PRESSURE VS TEMPERATURE FOR TITANIUM & TITANIUM ALLOYS

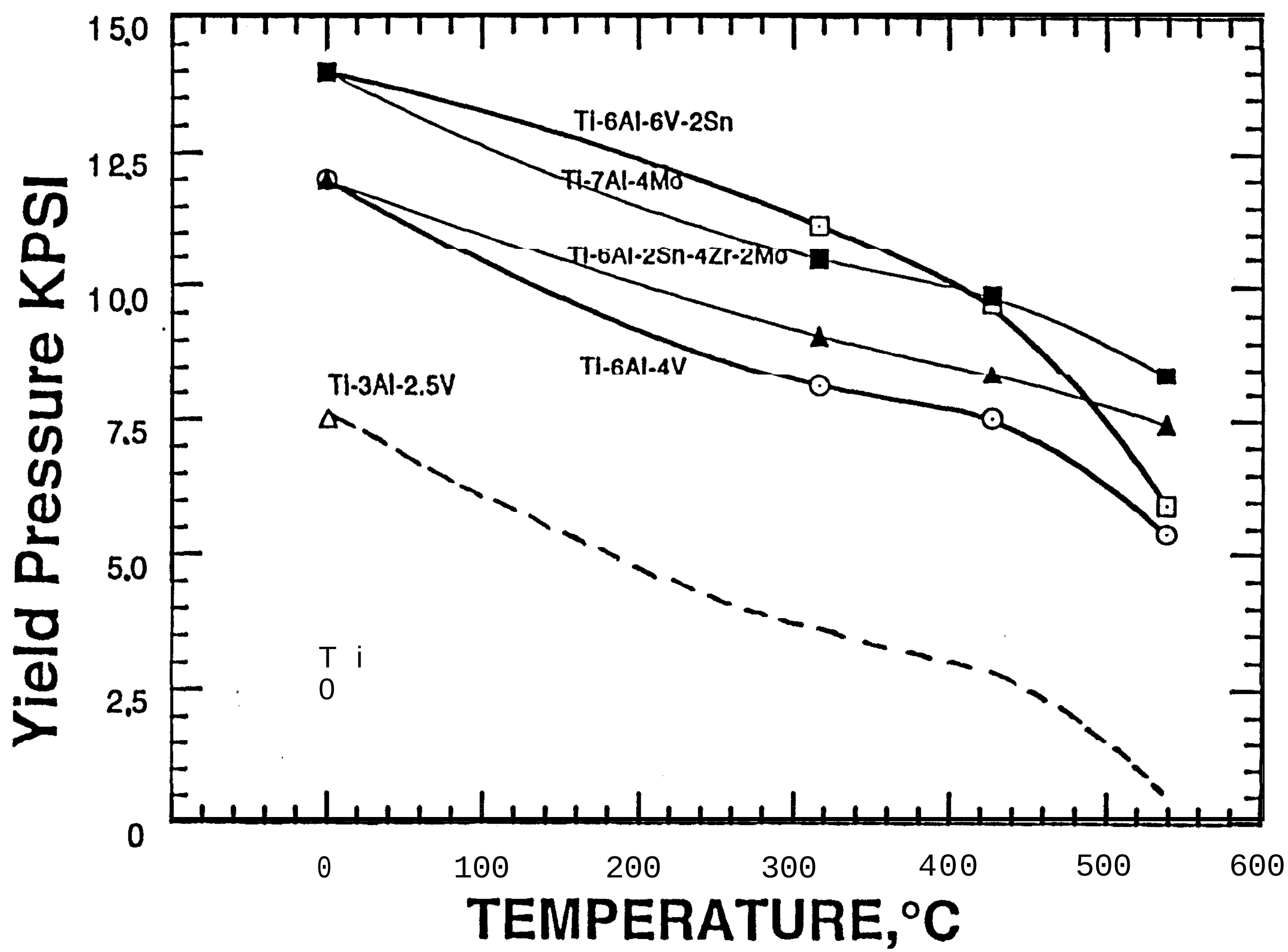


Figure **F6-20.** Yield Pressure for Titanium and Titanium Alloys as a Function of Temperature specifies the fraction of the liquid present that resides in droplets. The remainder of the liquid takes up volume, coating the tubing walls. The liquid density for both components of **NTO** and both components of MMH **vary** with density according to the appropriate Tait's Law.

The **liquid droplets** vaporize if the vapor pressure at the current gas temperature is higher than the **actual** species partial pressure. Additional gas reactants are also assumed to be generated by hypergolic interaction of the two liquids if they are both present in droplet form. These two effects **are** represented by the simple equations:

$$\frac{\partial N^{d}_{NTO}}{\partial t} = -3H_R\,\gamma(v)\,N^{d}_{NTO}\left[\frac{N^{d}_{MMH}}{10^{22}}\right] \cdot V_R\,\beta(v)N^{d}_{NTO}\,\frac{\left[N^{vp}_{NTO}(T) - N^{g}_{NTO}\right]}{N^{max}_{NTO}}$$

$$\frac{\partial N^{d}_{MMH}}{\partial t} = -2H_R\,\gamma(v)\,N^{d}_{MMH}\left[\frac{N^{d}_{NTO}}{10^{22}}\right] - V_R\,\beta(v)N^{d}_{MMH}\,\frac{\left[N^{vp}_{MMH}(T) - N^{g}_{MMH}\right]}{N^{max}_{MMH}}$$

Here $N^{vp}_{NTO}(T)$ and $N^{vp}_{MMH}(T)$ am the temperature-dependent vapor pressures of **NTO** and MMH respectively varying the term β (v) between 0.1 and 1 allows for the increase of the droplet vaporization rate due to fluid motion and turbulence. The nondimensional factor for the hypergolic breakdown, g(v), is set to 1 + $\beta(v)$ for simplicity, ensuring that mixed fluids generate reactants even in the absence of motion.

There is also a reaction specified for decomposition of the MMH in gas phase with energy release, provided the temperature is above 300°C:

$$\frac{\partial N^{g}_{MMH}}{\partial t} = -D_R\,\delta(T)\,N^{g}_{MMH}$$

$$\frac{\partial N^{g}_{prod}}{\partial t} = 3\,\frac{\partial N^{g}_{MMH}}{\partial T},$$

$$\frac{\partial E}{\partial t} = -40\,KCAL/Mole\,\frac{\partial N^{g}_{MMH}}{\partial T}$$

Here the nondimensional factor $\delta(T)$ is zero for T $\leq$ 300°C, increases to 1.0 at 500°C, and asymptotes to 10.0 as the temperature approaches infinity. This reaction is fed by vaporization of the liquid MMH present when the temperature is high enough, and thus is effectively a slow two-stage process as modeled here.

The rates in the numerical model are controlled globally by the four constants indicated above, C_R for the exothermic chemical reaction, H_R for the hypergolic breakdown of the two liquids, V_R for the vaporization of the two liquids, and D_R for the high-temperature, exothermic decomposition of MMH. The nominal values used for these four rates are C_R = 103, H_R = 103, V_R = 103, and D_R = 102. These are 1 ms, 1 ms, 1 ms, and 10 ms respectively, for the characteristic reaction times at moderate to low temperature. The decomposition of MMH is known experimentally to take on the order of 10 ms, even though it is quite energetic.

This simplified chemical kinetics model was incorporated into a one-dimensional fluid dynamics code for calculating the flow along the pipe. Thermal excursion of the pipe wall was not treated in this model. The computations were performed assuming a 2-meter-long pipe blocked at one end, and dumping into a 1000-cc volume (representing the MMH tank) at the other end A series of simulations were performed with various amounts of NTO distributed through the volume of the first 5 cm of the blocked pipe. In the calculation where the line was 10% filled with MMH, the several values of NTO introduced into the first 5 cm of the tube were 2 grams, 0.2 grams, and 0.02 grams. Figure F6-21 shows the pressure predicted at the closed end of the tube as a function of time. Similarly, Figure F6-22 shows the temperature at the closed end of the tube as a function of time. In all cases, substantial pressures and temperatures were predicted (Recall, however, that the 20 milligram case does not release enough total energy to significantly heat the tubing, unless decomposition of MMH actually occurs.) Figure F6-23 shows the integrated product mass produced. It is clear that late in these simulations, MMH has begun to decompose as a result of the high temperature predicted Figures F6-24 through F6-29 provide the predicted pressure and temperature profiles down the pipe at various times for the several cases simulated.

In the calculation where the line was 90% filled with MMH, the quantities of NTO introduced into the first 5 cm of the tube were 2 grams, 0.2 grams, 0.02 grams, and 0.002 grams. Figure F6-30 shows the pressure-vs.-time predicted at the closed end of the tube. Similarly, Figure F6-31 shows the temperature-vs.-time at the closed end of the tube. In this case, substantially higher pressures are reached than was the case in which the tube contains only 10% MMH. This is because of the tamping by the high MMH fill, which slows pressure relief and reduces the volume available to the expanding gas. Only the 0.002 gram case showed little effect on a 10 millisecond-time scale. Figure F6-32 shows the integrated product mass produced. Here again, decomposition of MMH is evident late in the simulations. Figures F6-33 through F6-40 provide the predicted pressure and temperature profiles down the pipe at various times for the several cases simulated.. In this case, the presence of shock waves is evident.

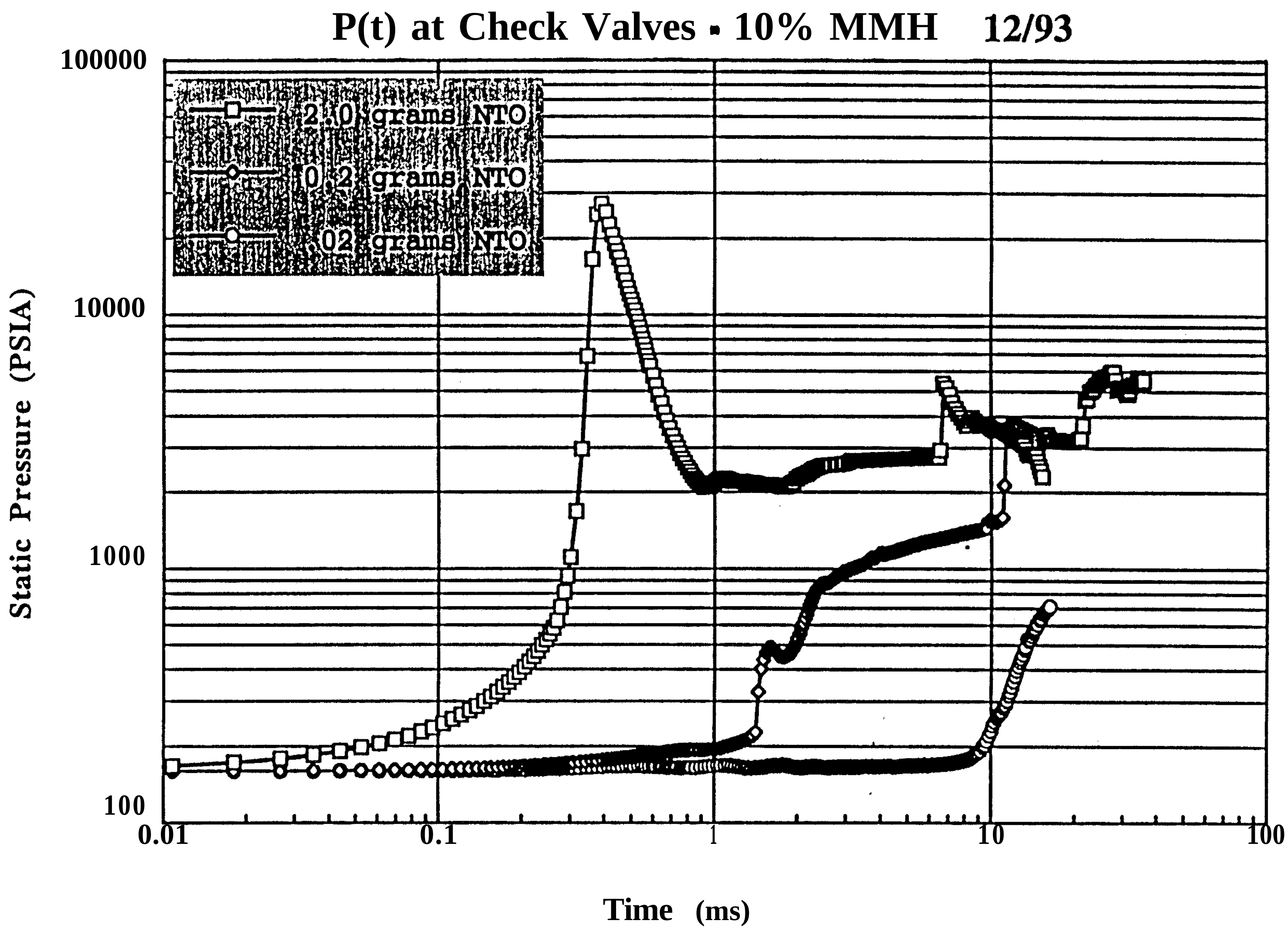


Figure F6-21. Predicted Pressure vs Time Check Balves: NTO Mixing in Tube 10% Filled with MMH

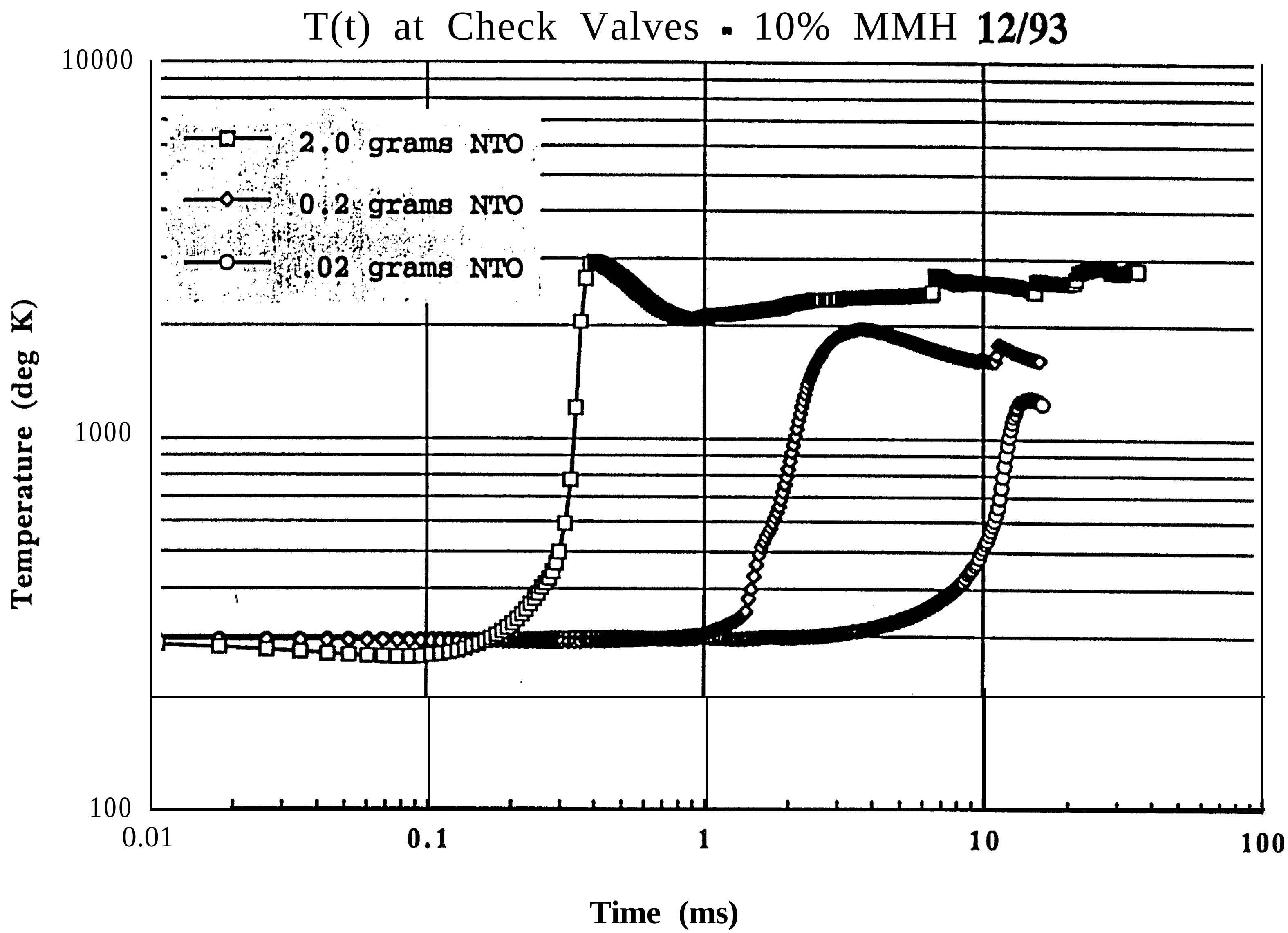


Figure **F6-22.** Predicted Temperature **vs** Time at Check Valves: **NTO** Mixing in Tube 10% **Filled with** MMH

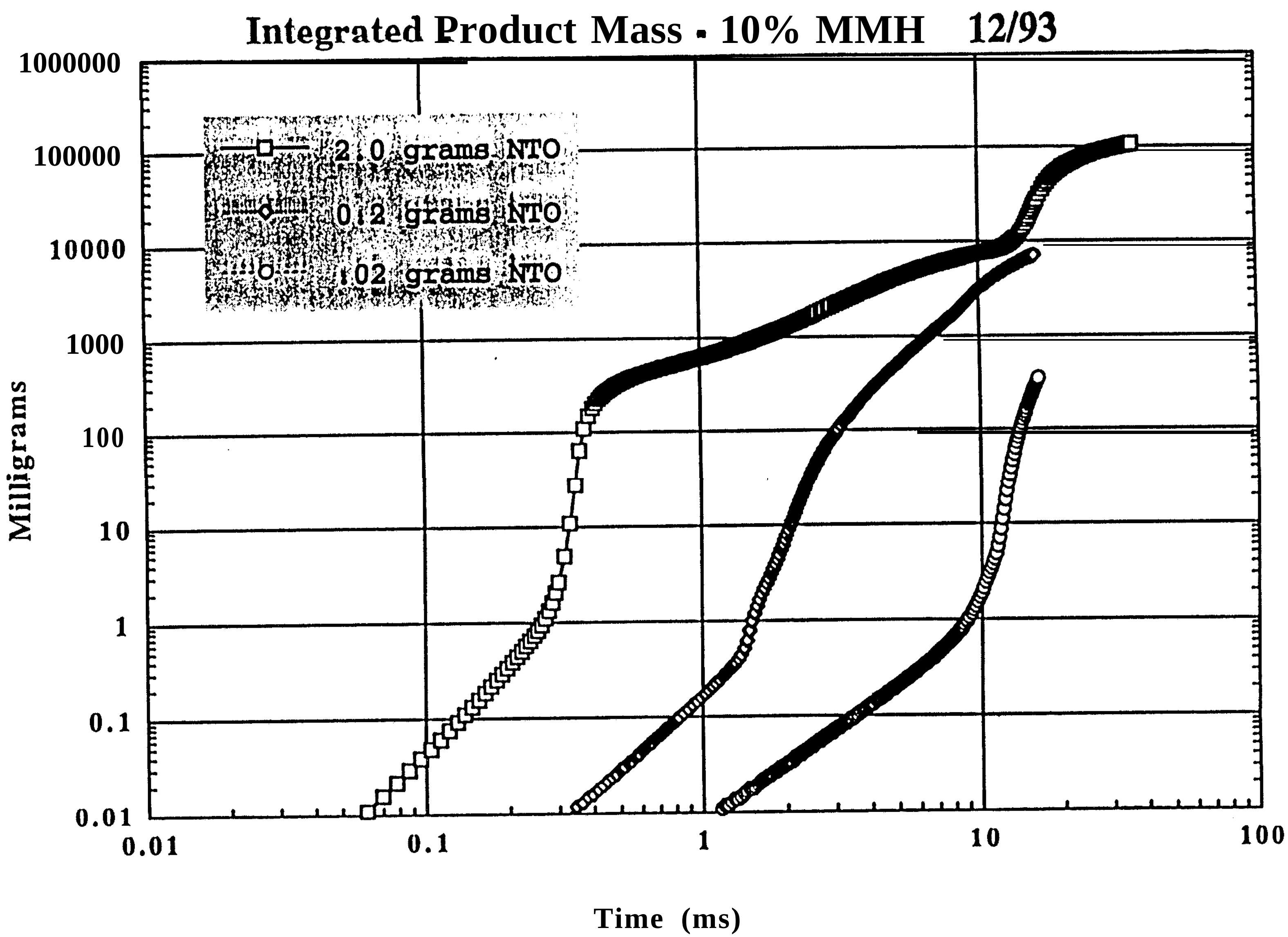


Figure F6-23, Predicted Integrated Product Mass, NTO Mixing in Tube 10% Filled with MMH

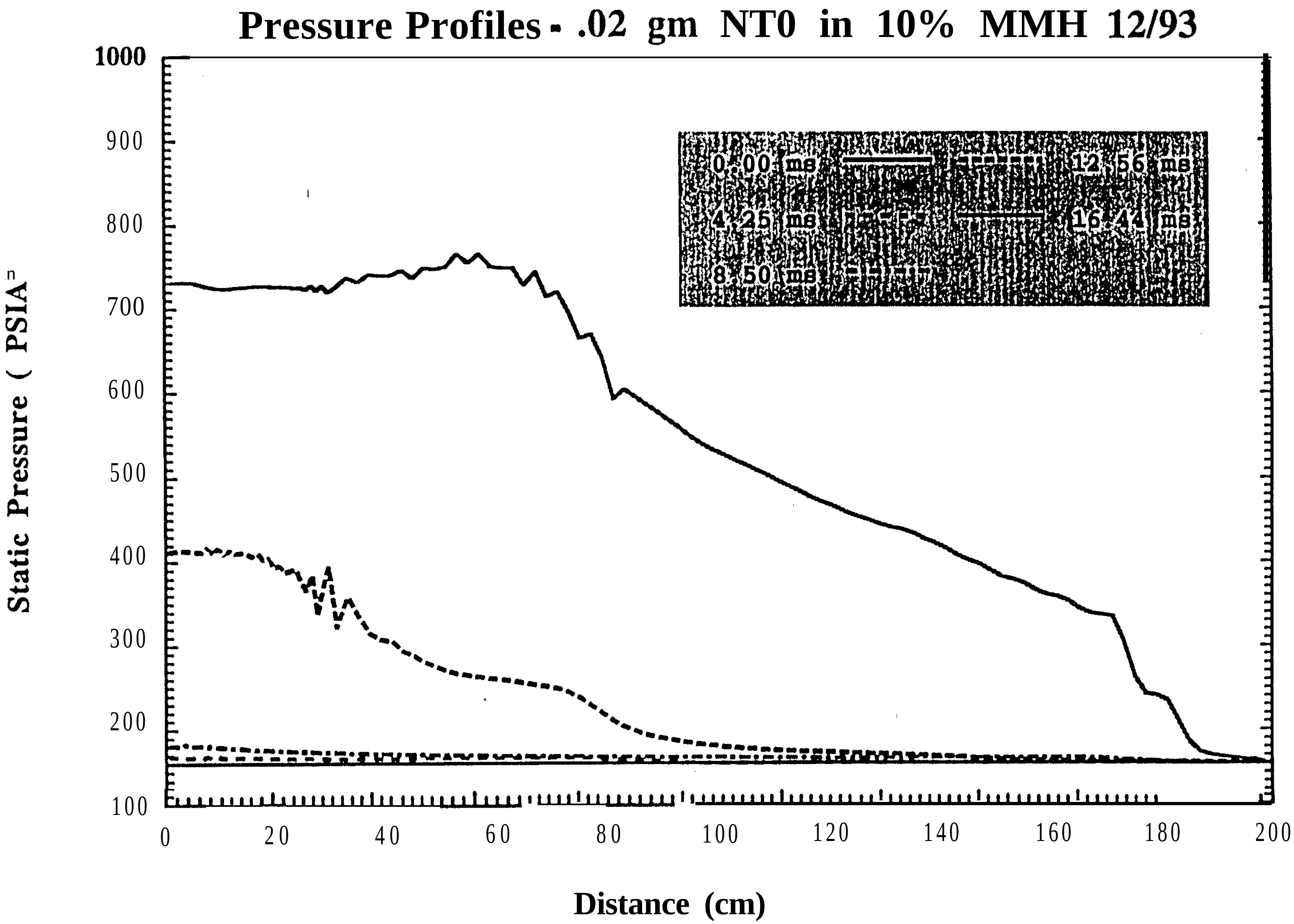


Figure F6-24. Predicted Pressure Profiles: 0.02 Grams of NTO Mixing in Tube 10% FIlled with MMH

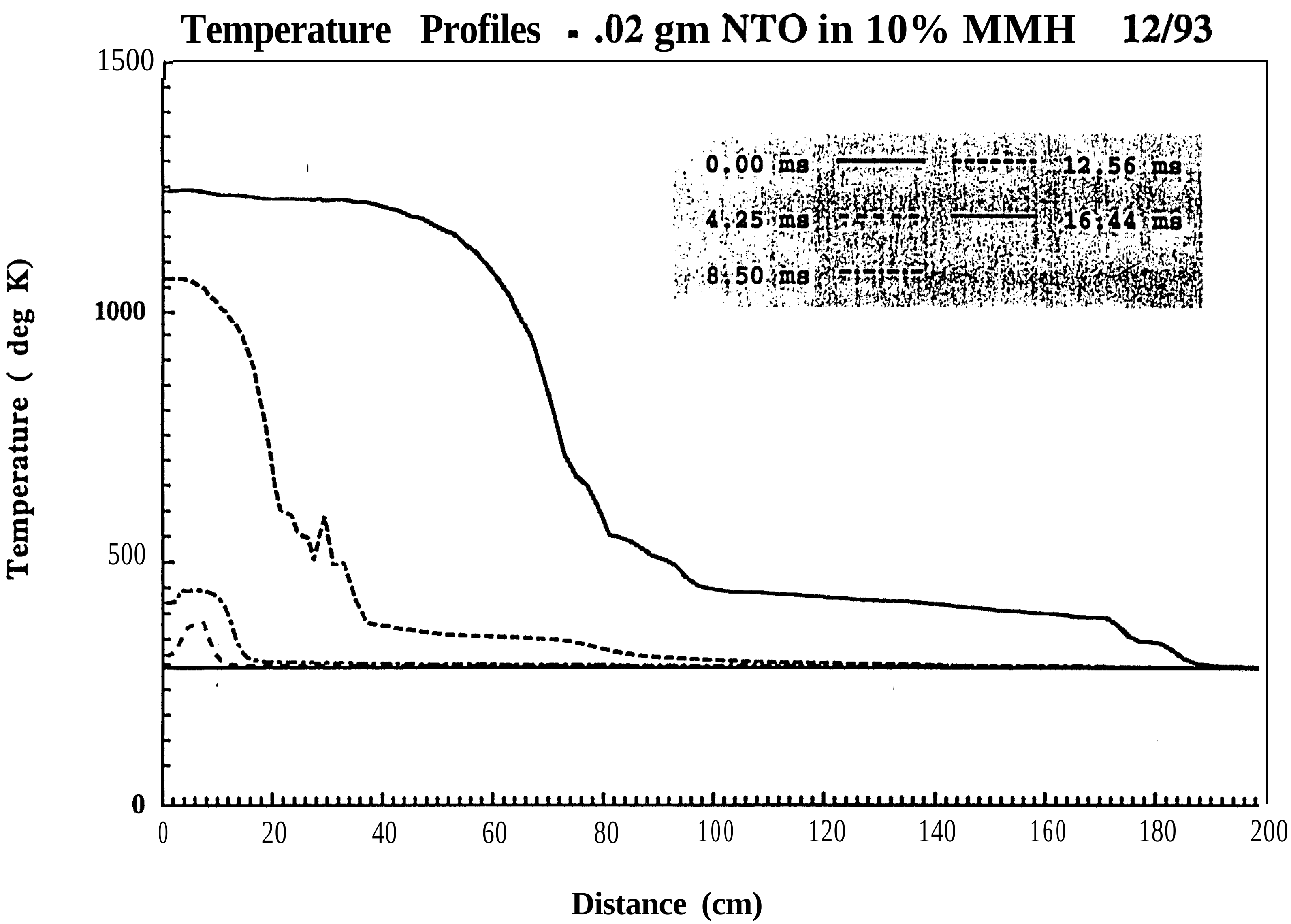


Figure **F6-25.** Predicted Temperature Profiles: 0.02 Grams **of NTO** Mixing in Tube 10% **Filled** with MMH

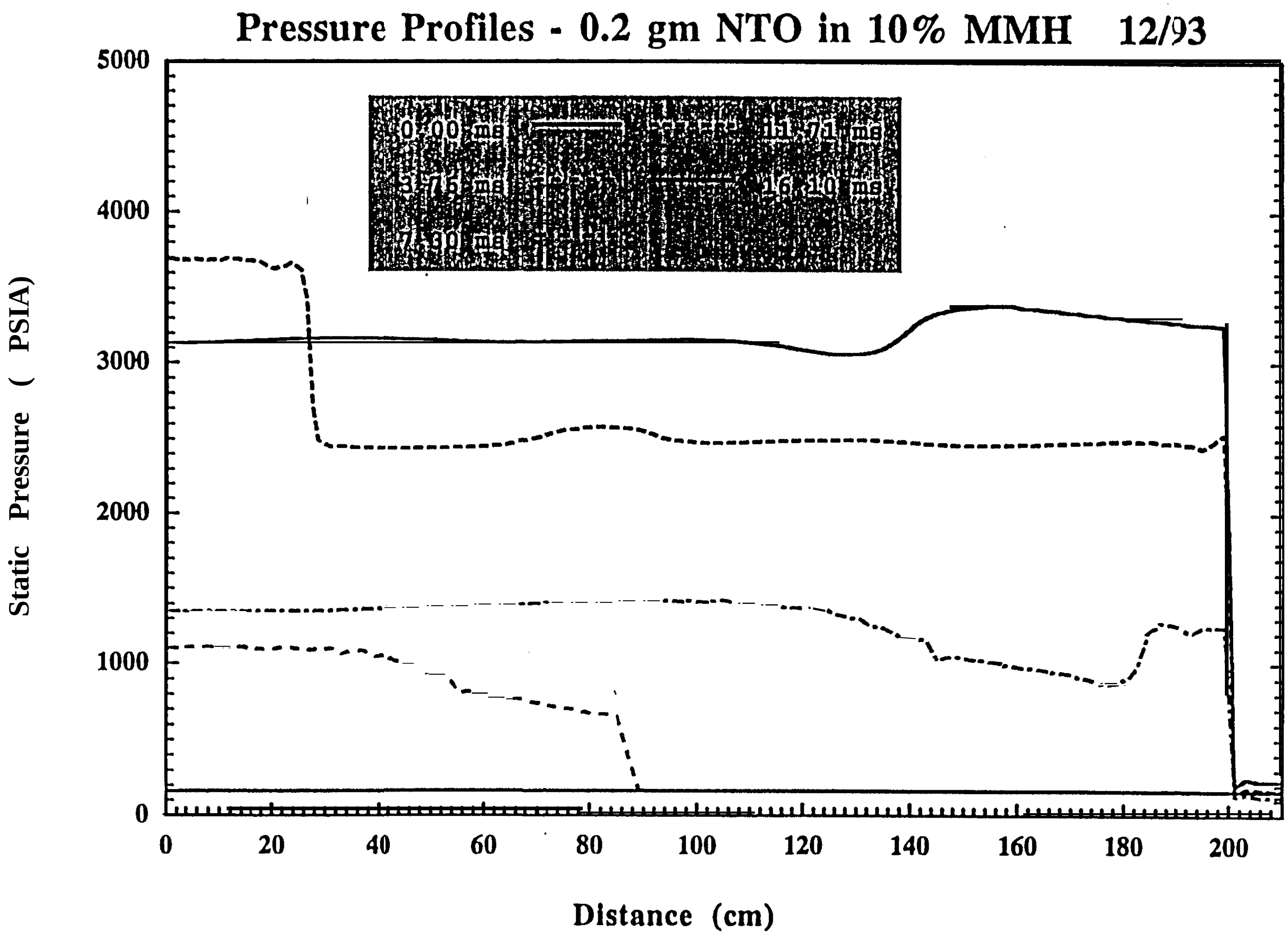


Figure F6-26. Predicted Pressure Profiles: 0.02 Grams of NTO Mixing in Tube 10% Filled with MMH

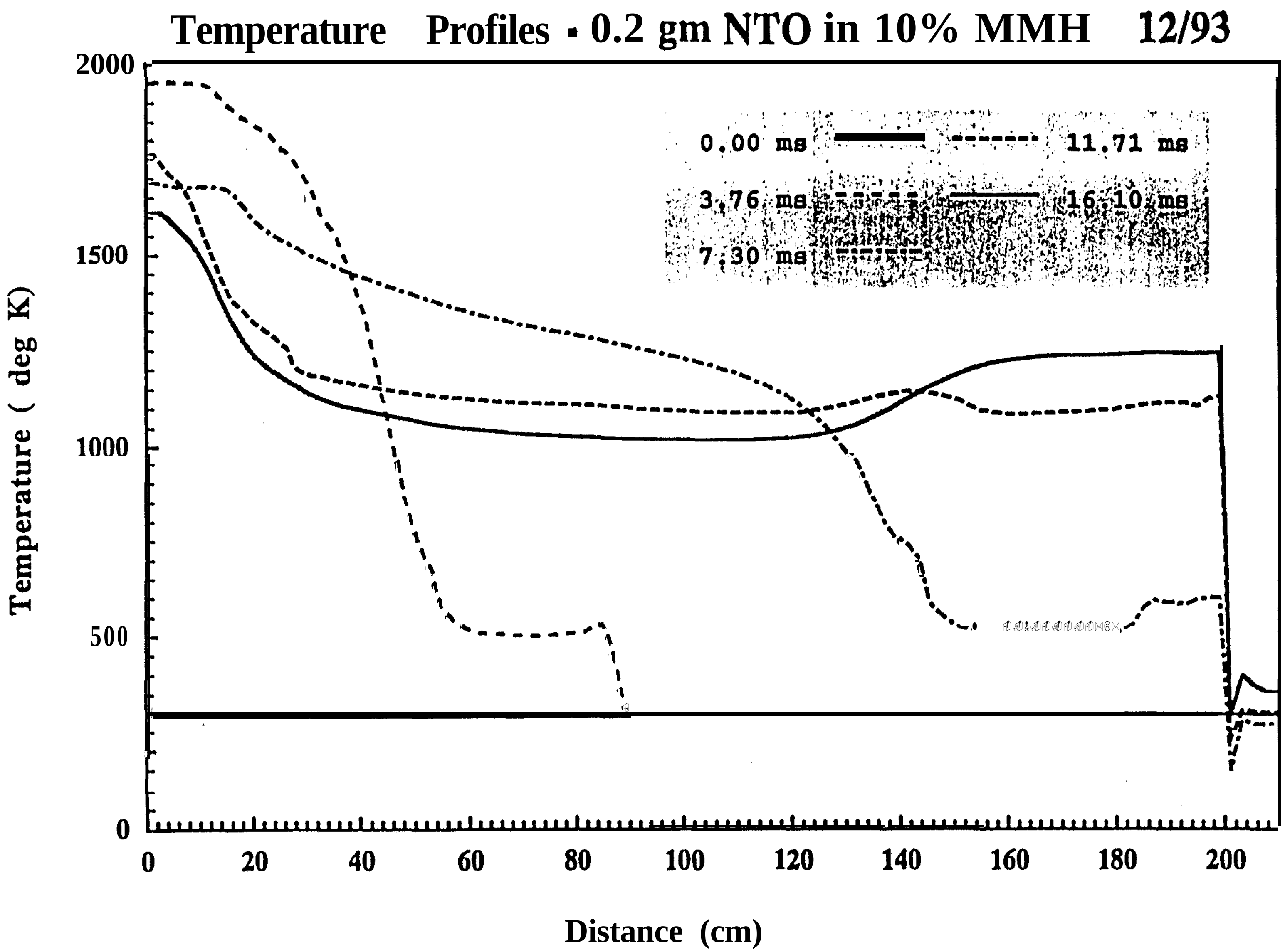


Figure F6-27: Predicted Temperture Profiles: 0.02 Grams of NTO Mixing in Tube 10% Filled with MMH

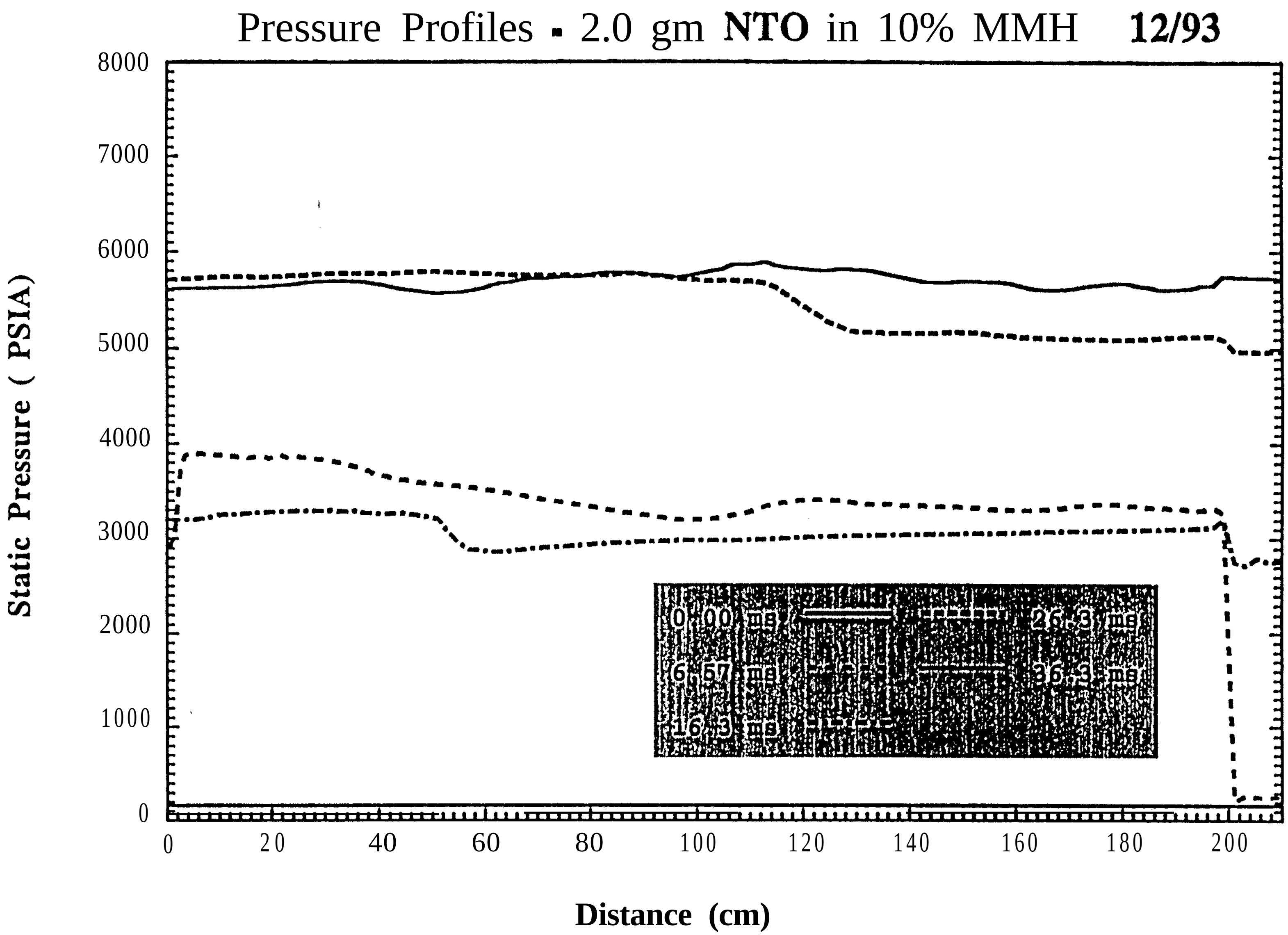


Figure **F6-28:** Predicted Pressure Profiles: 2.0 Grams of **NTO** Mixing in Tube
10% Filled with MMH

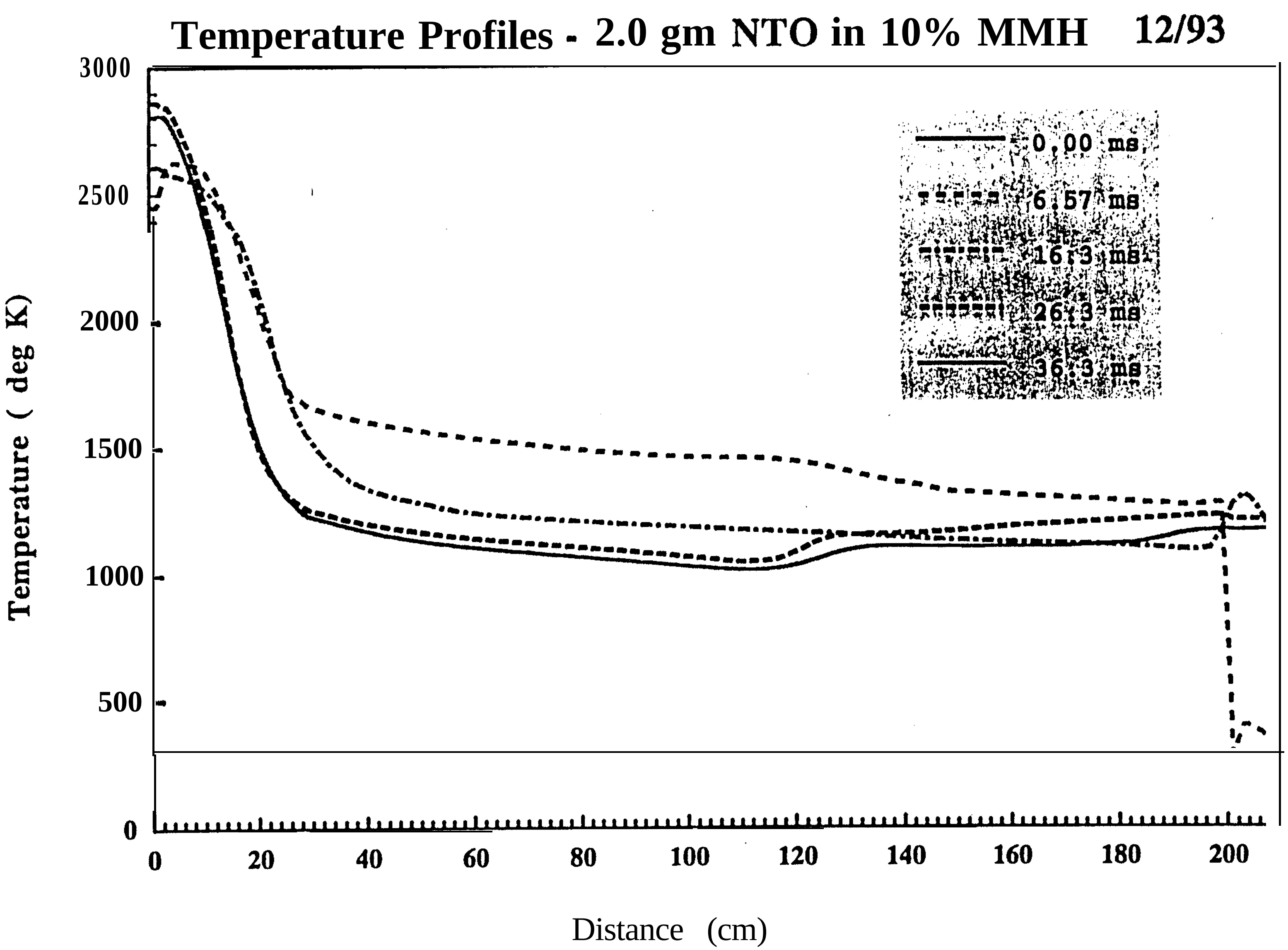


Figure F6-29: Predicted Temperature Profiles: 2.0 Grams of NTO Mixing in Tube 10% Filled with MMH

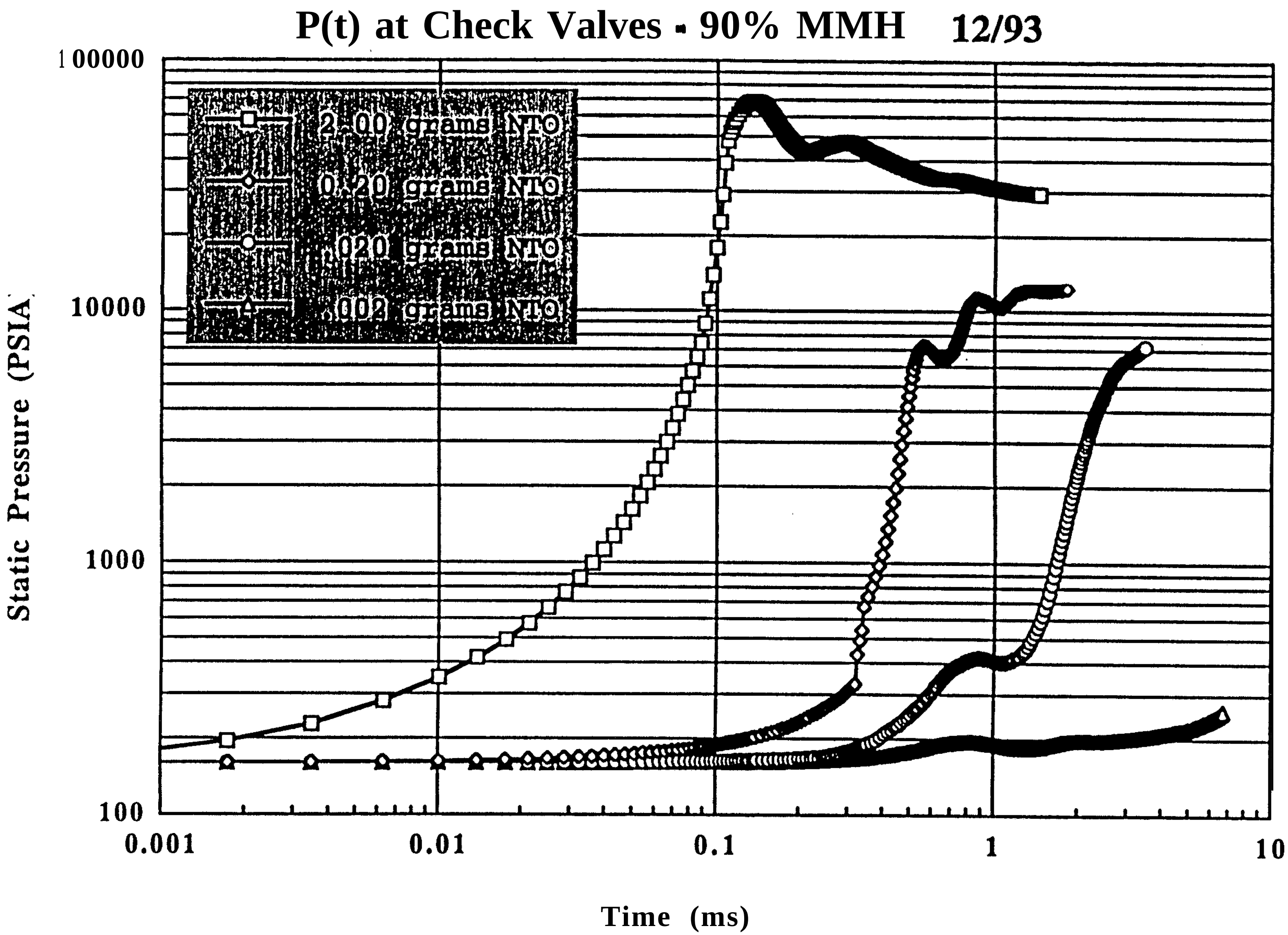


Figure F6-30: Predicted Pressure vs Time at Check Valves: NTO Mixing in Tube 90% Filled with MMH

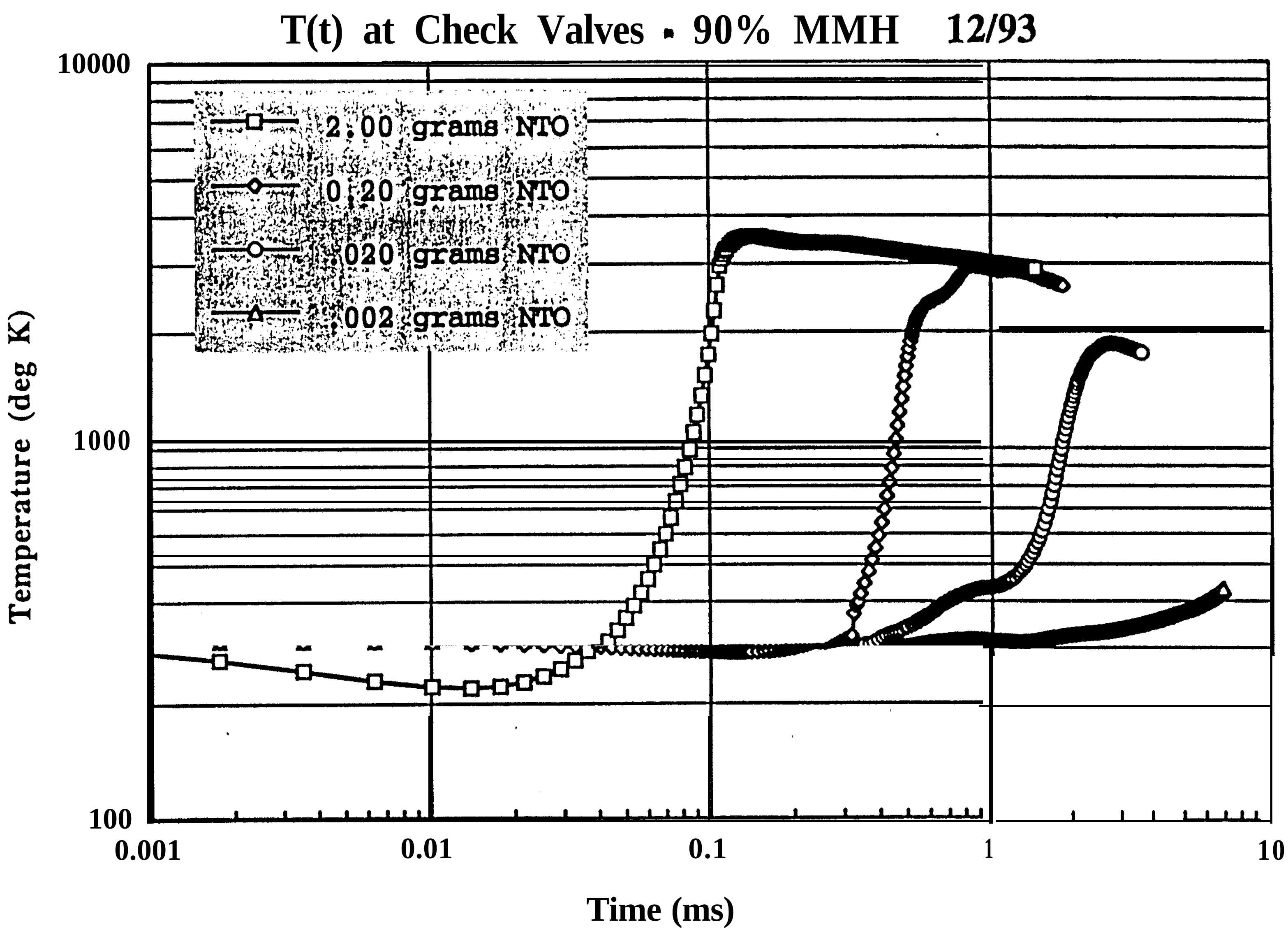


Figure F6-31: Predicted Temperature vs Time at Check Valves: NTO Mixing in Tube 90% Filled with MMH

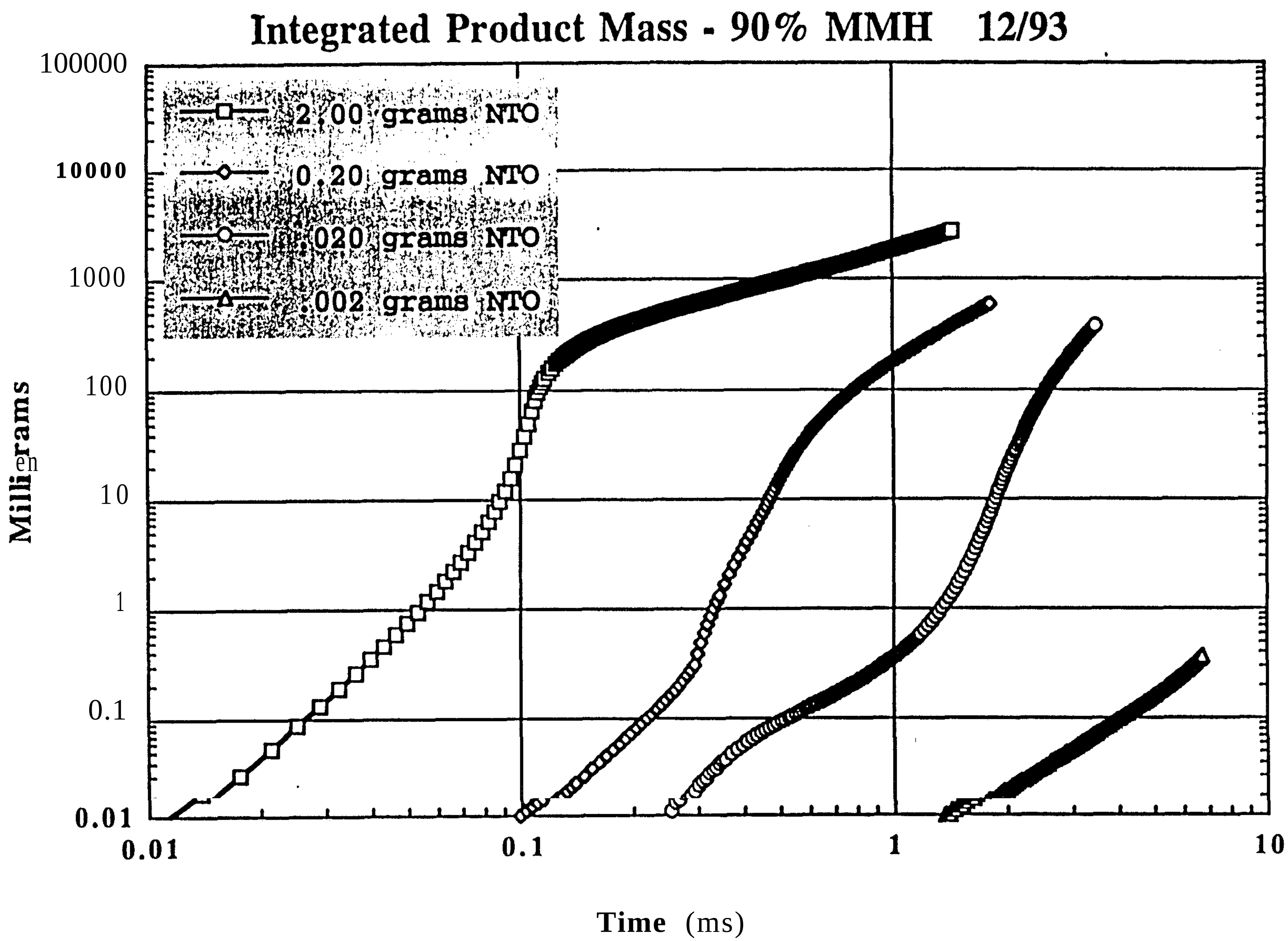


Figure **F6-32:** Predicted Integrated Product Mass, **NTO** Mixing **in** Tube 90% Filled with MMH

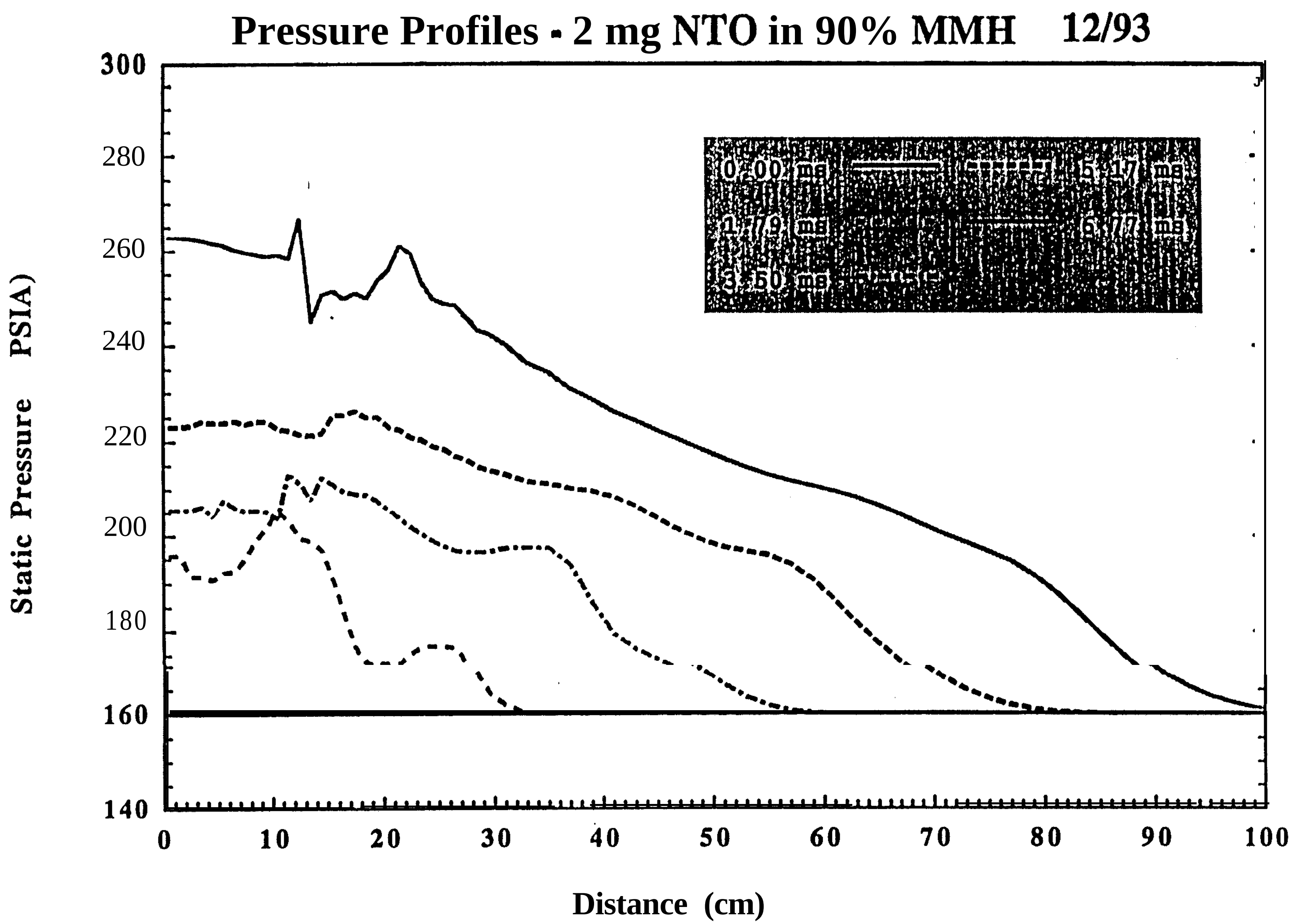


Figure **F6-33:** Predicted Pressure Profiles: 2 Milligrams **of NTO** Mixing in Tube 90% Filled with MMH

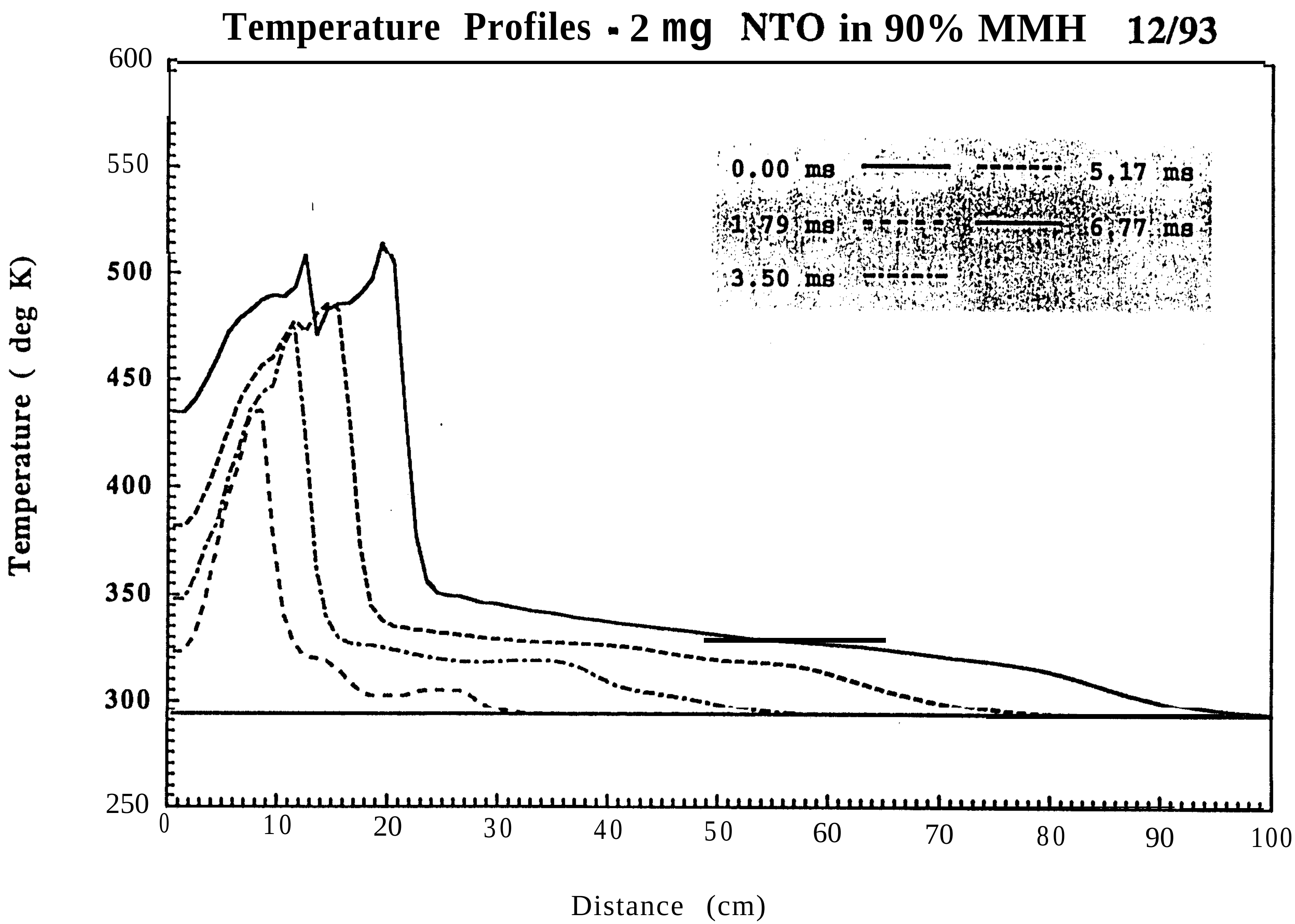


Figure **F6-34:** Predicted Temperature Profiles: 2 Milligrams of **NTO Mixing in** Tube 90% Filled with MMH

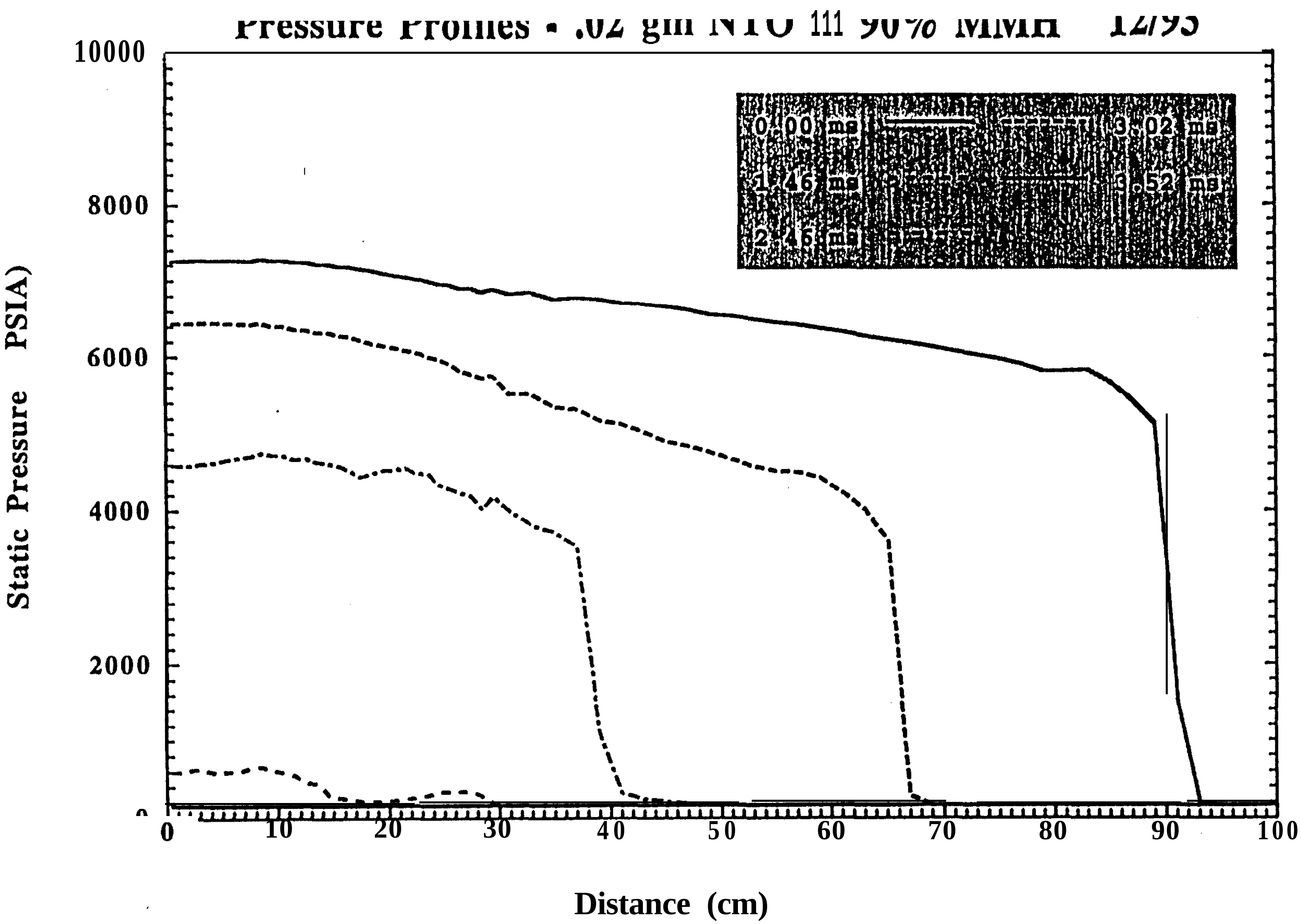


Figure F6-35: Predicted Pressure Profiles: 0.02 Grams of NTO Mixing in Tube 90% Filled with MMH

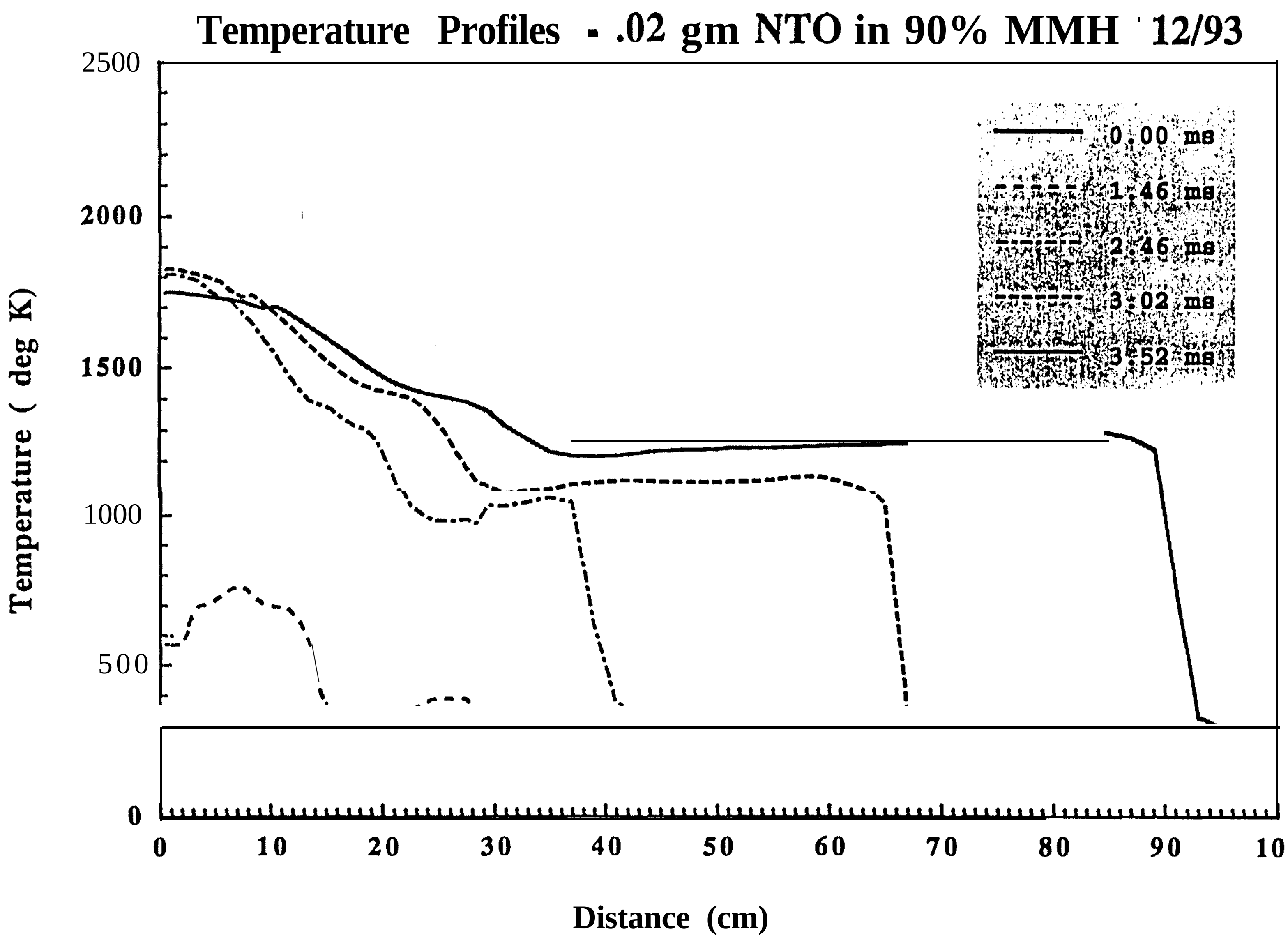


Figure **F6-36: Predicted** Temperature Profiles: 0.02 Grams of **NTO** Mixing in Tube 90% Filled with MMH

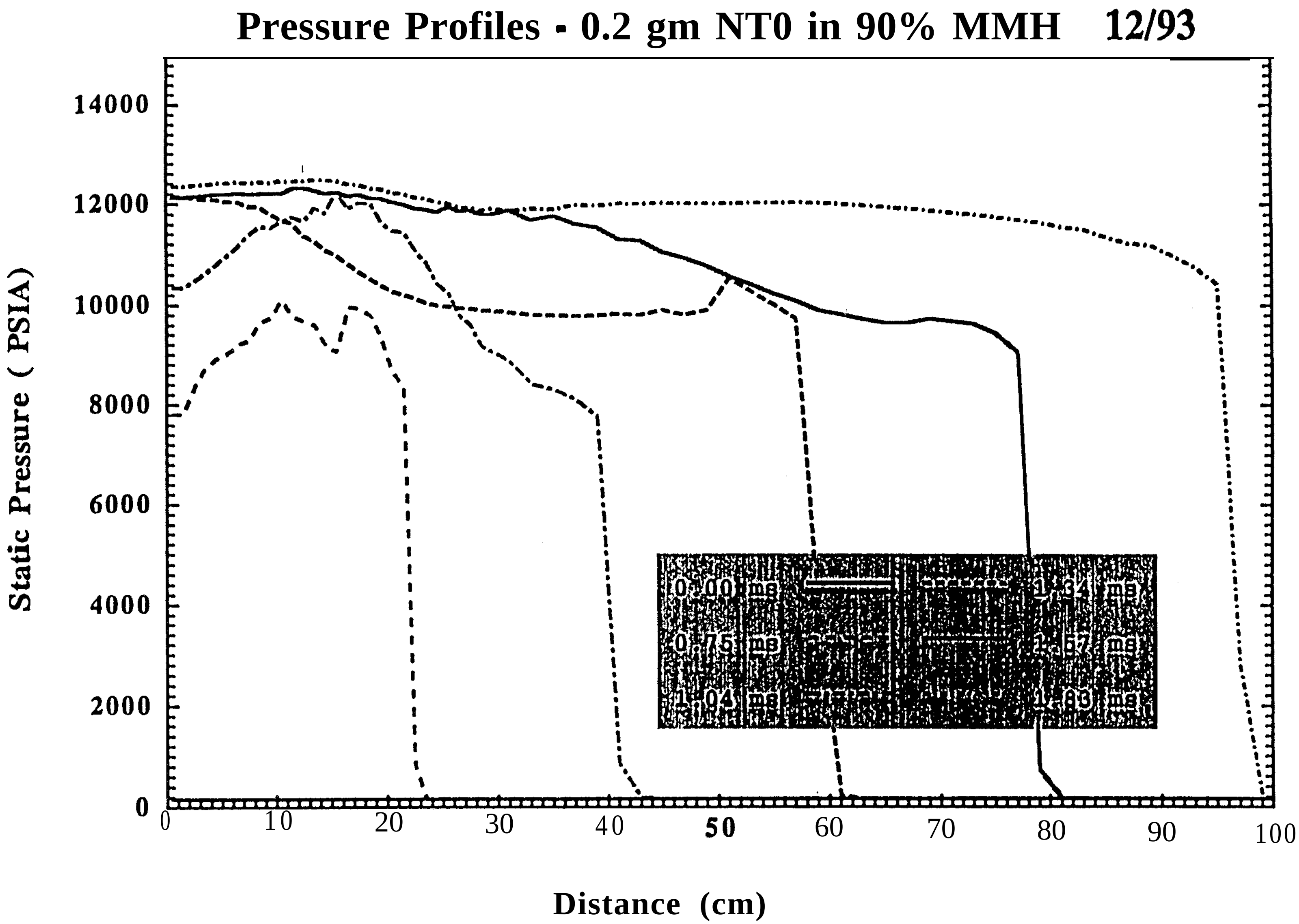


Figure **F6-37:** Predicted Pressure Profiles: 0.02 Grams of **NTO** Mixing in Tube 90% Filled with MMH

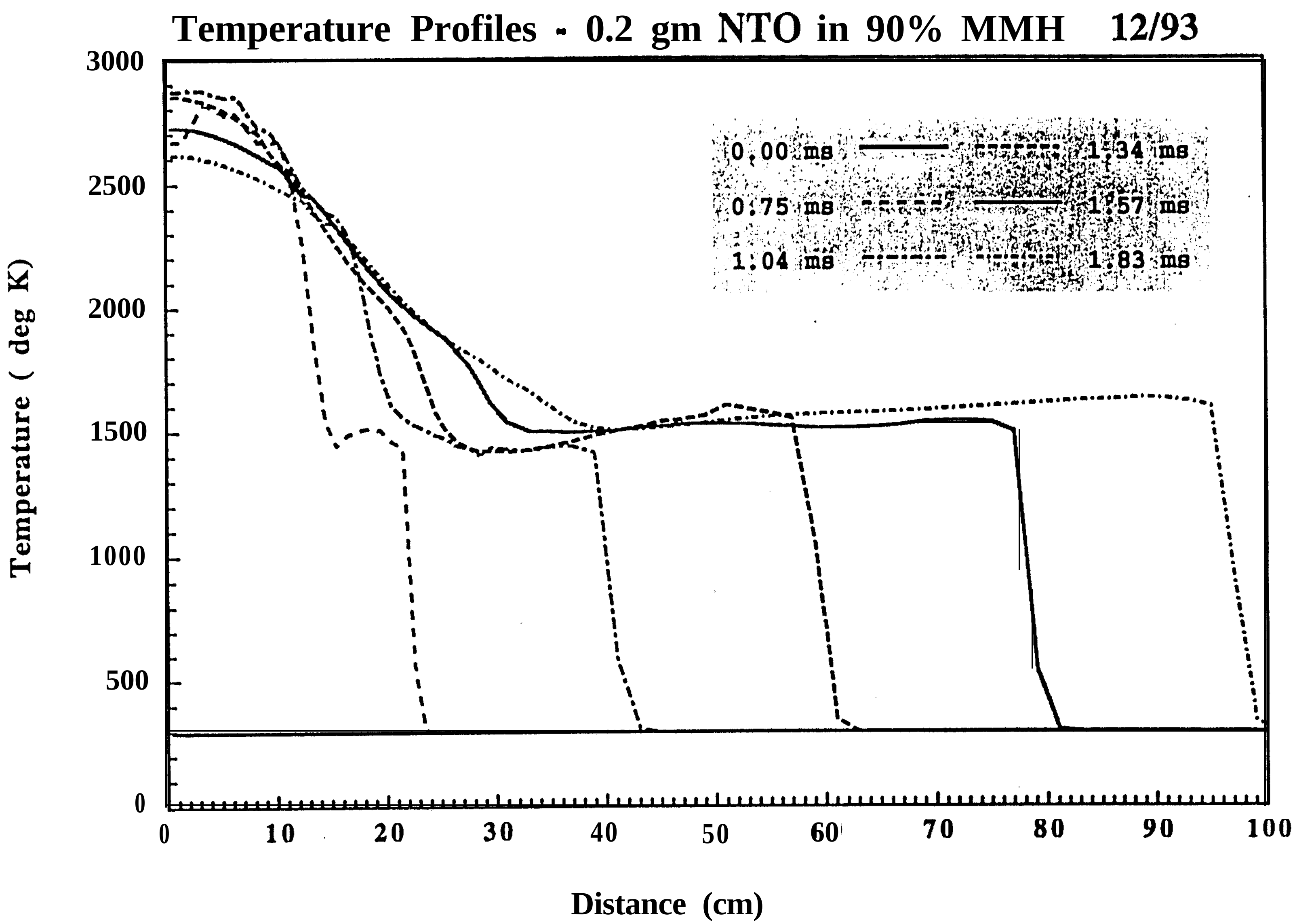


Figure F6-38: Predicted Temperature Profiles: 0.02 Grams of NTO Mixing in Tube 90% Filled with MMH

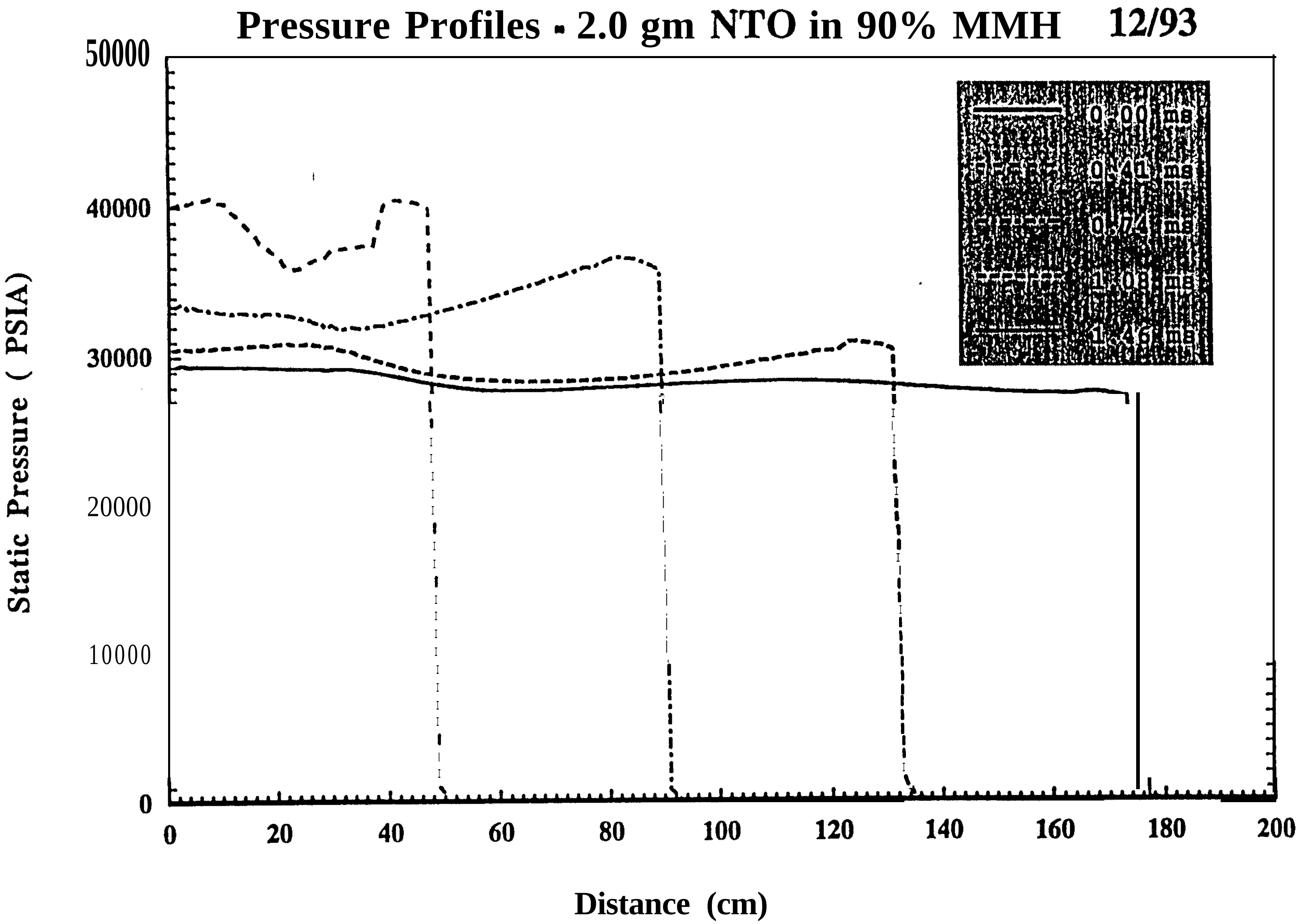


Figure **F6-39:** Predicted Pressure **Profiles:** 2 **.0** Grams of **NTO** Mixing in Tube 90% Filled with MMH

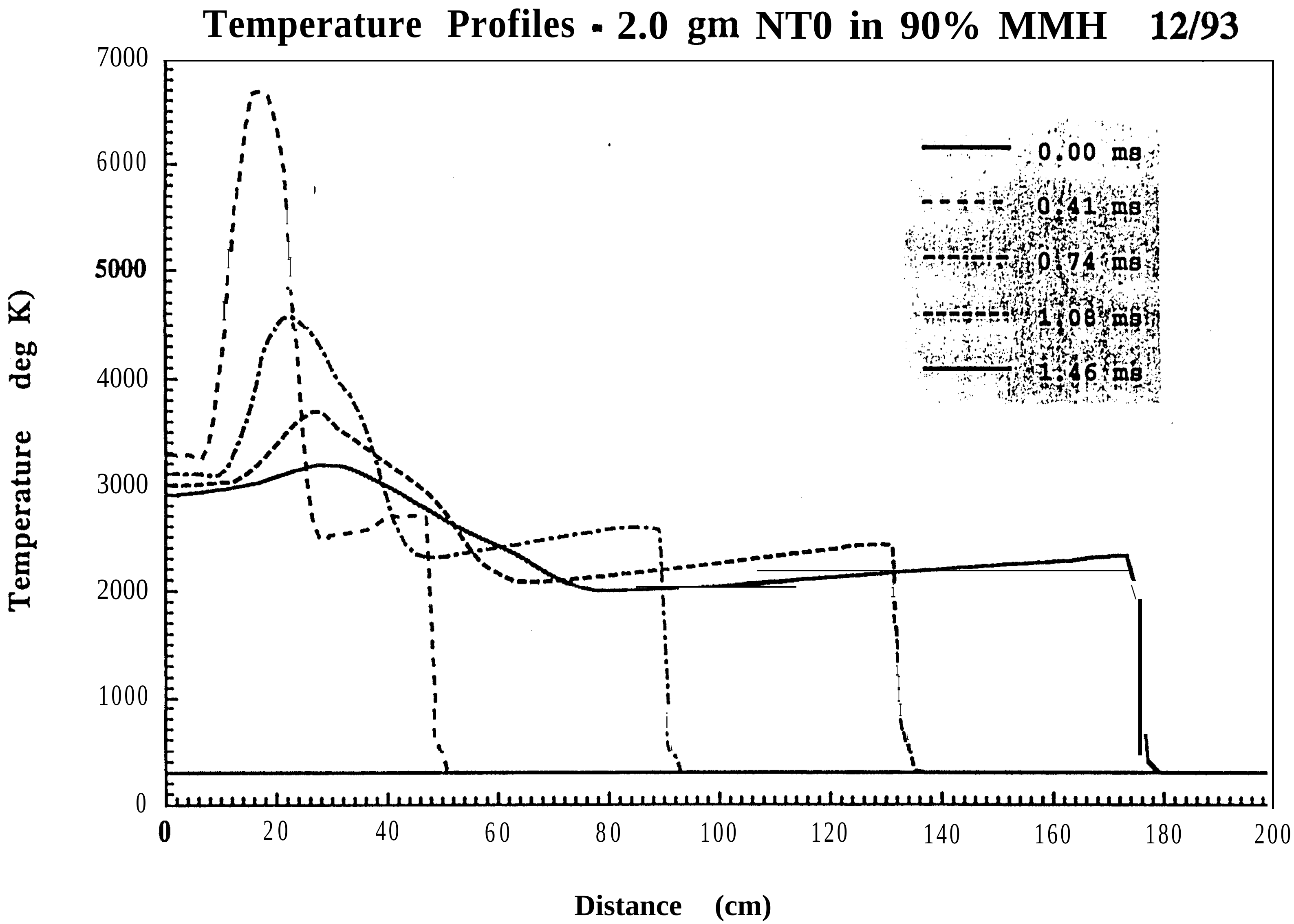


Figure F6-40: Predicted Temperature Profiles: 2 .0 Grams of NTO Mixing in Tube 90% Filled with MMH

In the cases simulated (10 percent MMH and 90 percent MMH), the tubing would have reached a temperature well above 500°C by ten milliseconds, by which time at least a few tenths of grams of NTO would have burned (smaller amounts of NTO do not release enough energy to heat the pipe unless the MMH decomposes). Hence, the tubing would have lost its strength within that time (see Figures F6-19 and F6-20). The pressures on the walls of the tubing would far exceed the yield strength. The question then becomes: will the transient pressure last long enough to disrupt the tube?

When the pressure far exceeds the yield pressure, one can estimate the acceleration of the tubing by ignoring the tensile strength and treating the tubing as a fluid shell. Furthermore, one could expect that the tubing will rupture if the tubing shell is accelerated, say, ten times its thickness. Under these assumptions, the acceleration of the tubing can be estimated in a planar approximation. If r is the radial location of the shell, P is the pressure in the tube, M is the mass per square centimeter of the tube and t is time, then:

$$\frac{d^2r}{dt^2} \cong \frac{P}{M}$$

or

$$r - r_0 \cong \frac{P}{2M} t^2$$

where ro is the initial location of the tubing shell. For the tubing used on Mars Observer M = .17 grams. For the purpose of calculation, let $P \sim 10^4$ psi (7×10^8 dynes/cm^2). The tubing has a shell thickness of .015 inches (.0381 cm). Thus the time tl required to displace the tubing shell by one shell thickness is:

$$t_1 \cong 4 \times 10^{-6} \text{ seconds}$$

Hence at 10,000 psi, the tubing shell will be displaced by ten times its own thickness in about 10 microseconds (or one hundred times its own thickness in 40 microseconds). These times are much less than the duration of the pressure and thermal pulses. In all likelihood, the tubing would rupture before it could cool down and regain its strength. Indeed, if the tubing yield strength were low enough (i.e., temperature high enough), the accelerated tubing wall would be Rayleigh-Taylor unstable. The growth time for a mode whose wavelengths equals the shell thickness would be a few microseconds. For the cases calculated, the tube would be expected to rupture within a few tens of microseconds of the time the gas pressure exceeded the yield pressure/temperature requirements discussed above.

The idealized calculations presented above are indicative of the problems that could be encountered if NTO were rapidly injected into the tubing leading to the MMH tank and thoroughly mixed with the MMH. The calculations'were performed for the situation where the mixing time of NTO with MMH is short, compared with other characteristic times (e.g., reaction times, pressure-relief times, thermal-diffusion times, etc.). The energy release in this situation is governed by the other characteristic times. If, however, the mixing time is long compared with the other characteristic times, then the mixing will govern the energy release (for example if the NTO and MMH never mix, i.e., infinite mixing time, then no energy is released). The calculations presented probably represent the most stressful situation that could have developed for the amounts of NTO introduced. In this sense, the calculations should be interpreted as an upper bound The calculations demonstrate that quantities of a few tenths of a gram or greater of NTO are required if direct burning of NTO with MMH is to be considered a threat to spacecraft health. For quantities of NTO much less than a few tenths of a gram, selfdecomposition of MMH would be required in order to create a threat to the spacecraft. The AFPL mixing test provided no evidence that self-decomposition of MMH occurred=

If the tubing ruptured or melted, then the helium pressure tank would vent through the ruptured tubing, spinning the spacecraft up. As a worst case, the Board considered a rupture downstream of the MMH pyro valves. This would result in a complete severance of the pressurization line, creating a clean, 3/8-inch diameter orifice with unidirectional gas expulsion, A failure of this type would be far more disruptive to the initial spacecraft dynamics than the development of fissures or cracks spewing gas in multiple directions.

The helium pressure regulator operates with the characteristics shown in Figure F6-41, assuming a 5-psia outlet pressure and a 70°F operating temperature. The average inlet pressure to the regulator is 2000 psia, and the average flow rate from Figure F6-41 is approximately 0.0125 lbs/sec through the 3/8-inch diameter line.

The time to expel the entire 10.7 Ibs of GHe is then:

$$At = \frac{10.7 \text{ Ibs}}{0.0125 \text{ lbs/sec}} = 856 \text{ seconds}$$

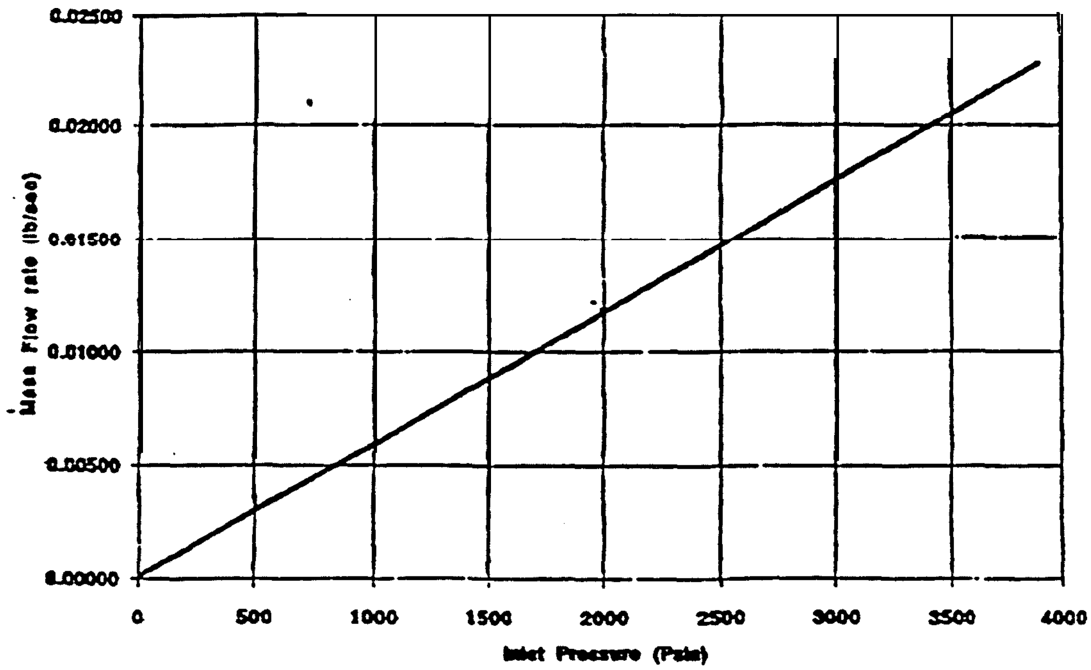


Figure F6-41. Flow Rate of Mars Observer Pressure Regulator at 70°F and 5 psia Outlet Pressure

The average effective thrust through the 3/8-inch diameter orifice is:

$$\bar{F} = pA = 5 \text{ psia} \times \frac{\pi\,(0.375)^2}{4} = .552 \text{ lbs}$$

The total delivered impulse in depleting all of the GHe is:

$$\bar{F}\Delta t = 0.552 \text{ Ibs} \times 856 \text{ sec} = 472 \text{ lb-see}$$

The effective Isp of the GHe is:

$$Isp = \frac{472\ lb_f\text{-sec}}{10.7\ lb_m\ GHe} = 44.1\ sec$$

The change in spacecraft spin rate may be estimated from:

$$T = I\alpha = I\frac{dw}{dt}$$

$$Fl\,dt = I\,dw$$

$$\Delta w = \frac{Fl\ \Delta t}{I}$$

The spacecraft mass properties at the time of the pressurization event are specified in Table 6-2

Assuming rupture at a 4-ft moment arm with respect to either the X or Z axes, the maximum spin rates achievable can be calculated as follows:

Spin About **Minimum** Moment of Inertia Axis:

$$\Delta w = \frac{472 \times 4}{1592.3} \times 1.3557\% = 1.6076\ \text{rad/sec} = 15.3\ \text{rpm}\ (92^\circ/\text{sec})$$

Spin About Maximum Moment of Inertia Axis:

$$\Delta w = \frac{472 \times 4}{2952.5} \times 1.3557 = 0.8669\ \text{rad/sec} = 8.3\ \text{rpm}\ (50^\circ/\text{sec})$$

TABLE F6-2 • MARS OBSERVER MASS PROPERTIES ($kg \cdot m^2$); CRUISE PHASE AT PV7 FIRING.

Ixx = 2259.6	**Ixy=** -198.7
Iyy = 2952.5	Ixz = -233.7
Izz = 1592.3	Iyz = -120.3

The above computed spin rates represent the maximum spin range achievable, given a unidirectional gas-expulsion stream with no obstructions in its path. This situation is unlikely to occur in practice, as the pressurization lines are well-covered with thermal insulation blankets. The initial rupture could have blown off the insulation blankets in the vicinity of the leak, or left the blanket partially attached, splaying the exhaust plume. Exhaust gases that hit the insulation blanket, the upper bulkhead of the spacecraft, or any other obstacles or appendages in their path would exert forces tending to cancel the spin that was induced from the initial thrust at the break. Since the gas is not likely to be released in a directed beam, but in a widening plume with a high likelihood of hitting obstacles in its path, it is reasonable to conclude that only a fraction of the energy stored in the GHe would be converted to spacecraft angular momentum.

The problem is extremely complicated to analyze and is highly dependent upon many unknown factors, including the size and form of the rupture, the location of the rupture, the direction of the gas jets, and the geometry of obstacles in the line of the plume. If only one-third to one-half of the total energy were converted to angular momentum (a somewhat more reasonable assumption given the arguments above), then the predicted range of the resultant spin rate would become:

Spin About Minimum Moment of Inertia Axis:

$$30°/\text{sec} < \text{Aw} < 46°/\text{sec}$$

Spin About Maximum Moment of Inertia Axis:

$$17°/\text{sec} < \text{Aw} < 25°/\text{sec}$$

The question that must now be asked is: what is the maximum spacecraft body rate at which the spacecraft is controllable? The answer to this question depends on the total net disturbance torque on the spacecraft, and the absolute angular rate of the vehicle. The net torque levels affect the transient spacecraft attitude, while the magnitude of the high body rates would influence the longer-term behavior.

If the disturbances on the spacecraft were significantly less than the maximum RWA torque capability of 0.14 N-m, and if the disturbance torque were applied slowly, the spacecraft would be able to control the attitude very well in the short term (minutes). However, as noted in Chapter F3 and Appendix L, the RWA was not reactivated until 234:01:00:17 UTC, ten minutes after firing pyro valve PV5. High-frequency disturbances may be beyond the control bandwidth (-0.015 Hz) of the system, even if the torque and momentum were well within the reaction wheel capability. In the scenario described above, the maximum torque available is about 3 N-m The reaction wheels, even if activated, could not control this amount of torque.

If the body rates exceed 7.4°/sec, the digital electronics in the IMU would become saturated, causing inertial reference to be lost. Since the momentum would also be high, emergency unloading would be triggered. As long as the body rates are below 9°/sec, the polarities of the rates are still available from the IMU. The monopropellant thrusters would ordinarily fire and reduce the total system momentum to less than 0.5°N-m-sec per axis. However, the attitude control system, including the monopropellant thrusters, was disabled daring the Pressurization Sequence. In addition, the thrusters normally require a 15-minute warm-up period before they can be fired. As a result, in the scenario postulated here, the spacecraft would have spun up to its maximum rotation rate (i.e., all of the GHe and MMH would have escaped) before the thrusters could be activated to unload the momentum.

A body rate in excess of 9°/sec would saturate both the digital and analog electronics of the IMU. The momentum unloading logic does not take action if more than one axis is in saturation. As the nutation was damped out by energy dissipation, the spacecraft would tend to spin about the axis of maximum inertia (15.35' from the Y-axis), bringing all but one axis out of saturation in most scenarios. With only one axis in saturation, unloading in the X and Z axes would commence. As the spacecraft was precessed back to Earth-pointing, more of the momentum would appear in the X and Y axes and would be desaturated with the thrusters. When the Y axis spin was reduced below 7.4°/sec, that axis would also be desaturated, since the polarity would be available.

Higher body rates in excess of about 36°/sec would cause all axes to remain in saturation due to the coupling from the maximum principal axis into the X and Z body axes. In this case, desaturation would not occur and the spacecraft would continue in an unrecoverable tumble. Such rates are clearly achievable in the scenario under consideration here.

When two gyros give saturated readings, the spacecraft responds by entering Contingency Mode. If Contingency Mode were entered within about four minutes of firing pyro valve PV5, then the expected switch-over from the HGA to the LGA would not occur. As discussed under AACS Failure #5 in Chapter F3, VTL simulations of a saturated IMU showed that the entry into Contingency Mode would prevent the execution of the script to turn on the RPA beam. This condition not only does not allow the HGA to be turned back on as planned when coming out of the blackout period, it also does not permit switch-over to the LGA upon entering Contingency Mode. As long as two axes remained saturated, the spacecraft would not transmit on either the high-gain or the low-gain antenna. Under these conditions, Mars Observer would also be spinning so rapidly as to prevent ground commands from being loaded into the spacecraft. Further, in all likelihood, the solar panels would no longer be receiving enough sunlight to recharge the batteries on the spacecraft. The net result of these events would be that the spacecraft would be rendered useless, and would likely never communicate its fate.

There is enough energy available in the GHe tank to force the spacecraft into Contingency Mode in less than four minutes after firing pyro valve PV5. However, if Contingency Mode were entered later than four minutes after firing pyro valve PV5 (by which time the downlink should have reappeared on the HGA), then the switchover from the HGA to the LGA would be executed. Since Contingency Mode would have been triggered within minutes of the HGA beiig reactivated, and since the spacecraft would be spinning about an unknown axis, it is unlikely that the short-duration HGA downlink would have been detected.

Switchover to the LGA would greatly broaden the antenna-boresight-to-Earth angles in which the downlink would be detected (see Figure D-3). However, while the LGA beam pattern is large, the link margin is small. If the LGA had been activated by entry into Contingency Mode, then the downlink should have been on for a minimum of several hours, until low battery charge state forced the RPA beam to be turned off. Whether the LGA signal would have irradiated the Earth long enough for the DSN to have detected it would depend on the spinning geometry of the spacecraft. There clearly are geometries (Earth-pointing) in which the transmission from the LGA should have been detected, and other geometries (LGA pointing away from Earth) in which it would never be detected. The more likely situation would be one in which the LGA beam pattern swept through these extremes. If this were to be the case, detection would depend upon how long it took for the LGA beam pattern to sweep by the DSN receivers. As discussed in Sections a.(8) and a.(9) of Chapter F5, it is unlikely that the LGA downlink from a rapidly rotating spacecraft would be detected by the DSN prior to the discharge of the batteries. Rough estimates considering the LGA beam pattern and the range of spin rates suggest a DSN detection probability of less than 25%.

In some sense, the above discussion about spin rates and downlink characteristics may be academic. Had a rupture occurred in the line between check valve CV2 and the MMH tank, then the contents of the MMH tank would have sprayed out through the rupture along with the GHe. Any MMH coming into contact with electrical wiring would probably damage the insulation, causing a short circuit and a loss of downlink.

It was also noted in some of the simulations that MMH had begun to decompose energetically due to the high temperature. If this decomposition were able to propagate to the MMH tank and into its contents, then the spacecraft might literally blow up. The simulations as performed are not able to handle the propagation of an MMH decomposition wave into the MMH tank, and the AFPL interaction tests showed no selfdecomposition of MMH; therefore, no conclusions can be reached in this regard

After reviewing the results of the tests and simulations discussed above, it was the Board's opinion that the unintended mixing of NTO oxidizer and MMH fuel, enabled by migration of NTO through the check valves into the pressurization plumbing during the 1 l-month cruise phase, is the most probable cause of the mission failure of the Mars Observer spacecraft.

f. PROPULSION SYSTEM TESTS AND ANALYSES

Several tests were conducted in support of the Propulsion System Technical Team. They are described in the Section e. above.

(1) Check Valve Leak Tests

A series of tests of check valves identical to those used aboard Mars Observer were conducted by JPL. Test results are presented in Figure D-6. A more complete report is pending and will be included in Appendix Q if received before distribution of this report as JPL memo IMO-353MO-93-024, "Mars Observer Check Valve Test Report." In addition, JPL will perform a propellant migration analysis, based upon propellant leakage/&fusion data determined by the check valve leak tests to determine the quantity of NTO that could be present in the pressurization manifold The analysis will be presented in JPL memo IMO-353MO-93-025.

(2) Propellant Interaction Tests

As of the time of writing, 12 tests had been conducted by AFPL. The results are discussed in the section of this report describing Propulsion System Failure #12, above.

Liquid Bullet" Test

A test was performed at JPL to determine if a slug of 1 or 2 grams of NTO could be accelerated by the He pressurization gas in-rush, and achieve enough kinetic energy to rupture a pressurization line. The test failed to produce any damage to the test tube. Test results will be released in JPL IOM353MO-93-026 "MARS OBSERVER "LIQUID BULLET" TEST RESULTS,"

(4) Pvro Valve Body Thread Erosion Inspections

The Mars Observer lot acceptance test valves, system test valves, and flight-backup valves (after firing) have/will receive X-ray, chemical and physical inspections at JPL and MMAC Reports of these inspections will be included in Appendix Q if received before publication.

In addition, JPL will perform a thread-strength margin analysis based upon pyro-valve inspection data. The analysis will be presented in JPL-IOM-353MO-93-030 "Investigation of Pyro Valve Initiator Thread Margin: Post Test Analysis."

(5) Pyro Shock Tests

A Mars Observer structural and pressurization system mock up has been built by MMAC to determine the shock levels that result from pyro valve firing. Five firings were expected planned Additional data will be collected to investigate ground currents, regulator response and pyro valve body degradation. These tests were authorized by the Board and the expenditure of each Mars Observer backup pyro valve was individually approved. Test results will be released in a JPL report.

(6) NTO/MMH Reaction Simulations

NRL prepared the simulations and calculations included under Propulsion System Failure #12 in Section e. above.

(7) Motion Produced bv a Ruptured Pressurization Line

NRL produced the calculations of spacecraft spin-up included in Section e. above. JPL will perform analyses to predict with more precision the motion produced by a ruptured pressurization line. The analysis will be presented in memo JPL-IMO-353-A-93-351.

(8) NASA Standard Initiator Mounting Thread Stress

NRL conducted an analysis of the stress levels imposed on NSI mounting threads. It is attached in Appendix Q as: "Stress Levels in NASA Standard Initiator Mounting Threads," by Robert B. Patterson, NRL memo of 3 November 1993.

(9) Regulator Valve Temperature

NRL conducted an analysis of the likely temperature of the Mars Observer pressure regulator. It is attached in Appendix Q as: "Mars Observer: Regulator Valve Minimum Temperature," by Nelson Hyman, NRL memo 8220-374:NLH:nlh of 28 October 1993.

g. PROPULSION SYSTEM ASSESSMENT

The Propulsion System used aboard Mars Observer has several weaknesses that, even if they did not contribute to the mishap, should be corrected before an identical or similar spacecraft is f lown again on an interplanetary mission. In addition, the Propulsion System Technical Team and Investigation Board developed several recommendations to NASA regarding these weaknesses.

(I) Achieving proper isolation between fuel and oxidizer is extremely important, especially on missions where the system will not be used for an extended period of time (e.g., an 1 l-month cruise phase to Mars). NASA should establish standards (maximum amounts) for the amount of oxidizer and fuel that would be permitted to migrate through check values, and/or react in the pressurization manifold of a propulsion system

(2) NASA should validate the NTO migration test data obtained by JPL, and obtain similar data for MMH migration.

(3) NASA should alert the users of NASA Standard Initiators of the possibility of damage from ejected NSIs. In addition, NASA should identify the reasons for OEA/Pyronetics pyro valve/initiator failures in order to determine the likelihood of a similar failure of NSIs.

(4) On missions where the temperature of a component or part of a system (e.g., the pressurization manifold) is critical to its performance, temperature sensors should be provided to obtain accurate temperature data. --

(5) A comprehensive analysis of the thermal environment of the propulsion system, including its pressurization manifold, should be required for all spacecraft on interplanetary missions.

(6) Propulsion system components and plumbing should be completely tested after assembly to ensure proper operation.

(7) Fuel and oxidizer should be tested for cleanliness both before and after loading aboard the spacecraft-

(8) Spacecraft propulsion system plans and documentation should be updated to reflect the as-built, as-flown configuration.

(9) The heritage of propulsion system used aboard Earth-orbiting spacecraft does not automatically qualify it for use on an interplanetary mission.

PART G

FINDINGS AND OBSERVATIONS

PART G

FINDINGS AND OBSERVATIONS

I, PRINCIPAL FINDINGS:

a. The Board concludes that the most probable cause of the loss of downlink from the Mars Observer spacecraft was a rupture of the pressurization side of the Propulsion System The most probable cause of the rupture was unintended mixing of NTO and MMH in the titanium tubing of the pressurization side of the Propulsion System. Mixing was enabled by significant migration of NTO across the check valves during the 1 l-month cruise phase.

b. Any one of three additional failure scenarios remain as a plausible explanation for the loss of downlink from the Mars Observer spacecraft:

(1) An Electrical Power System failure resulting from a short-circuit on the regulated power bus.

(2) A regulator failure resulting in over-pressurization of the NTO and MMH tanks.

(3) A pyro valve failure resulting in an NSI being expelled, damaging some other spacecraft component;

c. A number of spacecraft design flaws and poor operating procedures were identified that should receive close attention and resolution prior to further use in the same or derivative-design spacecraft for similar mission applications:

(1) The propulsion system design does not provide appropriate isolation between fuel and oxidizer.

(2) The differences in pyro initiator characteristics between the OEA/Pyronetics used by European Space Agency and the NSI used on U.S. spacecraft must be understood and resolved.

(3) Thermal instrumentation and control are not appropriate to an interplanetary mission profile.

(4) The power bus is susceptible to a short circuit resulting from a single component or insulation failure.

(5) Critical redundancy control functions can be disabled by a single-part failure or logic upset.

(6) The RXO can lose one of its two outputs without remedy of fault protection.

(7) There is no method of determining the health (proper operation) of the backup crystal oscillator in the RXO.

(8) A top-down audit of fault-protection requirements, implementation, and validation is needed.

(9) The system is not qualified to provide telemetry during all critical events.

(10) There is no flow-down and verification of system-level shock, loads, and thermal environment to the subsystem/box level.

(11) Spacecraft attitude is allowed to drift during critical operations.

(12) The documentation does not in all cases reflect the as-built, as-flown hardware.

2. GENERAL OBSERVATIONS:

a. The Mars Observer that was built departed significantly from the guiding principals originally established for the program. The use of a firm, fixed-price contract was inappropriate to the effort as it finally evolved. The role of JPL in this program was a best cumbersome, and did not take full advantage of its unique experience and expertise in interplanetary missions.

b. The original philosophy of minor modifications to a commercial, production-line spacecraft was retained throughout the program. The result was reliance on design and component heritage and qualification which were inappropriate for the mission. Examples include the failure to fully qualify the TWTs for operation during pyro-firing events, the design of the propulsion system, and the use of fault-management software that was not fully understood-

c. Caution should be exercised when assessing industry expertise in delivering certain classes of spacecraft and extrapolating that capability to completely different mission requirements. As an example, the processes, documentation, and culture associated with, and appropriate for, commercial production-line spacecraft are basically incompatible with the discipline and documentation required for a one-of-a-kind complex mission. The Mars Observer was not a production-line spacecraft.

PART H
ACRONYMS and ABBREVIATIONS

ACRONYM	DEFINITION
4πSS	4π(Steradian) Sun Sensor
A/D	Analog-to-Digital
AACS	Attitude and Articulation Control System
AFPL	Air Force Phillips Laboratory
AGC	Automatic Gain Control
ANS	Array Normal Spin
AO	Announcement of Opportunity
AOS	Acquisition of Signal
APL/JHU	Applied Physics Lab, Johns Hopkins University
AUXOSC	Auxiliary Crystal Oscillator
BCA	Battery Charge Assembly
BCR	Battery Charge Regulator
BIT	Bench Integration Test
BLF	Best-Lock Frequency
bps	Bit Per Second
BPSK	Biphase Shift Key
BRE	Bipropellant Rocket Engine
Bus	Reference to Spacecraft (excluding payload/instrument)
BVR	Bus Voltage Regulator
C&DHS	Command & Data Handling System
C/N	Carrier to Noise
CD1/2	Clock Divider 1 (or 2)
CDR	Critical Design Review
CDU	Command Detector Unit
CIU	Controls Interface Unit
CIX	Controls Interface Extender
CLT	Command Loss Timer
CMD	Command
CMOS	Complementary Metal Oxide Semiconductor
CNES	Centre National d'Etudes Spatiales (French Nat'l Space Agency)
COCOMO	Constructive Cost Model
CPLR 4	Hybrid coupler
CPLR 6	LGA Coupler
CRC	Cyclic Redundancy Code
CSA	Celestial Sensor Assembly
c v	Command Verification
c w	Continuous Wave
D/A	Digital-to-Analog
dB	Decibels
dB/K	Decibel per degree kelvin
dBc	Decibels above Carrier
dBi	Decibels above Isotropic
dbm	Decibels referenced to 1 milliwatt
dBw	Decibels referenced to 1 watt

DC	Direct Cunent
DMSP	Defense Meteorological Satellite Program
DOR	Differential One-Way Ranging
DOY	Day of Year
DPRO	Defense Plant Representative Office
DSN	Deep Space Network
DSS	Deep Space Station
DTR	Digital Tape Recorder
EDAC	Error Detection and Correction (EDF RAM Software)
EDF	Engineering Data Formatter
EED	Electra-Explosive Device
EIRP	Effective Isotropic Radiated Power
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EPC	Electrical Power Converter
EPET	Electrical Performance Evaluation Test
EPS	Electrical Power System
ERT	Earth-Received Time
ESA	European Space Agency
F/D	Focal Length-to-Diameter
FB	Fuse Box
FB2	Fuse Board 2
FET	Functional Electrical Test
FLTSAT	Fleet Satellite
FMECA	Failure Modes Effects and Criticality Analysis
FOV	Field of View
FP	Fault Protection
FSW	Flight Software
FY	Fiscal Year
G	Gravity
g	Gram(s)
G A	Gain-to-temperature Ratio
Gb	Giga bit
GB	Gigabytes
GDA	Gimbal Drive Assembly
GDE	Gimbal Drive Electronics
GE	General Electric
GE ASD	General Electric - Astro Space Division
GFE	Government Furnished Equipment
GGS	Global Geospace Science
GHe	Gaseous Helium
GHz	Gigahertz
GMT	Greenwich Mean Time
GPS	Global Positioning System
GRS	Gamma Ray Spectrometer
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
HEF	High Efficiency
HGA	High Gain Antenna

HRMS	High-Resolution Microwave Survey
HZ	Hertz
I&T	Integration & Test
v o	Input/Output
IC	Integrated Circuit
IMU	Inertial Measurement Unit
IPTO	initial Power Turn-On (test)
IV&V	Independent Verification and Validation
JOVIAL	Jules' Own Version of an Interactive Algorithmic Language
JPL	Jet Propulsion Laboratory
KABLE	Ka-Bank Link Experiment
Kb	Kilobit
KB	Kilobytes
Kbd	Kilobaud
kbp	Kilobit per Second
kg	Kilogram
kHz	Kilohertz
km	Kilometer
ksps	Kilo-symbol per second
LGA	Low Gain Antenna
LMC	Link Monitor Console
LOD	Loss of Downlink
LOS	Loss of Signal
LRE	Liquid Rocket Engine (TITAN)
LRR	Launch Readiness Review
m	Meter
m/s	Meter per Second
MAG/ER	Magnetometer/Electron Reflectometer
MB	Magabytes
MBR	Mars Balloon Relay
MEOP	Maximum Expected Operating Pressure
MEU	Memory Extender Unit
MGCO	Mars Geoscience Climatology Orbiter (now Mars Observer)
MHSA	Mars Horizon Sensor Assembly
MIL-STD	Military Standard
MIPS	Mega Instructions Per Second
MJ	Mega joule
MMAS	Martin Marietta Astro Space
MMH	Monomethylhydrazine
MO	Mars Observer
MOC	Mars Observer Camera
MOI	Mars Orbit Insertion
MOLA	Mars Observer Laser Altimeter
MOT	Mars Observer Transponder
mrad	Milliradian
MWA	Momentum Wheel Assembly
N	Newton(s)
NASA	National Aeronautics & Space Administration
NATO IV	North Atlantic Treaty Organization Satellite 4

nm	Nautical Mile
NOAA	National Oceanic and Atmospheric Administration
NRL	Naval Research Laboratory
NRZ-L	Non-Return-to-Zero Level
NSI	NASA Standard Initiator
NTO	Nitrogen Tetroxide
ORS	Offset Radiation Source
OTM	Orbit Trim Maneuver
P/N	Part Number
Payload	Reference to device or instrument used for mission
PDR	Preliminary Design Review
PDS	Payload Data Subsystem
PI	Principal Investigator
PLL	Phase-Locked Loop
PMD	Propellant Management Device
POR	Power on Reset
PROM	Programmable Read Only Memory
PSA	Partial Shunt Assembly
PSE	Power Supply Electronics
PSK	Phase Shift Keyed
R-S	Reed-Soloman (decoder; encoding)
RAID	Real-Time Applications Interactive Debugger
RAM	Random Access Memory
RCS	Reaction Control System
REA	Rocket Engine Assembly
REDMAN	Redundancy Management
RF	Radio Frequency
RFI	Radio Frequency Interference
RFP	Request For Proposal
RHCP	Right-Hand Circular Polarized
RLC	Receive LGA Cycling
rms	Root Mean Square
ROM	Read Only Memory
RPA	RF Power Amplifier
RPM	Revolutions per Minute
Rs	Radio Science
RT	Real Time
RTC	Realtime Command
RTLT	Round-Trip Light Time
RWA	Reaction Wheel Assembly
RXO	Redundant Crytal Oscillator
s/c	Spacecraft
s/w	Software
SA	Solar Array
SAD	Solar Array Drive
SAGD	Solar Array Gimbal Drive
SAGDE	Solar Array Gimbal Drive Electronic
SCP	Spacecraft Controls Processor
SCU	Signal Conditioning Unit

SELTS	Self-Test Software
SEPET	System-Level Electrical Performance Evaluation Test
SETI	Search for Extra-Terrestrial Intelligence
SEU	Single Event Upset
SLOC	Source Lives of Code
SME	Sun-Mars-Earth (angle)
SNR	Signal-to-Noise Ratio
SPF	Single Point Failure
sps	Symbols per Second
SRR	System Requirements Review
SSA	Sun Sensor Assembly
SSI	Spectral Signal Indicator
SW1/2	Input Waveguide Transfer Switch 1 (or 2)
TCC	Time Code Counter
TCM	Trajectory Correction Maneuver
TES	Thermal Emissions Spectrometer
TIROS	Television and infrared Observation Satellite
TOS	Transfer Orbit Stage
TSF	Track Static Frequency
TVC	Thrust Vector Control
TWT	Traveling Wave Tube
TWTA	Traveling Wave Tube Amplifier
UHF	Ultra High Frequency
USO	Ultra Stable Oscillator
UTC	Universal time Coordinated
VIMS	Visual Infrared Mapping Spectrometer
VSWR	Voltage Standing Wave Ratio
VTL	Verification Test Laboratory
XMT	Transmit
xsu	Cross Strap Unit

VOLUME II

MARS OBSERVER Mission Failure . Investigation Board Report

Book 1 of 2

31 DECEMBER 1993

NOTE

Some sections in this Volume contain presentations that have been reproduced exactly as provided, by the presenter, during the investigation process.

The document has been bound to retain the order, within each section, as it was officially submitted.

TABLE OF CONTENTS
VOLUME II APPENDICES (BOOK 1 OF 2j

APPENDIX (A)

MISHAP REPORT

TABLE OF CONTENTS
APPENDIX (A)

National Aeronautics and Space Administration

NASA Mishap Report

NOTE: Fill in unshaded blocks within 24 hours. Please print or type. See reverse for instructions.

GENERAL INFORMATION

1. NAME OF ORGANIZATION: Solar System Exploration Division
2. MISHAP DATE (MDY): 08 21 93
3. MISHAP TIME (24 hrs): 2100EST
4. ORG. FILE No.: SSED-MO-1

5. MISHAP CATEGORY (Check as appropriate)

TYPE A	TYPE B	TYPE C	INCIDENT
1 ☐ DEATH	2 ☐ LOST TIME	2 ☐ LOST TIME	4 ☐ INJURY
2 ☐ LOST TIME	3 ☐ PERM. DISABILITY	4 ☐ INJURY	6 ☐ DAMAGE
4 ☐ INJURY	4 ☐ INJURY	6 ☐ DAMAGE	
6 ☐ DAMAGE	5 ☐ HOSPITALIZATION	7 ☐ TEST FAILURE	MISSION FAILURE
7 ☐ TEST FAILURE	6 ☐ DAMAGE		X
	7 ☐ TEST FAILURE		

6. CLOSE CALL ☐
7. LEVEL OF POTENTIAL: Mission Failure
8. BLDG. NO./LOCATION:
9. SPECIFIC AREA:
10. MISSION AFFECTED: Mars Observer
11. PROGRAM IMPACT: Possible loss of mission

12. DESCRIPTION OF MISHAP (Sequence of events, extent of damage and injuries, cause, if known, etc. Use additional sheets if necessary)

Communications have not been established following A propellent pressurization requencr on 8/21/93. AA/OSS declared a mission contingency on 8/26/93.

PERSONNEL INVOLVED

13. NAME (Last, first, middle initial)
14. AGE
15. SEX ☐ M ☐ F
16. ORGANIZATION (CODE)/POSITION
17. SHIFT WORKED ☐ 1 ☐ 2 ☐ 3
18. HOURS OF CONTINUOUS DUTY BEFORE MISHAP
19. FIRST AID ONLY ☐ YES ☐ NO
20. FATALITY ☐ YES ☐ NO
21. INJURY TYPE (Code)
22. BODY PART(S) AFFECTED (Codes)
23. DAYS LOST: NO. ☐ TOTAL ☐ CONTINUING
24. CAUSE(S) OF INJURY (Codes): PRIMARY / CONTRIB. / POTENTIAL
25. MISHAP ENVIRONMENT (Codes): AGENCY / ACTIVITY
26. HAS EMPLOYEE RECEIVED TRAINING/CERTIFICATION APPLICABLE TO TASK? ☐ YES ☐ NO

EQUIPMENT/PROPERTY DAMAGED

27. CLASS OF EQUIPMENT/PROPERTY DAMAGED

1 ☐ FLIGHT HARDWARE 4 ☐ PRESURE VESSEL 7 ☐ OTHER
2 ☐ GROUND SUPPORT EQUIPMENT (GSE) 5 ☐ MOTOR VEHICLE
3 ☐ FACILITY 6 ☐ AIRCRAFT

28. SPECIFIC ITEM DAMAGED
29. SERIAL/NEMS NO.
30. SYSTEM/SUBSYSTEM AFFECTED
31. CAUSE(S) OF DAMAGE (Codes): PRIMARY / CONTRIB. / POTENTIAL
32. COST: ESTIMATE $ FINAL $

33. SUBMITTED BY (Name, title, mail code)	SIGNATURE	PHONE NO.	DATE
William C. Panter, Manager Mars Observer Program, Code SLP	W C Panter	358-0315	8/30/93

CORRECTIVE ACTION

34. ACTION PLAN (Provide estimated completion date for each action. Use extra sheets if necessary)

35. APPROVED (Name, title, mail code)	SIGNATURE	PHONE NO.	DATE

36. NASA SAFETY CONCURRENCE WITH CORRECTIVE ACTION PLAN (Branch chief or higher)

CONCUR (Name, title, mail code)	SIGNATURE	PHONE NO.	DATE

NASA SAFETY OFFICE USE ONLY

37. LESSONS LEARNED ☐ YES ☐ NO REF. NO. (If Yes)
38. TYPE OF INVESTIGATION 1 ☐ BOARD 2 ☐ TEAM 3 ☐ INVESTIGATOR
39. STATUS ☐ OPEN ☐ CLOSED
40. APPROVAL FOR CLOSURE: NAME AND TITLE / PHONE NO. / SIGNATURE / DATE

MRB00332

NASA FORM 1627 (JAN 90 PREVIOUS EDITIONS ARE OBSOLETE) RCS 100000

SH

APPENDIX (B)

DIRECTIVE

TABLE OF CONTENTS
APPENDIX (B)

ARCHIVE

NASA

National Aeronautics and Space Administration

Washington, DC. 20546

Office of the Administrator

9/16

SEP 10 1993

Dr. Timothy Coffey
Director of Research
Naval Research Laboratory
Department of the Navy
Washington, DC 20375-5320

Dear Dr. Coffey:

In accordance with the Mars Observer Contingency Plan, NASA is establishing a "Mars Observer Mission Failure Investigation Board.' I am hereby appointing you to serve as the Chairman of this Board. Comprised of Government employees, this Board will be a working group charged to review, analyze, and evaluate the facts and circumstances regarding the loss of spacecraft communications and the failure of the Mars Observer mission. Your charge as Board Chairman is to determine the cause of this failure and to report the results of the evaluation directly to me. Additional information on the authorities and responsibilities of the Board is outlined in the enclosed Investigation Board Charter.

NASA will make available a team of support staff to assist the Board and will work with you to identify and support any financial requirements associated with Board travel and the initiation of any special analyses. The immediate point of contact at NASA Headquarters for information, assistance, and support will be the Mars Observer Program Manager, William Panter. He can be reached at 202/358-0310 (office) or 703/590-0552 (residence).

Again, I want to convey my appreciation for your willingness to chair the Board. Your leadership of this group will be instrumental in assuring a systematic review of the Mars Observer failure and any causes associated with it.

Sincerely,

Daniel S. Goldin
Daniel S. Goldin
Administrator

Enclosure

REFERENCE COPY

MRB11115

MARS OBSERVER MISSION FAILURE INVESTIGATION BOARD CHARTER

1. PURPOSE

This establishes the Mars Observer Mission Failure Investigation Board and sets forth its responsibilities and membership.

2. ESTABLISHMENT

a The Mars Observer Mission Failure Investigation Board is hereby established in the public interest to gather information, analyze, and determine the facts as well as the actual or probable cause(s) of the Mars Observer loss of communications in terms of (1) Primary Cause, (2) Contributing Cause(s), and (3) Potential Cause(s) (pertinent observations may also be addressed) and to recommend preventive and other appropriate actions to predude recurrence of a similar mishap.

b. The Chairperson of the Board will report to the NASA Administrator.

3. AUTHORITIFS AND RESPONSIBILITIES

a. The Board will-

(1) Obtain and analyze whatever evidence, facts, and opinions it considers relevant by relying upon reports of studies, findings, recommendations, and other actions by NASA officials, contractors, subcontractors, or others by conducting inquiries, hearings, tests, and other actions it deems appropriate. In so doing, it may take testimony and receive statements from witnesses.

(2) Impound property, equipment, and records to the extent that it considers necessary.

Note: Impoundment may not necessarily predude release of information General information which would normally be released or had been released previously can continue to be released.

(3) Determine the actual or probable cause(s) of the Mars Observer mission failure and document and prioritize its findings in terms of (a) the Primary Cause(s) of the Mishap, (b) Contributing Cause(s), and (c) Potential Cause(s). Pertinent observations may also be made.

(4) Develop recommendations for preventive and other appropriate actions. A finding may warrant one or more recommendations, or it may stand alone.

(5) Provide a final written report to the NASA Administrator by November 20, 1993. The requirements in NMI 8621 .1 F wilt be followed

b. The Chairperson will--

(1) Conduct Board activities in accordance with NMI 862 1.1 F and any other instructions that the NASA Administrator may issue.

(2) Establish and document, to the extent considered necessary, rules and procedures for the organization and operation of the Board, including any subgroups, and for the format and content of oral or written reports to and by the Board.

(3) Designate any representatives, consultants, experts, liaison officers, or other individuals who may be required to support the activities of the Board and define the duties and responsibilities of those persons.

4. MEMBERSHIP

The Chairperson, members of the Board, and supporting staff are designated in Attachment A.

5. MEETINGS

The Chairperson will arrange for, and record the transactions of, all meetings held in conjunction with Board proceedings.

6. ADMINISTRATIVE AND OTHER SUPPORT

a. The Director of Research of the Naval Research Laboratory will arrange for office space and other facilities and services that may be requested by the Chairperson or designee.

b. All elements of NASA will cooperate fully with the Board and provide any records, data, and other administrative or technical support and services that may be requested.

c. The NASA support personnel as specified in Attachment A can be augmented by NRL as appropriate.

7. DURATION

The NASA Administrator will dismiss the Board when it has fulfilled its requirements.

8. CANCELLATION

This appointment letter is automatically canceled 1 year from effective date of the publication, unless otherwise specifically extended by the establishing authority.

Daniel S. Goldin
NASA Administrator

ATTACHMENT A

Members and Supporting Staff
Mars Observer Mission Failure investigation Board

MEMBERS

Chairperson: Tiiothy Coffey, Director of Research, Naval Research Laboratory

Members: Thomas C. Betterton, Rear Admiral, USN
Peter G. Wilhelm, Director of Naval Center for Space Technology, NRL
Michael D. Griffin, Chief Engineer, NASA
Joseph Janni, Chief Scientist, Air Force Phillips Laboratory
Kathryn D. Sullivan, Chief Scientist, National Oceanic and Atmospheric Administration

Ex Officio Member (non-voting): Leven B. Gray, NASA Code Q

SUPPORTING STAFF

Executive Secretary:	Joe G. Foreman, NRL Code 8001 .1
Program Technical Liaison:	William C. Panter, NASA Code S
NRL Administrative Support:	Kenneth W. Lackie, NRL Code 1001 .1
NASA Administrative Support:	Carrie L Sorrels, NASA Code S

Advisors:

Counsel •	George E. Reese, NASA Code G
Public Affairs •	Paula Clegget-lialeim, NASA Code P
	James Gately, NRL Code 1230

APPENDIX (D)

MAINTENANCE/TEST/ QUAL RECORDS DIRECTLY RELATED TO SURVIVING SCENARIOS

TABLE OF CONTENTS
APPENDIX (D)

31 AUG '93 16:22 P.1

#18

esa estec

TELEFAX / TELEX / TELEGRAM FORM.

(to be typewritten)

FROM : D DALE	PRIORITY	For telex or fax, use only
mailcode:	Normal	
	Urgent	
Ref. Date:	Page No 1	
Subject	No. of pages 3	
Fax/telex no.	Info.	distribution (by mail)
TO : A. DIAZ.		

Al

As discussed on the phone.

Regards,

DAVE

1467-19
1467-20 } OEA/Denver

Originator's signature .... Approved by

ESA / ESTEC, Postbus 299, 2200 AG Noordwijk, The Netherlands, telex no

Table III

Normally cloned Valves

Part Number	Size (in)	Material	Program	Temperature
1467	3/8	Titanium	Olympus	-76°F, single 80% +170°F, dual 130%
1467-5	3/8	Titanium	KU3/KU4, ANIK-E, TELSTAR	+71°C (+160°F) +25°C (+77°F) -34°C (-29°F)
1467-6, -7, -8	3/8	Titanium	SABIR-FE	ambient, ambient, no data
1467-9	3/8	Titanium	DSCS-LABS, GBI (LMSC/Aerojet)	+25°C (+77°F) +71°C (+160°F) -34°C (-29°F)
1467-10, 1467-11	1/4	Titanium	UHF(F1), UHF(F2-11 F9) NS-601(HF), AUSSAT, BRASILSAT	single -30°C dual +171°C single -10°F dual +150°F
1467-14	1/8	Titanium	Miniature Propulsion Subsystem, GBI (LMSC/Aerojet)	ambient
1467-15	3/8	Titanium	UHF(F1), UHF(F2-F9)	single -10°F dual +150°F
1467-16	1/4	Titanium	Miniature Propulsion System	ambient
1467-17	1/8 (orificed)	Titanium	Miniature Propulsion System, GBI, (LMSC/Aerojet)	ambient
1467-18	3/16	Titanium	SCIT	ambient
1467-19	3/8	Titanium	Mars Observer	see attached
1467-20	3/8	Titanium	Mars Observer	see attached
1467-21	3/16 Vent	Titanium	SCIT	ambient
1467-22	3/8	Titanium	TELSTAR	+71°C (+160°F) +25°C (+77°F) -34°C (-29°F)
1467-23	1/4	Titanium	Cluster	+122°F -22°F +68°F
1467-24	3/8	Titanium	Cluster	+122°F +32°F +68°F
1467-28	3/16	Titanium	ALAS	?
1467-29	1/4	Titanium	ALAS	ambient
1467-30	3/16	Titanium	ALAS	ambient
1467-31	3/16 (orificed)	Titanium	ALAS	ambient

FIRING HISTORY OF 3/8" ALL-TITANIUM PYROVALVES WITH 325 MG HI-TEMP BOOSTER CHARGE

PART NUMBER	QUANTITY DELIVERED	QUANTITY FIRED BY OEA	FIRING CONDITIONS (TEMPERATURE, OVER/UNDER CHARGE, ETC.)
1467*	---	6	Single initiator: -60'C/80%, +77'C/100%, ambient/80%
			Dual initiator: -60'C/130%, +77'C/130%, ambient/100%
14 67-5	34	18	71'C, 25'C, -34'C
1467-9	65	32	71'C, 25'C, -34'C
1467-15	21	3	Dual initiator: 66'C, Single initiator: -23'C
1467-19	26	10	-45'C/80% &120%, +75'C/80% & 120%, +25°C/100%/Dual
1467-20	4	---	---
1467-22	6	---	---
1467-24	9	3	Single initiator: 0'C/80%, 22°C/100%
			Dual initiator: 50'C/130%
1467-37	24	4	71'C, 24'C, -34'C
1467-38	12	---	71'C, 24'C, -34'C
Totals	201	76	
1466**	---	6	Single initiator: -60'C/80%, +77'C/100%, ambient/80%
			Dual initiator: -60'C/130%, +77'C/130%, ambient/100%
1466-5	20	18	71'C, 25'C, -34'C
1466-15	40	6	Dual initiator: 66'C, Singie initiator: -23'C
1466-16	3	---	---
1466-18	12	3	71'C, 24'C, -34'C
1466-19	22	---	---
Totals	97	33	

* see LO-1467
** see LO-1466

#36
SH

INFORMATION FOR NASA REVIEW GROUP

a) Provide **pressure test data for propellant tanks** *(bipropellant tanks).*

The data generated during the tank qualification program is contained qualification report. System level test data of the tanks during spacecraft Integration and Test resides at the spacecraft manufacturer. The following is a simplified history of tank testing at the component. propulsion system. propellant loading and flight levels. The tank qualification report and sysrem test data should be obtained from the Martin Marietta Astro-Space Corporation.

The bipropellant tanks are constructed of Titanium. The tanks have a Maximum Expected Operating Pressure (MEOP) of 300 psia Borh tanks were subjected to a proof pressure test of 375 psia. during component acceptance test. During qualification test the qualification unit was subjected to a burst test at 450 psia.

During Propulsion sysrem level testing the propellant tanks were pressurized to approximately 260 psia (regulated pressure). several rimes. primarily for system level leak tests. During propellant loading. approximately 60 days before launch. rhe Oxidizer (NTO) tank was filled with X38.4 KG of N_2O_4.the tank was then pressurized to 300 psig with Gaseous Helium (GHe). The Fuel tank was filled with 5 l 2.06 KG of onomethylhydraune (MMH). the tank was then pressurized with Helium to 290 psia. At launch. the pressure in the NTO tank was 250.8 psia the pressure in the MMH tank was 257.9 psia The change in pressure between launch and propellant loading was due to helium saturation of the propellants.

The following table shows the pressure and temperature of rhe NTO and MMH ranks during flight.

Date	Action	NTO Tank Pressure (psia)	NTO Tank Temp (°C)	MMH Tank Pressure (psia)	MMH Tank Temp 0)
9/25/92					
10/10/92	Launch l	250.8 287.8	30	257.9 259.7	22 26
10/10/92	postTCM-TCM-	230.6	30	206.9	25
01/04/93	(data point)	208.7	20	203.3.	24
02/08/93	- preTCM-2	188.6	13	197.8	21
02/08/93	postTCM-2	183.1	12	190.6	11
03/18/93	TCM-3	166.6	3	l ii.8	13
05/03/93	(data point)	137.4	0.4	163.3	3.5
08/19/93	pre Press.	159.2	I.!	165.1	4.3

b) **Provide** test history of regulator.

The basic regulator was qualified for the Space Shuttle for use in the Reaction Control System (RCS) the qualification tests included extensive testing in Oxidizer and MMH environments.

The official test history of the regulator is contained in the regulator qualification report and the qualification by similarity report for Mars

REFERENCE COPY

Observer. Copies of the qualification reports should be obtained from the Martin Marietta Astro-Space Corporation.

c) Susceptibility of pressure regulator to build-up of NTO Corrosion products.

The qualification program of the regulator for the STS application included extensive testing of the regulator in an Oxidizer vapor environment. The data from these tests arc located in the qualification report. Preliminary results of work in this area by JPL and Martin Marietta Astro Space is as follows.

Attachment 1, "Oxidizer Helium Regulator Flight Experience", from NASA JCS shows the history of this regulator during the Shuttle program usage. One ground test regulator, S/N0035 suffered a failed closed condition during ground test at White Sands Test Facility (WSTF). The unit was removed from the test setup and placed in storage for several years. The unit was removed from storage and retested and found to be failed in the open position. A failure investigation report of the open condition was written and can be obtained from WSTF. The conclusion in the report was "... it is concluded that S/N 0035 failed to lockup in all inlet pressure conditions was because of the crystalline formation on the outer convolute of both main bellows assemblies." This report fails to make a conclusion as to the source of the contamination or to note that (after a period of storage) the regulator was found to be stuck in the open position, not the closed which led to its removal from the WSTF test. The conclusion by NASA/JCS (attachment 1, chart 2) concludes that the failure(s) were due to;

(1) The regulator was subjected to non flight representitive test environment.

(2) The regulator remained in storage for approximately 3 years, most probably contaiminated with residual oxidizer.

The failure investigation included an analysis of the contamination. Calcium and Aluminum were found by Scanning Electron Microscope (SEM) and Energy Dispersive X-Ray (EDAX) examination. These two elements are not used in propulsion systems or test facility's. This tends to support the NASA/JCS conclusions.

These results arc considered preliminary further investigation is on going by the JPL review board and Martin Marietta Astro-Space.

d) Other Data: Water Hammer Analysis

Prior to launch a test was performed, using spare flight hardware, to determine the effects of water hammer due to opening the bipropellant latch valves down stream of the propellant tanks, during priming of the bipropellant system. The test was performed at WSTF the data can be obtained from the Martin Marietta Corporation. An analysis/computer simulation was performed by TRW under contract to JPL. The results are detailed in TRW Final Report, "TOPEX and Mars Observer Waterhammer Analysis", 25 August 1993. E

Y. Wong and, H. W. Behrens, Contract No. 959293, Sales No. 60323.000 and 60323.00 1.

Both the analysis and the hardware test at WSTF indicated that pressure spikes caused by Water/Propellant hammer during priming of the system were well within the tolerances of the system. Priming took place approximately 7 days after launch.

e) Other Data: Flow analysis for reaction Control System.

More definition of information requested is necessary.

NASA
National Aeronautics and Space Administration

Facsimile Transmission
(Instructions on Reverse)

Attachment 1

1. TRANSMITTING STATION NO.	2. VOICE PHONE CONTACT NO.	3. MESSAGE NO.	4. TOTAL PAGES (Including lead)	5. DATE
			12	9-3-93

6. FROM (Name, organization and location):	7. OFFICE CODE / 8. OFFICE PHONE NO.
John B. Henderson NASA-JSC	EP4 (713) 483-900[illegible]

9. TO (Include Office Code and telephone number.) (May also be used for remarks):

Edmund Stalof

Martin Marietta Astro Space

(609) 490-6517 PHONE
(609) 490-3963 FAX

Carl Guernsey / Phil Garrison

NASA-JPL

(818) 354-9144 Phone
(818) 393-6682

HERE are our summary charts for Shuttle RCS regulator history for both the flights systems & ground test articles, along with a copy of the pertinent pages from the S/N 035 draft test report. We would like to go over these with you someti[me] today. Give me a call at (713) 483-9008 or John Griff[in] at (713) 483-9003.

[signature]

RCS Oxidizer Helium Regulator Flight Experience

Propulsion and Power Division

RCSubsystem | September 2, 1993

- SPACE SHUTTLE RCS OXIDIZER HELIUM REGULATORS HAVE AN EXTENSIVE OPERATIONAL HISTORY

  - FOUR ORBITERS, EACH WITH SIX SERIES REDUNDANT HELIUM REGULATORS FOR THE OXIDIZER SYSTEMS

  - OF THE 24 OX HELIUM REGULATORS ON THE FOUR VEHICLES, ONLY SEVEN HAVE BEEN REMOVED DUE TO FAILURES

    - MAJORITY ATTRIBUTED TO PARTICULATE CONTAMINATION ON SEAT CAUSING LEAKAGE
    - 1 CASE OF LOW REGULATED PRESSURE DUE TO CRACKED BELLEVILLE
    - 1 CASE OF HIGH LOCK-UP (-10 PSIA HIGH) DUE TO AN OUT OF CONFIGURATION PRIMARY SPRING SUPPORT
    - 1 CASE (LEAKAGE) WHERE CORROSION PRODUCTS WERE FOUND THROUGHOUT THE REGULATOR; SUSPECT H20 INTRUSION DURING BUILD-UP

  - 17 REGULATORS FROM ORIGINAL VEHICLE BUILD ARE STILL IN SERVICE, WITH SYSTEM EXPOSURE PERIODS RANGING FROM 433 DAYS TO 3803 DAYS

    - MARGINAL CHECK VALVE PERFORMANCE HAS RESULTED IN OXIDIZER VAPOR MIGRATION TO REGULATORS ON ALL VEHICLES

Oxidizer Helium Regulator Flight Experience

Propulsion and Power Division

RCSubsystem | September 2, 1993

- WSTF S/N 0035 REGULATOR HAS BEEN SUBJECTED TO HARSH TEST ENVIRONMENTS THAT ARE NOT CONSIDERED REPRESENTATIVE of ACTUAL FLIGHT SERVICE

  - OPF PURGE TESTING WHERE REGULATOR (AND ENTIRE HELIUM SYSTEM) WAS 'EXPOSED 'TO 100% OX SATURATED HELIUM OVER AN EXTENDED PERIOD OF TIME

  - SPECIAL TESTING IN WHICH REGULATOR WAS EXPOSED TO CONDITIONS THAT WOULD BE EXPECTED TO INTRODUCE LIQUID OXIDIZER INTO THE REGULATOR

- S/N 0035 FAILED OPEN FAILURE OCCURRED AFTER REGULATOR SAT FOR THREE YEARS ON THE MOTH-BALLED FORWARD TEST ARTICLE

  - RESIDUAL OXIDIZER MOST PROBABLY REMAINED IN REGULATOR FROM THE PREVIOUSLY TESTING

CAR #	S/H	Install	Remove	Days	# Flts	Comments
WSTF Regulator Failures Resulting In R&R						
AB5315	22				WSTF	EI081; leakage; N2O4 attack of main seat gold plate
A86692	22				WSTF	EI-081; leakage; gold plate
AB7558	20				HSTF	EI-081; leakage; gold plate
AB9769	20				WSTF	EI-081; leakage; gold plate
AC0653	22				WSTF	EI-081' leakage; gold plate; no action (waived)
AC0746	20				WSTF	EI-081; leakage; gold plate; seal damage
AD8120		10/1/82	e/30/91		WSTF	High leak and lock-up; exp. off-nominal testing
KSC Regulator Failures Resulting In R&R						
02F011	4[illegible]	4/12/81	12/1/82	598	5	High lock-up due to out-of-config prim spring supp.
AC9134	4[illegible]	11/28/83	12/1/83	3	1	Leakage due to corrosion generated particles; moist
AC9134	5[illegible]	4/1/81	12/1/83	974	6	ure intrusion suspect; tube clogging noted (AC9134)
AD0134	40	4/1/83	8/21/85	873	7	Internal leakage; waived; not removed at this time
AD1834	20	6/1/85	2/1/86	245	3	Trans. leak; Corr. products; Suspect H2O at b/u
AD1882	44	4/1/83	12/1/88	2071	8	White contam between pilot ball and seat
AD1883	40	4/1/83	11/30/88	2070	8	Probably pulled during downtime for earlier leakage
27RF10	42	4/1/83	12/31/88	2101	9	Cracked belleville spring; H2 embrittlement
53RF02	66	6/1/85	3/15/93	2844	12	Metallic particulate contam on prim. pilot poppet

. Note : Install / Remove dates are in some cases 'best guess' numbers based on initial flight dates.

Vehicle Downtime After 51-L And Associated Regulator Failures

Veh		Downtime				Regulator Failures
102		1297				s/n 20 removed right after 61-C flight
103		1122				None noted
104		1095				s/n 44 (AD1682) - not clear why removed from veh.
						s/n 40 (AD1883)
						s/n 42 (27RF10) - cracked belleville

MARS Obs/Reg Fail History

Original Veh		Regulators Still On Vehicle 1st Flight	Current	Days	Original Regs Still In Service
99	LP01	4/4/83	9/1/93	3803	0 of 2
	RP01	4/4/83	9/1/93	3803	1 of 2
102	FRC2	4/12/81	9/1/93	4525	0 of 2
103	FRC3	8/30/84	9/1/93	3289	2 of 2
	LP03	8/30/84	9/1/93	3289	2 of 2
	RP03	8/30/84	9/1/93	3289	2 of 2
104	FRC4	10/3/85	9/1/93	2890	2 or 2"
	LP04	6/17/85	9/1/93	2998	0 of 2
	RP04	4/29/85	9/1/93	3047	2 of 2
105	FRC5	s/7/92	9/1/93	482	2 of 2
	LP05	6/25/92	9/1/93	433	2 of 2
	RP05	6/25/92	9/1/93	433	2 of 2

Total 17 of 24 (70%)

-IN U33 TEST REPORT (DRAFT)
(PR104)

1.0 Introduction

In April 1987, the NASA Johnson Space Center (JSC) Propulsion and Power Division requested that the White Sands Test Facility (WSTF) perform a helium system surge activation test on the forward reaction control system (FRCS) qualification test article installed in WSTF Test Stand 328 (TS-328). The results were reported in TR-523-001.

In July 1987, Rockwell International (RI) requested additional FRCS testing concerning helium regulator response characteristics. Consequently, additional testing was added to the WSTF TD-523-001, via test change request 1 (TCR-1). The testing consisted of:

- Nonfiring regulator flow tests at various inlet pressures simulating Kennedy Space Center (KSC) Orbiter processing facility (OFF) ground support equipment (GSE). An identical system was built at the launch pad (PAD) after this testing was completed.

- Investigation of procedures for purging propellant contamfnanu from the helium system.

During investigation of purging procedures, an anomaly on the oxidizer "B" secondary pressure regulator PR104 was discovered. Tut change request 2 (TCR-2) added the PR104 exposure tests to investigate this anomaly. Test results from both TCR-1 and TCR-2 are discussed in this test report.

2.0 Test Objectives

The test objectives were to:

- Determine the validity of current Operations and Maintenance Rquiremenu and Specification Document (OMRSD) regulator checkout requiremenu and establish revised criteria if necessary.

- Obtain engineering data to define procedures and equipment to screen regulators for slow response at the KSC OPF.

- Determine the effectiveness of the helium blowdown purge for removal of propellant vapors from the helium pressurization system lines.

- Investigate the PR104 anomaly.

3.0 Test Summary

Tasting began in July 1987 and ended in March 1988, and was divided Into two categories. check valve functional tests and regulator response tests. Check valve functional tests were required to provide baseline data on the condition of each check valve before regulator flow testing. Regulator response tests ware conducted In four seriu: OPF regulator response tests, 800-psi helium tank blowdown tests, PAD regulator response tests, and the OPF purge tests.

The OPF regulator response test data showed that the regulators, with the *exception* of PRIM, were generally within accepted test criteria and flow traces were easily discernible. Additionally, the regulators especially sensitive to the flow rate used at lower Supply pressures.

U.S. Gov't

The 800-psi helium tank blowdown tests used the tank vent Quick Disconnect (QD). Most regulators could not maintain regulated pressure within specification at flow In excess of 130 scfm.

The PAD regulator response tests used the regulator checkout QD, and like the 800-psi helium tank blowdown tests, most regulators could not maintain regulated pressure within specification at flow in excess of 150 scfm.

The OPF purge tests evaluated the effectiveness of off-loading residual helium through the regulator checkout QD to purge propellant vapors from the helium prurant lines. Test data indicated that this method was not effective in removing propellant vapors from the helium system. However, a system was successfully developed to sample propellant contamination in very small line sections and used for the contamination purge tests.

The regulator checkout requirement procedures verified at WSTF are now used at KSC to screen regulators with slow response and the OMRSD was changed accordingly.

Regulator follow-on tests were added and centered around the PR104 oxidizer "B" secondary regulator failed-closed anomaly that occurred during the PAD regulator tank vent tests. These tests were performed using different contamination and flow pressures in an anempt to re-create the anomaly. Despite all efforts, the anomaly could not be re-created and the rwon for this regulator failure remains unexplained.

4.0 Test Configuration

The test configuration included the test article, test facility, propellant sample, and environmental conditions.

4.1 Test Article

The FRCS EI-081 configuration was in accordance with drawings VT70-421002 (test article complete) and VI70421003 (RCS), with the oxidizer tank swirl diffuser (SK5C3372000-11) induction system installed. The oxidizer system had been modified to the OY-99 configuration. except for the unique fill/spill and regulator checkout tube routing with QD locations *on* opposite panels. The modification was for the acoustic fatigue test.

Modifications are detailed in the following drawings:

VO70-421702 -- Tubing and installation modification
VO70-0421406-049, -058, and -060 -- Tubing
VO70-316235 -- Added QD mounting hole

The fuel tank swirl diffuser, drawing VT70-421303, was installed to facilitate OV-99 modification& but the flange was not connected to the module pressurization system. Appendices A, B, and C show the instrumentation list, discrepancy record summary, and tank cycle record, respectively.

During the tests, the following helium regulators were present in the test article:

PART NUMBER	SERIAL NUMBER	LOCATION
K-284-0418-0002	0023	PR-101
MC-284-0418-0001	0007	PR-102
K-284-0418-0002	0013	PR-103
MC-284-0418-0001	0035	PR-104

Before the regulator response tests, the FRCS helium system was modified to the full OV-99 configuration. TPS 3FTQA-099 and 3FTQA-101. The modification, performed in suppon of the KSC propellant tank checkout GSE verification test, convened the FRCS test article to a flight tubing configuration for future testing.

43 Tat Facility

The test facility configuration for regulator flow tests is shown on the fuel and oxidizer flow system schematics provided in appendix D. The flow system was fabricated, assembled. and installed at TS-328. Line length and component simulation duplicated the KSC OPF regulator test facility configuration.

4.3 Propellant Samples

Oxidizer samples were taken from the auxiliary conditioning unit (ACU) at TS-328 before the test article tank loading. Oxidizer sample analysis indicated that propellant iron content was not within specified limits. Molecular sieve equipment installed at the ACU removed the excess iron from the oxidizer. Initial and final oxidizer sample reports are included in appendix E. Fuel samples were not taken because the fuel system tests were deleted.

4.4 Test Conditions

Tests were performed at various temperatures and pressures. Conditioned temperatures in the test stand were between 40 and 70 °F. Test stand ambient atmospheric pressure ranged between 12.23 to 12.30 psi.

5.0 Test Description

Two groups of tests were performed; the check valve functional tests helped to establish baseline criteria for the four series of regulator response tests. Shortly after testing was completed, original data listings and plots of the regulator flow tests were sent to JSC and RI representatives for analysis. Test compilations and data plots showing pressure and flow transients on the test article regulators during different flow conditions are provided in appendices G through L.

5.1 **Check** Valve Functional Tests

Functional checks of the test article quad check valves were performed to verify that leakage through the poppet was within acceptable limits for the various regulator and contamination purge tests. Valves were **checked** individually and combined, from the upstream and downstream sides, and the leak was measured by a volumetric leak detector. Leak checks were accomplished using high-pressure helium at 250 ±6 psig and low-pressure helium at 1 ±0.5 psig. Cheek valve functional test results are contained in appendix G.

5.2 Regulator Response Tests

Regulator response tests consisted of five individual test series: OPF regulator response tests, 800-psi helium tank blowdown tests, PAD regulator response tests, OPF purge tests, and the PR104 regulator anomaly and follow-on contamination tests.

Regulator response tests were performed according to this typical regulator flow test sequence:

- The manual valve was verified closed. When it was necessary to open the manual valve, opening and running torque readings wera taken.
- The regulator flow panel system shown in the schematics in appendix D was connected to the checkout QD at MD123 and MD126, then configured for the tests. Flow test systems were a close simulation of the actual OPF GSE. The actual KSC system was measured to verify that the WSTF system configuration was functionally identical. Quick disconnects MD123 and MD126 were opened and the standard 0.5 ft^3 regulator flow system ullage was placed on line.
- According to specific test requirements, the helium tanks were adjusted to test pressure.
- Depending on the test, the regulators were locked up at varying pressures. Pressurization was accomplished using the helium checkout panel through fuel QD MD101 and MD103 and oxidizer QD MD102 and MD104. Ambient pressure was verified on the sense port and the helium isolation valve was activated. Then the second helium isolation valve was activated.
- The secondary sense port of the test regulator, usually regulator "B," was pressurized to 40 ±1 prig and the secondary lockup pressure was verified.
- Test flow was set on the test regulator using repetitive trials as required. Flow was initiated and terminated using the demand valve in the flow test GSE.
- The data system was then enabled and when flow was initiated, the test regulator condition was recorded.
- The helium regulator leg was changed, usually from the "B" leg to the "A" leg, and the second test regulator was flow-checked as before.
- The secondary sense port was then depressurized from 40 psia to ambient pressure and the regulator flow test was repeated.

In the OPF purge tests, the following method was used to introduced saturated vapors into the helium system:

- The area above and below the quad check valves was interconnected and then connected to an aspiration point/isolation valve fuel QD MD123 to QD MD111 and oxidizer QD MD126 to QD MD112.

- The tank manual valves, fuel MV101 and oxidizer MV102, were closed and the tanks were verified by chemical analysis to contain helium 100 percent saturated with Propellant vapors.
- The area above and below the quad check valves was evacuated by opening QD MD123 and MD1 11, and MD126 and MD1 12, allowing the respective system to evacuate to approximately 1 psia pressure.
- The aspiration/isolation valve was closed and the system vacuum was verified not to degrade.
- The respective manual valves, MV101 and MV102, were opened to allow propellant-saturated helium to be sucked into the helium systems.

6.0 Test Results

To ensure that test requirements were met, data was reviewed after each test. Final analysis of the test data was performed by RI and the OMRSD was changed accordingly.

6.1 Check Valve Functional Tests

Check valve functional tests provided baseline data on the check valve condition before regulator flow tests and arc shown.in appendix G.

6.2 Regulator Response Tests

A series of regulator response tests were conducted.

63.1 OPF Regulator Response Test

The OPF regulator response test simulated OPF conditions at KSC during regulator checkout. An initial WSTF review of the test data showed that although the regulator lockup pressure was high at times, the flow data was generally within acceptance test criteria and the traces were easily discernible. It was also shown that the regulators were especially sensitive to the flow rate used at lower supply pressures. Data was gathered as criteria for evaluating flight hardware performance.

The OPF regulator response test data and plots are provided in Appendix H.

6.2.2 800-psl Helium Tank Blowdown Response Test

The helium tank blowdown response test provided engineering information regarding regulator performance at low inlet pressures. Data was used to determined whether regulator performance could be evaluated during a 150-scfm flow with an initial helium tank pressure of 800 psia.

The flow rate used in performing the tests was critical. Most regulators could not maintain regulated pressure within specification at flow in excess of 150 scfm. Marginal regulators also could not regulate pressures during flow rates in the 130 to 150 scfm range. At the lower flow raw, the undershoot/overshoot was much less discernible.

Helium tank blowdown response data and plots are provided in appendix I.

U.S. Gov't

62.3 PAD Regulator Response Tests

The PAD regulator response tests simulated the conditions chat would exist if regulator checkout were required at the launch pad.

Two different techniques of testing regulator response characteristics at the PAD were planned. The first technique was to use the regulator checkout QD. The second technique was to use the tank vent QD, which is more accessible at cht PAD. However, a failure of PR104 caused the deletion of the tank vent QD tests and only one technique was evaluated. Test data also provided a baseline for comparison of flight regulators.

The PAD regulator response test data results and plots are provided in appendix 1.

6.2.4 OPF Purge Tats

The OPF purge tests evaluated the effectiveness of the present maintenance requirement. This requirement called for off-loading residual helium through the regulator checkout QD, in an effon to sweep propellant vapors from the helium pressuranc lines.

Significant time and effort were expended co verify that the systems *were* filled with a 100 percent saturated mixture of helium and propellant vapors.

Sampling small system volumes was difficult and initial sample results were frequently Inconsistent. However, refinement of chc method improved repeatability. providing sample rtsulu within an acceptable range.

Results from chc OPF purge tests indicated that the proposed high-flow purge would remove large amounts of the propellant vapors, but the purge would not clean cha system to the expected levels. Significant propellant vapors remained In those systems.

Apparently, coo many 'trap' areas not in the direct flow path existed to allow for more complete removal of the contamination with such a short duration helium tank blowdown.

The OPF purge test results and plou are provided in appendix K.

63.5 Regulator Anomaly and Follow-on Contamination Tests

During performance of the PAD regulator response tests, the PR104 oxidizer "B" secondary regulator failed-closed. Because this type of failure had never been seen in the Shuttle program, a careful diagnostic tart was performed on PR104 to determine the cause. Subsequent tests were performed using different states of contamination and flow pressures to try to re-create the anomaly.

A 3-liter Hoko bottle, half filled with N_2O_4, was connected to MD126-MD112 with a "Y" connection and the bottle was maintained approximately 10 °F above FRCS regulator temperature. Test volume was then allowed to cycle with the ambient temperature swings, approximately 35-75 °F.

Although the PR104 flow performance remained sluggish during these tests, daily cycling of stand temperatures and precipitation of oxidizer liquid into the regulator area apparently did not cause regulator performance to degrade further. Data showed that PR104 remained essentially the same and improved slightly after being exposed IO these conditions for 3 weeks. No instance Of the PR104 complete failure to open reoccurred.

During the last set of flow tests, it was noted that helium isolation valve LV102 stuck in the open position. After several attempts, LV102 closed but remained sluggish during subsequent valve cycles. This anomaly may have been caused by introduction of the oxidizer vapors into the helium isolation valve area.

Test data and representative test plots are provided in appendix L.

7.0 Conclusions

The OPF Regulator Response test data showed that the regulators were generally within the test criteria of the OMRSD and the flow traces were wily discernible. The regulators were especially sensitive to the flow rate used at the lower supply pressures.

The 800-psi Helium Tank Blowdown Response test showed that the flow rate was critical. Most regulators could not maintain regulated pressure within specification at flows in excess of 150 scfm.

The PAD Regulator Response test used a different outlet path and like the Helium Blowdown Response test, most regulators could not maintain regulated pressure with specification at flows in excess of 150 sch.

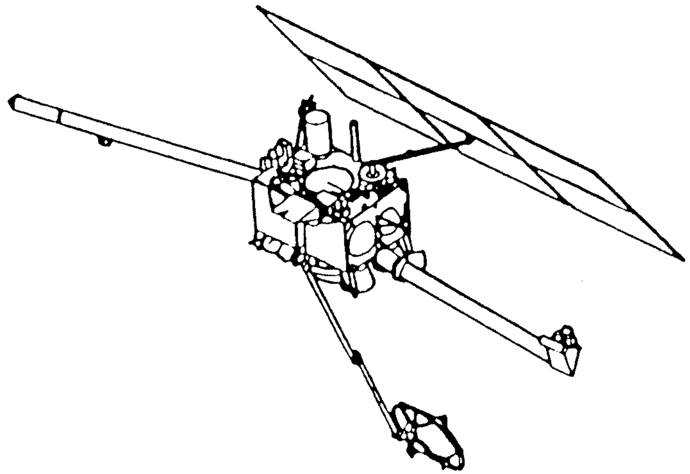
The OPF Purge test results indicated that a high-flow purge through the regulator checkout QD was not an effective method to clean the system to the expected level.

Follow-on tests were performed on the PR104 oxidizer secondary regulator when it failed closed during the PAD Regulator Vent test. Several attempts were made to re-create the anomaly, but the failure did not re-occur.

JPL

Skt

MARS OBSERVER
SPACECRAFT DEVELOPMENT



GEORGE PACE

APRIL 23, 1993

AGENDA

- DESIGN APPROACH
- **SYSTEM DESIGN**
- SUBSYSTEM DESIGN
- TEST PROGRAM
- CONCLUSION

DESIGN APPROACH

BASIC ASSUMPTION:

USE EXISTING EARTH ORBITER DESIGNS → MAKE MINIMUM CHANGES TO ACCOMPLISH THE MISSION → MARS OBSERVER SPACECRAFT

SATCOM K BUS

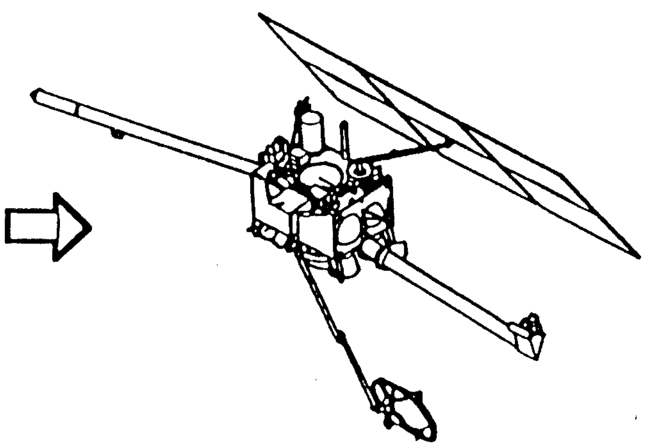
- •NADIR POINTING PANEL FOR INSTRUMENTS
- *LARGE PROPELLANT CAPABILITY
- ● STS LAUNCH QUALIFIED

DMSP/TIROS SUBSYSTEM COMPONENTS

- •28 V REGULATED POWER
- •3 AXIS STABILIZED
- *PRECISION NADIR POINTING
- *CENTRAL GENERAL PURPOSE FLIGHT COMPUTER
- *PROGRAMMABLE TELEMETRY FORMATTER
- *TAPE RECORDERS

DESIGN DRIVERS

- **CLASS A MISSION**
- **MISSION LIFE**
- **LAUNCH VEHICLE**
- **MISSION REQUIREMENTS**
- **PAYLOAD REQUIREMENTS**
- **DATA STANDARDS**



SUBSYSTEM

DETAILS

ADDENDUM

JPL COMMAND & DATA HANDLING (C&DH)

COMMAND FUNCTIONS:
- RECEIVES & ROUTES DEMODULATED INCOMING COMMANDS
- PERFORMS ONBOARD COMPUTATIONS
- EXECUTES RECEIVED/STORED COMMANDS
- PROVIDES CLOCKS FOR SCIENCE & ENGINEERING

DATA HANDLING FUNCTIONS:
- COLLECTS, DIGITIZES, MULTIPLEXES, SYNCHRONIZES, AND FORMATS ENGINEERING PACKETS/TRANSFER FRAMES
- ROUTES, STORES, AND PLAYS BACK SCI. & ENG. DATA
- PROVIDES COMMAND VERIFICATION
- PROVIDES MEMORY DUMP CAPABILITY

REQUIREMENTS:

COMMAND RATES/FORMATS:
- COMMAND RATES OF 7.6125 TO 500 B/S (NOMINAL @ 125 B/S)
- HARDWIRED 2000 B/S FOR GSE MEMORY LOADS
- REAL TIME AND STORED COMMANDS
- COMPUES WITH COMMAND STANDARDS
- STORES 1500-16 BIT PAYLOAD COMMANDS
- MAXIMUM SEQUENCE > 144 HOURS

TELEMETRY RATES/MODES:'

10 B/S	REAL TIME	ENGINEERING
250 B/S	REAL TIME & RECORD	
2 KB/S	REAL TIME a RECORD	
8 KB/S	PLAYBACK	
16 KB/S	RECORD	SPACECRAFT-TOS
32 KB/S	PLAYBACK	
4 KS/S	REAL TIME & RECORD	S&E-1
8 KS/S		
16 KS/S		
32 KS/S		
42.7 KS/S	PLAYBACK	
85.3 KS/S		
40 KS/S	REAL TIME	S&E-2
64 KS/S		
80 KS/S		

ASSEMBLIES:

STANDARD CONTROLS PROCESSOR UNITS (SCP):
- 96 K WORDS OF RAM-LOADABLE VIA CDU/CIU, 20 K WORDS PROM
- 1750-A INSTRUCTION SET, MARCONI CPU, 16/32/48 BIT ARITHMETIC
- 16 LEVEL INTERRUPT SYSTEM
- SEU RESISTANT PROCESSOR AND PERIPHERALS
- CONTAINS ALL COMMAND & CONTROL SOFTWARE

CONTROLS INTERFACE UNIT (CIU)/INTERFACE EXTENDER (CIX):
- INPUT/OUTPUT CONTROL FOR SCP
- COMMAND DECODING AND ROUTING
- SINGLE BIT ERROR CORRECTION/DOUBLE BIT ERROR DETECTION
- CLOCK GENERATION
- DISCRETE/LOW LEVEL COMMAND INTERFACES

SIGNALS CONDITIONING UNIT (SCU):
- HIGH LEVEL COMMAND INTERFACES
- PRE-ARM AND ARM RELAYS FOR PYROTECHNICS
- THRUSTER FIRE SIGNALS

DIGITAL TAPE RECORDERS (DTR):
- NASA STANDARD 1.38 x 10^9 BITS
- 3 CROSSSTRAPPED EU TO 4 TUs

CROSS STRAP UNIT (XSU)
- TRANSFERS DATA TO TELECOMM,

ENGINEERING DATA FORMATTER (EDF):
- 32 K WORDS RAM, 22 K WORDS PROM
- 1750 A INSTRUCTION SET, MARCONI CPU
- 364 ANALOG, 256 DIGITAL INPUTS
- TIME CODE GENERATION FOR PDS/TRANSFER TO INSTRUMENTS
- SENDS SELECTED T/M DATA TO SCP FOR FAULT MONITORING

REDUNDANT CRYSTAL OSCILLATOR (RXO):
- PROVIDES BASIC SIC CLOCK @ 5.12 MHz
- 1 PART IN 10^8 STABILITY PER DAY

HERITAGE:
- PDS - GFE FROM JPL
- CPU/EDF . NEW DESIGN FOR MARS OBSERVEWLANDSAT
- CIU/DTR/RXO/SCU/XSU - DMSP/ATN

TELECOMMUNICATIONS

FUNCTIONS:

- RECEIVE X-BAND UPLINK, DEMODULATE COMMAND SUBCARRIER AND/OR RANGING SIGNAL
- GENERATE X-BAND DOWNLINK (COHERENT OR VIA USO)
- PHASE MODULATE DOWNLINK CARRIER FOR REAL TIME/DTR PLATBACK
- PROVIDE MODULATION FOR DIFFERENCED ONE-WAY AND TWO-WAY TURNAROUND RANGING
- TRANSMIT TELEMETRY/RECEIVE EMERGENCY UPLINKED COMMANDS DURING EMERGENCY/SAFE MODES
- ACCOMMODATE Ka-BAND BEACON ENGINEERING DEMONSTRATION
- TRANSMIT S/C-TOS PLAYBACK DATA

UPLINK REQUIREMENTS:

FREQUENCY: 71457190 MHz

G/T IN dB/K
- > -17.5 OUTER CRUISE/MAPPING
- > -28.2 EMERGENCY MODE

DOWNLINK REQUIREMENTS:

FREQUENCY: 8400-8450 MHz X-BAND
33.6 GHz Ka-BAND

X-BAND EIRP IN dBm
- > 37.5 INITIAL ACQUISITION
- > 46.5 INNER CRUISE
- > 81.4 OUTER CRUISE/MAPPING
- > 46.0 10 BPS EMERGENCY MODE

K-BAND EIRP IN dBm
- > 50 (GOAL)

ASSEMBLIES:

ANTENNA:
- 1.5 M P-AXIS CASSEGRAIN X-BAND ANTENNA ASSEMBLY (INCLUDES Ka-BAND ANTENNA ON SUBREFLECTOR)
- TWO HEMISPHERICAL LGAs FOR EMERGENCY/BACKUP RECEIVE
- ONE HEMISPHERICAL LGAs FOR EMERGENCY/BACK- UP TRANSMIT

MARS OBSERVER TRANSPONDERS (MOT):
- COHERENT TRANSLATION OF RECEIVED X-BAND CARRIER TO X-BAND TRANSMIT FREQUENCY
- DEMODULATE COMMAND/RANGING FROM RECEIVED CARRIER
- MODULATE TELEMETRY AND RANGING ONTO TRANSMITTED CARRIER

COMMAND DETECTOR UNITS (CDU):
- DEMODULATE THE BCPHASE MODULATED SUB- CARRIER
- COMMANDS AND CLOCK SIGNALS SENT TO C&DH
- TELEMETRY AND STATUS DATA ROUTED TO EDF

RF POWER AMPLIFIERS (RPA):
- TWTs OPERATE AT 44 W (RF)

ULTRA STABLE OSCILLATOR (USO):
- USED FOR RADIO SCIENCE: GENERATES DOWNLINK FREOUENCY REFERENCE

HERITAGE:
- TRANSPONDER - MOTOROLA - DERIVED FROM NASA STD & MAGELLAN
- CDU -NEW DESIGN - GFP
- USO - PAYLOAD GFP
- RPA - VARIAN - DERIVED FORM HERITAGE DESIGN
- LGA, HGA - DERIVED FROM HERITAGE DESIGNS
- RF COMPONENTS - DSCS III, DBS, SATCOM

JPL ATTITUDE & ARTICULATION CONTROL

FUNCTIONS:

3 AXIS SPACECRAFT ATTITUDE CONTROL

- INITIAL ACQUISITION
- CRUISE/DRIFT "CONTROLLED ROLL"
- MANEUVERS
- MAPPING PHASE: AUTONOMOUS NADIR POINTING
- SAFE MODE

SPACECRAFT ATTITUDE DETERMINATION

HIGH GAIN ANTENNA POINTING

SOLAR ARRAY POINTING

PROVIDE TELEMETRY DATA FOR ATTITUDE RECONSTRUCTION

REQUIREMENTS:

POINTING ACCURACY (/AXIS, 3σ):

NADIR MOUNTED

CONTROL: <10 MRAD

KNOWLEDGE: <3 MRAD

BOOM MOUNTED

CONTROUKNOWLEDGE: <25 MRAD

HGA CONTROL: <8.7 MRAD

HGA KNOWLEDGE: <3 MRAD

POINTING STABILITY (3σ):

OVER 5 SEC: (0.5 MRAD R+P

<1 .O MRAD Y

OVER 12 SEC: <3 MRAD/AXIS

HGA OVER 300 SEC: <3 MRAD

MANEUVER ACCURACY (3σ):

SIDE VELOCITY ERROR:

0.01 m/s FIXED

25 MRAD PROPORTIONAL

MAGNITUDE ERROR:

0.05 m/s FIXED

2% PROPORTIONAL

ASSEMBLIES:

INERTIAL MEASUREMENT UNIT (IMU):

MEASURES BODY RATES AND ACCELERATIONS

UTILIZE 3 DRIRU II GYROS, 4 ACCELEROMETERS

MARS HORIZON SENSOR ASSEMBLY (MHSA):

MODIFIED BARNES EARTH SENSOR

4 QUADRANTS

OPERATES IN CO_2 WAVELENGTH

OPERABLE OVER 350 TO 370 KM ORBIT ALTITUDE

4 PI STERADIAN SUN SENSOR (4ΠSS):

MEASURED SUN ANGLE AT 4.4 MR ACCURACY

5 REDUNDANT SENSOR DETECTORS

CELESTIAL SENSOR ASSEMBLY (CSA):

INTERNALLY REDUNDANT

STAR MAPPER FOR INERTIAL ATTITUDE REFERENCE

REACTION WHEEL ASSEMBLY (RWA):

4 BRUSHLESS DC MOTORS AND DRIVE FLYWHEELS FOR ATTITUDE CONTROL TORQUING

AUTONOMOUS UNLOADING: PROGRAMMABLE SET POINTS

FLIGHT SOFTWARE:

RESIDENT IN C&DH SCP

HERITAGE:

IMU/RWA: DMSP/ATN

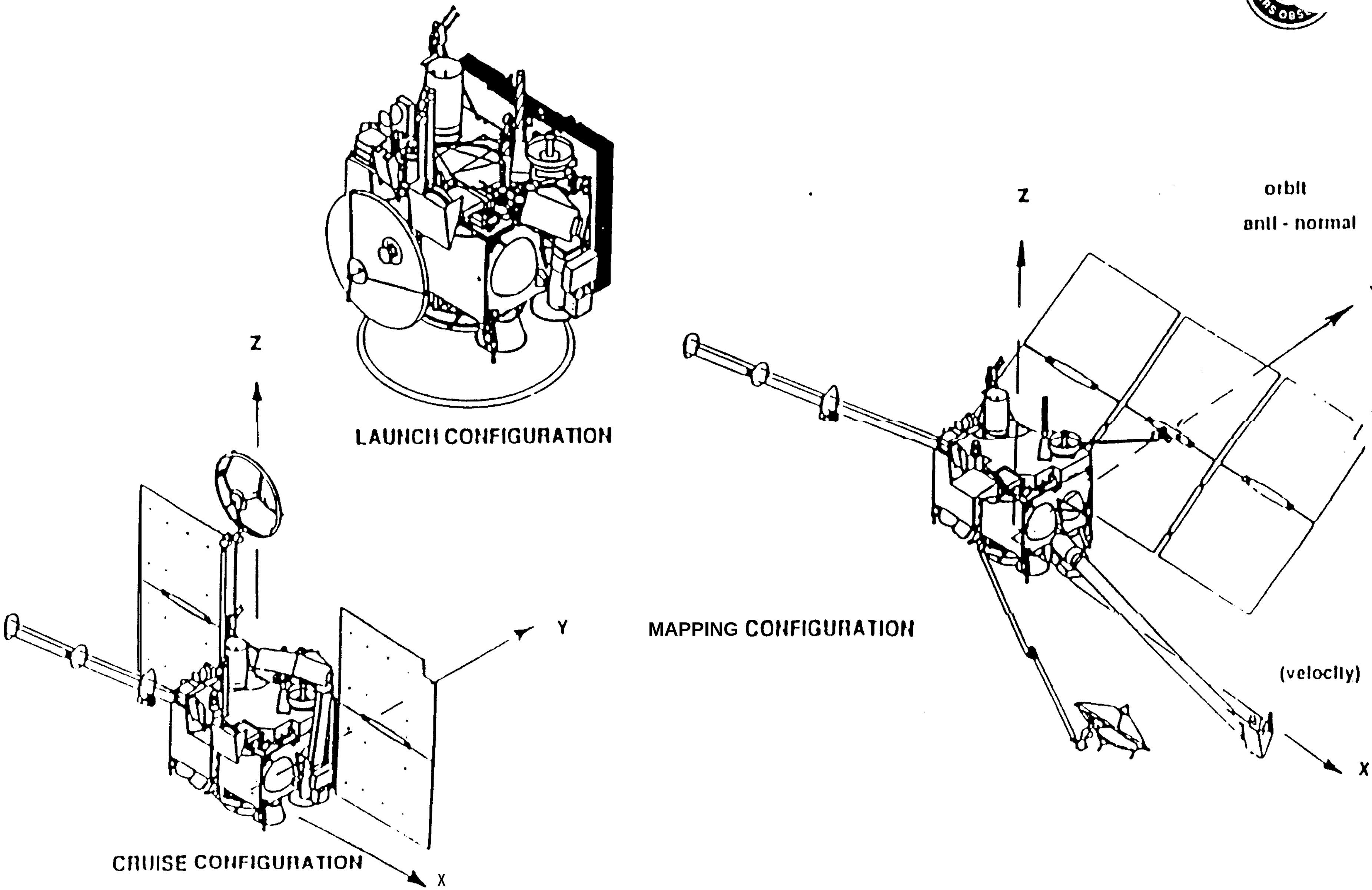
CSA: DMSP

SSA: NTS/SAGE/HCCM

MHSA: DMSPIATN, MODIFIED OPTICS AND PREAMP

SOFTWARE: DMSPIATN, NEW

SPACECRAFT C(IFIGURATIONS



MISSION CRITICAL SINGLE FAILURE POINT POLICY

- NO SINGLE FAILURE SHALL CAUSE:
  - PERMANENT LOSS OF DATA FROM MORE THAN ONE INSTRUMENT
  - FAILURE TO ACHIEVE & MAINTAIN MAPPING ORBIT
  - LOSS OF POINTING CONTROL
  - LOSS OF ATTITUDE RECONSTRUCTION TELEMETRY DATA
  - FAILURE TO ACHIEVE QUARANTINE ORBIT
- IMPLEMENTED THROUGH BLOCK, FUNCTIONAL, AND ALTERNATE MODE REDUNDANCY
- BLANKET WAIVER FOR LOW RISK ITEMS
  - STRUCTURE, BOOMS, BOOM HINGE BEARINGS, CABLING
  - PROPELLANT/PRESSURANT TANKS, LINES
  - PASSIVE RF COMPONENTS, HGA, LGAT
  - THERMAL BLANKETS, HEAT SHIELDS
  - ACTUATOR BEARINGS & ENCODER DISK, CSA OPTICS
- SPECIFIC WAIVERS FOR REMAINING SFPs
  - HINGE ASSEMBLIES, DEPLOY DELAY ASSY, ROTARY WAVEGUIDES
  - PRESSURE REGULATOR, FILTERS
  - CIU TO RXO & IMU INTERFACES, IMU SPIN MOTOR DURING MOI
  - S/A TELEMETRY SHUNT

SPACECRAFT DESIGN HERITAGE - ELECTRONICS

ITEM	HERITAGE'	PROGRAM
TELECOMM		
TRANSPONDER	2C	MAGELLAN
POWER AMPLIFIER	2B	LANDSAT/DSCS III
HGA	2C	STC/DBS/LANDSAT
LGA	3	
C&DH		
COMPUTER	3	
I/O UNITS	2C	DMSP/ATN
TELEMETRY PROCESSOR	3	
CLOCK	2B	DMSP/ATN
GIMBAL DRIVE ELECTRONICS	2C	SATCOM/GSTAR
TAPE RECORDERS	2A	ATN
ATTITUDE CONTROL		
INERTIAL MEASUREMENT UNIT	2A	DMSP/ATN
HORIZON SENSOR	2B	DMSP/ATN
CELESTIAL SENSOR	2A	DMSP
SUN SENSOR	2B	NTS/SAGE/HCMM
REACTION WHEELS	2B	DMSP/ATN
POWER		
ELECTRONICS	2C	DMSP/GSTAR
SOLAR ARRAY	2C	SATCOM
BATTERIES	2C	DMSP
SOFTWARE		
COMMAND & CONTROL	2C	DMSP/ATN
TELEMETRY	3	

SPACECRAFT DESIGN HERITAGE - MECHANICAL

ITEM	HERITAGE'	PROGRAM
PROPULSION		
490 N BI-PROP THRUSTERS	1/2A	IABS
22 N BI PROP THRUSTERS	2A	INTELSAT VI
BI-PROP TANKS	3	
PRESSURANT TANK	2A	SATCOM
4.4 N MONO-PROP THRUSTERS	2A	SATCOM
0.9 N MONO-PROP THRUSTERS	2A	SATCOM/VOYAGER
MONO-PROP TANKS	2A	SATCOM
MECHANISMS		
DEPLOYMENT	2C	SATCOM
CANISTER BOOMS	3	
SOLAR ARRAY & HGA BOOM	3	
GIMBAL DRIVES	2C	SATCOM
STRUCTURE		
PRIMARY	2C	SATCOM
SECONDARY	3	

'KEY: 1 -HARDWARE HERITAGE
- 2A - DESIGN HERITAGE, NO CHANGE
- 2B - DESIGN HERITAGE, MINOR MOD
- 2C - DESIGN HERITAGE, MAJOR MOD
- 3 - NEW DESIGN

JPL

POWER SUBSYSTEM

FUNCTIONS:

SUPPLY, CONTROL, CONVERT, REGULATE, DISTRIBUTE ALL ELECTRICAL POWER REQUIRED BY SPACECRAFT BUS AND PAYLOAD DURING ALL MISSION PHASES

CONTROL BATTERY CHARGE/DISCHARGE

PROVIDE COMMANDS FOR GROUND CONTROL

PROVIDE TELEMETRY FOR PERFORMANCE AND FAULT MONITORING

CONDITION AND DISTRIBUTE EXTERNALLY PROVIDED POWER FOR BUS SUBSYSTEMS AND SCIENCE PAYLOAD DURING GROUND TEST, PRELAUNCH AND LAUNCH PHASES

PROVIDE OVER/UNDER VOLTAGE AND OVERCURRENT PROTECTION

POWER MANAGEMENT SOFTWARE IN C&DH

REQUIREMENTS:

28 V DC ± 2% REGULATED POWER
136.5 W CONTINUOUS POWER FOR PAYLOAD DURING MAPPING
158 W PAYLOAD PEAK POWER
RIPPLE: 170 mV (p-p)
MAXIMUM TRANSIENT: 7.5 A
MAXIMUM RATE OF CURRENT RISE: 100 mA/μs
MINIMUM UNDERVOLTAGE: 15 V
MAXIMUM OVERVOLTAGE: 38 V
FUSES: 160

ASSEMBLIES:

SOLAR ARRAY:
- TWO-AXIS GIMBALED DRIVE, SINGLE PANEL 2.3 x 1.8 M
- 6 PANELS, PARTITIONED DEPLOYMENT
- REDUNDANT DIODE ISOLATION BETWEEN CIRCUITS
- 1400 W @ 1.4 AU, 1150 W @ 1.7 AU

BATTERIES:
- TWO NI-Cd 42 A-HR, 17 CELLS EACH
- 0 TO 5°C TEMPERATURE RANGE
- POWERS PYRO FUNCTIONS DIRECTLY

PARTIAL SHUNT ASSEMBLY (PSA):
- REGULATES SOLAR ARRAY OUTPUT
- CONTROLS via ELECTRICAL MISMATCHING OF SOLAR ARRAY CIRCUITS

POWER SUPPLY ELECTRONICS (PSE):
- CONTROL PSA
- BOOST BATTERY VOLTAGE DURING ECLIPSE
- CONTROL BATTERY CHARGER OPERATIONS
- FUSED OUTPUTS

BATTERY CHARGER ASSEMBLY (BCA)
- 4 CHARGE RATES: 0.85, 10, 12.5, 15 A
- 16 V/T TAPER MODES

PYROTECHNIC FUNCTIONS:
- SOLAR ARRAY RESTRAINT CABLE CUTTERS
- HGA DEPLOYMENT ASSEMBLY
- MAG/GRS BOOM DEPLOYMENTS
- PROPULSION PYRO VALVES
- V-BAND

HERITAGE:

SOLAR ARRAY: SATCOM-K
ELECTRONICS: DMSP, GSTAR
BATTERY: NASA STANDARD SPECIFICATION

JPL

PROPULSION SUBSYSTEM

FUNCTIONS:

PRODUCE VELOCITY CHANGES FOR:

- TRAJECTORY CORRECTION MANEUVERS (TCM)
- MARS ORBIT INSERTION (MOI)
- ORBIT TRIM MANEUVERS (OTM)
- QUARANTINE ORBIT RAISE MANEUVER

PRODUCE TORQUE FOR:

- THRUST VECTOR CONTROL DURING THRUSTING MANEUVERS
- REACTION WHEEL MOMENTUM UNLOADING
- BACKUP SPACECRAFT SLEWING

REQUIREMENTS:

BIPROP

TOTAL A',': 2.7 KM/SEC

CAPACITY: 1364 KG

ISP 490 N: 308 SEC(MIN)

22 N: 280 SEC

PULSE CAPABILITY (MIN):

490 N: 18,000

22 N: 66,450

MINIMUM IMPULSE BIT:

490 N: 12.25 N-S

22 N: 0.057 N-S

MONOPROP

CAPACITY: 84 KG

ISP -SS @ MAX P (MIN):

4.5 N: 220 SEC

0.9 N: 225 SEC

PULSE CAPABILITY (MIN):

4.5 N: 73,150

0.9 N: 410,000

ASSEMBLIES:

BIPROPELLANT:

ULTILIZES TWO 1.07 M TITANIUM TANKS FOR MMH (FUEL) AND N_2O_4 (OXIDIZER) LOCATED ON CENTER CYLINDER

BLOWDOWN FOR TCMs, REGULATED 255 PSI FOR MOI

FOUR 490 N AND FOUR 22 N ENGINES

ENGINE HALF SYSTEMS USED FOR REDUNDANCY MANAGEMENT

PARALLEL AND SERIES COMPONENTS ARRANGED TO PRECLUDE SINGLE FAILURE POINTS

PRESSURANT EQUIPMENT:

HELIUM PRESSURANT

SINGLE (0.66 M) TANK, SERIES REDUNDANT REGULATOR

GRAPHITE COMPOSITE OVERWRAP STAINLESS STEEL

MONOPROPELLANT EQUIPMENT:

TWO 0.48 M TITANIUM TANKS

EIGHT 4.5 N AND FOUR 0.9 N CATALYTIC REAs

BLOWDOWN OPERATION

HERITAGE:

BIPROPELLANT:

TANK: NEW

THRUSTERS: MARQUARDT, IABS

PRESSURANT:

TANK: GE/ASD SATCOM, SERIES 5000

REGULATOR: IABS

MONOPROPELLANT: GE/ASD SATCOM, SERIES 3000/4000/5000

JPL

STRUCTURE SUBSYSTEM

FUNCTIONS:

- PROVIDE FOR STRUCTURAL MOUNTING OF ALL ASSEMBLIES
- PROVIDE STABLE MECHANICAL INTERFACE FOR ALL SENSORS
- ENSURE STRUCTURAL INTEGRITY FOR ALL MISSION PHASES
- PROVIDE BASIC SYSTEM ALIGNMENTS
- PROVIDE FOR UNOBSTRUCTED SENSOR FOV
- PROVIDE CENTER OF MASS CONTROL

REQUIREMENTS:

- 2500 KG TOTAL INJECTED MASS
- COMPATIBLE WITH TITAN III
- ACCOMMODATE 166 KG OF GFP
- OPTICAL ALIGNMENT OF REFERENCE MIRRORS TO PRIMARY **MIRROR**
- PIN AND BOLT INSTRUMENT ATTACHMENT

ASSEMBLIES:

PRIMARY **STRUCTURE:**

- **MAGNESIUM ALLOY CENTER CYLINDER FOR PRIMARY LOAD PATH**
- **2.1 x 1.5 x 1.0 M RECTANGULAR MODULE**
- **EIGHT MODULAR ALUMINUM HONEYCOMB EQUIPMENT PANELS**

SECONDARY **STRUCTURE:**

- **THRUSTER SUPPORT BRACKETRY AND HEAT SHEILD**
- **TANK SUPPORTS**
- **S/A INBOARD SUPPORT**
- **CSA BRACKET AND SUN SHEILD SUPPORT**
- **PURGE LINE, HARNESS, & THERMAL SUPPORTS**
- **LGA SUPPORT BRACKETRY**

TOS ADAPTER

- MONOCOQUE **WITH SKINS AND STRINGERS**
- **"V" BAND SEPARATION CLAMP ASSEMBLY WITH RETENTION SPRINGS**
- **SPRING-ASSISTED SEPARATION**
- BOUND **120 BOLT INTERFACE WITH TOS**

HERITAGE:

PRIMARY STRUCTURE: **SATCOM K**
SECONDARY STRUCTURE & ADAPTER: NEW

MECHANISMS

FUNCTIONS:

PROVIDES DEPLOYMENTS FOR:
- INSTRUMENT BOOMS (MAG/ER, GRS)
- HGA
- SOLAR ARRAY AND PANELS

PROVIDES ARTICULATION FOR:
- SOLAR ARRAY
- HGA

REQUIREMENTS:

TITAN III COMPATIBLE
MULTIPLE POSITIONS FOR GRS CALIBRATIONS DURING CRUISE AND MAPPING
SURVIVE MOI

ASSEMBLIES:

SOLAR ARRAY MECHANISMS:
6 PANELS AND 2 BOOM SECTIONS (2.8 M) CONSTRAINED FOR LAUNCH
PARTIAL DEPLOYMENT DURING CRUISE AND TRANSITION
DEPLOYMENT vla REDUNDANT CUTTING OF RESTRAINT CABLES
BOOMS/PANEL DEPLOYMENT CONTROLLED vla HINGE SPRINGS AND PASSIVE DAMPERS
DRIVE MECHANISM IS P-AXIS GIMBAL
(MINIMIZES SWEPT VOLUME WHILE TRACKING THE SUN)
- HARMONIC DRIVE WITH REDUNDANT MOTOR WINDINGS
- SHAFT POSITIONS DETERMINED vla OPTICAL ENCODERS

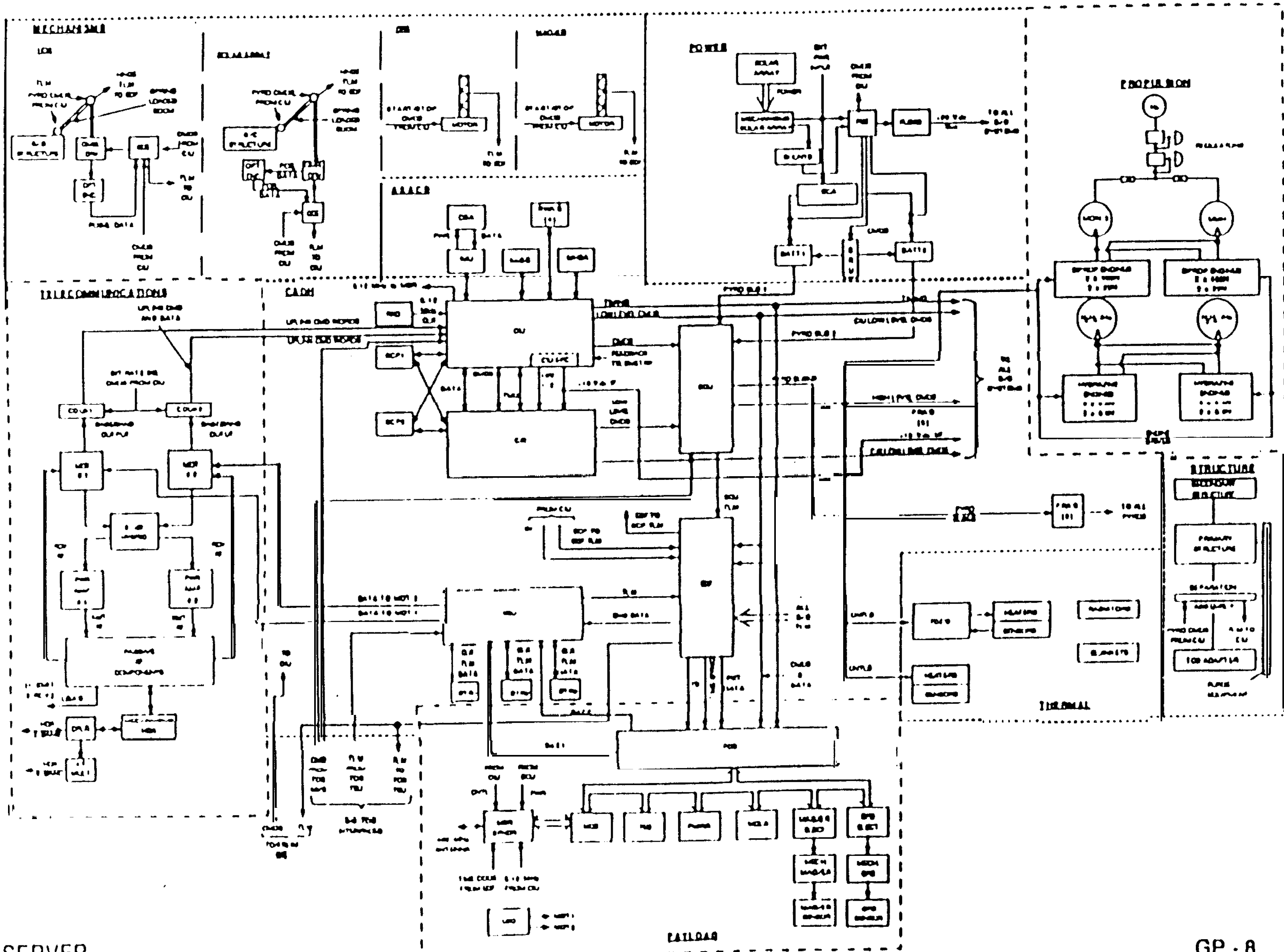
HIGH GAIN ANTENNA:
STOWAGE DESIGNED TO ACCOMMODATE TITAN ENVELOPE
CABLE-AND-SHEAR TIES USED FOR RESTRAINT
DEPLOYMENT vla REDUNDANT CUTTING OF RESTRAINT CABLES
5.3 M BOOM OF TUBULAR GRAPHITE EPOXY COMPOSITE LAMINATE
PARTIAL DEPLOYMENT FOR CRUISE- BODY FIXED POINTING MAINTAINED
2-AXIS GIMBAL DRIVE MECHANISM (SAME AS S/A) FOR MAPPING

INSTRUMENT BOOM:
2 MOTOR DRIVEN CANNISTER SYSTEMS FOR MULTIPLE DEPLOYMENT POSITIONS
2 EXTENSIONS FOR MAGNETICS AND 3 EXTENSIONS FOR GAMMA RAY BACKGROUND MASKING
6 M TOTAL EXTENSION
SEQUENCED DEPLOYMENT OF BOOMS FOR SPF PROTECTION
MIDSPAN PICKUP FOR CEA, ER, INBOARD MAG, AND HARNESS
BOOM RETRACTION LIMITED BY NO CABLE RETRACTION

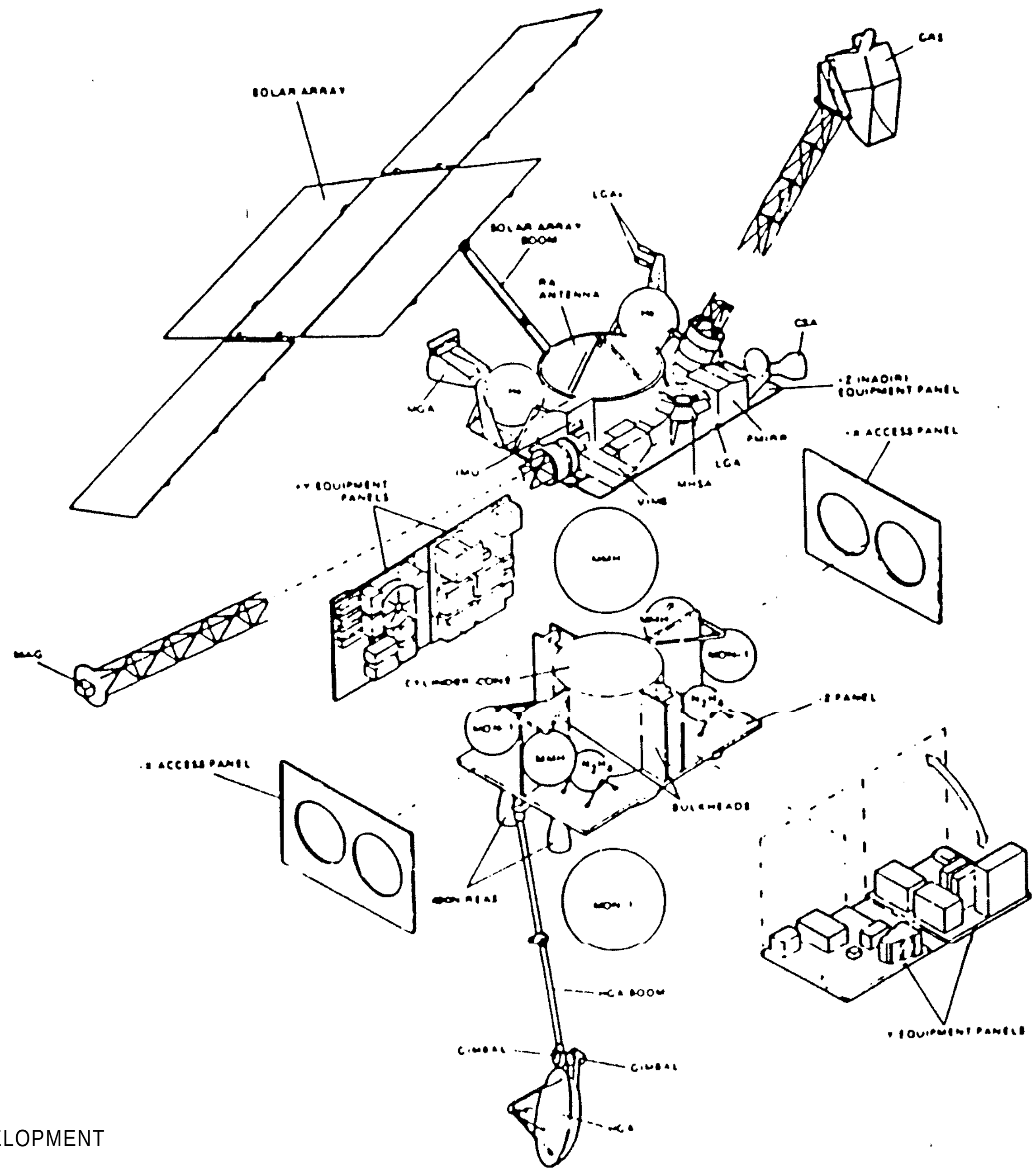
HERITAGE:

DEPLOYMENT MECHANISMS: SIMILAR TO SATCOM & DMSP
BOOMS: NEW

SYSTEM BLOCK DIAGRAM



JPL SPACECRAFT EXPLODED VIEW - INITIAL



THERMAL CONTROL

FUNCTIONS:

MAINTAIN SPACECRAFT EQUIPMENT TEMPERATURES WITHIN OPERATING AND NON-OPERATING LIMITS

MINIMIZE THERMALLY INDUCED DISTORTIONS OF DEPLOYED BOOMS, PAYLOAD MOUNTING AREAS, AND SUPPORTS

ASSEMBLIES:

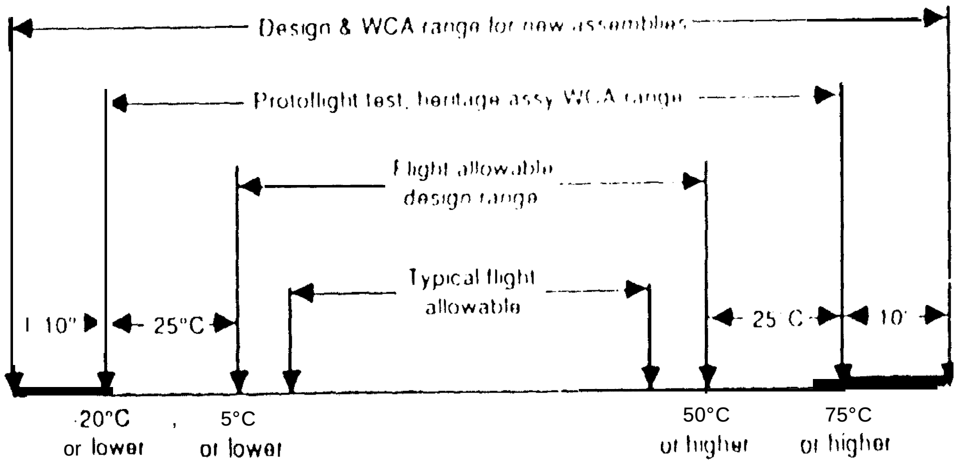
PASSIVE CONTROL ELEMENTS
- RADIATORS
- SHIELDS
- MULTILAYER BLANKETS
- PAINT
- TAPES
- HEAT SPREADERS

ACTIVE CONTROL ELEMENTS:
- HEATERS
- DUAL THERMAL CONTROL DEVICES (OTC)
- THERMOSTATS
- COMPUTER MONITORING AND COMMANDS

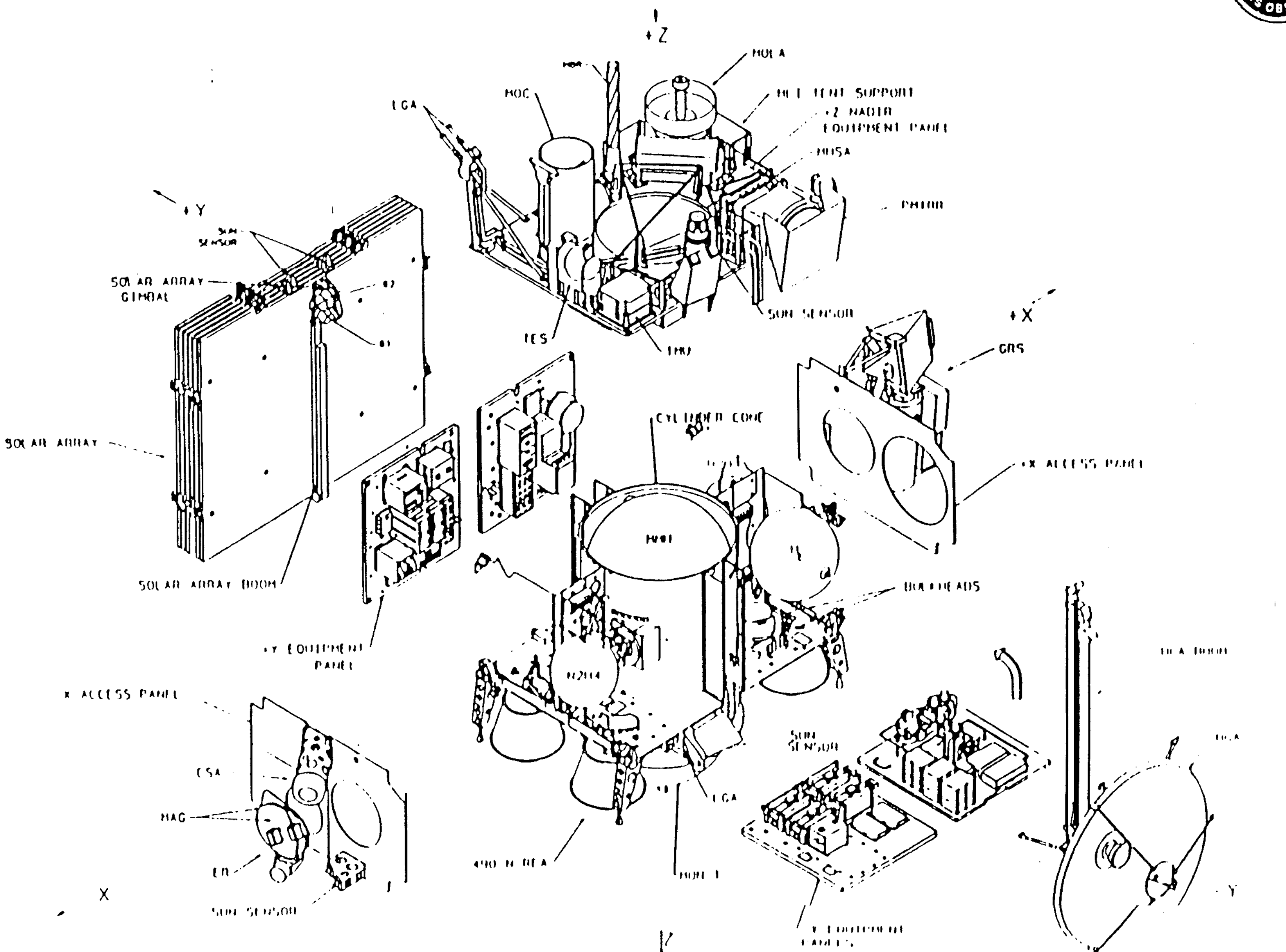
HERITAGE:

PASSIVE AND ACTIVE ELEMENTS: DMSP, SATCOM
COMPUTER MONITORING: NEW

REQUIREMENTS:



SPACECRAFT EXPLODED VIEW - FINAL



JPL

SUBSYSTEM DESIGN CHANGES

SUBSYSTEM	INITIAL PROPOSED	FINAL LAUNCHED
COMMAND & DATA HANDLING (C&DH)	CUSTOM LSI CPU 64K RAM, 256 PROM 1802 EDF 2 DTRs	1750 CPU 96K RAM, 20K PROM 1750 EDF 4 TUs, 3 EUs
TELECOMMUNICATIONS	NXT 6 LGA (2 R, 4 T) 1.0 M HGA 1 MGA	MOT 3 LGA (2R, 1 T) 1.5 M HGA
ATTITUDE & ARTICULATION CONTROL (AACS)	4 GAS BEARING GYROS 3 ACCELEROMETERS	3 DRY TUNED-ROTOR GYROS 4 ACCELEROMETERS
POWER	5 SOLAR PANELS 2 x 26.5 A-HR BATTERIES	6 SOLAR PANELS 2 x 42 A-HR BATTERIES
STRUCTURE, MECHANISMS & THERMAL CONTROL	STS ENVELOPE & LOADS RETRACTABLE LANYARD BOOM LOUVERS	TITAN III ENVELOPE & LOADS CANISTER BOOM
PROPULSION	6 BI-PROP TANKS 3 MONO-PROP TANKS MONO-PROP THRUSTERS: 4 x 22N 12 x .9N	2 BI-PROP TANKS 2 MONO-PROP TANKS MONO-PROP THRUSTERS: 8 x 4.5N 4 x .9N

JPL

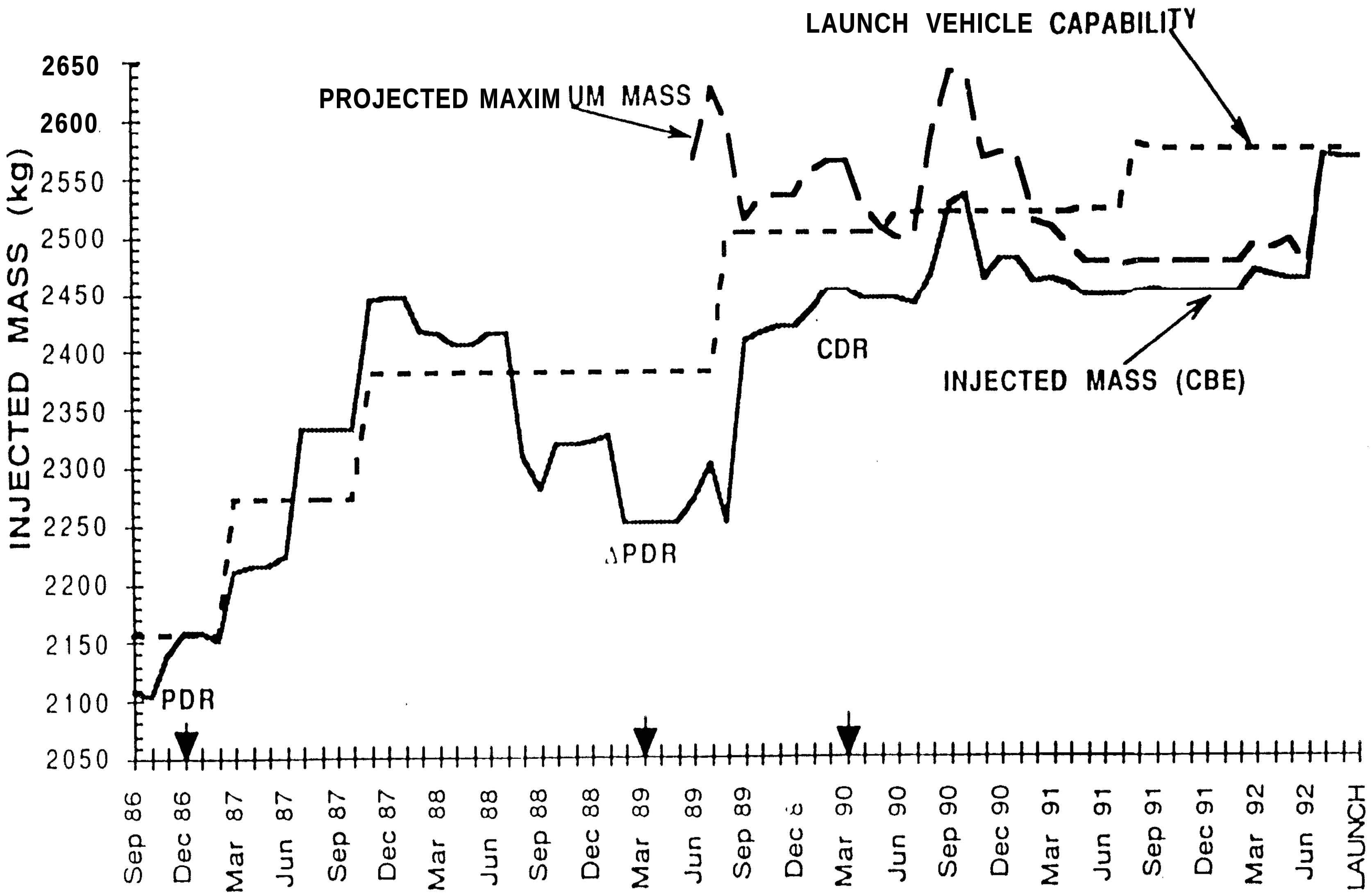
PAYLOAD ACCOMMODATION CHANGES

PAYLOAD ITEM	RFP	CONTRACT START	UM # 9	UM #16, 19, 20
INSTRUMENT COMPLEMENT	8 INSTRUMENTS	DELETE UVS AND UVP, ADD TES AND MOC	ADD ER	DELETE VIMS, REPLACE RA WITH MOLA, ADD MBR
MASS (KG)	108.5	120	168	166
POWER(W)	115	130	136.5 PULSE LOADS TO 158	42 A HR BATTERIES
MAX DATA RATE (KBPS)	32	64	85.3	SAME
TAPE STORAGE (BITS)	0.52×10^9	1.04×10^9	1.38×10^9	1.84×10^9
PAYLOAD COMMANDS	1024	1500	PDS BURST MODE	SAME
MAX CMD RATE (BPS)	32	32	500	SAME
SIMULTANEOUS CMDS	5	10	SAME	SAME
MAX SEQUENCE (HRS)	96	144	SAME	SAME
FOV	STRAWMAN	SELECTED	LARGER MOC FOV, TES DESIRED FOV	MBR FOV PMIRR DESIRED FOV
CONFIGURATION	STRAWMAN	SELECTED	SPLIT PACKAGE FOR GRS, MOC, AND PMIRR	LARGE BALLAST MASS
TOTAL VOLUME (M^3)	0.47	0.76	0.78	TBD
THERMAL	-20” TO +30 C	SAME	SOME EXCEPTIONS	CUSTOMIZED INTERFACES
PURGE	VIMS & PMIRR		TES & MOC	MOLA

MASS SUMMARY

ELEMENT	PROPOSED INITIAL,	FINAL LAUNCH
PAYLOAD	108.5	156.6
ATTITUDE CONTROL	48.4	57.8
POWER	130.5	203.7
PROPULSION	131.5	127.3
STRUCTURE	154.0	223.3
THERMAL CONTROL	33.0	36.5
C&DH	64.4	80.0
TELECOMMUNICATIONS (INCL GFP)	26.6	43.8
MECHANISMS	52.7	102.1
HARNESS	43.8	81 .0
BALANCE	-0-	12.4
MARGIN	33.6	-0-
SUBTOTAL DRY MASS	827.0	1124.5
PROPELLANTS & PRESSURANTS	1328.0	1440.5
TOTAL INJECTED MASS	2155.0 KG	2565.0 KG

MASS HISTORY

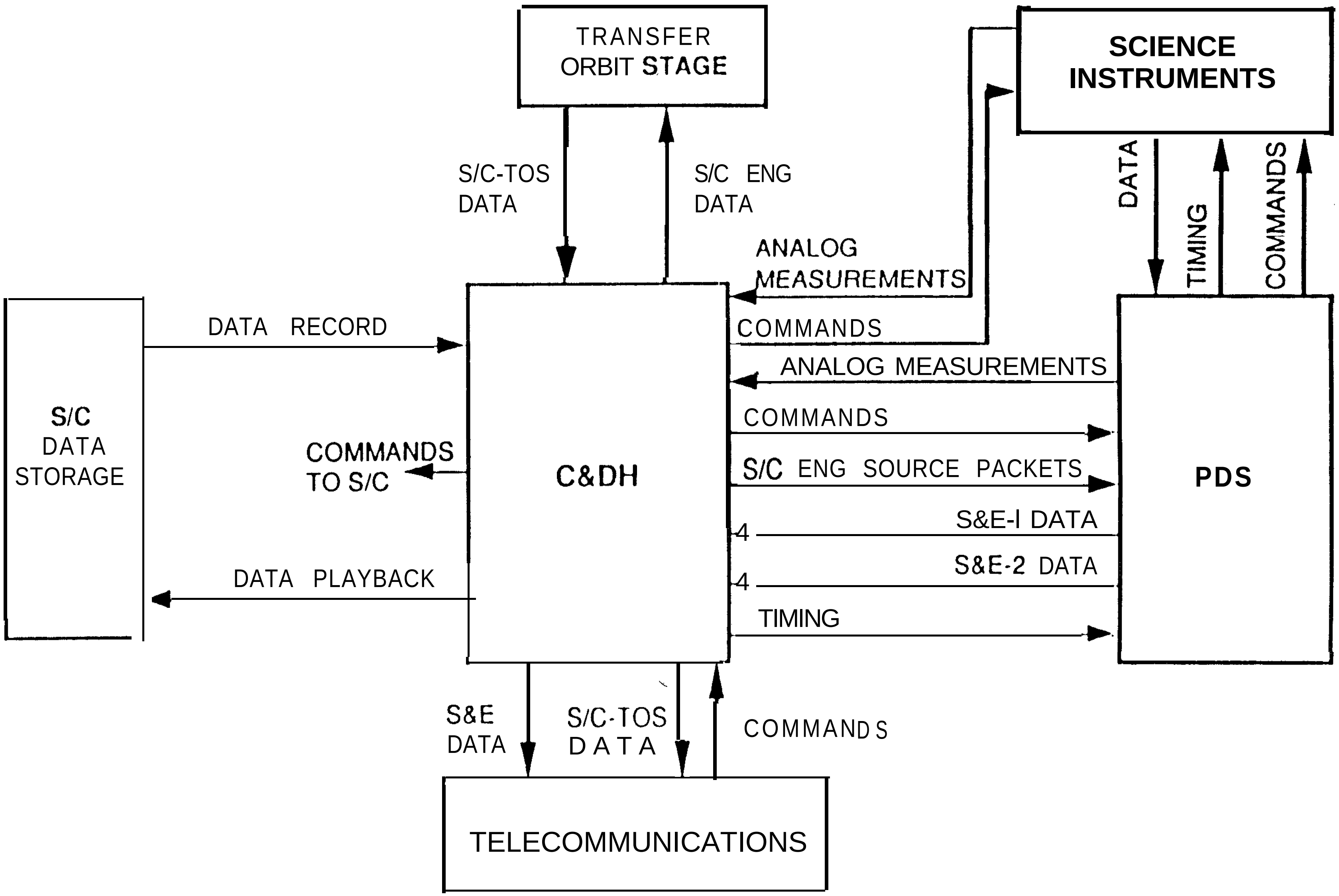


POWER SUMMARY

MAPPING @ APHELION WITH 1 TAPE RECORDING AND I TAPE IN PLAYBACK

ELEMENT	INITIAL PROPOSED	FINAL LAUNCH
PAYLOAD	109.0	130.4
TELECOMM	154.0	144.5
C&DH	73.6	73.5
AACS	63.7	51 .0
PROPULSION	0	13.4
POWER	4.2	6.2
MECHANISMS	0	22.6
THERMAL	43.0	162.3
NOMINAL LOAD	447.5	603.9
BATTERY CHARGING	252.5	451 .0
LOSSES	0	39.0
TOTAL POWER REQUIRED	700.0	1093.9
TOTAL POWER AVAILABLE	710.0 W	1147.0 W

COMMAND & DATA FLOW

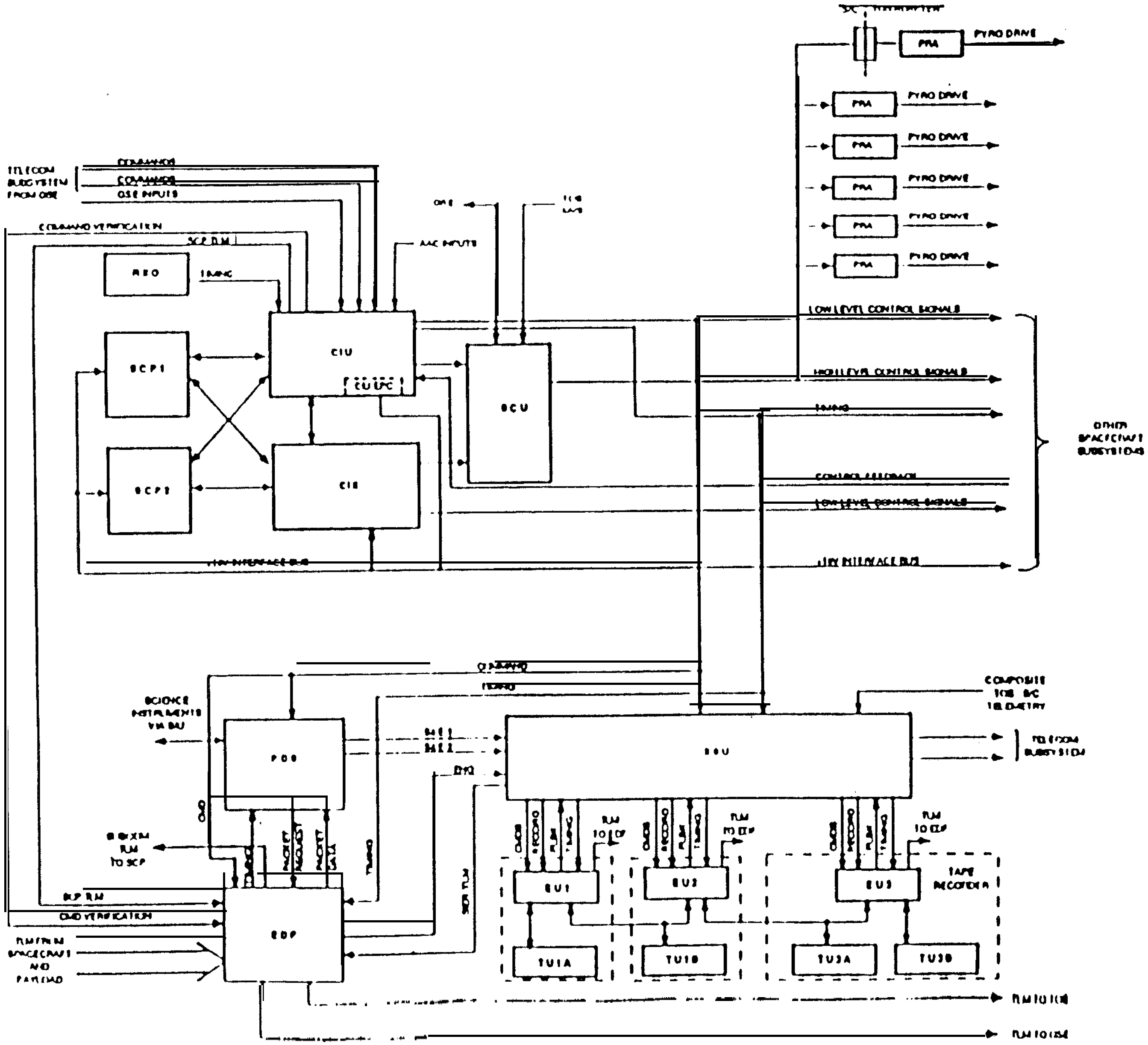


JPL COMMAND & DATA HANDLING SUBSYSTEM

- **DESIGN DRIVERS**
  - DATARATES
  - DATA STORAGE REQUIREMENTS
  - DATA STANDARDS
  - FAULT PROTECTION
- **CHARACTERISTICS**
  - CENTRAL COMPUTER FOR COMMAND AND CONTROL
  - MEMORY: 96K RAM, 20K PROM
  - UTILIZE GFP PDS AS INTERFACE TO INSTRUMENTS
  - DATA RATES: 10 TO 80K BPS
  - DATA STORAGE: 1.38×10^9 BITS, SIMULTANEOUS RECORD & PLAYBACK
  - CENTRAL CLOCK
- **DEVELOPMENT HIGHLIGHTS**
  - 1750 PROCESSOR REPLACED CUSTOM LSI HERITAGE CPU. COMMON DESIGN BETWEEN SCP & EDF. SHARED DEVELOPMENT WITH LANDSAT.
  - MEMORY ADDED TO ACCOMMODATE SOFTWARE GROWTH
  - SPARE TAPE TRANSPORT FLOWN TO IMPROVE RELIABILITY

JPL

C&DH BLOCK DIAGRAM



JPL

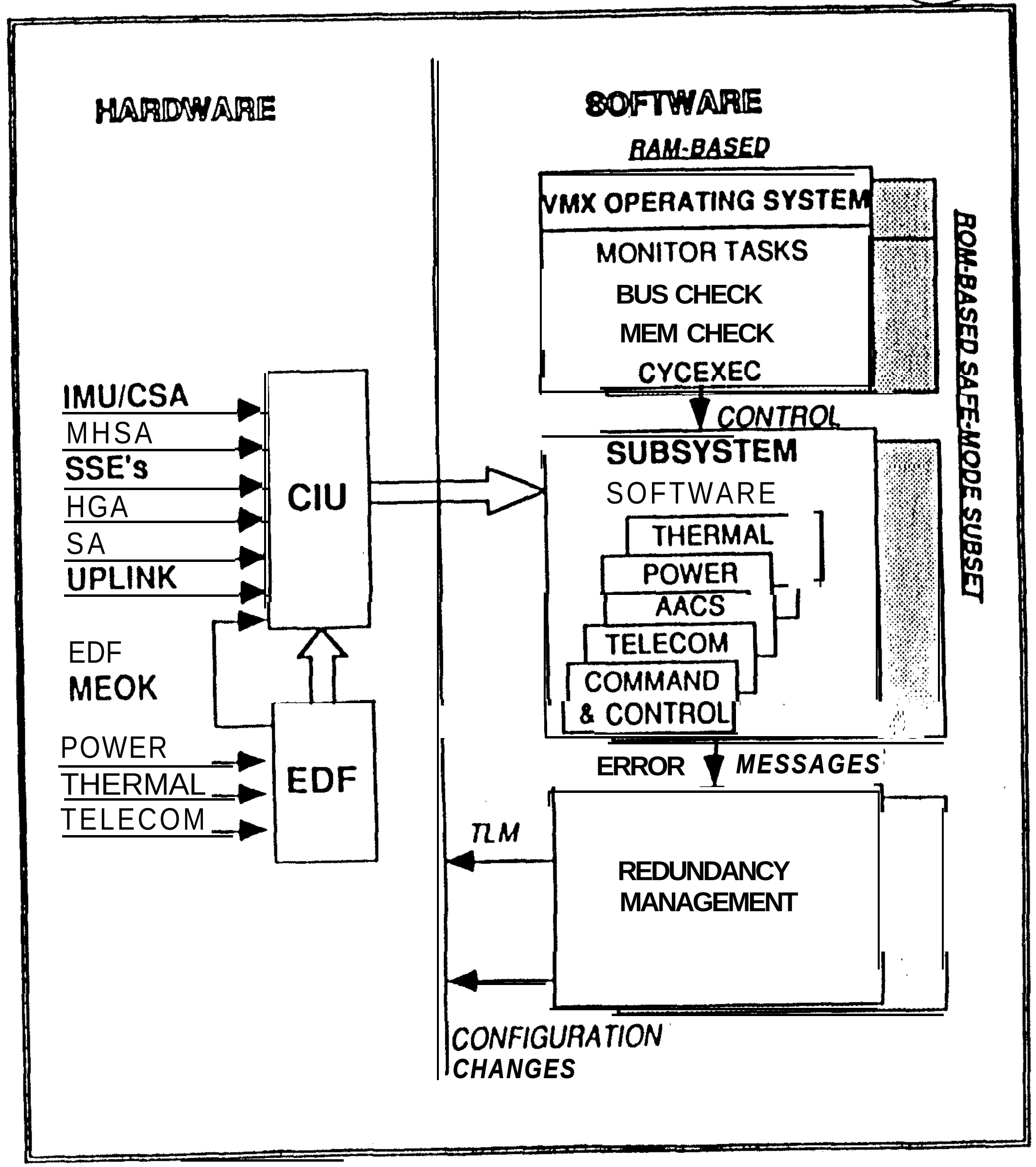
FLIGHT SOFTWARE

FUNCTIONS:

COMMANDING	• RECEPTION, VERIFICATION, STORAGE, DISTRIBUTION
TELEMETRY	• S/C DATA FROM EDF, SIC TIMING, ATTITUDE, CONFIGURATION DATA
AACS	• SENSORS, ALGORITHMS, COMMAND EFFECTORS
POWER	• BATTERY CHARGE, LOAD CONTROL, S/A POINTING
THERMAL	• S/C ELECTRONICS & TANK THERMAL MANAGEMENT
TELECOM	• CYCLIC ORBITAL OPS, HGA POINTING
FAULT PROTECTION	• REDUNDANCY MGMT AND ANOMALY MODE OPS

CHARACTERISTICS:

WRITTEN IN JOVIAL & SOME ASSEMBLY
VMX OPERATING SYSTEM
FITS IN 96K RAM, 20K PROM
HOT-SPARE CONCEPT • BOTH SCPs RUN SAME SOFTWARE; ONLY ONE IN CONTROL



FAULT PROTECTION APPROACH

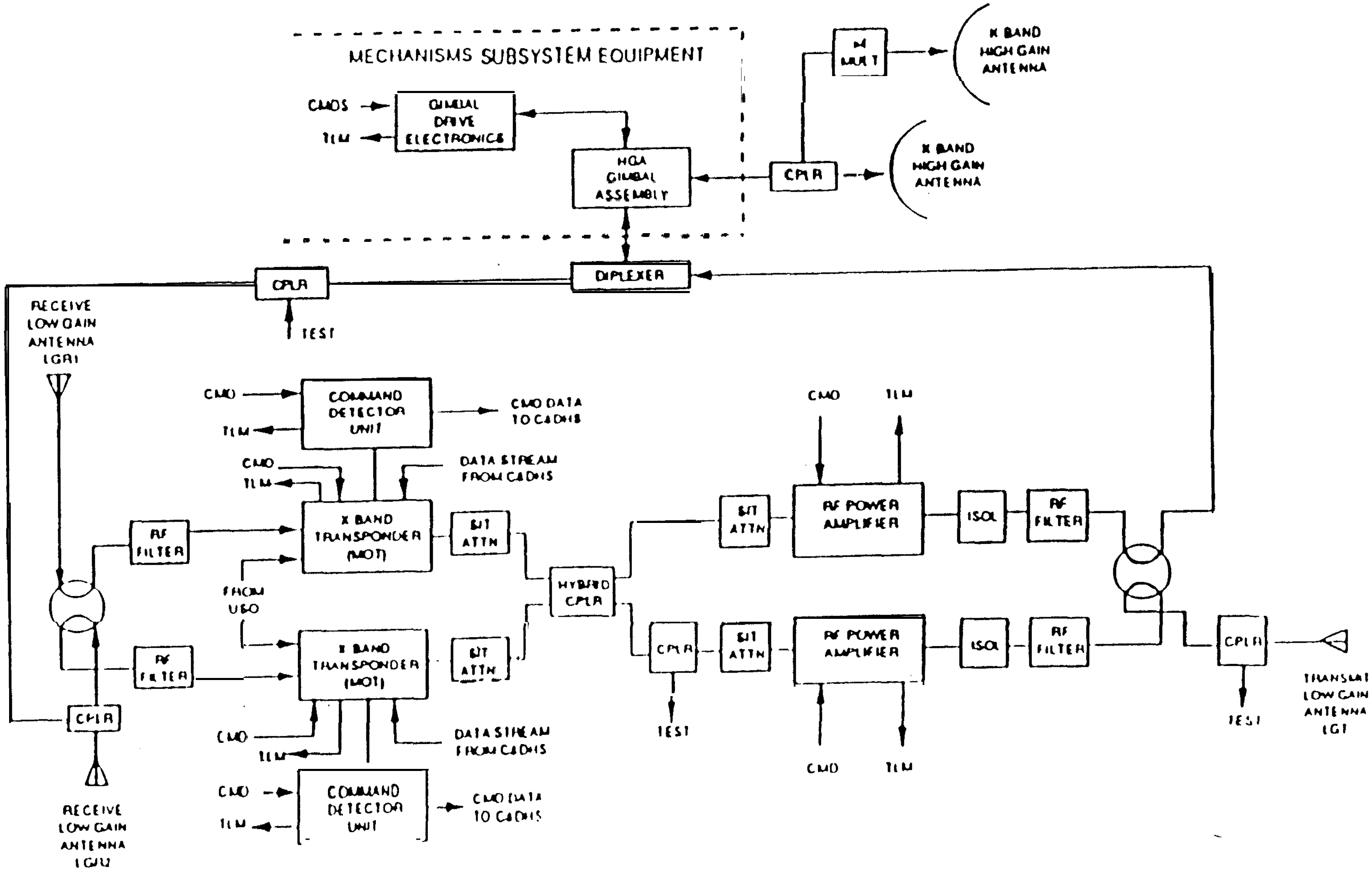
- BLOCK AND FUNCTIONAL REDUNDANCY PROVIDED TO ELIMINATE MISSION CRITICAL SINGLE FAILURE POINTS. PROJECT POLICY EXEMPTS LOW RISK DIFFICULT TO ELIMINATE SFPs. REMAINING SFPs ARE HIGHLY UNLIKELY. CONSERVATIVE DESIGN MARGINS REQUIRED TO REDUCE RISK.
- OPERATION IN DEGRADED MODES ALLOWED FOR MISSION PROTECTION
- FAULT MANAGEMENT APPROACH
  - IMMEDIATE RESPONSE - HARDWARE PROTECTION CIRCUITS (e.g. FUSES, OSCILLATORS, OVER/UNDER VOLTAGE CIRCUITRY)
  - INTERMEDIATE RESPONSE - FLIGHT SOFTWARE
    - » MONITORS OUTPUTS TO CONTROL SWITCHING FROM PRIMARY TO BACKUP OR FUNCTIONALLY REDUNDANT UNIT
    - » INVOKES SUCCESSIVE DEGRADED MODES IF NO RECOVERY - PRIORITY: 1) SAFE THE S/C, 2) SAFE THE PAYLOAD, 3) ASSURE COMMAND LINK, 4) PROVIDE TELEMETRY
      - EMERGENCY - PROTECTS **AGAINST LOSS OF** LINK
      - **CONTINGENCY - PROTECTS AGAINST EXCESSIVE POWER DRAIN**
      - **SAFE - PROTECTS AGAINST CATASTROPHIC POWER DRAIN OR HARDWARE LOSS**
  - **SLOWEST RESPONSE - GROUND OPERATIONS**
    - **» TREND AND FAILURE ANALYSIS**
    - **» LONG-TERM REDUNDANCY MANAGEMENT (e.g. DTR)**
    - **» RECOVERY AFTER AUTONOMOUS ACTION**

JPL

TELECOMMUNICATIONS SUBSYSTEM

- **DESIGN DRIVERS**
  - ALL X-BAND SYSTEM
  - DATARATES
  - COMMAND RATES
  - CRUISE AND MAPPING GEOMETRIES
- **CHARACTERISTICS**
  - TRANSPONDER: MOT
  - UTILIZE GFP CDU AND **USO**
  - POWER AMPLIFIER: 44W TWT
  - ANTENNAS: 2 DOF HGA (1.5 M), CGA (2 RCV, 1 TRAN)
- **DEVELOPMENT HIGHLIGHTS**
  - NXT DEVELOPMENT FAILED. MOT DERIVED FROM MAGELLAN TRANSPONDER
  - DESIGN SIMPLIFIED TO ELIMINATE ANTENNAS BUT ASSURE COVERAGE
  - **CONFORMAL** COATING OF TWT POWER SUPPLY ADDED AFTER SWITCH TO TITAN III
  - KA-BAND BEACON ADDED AS ENGINEERING DEMONSTRATION
  - DUAL SUBCARRIER ADDED FOR LOW DATA RATES
  - OPERATIONAL WORKAROUND DEVELOPED FOR **BOTH CDUs LOCKING UP**

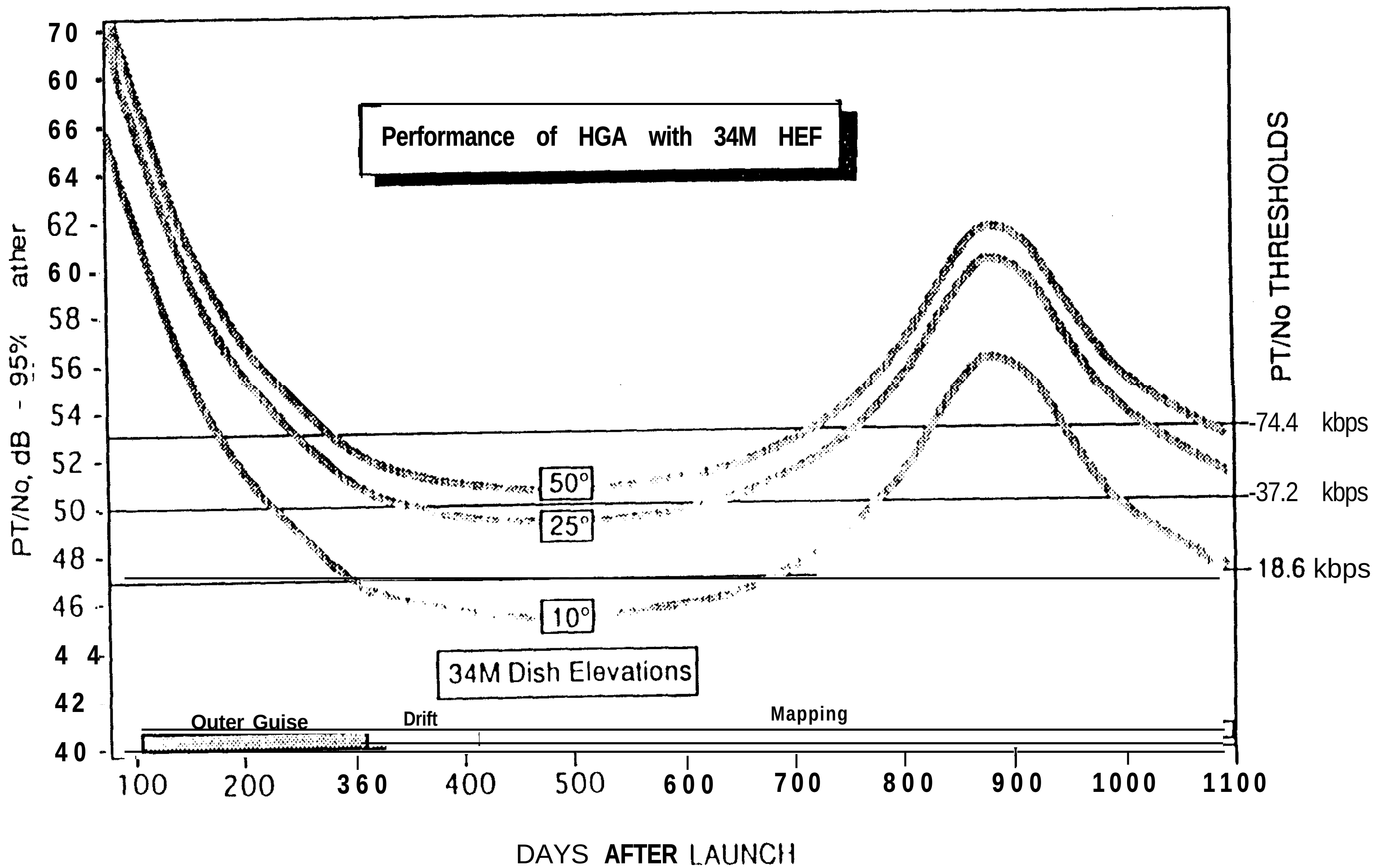
TELECOMMUNICATIONS BLOCK DIAGRAM



MARS OBSERVER
SPACECRAFT DEVELOPMENT

GP-22
4-23-93

COMMUNICATIONS MARGINS



ATTITUDE & AKTICULATION CONTROL SUBSYSTEM

. DESIGN DRIVERS

- MULTIPLE ‘FLIGHT MODES: LAUNCH, CRUISE, TCM, MOI, **MAPPING, OJ-M**
- BOOM INTERACTION
- AUTONOMOUS OPERATION
- MARS ENVIRONMENT

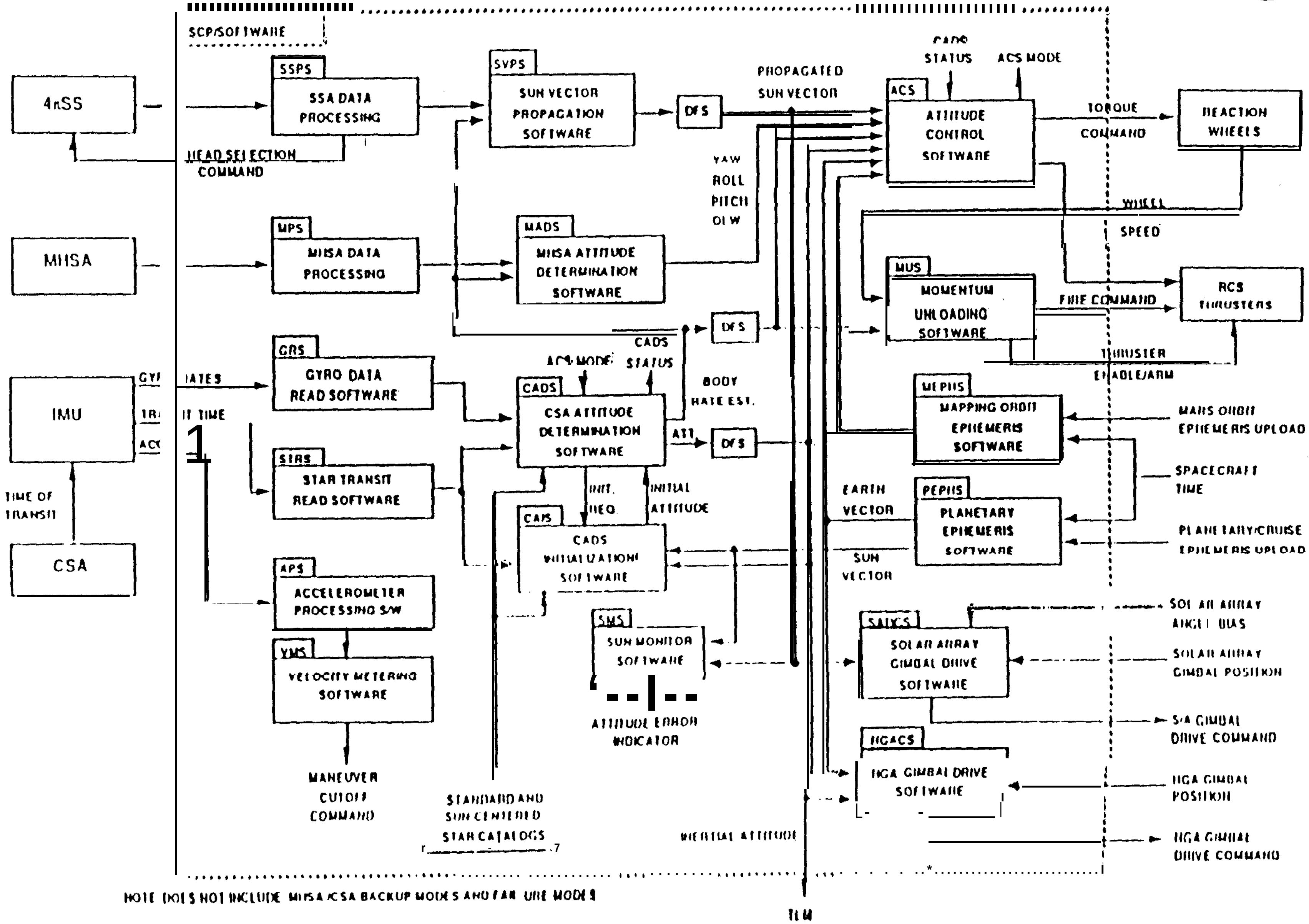
• CHARACTERISTICS

- 3 AXIS STABILIZED: 10 MRAD CONTROL, 3 MRAD KNOWLEDGE
- **FLIGHT SOFTWARE CONTROLLED**
- SENSORS: 4π SUN, CELESTIAL, MARS HORIZON, IMU (GYROS & ACCEL)
- ACTUATORS: THRUSTERS, 4 REACTION WHEELS

. DEVELOPMENT HIGHLIGHTS

- EARTH SENSOR MODIFIED FOR MARS ENVIRONMENT
- SDOF GAS-BEARING GYROS REPLACED BY 2DOF DRY TUNED-ROTOR GYROS
- EXTENSIVE ANALYSIS & PARAMETER SPECIFICATION TO MINIMIZE CONTROL INTERACTIONS
- TIGHT TOLERANCE REACTION WHEEL BEARINGS

AACS BLOCK DIAGRAM



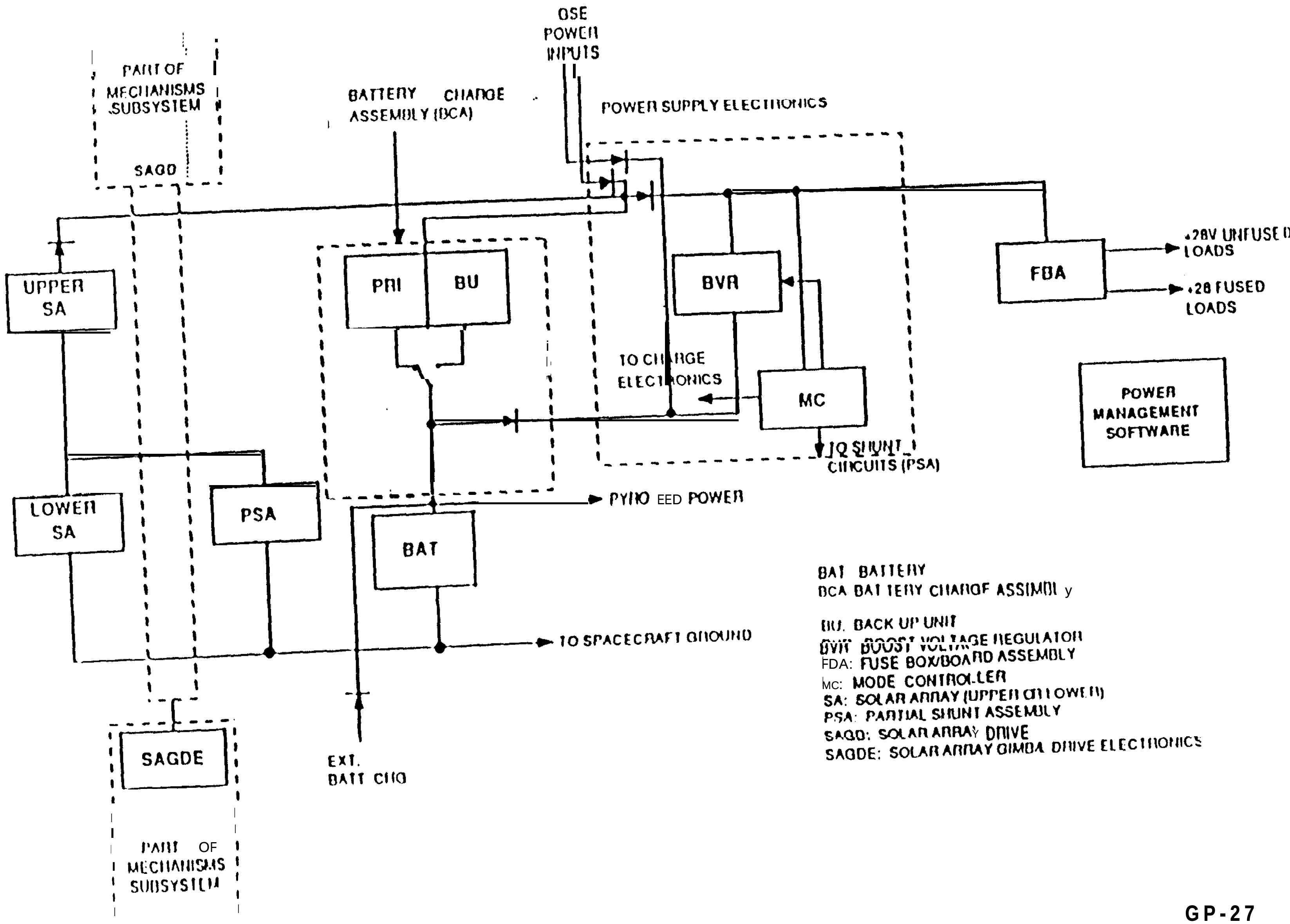
JPL

POWER SUBSYSTEM

- **DESIGN DRIVERS**
  - CHANGING SUN INTENSITY: 1.0 -> 1.7 AU
  - SUBSYSTEM AND PAYLOAD POWER REQUIREMENTS
  - BATTERY CHARGE/DISCHARGE CYCLES
- **CHARACTERISTICS**
  - REGULATED 28 V DC ± 2% BUS
  - 136.5 W CONTINUOUS FOR PAYLOAD DURING MAPPING
  - 2 DOF SOLAR ARRAY
  - 2 x 42 AMP-HR NI-CAD BATTERIES
- **DEVELOPMENT HIGHLIGHTS**
  - SLIP RING POWER TRANSFER REPLACED BY FLEX CABLES
  - S/A AND BATTERY SIZE GREW AS REQUIREMENTS GREW
  - STANDARD NI-CAD CELL DEVELOPMENT PROBLEMS LEAD TO DEVELOPMENT OF SUPER NI-CAD BATTERIES AS BACKUP
  - BATTERY RECONDITIONING UNIT DELETED DURING MASS REDUCTION RE-DESIGN
  - DESIGN ALLOWED INADVERTENT POWER TURN-ONS DURING TEST

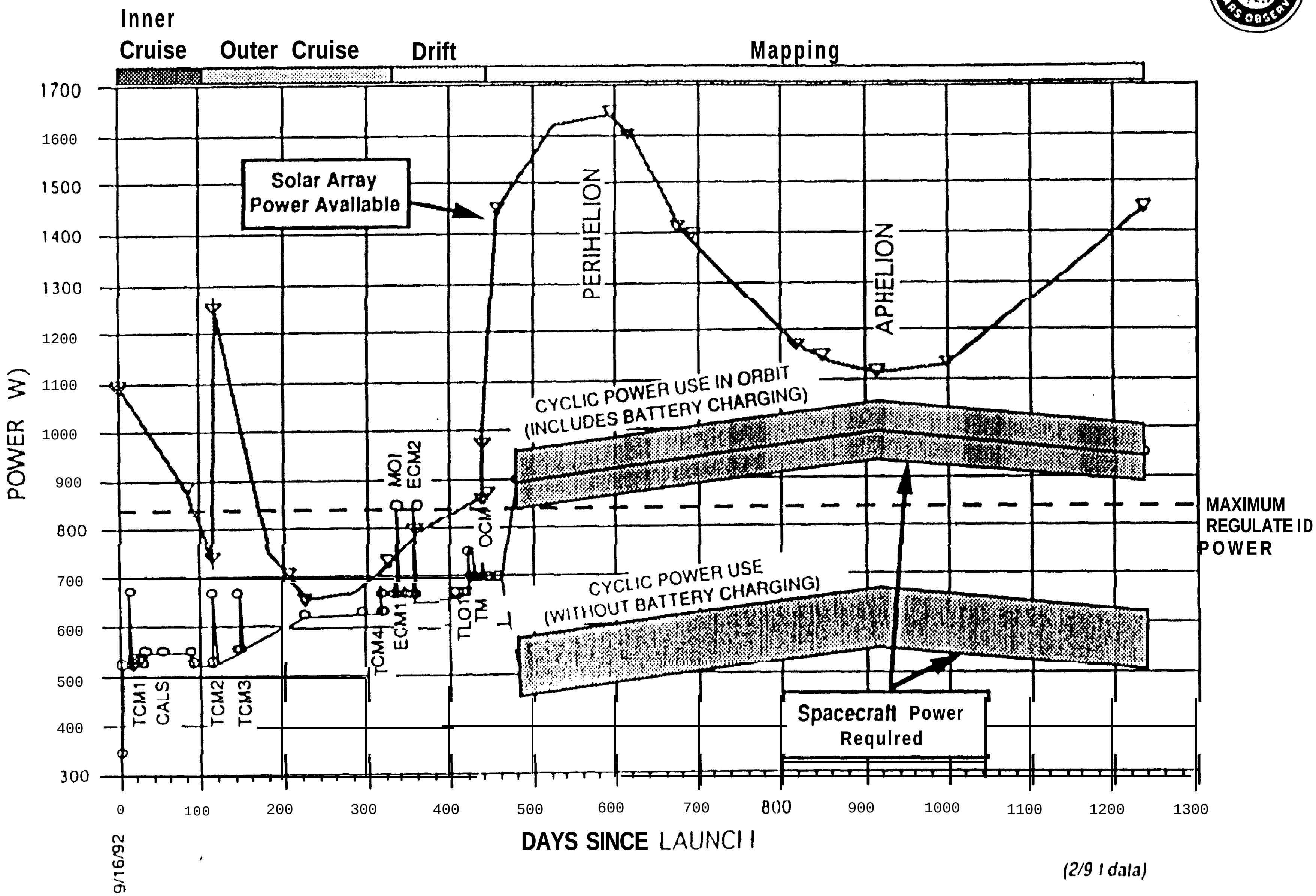
JPL

POWER BLOCK DIAGRAM



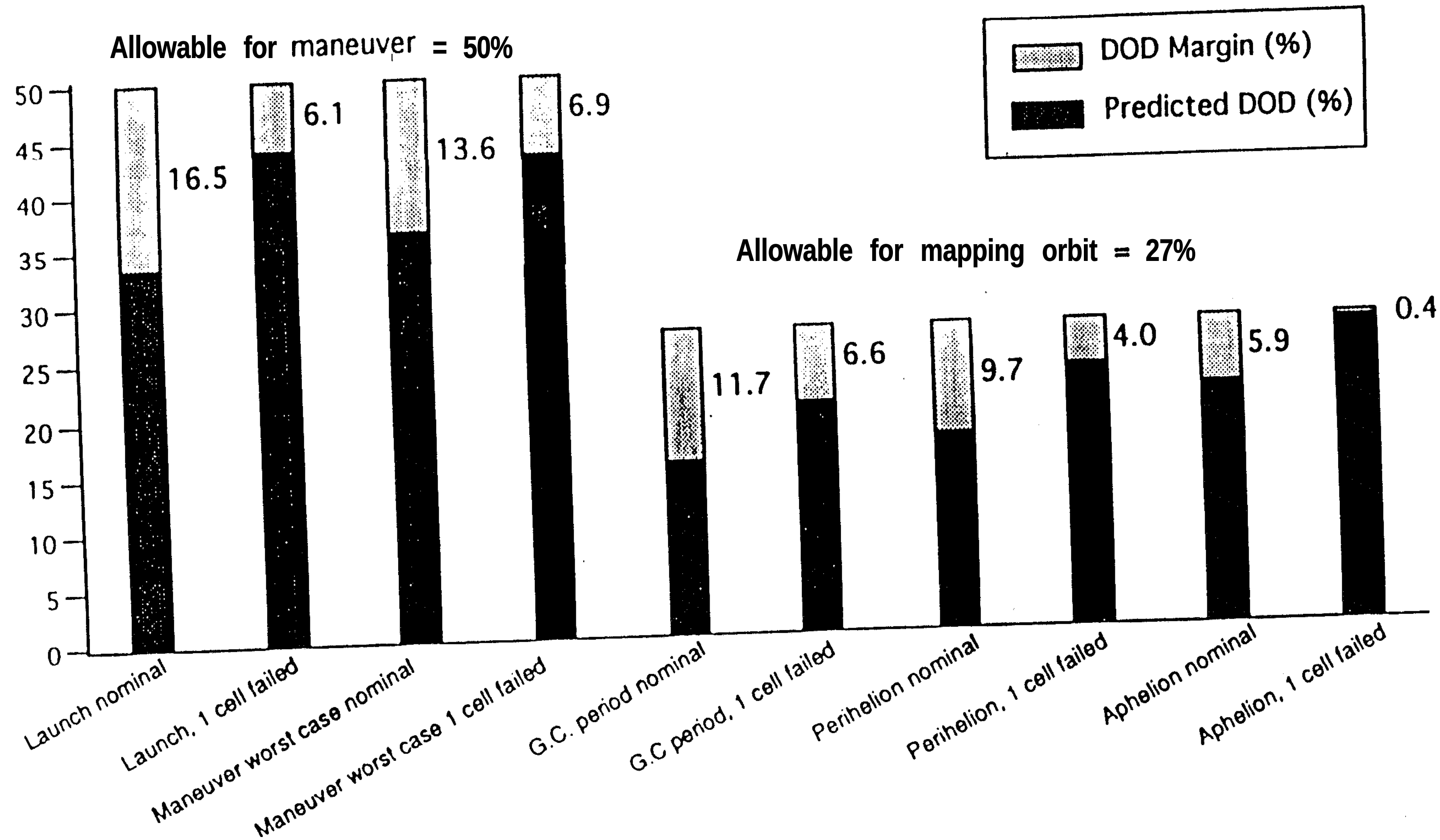
BAT BATTERY
BCA BATTERY CHARGE ASSEMBLY
BU. BACK UP UNIT
BVR BOOST VOLTAGE REGULATOR
FBA: FUSE BOX/BOARD ASSEMBLY
MC: MODE CONTROLLER
SA: SOLAR ARRAY (UPPER OR LOWER)
PSA: PARTIAL SHUNT ASSEMBLY
SAGD: SOLAR ARRAY DRIVE
SAGDE: SOLAR ARRAY GIMBAL DRIVE ELECTRONICS

SOLAR ARRAY POWER MARGINS



JPL

BATTERY MARGINS



JPL

CONTINGENCY STATUS

ANTICIPATED LIENS

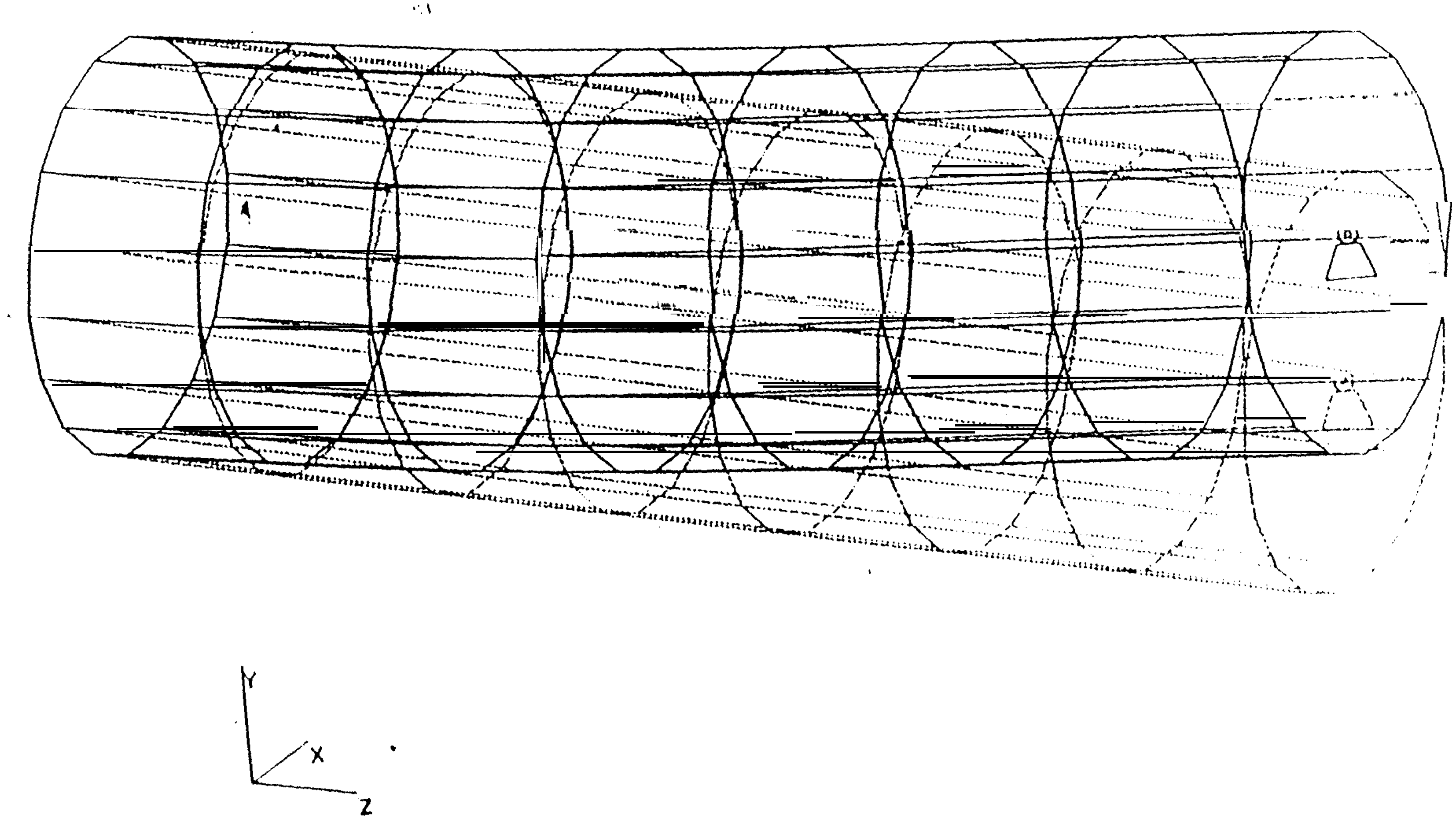
	FY91	FY92	TOTAL	REASON
LIENS • CONTRACT				
A003 ADDITIONAL S/C TESTING (QUAL&SYS)	100	500	600	RECOMMENDATION OF REVIEW BOARDS AND SECTIONS
A004 CORRECT PFR CLOSEOUT DEF.	50	100	150	JPL WON'T LIKE TDR CLOSEOUTS
A005 CORRECT RELIABILITY ANALYSIS DEF.	50	0	50	ADDITIONAL FMECA, WCA BASED ON JPL REVIEW
A014 I&T WORKAROUNDS FOR INSTR. PROBLM	0	500	500	ADDITIONAL SYS TEST TIME TO ISOLATE INSTR UNIQUE PROBLMS
A015 I&T WORKAROUNDS FOR LATE P/L	100	1,500	1,600	INSTRUMENTS ARRIVE LATE AT GE
A016 JPL BCE TO ASD SCS ICD'S	100	0	100	DOCUMENTATION FOR TEST EQUIPMENT INTERFACES
A018 LARGER S/C TEAM	90	130	220	PRE-LAUNCH MOS TEAM NEEDS TO BE LARGER
A019 LATE MAG & RAD CHARC TESTS	40	0	40	WORKING IN TESTS IF LATE
A036 PAS SCOPE	240	240	480	SFOC I/F SCOPE
A037 NEW FLIGHT S/W REQUIREMENTS	100	200	300	DESIGN CHANGES TO FLIGHT S/W AFTER CDR
A044 PAYLOAD NEW I&T REQUIREMENTS	200	1,500	1,700	NEW REQUIREMENTS
A049 S/C IMPACT OF P/L DESIGN DELTAS	100	0	100	ADDL MASS, POWER, FOV, ETC. I.E. 49079 AND 49033
A052 SEQUENCE TEST REQUIREMENTS	100	400	500	FLT SEQ VERIFICATION TESTING AT GE
A054 STUDY/IMPLEMENT SCIENCE ECR'S	170	0	170	NEW REQUIREMENTS/SCIENCE ENHANCEMENTS
A056 SYSTEM CONTAMINATION ANALYSIS	50	0	50	PRESENT CONTAMINATION APPROACH INADEQUATE
A071 TRANSPORT FIXTURE	100	0	100	SHIPPING CONTAINER DELETED FORM GE-ASD CONTRACT
A074 BRE CONFIDENCE TEST	200	0	200	MARQUARDT 490N TEST
A075 TWT PYRO SHOCK TEST	60	0	60	TEST TWT ON DURING SHOCK
A076 FAULT PROTECTION REVIEW/FIXES	300	100	400	ADD FORMAL REVIEW OF SIC FAULT PROTECTION
TOTAL • ANTICIPATED =	7,060	10,051	17,111	

	FY91	FY92	TOTAL
PROJECT CONTINGENCY	8,532	13,263	21,795
AUTHORIZED & PENDING LIENS	8,268	1,540	9,808
ANTICIPATED LIENS	7,060	10,051	17,111
TRUEUNENCUMBEREDBALANCE	-6,796	1,672	-5,124

ATTACHMENT 6

Figure 5

MARS OBSERVER TWTA
MODAL ANALYSIS
F1 = 18000 HZ

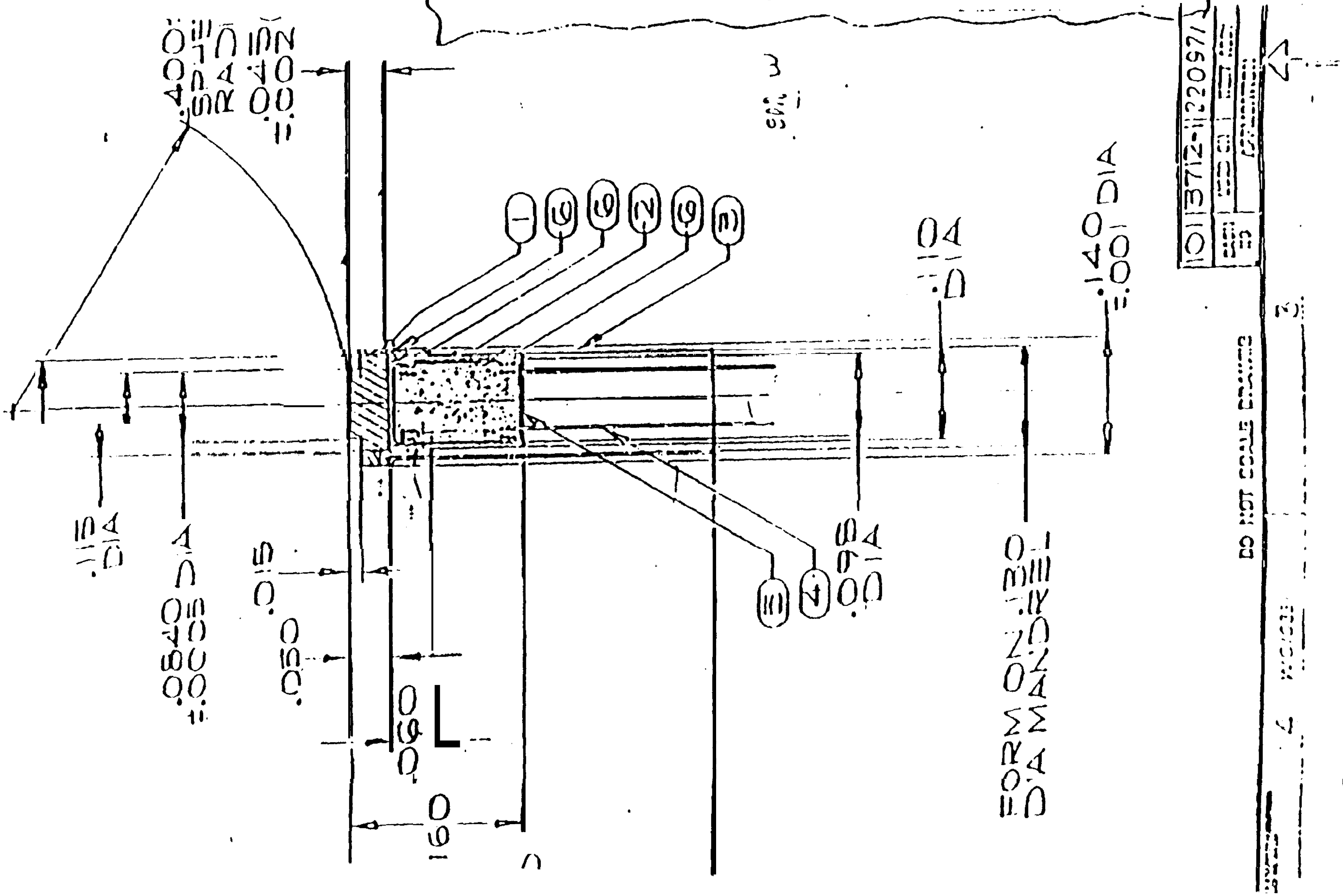


45

LTSAT Cathode & Support Details

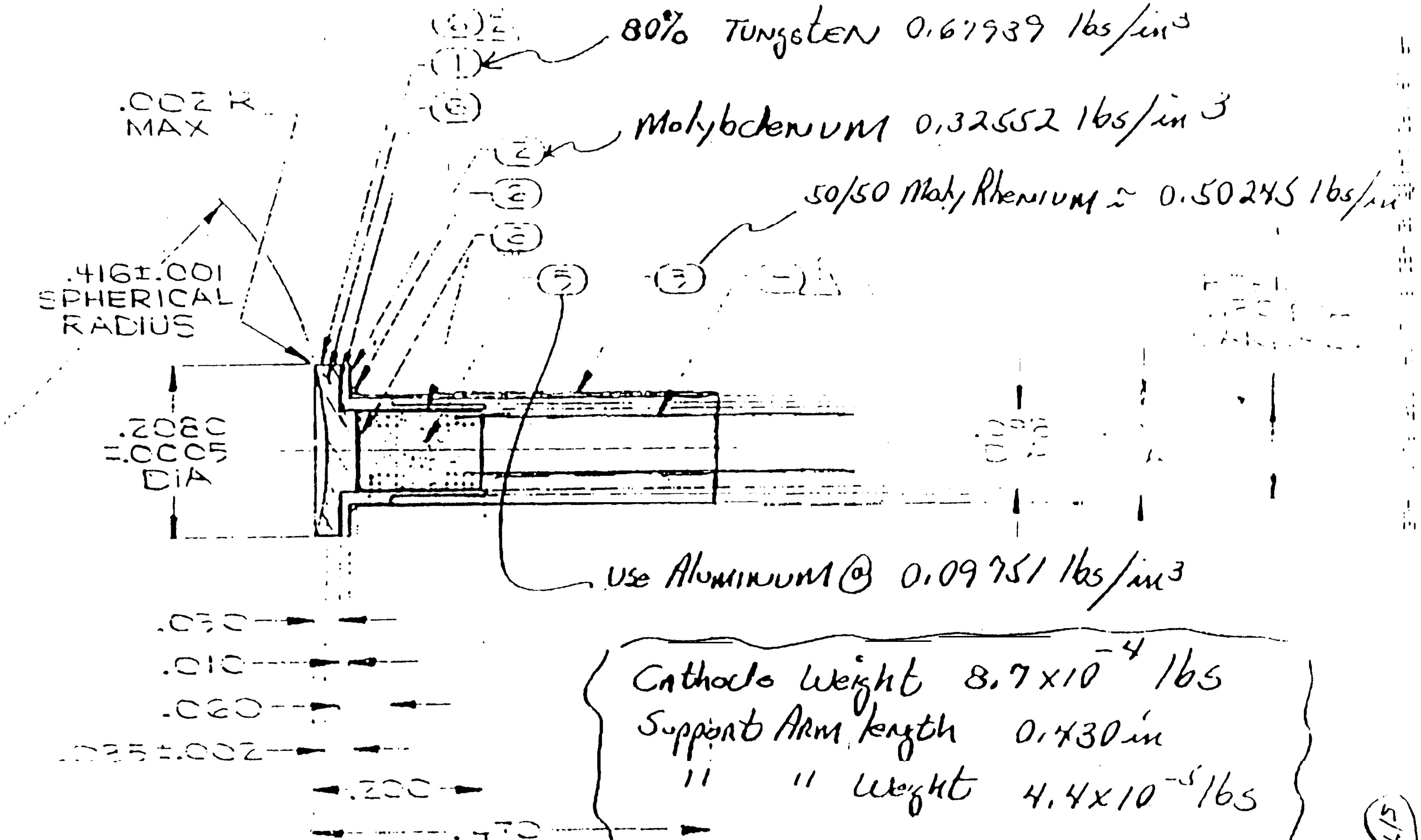
FIGURE 2

Cathode Weight = 6.27×10^{-4} lbs
Support Arm Length = 0.390 in
" " Weight = 4.0×10^{-5} lbs



M.O. Cathode & Support Details

FIGURE 1



Cathode Weight 8.7×10^{-4} lbs
Support Arm length 0.430 in
" " Weight 4.4×10^{-5} lbs

SOURCE(S)
CONST ED
OR

Figure 4

Mars Observer TWTA Cathode Support Tube

Finite Element Model

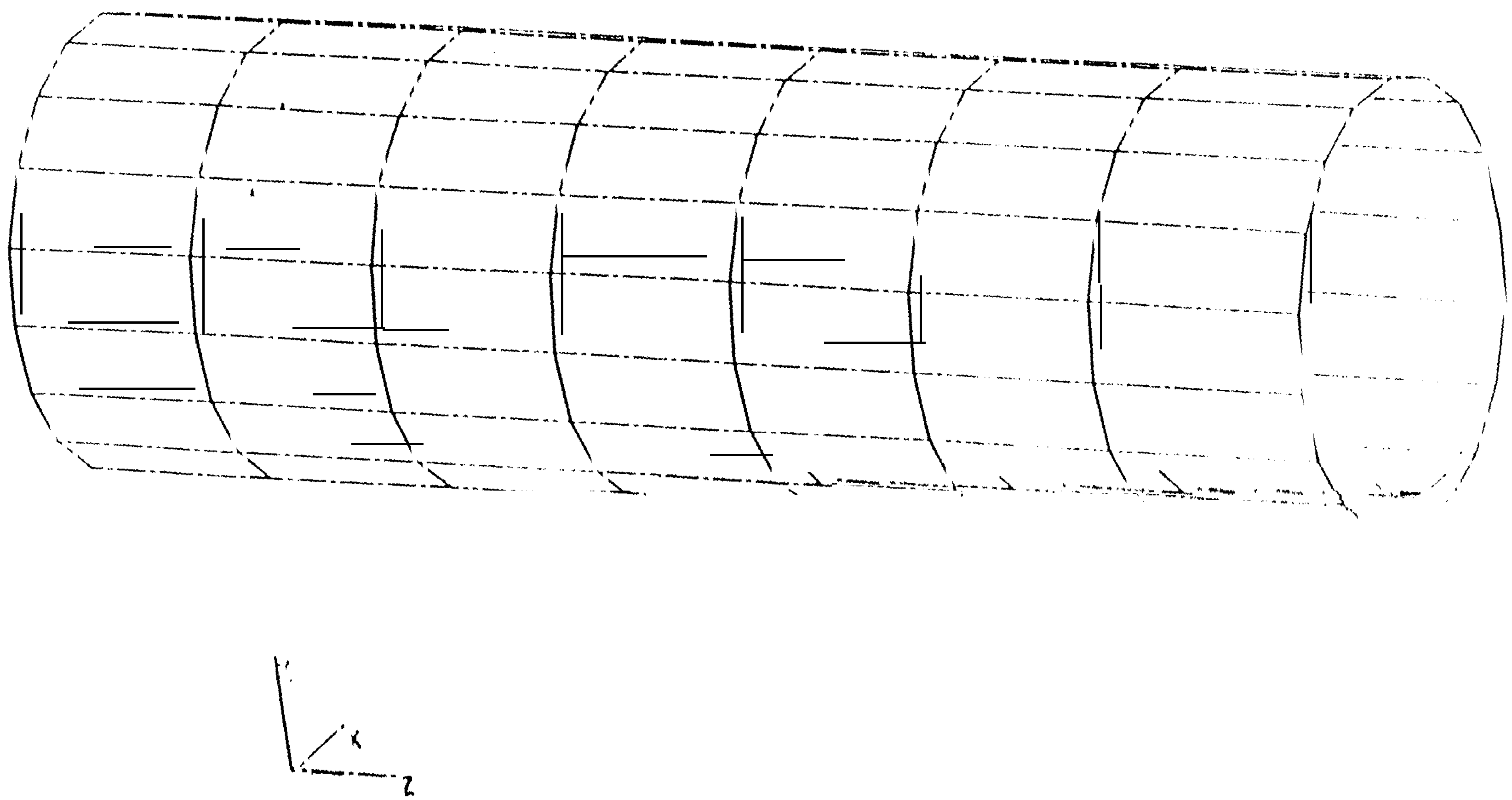
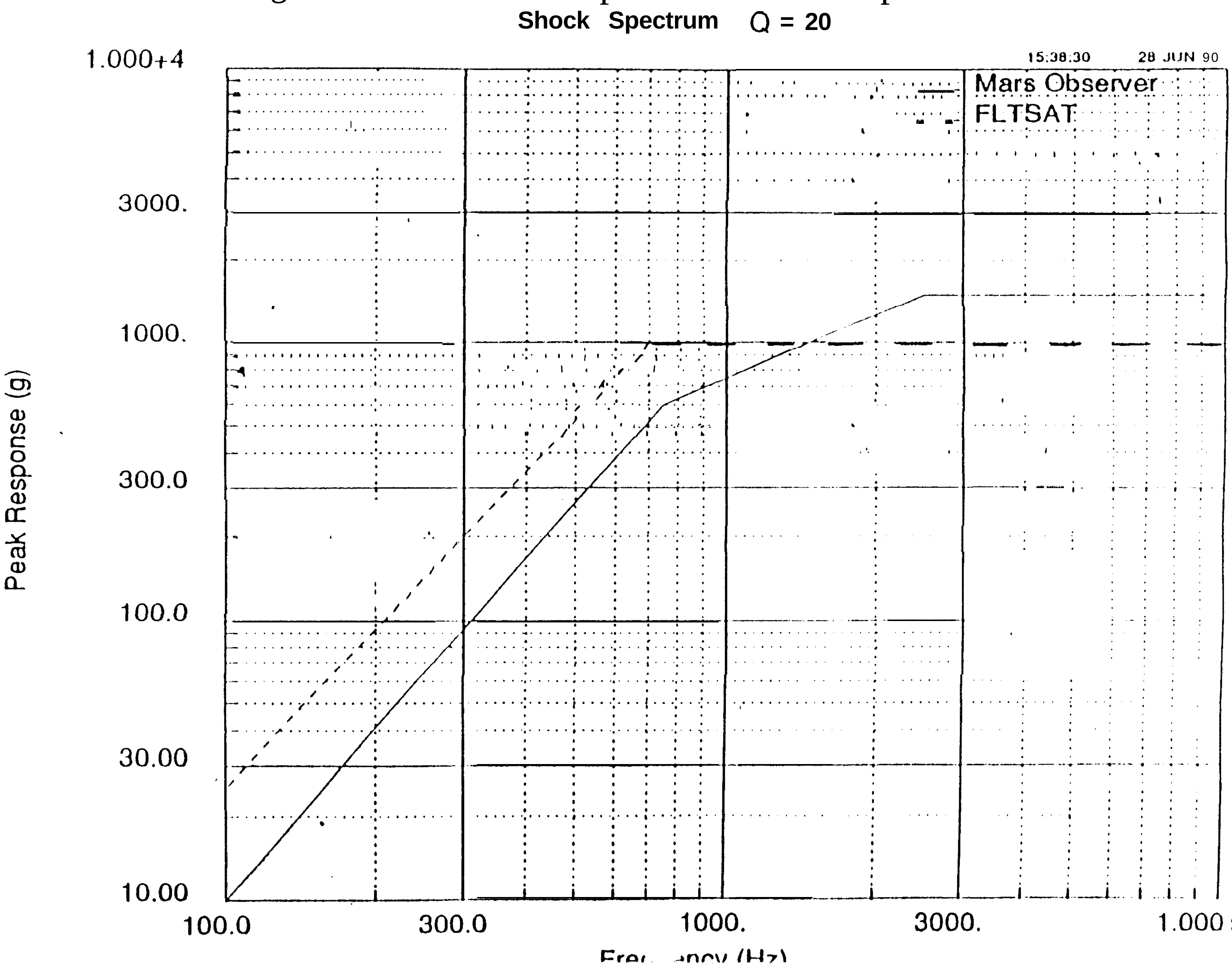


Figure 3 Shock Spectrum Envelope



45

ATTACHMENT 5

M. Burrow 7-1?
cc. J. Abraham
R. Brace
R. Schreib

MA90-MO-23 7
MO-SD/STR-41

GE ASTRO SPACE

To : W. Mandale

Date : 24-JUN-1990

From : J.E. Voce

Ext : 3848

Re : Mars Observer Shock Analysis on TWTA Cathode Support Tube

cc : D.Anderson, P.Kaskeiwicz, A.Martz, J.Matlak, G.Owen, S.Teitelbaum, Linda (MS 97)

References :

1) MA90-MO-22, "Mars Observer Shock Levels", J.Voce, 4th June 1990

2) Watkins-Johnson Company IOM, "Mars/Topex Cathode Support Struct Analysis", J.F.Wilson, 14th Sept 1988

The transfer wave tube assembly (TWTA) on Mars Observer (MO) is of slightly different design to prevoiusly flown assemblies. The area most changed and most susceptible to pyro shock inputs is the cathode support tube.

The cathode tube assembly for MO is shown in Figure 1. The support tube is cantilevered from its attachment to the base assembly, and supports its own weight. A similar arrangement is used on the FLTSAT TWTA, which is shown in Figure 2. This assembly has been subjected to shock without failure. The shock environment was however lower than the MO requirement.

A dynamic analysis was thus carried out to determine the correct input to the MO tube, and a stress analysis was performed to determine margins of safety and to make a recommendation on whether a shock test is necessary prior to spacecraft system pyro test.

Shock Inputs

Shock Inputs occur from pyro firings on the MO spacecraft and may be grouped into three categories:

i) Cable Cutters
ii) Propulsion Pyros
iii) V-Band Pyros

The enveloping shock spectrum for these events, developed in Reference 1, is shown in Figure 3, together with the FLTSAT shock input.

Dynamic Analysis

A modal analysis of the support tube was carried out by making a Finite Element (FE) model. Plate elements were used in the model which is shown in Figure 4. The first mode frequency was 18 KHz and the mode shape is shown in Figure 5. This first resonance shows that the shock input to the tube is at its maximum (1500 G from Figure 3).

Static Analysis

The static analysis was carried out by applying 1500G to the tube in the direction that simulates the firsts mode of the structure. The maximum stresses occurred at the fixed end of the tube and are summarized below:

Stress Type	Calculated	Allowable
Shear	650 psi	32000 psi
Tensile	2180 psi	48000 psi
Compressive	2180 psi	48000 psi

The allowables are derived from Reference 2 and include a 1.7 Ultimate material factor of safety. The margins can be seen to be ample even if further factors should be applied (a stress concentration factor of 10 is used in Reference 2).

Recommendation

The analysis shows that the cathode tube can easily withstand the MO shock environment. It is therefore sufficient to test the assembly at system shock test only. Previous testing of similar assemblies without failure gives further confidence in the recommendation.

John Voce
Mechanical Analysis

David Chu, Manager
Mechanical Analysis

...-988-2579

MARS OBSERVER PYROTECHNIC SHOCK TEST PROGRAM

Reference: Matlack/Egan GEASD memo MO-PRO-27 dated April 20, 1990 (request for proposal).

Description of RFP:

1. VSCO to provide test plans, test procedures, test vehicles, test fixtures, test implementation, test execution, and post-test analyses to demonstrate that the 1298-1 Mars Observer TWTA can survive pyrotechnic shocks of TBD peak gravities while operating without sustaining any damage or suffering any permanent degradation in performance.

2. VSCO to provide recommendations regarding shock testing of the flight TWTA's.

The referenced memorandum referred in turn to a technical discussion held at VSCO Palo Alto on March 23, 1990. In this discussion, VSCO suggested that if GEASD decided to pursue operating pyroshock testing, a NATO non-flight residual TWT could be provided for testing to 8ti8q concerns regarding the mechanical strength of the cathode support system in the TWT. The candidate NATO 3618-8 TWT, S/N 337R, has been successfully tested to NATO/SKYNET qualification requirements for sine and random vibration (non-operating), temperature, and thermal vacuum. The TWT cathode and gun structure are identical to the 3618-9 TWT produced for the Mars Observer program.

If it is deemed necessary to demonstrate pyroshock survivability on a Mars Observer configuration TWTA, the only candidate available for this purpose is the Engineering Model TWTA utilized for the High Voltage Verification program.

Rough Order of Magnitude proposal:

Task 1:

A:	Test plans, procedures, and analyses:	$43.5K
B:	Pretest of unit to be tested:	5.9K
	TWT only	5.9K
	Full TWTA	8.5K

C: Transport of equipment, set-up, and performance of test at test facility: 5.1K
Note: Requires full support station including rack power supply if TWT only or TWTA test rack if full TWTA is to be tested.

D: Transportation and support costs: 5.0K

E: Test Vehicle cost: TWT only $ 85.5K
Full TWTA $293.0K

NOTE: Test may be to destruction - GEASD may retain possession of test vehicle after test is concluded.

F: Outside Lab costs: 15.0K

G: Design and fabricate test fixtures and adaptor plates: 5.0K

Total ROM cost: TWT only $175,000
Full TWTA $385,100

VSCO [illegible] responsibility implicit or otherwise regarding the ability of the test vehicle to successfully survive the proposed 27 exposures to the TBD pyroshock test levels but will support the test program on a [illegible] efforts basis to help determine if the flight hardware will survive mission requirements as currently being determined. If the test vehicle does not survive the test program, VSCO will support customer's efforts to determine what modifications to the mission operational requirements will ensure sufficient margin of safety for the TWTA's.

Task 2:

VSCO does not recommend pyroshock testing of the flight units. Past programs requiring pyroshock testing wrote off the test vehicles as non-flight worthy following exposure to test levels even though the tests were successful (and these were non-operating tests).

If GEASD/JPL wish to perform out-of-scope operating pyroshock testing on the flight TWTA's, it is VSCO's position that such testing will have to be performed post-delivery at customer's risk and would be a violation of VSCO warranty provisions.

VSCO will support and perform such testing only if so directed. Price, schedule, and warranty impact will be provided upon receipt of such direction.

RMS/rms pyroshok

05/31/90 09:15

ATTACHMENT 4

cc: J. Abraham
R. Grace
R. Schneider

GE Astro Space

Astro Space Division
General Electric Company
P.O. Box 800, Princeton, NJ 08543-0800
609 426-3400

MO-CNTA-554

Jet Propulsion Laboratory
California Institute of Technology
4800 Oak Grove Drive
Bldg. 264, MS. 627
Pasadena, CA 91109

Attention: Mr. Robert Kinkade
Contract Negotiator Specialist

Subject: [redacted]

Reference: MO Contract No. 957444

30 May 1990

Gentleman:

Enclosed is a copy of information requested by System Tradeoff studies SYS-27 "Mars Observer TWT Mechanical Design Evaluation with Respect to Pyro-Shock and Random Vibration. This information includes a ROM from VARIAN, to provide Pyro-Shock Testing ror the Mars Observer program. Thia completes the above task requirements.

Any additional questions, please contact the undersigned.

Very truly yours,

S. Danner

S. Danner
Manager, Program Financial control

cc: K. Byrne - w/encls.
N. Gauss - "
S. Danner - "
N. Miles
J. Matlack

A4

2

...-988-2579

varian

microwave equipment division

MATLACK 5-14-90

RECEIVED
MAY 23 1990
J. MATLACK

11 May 1990

In reply refer to
NKS-90-224

cc: N. Miles
S. Denn
N. Gauss
A. Martz
W. Menda

General Electric Company
Astro Space Division
P. O. Box 800
Princeton, New Jersey 08543-0800

Attention: Mr. J. Egan

Subject: Rough Order of Magnitude Proposal for Pyroshock Testing, Subcontract No. G-1-P084-4001-00-F44

Reference: GEASD Memo MO-PRO-27 dated 20 April 1990

Gentlemen:

In accordance with the referenced memo, attached is a rough order of Magnitude (ROM) estimate for performing pyroshock testing on the Mars Observer program. This is an estimate only and is not a commitment by Varian Associates, Inc.

Please contact the undersigned if you have any questions.

Sincerely,

Robert W. Wolf
Contract Specialist

M.Q. TWTAS - PYRO SHOCK (APRIL 90) CONT'D

- RFQ FOR PYRO SHOCK TESTING **WAS FORWARDED TO** VARIAN 23 APRIL.
  - USE NON-FLIGHT **NATO TWT AS** INITIAL **TEST ARTICLE**
  - TEST OUTLINE AS FOLLOWS:
    1) 3 SHOCKS IN EACH AXIS (BEGINNING AT **6db** 'BELOW SPECIFIED SHOCK LEVEL)
    2) 1ST SHOCK NON-OPERATING FOLLOWED **BY FUNCTION** TEST TO VERIFY PASS/FAIL'
    3) 2ND & 3RD **SHOCKS IN** OPERATING CONDITION WITH **RF** APPLIED AND PERFORMANCE MONITORED
    4) REPEAT STEPS 1 THRU 3 AT SHOCK LEVEL **OF SPEC-3db** AND AGAIN AT SPEC SHOCK LEVEL
  - TEST FACILITY/LOCATION OPTIONAL WITH **ASTRO, JPL** OR LINCOLN LABS AS POSSIBLE TEST SITES
- RFQ REQUESTS VARIAN RECOMMENDATION FOR VERIFICATION **OF** FLIGHT MODEL TWTA OPERATING . SHOCK SURVIVABILITY.
- **RFQ** RESPONSE **REQUESTED BY 7 MAY.**

Paul Sarnoski
4-23-93

ATTACHMENT 5

MARS OBSERVER
RECOMMENDATION FOR ACTION (RFA)

REVIEW TITLE: SPACECRAFT SYSTEM CDR
REVIEW DATE: MARCH 20, 21, AND 22, 1990

SUBMITTED BY: Al Wolfe/etc. ORGANIZATION: ______

TOPIC: TWTA/RPA

STATEMENT OF CONCERN:
Inability to leave TWTA/RPA on during dynamic transients such as pyro shocks **impacts** requirement for initial acquisition by the DSN. It also may prevent project from obtaining real time telemetry data validating S/C pointing before trajectory correction maneuvers-

RECOMMENDED **ACTION:**

1) Reassess risk of leaving RPA on during some or all pyro shocks.
2) If reliability concern is still an issue,
   1) Resolve incompatibility with DSN initial acquisition requirements
   2) Consider redoing maneuver sequence so the RPA is not turned off until after the turns have been completed.

FOR PROJECT USE ONLY. DO NOT WRITE BELOW THIS LINE.

RFA DISPOSITION:
☐ Accepted for Action ☐ Accepted for Advisory ☐ Rejected

RFA NUMBER: 34 **ACTION ITEM NUMBER:** ______

RATIONALE FOR DISPOSITION:
See RFA #1 for DSN initial acquisition. Project is considering shock testing on powered TWT. Maneuvers are not considered to be a severe shock environment. Obtaining real-time telemetry during maneuvers will be considered on a case-by-case basis depending on power availability and mission risk.

COGNIZANT MANAGER: Pace/Gauss

ACTION **ASSIGNED** TO: Potts/Davis DUE DATE: ______

We reviewed prior shock teat spectra which appear in various M.O. documents and concluded that further clarification by M.O. engineering would be *helpful* in determining the shock teot levels. Sample data attached.

PRS/pjl

Attachments

cc: R. Swan/Varian
F. Ott/JPL

M.O. TWTAS - PYRO SHOCK (APRIL 90)

- ABILITY TO SURVIVE OPERATING PYRO SHOCK IS BEING INVESTIGATED,
  - VARIAN HAS NO T&ST DATA FOR OPERATING SHOCK ON ANY TWT OR TWTA
  - ALL HERITAGE IS NON-OPERATING SHOCK
  - VARIAN CONFIDENT M.O. &PC WOULD' SURVIVE, BUT HAS RESERVATIONS ABOUT TWT
  - ANALYSIS INDICATES TWT CATHODE SUPPORT STRUCTURE IS WEAK&ST LINK

- REVIEW MEETING HELD 22 MAR AT VARIAN WITH JPL AND GE PERSONNEL.
  - PRELIMINARY MECHANICAL ANALYSIS INDICATES OPERATING TWT HAS POSITIVE MARGIN FOR 3,000G ACCELERATION LOAD
  - CONCENSUS OPINION THAT DEMONSTRATION BY TEST SHOULD BE PERFORMED IF MISSION REQUIRES SHOCK WHILE OPERATING
  - NO NON-FLIGHT M.O. TWTS AVAILABLE FOR TEST SAMPLES
  - NATO TWT CATHODE 8 SUPPORT IS IDENTICAL DESIGN AND NON-FLIGHT NATO TW-1-S ARE AVAILABLE

PAGE 1

Paul Sarnoski
4-23-93

The meeting attendees agreed that the optimum test would be to operate the TWT as it would be operated during flight. We agreed that a test without any power applied should be performed before the operating test. Two power on test should follow the unpowered test to establish the desired margin. Each test would consist of three shock levels of -6 dB, -3 dB and full amplitude.

It is recommended that such a test be performed, preferably fully powered, and a ROM be requested from G.E. relative to the cost of the different test options.

A copy of the March 22, 1990 meeting minutes are attached for reference.

FO:sb

90-030.IOM

attach.

GE Aerospace

General Electric Company
... Moffett Park Drive, ...
Sunnyvale, CA 94089-1322
408 734-0523

TO: J. Matlack

FROM: P. Sarnoski

DATE: 22 March 1990

SUBJECT: Meeting Minutes - TWTA Shock - M.O. Program

Attendees:

NASA-JPL	GE	VARIAN
F. Ott	M. Hammer	R. Swan
A. Kissle	P. Sarnoski	J. Wilson

The meeting was held at Varian Space TWT facility in Palo Alto, CA, on Thursday 3/22/90. Mr. Frank Ott stated that the meeting was requested by JPL in order to assess the M.O. TWT design regarding operation thru the shock environment and to provide recommendations. NASA would like to operate the TWTA during deployments (shocks) of the M.O. mission.

The analysis by Varian (Attachment #1) indicates that the design h margin remaining for operating vibration. Hr. Kissle JPL indicated that his assessment also showed positive margin for 3,000 G shock with the cathode hot (operating).

A consensus opinion was reached that based upon existing analytical data, it is likely that the H.O. TWT would survive operating shock. However, everyone felt that if it war intended for the M.O. mission to rely on TWT success thru operating shock, then the capability should be demonstrated by teat of a sample TWT.

Varian indfcatrd that there were no M.O. TWTs available for testing except flight models. They proposed that sample(r) of non-flight NATO TWT(s) could be used as test samples. We then reviewed significant aspects. of the NATO design and concluded that the NATO TUT would be an adequate representation of the M.O. design regarding shock survivability.

Prior shock tests have been performed on TWTs at GE, JPL, and Lincoln Labs and either would be acceptable for M.O. tasting, if required.

The group briefly described a shock test outline which would provide needed information as follows:

1) 3 shocks each axis;
2) 1st rhock each axis in non-op condition followed by test to verify pas8 or fail;
3) 2nd and 3rd shocks each axis in operating condition with RF applied;
4) Repeat steps 1 thru 3 tar progressively increased shock levels starting with levels 3 to 6 db below the specified test levels and progressing to the specified test levels.

ATTACHMENT 2

JET PROPULSION LABORATORY

INTEROFFICE MEMORANDUM
330-NAB-90.19

18 April, 1990

TO: G. Pace

FROM: N. Burow

SUBJECT: RECOMMENDED TEST TO DEMONSTRATE MARS OBSERVER TWT DESIGN QUALIFICATION FOR OPERATION IN SHOCK ENVIRONMENTS

REFERENCE: IOM 3366-90-030, "Mars Observer TWT Ability to Survive *Pyro* Shock While the TWT is Operating," from F. Ott/A. Kissle, 4/10/90 (attached).

A review of the Mars Observer TWT design was held 3/22/90 at Varian Associates in San Jose, Ca. (Varian is providing the M.O. TWTA under subcontract to GE-ASD). The purpose of the review was to determine if the TWT design would permit operation without damage in the Mars Observer shock environment. Technical representatives from GE-ASD, JPL and Varian were present. A consensus was reached (based on analysis) that the MO TWT would survive the required shock environment without damage with the TWTA in a normal operating condition (i.e., powered). It was also the consensus that this capability should be demonstrated by margin testing before committing to use in flight.

The referenced IOM provides more details from the meeting and the recommended actions to be taken to obtain the necessary test verification. A shock test with the TWT fully powered (including high voltage and RF drive) is strongly recommended.

DISTRIBUTION:

J. Abraham
R. Brace
S. Butman
K. Curry
H. Detweiler
R. Green
C. Hamilton
R. Horttor
R. Jones
A. Kissle
T. Komarek
J. Xoukos
B. Madsen
J. Meeker
W. Moore
F. Ott
D. Potts
R. Schoenbeck
M. Traxler

A 2

JET PROPULSION LABORATORY

INTEROFFICE MEMORANDUM
3366-90-030

April 10, 1990

TO: N. Burrows

FROM: F. Ott/A. Kissle F. Ott/A. Kissle

SUBJECT: Mars Observer TWT Ability to survive Pyro Shock while the TWT is operating.

On March 22 1989, a meeting was conducted at Varian relative to Mars Observer (MO) TWT shock capability. Varian was represented by R. Swan and J. Wilson; G.E. by M. Hammer and P. Sarnoski; and JPL by A. Kissle (a mechanical engineer) and F. Ott. J. Wilson of Varian, P. Sarnoski of G.E. and A. Kissle of JPL had independently performed calculations prior to the meeting.

Varian and JPL calculations indicated that the TWT would survive operating shock levels which conformed to the typical pyro shock curve with a peak amplitude of 3,000 gs. G.E. calculations evaluated and concurred with the Varian/JPL calculations. Worst case estimated values were used for the calculations. Most of the margin above the 3,000 G level would be expected to exist because many of the parameters would not be at their worst case value.

No person at the meeting recommended operating the TWT during pyro-shock on a flight mission without performing margin testing. Varian stated that a NATO out-of-specification TWT could probably be made available for the test. The NATO TWT design relative to the MO TWT was reviewed and the group agreed that such a TWT could be used for a representative test. No MO TWTs are available for pyro shock testing and the cost of fabricating one would be high.

The least expensive operating shock test that could be performed would involve cathode heater power only. The TWT shock capability is Lowered by heating the cathode support during operation, Deflection of the TWT beam during shock is not expected to be a problem. If beam interception currents exceed safe levels, TWTA protection circuits should remove the TWT voltages: but, a cathode heater power test would not indicate whether the TWTA would turn off because of pyro shock.

A more expensive shock test would be with the TWT functioning as it would in flight. An engineering power supply exists that could be modified to mate with the test TWT. Probably new HV transformers would have to be manufactured and installed. Another possible approach would be to use the TWT Test Rack that was used to test the TWT. Nothing was said about using such a rack for the shock test.

A1 -

MARS OBSERVER

GE-ASD LEVEL OF EFFORT TASK STATEMENT

TYPE: _X_ SYSTEM TRADEOFF ___ MISSION DESIGN ___ MOS

DATE: 9 March 1990 AUTHORIZATION NO. _SYS-27_
REVISION _(N/A)_

TITLE: TWT Mechanical Design Evaluation with Respect to Pyro-Shock. and Random Vibration

HOURS: 40

DESCRIPTION:

Provide support for a technical meeting (at Varian) to allow JPL to review the internal construction of the Mars Observer TWT, Insure that **appropriate** subcontractor tecnnrcal personnel and TWT drawings are available to support discussion of the **mechanical** design. Provide support for follow-up clarification teleconference calls as required.

An understanding of the Mars Observer TWT's internal construction is required **to evaluate** its ability to operate **during pyro-shock** and random vibration environments. Operation within specification during these environments is not required: only that the TWT **not sustain** permanent damage or performance degradation following **these events**.

REPORTS: N/A

SCHEDULE: This task; shall be completed by 15 April 1 I'??(I)-

APPROVALS: Nathan A. Bussey

George D. Pace Jr.
Spacecraft Manager

RECEIVED
GEORGE D. PACE, JR.
MAR - 9 1990
File

ATTACHMENT 1

Jet Propulsion Laboratory
California Institute of Technology
4800 Oak Grove Drive
Pasadena California 91109
(818) 354-4321

JPL

March 12, 1990

Refer to: 622-RCK76:emz

GENERAL ELECTRIC COMPANY
Astro-Space Division
P.O. Box 800
Princeton, New Jersey 08543-0800

Attention: Mr. Sterling Danner
Program Financial Control
M/S 63

Subject: Level-of-Effort (LOE) Authorization

Reference: (a) Contract No. 957444

Authorization No. SYS-27
Revision: ______

In performance of: _x_ System Trade-Off Studies
_____ Mission Design Support
_____ Mission Operations Support

You are hereby directed to expend up to 40 LOE work hours for the Mars Observer TWT Mechanical Design Evaluation with Respect to Pyro-Shock and Random Vibration as defined in the attached task statement.

It is requested that this LOE task be completed no later than 4/15/90.

Very truly yours,

Robert C. Kinkade

Robert C. Kinkade
Contract Negotiator Specialist

Attachment: as stated

cc: N. Burow
R. Jones
G. Pace
D. Potts
G. Robinson

considered vital. While the project understood the benefit of keeping the RPA on during these events, the loss of telemetry data would be an inconvenience but would not be in series with mission success. Avoiding the additional cost was consistent with other project actions at that time to contain the projected cost growth.

Based on the above, it was decided to forgo the powered-on pyro shock test and submit a flight rule (Attachment 8) that the TWT be turned off during pyro shock events. No attempt was made to "fine tune" this rule during flight for cases where the levels might have been low. To do so would have required additional analysis to determine the shock levels, a shock test of a powered-on TWT, and modification and test of the flight blocks and sequences. Funds were not available for this level of testing during flight either. Without this effort, JPL would have been at risk for the entire on-orbit performance award if we left the TWT on during pyro shock and it subsequently failed.

In conclusion, while I can't imagine us not qualifying the TWT for powered-on pyro shock should we have the opportunity to re-fly Mars Observer, the original decision was consistent with good engineering judgement for the conditions that existed at the time. Doing pyro shock testing at the system level and foregoing the powered-on TWT pyro shock test which was only an enhancement are examples of actions consistent with the "faster, better, cheaper" and "doing more for less" concepts now being espoused. !

Good point.

Distribution:
N. Burow
J. Casani
D. Evans
N. Gauss (ASD)
R Gibbs

four flight RPAs on order. The first RPA got delivered in March of 1991 and was used to support the DSN compatibility test. Subsequent deliveries were spaced over many months.

The project did not want to commit a flight tube to support this test. In their response, Varian had cast doubt on the subsequent flight worthiness of any tube undergoing powered-on pyro shock (see task 2 of Attachment 4). Since we were having problems delivering any RPAs at the time, procuring a spare tube was essential for the test.

A separate Astro analysis at the time (Attachment 5) indicated that the tube should easily survive the Mars Observer pyro shock environment and recommended that the assembly be tested at the system level only. Astro's position was that the normal operating mode based on their previous experience was to have the tube powered off during pyro shock. Their experience was based on having all the pyro events early in the mission. They were not concerned about turning the TWT off during these times, but supported doing a powered-on test if JPL was willing to pay for it. Powered-off pyro shock tests were subsequently conducted at the system level with many firings. No failures occurred during these tests.

In anticipation of a powered pyro shock test, a lien of $60 K was added to the project lien list on May 4, 1990. At that time, this lien did not anticipate the need for a spare tube. The lien was carried and reported to NASA through the August budget review (Attachment 6). The original estimate of $60 K was retained in the lien listing, but when the higher contractor estimates became known, they were used in making a decision. Attachment 6 also illustrates the difficult financial situation the project was in at this time. The reserve for the year had been consumed by the authorized and pending liens. Anticipated liens for a variety of development problems required an additional $6.8 M beyond the reserve.

The status was reported to the Mars Observer project senior review board on June 6, 1990 and the NASA program manager in the June 13, 1990 program review (Attachment 7). The project's plans to turn off the TWT during shock events and the prohibitive cost of the test were discussed.

The project had already accepted the fact that the TWT beam would be cycled off during every orbit of mapping due to power limitations. A development test of 11,450 on/off cycles of the TWT beam was successfully conducted. In addition, the RPA was expected to be cycled on and off during system testing. A few more cycles to turn the tube off during maneuvers and pyro shock was not considered a problem. By launch, each RPA had undergone 142 on/off cycles with no problem.

Since the project had already accepted the loss of telemetry data during large portions of the mission, the loss of data during pyro shock events was not

JET PROPULSION LABORATORY

INTEROFFICE MEMO
GDP: 93-15

To: G. Cunningham

October 28, i993

From: G. Pace

Subject: TWT Power State During Pyro Shock

The following is the background leading to the decision to turn the traveling wave tube (TWT) off during pyro shock events. This is my recollection based on my notes and memory, project documentation, and discussions with other project personnel.

Concern about turning the TWT's off during maneuvers and pyro shock events first surfaced after the Telecommunications Subsystem CDR on November 13 and 14, 1989. Division 33 brought this concern to the attention of the project after reviewing the RF power amplifier (RPA) CDR package. The RPA consists of the TWT and a matched power supply. The main topic being investigated at the time was the potting of the high voltage power supply, but it was discovered during review of the test plan that the TWT was to be turned off during vibration and pyro shock testing. The Division's main concern was the lack of telemetry data when the TWT was turned off during maneuvers and pyro shock events.

I brought this concern to the attention of Dave Evans on March 7, 1990. On March 9, 1990, I initiated a level-of-effort SYS27 (Attachment 1) for Astro and their subcontractor, Varian, to support a meeting to discuss the shock qualification of the TWT. That meeting was held on March 22, 1990, at Varian with JPL, Astro, and Varian in attendance. A consensus from that meeting (Attachment 2) indicated that the TWT should survive pyro shock in the powered condition, but that a test was required to demonstrate this capability. Varian stated that all of their heritage for pyro shock was based on an unpowered TWT.

The concern was also raised at the Spacecraft System CDR on March 20, 21, and 22, 1990 (Attachment 3). The project responded that turning the RPA off during maneuvers was a power concern not a question of survival. The project stated that a pyro shock test with the TWT on was under consideration.

I subsequently verbally directed Astro in late March as part of LOE SYS-27 to obtain a cost estimate from Varian to perform a pyro shock qualification test with a powered TWT. The results of the LOE were submitted on May 30, 1990 (Attachment 4). Varian's cost to do the test was $175 K for the TWT only and $385 K for the full RPA. With Astro and JPL loadings the full cost to the project would have been approximately $250 K to $550 K. This cost was high because a spare tube had to be purchased from Varian to support the test. Recall at this time, we were having technical and schedule problems at Varian and were concerned about delivery of the

Jet Propulsion Laboratory
California Institute of Technology
4800 Oak Grove Drive
Pasadena California 91109-8099
(818) 354-4321

JPL

Reply to: GEC:250-93-187

October 28, 1993

Mr. William C. Panter
Code SLP
National Aeronautics and Space Administration
Washington, D.C. 20546

Dear Bill:

Pursuant to your request of October 15, 1993, I have attached Mr. George Pace's summary of the issues and decisions associated with the flight rule to turn the transmitters off during pyro shock events.

If there is additional information or clarification required, please do not hesitate to call.

Very truly yours,

Glenn E. Cunningham
Manager
Mars Observer Project

Attachment

-MR300697-

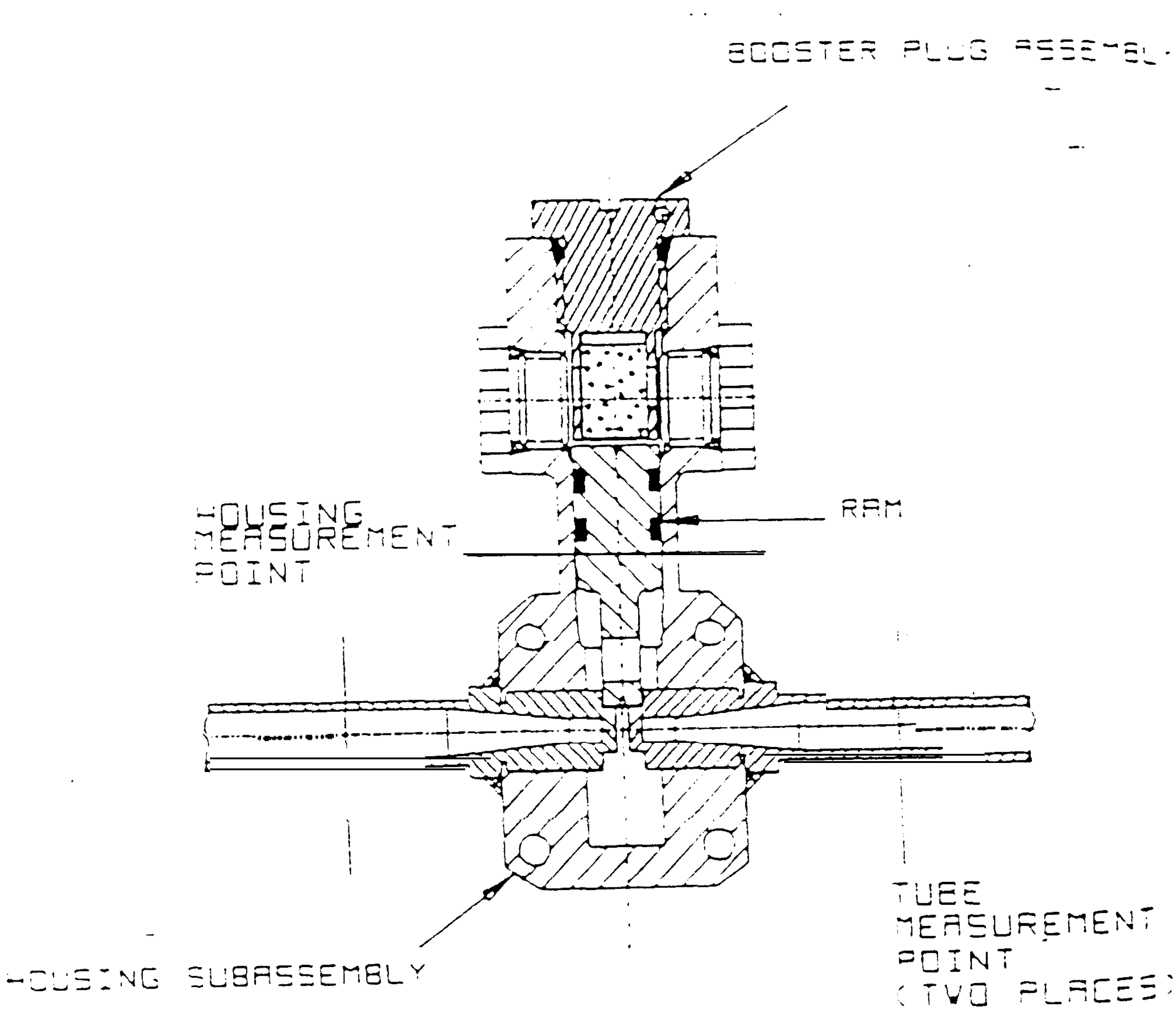


Figure 3.2-7 Normally Closed Pyrovalve

3.2.1.6 GHe Pressure Regulators

The regulator is a series-redundant regulator. Figure 3.2-8 provides a sectional view of the regulator. Each regulator stage is capable of maintaining low system pressure within MEOP. The worst-case regulator lockup pressure assuming a failed primary regulator stage is 276 psia ([illegible] MPa). The regulator is not pressurized until following MO/TOS separation when the pyrotechnic valves are initiated. The max regulator leak rate is 30 cm3/hr. The regulator is designed to maintain the MMH tank and NTO tank delivery pressure to the 490-N engines at 255 psia during engine firings. The regulator MEOP is 4,500 psia, proof pressure is 6,750 psia, burst pressure is [illegible] psia.

3.2.1.7 Check Valves

Two groups of two series connected check valves preclude mixing of MMH and NTO after the pyroisolation valves are fired. Figure 3.2-9 provides a cross-sectional view of the check valve. Check valve materials are compatible with MMH and NTO. Check valve MEOP is [illegible] psia, proof pressure is 450 psia, and burst pressure is 20,000 psia.

Two series connected check valves and one NC pyrotechnic actuated isolation valve preclude mixing of bipropellant vapor during ground operations and launch. The pyrotechnic valve design provides a separate seal for the inlet and outlet sides of the valve. The combination of these devices makes the system three fault tolerant against mechanical failure. The pyrovalve is not operated until after launch.

3.2.1.8 Latch Valves

Four single seat torque motor actuated latch valves isolate MMH and NTO from the four 490-N thrusters and from the four 22 N thrusters. Figure 3.2-10 provides a cross-sectional view of the latch valves. The valves are normally closed until separation from the TOS. The position of the valve is sensed through a microswitch position indicator and valve status is part of the spacecraft telemetry stream. Valve actuation time is 50 milliseconds. The valve MEOP is 300 psia (inlet) and 600 psia (outlet). Proof pressure is 900 psia. Burst pressure is 1500 psia. The 600 psia outlet pressure takes water hammer spikes into account. The valves provide back pressure relief capability. The valve is compatible with NTO, MMH, GHe, GN2 and isopropyl alcohol.

3.2.1.9 Service Valves

Three types of service valves are used on the MO spacecraft; 0.25 inch high pressure service valves, 0.25 inch low pressure service valves, and 0.5 inch service valves. Figure 3.2-11 shows a typical MO spacecraft service valve. All three types of service valves are manually operated and constructed of 6Al-4V titanium. Each service valve will provide two mechanical seals to prevent propellant or pressurant leakage. The two seals are provided by the valve seat when closed and the installation of an internal cap. The 0.25 inch high pressure service valves are used to pressurize the helium pressurant tank and to perform pressure tests on the subsystem. The 0.25 inch low pressure valves are used for venting the MMH and NTO tanks during loading and to conduct pressurized tests of the subsystem. The 0.25 inch low pressure service valves used in the MMH manifolds, NTO manifolds and GHe manifolds (as well as the N_2H_4 monopropellant service valves) have unique inlet fittings to prevent mismating GSE couplings. The 0.5 inch service valves are used to load and offload the MMH and NTO tanks and each has a unique inlet fitting to prevent mismating. All service valves are compatible with MMH, NTO, water, isopropyl alcohol, GHe, and GN2. MEOP, proof, and burst pressures for the various service valves can be found in Table 3.2-1.

3.2.1.3 490-N Thruster

Four 490-N thrusters are provided. Figure 3.2-5 provides a cross-sectional view of the bipropellant thruster valve. Each thruster has two valves that control the flow of NTO and MMH. Each valve has an armature that causes a poppet to open and close the port, causing oxidizer or fuel to flow from the valve, through an injector, and into the thrust chamber where combustion takes place. The valves, which were factory EB-welded and hydrostatically tested to assure leak free performance, are designed to fail safe (fail closed) when control signal is lost. The engine electrical components are explosion proof. The thruster is compatible with MMH, NTO, GHe, GN2, isopropyl alcohol, and Freon 113. The thruster MEOP is 400 psia. It is proof tested to 600 psia and has a burst pressure equal to or greater than 1,000 psi.

3.2.1.4 22-N Thruster

The spacecraft has four 22-N thrusters. A cross-sectional view of the thruster valve is shown in Figure 3.2-6. Each thruster has a torque-operated valve that controls the flow of NTO and MMH. The valve has an NTO port and MMH port. A torque motor [illegible] assembly to open and close the ports, causing oxidizer and fuel to flow from the valve through an injector and into the thrust chamber where combustion takes place. The valves are designed to fail safe (fail closed) when control signal is lost. The valves are EB welded and hydrostatically tested at the factory to assure leak free performance. The thruster electrical components are explosion proof. The thruster is compatible with MMH, NTO, GHe, GN2, isopropyl alcohol and Freon 113. The thruster MEOP is 400 psia. Proof tested to 600 psia, it has a burst pressure equal to or greater than 1,000 psi.

3.2.1.5 Pyrotechnic Valves (Normally Closed)

Two parallel normally closed pyrovalves isolate the high pressure GHe tank from the regulator inlet. Another two such valves isolate the MMH and NTO sections of the GHe delivery manifold to prevent mixing of MMH and NTO vapors. Figure 3.2-7 shows a cross-sectional view of one of the pyrotechnic valves, which are opened after launch. Dual O-ring seals stop external leaks, individually sealed inlet and outlet lines preclude internal leaks. The valves are actuated by an NSI (see Section 3.3). The valve is considered dual fault tolerant against mechanical failure. The valve-firing circuit (see Section 3.3.1.4) provides dual fault tolerance against inadvertent initiation. The valve MEOP is 4500 psia, proof pressure is 6750 psia and burst pressure is 11250 psia.

During ground operations and prior to separation, low pressure components are isolated from high pressure components by the NC pyrotechnic isolation valve, and two regulator stages in series. Each regulator stage is capable of regulating full helium pressurant tank pressure down to below the MEOP of the bipropellant tanks. The pyrotechnic isolation valve design provides additional fault tolerance by redundantly sealing both the inlet and outlet ports internally. Any pressure increase due to pyrovalve failure or inadvertent pyrovalve initiation can be detected by the low pressure transducers in the bipropellant manifold. The pressurant loading equipment [illegible] per TP-BPLOA-3271152) to perform emergency venting of the system [illegible]. The high pressure GHe subsystem can be vented via the service valves. [illegible] the propellant manifold will measure propellant tank pressure. This will permit verification [illegible] tank MEOP has not been exceeded prior to attempting to vent the system. [illegible] not actuated until after TOS separation. Failure of two regulator stages and [illegible] before the low pressure system MEOP can be exceeded. The proper function and [illegible] propulsion system components and flow control devices is verified by test and the system [illegible] at the launch site prior to fueling per TP-BPLOA-3271152.

JPL

MEDIUM TO HIGH RISK PARTS WAIVERS

- MEDIUM
  - WAIVER 59171; TES CUSTOM HYBRID: PHOTODIODE AND OP AMPL. NO ELEMENT EVALUATION OF PHOTODIODE, AMPLIFIER OR RESISTOR; NO PIND TEST, NO X-RAY INSPECTION
  - WAIVER 59183; MOC ADC 080 MICROCIRCUIT. AL 15 PARTS FAILED LIFE TEST CATASTOPWICALLY DUE TO A POWER OUTAGE AND UNCONTROLLED REAPPLICATION OF POWER. NOT ENOUGH PARTS REMAINED FOR LIFE TEST. NO LONG TERM LOT LIFE DATA
  - WAIVER 59184; MOC DFP16 RESISTOR NETWORK. PARTS WERE NOT SCREENED 50 MIL-R-83401 OR JPL REQUIREMENTS. PARTS CAME FROM THE SAME MANUFACTURING LINE USED TO MAKE MIL-R-83401 PARTS
  - WAIVER 591878; MOC TCD SERIES CAPACITORS. CAPACITOR TYPE WAS NEW 7'0 JPL AND HAD NO ESTABLISHED RELIABILITY STATUS. RESULTS OF MANUFACTURER'S SCREENING AND LOT QUALITY CONFORMANCE TESTING ARE NOT KNOWN

- MEDIUM TO HIGH
  - WAIVERS 59230A, 59231A, 59232A; TEST CUSTOM HYBRIDS. PYROELECTRIC DETECTORS RECEIVED INADEQUATE QUALIFICATION TESTING AND SCREENING. Do NOT MEET THE REQUIREMENTS FOR CLASS B HYBRIDS

Spacecraft Propulsion Loading

- The MMH branch of the bl-propellant subsystem was loaded wet to latch valves 3 & 4. The MMH load was [illegible] lbs. The MMH tank was pressurized to 281 psi.
- The NTO branch of the bipropellant subsystem was loaded wet to latch valves 1 & 2. The NTO load was 1848.5 lbs. The NTO tank was pressurized to 314 psi.
- The GHe tank was pressurized to 4130 psi at 23%.
- All lines downstream of **latch valves** 1 thru 4 were pressurized to 80 psia.

Spacecraft Propulsion Loading

- GHe lines downstream of pyro valves 7 & 8 were pressurized to 4000 psla.
- GHe lines downstream of pyro valves 5 & 6 were pressurized to 265 psia.
- Each half system branch of the monopropellant subsystem was vacuum loaded to the engine valves with 93 lbs of N_2H_4.
- Each monopropellant half system was pressurized to 360 psia.
- After loading was completed monopropellant latch valves 9 & 10 were closed.
- Final closeout operations torqued and lock wired all service valves, service valve inner and outer seal caps.

(4)

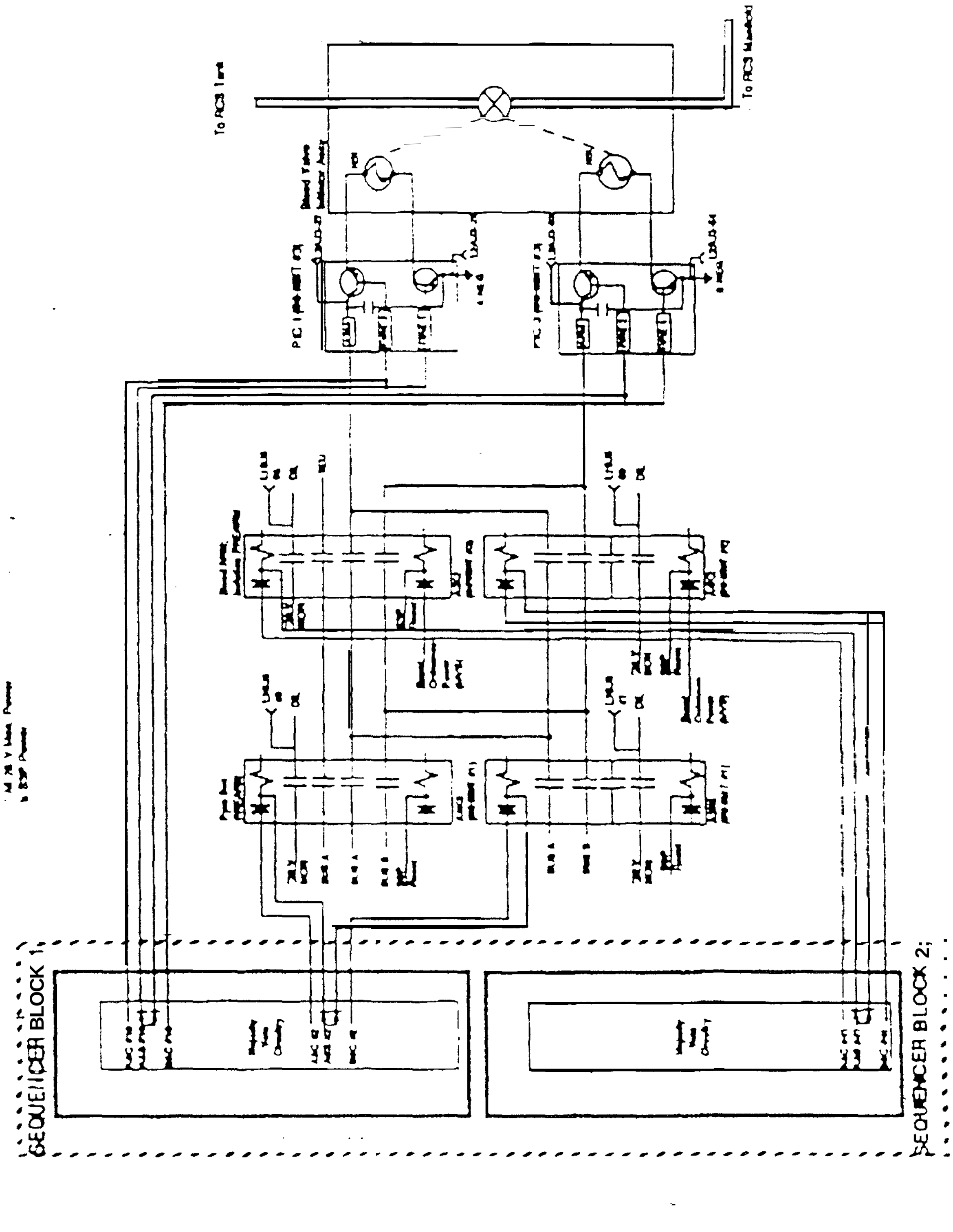


FIGURE 4.3-1 FIRING CIRCUIT DIAGRAM, RCS BLEED VALVE

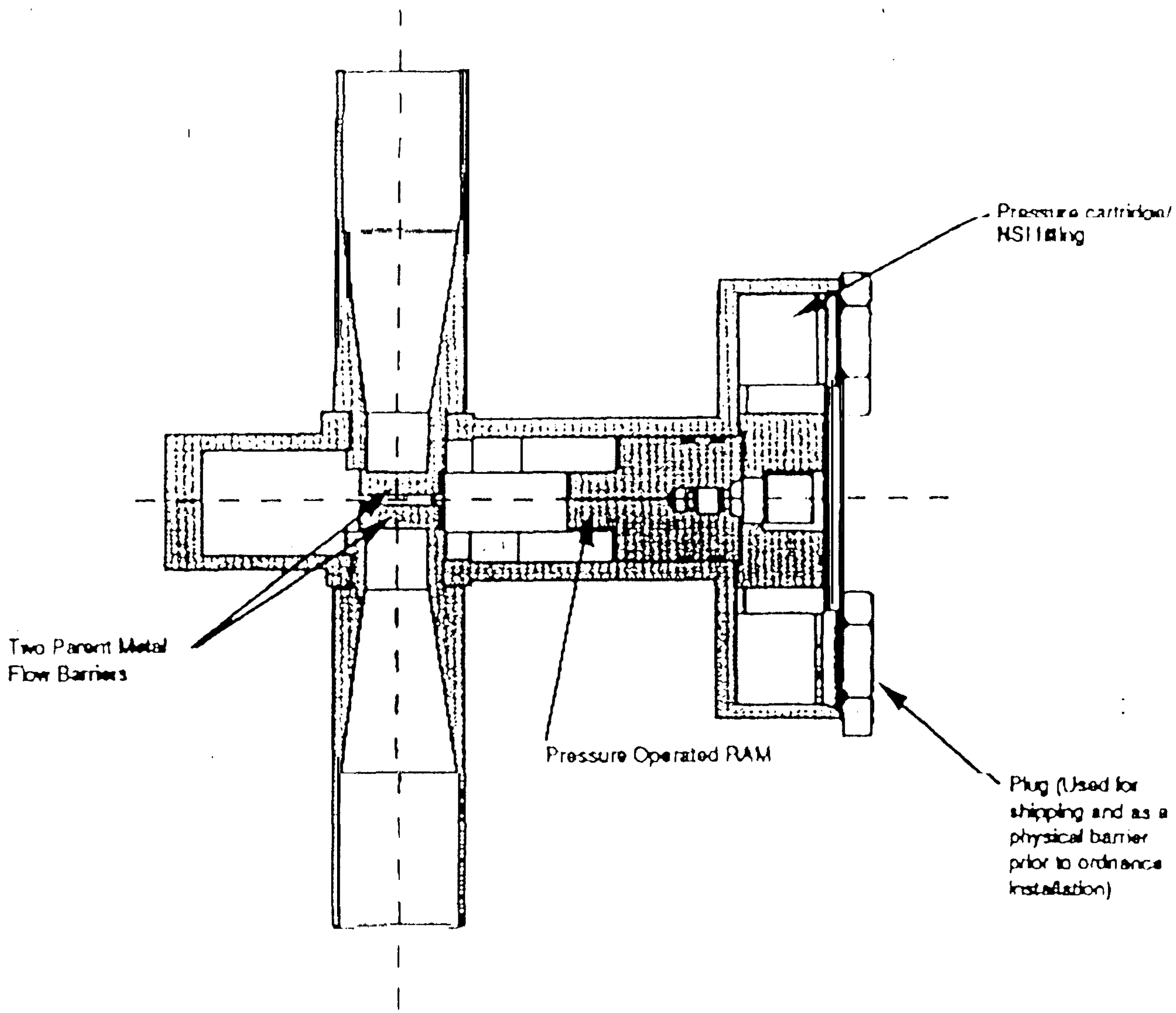


FIGURE 7 PYRO OPERATED 3/4 INCH VALVE

Enclosure 2

NASA
National Aeronautics and Space Administration

Facsimile Transmission Cover Sheet

(1)

FROM

NAME AND TITLE		INSTALLATION	OFFICE CODE
Rich Gladwin		MSFC	CT21
TELEPHONE NUMBER	FAX NUMBER	TOTAL PAGES (including cover sheet)	DATE
205 544 9407	205 544 4155	4	9-3-93

TO

NAME AND TITLE		INSTALLATION	OFFICE CODE
Mowinchini		HQ	
TELEPHONE NUMBER	FAX NUMBER		
202 358 0573	— 2776		

REMARKS/ADDITIONAL ADDRESSES

TOS uses pyro isolation valves made by Pyronetics. As shown on page (2), part numbers are 1420 and 1430. The redundant NSI's are fired simultaineously and the valves were tested this way. There shouldn't be any problems with structural integrity

Call me if you have any questions.

WARNING
Most FAX machines produce copies on thermal paper. The image produced is highly unstable and will deteriorate significantly in a few years. It must be copied prior to filing as a record.

Table I

Normally Open Valves

Part Number	Size (in)	Material	Program	Temperature
1466	.375	Titanium	L-SAT	same as 1467
1466-5	.375	Titanium	KU3, KU4, ANIK-E	see attached
1466-11	.25	Titanium	UHF(F1-F9), HS-601(NF), AUSSAT, BRASILSAT	single -10°F dual +150°F
1466-14	.25	Titanium	UHF(F1-F9), HS-601(NF), AUSSAT, BRASILSAT	single -10°F dual +150°F
1466-15	.375	Titanium	UHF(F1-F9), AUSSAT	single -10-F dual +150°F
1466-16	.375	Titanium	TELSTAR	same as 1466-5

Table III

Normally Closed Valves

Part Number	Size (in)	Material	Program	Temperature
1467 1467-1	3/8	Titanium	Olympus	-76°F, single 80% +170°F, dual 130%
1467-5	3/8	Titanium	KU3/KU4, ANIK-E, TELSTAR	+71°C (+160°F) +25°C (+77°F) -34°C (-29-F)
1467-6, -7, -8	3/8	Titanium	SABIR-FE	ambient, ambient, no data
1467-9	3/8	Titanium	DSCS-LABS, GBI (LMSC/Aerojet)	+25°C (+77°F) +71°C (+160°F) -34°C (-29°F)
1467-10, 1467-11	1/4	Titanium	UHF(F1), UHF(F2-11 F9) HS 601(HF), AUSSAT, BRASILSAT	single -30°C dual +171°C single -10°F dual +150°F
1467-14	1/8	Titanium	Miniature Propulsion Subsystem, GBI (LMSC/Aerojet)	ambient
1467-15	3/8	Titanium	UHF(F1), UHF(F2-F9)	single -10°F dual +150°F
1467-16	1/4	Titanium	Miniature Propulsion System	ambient
1467-17	1/8 (orificed)	Titanium	Miniature Propulsion System, GBI, (LMSC/Aerojet)	ambient
1467-18	3/16	Titanium	SCIT	ambient
1467-19	3/8	Titanium	Mars Observer	see attached
1467-20	3/8	Titanium	Mars Observer	see attached
1467-21	3/16 Vent	Titanium	SCIT	ambient
1467-22	3/8	Titanium	TELSTAR	+71°C (+160°F) +25°C (+77°F) -34°C (-29°F)
1467-23	1/4	Titanium	Cluster Gas-Valves	+122°F -22°F +68°F
1467-24	3/8	Titanium	Cluster Liquid-Valves	+122°F +32°F +68°F
1467-28	3/16	Titanium	ALAS	?
1467-29	1/4	Titanium	ALAS	ambient
1467-30	3/16	Titanium	ALAS	ambient
1467-31	3/16 (orificed)	Titanium	ALAS	ambient

SLt 6

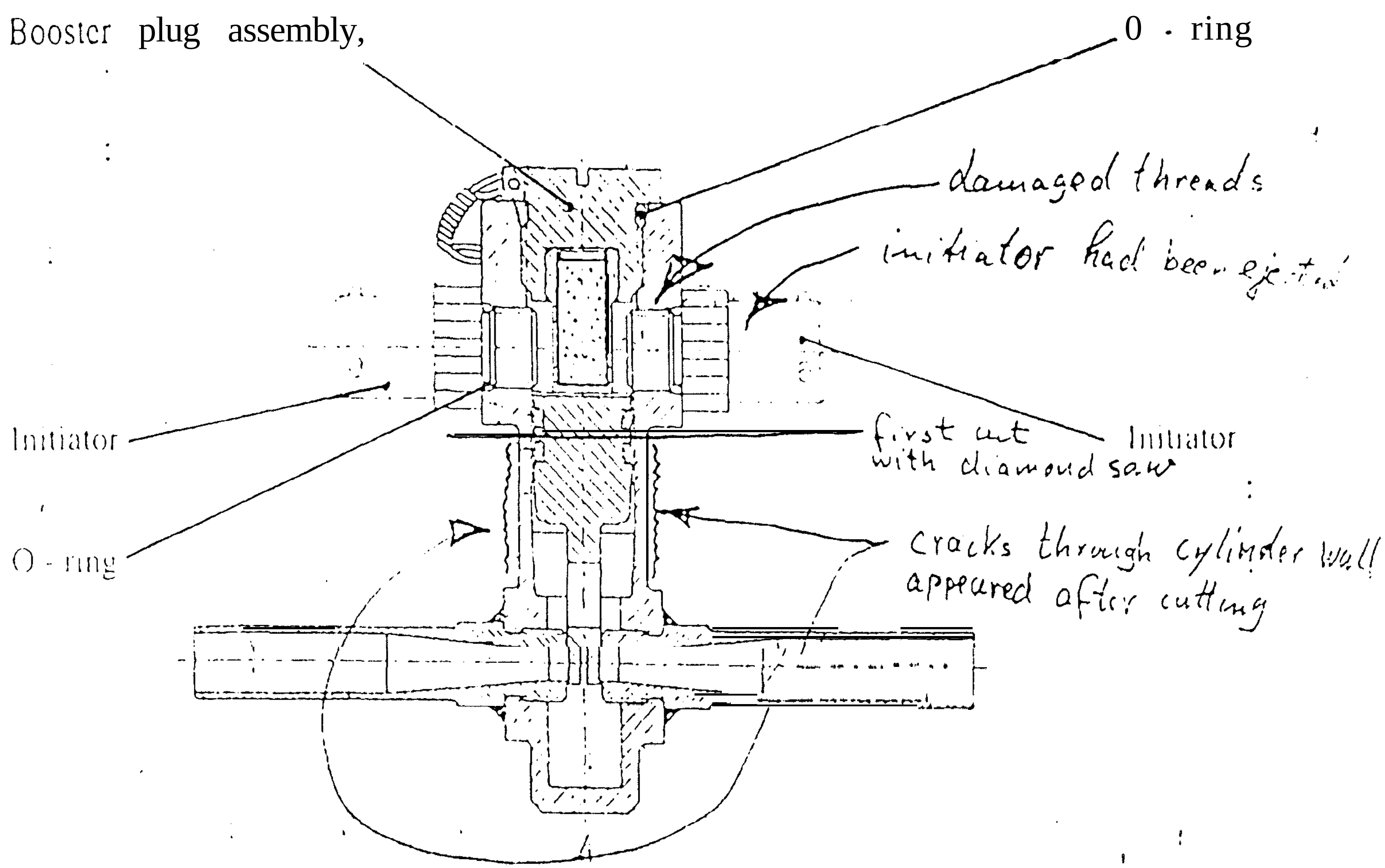
FIRING HISTORY OF 3/8" ALL-TITANIUM PYROVALVES WITH 325 MG HI-TEMP BOOSTER CHARGE

PART NUMBER	QUANTITY DELIVERED	QUANTITY FIRED BY OEA	FIRING CONDITIONS (TEMPERATURE, OVER/UNDER CHARGE, ETC.)
1467*	---	6	Single Initiator: -60°C/80%, +77°C/100%, ambient/80% Dual initiator: -60°C/130%, +77°C/130%, ambient/100%
1467-5	34	18	71°C, 25°C, -34°C
1467-9	65	32	71°C, 25°C, -34°C
1467-15	21	3	Dual initiator: 66°C, Single initiator: -23°C
1467-19	26	10	-45°C/80% & 120%, +75°C/80% & 120%. +25°C/100%/Dual
1467-20	4	---	---
1467-22	6	---	---
1467-24	9	3	Single initiator: 0°C/80%, 22°C/100% Dual initiator: 50°C/130%
1467-37	24	4	71°C, 24°C, -34°C
1467-38	12	---	71°C, 24°C, -34°C
Totals	201	76	
1466**	---	6	Single initiator: -60°C/80%, +77°C/100%, ambient/80% Dual initiator: -60°C/130%, +77°C/130%, ambient/100%
1466-5	20	18	71°C, 25°C, -34°C
1466-15	40	6	Dual initiator: 66°C, Single initiator: [illegible]
1466-16	3	---	---
1466-18	12	3	71°C, 24°C, -34°C
1466-19	22	---	---
Totals	97	33	

* see 10-1467
** see 10-1466

Liquid pyro valve design

1467 - 24 Cluster flight models
1467 - 15 SM valves (residuals from Hughs order)



2/5

ENCLOSURE 2

esa estec

european space research and technology centre

CLUSTER PROJECT OFFICE

DATAFAX

Ref : PK/7210/JB/aen/820 Date : 3 September 1993

From : J Bruggemann (PKQ)
Phone : +31-1719 844 31
Fax # : +31-1719-46280

To : Aldo Dominicini
NASA HQ, Washington D.C. - USA
Fax : +1 202- 358 2776

To : Joe Wonsever
NASA H.Q., Washington D.C. - USA
fax : +1-202-358 2778

Subject: CLUSTER Failure of Pyro-valves

Subsequent to the phone conversation today, I am providing you the following information (as a first step) on the failure of the CLUSTER pyro-valves.

1. The contacts in OEA/Pyrotecnics are:

   D I Fredericks (General Manager,)
   Dean Grages (Design Engineer)
   OEA/Pyronetics, Denver, Colorado - USA
   fax: 33-699 6991

   Normally we deal with OEA via our contractors Dornier in Germany and British Aerospace, Bristol, England.

2. Attached is a copy of a drawing of the pyro-valve (liquid valve) being used for Cluster, with some indication on the deficiencies which we have seen so far further details can be provided next week.

3. Attached are also two tables (3 pages) which we received from OEA and which include types which are similar to the CLUSTER valves and which had been tested under various conditions.

Kind regards,

J. Bruggemann

J. Bruggemann

Attachment

esa ESTEC/CLUSTER esa
Fax No. Page 1/5
Time
Date 3/9/93

a) During the test article firing of the initiators, the ESA design requires a 15 millisecond between each firing. NASA designs commonly fire both initiators in parallel. However, in the tests performed by ESA the ignition mechanism is triggered to only one initiator the other fires in sympathy.

b) Both initiators fired and the first initiator was expelled, causing damage to the test article. However, the valve operated and opened.

c) In the test of the -15 valves ESA experienced similar problems.

d) ESA has theorized that the initial firing of the first initiator caused damage to the threads which then failed when the second initiator fired. *Cross* sectioning of the threads show a sintered conglomerate of titanium and nickel at the threads. It appears that the titanium is being eroded by the hot gases.

e) ESA uses an OEA initiator, (P/N 4704100) which is supposed to be equivalent to the NASA Standard Initiators (NSI).

f) ESA valves are made of titanium and use the 4704100 initiators which have a wider variation in dynamic output than do the NSIs. NASA valves used on the Mars Observer (MO) were titanium valves with NSIs. The valves used on TOS were made from stainless steel and used NSIs. See enclosure 2 for comments from MSFC/SR&QA.

g) OEA tested two -15 valves with the 4704100, one valve expelled the initiator the other experienced a blow by, however all valves opened. In another test at OEA the valve was tested with brackets pressing against the ignitiator housings. This valve fired successfully.

Enclosures: (2)

cc: QR/C Schneider
QT/ J Wonsever

MEMO

NASA HEADQUARTERS/QR

DATE: September 7, 1993

TO: QT/ M Greenfield, Director, Payloads Division

FROM: QR/ A Domenichini Jr.

SUBJECT: Telecon w/ ESA, Results of investigation of Pyro-valve Failure.

Points of Contact: NASA/S A Diaz-358-1413
NASA/ B C LAM 358-2332
ESA/ESTEC J Bruggemann 31-1719 844 31
OEA D Franklin/D Gregas Program Engineering (303)693-1248 x417,
MSFC R Gladwin (205)544-4155

Background: Telecon from Mr Dave Dale of ESTEC/ESA to Mr Al Diaz, NASA/S informed NASA of problems experienced by ESA with OEA/Pyronetics pyre-valves. Pyro-valves are used in the Mars Observer, a satellite which has already been launched and TOS a system due to be launched during STS-51 mission.

Discussion: A conference call was setup with J Wonsever(TDY at LERC) Mr J Bruggemann(ESA/ESTEC/PKQ) and NASA Headquarters(QR) for September 3, 1993. Mr Bruggemann is responsible for the analysis of the pyro-valve failure and determining the causes. See enclosure 1 for depiction of ESA valve and comments on failure.

The failure occurred to a pyro-valve(liquid valve) for the CLUSTER program during testing. The valve, p/n 1467-24, supplied by OEA, was qualified using similarity. The failure was catastrophic and caused damage to the test article.

The investigation which followed the incident used OEA, p/n 1467-15, pyro-valves since there were no valves available from the lot used on the CLUSTER test article. Valve, p/n 1467-15, is similar to the -24 except the inlet fuel value is a smaller diameter, however the value housing is the same size and made from same material, TI6ALV. The valves are from a lot produced for Hughes Inc. During the investigation four pyro-valves were tested, three demonstrated the same 'results of the -24. The last valve was braced by steel plates on either side and no failure occurred.

Observations made concerning the failed valves revealed the following:

Electro Explosive Device System Testing

MARTIN MARIETTA

MARTIN MARIETTA ASTRO SPACE

- System Level EED Testing Consisted of the Following Steps:

1) <u>Verification of Ordnance Harness Output Via Test Rack</u>

Outputs of Ordnance Harness Verified using Test Rack Connected to all EED inputs simultaneously. Each Individual Ordnance output was verified to be Enabled, Armed, and Fired, and that no other EED input was stimulated.

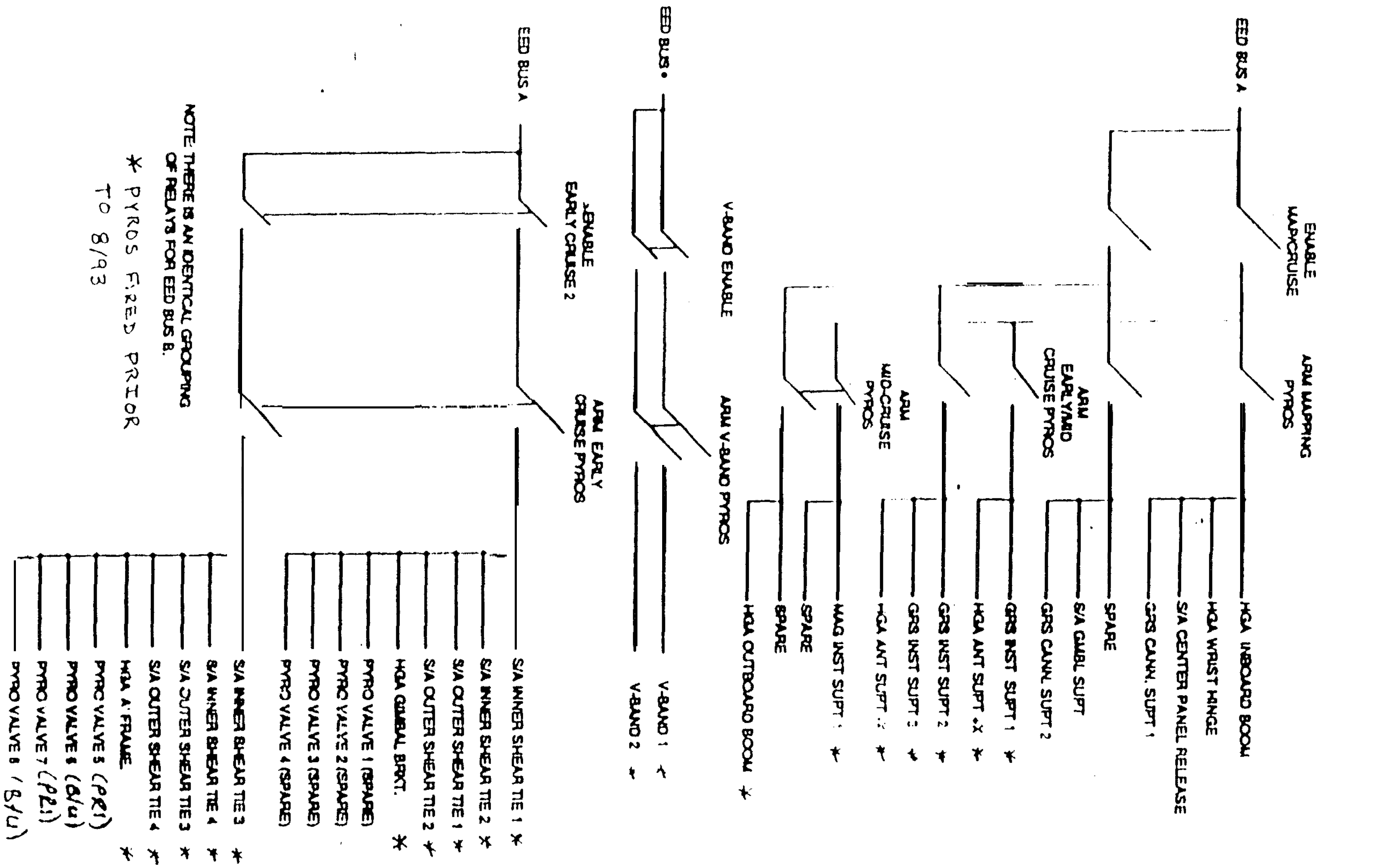
2) <u>Verification of EED Firings using Live EEDs</u>

During Deployment testing, Pyros were fired using flight sequences loaded into the SCP memory.

Pyro Bus Enable and Arm System Diagram (Primary Side Shown)

MARTIN MARIETTA

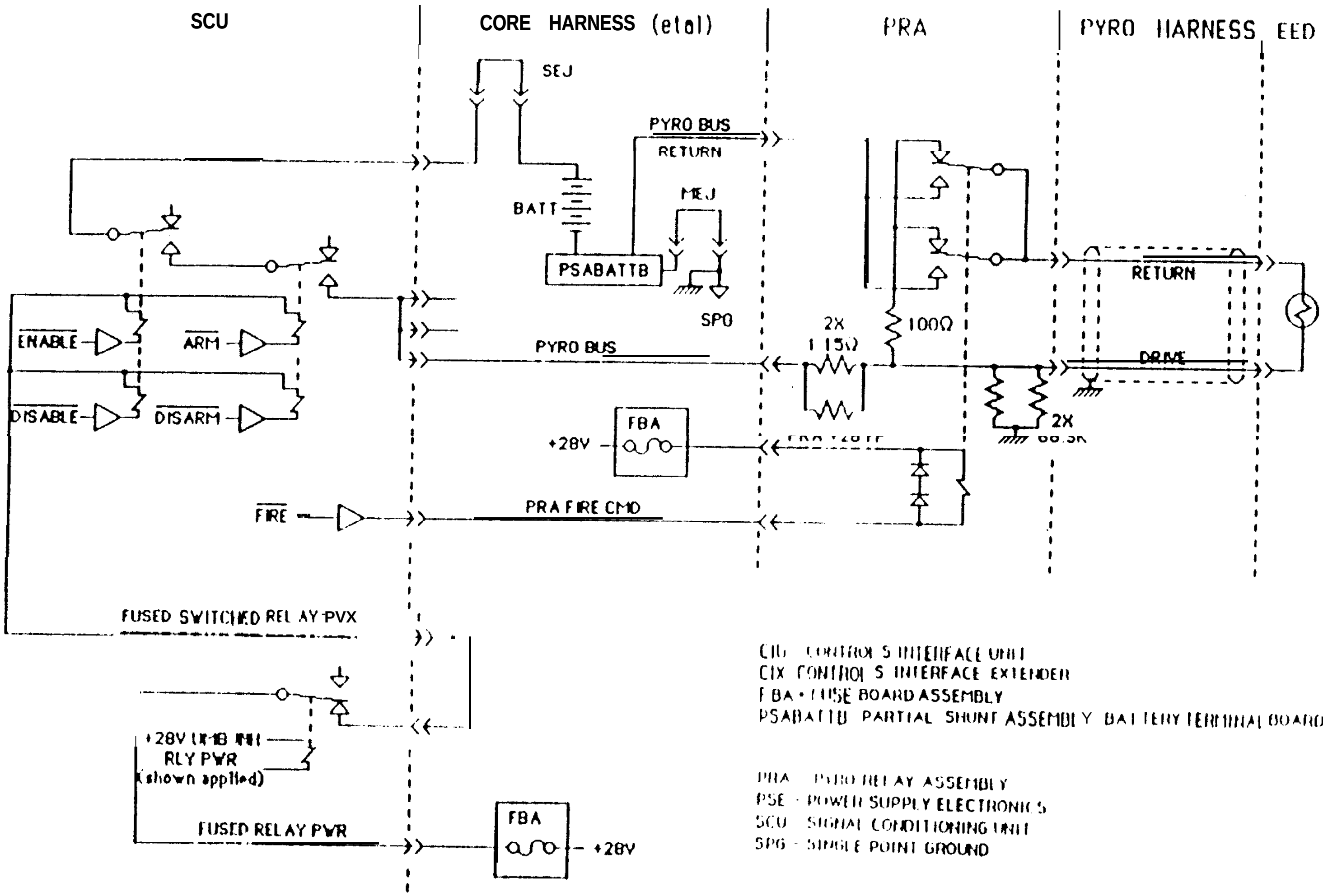
MARTIN MARIETTA ASTRO SPACE



Electro Explosive Device (EED) System Diagram (Primary Side Shown)

MARTIN MARIETTA

MARTIN MARIETTA ASTRO SPACE



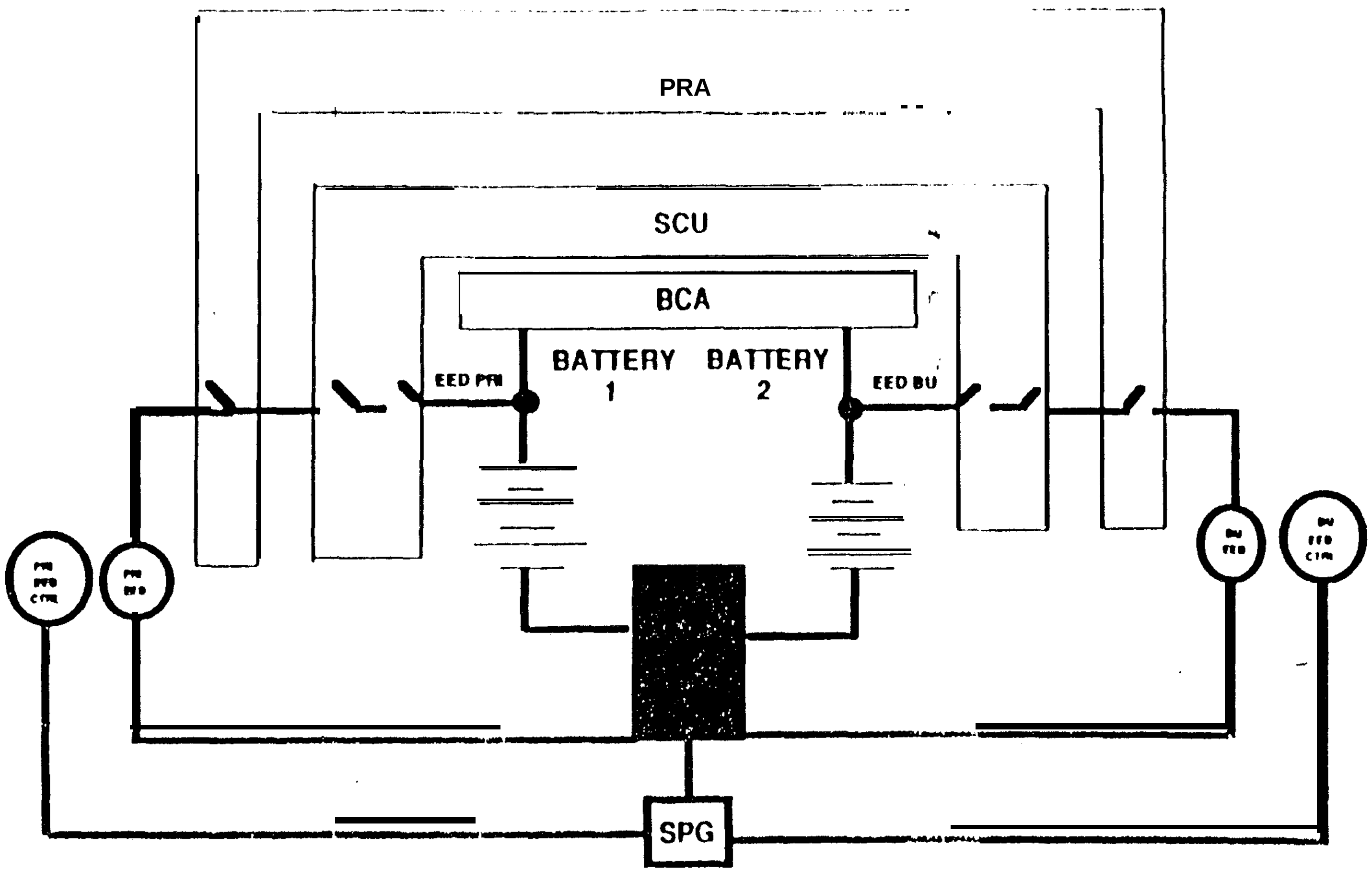
CIU - CONTROLS INTERFACE UNIT
CIX - CONTROLS INTERFACE EXTENDER
FBA - FUSE BOARD ASSEMBLY
PSABATTB - PARTIAL SHUNT ASSEMBLY BATTERY TERMINAL BOARD

PRA - PYRO RELAY ASSEMBLY
PSE - POWER SUPPLY ELECTRONICS
SCU - SIGNAL CONDITIONING UNIT
SPG - SINGLE POINT GROUND

Electro Explosive Device (EED) Bus Configuration

MARTIN MARIETTA

MARTIN MARIETTA ASTRO SPACE



- PRIMARY BUS - ONE BATTERY
- BACK-UP BUS - OTHER BATTERY
- REDUNDANT EEDS ON SEPARATE BUSES
- EED RETURNS TO BATTERY NEGATIVE
- EED CONTROL RETURNS TO SPG
- BUS VOLTAGE: 18.5V TO 25.5V

ATT

MARTIN MARIETTA

MARTIN MARIETTA ASTRO SPACE

Pyrotechnic Electrical System

Dan Hoff man
9/8/93

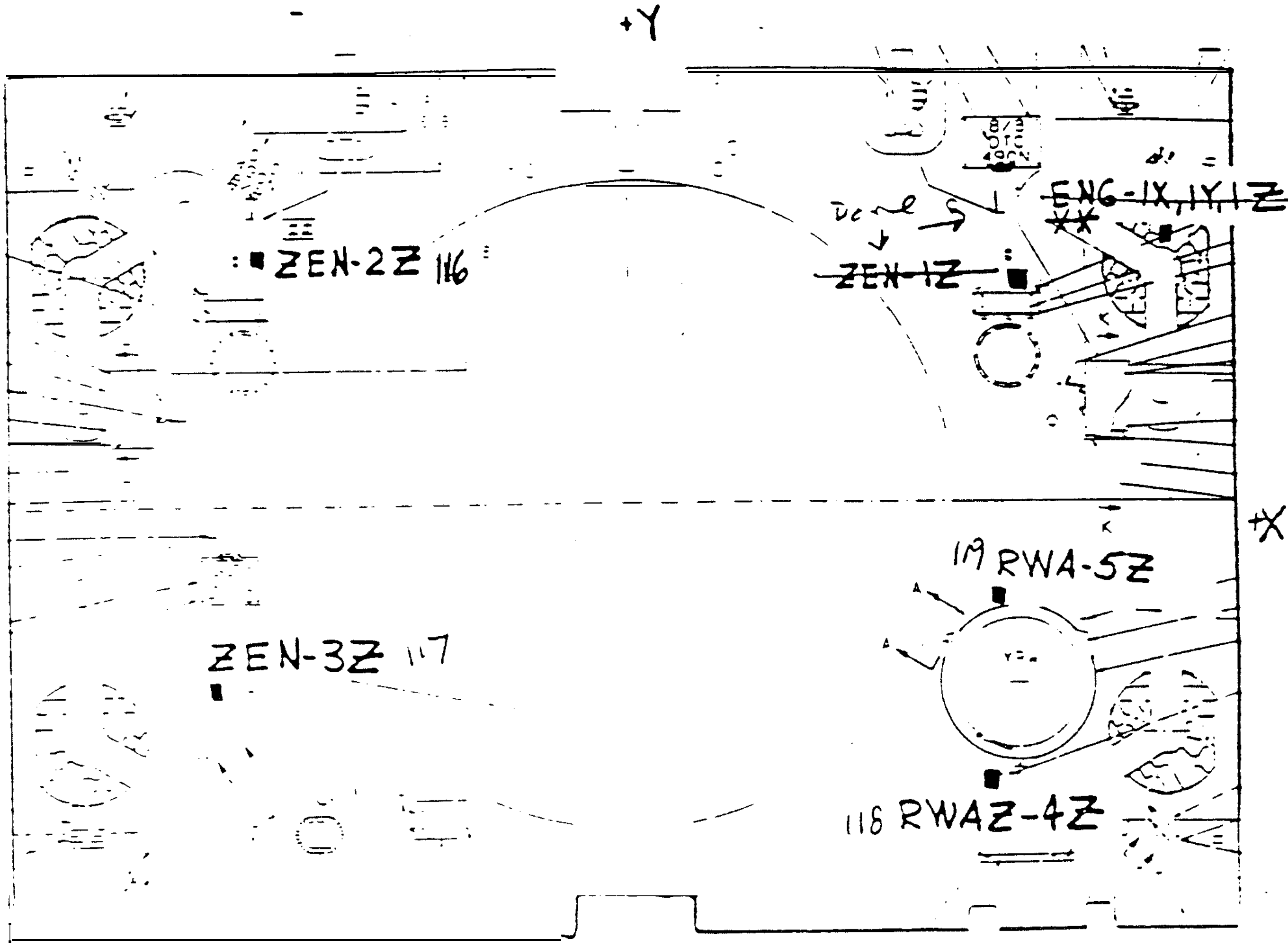
MRB00166

RXO LOCATION

AVEL-SUN-1 Y

Shock Response Spectrum (G's)

Pyrovalve Shock Test	4.6	9110
Solar Arrav Shock	64.3	2550
HGA Pyroshock	11.7	5000
MAG Pyroshock	6.5	3830
Separation Shock	51	3040
GRS Shock	12.9	3220



** - THESE ACCELS WILL BE MOUNTED ON THE ENGINE MOUNTING FLANGE FROM THE OUTBOARD SIDE OF THE ZENITH PANEL.

INSIDE

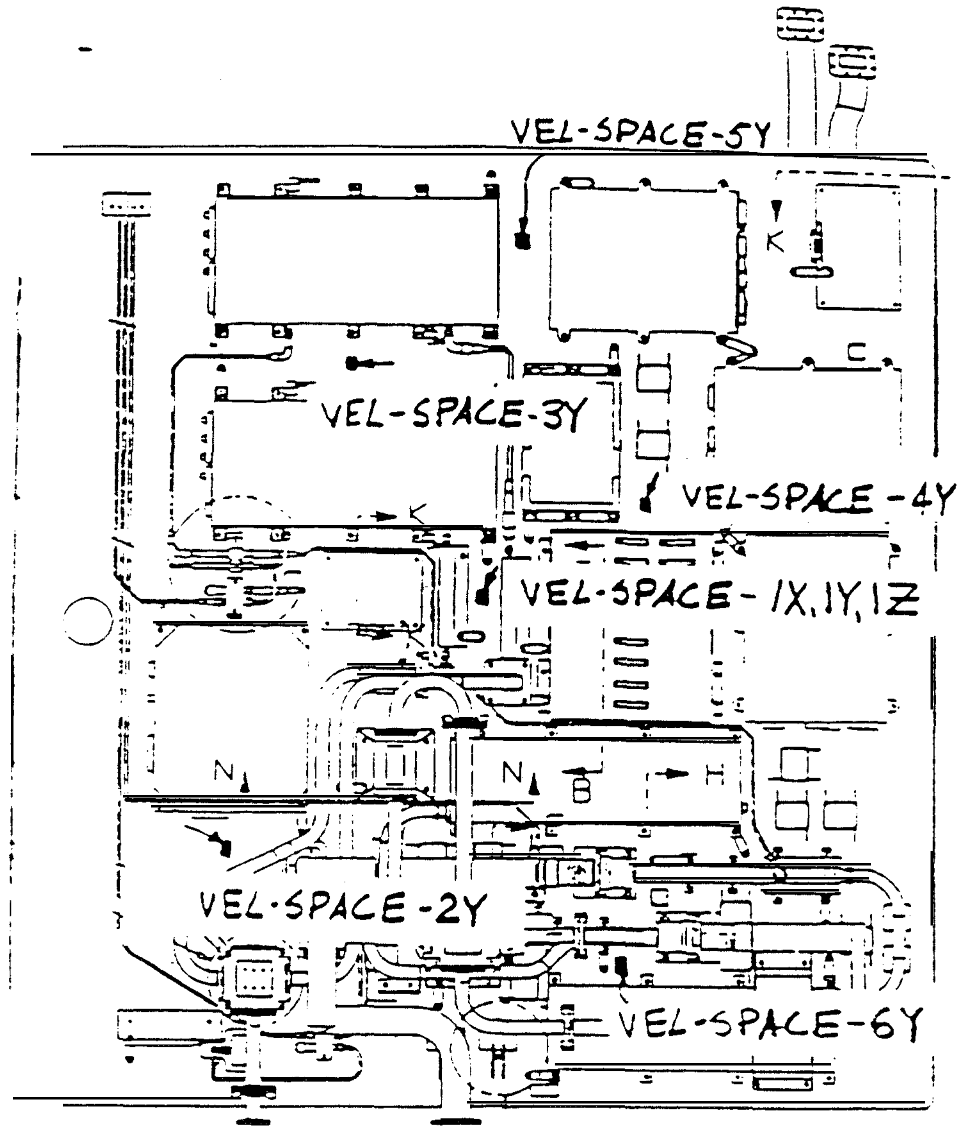
FIGURE 17 - ZENITH PANEL

Done

I.D. 0128M

Size	Code Ident No.	
A	49671	TP-AI-3271152
		Sheet 29

Date 8/8/91



Done

FIGURE11 . VELOCITY SPACE PANEL.

I.D. 0128M

Size	Code Ident No.	
A	49671	TP-AI-3271152

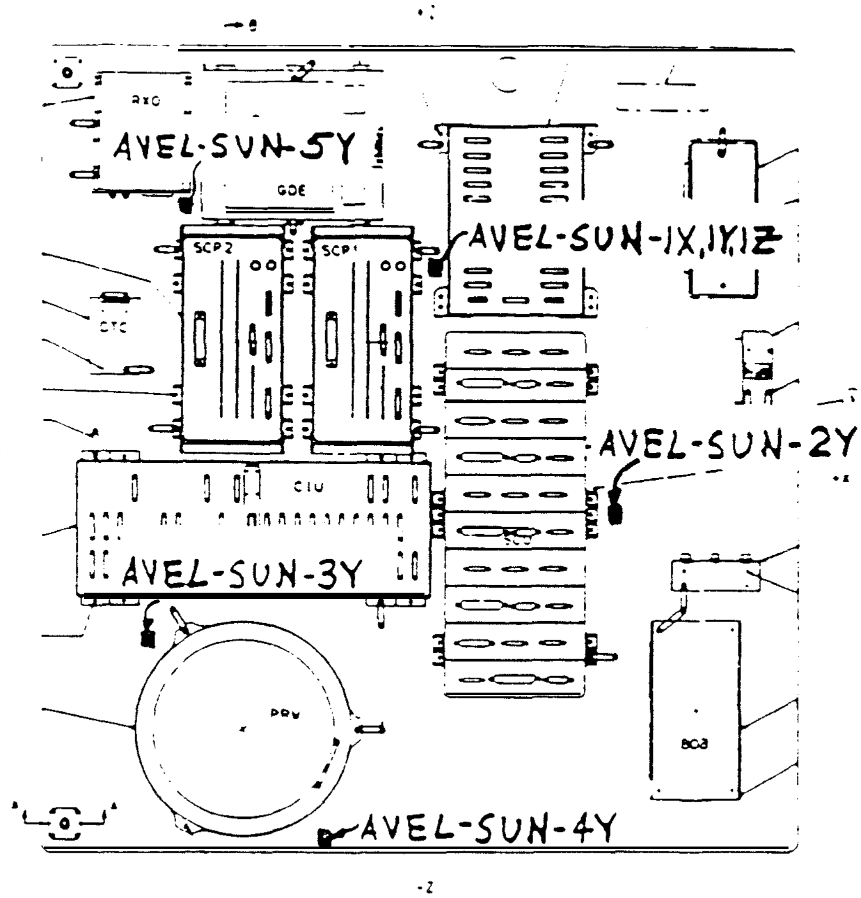


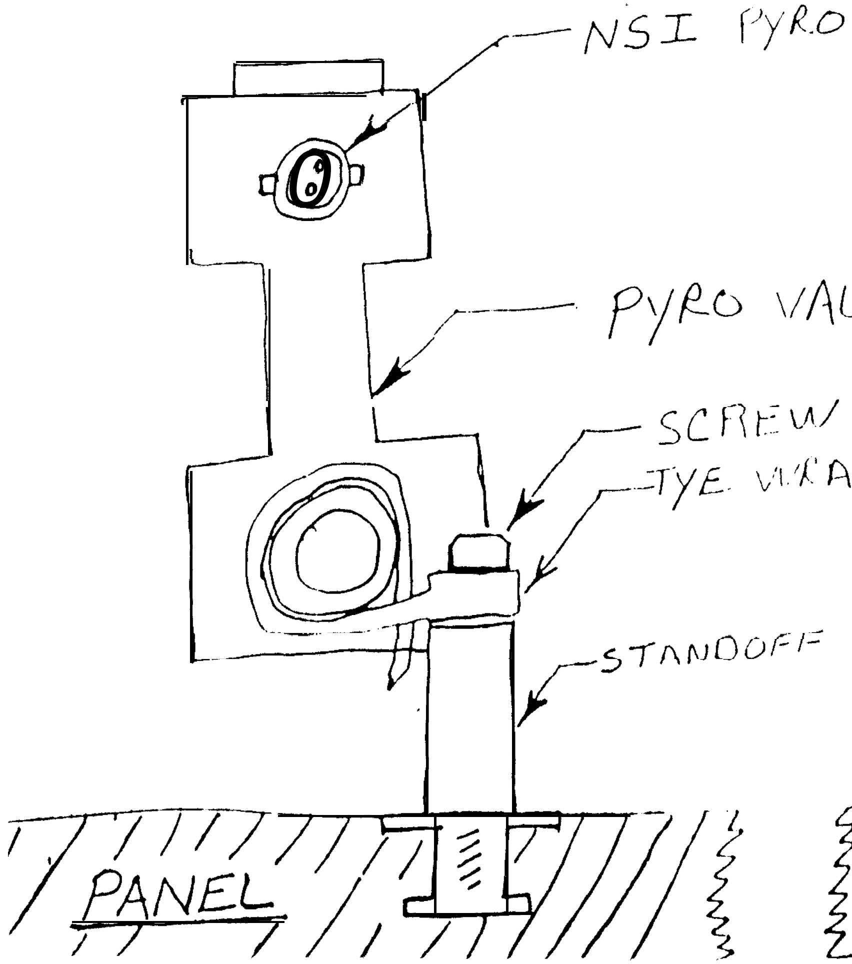
FIGURE 14 - ANTI-VELOCITY SUN PANEL

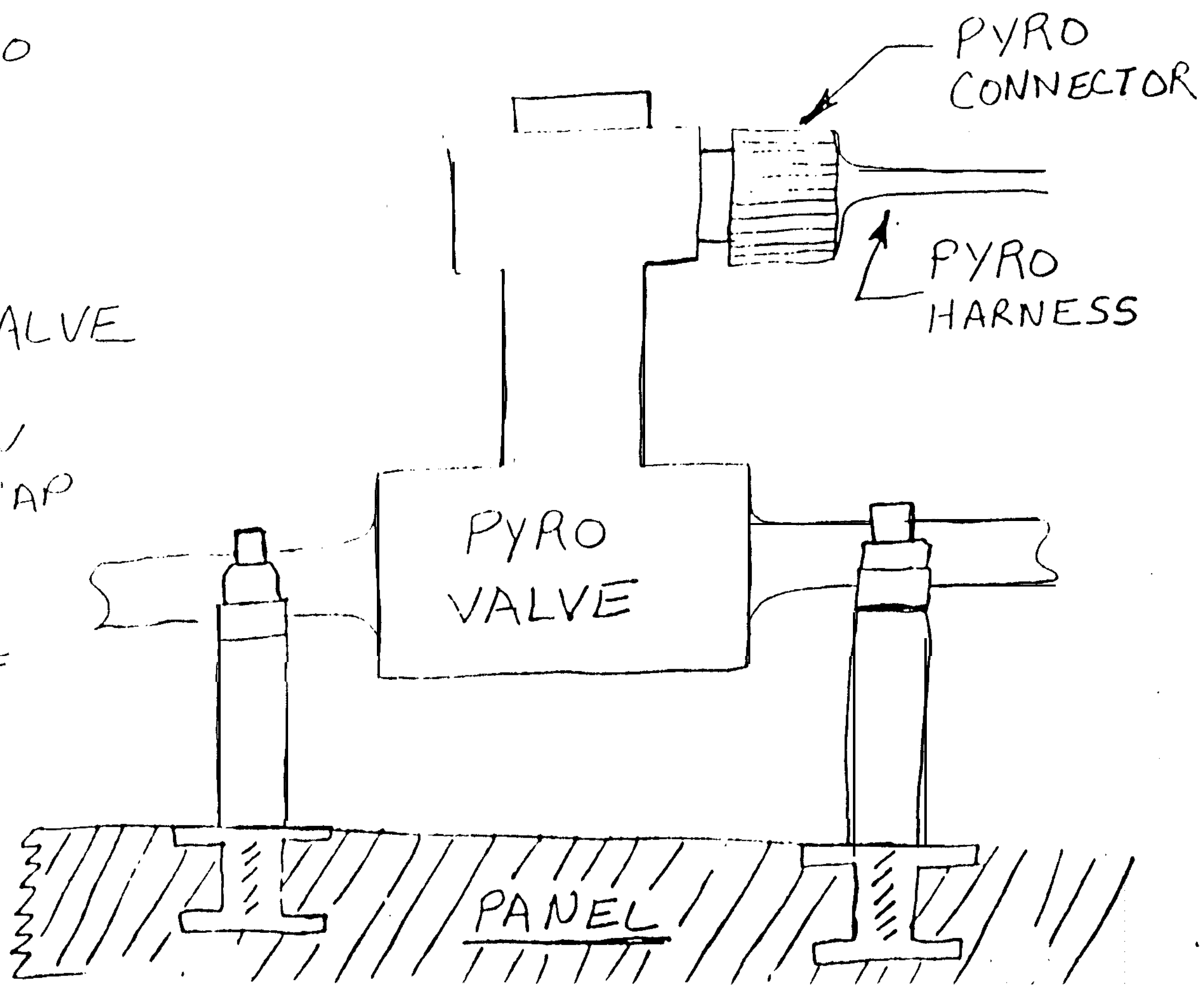
I.D. 0128M

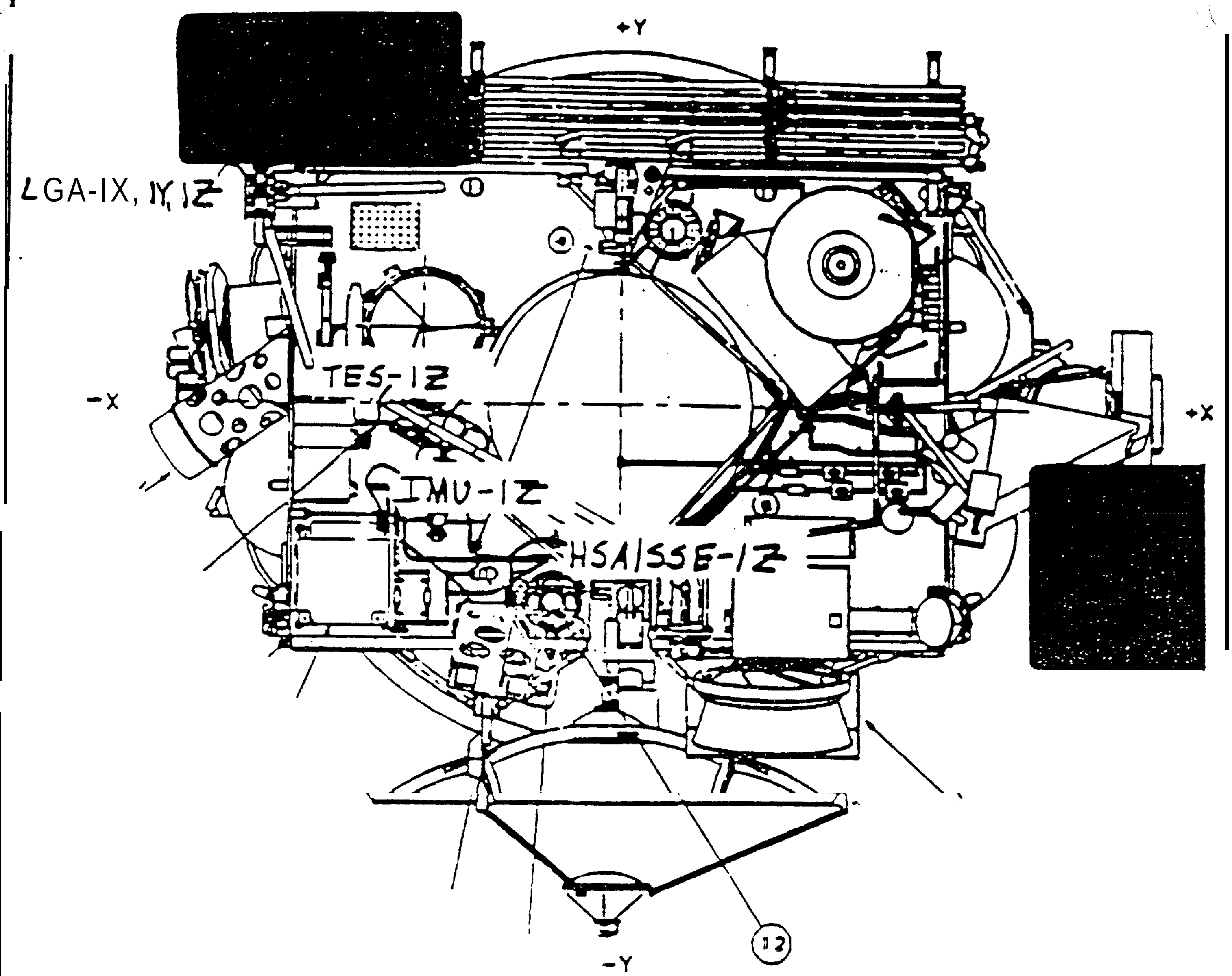
Size	Code Ident No.	
A	49671	TP-AI-3271152

G FAUDION
9/1/93

PYRO VALVE
MOUNTING CONFIGURATION







[redacted]

OUTSIDE

FIGURE 5 - NADIR PANEL - TES,IMU,HSA/SSA,LGA

I.D. 012811

Size	Code Ident No.	
A	49671	TP-AI-3271152
		Sheet 17

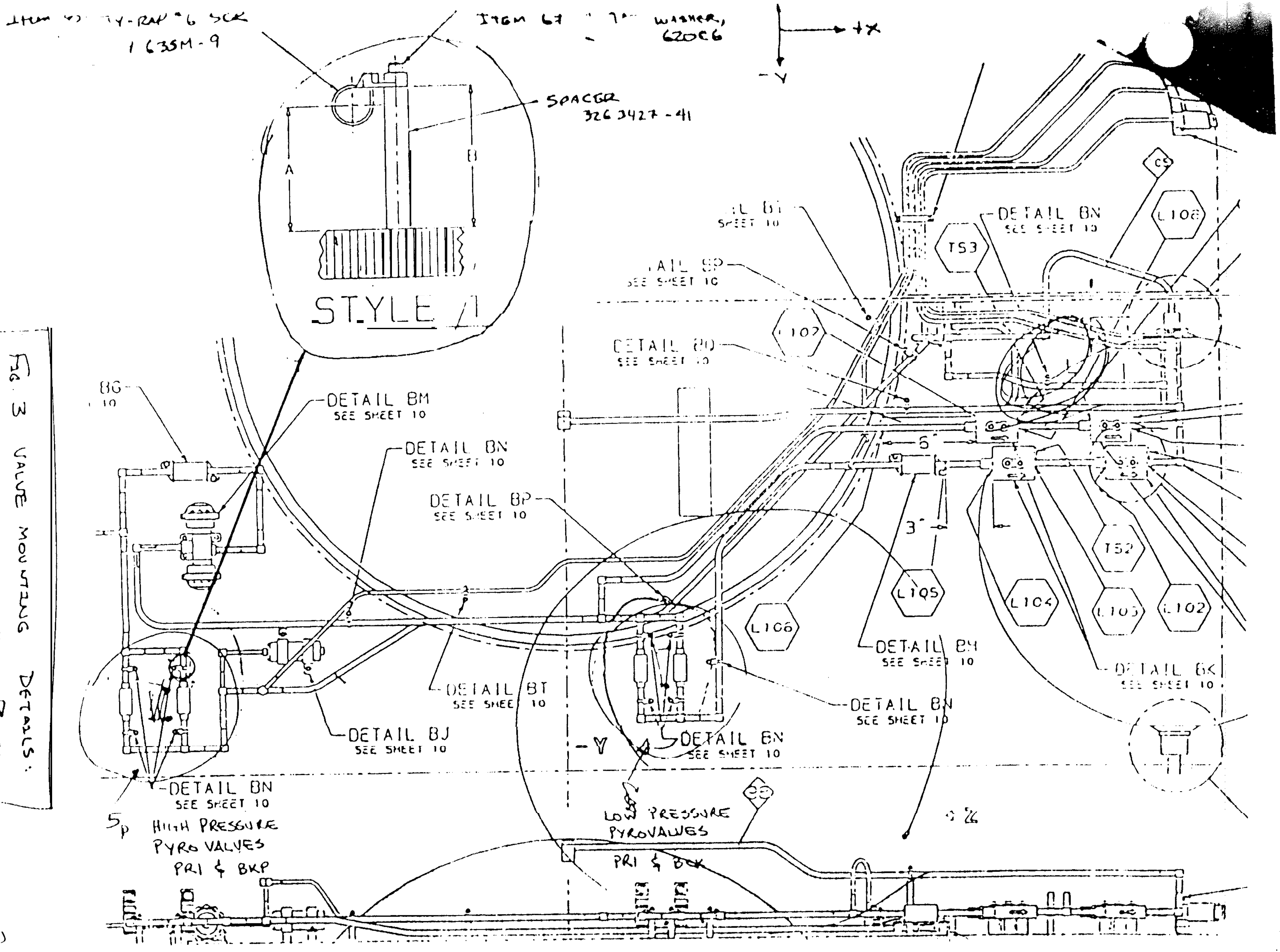
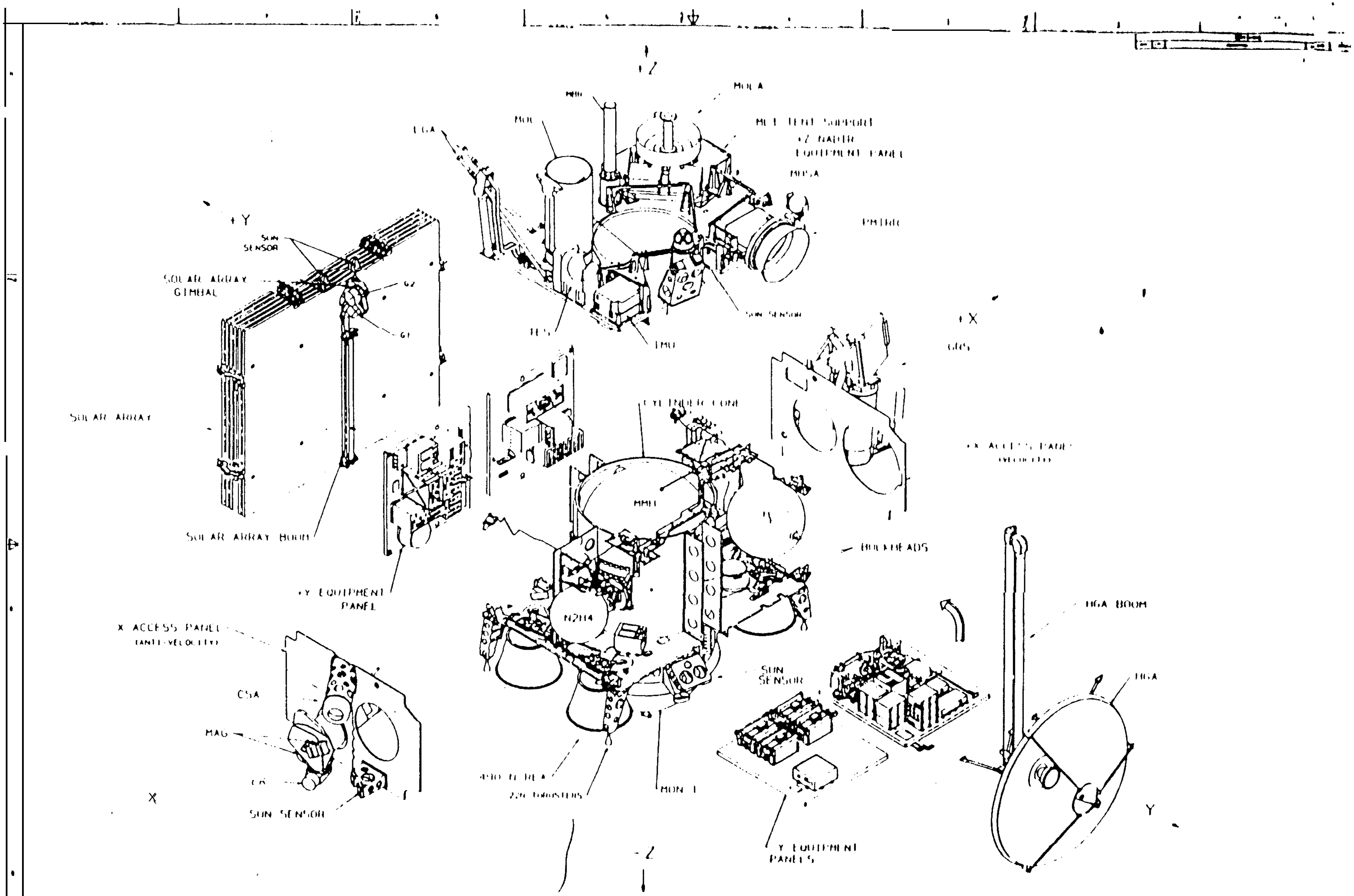


FIG. 3 VALVE MOUNTING DETAILS: OUTBOARD NADIR PANEL

DRAWING 3267342



GE Astro Space

ODED VI

OBSER

Pyro Shock Data

PYROVALVE

MARTIN MARIETTA

MARTIN MARIETTA ASTRO SPACE

Accei Location	Accel Name	Peak Level @ Freq (SRS)
Nadir Panel	STR-NAD-2Y	24.22g @ 50 hz*
	TES-1Z	1604g @ 4058 hz
Center Cylinder	RWA-CY L-l Y	1.7g @ 3830 hz
Velocity-Space Panel	VEL-SPACE-1 Y	3.1g @ 9110 hz
Anti-Vebcity-Space Panel	AVEL-SPACE-1 Y	2.3g @ 1520 hz
Velocity-Sun Panel	VEL-SUN-1 Y	0.52g @ 9110 hz
Anti-Velocity-Sun Panel	AVEL-SUN-l Y	4.7g @ 9110 hz

RF/TWTA Vel. Space Panel

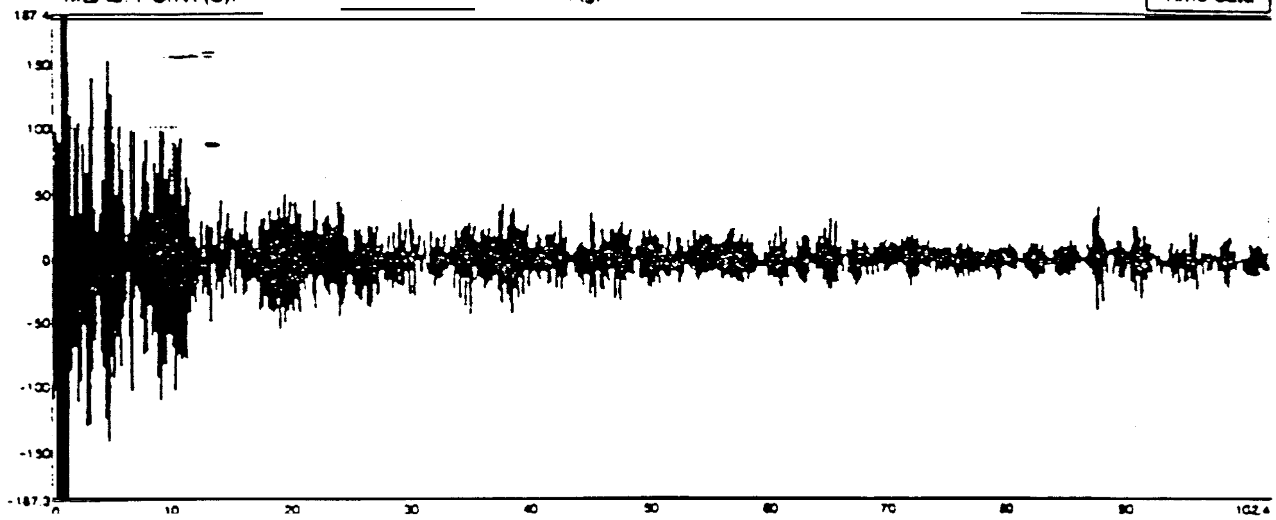
MARTIN MARIETTA

MARTIN MARIETTA ASTRO SPACE

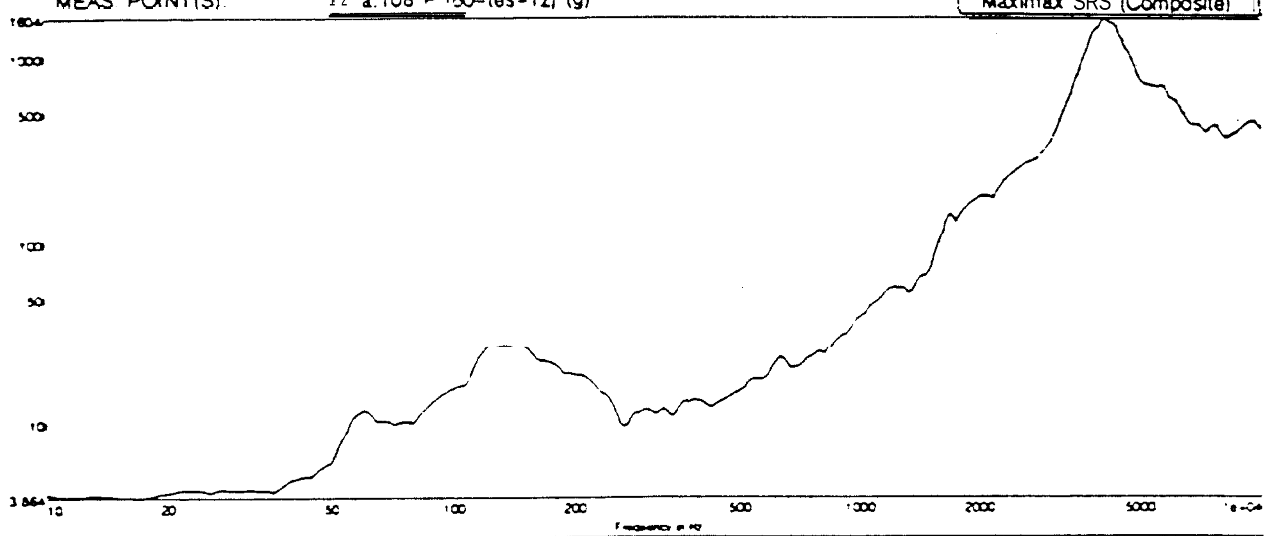
Sig. Sources: HGA Gimbal, HGA **Boom, Sep. Shock,** Pyrovalve

EVENT	ACCEL	SRGS (g)	FREQ (Hz)
HGA Gimbal #1	A Vel Space 1 Y	362.6	1706
	" " 3Y	-1 mu--	7231 - N.G.
	VEL. SP. 5Y	62.6	4555
	VEL. SP. 6.1	10.6	6080
HGA Boom #1	Vel Space 1 Y	29.3	3030
	5Y	12.9	4826
	6Y	1 a.4	8116
HGA Boom #3	Vel Space 1Y	54.6	3830
	5Y	58.3	4555
	6Y	29.9	4299
Pyro valve	Vel Space 1Y	2.7	9652
#1	Vel Space 1Y	3.5	9110

MEAS. POINT(S): 12: ● :loa: P (60-tes-1z) (g) | Time data



MEAS. POINT(S) 12 a:108 P (60-tes-1z) (g) | Maximax SRS (Composite)



MEAS. POINT(S): 12 a:108 P (60-tes-1z) g Damping=5 % (Reso: =1/12)

Maximax SRS (Composite) values table

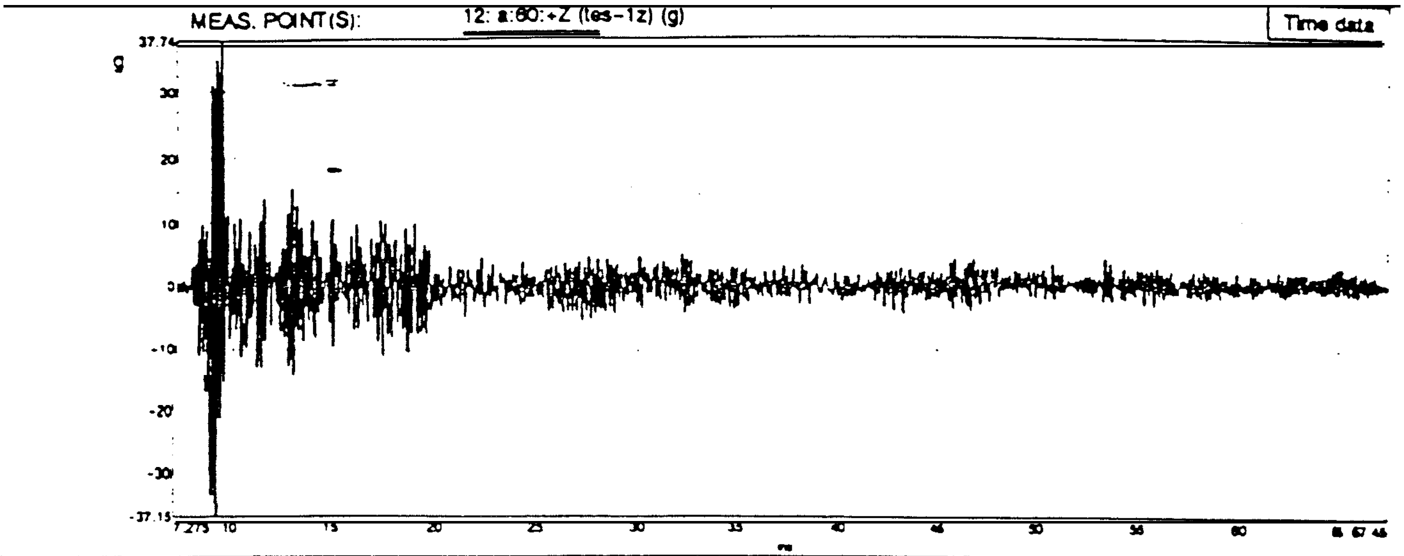
Freq	Level	Freq	Level	Freq	Level	Freq	Level	Freq	Level
9.988	4.060	39.95	4.876	159.8	21.86	639.1	22.41	2556	[illegible]
10.54	4.024	42.32	5.080	169.3	21.29	677.1	19.6	2708	273.8
11.21	3.876	44.84	5.181	179.3	20.13	717.4	20.91	2870	326.8
11.88	4	47.5	5.767	190	18.32	760.1	23.81	3040	386.3
12.58	4.080	50.33	6.301	201.3	18.17	805.2	25.38	3221	500.1
13.33	4.032	53.32	8.128	213.3	17.22	853.1	27.46	3413	686.8
14.12	4.013	56.48	10.42	226	16.1	903.8	31.2	3615	888.9
14.96	3.965	59.85	11.84	239.4	13.57	957.6	35.2	3830	1370
15.85	3.881	63.41	10.82	253.6	10.42	1016	40.17	4058	1600
16.79	3.882	67.18	10.29	268.7	10.18	1075	46.23	4299	1624
17.78	3.88	71.17	8.747	284.7	11.33	1139	51.84	4556	1140
18.85	4.070	75.41	10.12	301.6	11.60	1207	56.7	4826	894
19.97	4.172	79.88	10.24	319.6	11.53	1278	57.13	5113	717.8
21.16	4.230	84.64	11.82	338.6	10.87	1354	54.67	5417	690.9
22.42	4.285	89.67	13.38	358.7	12.42	1436	65.42	5739	683.4
23.75	4.261	95.01	14.84	380	12.84	1520	61.23	6080	562.5
25.16	4.186	100.7	15.48	402.6	13.06	1610	113	6442	478.7
26.66	4.312	106.6	15.88	426.6	12.08	1706	136.6	6825	431.6
28.25	4.251	113	20.82	451.9	13.01	1808	147.6	7231	400.2
29.93	4.290	119.7	25.08	478.8	14	1916	168.2	7661	428.1
31.7	4.3	126.8	27.28	507.3	16	2029	177.9	8116	384.8
33.58	4.28	134.4	27.7	537.4	17	2150	172.4	8599	394.2
35.58	4.208	142.3	26.83	569.4	17.12	2278	207	9110	430.6
37.7	4.535	150.8	25.08	603.3	16.40	2413	252.2	9652	448.9

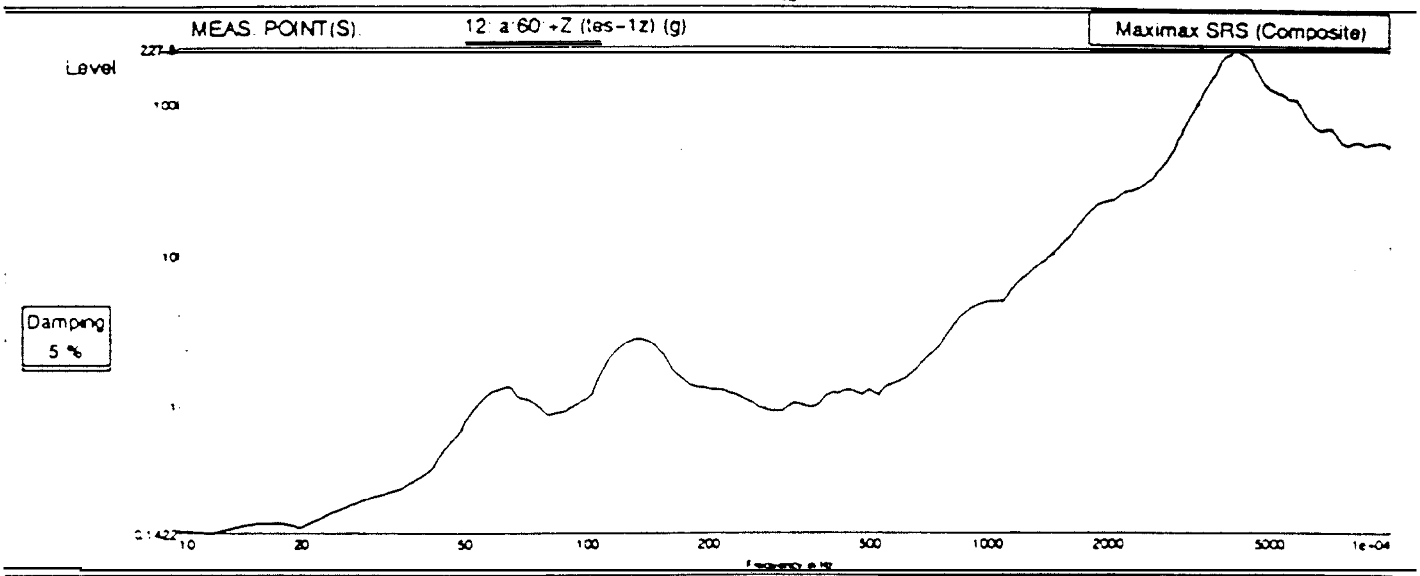
GE ASTRO SPACE

PROJECT : mars observer
ITEM(S/N) : space craft
MODE : Shocks

AXIS : shock
TEST TYPE : pyro valve #1
REF. POINT :

E.T.P..: 5936
SEQ. : 16 (4-24-92)





MEAS. POINT(S): 12: a:60:+Z (tes-1z) (g) Damping=5 % Resol =1/12)

Maximax SRS (Composite) values table

Freq.	Values	Freq.	Values	Freq.	Values	Freq.	Values	Freq.	Values
[illegible]	0.142	[illegible]	[illegible]	[illegible]	1.340	638.1	1.673	[illegible]	32.40
10.50	0.140	42.32	0.4280	169.3	1.340	677.1	1.848	2700	38.86
11.21	0.1462	44.84	0.5310	179.3	1.388	717.4	2.281	2870	46.81
11.88	0.1462	47.5	[illegible]	190	1.383	760.1	2.642	3040	[illegible]
12.58	0.1461	50.33	0.6128	201.3	1.306	805.2	3.3	3221	[illegible]
13.33	0.1642	53.30	1.006	213.3	1.384	853.1	3.836	3413	119.3
14.12	[illegible]	56.40	1.176	[illegible]	1.233	903.8	4.452	3616	162.0
14.96	0.1636	59.86	1.281	239.4	1.167	957.6	4.777	3830	198.7
15.85	0.1687	63.41	1.364	253.6	1.083	1016	4.881	4058	224.1
16.79	[illegible]	67.18	1.183	268.7	0.8845	1075	4.982	4300	214
17.79	0.1656	71.17	1.110	284.7	0.9614	1139	5.810	[illegible]	184.2
18.85	0.1654	75.41	1.084	301.6	0.9646	1207	6.85	[illegible]	129.4
19.97	0.1674	79.88	0.8798	319.6	1.094	1278	7.775	5113	121.8
21.16	0.1784	84.64	0.817	338.6	1.058	1354	8.781	5417	114.1
22.42	0.1807	89.67	0.8811	358.7	0.9807	1436	9.89	5738	108.1
23.75	0.1871	95.01	1.094	380	1.086	1520	11.8	6080	87.82
25.16	0.2104	100.7	1.161	402.6	1.23	1610	13.75	6442	70.88
26.66	0.2230	106.6	1.572	426.6	1.364	1706	16.86	[illegible]	67.57
28.25	0.237	113	2.057	451.9	1.274	1808	18.72	7231	66.38
29.93	0.2487	119.7	2.481	478.8	1.211	1916	20.63	7661	64.31
31.7	0.2616	126.8	2.733	507.3	1.288	2029	22.75	8116	56.86
33.59	0.2742	134.4	2.816	537.4	1.298	2150	25.27	[illegible]	64.07
35.59	0.2864	142.3	2.887	569.4	1.4	2278	28.5	9110	66.02
37.7	0.324	150.8	2.326	603.3	1.480	2413	28.7	[illegible]	56.07

GE ASTRO SPACE

PROJECT : MO

ITEM(S/N) : S/N 001

MODE : Shocks

AXIS : Shock

TEST TYPE : Pyro Valve

REF. POINT :

E.T.R.: 5936

SEQ. : 2 (4/22/92)

RXO Anti-Vel. Sun Panel

MARTIN MARIETTA
MARTIN MARIETTA ASTRO SPACE

Sig. Sources: S/A Separation, Pyro Valve

EVENT	ACCEL	SRS (g)	FREQ (Hz)
S/A #9	AVEL-SUN-1Y	~~4.6~~ 61.9	~~9110~~ 2520
	~~TES-1Z (source)~~	~~1603~~	~~4058~~
MAG	AVEL-SUN-1 Y	~~4.4~~ 6.5	~~5113~~ 3830
	AVEL-SUN-3Y	108.7	6442
Pyro valve	AVEL-SUN-1Y	4.6	9110
	TES-1 Z (source)(Lo)	224	4058
	" (source)(hi)	1603	4058

Spacecraft Configuration for Pvro Shock Test

MARTIN MARIETTA

MARTIN MARIETTA ASTRO SPACE

Spacecraft powered with C&DH Subsystem operational:

- o Both SCPs on and active, SCP1 in Control,
- o Payload off,
- o All unused pyro and thruster equipment disabled and disarmed,
- o RWAs on and in Launch Tach, ≈200 rpm,
- o DTRs on,
- o RPAs off.

Pyro firing controlled via Stored Command Sequence (script) loaded into SCPs. Script is nominally the same as the one used in flight.

Spacecraft Pyro Firing

PYRO SHOCK TEST RESULTS SUMMARY

- Shock Tests Performed
  - Separation
  - GRS Sensor & Canister
  - HGA
  - Solar Array
  - Magnetometer
  - Pyrovalve

- Separation shock had less attenuation than other shocks.

- GRS, HGA Mag and Solar Array shocks had high attenuation rates.

- Pyrovalve shock produced low responses.

- Peak shock responses are less than the design shock spectrum of ENV-RQM-3271152.

- All systems functioned normally after pyro firings.

Spacecraft Pyro Firing Plan

Modified Spacecraft Pyrotechnic Device Firing Plan

PYROTECHNIC DEVICE / LOCATION	PRE-DYNAMIC DEPLOYMENT	POST DYNAMIC DEPLOYMENT	PYRO-SHOCK
HGA			
Gimbal Support #1	0 (a)	1	0
Gimbal Support #2	0 (a)	1	0
Wrist Hinge	0 (a)	1	0
Boom Support #2	0 (a)	1	0
Boom Support #1	0 (a)	1	0
Tripod (Velocity)	0	1	0
Tripod (Anti Velocity)	0	1	0
GRS			
Bracket (Nadir)	0	1	0
Bracket (Nadir)	0	1	0
Bracket (Velocity)	0	1	0
Canister Support #1	0	1	0
Canister Support #2	0	1	0
MAGNETOMETER			
Canister	0	1	0
SOLAR ARRAY			
Outer Shear Ties (4)	0 (b)	1	0
Inner Shear Ties (4)	0 (b)	1	0
Center Release	0 (b)	1	0
Gimbal Tripod Shear Tie	0 (b)	0	0
SEPARATION CLAMP			
(+X) Primary	0	1	0
(X) Primary	0	1	0
(+X) Secondary	0	1	0
(X) Secondary	0	1	0
PYROVALVES			
High Pressure #1	N/A	N/A	2
High Pressure #2	N/A	N/A	0
Low Pressure #1	N/A	N/A	2
High Pressure #2	N/A	N/A	0

Notes

(a) The HGA Deployment Assembly is to be deployment tested at the assembly level only prior to Spacecraft Dynamic testing. Testing is to be accomplished without the HGA at the assembly level.

(b) The Solar Panel Deployment Assembly is to be deployment tested in conjunction with the second spacecraft • [illegible] only prior to Spacecraft Dynamic testing

Spacecraft Pyro Firing Plan

WAIVER WD 22785 Spacecraft Pyrotechnic Device Firing Plan

PYROTECHNIC DEVICE /LOCATION	PRE-DYNAMIC DEPLOYMENT	POST DYNAMIC DEPLOYMENT	PYRO-SHOCK
HGA			
Gimbal Support #1	0 (a)	1	0
Gimbal Support #2	0 (a)	1	0
Wrist Hinge	0 (a)	1	0
Boom Support #2	0 (a)	1	0
Boom Support #1	0 (a)	1	0
Tripod (Velocity)	1 (a)	1	0
Tripod (Anti-Velocity)	1 (a)	1	0
GRS			
Bracket (Nadir)	1	1	0
Bracket (Nadir)	1	1	0
Bracket (Velocity)	1	1	0
Canister Support #1	1	1	0
Canister Support #2	1	1	0
MAGNETOMETER			
Canister	1	1	0
SOLAR ARRAY			
Outer Shear Ties (4)	0 (b)	1	0
Inner Shear Ties (4)	0 (b)	1	0
Center Release	0 (b)	1	0
Gimbal Tripod Shear Tie	0 (b)	1	0
SEPARATION CLAMP			
(+X) Primary	0	1	0
(-X) Primary	0	1	0
(+X) Secondary	0	1	0
(-X) Secondary	0	1	0
PYROVALVES			
High Pressure #1	N/A	N/A	3
High Pressure #2	N/A	N/A	0
Low Pressure #1	N/A	N/A	3
High Pressure #2	N/A	N/A	0

Notes.

(a) The HGA Deployment Assembly is to be deployment tested at the assembly level only prior to Spacecraft Dynamic testing. Testing is to be accomplished without the HGA at the assembly level S/C tripod Pre dynamic deployments are limited to "pop & catch" tests

(b) The Solar Panel Deployment Assembly is to be deployment tested in conjunction with the second spacecraft structure only prior to Spacecraft Dynamic testing

Spacecraft Pyro Firing Plan

- ENV-SS-3271152 , MO System level environmental test specification directs the number of pre and post dynamic pyro device firings to be performed at the Spacecraft level.

- Waiver WD 22785 proposed that the specified firing plan be modified due to changes in Spacecraft pyro initiated deployments. This waiver has been approved by JPL.

- The revised plan has been further modified as shown by Revision A dated 5/27/92. The Spacecraft I &T schedule mandated additional modifications to the plan.

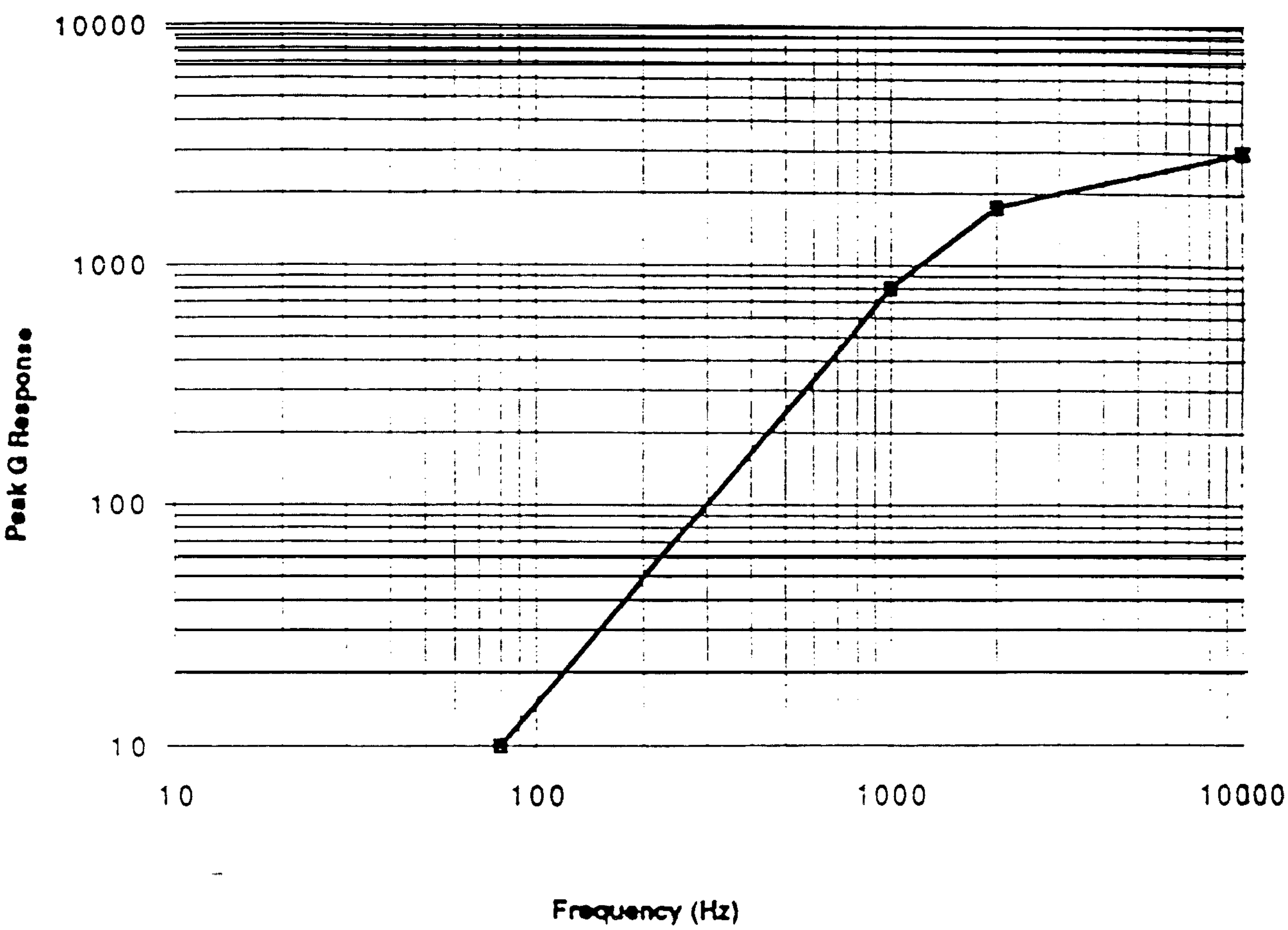


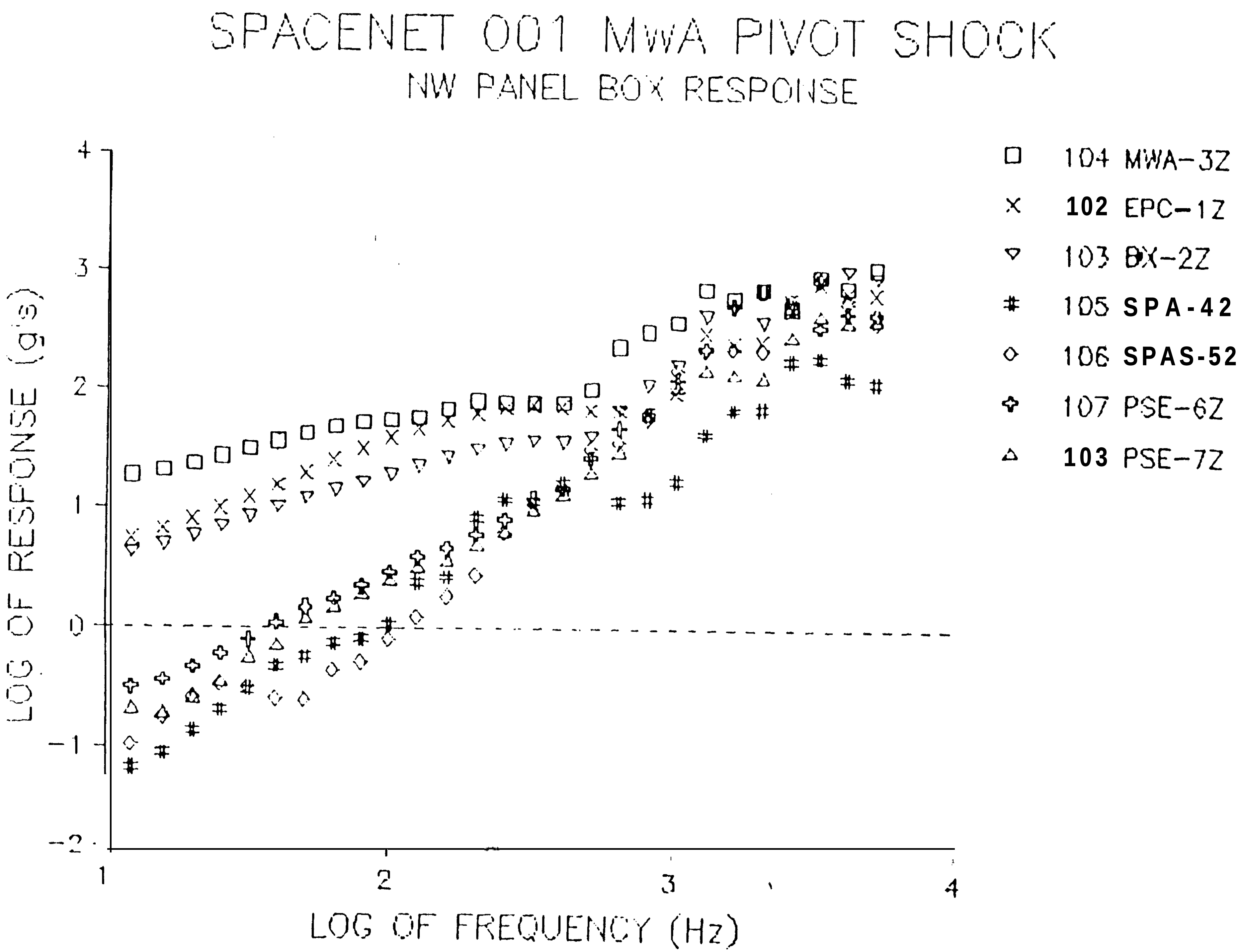
Figure 1. Pyrotechnic Response Spectra (Q = 20)

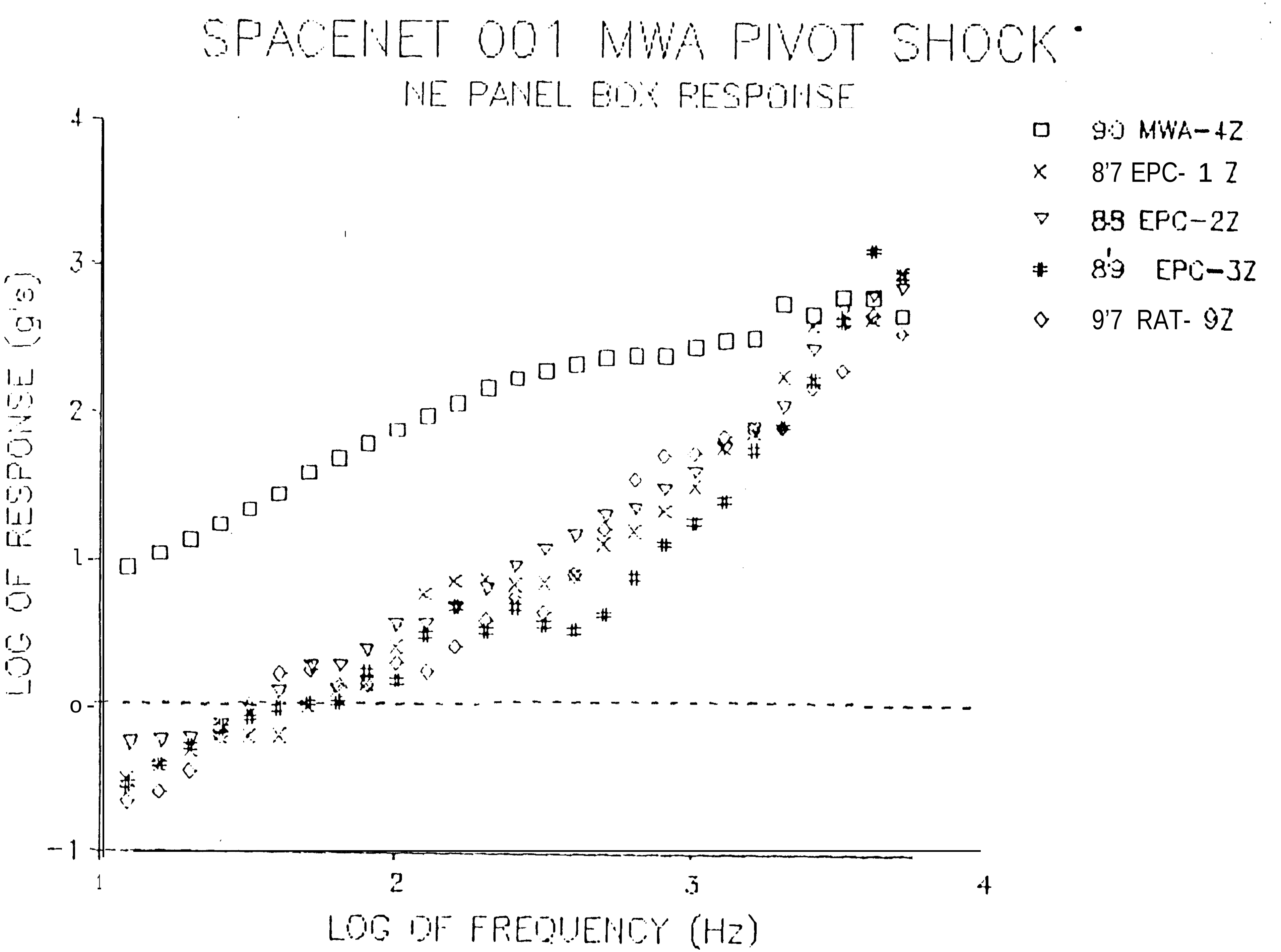
I.D. 1797M

Contractual Requirements
Required Pyro Tests

MARTIN MARIETTA ASTRO SPACE

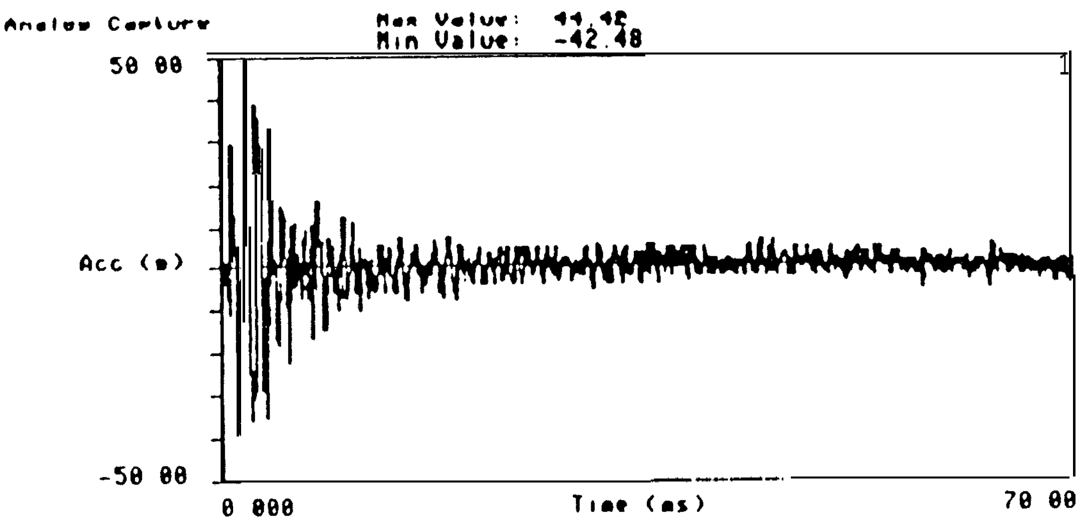
- ENV-PPR-3271152, Mars Observer Environmental Program Policy and Requirements.
- ENV-DR-3271152, Performance Specification: Environmental Design Requirements.
- ENV-SS-3271152, Mars Observer System Level Environmental Test Specification.
- ENV-RQM-3271152, Mars Observer Assemblies Environmental Test Specification.
- PLN-SVER-3271138, Mars Observer Spacecraft Structural Verification Plan.


SPACENET 001 MwA PIVOT SHOCK
NW PANEL BOX RESPONSE
LOG OF RESPONSE (g's)
LOG OF FREQUENCY (Hz)
4
3
2
1
0
−1
−2
1
2
3
4
104 MWA-3Z
102 EPC-1Z
103 BX-2Z
105 SPA-42
106 SPAS-52
107 PSE-6Z
103 PSE-7Z


SPACENET 001 MWA PIVOT SHOCK
NE PANEL BOX RESPONSE
9.0 MWA-4Z
8'7 EPC- 1 Z
8.8 EPC-2Z
8'9 EPC-3Z
9'7 RAT- 9Z
LOG OF RESPONSE (g's)
LOG OF FREQUENCY (Hz)
-1
0
1
2
3
4
1
2
3
4

SPACECRAFT MECHANICAL TESTING

SEPARATION SHOCK



03-Aug-85 AP-BAT1B-7X
SATCOM K HARMON CLAMP SHOCK

SRS Listing 03-Aug-85
Analog Capture

1.0% Damping Absolute Acceleration
1/3 Octave Maxi-Max

Freq	Ampl	Freq	Ampl
3.9811	0.163765	158.49	3.31773
5.8119	8.240726	199.53	6.89860
6.3896	0.499409	251.19	3.83381
7.9433	0.729309	316.23	8.31855
10.000	1.17378	398.11	13.5413
12.589	1.79529	301.19	25.4492
15.849	2.45508	630.96	61.4785
19.953	2.61865	794.33	187.227
25.119	2.89180	1000.0	312.719
31.623	1.81110	1258.9	135.797
39.811	2.03264	1584.9	77.7852
50.119	1.74113	1993.3	91.2188
63.096	3.74890	2511.9	148.914
79.433	2.65344	3162.3	144.477
100.00	7.29370	3981.1	100.523
125.89	4.33936		

DEPLOYMENT RELEASE & SHOCK

OBJECTIVE	• VERIFY OPERATION OF SEPARATION MECHANISM • VERIFY NO DAMAGE
ARTICLE	• **MWA'S, ANTENNAE,** SOLAR ARRAYS ON **PROTO** S/C
FACILITY	• **ACOUSTIC CHAMBER** • STE TO FIRE SQUIBS
LOADS	• **LIVE SQUIBS (TYPICAL FLIGHT)** • **FLIGHT SEPARATION HARDWARE** - **CUTTERS: BOLT, CABLE** - SEPARATION NUTS
MEASUREMENTS	• ACCELEROMETERS ~ 90
CRITERIA	. **MECHANICAL INSPECTION** CHECK LIST

WWM

V-BAND SEPARATION & SHOCK

OBJECTIVE
- VERIFY V-BAND SEPARATION SYSTEM FUNCTION
- VERIFY NO HARMFUL RESULT
- DETERMINE SHOCK INPUTS TO COMPONENT;

ARTICLE
- PROTOFLIGHT S/C MOUNTED ON ADAPTER
  - STOWED DEPLOYABLES: SA, REFLECS & ANT
  - TANKS EMPTY
  - PYROS INSTALLED
  - AKM SIMULATOR INSTALLED

FACILITY
- ACOUSTIC CHAMBER
- STE TO FIRE SQUIBS
- **BAND TIGHTENING RIG**

LOADS
- LIVE SQUIBS (TYPICAL FLIGHT)
- TWO TESTS

MEASUREMENTS
- ACCELEROMETERS $\simeq$ 90

CRITERIA
- MECHANICAL INSPECTION CHECK LIST

<u>KEY POINT: TEST LOAD IS NOMINAL</u>

WWM

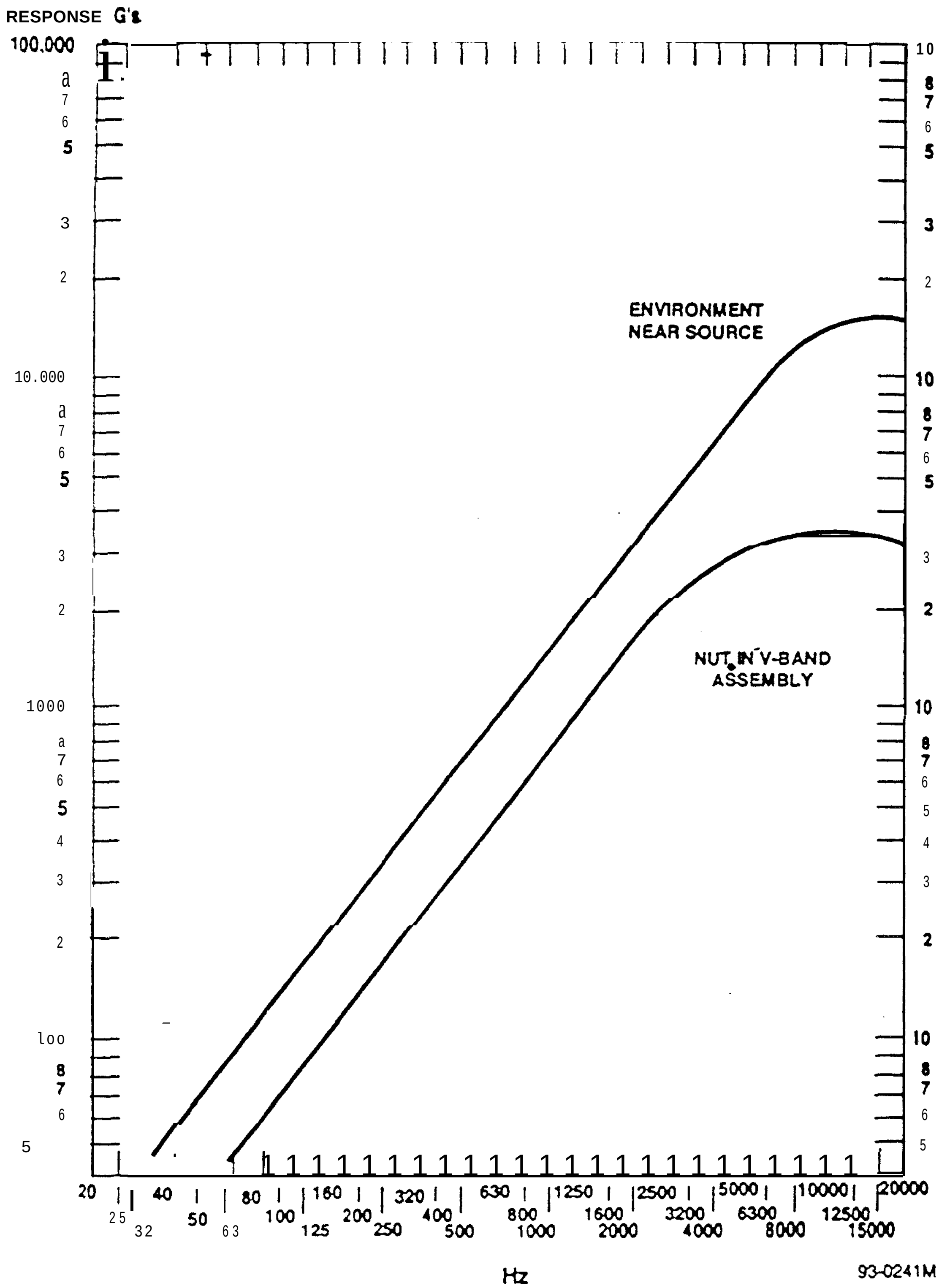


Figure 3.9- 1. Suggested Environment Produced by Expolsive Bolts and Separation Nuts

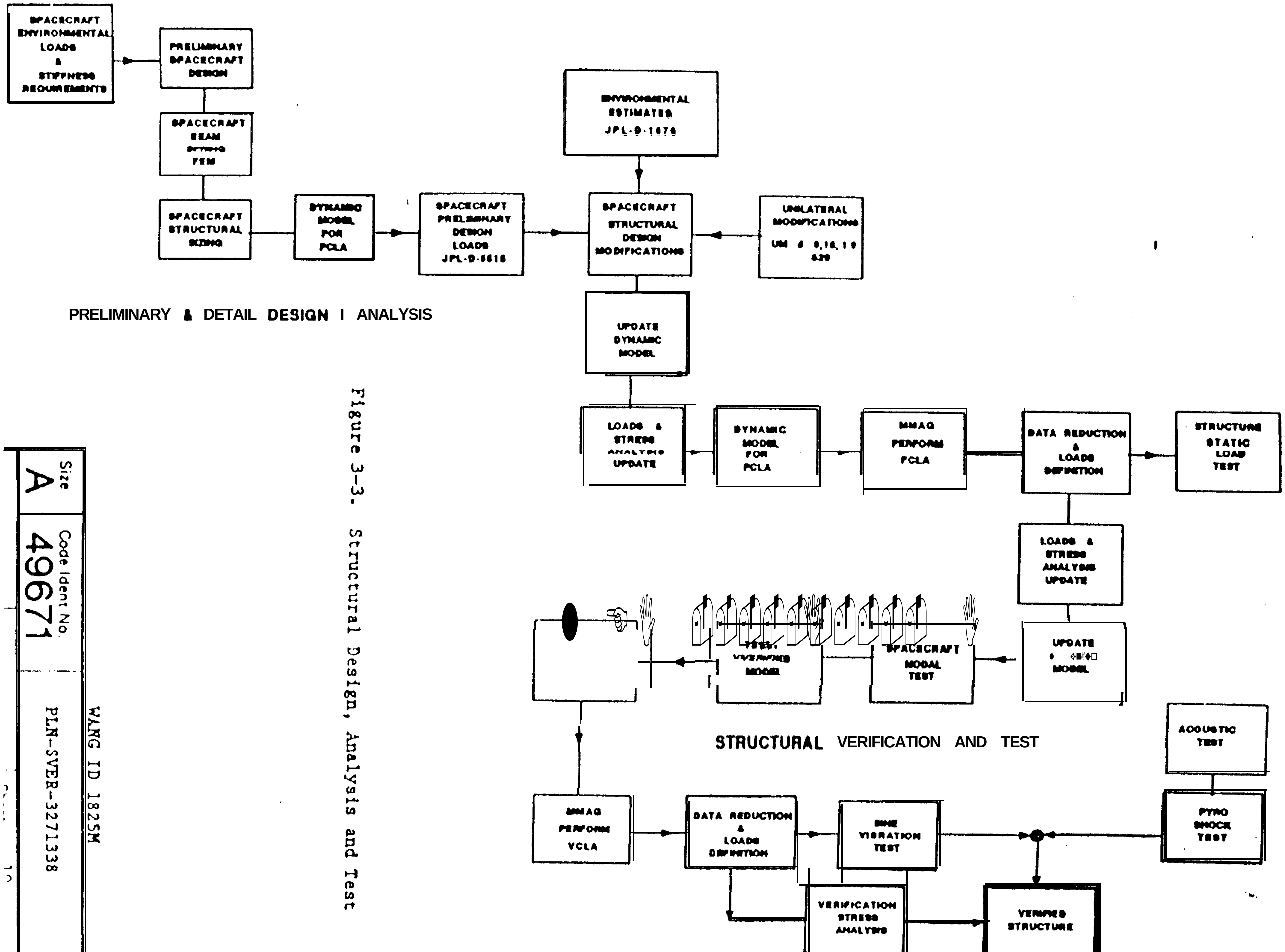


Figure 3-3. Structural Design, Analysis and Test

Att. 3 Aug '93

Standard Astro Space Practice Regarding Shock Environments

MARTIN MARIETTA

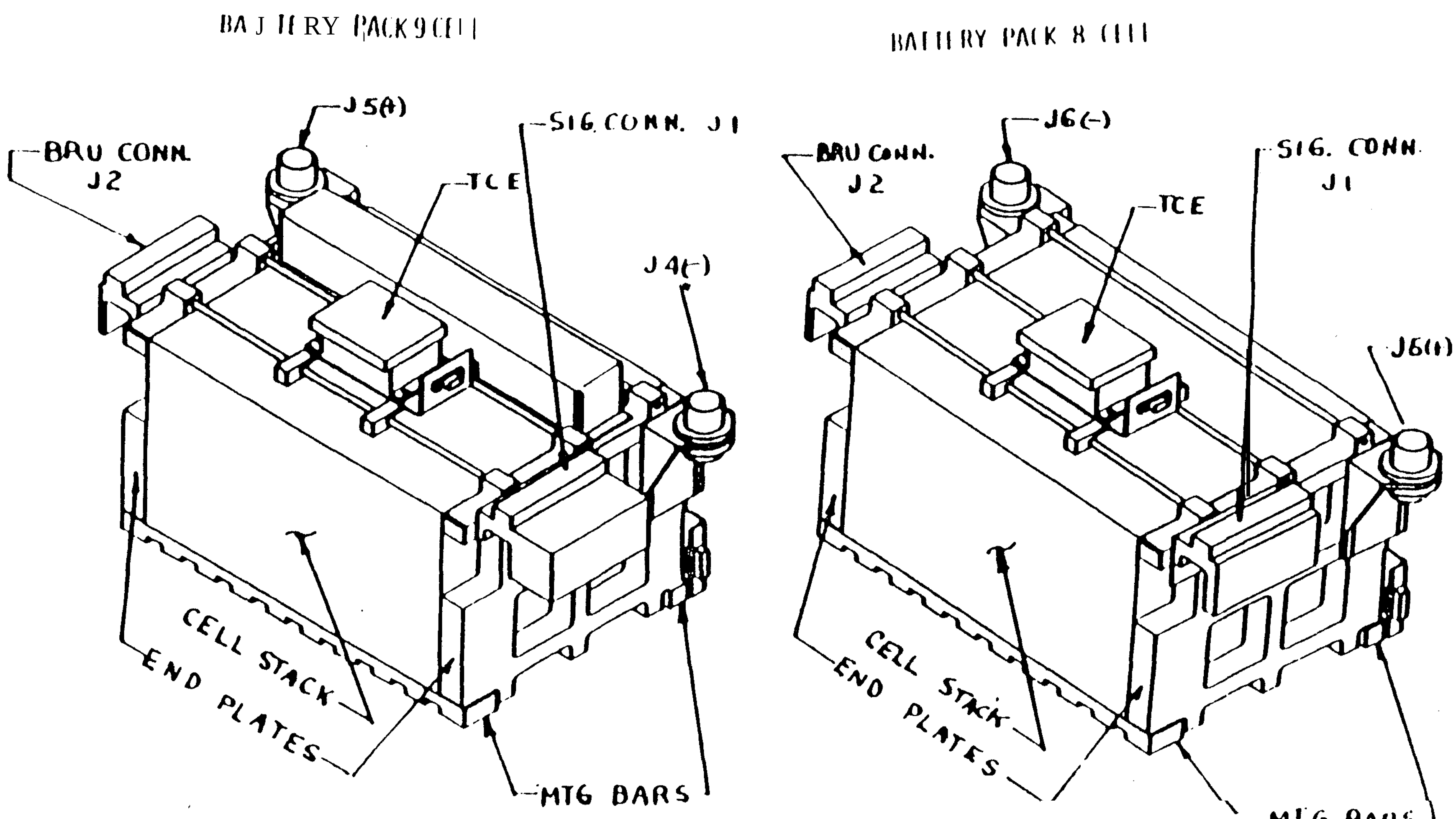
MARTIN MARIETTA ASTRO SPACE

- Environment Predictions
  - Customer Specifications/Requirements
  - Design Database
  - Shock Propagation
- Design Requirements
  - System Level
  - Component Level
- Structural Verification & Test
  - Component Level Tests
  - System Level Tests

42 AH Battery Pack

MARTIN MARIETTA

MARTIN MARIETTA ASTRO SPACE



SIZE: 40.00 x 18.80 x 18.42 cm (15.75” x 7.40” x 7.25” 9 CELL PACK)
38.10 x 18.80 x 18.42 cm (15.00” x 7.40” x 7.25” 8 CELL PACK)

MASS: 18.44 kg (40.66 lb) 9 CELL PACK

16.89 kg (37.24 lb) 8 CELL PACK

TOTAL 70.7 kg

Component Descriptions

- Solar Array ,
  - -Six panel using 63 series cells by 416 parallel strings, 30 circuits
  - -2 cm by 4 cm BSR type cells (26202)
  - -Four panels deployed for cruise crus GCO phases
  - -Six panels along with the boom deployed for mapping
  - -Two axis gimbal points the array at the sun
  - - Panel wiring minizes magnetic field
- Batteries
  - -Nickel Cadmium cells 42 AH
  - -Each battery in two packs, one 8 cell pack and the other 9 cell pack
  - -Temperature control by TCE on each pack, thermostat backup

Summary of Solar Array Capability and Power Margins

MARTIN MARIETTA

MARTIN MARIETTA ASTRO SPACE

MISSION PHASE	MINIMUM AVERAGE SOLAR ARRAY POWER AVAILABLE (WATTS)	POWER MARGIN (WATTS)			
		W.C.(3)	1 SHORTED BATT CELL	LOSS OF 1 S/A CIRCUIT	NOMINAL
INNER CRUISE	740(1)	203	-	-	
OUTER CRUISE	661(1)	15	-	-	
DRIFT	838(1)	152	-	-	
GC	938(2)	28	45	56	81
PERIHELION	1408(2)	42	48	129	147
APHELION	1147(2)	6	14	45	59

NOTES:

(1) WATTS AT 29.4V AT W.C. TEMPERATURE

(2) SA CAPABILITY AT 29V USING ORBITAL AVERAGE SUNTIME VALUES WITH NO CIRCUIT FAILURES.

(3) WORST CASE MARGINS WITH BOTH ONE SOLAR ARRAY CIRCUIT AND ONE BATTERY CELL SHORTED. AT WORST CASE SOLAR TEMPERATURE/PROFILE.

Spacecraft Fusing

MARTIN MARIETTA

MARTIN MARIETTA ASTRO SPACE

- 0 FUSES ON THE 28V MAIN BUS
- 0 FUSES ON THE 28V PAYLOAD BUS GROUPED TOGETHER ON FBA-1
- 0 SINGLE FUSE FOR EACH CIRCUIT/BOX/LOAD
- 0 SEPARATE FUSES FOR PRIMARY AND BACK-UP CIRCUITS
- 0 STANDARD FUSE RATINGS AND SIZES DERATED FOR SPACE APPLICATION
- 0 MOUNTED ON COVERED BOARDS FOR FUSE ACCESSIBILITY
- 0 FUSES SIZED TO PROTECT THE BUS, CONDUCTORS, AND PINS
- 0 FUSES SELECTED IN ACCORDANCE WITH PAPL AND SCD RCA 496712629518 DERATING
  - ADDITIONAL DERATING FOR TEMPERATURE
  - MINIMUM FUSE 2.0A, MAX FUSE 7.0A (NOMINAL RATINGS)

Summary of Nominal Loads by Phase and Modes (@ 28V)

MARTIN MARIETTA

MARTIN MARIETTA ASTRO SPACE

Table 3A Summary of Nominal Loads By Phase and Modes (Watts @28V) Rev 8 11 2/26/90

	INNER CRUISE (INIT)	SAFE MODE I	OUTER CRUISE (INIT)	SAFE MODE t	DRIFT (INIT)	DRIFT (NO INIT)	GCO INT	GCO NO INT	MAPPING PERIHELION INT+PB	PERIHELION INT	PERIHELION NO INT part orb	PERIHELION NO INT full orb	APHELION INT+PB	APHELION INT	APHELION NO INT part orb	APHELION NO INT full orb	SAFE MODE C
PAYLOAD	1.5	3.5	1.5	3.5	11.0	11.0	3.5	3.5	130.0	130.4	130.t	130.0	130.t	130.4	130.4	130.4	3.5
TELECOMM	155.1	39.1	144.5	39.1	144.5	39.1	144.5	39.1	144.5	144.5	39.1	39.1	144.5	144.5	39.1	39.1	39.1
CDM	51.3	49.1	51.3	49.1	73.5	73.5	58.4	58.4	73.5	58.4	5t.4	5t.4	73.5	5t.4	58.4	58.4	49.1
ACS	49.5	49.5	49.5	49.5	49.5	49.5	49.5	49.5	51.0	51.0	51.0	51.0	51.t	51.t	51.0	51.0	49.5
PROPULSION	3.t	3.0	3.t	3.t	1.0	3.0	13.4	3.0	13.4	13.4	3.0	3.0	11.4	11.4	3.0	3.0	3.t
POWER	6.2	6.2	6.2	6.2	6.2	6.2	6.2	6.2	6.2	6.2	6.2	1.2	1.2	1.2	6.2	6.2	6.2
MECHANISMS	t	0	t	0	0	0	0	0	22.6	22.6	1.t	7.0	22.6	22.6	7.0	7.0	0
THERMAL	160.t	194.0	254.0	303.t	254.0	111.0	202.t	199.0	104.0	104.0	102.0	135.0	137.0	137.0	120.0	170.5	238.1
Current Total	431.4	345.2	510.0	454.2	505.5	493.1	478.3	359.5	545.6	530.5	391.t	430.t	570.t	563.5	420.0	466.4	388.6
Total @ delta PDB	442.6	354.6	550.t	469.t	556.1		494.3	371.0	570.0	556.1	612.2	448.9	604.7	584.7	440.6	484.2	367.4
Margin from PDB	11.7	9.4	36.0	11.0	6.6		16.0	11.5	24.4	19.6	14.3	16.0	26.1	21.2	15.7	17.8	21.2

MRB00164

JPL

LESSONS LEARNED SUMMARY

- SYSTEM CONTRACT APPROACH SAVED COST
  - HERITAGE HARDWARE & S/W DESIGN REDUCED MO COSTS
  - CONCURRENT PROGRAMS ALLOWED FOR SHARED DEVELOPMENT
  - PRODUCTION LINE APPROACH SAVED RESOURCES IN MANUFACTURING AND TESTING
- ELECTRONIC PARTS INDUSTRY HAS MATURED. CLASS S PARTS MAY BE DIFFICULT TO GET, BUT ONCE YOU **GET** THEM **THEY ARE GOOD**
- BIGGEST PROBLEMS USUALLY OCCUR WITH NEW DEVELOPMENTS OR UNIQUE PROCESSES
- INSTRUMENTS SHOULD BE THERMALLY ISOLATED FROM SPACECRAFT
- THE PROJECT MANAGER SHOULD HOLD ALL WEIGHT RESERVE
- THE LAUNCH VEHICLE RESERVE SHOULD HAVE BEEN RELEASED EARLIER

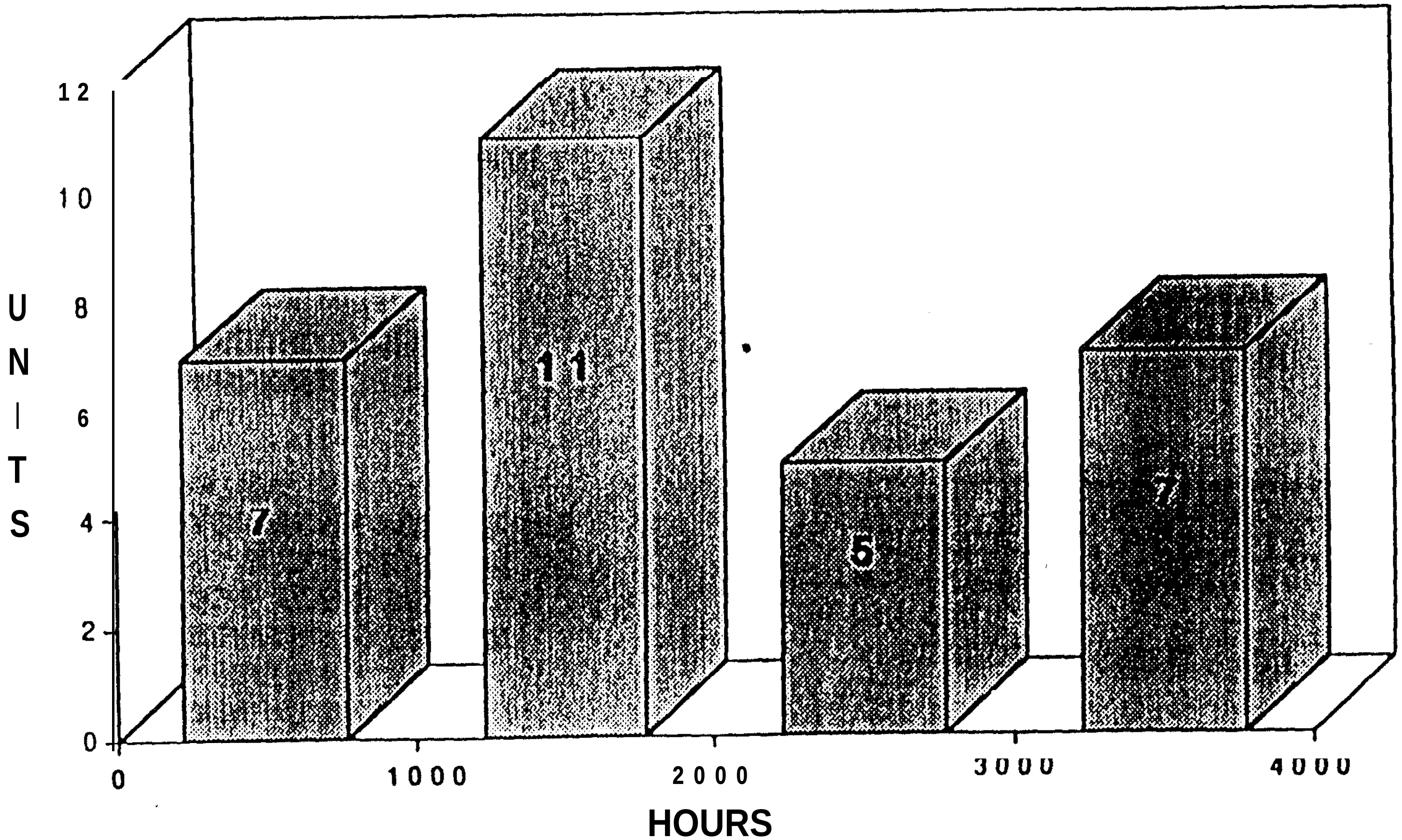
JPL

PFR COMPARISON

CAUSE CATEGORY	PROJECT			
	VOYAGER	GALILEO	MAGELLAN	MO
DESIGN	33.5 %	19.9 %	24.7 %	23 %
SOFTWARE	4.0 %	6.1 %	18.1 %	5 %
WORKMANSHIP	13.9 %	16.1 %	7.0 %	18 %
PIECE PART	11.3 %	6.4 %	2.6 %	2 %
MANUFACTURING	4.2 %	2.3 %	2.5 %	3 %
SUPPORT EQUIP.	5.1 %	14.4 %	9.0 %	13 %
DAMAGE	3.0 %	2.1 %	2.3 %	1 %
TEST ERROR	8.4 %	10.8 %	12.9 %	21 %
ADJUSTMENT	2.0 %	1.4 %	0.6 %	1 %
OPERATING TIME	0.1 %	0.1 %	0.0 %	0 %
OTHER	12.9 %	18.0 %	17.9 %	13 %
NONE	1.7 %	2.4 %	2.3 %	0 %
TOTAL NUMBER	3433	4500	1600	1378

JPL

SPACECRAFT ASSEMBLY OPERATING HOURS



JPL

ENVIRONMENTAL TEST PROGRAM

ASSEMBLY LEVEL PROTOFLIGHT TESTS	APPLICABILITY
• SINE VIBRATION - @ 1.5 X FA	NEW ASSEMBLIES ONLY, ASD 1ST UNITS
• ACOUSTICS - FOR A/M > .3	HGA & S/A
• RANDOM VIBRATION - @ FA+4DB	MOST ASSEMBLIES, SOME @ < 4DB
• ACCELERATION - FOR G SENSITIVE ONLY	ANALYSIS ONLY
• PYRO SHOCK - SHOCK SENSITIVE ONLY	MOST @ SYSTEM LEVEL ONLY
• THERMAL/VACUUM ALL FOR 168 HRS @ -20 TO +75°C	NEW ASSEMBLIES ONLY, ASD 1ST UNITS . HERITAGE: THERMAUATMOS, 80 HRS -20 TO + 75°C IF WCA OK, OTHERWISE, FA ± 25°C
• THERMAL SHOCK (SELECTED)	HGA & S/A
• LAUNCH PRESSURE PROFILE	ANALYSIS ONLY
• EMC/EMI	MOSTLY SAMPLED OR @SYSTEM
• OTHER	• MAG & RAD CHARACTERIZATION

SYSTEM LEVEL PROTOFLIGHT TESTS	APPLICABILITY
• SINE VIBRATION @ 1.5 X FA	WITH NOTCHING
• ACOUSTICS @ FA + 4DB	ACOUSTICS
• RANDOM VIBRATION - ACOUSTICS ONLY OK	ACOUSTICS
• PYRO SHOCK - 3 EACH	SELECTED
• THERMAL/VACUUM FOR 300 HRS	THERMAL BALANCE & THERMAL/VACUUM
• EMC/EMI	SELECTED
• OTHER	MAG & RAD CHARACTERIZATION

JPL

TEST PROGRAM
HIGHLIGHTS & OBSERVATIONS

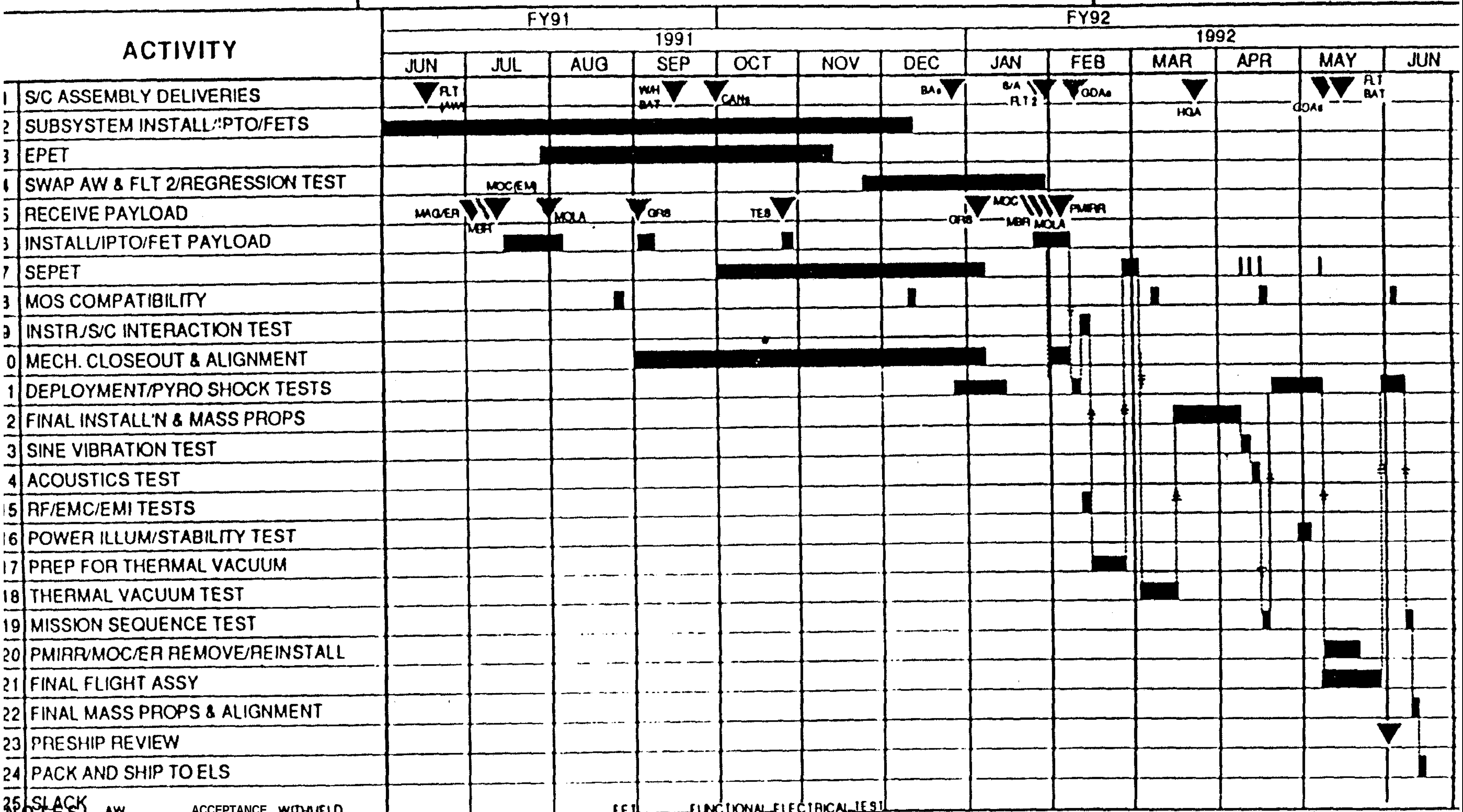
- MUCH OF BOARD AND ASSEMBLY-LEVEL TESTING DONE ON AUTOMATED TEST STATIONS
- SUBSYSTEM TESTING CAN BE DONE AT SYSTEM LEVEL
- BENCH INTEGRATION TEST (BIT) ADDED TO START ELECTRICAL **SYSTEM TESTING**
- SPARES AND VTL NON-FLIGHT ASSEMBLIES WERE VITAL IN KEEPING FLOW MOVING
- AN INTEGRATED PROPULSION SUBSYSTEM SAVES MASS BUT SLOWS TESTING
- PACKET TELEMETRY MAY SAVE EFFORT IN MOS, BUT IT SIGNIFICANTLY SLOWED SYSTEM TEST
- THE SYSTEM THERMAL-VACUUM TEST WAS KEY TO BRINGING **SYSTEM TOGETHER**
- SYSTEM TESTING ORDER CAN BE CHANGED TO MINIMIZE TIME WITH LITTLE ADDED RISK
- SYSTEM SINE VIBRATION TEST REVEALED NO PROBLEMS
- SIGNIFICANT EFFORT ON DEPLOYMENT TEST AND INSPECTION WAS TIME WELL SPENT
- COMMON WORK STATIONS FOR SYSTEM TEST, VTL, & MOS WOULD SAVE RESOURCES
- COMMON S/C & MOS MISSION SEQUENCES WOULD SAVE RESOURCES.
- MISSION SEQUENCES MUST BE SHORT ENOUGH OR RE-STARTABLE TO ALLOW FOR TEST INTERRUPTIONS (e.g. LIGHTNING AT CAPE)

JET PROPULSION LABORATORY

MARS OBSERVER
SYSTEM TEST

CREATED: DECEMBER 12, 1991
DATE

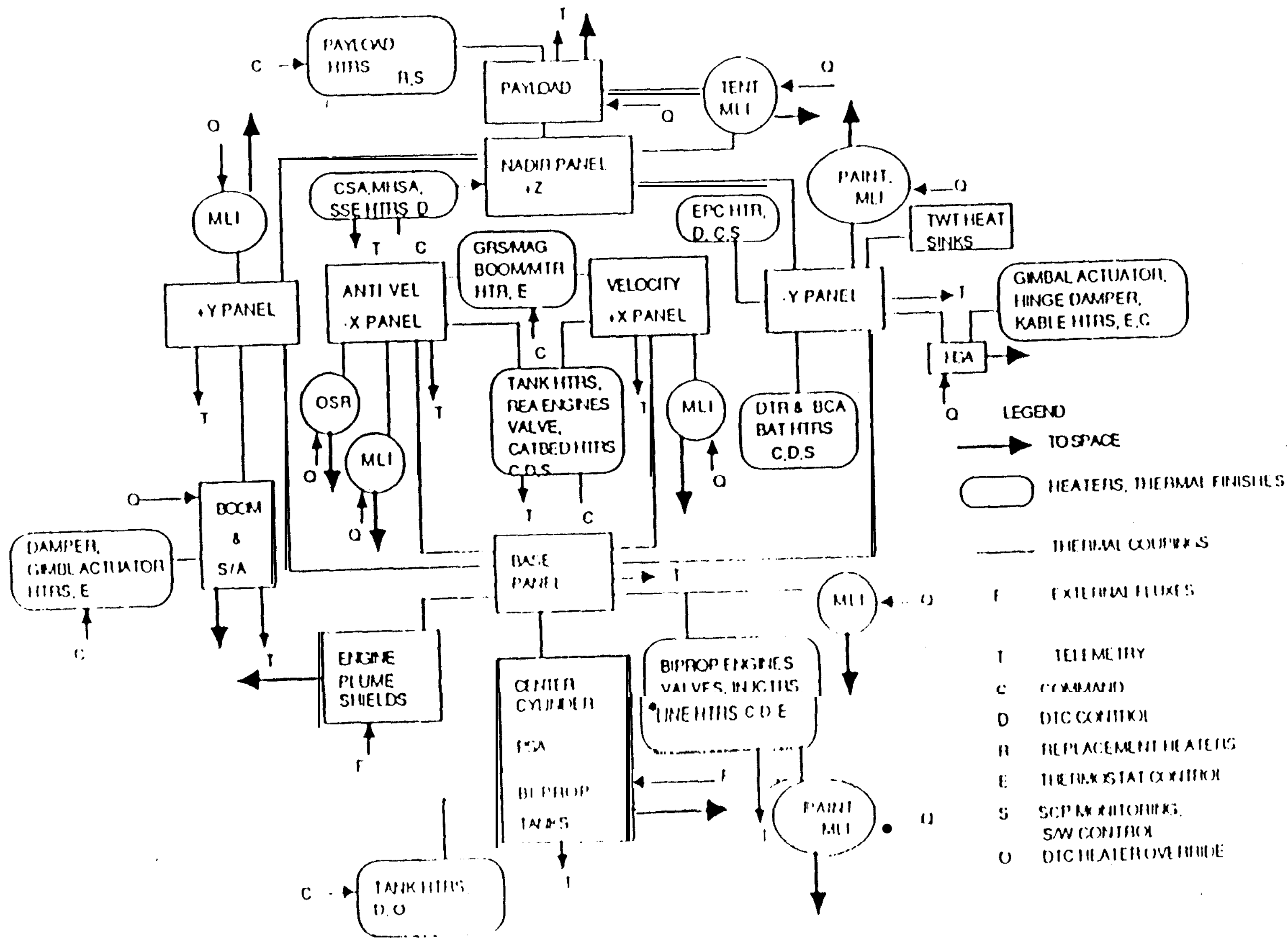
STATUS AS OF: JUNE 26, 1992
DATE



NOTES:

AW	ACCEPTANCE WITHHELD	FET	FUNCTIONAL ELECTRICAL TEST
BAs	BOOM ASSEMBLIES (HGA & S/A)	FLT	FLIGHT
BAT	BATTERIES	GDA	GIMBAL DRIVE ASSEMBLY
CAN	CANNISTERS (MAG/ER & GRS)	HGA	HIGH GAIN ANTENNA
ELS	EASTERN LAUNCH SITE	IPTO	INITIAL POWER TURN ON
EM	ENGINEERING MODEL	S/A	SOLAR ARRAY
EMC/EMI	ELECTROMAGNETIC COMPATIBILITY/INTERFERENCE	SEPET	SYSTEM ELECTRICAL PERFORMANCE EVALUATION TEST
EPET	ELECTRICAL PERFORMANCE EVALUATION TEST	W/H	WORKHORSE

THERMAL CONTROL BLOCK DIAGRAM

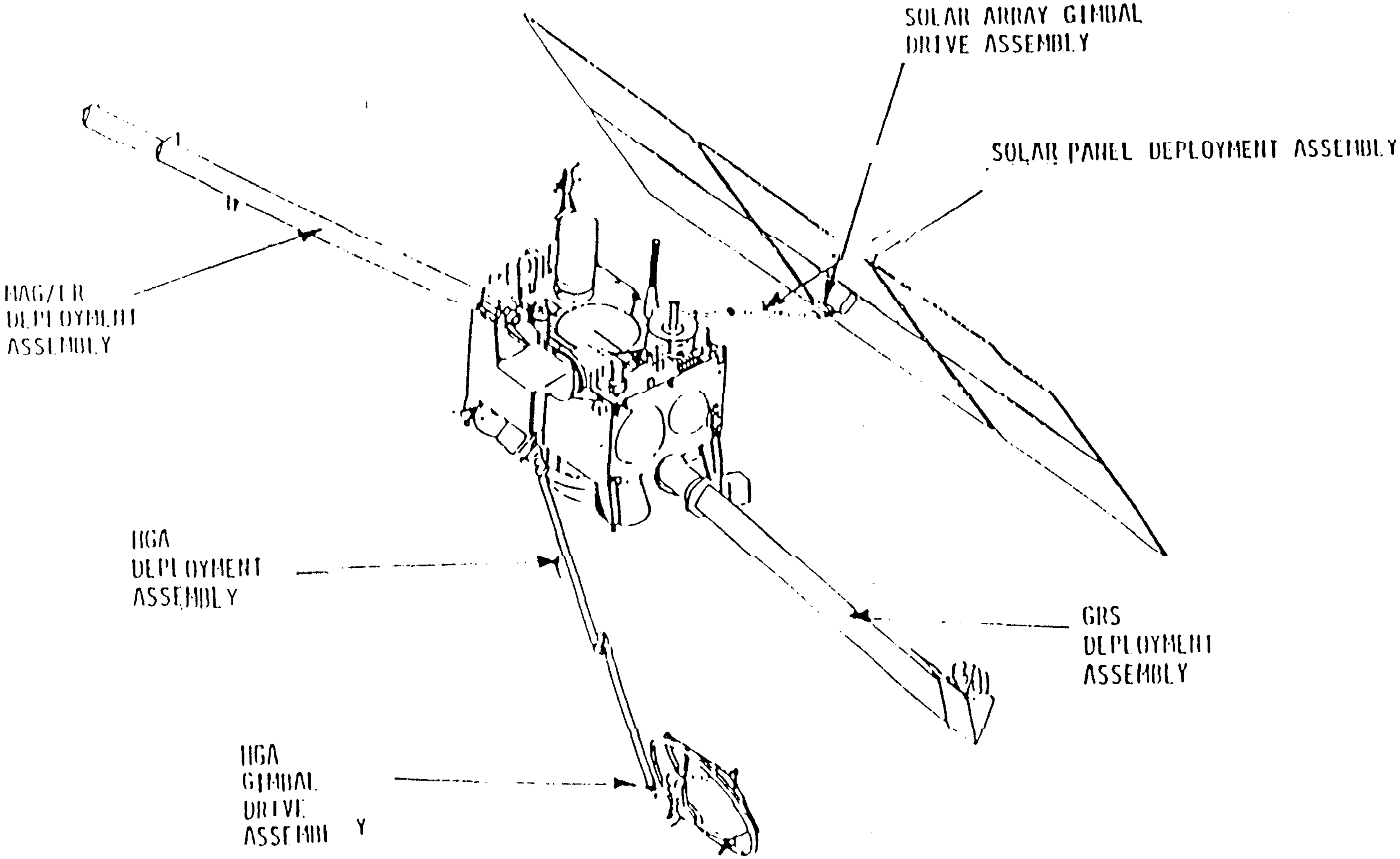


THERMAL CONTROL SUBSYSTEM

- **DESIGN DRIVERS**
  - VARYING SUN DISTANCE
  - SHUNT DISSIPATION
  - PAYLOAD THERMAL REQUIREMENTS
  - SUBSYSTEM THERMAL REQUIREMENTS
- **CHARACTERISTICS**
  - PASSIVE CONTROL: MULTI-LAYER INSULATION BLANKETS (MLI), PAINT, TAPE, OPTICAL SOLAR REFLECTORS (OSRs)
  - ACTIVE CONTROL: THERMOSTATS, FIXED & PROPORTIONAL HEATERS
- **DEVELOPMENT HIGHLIGHTS**
  - DELETED LOUVERS FROM ORIGINAL DESIGN
  - CUSTOMIZED PAYLOAD THERMAL INTERFACE
  - PROPELLANT TANK TEMPERATURES

JPL

MECHANISMS

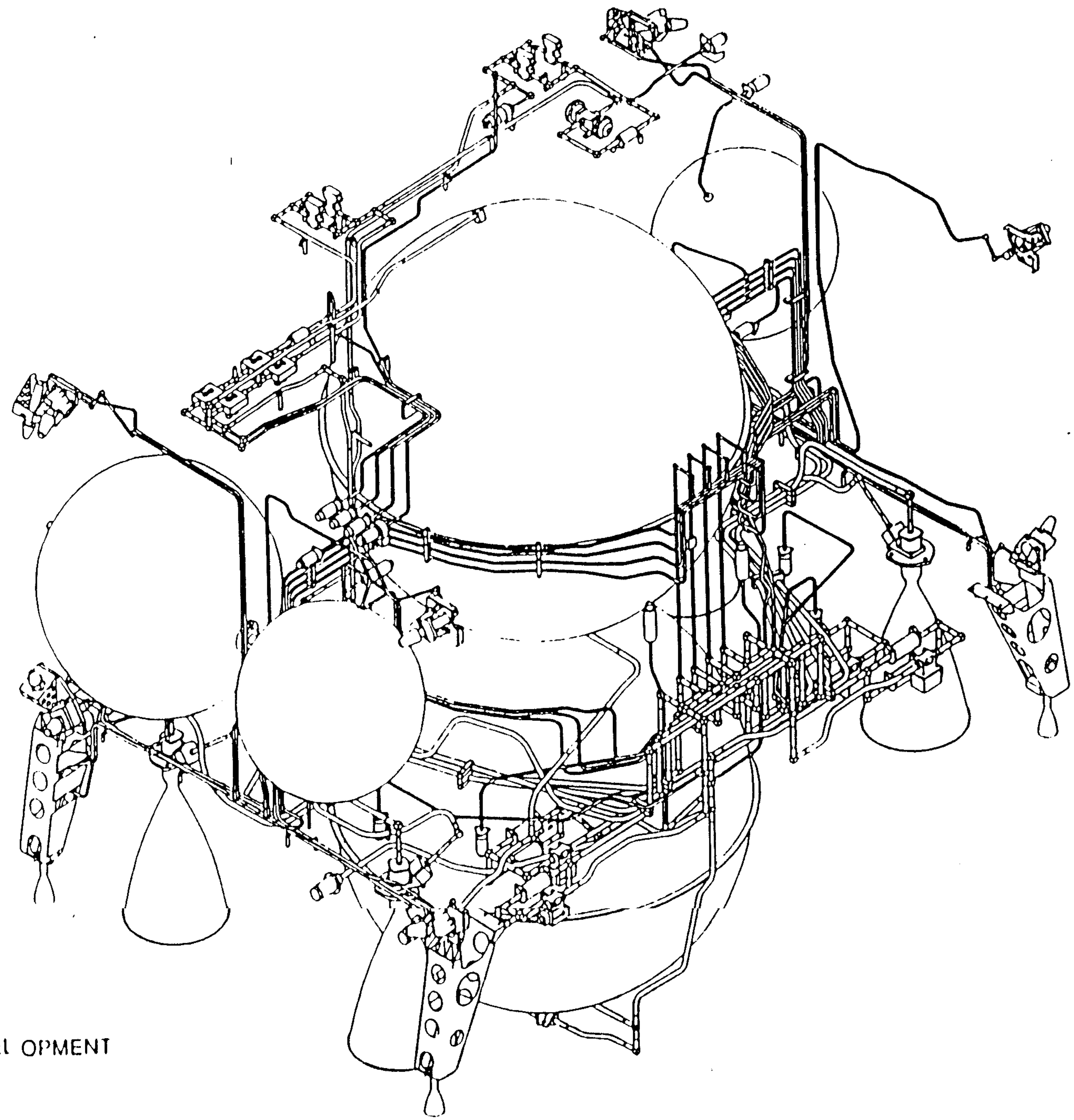


MECHANISMS SUBSYSTEM

- **DESIGN DRIVERS**
  - MAG/ER & GRS CRUISE CALIBRATION REQUIREMENTS
  - MAG/ER & GRS MAPPING REQUIREMENTS
  - NADIR POINTING BUS CAUSES VARYING EARTH & SUN GEOMETRY @ MARS
  - LAUNCH & MOI LOADS
- **CHARACTERISTICS**
  - 2 x 6M CANISTER BOOMS (MAG/ER & GRS)
  - 5.7M, 2 SECTION, HGA BOOM, 2 AXIS GIMBAL
  - 2.8M, 2 SECTION, S/A BOOM, 2 AXIS GIMBAL
- **DEVELOPMENT HIGHLIGHTS**
  - SCIENCE BOOMS TYPE CHANGED FROM RETRACTABLE LANYARD TO CANISTER
  - RE-SIZING OF BOOM SECTIONS FOR INCREASED TIP MASS
  - SEVERAL ITERATIONS ON HGA BOOM LENGTH
  - SEVERAL DEVELOPMENT PROBLEMS WITH OPTICAL ENCODERS
  - EXTENSIVE TEST PROGRAM

JPL

PROPULSION ASSEMBLY

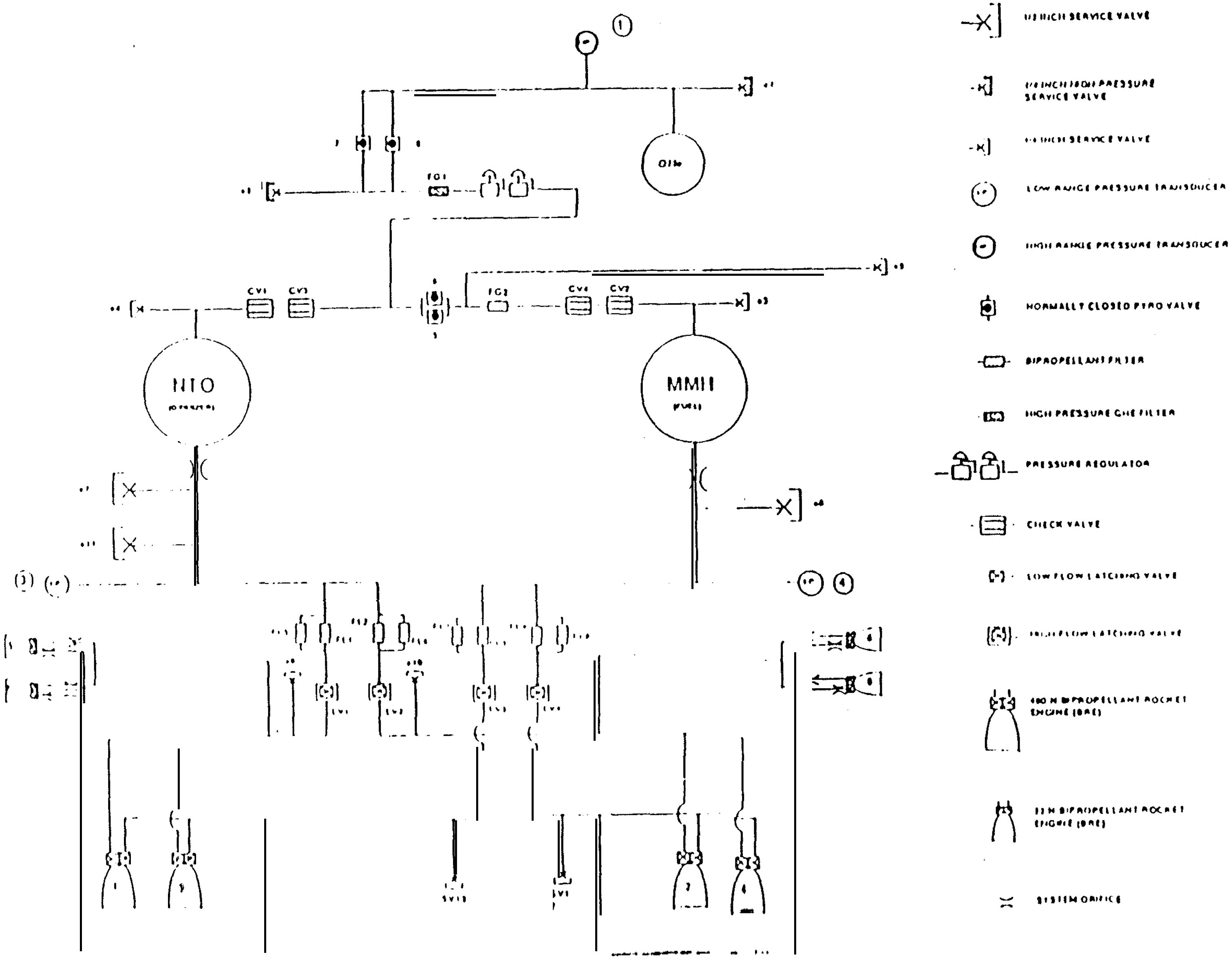


MARS OBSERVER
SPACECRAFT DEVELOPMENT

GP - 35
4-23-93

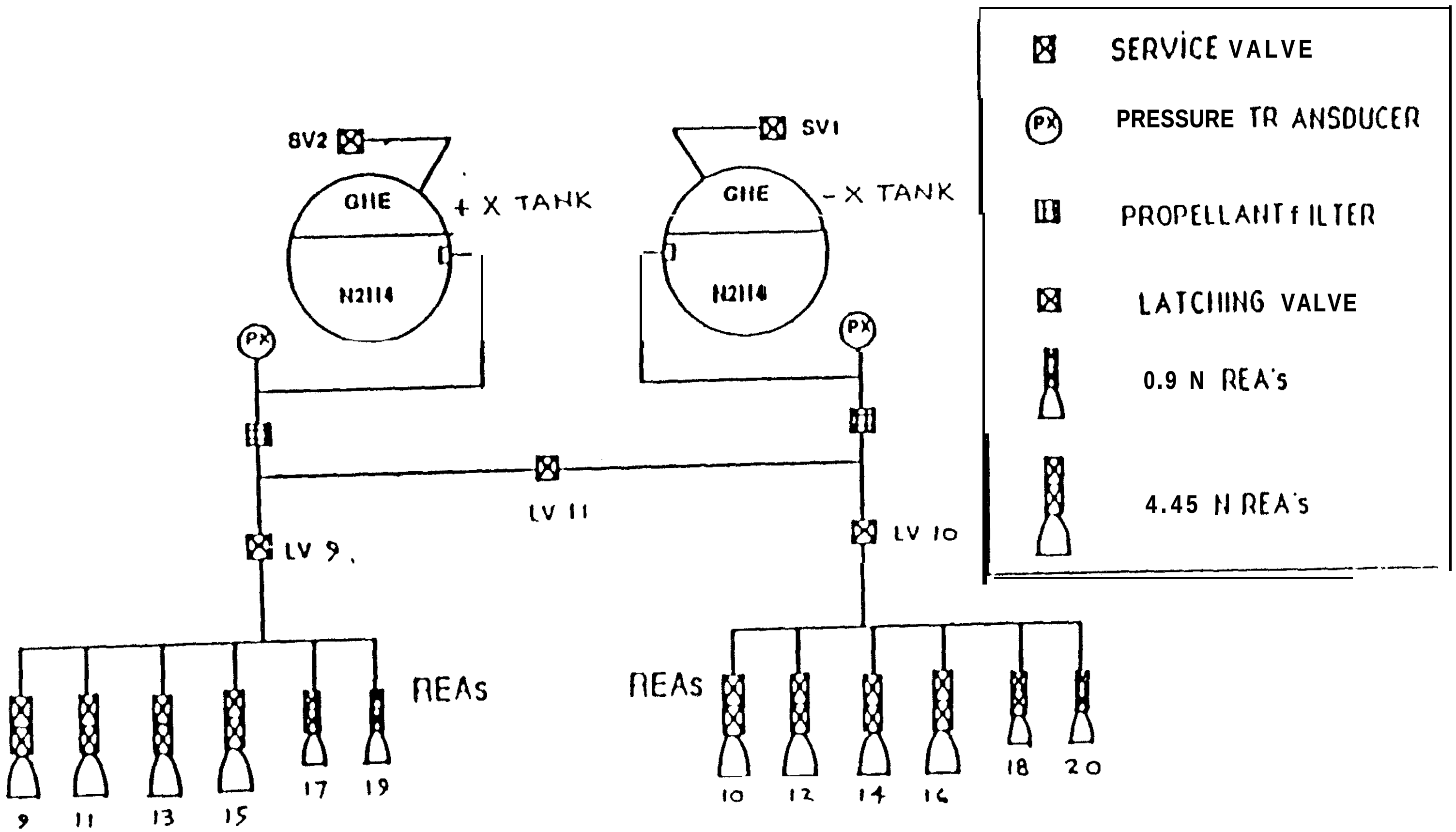
JPL

BI-PROPELLANT SUBSYSTEM BLOCK DIAGRAM



JPL

MONO-PROPELLANT SUBSYSTEM BLOCK DIAGRAM

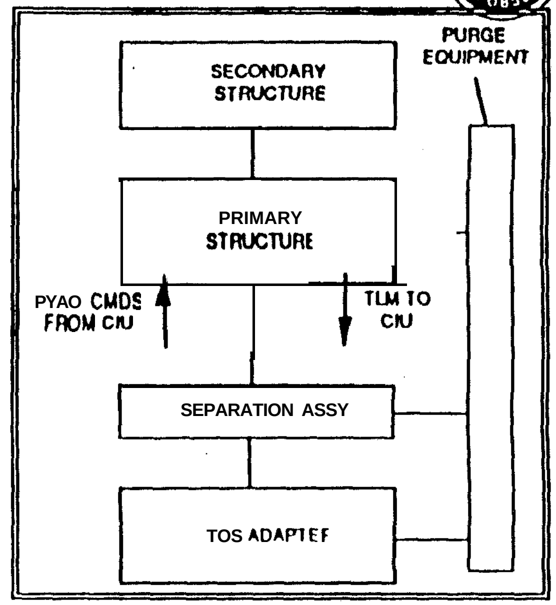


PROPULSION SUBSYSTEM

- **DESIGN DRIVERS**
  - MULTIPLE MISSION MANEUVER REQUIREMENTS: TCM, MOI, OTM, WHEEL UNLOADING
  - CONTAMINATION: MONO-PROP IN ORBIT, NO THRUSTERS ON NADIR PANEL
- **CHARACTERISTICS**
  - MONO-PROP: 84 KG PROPELLANT, 8 x 4.5 N & 4 x .9 N THRUSTERS
  - BI-PROP: 1363 KG (2.7 KM/SEC), 4 x 490 N & 4 x 22 N THRUSTERS
- **DEVELOPMENT HIGHLIGHTS**
  - NUMBER OF TANKS REDUCED TO SIMPLIFY DESIGN
  - ORIGINAL 490 N DEVELOPMENT FAILED, ALTERNATE SOURCE SELECTED
  - PROPELLANT MANAGEMENT DEVICE (PMD) REDESIGNED TO PREVENT SCREEN TEARING
  - TANK PMD TAB BROKEN DUE TO STRESS CORROSION FROM CONTAMINATED FREON
  - LEAKING He TANK REPLACED BY SPARE
  - PARALLEL BI-PROP FILTERS ADDED
  - SUBSYSTEM COMPONENTS DELETED DURING MASS REDUCTION REDESIGN
  - WAIVER REQUIRED FOR MONO-PROP CONTINGENCY OFF-LOADING
  - BI-PROP VALVE REQUALIFIED
  - BLOWDOWN MODE IMPLEMENTED TO AVOID POTENTIAL REGULATOR LEAK
  - PLUME SHIELDS ADDED LATE TO PREVENT BLANKET EROSION

JPL

STRUCTURE



FUNCTIONS

PROVIDE FOR STRUCTURAL MOUNTING OF ALL ASSEMBLIES
PROVIDE STABLE MECHANICAL INTERFACE FOR ALL SENSORS
ENSURE STRUCTURAL INTEGRITY FOR ALL **MISSION** PHASES
PROVIDE BASIC **SYSTEM** ALIGNMENTS
PROVIDE FOR UNOBSTRUCTED SENSOR FOV
PROVIDE CENTER OF MASS CONTROL

REQUIREMENTS

COMPATIBLE **WITH TITAN III**
ACCOMMODATE 166 KG OF PAYLOAD
OPTICAL ALIGNMENT OF REFERENCE MIRRORS TO PRIMARY MIRROR
PIN AND BOLT INSTRUMENT ATTACHMENT.

STRUCTURE COMPONENTS

PRIMARY STRUCTURE:

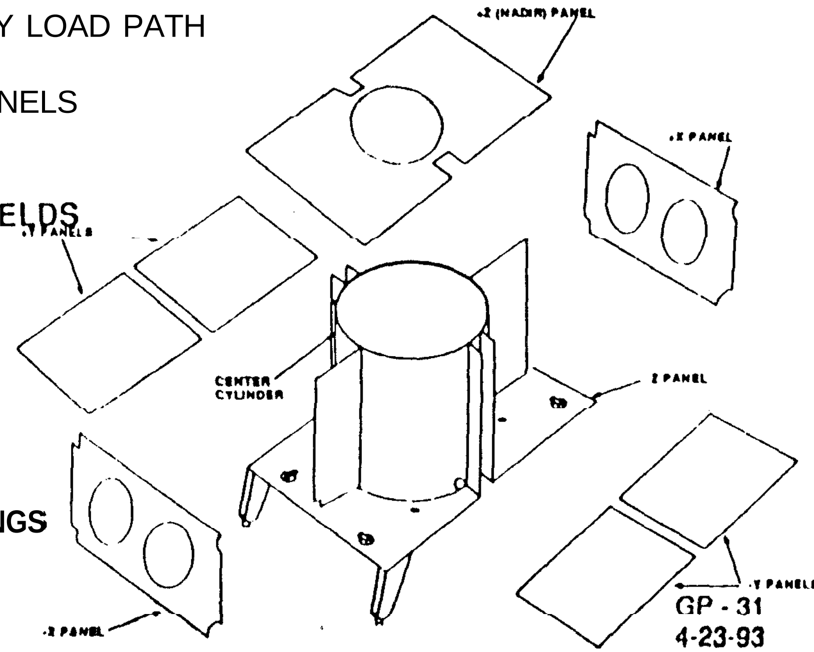
MAGNESIUM ALLOY CENTER CYLINDER FOR PRIMARY LOAD PATH
2.1 x 1.5 x 1.0 M RECTANGULAR MODULE
8 MODULAR ALUMINUM HONEYCOMB EQUIPMENT PANELS

SECONDARY STRUCTURE:

THRUSTER SUPPORT BRACKETS, HEAT **& PLUME** SHIELDS.
TANK SUPPORT BRACKETS
PURGE LINE, HARNESS, **&** THERMAL SUPPORTS
LGA SUPPORT BRACKETS

TOS ADAPTER:

MONOCOQUE WITH SKINS AND STRINGERS
BOUND BOLT INTERFACE WITH TOS
"V" BAND CLAMP **ASSEMBLY WITH RETENTION SPRINGS**
SPRING SEPARATION



GP - 31
4-23-93

STRUCTURE SUBSYSTEM

- **DESIGN DRIVERS**
  - LAUNCH VEHICLE LOADS & INTERFACE ENVELOPE
  - WEIGHT CONSTRAINED
  - SUBSYSTEM & PAYLOAD FIELDS-OF-VIEW
  - CENTER-OF-MASS CONTROL
- **CHARACTERISTICS**
  - CENTRAL CYLINDER FOR BI-PROP TANKS AND PRIMARY LOAD PATH
  - MODULARIZED EQUIPMENT PANELS
  - ALUMINUM WIRE MAIN HARNESS
- **DEVELOPMENT HIGHLIGHTS**
  - LAUNCH VEHICLE CHANGED FROM STS TO TITAN III
  - INSTRUMENT COMPLEMENT & MASS CHANGED
  - SEVERE MASS REDUCTION EFFORT AFTER SYSTEM CDR
  - BOUND BOLT INTERFACE BETWEEN ADAPTER AND UPPER STAGE (TOS) REQUIRED ALOT OF EFFORT
  - ALUMINUM HARNESS HAD NO PROBLEMS WHEN HANDLED CAREFULLY