THE INTERPLANETARY PIONEERS

VOLUME II: SYSTEM DESIGN AND DEVELOPMENT

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

NASA SP-279

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VOLUME II: SYSTEM DESIGN AND DEVELOPMENT

by

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Scientific and Technical Information Office 1972
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Washington, D.C.

For sale by the Superintendent of Documents,
U.S. Government Printing Office, Washington, D.C. 20402
Price $2.50 Stock Number 3300-0452
Library of Congress Catalog Card Number 74-176234

Foreword

SOME EXPLORATORY ENTERPRISES start with fanfare and end with a quiet burial; some start with hardly a notice, yet end up significantly advancing mankind's knowledge. The Interplanetary Pioneers more closely fit the latter description. When the National Aeronautics and Space Administration started the program a decade ago it received little public attention. Yet the four spacecraft, designated Pioneers 6, 7, 8, and 9, have faithfully lived up to their name as defined by Webster, "to discover or explore in advance of others." These pioneering spacecraft were the first to systematically orbit the Sun at widely separated points in space, collecting information on conditions far from the Earth's disturbing influence. From them we have learned much about space, the solar wind, and the fluctuating bursts of cosmic radiation of both solar and galactic origin.

These Pioneers have proven to be superbly reliable scientific explorers, sending back information far in excess of their design lifetimes over a period that covers much of the solar cycle.

This publication attempts to assemble a full accounting of this remarkable program. Written by William R. Corliss, under contract with NASA, it is organized as Volume I: Summary (NASA SP–278); Volume II: System Design and Development (NASA SP–279); and Volume III: Operations and Scientific Results (NASA SP–280). In a sense it is necessarily incomplete, for until the last of these remote and faithful sentinels falls silent, the final word is not at hand.

HANS MARK
Director
Ames Research Center
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Contents

CHAPTER 1

Defining the Pioneer System

MOTIVATION FOR PIONEER

THE SCIENTIFIC MISSION of Pioneers 6 through 9 was the synoptic measurement of the interplanetary milieu as it is affected by the Sun. The Pioneers measured and transmitted back to Earth data on solar plasma, cosmic radiation, magnetic and electric fields, and the specks of cosmic dust that drift through interplanetary space. All of these physical phenomena are dominated by the Sun. The Pioneer spacecraft described here were akin to weather satellites, except that they were artificial planets of the Sun rather than satellites of the Earth. Spotted strategically around the Sun in the plane of the ecliptic, they monitored the ever-changing fluxes and fields that wax and wane with solar activity.

Solar activity follows an eleven-year cycle of sunspot numbers—a periodic phenomenon felt throughout the solar system. In 1962, when NASA began to formulate its "follow-on" Pioneer Program, which would extend the earlier International Geophysical Year (IGY) Pioneers (Pioneers 1–5), scientists around the world were organizing an investigation of solar problems to take place during the solar minimum expected during the 1964–1965 period. They hoped to further the scientific advances recorded during the IGY (18 months in the span 1956–1958), a period that also saw the first satellites and the formation of NASA. The new effort was labeled the International Quiet Sun Year (IQSY). The five Pioneers planned in 1962 would be in direct support of the IQSY, supplementing NASA's Orbiting Geophysical Observatory (OGO) series, Orbiting Solar Observatory (OSO) series, and Explorer series in orbit around the Earth, and a worldwide array of scientific sensors on the ground and on sounding rockets. The Pioneers' unique value to the IQSY lay in the fact that they would range far ahead and behind the Earth as it swung around the Sun. Further, they would make *in situ* measurements of deep-space phenomena, unperturbed by the Earth's magnetic and gravitational fields.[1]

As the IQSY or Interplanetary Pioneer Program developed, it became apparent that the long lifetimes of the spacecraft and the schedule changes would extend deep-space solar monitoring through the 1969–1970 solar

[1] See Volume I for the detailed scientific objectives of the Pioneer Program, and Volume III for a summary of scientific results.

maximum—a scientific bonus. This extension of coverage also aided the Apollo lunar exploration effort. Scientists, as they take the Sun's pulse with their manifold instruments, are beginning to predict solar activity in much the same way that the Weather Bureau predicts tornado and hurricane activity from its array of terrestrial meteorological sensors. A violent storm on the Sun could endanger astronauts on the Moon with an intense burst of solar cosmic rays. The prediction of a severe solar disturbance a few hours ahead of time would give astronauts time to take shelter in the relative safety of their spacecraft.

Weather prediction of any kind is more reliable if data can be obtained from widely separated sites. The "inward" Pioneers (6 and 9) and "outward" Pioneers (7 and 8) led and lagged the Earth by tens of millions of miles, respectively, providing a much broader data base than terrestrial sensors. Selected scientific parameters from the Pioneer spacecraft were teletyped from tracking and data acquisition sites to Ames Research Center, forty miles south of San Francisco, where they were processed and analyzed prior to transmission to the Space Disturbance Center of Environmental Science Services Administration (ESSA), at Boulder, Colorado. After combining Pioneer data with that from other sensors, in space and on the ground, ESSA issued daily Space Disturbance Forecasts. These forecasts not only alerted astronauts, but also signaled scientists around the world that interesting events were about to happen on the Sun. Solar weather monitoring was not one of the original objectives of the Pioneer Program, but the obvious value of Pioneer deep-space data led to its use in preparing the Space Disturbance Forecasts.

DESIDERATA AND CONSTRAINTS: SOME EARLY THOUGHTS

The Pioneer mission as defined above was far too general to enable engineers to sit down and draw up a system design. A wide variety of spacecraft, weighing pounds or tons, costing millions or billions, could monitor interplanetary weather. In 1962, the practical considerations of money and available launch vehicles dictated that the spacecraft weigh only about 100 lb. The investment of resources had to be commensurate with the potential scientific payoff and not detract from NASA's major mission, the manned lunar landing. In this light, the Pioneers were closely related to the small Explorer-class satellites that NASA launches on geophysical missions.

As the scientific desiderata were defined more closely, the main engineering features of the Pioneers began to come into focus. As instrument carriers, scientists wanted the Pioneers to have:

(1) The ability to point instruments in all directions, particularly all azimuths in the plane of the ecliptic

(2) Capability for continuous data sampling of the experiments

(3) High data transmission rates back to Earth

(4) Many commandable modes of operation to permit them to modify their instruments from Earth

(5) A stable environment for the instruments without temperature extremes, electromagnetic interference, or other perturbing forces

(6) A very low residual magnetic field that would not obscure the slight interplanetary magnetic fields

(7) A long reliable life, preferably a year or more

(8) A maximum spacecraft penetration toward and away from the Sun

(9) A wide variety of scientific instruments to measure the many interrelated features of interplanetary space

Vannevar Bush once described science as "an endless frontier." In this context, it would be desirable to carry a hundred instruments, approach the Sun to within a tenth of an Astronomical Unit (AU), and take excursions from the plane of the ecliptic. Such objectives were obviously beyond the scope and assigned resources of the IQSY Pioneers, though certainly not beyond man's increasing capabilities in space. Despite the limitations, the final spacecraft were excellent instrument platforms that more than fulfilled all scientific objectives.

With the Pioneer Program thus characterized as a modest effort, available launch vehicles and proven technology had to be applied. The engineers on the Pioneer Program did not attempt to make technological breakthroughs. In addition, the horizons of the Pioneers had to be limited by available tracking and data acquisition facilities. The basic factor creating these constraints was, of course, cost. Scientific return had to be maximized within a framework of resources in the $50 to $100 million category.

Every engineer recognizes the trade-off problem just posed; that is, maximizing performance within fixed technological and financial constraints. The detailed trade-offs and design philosophy employed during the evolution of the Pioneer spacecraft will be described in later chapters. Here, only the identification of the major elements of the overall Pioneer system is important. The resources of NASA in 1962 allowed the Pioneer Project the following system elements (fig. 1–1):

(1) The Delta launch vehicle, a low cost, highly reliable rocket, capable of propelling about 150 pounds into orbit around the Sun, available, and well proven in many satellite launches.

(2) The Deep Space Network (DSN), comprising the Deep Space Instrumentation Facility (DSIF) of tracking and data acquisition antennas and the Space Flight Operations Facility (SFOF) at the Jet Propulsion Laboratory (JPL) in Pasadena, California. The DSN was built for NASA's planetary and lunar programs. It was the only NASA tracking and data data acquisition network, capable of handling a small space probe tens of millions of miles from the Earth, and a reality that helped shape the Pioneer Program.

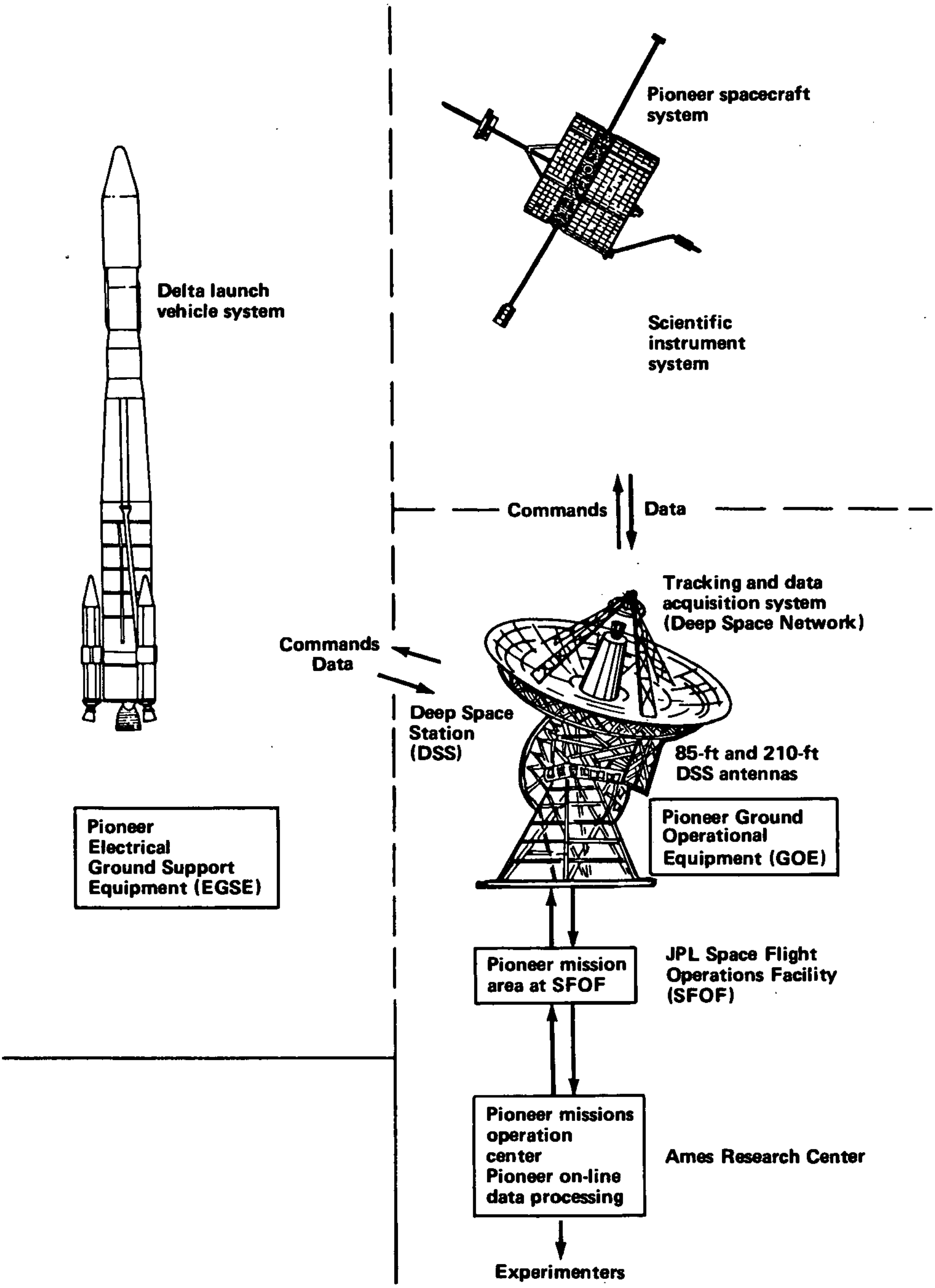

FIGURE 1-1.—The four Pioneer systems.

(3) The spacecraft system, a stable instrument carrier weighing something around 100 pounds, capable of communicating with experimenters through the DSN, and providing room for about 20 to 40 lb of scientific instruments.

(4) The scientific instrument system, made up of magnetometers, particle detectors, and whatever other instruments the scientific community deemed feasible and worthwhile within the payload limitations set by the launch vehicle, orbit, and spacecraft weight requirements.

Obviously, this is only the crudest sketch of the Pioneer System—about as far as one could go at that time, given NASA's resources and the broad scientific objectives. This was the starting point. To go further, someone needed to express things in numbers or "engineer" the system.

The purpose of this volume is to describe how this engineering was performed, what the critical design decisions were, and what the final system looked like. Pioneer operations and scientific results are related in Volume III.

A HIERARCHY OF SYSTEMS

Before a description of the Pioneer feasibility study and subsequent design, fabrication, and test activities, a model of the spacecraft program is desirable. This model should not only define the various equipments and how they mesh physically, but also how the spacecraft project moves through the time dimension from feasibility study to launch pad.

The Pioneer spacecraft with about 20 lb of scientific instruments may be likened to the apex of a large pyramid. The small point of the pyramid depends completely upon the large supporting foundation. In this analogy, the base of the pyramid is represented by the launch vehicle, the ground support equipment, the test facilities, and the multifarious activities involved in the design, construction, and operation of the spacecraft. The spacecraft receives the fanfare, but thousands of people on the ground and hundreds of millions of dollars worth of facilities are also essential to success.

In the formal language of engineering, the complete system begins at the spacecraft sensor and ends with the publication of the scientific results in the literature. The major elements in the overall Pioneer System are the spacecraft itself, its cargo of scientific instruments, the launch vehicle, and the tracking and data acquisition network, as diagrammed in figure 1–1. Because these elements are frequently called systems in their own right, it is more proper to refer to them overall as the Pioneer supersystem. Nevertheless, to adhere to Pioneer Program terminology, the Pioneer System will be understood to consist of four lesser systems. Each of the four Pioneer systems can be further subdivided into subsystems, such as the spacecraft communication subsystem. These distinctions may seem overly complicated, but it is important to sketch out a general framework for the descriptive material that follows. And, of course, from the standpoint of program management, the supersystem work must be parceled out to engineering groups in conveniently sized systems, subsystems, and even smaller pieces.

Engineers like to subdivide large supersystems into smaller pieces because this dissection helps them to see and to understand the inner workings of each part. The problem is putting the pieces back together again. The designer of the spacecraft's communication subsystem cannot ignore the antenna of the terrestrial data acquisition equipment, even though it may be designed by a company in another part of the country. Boundary regions between subsystems and systems are termed interfaces. The proper matching of interfaces is vital to the successful operation of the complete supersystem.

The management device used to ensure matching interfaces in many NASA programs is called the interface specification, a carefully written description of how various subsystems must fit together. The interface between a Pioneer spacecraft far out in space and the DSIF back on Earth involved such matters as radio frequencies, the type of telemetry employed, and the many other aspects of radio communication. The Pioneer supersystem was, in fact, a huge, remotely controlled, information-gathering machine. It is not surprising to find that communication and information interfaces existed between almost all systems and subsystems. A mechanical

TABLE 1–1.—*Types of Interfaces in Spacecraft Systems*

Type	Design considerations
Mechanical	Physical dimensions of mating parts must match. Shock and vibration may cause damage during launch.
Spatial	Competition for solid angle (view cones) by scientific instruments, solar cells, and navigation sensors.
Thermal	Heat flow across interfaces may degrade equipment; viz, aerodynamic heating during launch.
Electrical	Voltages, currents, and ac frequencies must match. The summation of the power profiles of equipment as functions of time must not exceed power subsystem capacity.
Magnetic	Magnetic materials and current loops must not interfere with magnetic instrumentation.
Electromagnetic	Crosstalk between neighboring circuits is a common spacecraft problem.
Radiative	Particulate radiation from nuclear power supplies or the environment may interfere with instrumentation or, in extreme cases, damage materials and components. (Not a consideration on the IQSY Pioneers.)
Information	Data flow across interfaces must be matched in terms of word format, the rate of data transmission, etc.

interface obviously must be matched when the launch vehicle and spacecraft come together at the launch pad. Within the spacecraft itself, the interfaces are more subtle, as illustrated in table 1–1. Magnetic cleanliness, for example, was a major goal on the Pioneer spacecraft. This requirement led to the establishment of magnetic interface specifications stipulating the maximum magnetic fields tolerable from each spacecraft subsystem. In the large sense, the spacecraft systems also had to be matched to the environment; that is, environmental forces, such as solar plasma, could not degrade spacecraft performance. The interface specifications strongly influenced the design of each component on the spacecraft and on the ground.

A MODEL OF THE SPACECRAFT SYSTEM

The definition of the spacecraft subsystems is essential to the understanding of the chapters that follow. Subsystem definition varies somewhat from design group to design group. Generally, one attempts to lift out a well-defined piece of equipment with well-defined functions, and label it a subsystem. The electric power subsystem is a typical subsystem. After the electric power subsystem, the communication subsystem, and the other "removable" subsystems listed in table 1–2 are extracted, only the structure subsystem is left. The structure subsystem is the shell and/or framework that holds the other subsystems in place. Its design is just as critical to success as any other subsystem.

TABLE 1–2.—*Definition of Pioneer Spacecraft Subsystems*

Subsystem	Functions performed
Communication	Relays scientific and spacecraft status data from the spacecraft to Earth; receives commands from Earth.
Data-handling	Accepts data from scientific and housekeeping instruments and arranges them in proper format for transmission back to Earth; provides for limited data storage.
Electric-power	Provides electrical power to all spacecraft subsystems and the scientific instrument system.
Orientation	Orients the spacecraft spin axis as required; damps out wobble. Attitude sensors and gas jets are included within this subsystem under Pioneer Program terminology.
Thermal-control	Maintains temperatures within specified ranges within the spacecraft.
Command	Decodes and distributes commands received via the communication subsystem to the spacecraft subsystems specified in the command addresses.
Structure	Supports and maintains spacecraft configuration under design loads; provides booms for instrument isolation.

The Pioneer spacecraft subsystems delineated in figure 1–2 and table 1–2 are fairly consistent with nomenclature in other spacecraft projects. The major differences are as follows:

(1) The Pioneer scientific instruments were considered to constitute a full-scale system by themselves and not a spacecraft subsystem as on many Earth satellites.

(2) An onboard data-handling subsystem was separated from the communications subsystem.

(3) The attitude-control subsystem was termed the "orientation subsystem" on Pioneer spacecraft.

(4) Onboard propulsion and centralized onboard computer subsystems were not needed on Pioneer spacecraft.

(5) Housekeeping sensors were included within each subsystem rather than considered collectively as a separate subsystem.

Interfaces had to be matched between each of the seven subsystems portrayed in figure 1–2 and table 1–2. Almost all of the Pioneer spacecraft subsystems required electrical power, and most also exchanged data and commands with the data handling and command subsystems. All subsystems had to fit together mechanically. (This is not so elementary a problem as it seems. Each spacecraft contains tens of thousands of parts, and cases have occurred where parts did not mesh properly the first time.) In view of spacecraft complexity, interface specifications were both voluminous and indispensable.

THE PROPER ORDER OF THINGS

Equipment specifications stipulate what the equipment should be like; interface specifications insure that the various pieces of equipment will fit and work together satisfactorily. The omitted dimension is time. The flow of project events is specified by a milestone series familiar to every engineer and project manager.

As related in Volume I, the Pioneer Project began informally as a concept in Ames Research Center, NASA Headquarters, and industry during 1962. After considering the broad scientific objectives and its available resources, NASA management selected the major features of the Pioneer Project in 1962, as described in the preceding sections. However, many features of the spacecraft and mission-dependent equipment remained undefined. The next logical step was a feasibility study. The Pioneer feasibility study was made at Space Technology Laboratories, Inc. (STL)[2] and it went much further than the confirmation of feasibility; many design decisions were made and the spacecraft and other systems took on more detailed focus.

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[2] Later renamed TRW Systems (July 1, 1965).

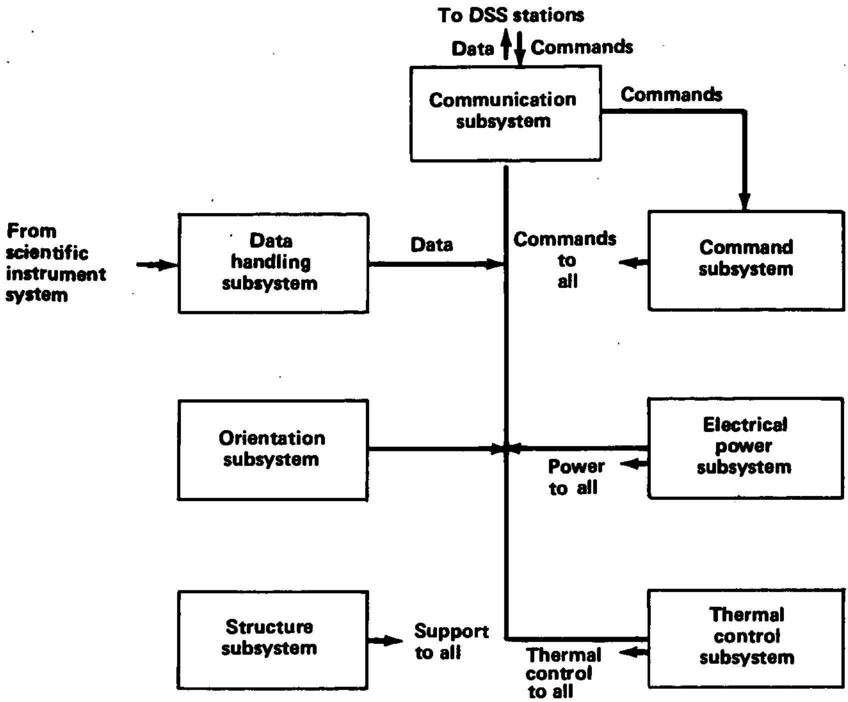

FIGURE 1–2.—Generalized block diagram showing Pioneer spacecraft subsystems. Magnetic, thermal, and other forces crossing subsystem interfaces are not shown.

With the feasibility study as a basis, Ames Research Center was able to draw up specifications to serve as the basis for hardware contracts. Two mainstreams of activity began with project approval on November 9, 1962; one stream each for the spacecraft and the scientific instruments. The evolution of the two other major systems, the Delta launch vehicle and the DSN, were not dictated by the Pioneer Program. The spacecraft and scientific instruments progressed through the phases of:

(1) Contractor competition and selection
(2) Detailed design and development
(3) Hardware procurement and fabrication
(4) Testing
(5) Integration and checkout

The spacecraft, its instruments, the Delta, and DSN utilization ultimately converged at Cape Kennedy at the time of launch, when the four systems were integrated and checked out as a single supersystem. The Pioneer mission-dependent equipment, including the spacecraft and its scientific instruments, was completely new; while the mission-independent equipment (the DSN and Delta) required what are termed "project-unique" modifications and auxiliaries. The general flow pattern of the Pioneer Project is illustrated for the first spacecraft, Pioneer 6, in figure 1–3. A major task of NASA Project management was the coordination of these four more-or-less parallel streams of effort. Specifications, schedules, and review meetings were the primary management tools employed in assuring that the four-way confluence was a successful one. The four straight Pioneer successes testify to the excellence of both engineering and management, in and out of the government, between Project approval in 1962 and the final launch in the series in 1969.

A LOOK AT THE STL FEASIBILITY STUDY

Before NASA could embark upon a full-scale hardware program, it required a more precise definition of the Pioneer System. The general objectives and the rough delineation of major system components described in the preceding sections had to be confirmed by a hard-headed preliminary engineering design and then sketched out in more detail. The 1962 Pioneer feasibility study performed these tasks.

Feasibility studies are common in aerospace projects. The essentials of a system have to be known before realistic cost and schedule estimates can be made. If the Pioneer Program actually proved feasibile within the

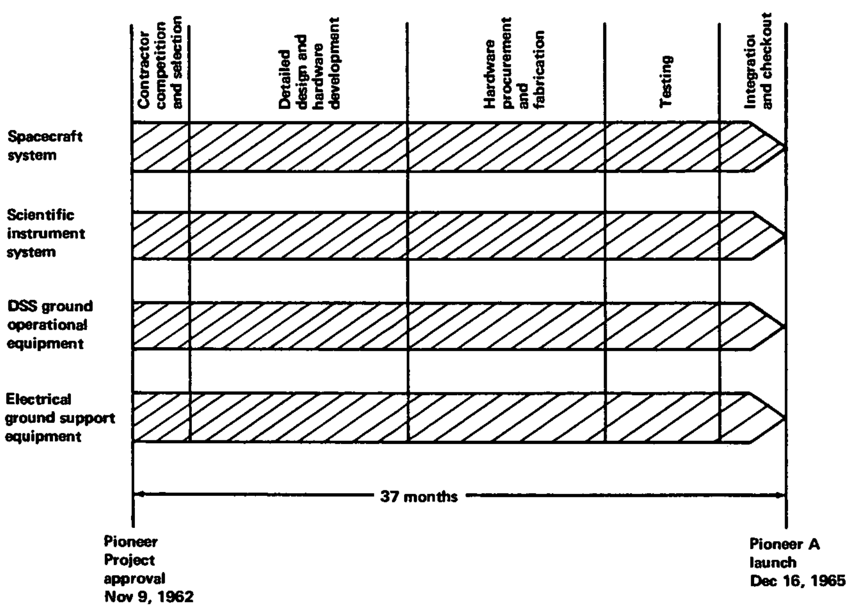

FIGURE 1–3.—Activity flow for the four Pioneer systems (shown for Pioneer A). Subsystems within each system followed similar paths. The phases were not synchronized precisely.

limitations of its resources, NASA intended to use the feasibility study as a basis for a Request for Proposal (RFP) which, in turn, would lead to contracts for the construction and testing of the spacecraft, the instruments, and the other project-unique equipment. This was, in fact, exactly what happened during 1962 and 1963.

In April 1962, STL completed a 2½-month study of an Interplanetary Probe under contract NAS2–884. That study was the basis for the IQSY Pioneer Program and the spacecraft now called Pioneers 6 through 9. Armed with the STL study, Ames Research Center issued RFP A–6669 on January 29, 1963. STL won the final competition for the design, development, and construction of the spacecraft and certain ground-support equipment. NASA signed a letter contract with STL on August 4, 1963, and the Pioneer Program began to move out of the paper-study phase. The definitive contract with STL was not signed until May 1964. During the nine-month letter contract phase, STL and Ames engineers refined the spacecraft design considerably, making what might be called a "second iteration" on the design presented in the feasibility study.

The Pioneer feasibility study is especially important because, during the 2½ months in early 1962, almost all of the important system-design decisions were made by STL engineers working in conjunction with Ames personnel. A brief review of the most significant of these system-wide decisions is in order, for they refined considerably the definition of all systems and subsystems, and also revealed how well the general scientific objectives could be met within the scope of NASA's resources.

SOME CONSTRAINTS ON THE FEASIBILITY STUDY

At the beginning of the feasibility study, NASA and STL personnel had at hand the general objectives and constraints mentioned earlier, the most important of which stipulated the use of the Delta (Thor-Delta) launch vehicle, the DSN, and, as far as possible, proven hardware. Applying numbers to its mission objectives, NASA specified that the Pioneer spacecraft should have a minimum probability of success of 0.8 for a 6-month life, with no absolute upper limit, and that it should be able to operate between 0.8 and 1.2 AU without spacecraft modifications.

The feasibility study proceeded on the basis of a contractual go-ahead on October 1, 1962, and a first flight in July 1964. Three other launches would follow at 6-month intervals and sufficient spare parts would be built for a fifth spacecraft. Furthermore, NASA imposed a fiscal constraint: the rate of cost buildup during the first 6 months of the program was not to be more than 15 percent of the total program cost.

The tight schedule, the desire to minimize costs, and the high spacecraft-reliability target defined a modest total program with a very simple spacecraft built from off-the-shelf components.

A NEW FLIGHT CONCEPT

The only significant experience U.S. astronautical engineers had with interplanetary spacecraft prior to early 1962 was with Pioneer 5 (launched June 26, 1960) and Mariner 2 (launched in August 1962). The Mariner was a sophisticated spacecraft, with solar panels that could be oriented toward the Sun and a high-gain, directional communication antenna that would point toward DSN antennas back on Earth. Weighing 447 pounds, Mariner 2 was too complex and too expensive under the Pioneer ground rules. Pioneer 5, on the other hand, was merely spin-stabilized in outer space and could not face its solar paddles to the Sun while rotating. It possessed an omnidirectional antenna that radiated radio energy wastefully in all directions. Still, Pioneer 5 had operated successfully for 2½ months (a good record in 1960) and had sent back signals from 22 million miles. At 75 lb, Pioneer 5 was a simple spacecraft and much closer than Mariner 2 to NASA's concept of the IQSY Pioneers. STL had built Pioneer 5; this was one reason why STL was awarded the feasibility study and, ultimately, the IQSY Pioneer spacecraft contract.

Weight and simplicity dictated a spin-stabilized spacecraft (almost all early U.S. spacecraft were spin-stabilized for this same reason). Spacecraft stabilization with gas jets and/or gyros was weight-consuming and risky from the reliability standpoint. Spin-stabilization also has the advantage of rotating the scientific instruments frequently through all azimuths. However, uncontrolled spin-stabilization entailed three problems:

(1) An ordinary dish-type directional antenna would be aimed at the Earth only once each rotation, a fact that would militate against achieving high data flow rates (high bit rates) over tens of millions of miles.

(2) The scientists preferred to have their instruments scan in the plane of the ecliptic, not any of the infinite number of other planes possible with a randomly oriented spacecraft.

(3) If the spin vector of the spacecraft was to be random, solar cells would have to be mounted on all sides of the spacecraft.

These thoughts led to the concept of an orientable, spin-stabilized spacecraft, with a spin axis that could be swung around with a simple gas jet until it was perpendicular to the plane of the ecliptic. The laws of motion predicted that torquing the spin vector would cause spacecraft precession or wobbling, but this could be largely eliminated by installing a simple "wobble damper." If the spacecraft were a cylinder (the preferred shape for many scientific spacecraft), with instruments and solar cells mounted around its curved surface, the last two of the three problems would be solved. The scientific instruments would scan in the plane of the ecliptic and solar cells would not be needed on the flat ends of the cylinder, freeing them for other components.

Only the antenna problem would remain. A paraboloidal spacecraft antenna sweeping the plane of the ecliptic would be wasteful of spacecraft power, as would an omnidirectional antenna. A rather inspired solution to this conundrum was a mast-like antenna (a modified Franklin array) mounted along the spin axis. This kind of antenna would radiate a flat fan of radio energy in the plane of the ecliptic. The Earth and its DSN paraboloidal antennas would always be in this fan if the spacecraft spin axis were properly oriented. Of course, some radio energy would be wasted in other spacecraft azimuths; but the antenna fan was so narrow ($\pm 5°$ to the 3 dB points) that it was much less wasteful (10 dB), better than an omindirectional antenna, and still far simpler than a pointable-dish antenna, such as that on Mariner 2. The combination of the unusual antenna plus spin stabilization perpendicular to the plane of the ecliptic were the key design decisions that conferred high reliability and the capability of very long distance communication on the IQSY Pioneers.

It is interesting to note how completely these elementary considerations define the spacecraft configuration (fig. 1–4). It had to have cylindrical symmetry (for spin stability); and it had to have a conspicuous antenna

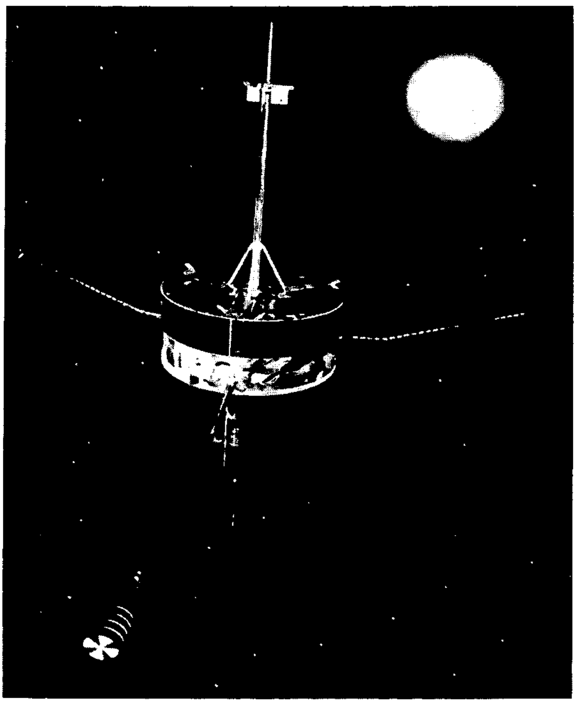

FIGURE 1–4.—View of the Pioneer spacecraft showing the three radial booms deployed, the telemetry antenna mast (top), and the Stanford radio propagation experiment antenna (bottom).

mast at one end. Actually, these early design decisions were felt throughout the entire spacecraft design period and at every subsystem level. The effects were important at the systems level, too. The instrument system was aided by the spinning spacecraft platform; the high-gain antenna made the DSN's task easier; and the axis of spin symmetry simplified the dynamic interface with the Delta launch vehicle.

ESCAPING THE EARTH'S GRAVITATIONAL FIELD

The STL feasibility study considered a launch from Cape Kennedy into the plane of the ecliptic. Cape Kennedy approaches to within about 5° of the plane of the ecliptic once each day. Propulsive requirements for Earth escape into the plane of the ecliptic are minimum at this time; however, the rocket must be fired in a given direction precisely at this moment. Payload can be traded for rocket propellant to gain the desired flexibility in launch time and direction. Rough calculations showed that a 9-lb payload penalty would permit a launch any time of day, in any direction from the Cape. (See ch. 2 for trajectory details.) Another launch trajectory trade-off concerned the altitude of main engine cutoff (MECO). Low-altitude MECOs (around 180 000 ft) produced important aerodynamic heating that required insulation on the Delta stages. The weight of this insulation reduced the payload. High-altitude MECOs demand less insulation, but payloads are again reduced because of higher propulsive requirements. These tradeoffs were investigated in detail in the feasibility study. The general conclusion was that there was ample margin in the Delta capabilities for launching a payload of about 126 lb.

Also examined was the possibility of exchanging the X–248 Delta third stage for the somewhat better X–258. Essentially NASA was offered a choice between a little more payload and less reliability with the newer stage. (This matter will be brought up again in ch. 7 because NASA ultimately did switch from the X–248 to the X–258 for Pioneer 6.)

COMMUNICATIONS RELIABILITY

In the feasibility study, STL calculated a 0.83 probability of a 6-month life for the spacecraft, once in orbit. To attain this level of reliability, STL engineers employed redundancy, particularly for such critical communication components as the traveling wave tube, receiver, decoder, and elements of the digital telemetry unit. Conservative selection of parts was also a factor. For example, the traveling wave tube was selected over the more efficient but relatively new amplitron because reliability data were lacking for the latter. There was also doubt that a sufficiently reliable amplitron could be delivered in time for the first flight. Further, the amplitron's stray magnetic field might have compromised magnetometer experiments in the payload.

GETTING RID OF WASTE HEAT

NASA wished to send the IQSY Pioneers on solar orbits both inside and outside that of the Earth. The accommodation of different thermal environments, without redesigning the spacecraft, dictated an active thermal control subsystem; that is, one with temperature-controlled vanes or louvers rather than static schemes employing fixed patterns of different surface absorptivity and emissivity. The relatively large variations in internal heat generation due to the variable transmitter power also added impetus to the choice of an active thermal control subsystem. The logical spot to install the vanes was on the bottom of the spacecraft which was unencumbered by solar cells or antenna. From here the spacecraft could radiate the waste heat directly to the cold sky seen perpendicular to the plane of the ecliptic.

ONBOARD DATA STORAGE

Although the feasibility study did not absolutely recommend onboard data storage, the subject was considered carefully and left an option for NASA—an option that NASA did take.

Most Earth satellites carry tape recorders which are read out whenever the satellite passes over a data-acquisition site. A tape recorder allows instruments to record continuously when the satellite is out of sight of a station. The DSN has several 85-ft and 210-ft paraboloidal antennas suitable for Pioneer data acquisition at various sites around the world (see ch. 8 for list), but often they are busy on high priority programs, such as manned and unmanned lunar spacecraft. As a result, a data handling subsystem with limited data storage appeared in the final spacecraft design.

AN EARLY WEIGHT BREAKDOWN

Of course, the feasibility study went into much more detail than the preceding paragraphs indicate. The chapters covering the various subsystems will trace the design from the feasibility study through final fabrication. The intent here has been to introduce the reader to more general considerations and the major design decisions that were made during the feasibility study.

The IQSY Pioneer spacecraft emerging from the feasibility study had the same basic geometry as the final flight versions, except that booms were installed to isolate instruments in the flight models (fig. 1–4). The feasibility study's weight breakdown is presented in table 1–3.

TABLE 1–3.—*Spacecraft Weight Breakdown; STL Feasibility Study*

Subsystem or component	Weight
Structure	17.4
Communications	27.6
Electrical system	15.8
Reorientation system	8.4
Temperature control	3.2
Solar cell array	17.7
Balance weights	1.5
	91.6 lb
5 percent contingency	4.6
	96.2 lb
Experiments, power conversion and cabling	20.0
	116.2 lb
Delta interstage structure	9.5
Total	125.7 lb

EXPERIMENTS SUGGESTED BY STL

The IQSY Pioneers were considered precursor instrument carriers for the purposes of the feasibility study. The thought at NASA at that time was that more sophisticated space vehicles carrying better, more precise instruments would follow once the Pioneers blazed a path and radioed back a rough picture of the interplanetary domain. Because the interplanetary environment was only known imperfectly, the experiments were designed with a high dynamic range rather than high precision. As mentioned previously, instruments strongly affect spacecraft design, particularly in the matters of scanning, communication, and power requirements. Furthermore, the command subsystem and experiments should possess sufficient flexibility to allow experimenters to step up the sampling rates for instruments recording solar plasma, solar radiation, and magnetic fields during periods of high solar activity. In other words, the Pioneers were not to be regarded as passive instrument platforms set adrift on the interplanetary sea, but rather flexible arrays of instruments responsive to experimenters on Earth.

With flexibility in mind, STL suggested three alternative data handling systems offering various combinations of real-time transmission, fast scanning of selected instruments, and data storage prior to transmission. Data storage allowed the instruments to record data faster than the communication subsystem could transmit it to Earth—a valuable feature during a solar flare, for example. As mentioned earlier, data storage capability also permitted data recording during periods when the spacecraft was not being tracked. The feasibility study, however, was based on a spacecraft without data storage for the sake of simplicity and reliability, although STL engineers clearly favored the addition of a data storage unit.

STL did examine specific types of experiments, although for the actual spacecraft NASA solicited experiments from the scientific community. The five types of instruments suggested by STL were:

(1) Magnetometers, both fluxgates and search coils
(2) Plasma probes
(3) Lyman-alpha detectors
(4) Micrometeoroid detectors
(5) Cosmic-ray detectors

Detailed instrument design was not part of the feasibility study. The study of instrument types was aimed solely at defining interface problems. The Pioneer spacecraft that actually flew carried all of the instrument types suggested by STL with the exception of the Lyman-alpha detectors. As a result of the deliberations of its Space Science Steering Committee, NASA also added radio-propagation and electric-field experiments to the Pioneers (see ch. 5).

IMPACT OF THE FEASIBILITY STUDY

The feasibility study was a solid foundation for the drawing up of specifications, the issuance of an RFP, and the eventual selection of a hardware contractor. The feasibility study did not provide all of the answers; some spacecraft features were changed later during the detailed design phase. Still, the basic concept was sketched out and strengthened by the application of STL's hardware experience with many other spacecraft in the same size class. The following chapters covering detailed system design will use the results of the feasibility study (ref. 1) as a point of departure in describing the technical evolution of the Pioneer interplanetary probe.

REFERENCE

1. ANON.: Final Report on the Interplanetary Probe Study. Space Technology Laboratories Rept., Redondo Beach, Aug. 15, 1962.

Pioneer Launch Trajectory and Solar Orbit Design

THE ORIGINAL PLAN for the Pioneer Program involved merely sending a small spacecraft into orbit about the Sun where it could monitor the environment in interplanetary space without the perturbations of the Earth's magnetosphere and atmosphere. Trajectory analysis soon showed that the scientific productivity of the missions could be increased greatly by shaping the trajectories and orbits to: (1) enhance solar system coverage, (2) create astronomical phenomena, such as solar occultations, and (3) study Earth-induced space phenomena, such as its magnetic tail. Trajectory and orbit planning thus became more complex as scientific objectives grew more ambitious.

Each Pioneer mission was different. Rather than burden the reader with the details of each, generalizations and summaries covering all Pioneer flights will be presented, supplemented by a detailed discussion of trajectory and orbit design for Pioneer 9.

SPECIFIC MISSION OBJECTIVES: A SCIENTIST'S VIEW

To set the stage for the general treatment of trajectory trade-offs and other factors that influenced Pioneer celestial mechanics, consider the following special requirements levied on the five missions. The special requirements for Pioneer 6 were:

(1) Inward trajectory, perihelion near 0.8 AU, in order to extend solar system coverage by Pioneer instruments into the sector ahead of the Earth as it plies its orbit about the Sun

(2) Solar occultation of the spacecraft as seen by the tracking antennas on Earth

The special requirements for Pioneer 7 were:

(1) Outward trajectory, aphelion near 1.1 AU, to extend solar system coverage in the Earth's "wake"—note that a lagging spacecraft actually detects solar events before terrestrial instruments because the outwardly spiraling solar magnetic lines of force sweep around the solar system faster than the planets due to the Sun's 28-day rotation.

(2) Geomagnetospheric tail analysis—an outward-bound Pioneer can be designed to swing through the Earth's magnetic tail.

(3) Lunar occultation analysis—on both inward- and outward-bound missions, scientists had a "sporting chance" to see an occultation of the Earth by the Moon through the sensors of the Pioneer instruments. Intrinsic launch vehicle inaccuracies precluded any guarantees. The first attempt was made with Pioneer 7.

Pioneer 8 special requirements were the same as for Pioneer 7; Pioneer 9 special requirements were the same as for Pioneer 6.

The special requirements for Pioneer E were:

(1) Inward-outward combination trajectory, with final near-Earth (1.0 AU) heliocentric orbit—the objective was to have the spacecraft linger in the vicinity of the Earth, allowing the use of high-bit-rate telemetry over a period of several hundred days.

(2) Geomagnetospheric tail analysis was to be similar to that of Pioneer 7.

OTHER FACTORS INVOLVED IN PIONEER TRAJECTORY AND ORBIT DESIGN

Before detailed trajectory studies could commence for any Pioneer mission, the science-oriented objectives had to be translated into quantitative goals which in turn were subject to quantitative constraints imposed by hardware and the laws of nature. Several new trajectory and orbit design factors are apparent in the following list of goals and constraints established for the Pioneer 9 mission, which is used here as an example:

(1) 0.76 AU nominal perihelion

(2) 0.00° inclination with respect to the ecliptic plane

(3) To maximize the time the spacecraft remains close to superior conjunction (solar occultation)

(4) Lunar occultation

(5) To provide station-look angles of less than 150° and greater than 30° at Deep Space Station 51 (DSS–51 at Johannesburg); DSS–41 (Woomera); and DSS–12 (Goldstone) from launch plus 90 minutes to launch plus 48 hours (This goal was established to facilitate tracking and data acquisition during the spacecraft orientation maneuvers.)

(6) To minimize the sensitivity of transit time to superior conjunction to deviations in launch vehicle performance

The following constraints or essential conditions were also established:

(1) Three-sigma probability (99.73 percent) of Earth escape based on *n*-body escape velocity

(2) Three-sigma probability of a Sun-look angle greater than 10° at spacecraft injection (This condition provides a high probability that the Sun will be seen by the spacecraft Sun sensors, which have a built-in 10° deadband where the Sun is invisible. Proper orientation of the spacecraft is impossible unless the Sun is in view of these sensors. See ch. 4.)

(3) Three-sigma probability of orbit inclination to the ecliptic plane less than 0.20°

(4) Total input heat rate to spacecraft less than 0.095 Btu/ft^2-sec due to rocket plume radiation, Earth albedo, etc.

(5) Maximum internal fairing temperature less than 275° F

(6) Spacecraft total angular momentum vector to point at the southern celestial hemisphere after the first orientation maneuver (Type-I maneuver)

(7) Three-sigma probability that Earth shadow period (time in umbra) is less than 15 minutes (The spacecraft battery is of limited capacity and must not be exhausted before sunlight activates the solar cells) (fig. 2–1).

(8) Three-sigma probability that the spacecraft will not impact the Moon

(9) Spacecraft spinup acceleration to be less than 25 radians/sec^2 to avoid undue stresses on the spacecraft

(10) Launch window to be greater than eight minutes

(11) Sixty seconds of tracking from Antigua to be available after second-stage engine cutoff (SECO)

(12) The orbit attained by the second stage prior to the injection of the Pioneer spacecraft into an escape hyperbola to be suitable for the piggyback Test and Training Satellite (TETR–2 on Pioneer 9)

Quite obviously there was a need for the trajectory designer to balance many parameters as he attempted to program the Delta launch vehicle for a Pioneer mission.

Early Phases of Trajectory

A general picture of the trajectory is useful amid these seemingly unrelated parameters. The Delta launch vehicles carrying Pioneer payloads were all launched southeastward from Cape Kennedy along the Eastern Test Range (ETR). During the flight, the Deltas passed over Ascension Island in the South Atlantic and tracking stations in the vicinity of Johannesburg, Republic of South Africa (fig. 2–2). Approximately 500 seconds after lift-off, the second-stage engines cut off (figs. 2–3 and 2–4). The Delta second and third stages, the Pioneer spacecraft, and any TETR piggyback spacecraft are then in Earth orbit over Johannesburg. This coast phase is essential if the spacecraft is to be launched properly into an orbital plane nearly parallel to that of the ecliptic. At a point before the spacecraft and attached Delta upper stages reach the plane of the ecliptic, the small rockets on the spin table on the Delta second-stage fire, imparting a spin to the spacecraft and Delta third stage. Next, the Delta third stage fires at that precalculated point in the coast trajectory where the velocity added by the third stage will carry the spacecraft into an escape hyperbola and thence into orbit around the Sun. Only after third-stage ignition is the second-priority TETR injected into Earth orbit from its berth near the top of the second stage. The inward Pioneers (6 and 9) were injected with

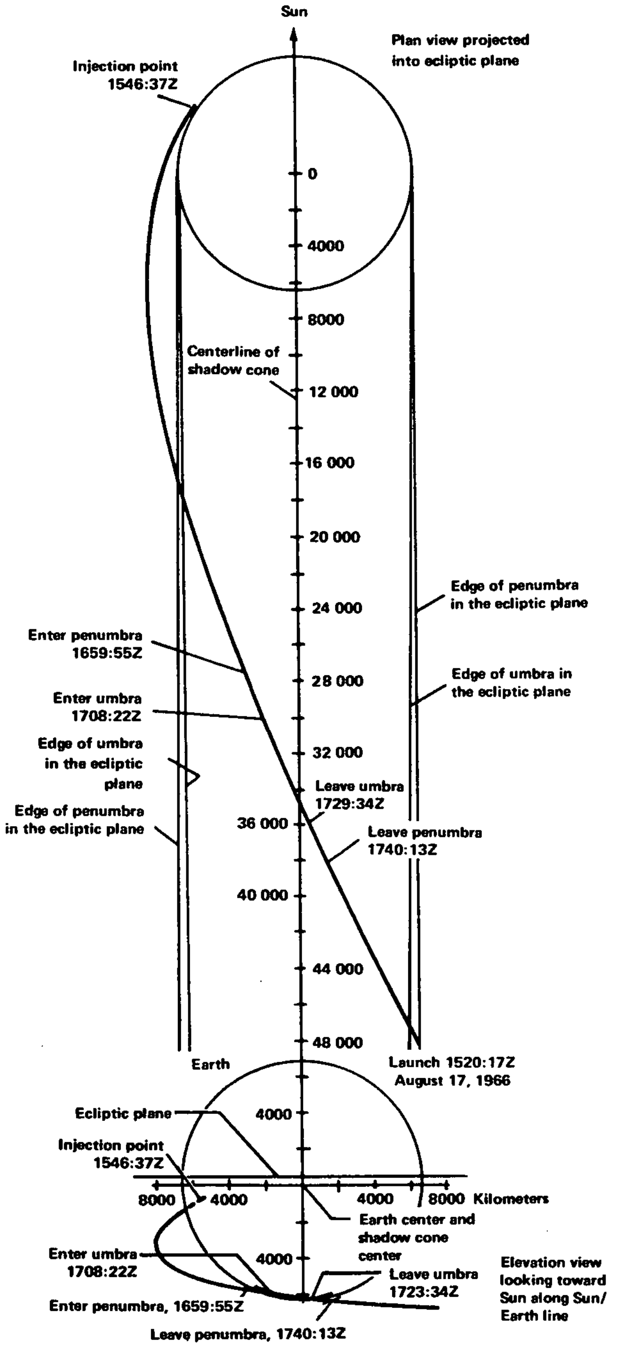

FIGURE 2–1.—The trajectory of Pioneer 7 as it passed through the Earth's shadow.

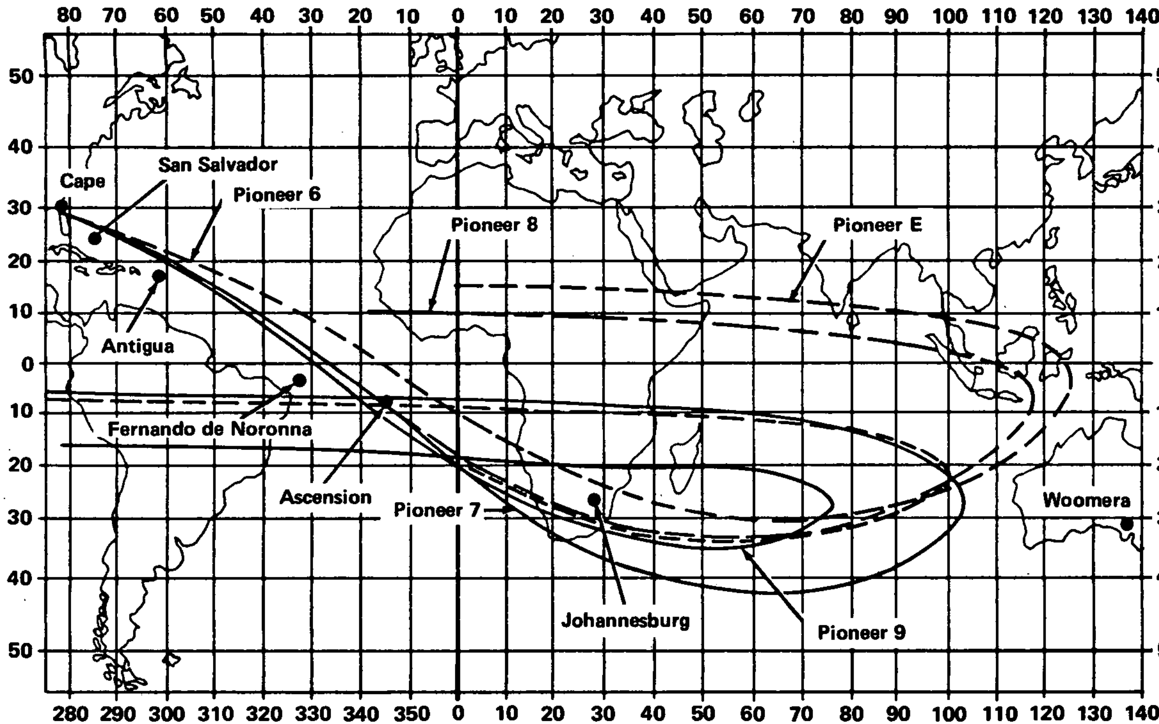

FIGURE 2-2.—Ground tracks for the four successful Pioneers plus the projected track of Pioneer E.

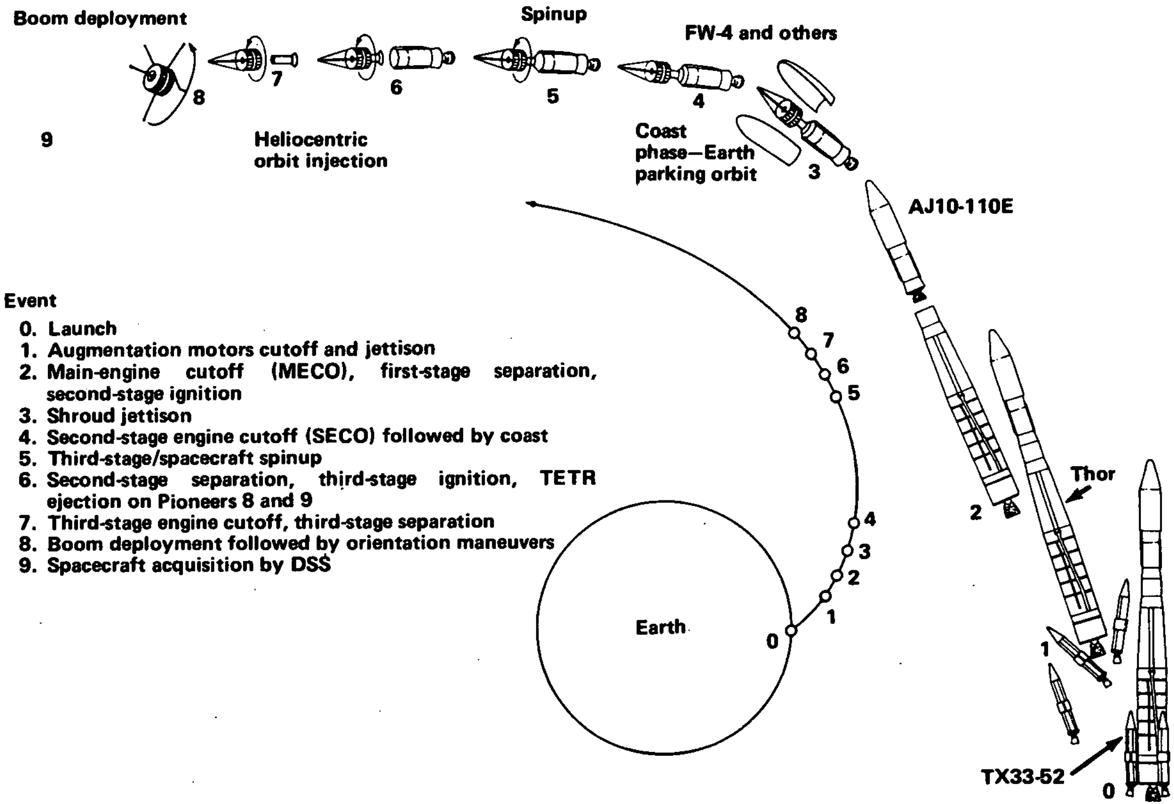

FIGURE 2-3.—The Pioneer launch sequence.

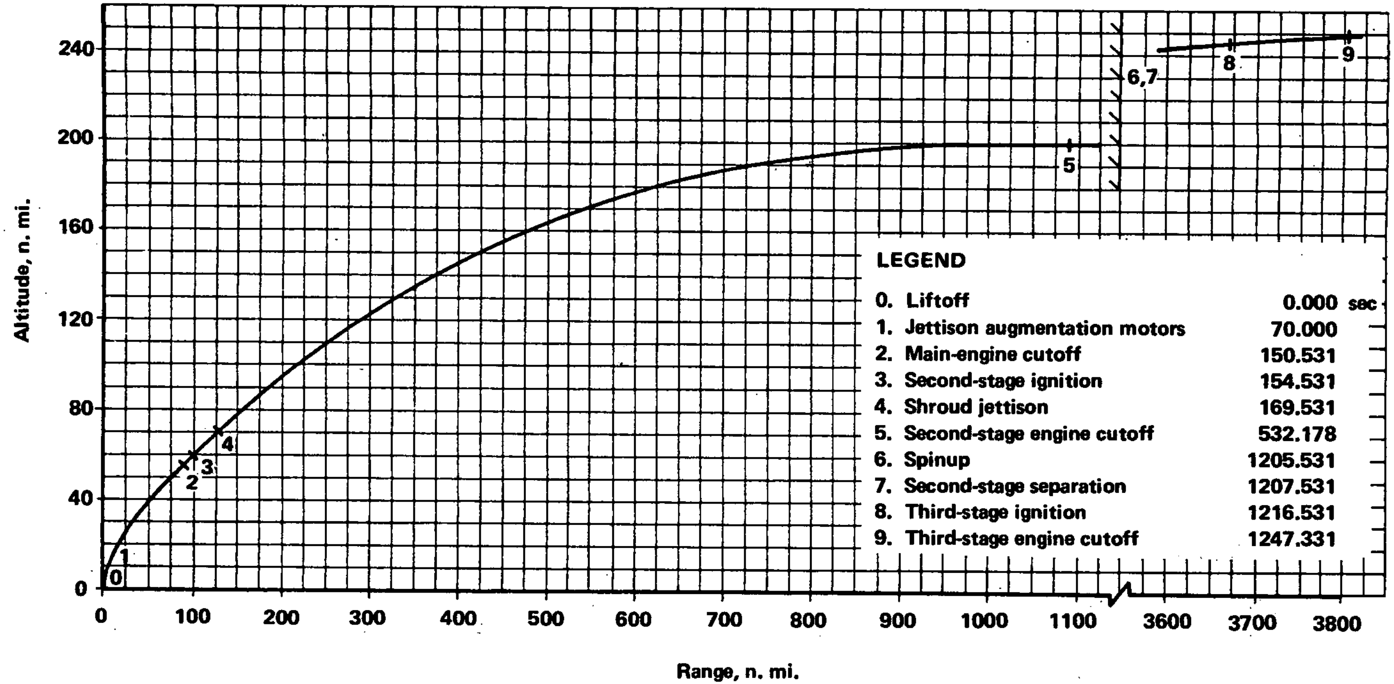

FIGURE 2–4.—The planned ascent profile for Pioneer 9.

velocity vectors approximately opposite to the Earth's velocity. Thus slowed, they "fall" in toward the Sun and initially fall behind or lag the Earth (fig. 2–5). The inward Pioneers essentially convert gravitational energy into orbital velocity and, after about 75 days, catch up with the Earth and lead it by ever-increasing distances in its journey around the Sun. The outward Pioneers (7 and 8) were injected with velocities parallel to that of the Earth. They initially lead the Earth but after 30 to 40 days they fall behind and, like the outer planets, lag the Earth.

The ground tracks of the Pioneers (or any other interplanetary probes) indicate a retrograde motion with respect to an observer on the Earth. This effect is due to the rotation of the Earth under the spacecraft as it moves off into deep space. The ground track is, of course, of vital importance in scheduling NASA's tracking and data acquisition stations around the world (ch. 8).

Launch Windows

As the Earth turns on its axis, it carries Cape Kennedy to a position within approximately 5° of the plane of the ecliptic once a day. This is the optimum period for Pioneer launches. At this moment, only about 5 min of coast time are required to reach the plane of the ecliptic. Twelve hours later, a 30-min coast period is necessary; this would cost extra payload pounds. Pioneer launches, therefore, were best made during launch windows a few minutes wide that occur only once a day. The Pioneer Project Office

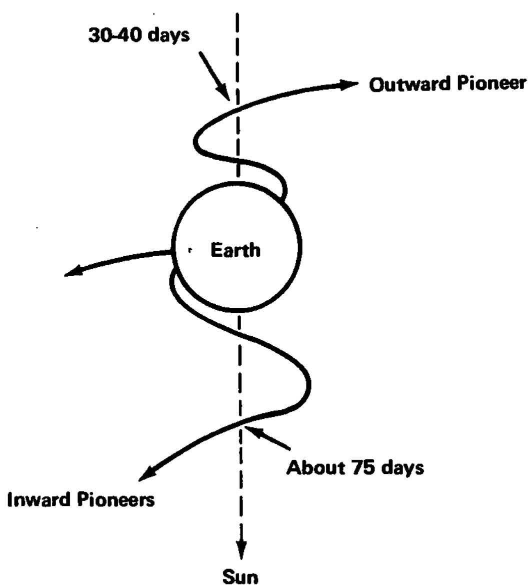

FIGURE 2–5.—Initial portions of inward and outward Pioneer trajectories (not to scale).

at Ames Research Center required that launch windows be greater than 8 min wide so that short holds would not scrub a mission for a whole day.

From the standpoint of scheduling the many intermeshing launch activities at Cape Kennedy, Pioneer launches had to be arranged months ahead of time. Precise times had to be specified so that tracking crews, range safety personnel, and all other Cape activities could be properly synchronized. Neither was a specific time window on a single specific day sufficient, because the launch might be postponed due to weather or some minor malfunction. In the Pioneer Program, "blocks" of launch windows, with windows about a day apart within each block, were established. If minor problems did crop up, it was hoped that they would be corrected in time for launch within a given block of days. If, however, serious difficulties arose affecting the launch—weather or a conflict with a higher priority program, for example—the first block of windows could be set aside and replaced by the second block. Two blocks of windows set aside for Pioneer 9 are described in table 2–1.

PIONEER 9 TRAJECTORY ANALYSIS

The ascent trajectory for the Delta launching Pioneer 9 was designed by McDonnell-Douglas Astronautics Co., the launch vehicle contractor, subject to the objectives and constraints specified by the Pioneer Project Office of Ames Research Center (ref. 1). This was the usual procedure for all Pioneer launches. The objectives and constraints listed in the preceding section are essentially those applied to Pioneer 9.[3] The two blocks of desired launch days were also specified by Ames in accordance with mission requirements. All data presented in this section were computed for the nominal launch time of November 6, 1968, 9:47 GMT unless otherwise specified. They are applicable even though the launch was delayed until November 8.

A characteristic of all Pioneer missions is the common ascent trajectory for all days within a given launch block. Timing of the launch was, of course, dictated by the approach of the plane of the ecliptic to Cape Kennedy. The ascent profile for the Pioneer 9 launch is presented in figure 2–4.[4] Further details about the launch vehicle and its operational constraints may be found in chapter 7. For this specific launch the planned liftoff weight was 151 761 lb with a liftoff thrust of 255 367 lb. In the launch plan, the main first-stage engine burned for 150.5 sec and the three solid augmentation rockets burned for approximately 40 sec each. During the first-stage burn, pitch and yaw control was accomplished by the automatic

[3] According to NASA terminology, this spacecraft was called Pioneer D until it was successfully injected into Earth orbit.

[4] Launch trajectories were computed from a three-dimensional n-body computer program developed by JPL and designated DBH07.

TABLE 2–1.—*Pioneer-D Launch Windows*

Block-IIA missions		
Date	Window opens (GMT)	Window closes (GMT)
November 6, 1968	0945	0959
November 7, 1968	0941	0955
November 8, 1968[a]	0937	0951
November 13, 1968	0917	0932
November 20, 1968	0849	0904
November 22, 1968	0841	0856
Block-III missions		
Date	Window opens (GMT)	Window closes (GMT)
November 27, 1968	0843	0857
December 4, 1968	0816	0829
December 11, 1968	0748	0802
December 18, 1968	0721	0734
December 22, 1968	0703	0717

[a] Actual launch date; time 0946:29 GMT.

gimballing of the main engine in response to signals from an inertial reference package. Roll control was maintained by gimballed vernier engines and the inertial reference package. A radio guidance system in the second stage also supplied steering correction signals to the first stage.

The second-stage engine was ignited at an altitude of about 60 n. mi. This motor burned for a nominal 377.6 sec with a thrust of 7803 lb. Again, the main engine was gimballed. Roll was controlled by four cold-gas jets. Control signals originated in a second-stage programmer, an inertial reference package, and the radio guidance system.

The coast period following second-stage cutoff was computed to be 684.4 sec for Pioneer 9. As the spacecraft and the attached second and third stages approached the point of injection, gas jets turned the spacecraft axis so that it had an elevation angle of $-2.0°$ and a yaw angle of $5.2°$ about the local vertical to the right of the trajectory plane looking downrange. Next, the spin-table rockets atop the second stage were fired to spin up the spacecraft and attached third stage for purposes of dynamic stability. Just 9 sec before reaching the point of injection, the second stage was jettisoned. The combination third-stage-plus-spacecraft weighed 878.2 lb at this point.

The solid third-stage engine fired for 30.8 sec with a thrust of approximately 5605.5 lb, imparting 3282 m/sec velocity to the spacecraft before cutoff. The resultant velocity vector was approximately parallel to the plane of the ecliptic. Attitude stability during third-stage burn was maintained by the spinning action. The third stage was jettisoned and the spacecraft headed on an escape hyperbola for heliocentric orbit. The ground trace for Pioneer 9 is shown in figure 2–2.

If all had gone well during the launch, the spacecraft would have embarked on the planned near-Earth trajectory illustrated in figure 2–6. As it turned out, a launch vehicle problem delayed the launch for about 48 hours. If the date in figure 2–6 is changed to Nov. 8, the figure applies equally well for the actual trajectory. The view in figure 2–6 is that seen from the north ecliptic pole with the Earth-Sun line fixed in space. As mentioned earlier, the inward-bound Pioneer 9 spacecraft initially lagged the Earth which was moving to the left.

One can see from figure 2–7 how lunar occultation by the Earth—as seen from spacecraft instruments—is possible. This astronomical event, however, is very sensitive to small dispersions in launch vehicle performance. A slight deviation from the nominal orbital plane, for example, will preclude occultation.

The actual trajectory of Pioneer 9 is shown in figure 2–8, on the same scale as figure 2–7. The critical difference is not the shape of the trajectory, which is almost identical, but the day of launch. During the two days' delay, the Moon had moved as shown. Further, a 9-min hold prior to launch resulted in excursions of 206 000 and 202 000 km below and above the plane of the ecliptic, respectively.

Figure 2–8 also has the Earth's magnetosphere (hardly spherical) superimposed upon it. The trajectory cuts through the "side" of the plasma sheath, but inward launches do not take the spacecraft far out into the magnetic tail like the outward launches do.

The event termed syzygy is noted on figure 2–9. This is simply that point in time when the Earth is between the Sun and the spacecraft and in a common plane perpendicular to the ecliptic. Unlike solar occultation, which occurs when the Sun is between the Earth and the spacecraft, syzygy holds little interest for the scientists.

Three other types of charts are commonly used in describing Pioneer heliocentric orbits. The first, figure 2–10, is based on a Sun-centered vernal equinox ecliptic reference. It shows clearly how Pioneer 9 draws farther and farther ahead of the Earth as both swing around the Sun. Perihelion for Pioneer 9 occurs roughly at the same spot in space where it was launched, but when it first returns to this location in 298 days, the Earth with its 365-day period will be far behind the spacecraft.

The second type of presentation shows Pioneer orbits plotted with respect to a fixed Earth-Sun line, figure 2–11. This figure is much like figures 2–6, 2–7, and 2–8 except that here the Sun is at the center of the polar coordinate paper. The distance between the Earth and Pioneer 9

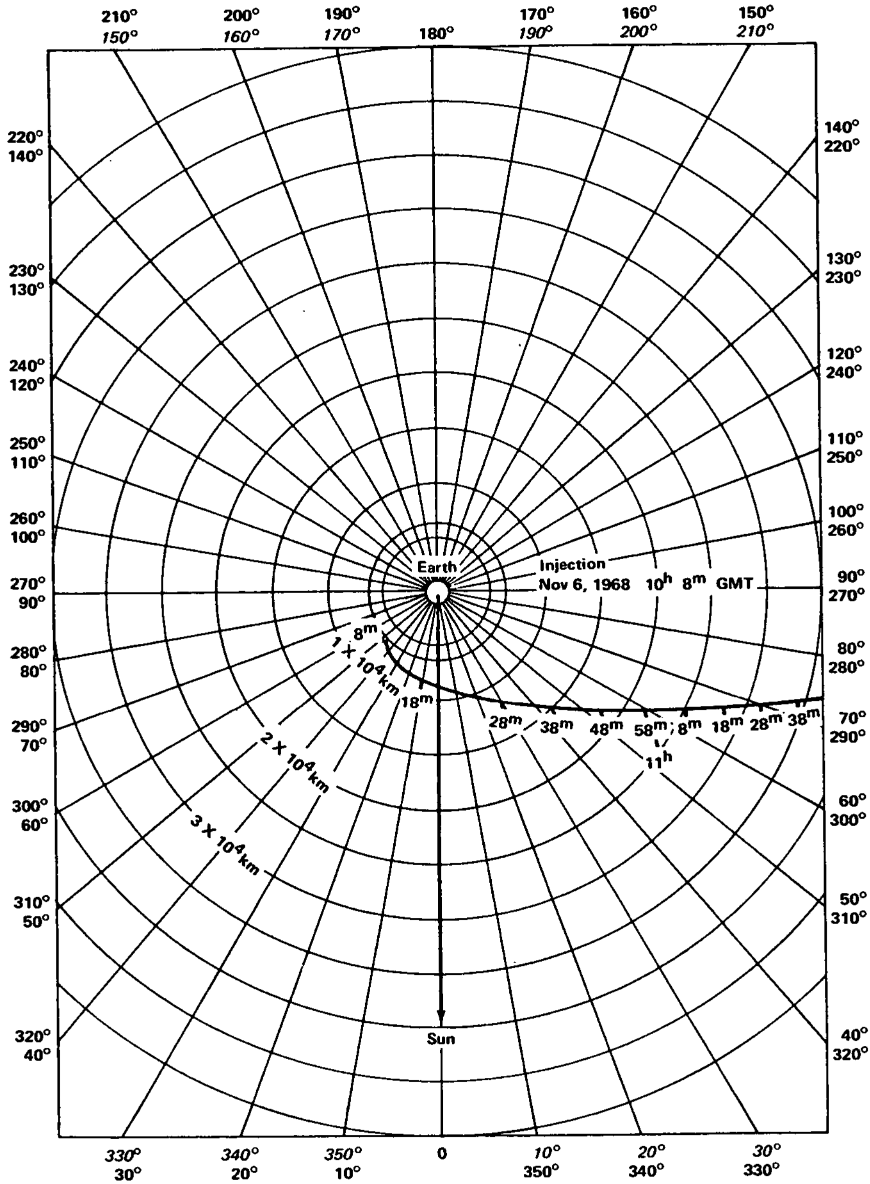

FIGURE 2–6.—The near-Earth trajectory of Pioneer 9 (as projected for a Nov. 6 launch).

grows ever greater, with the exception of the little loops of apparent retrograde motion at the aphelion points. Early in 1973, Pioneer 9 will lap the Earth for the first time. In the case of outward Pioneers, the Earth laps them, although not so quickly. The plots for outward-bound Pioneers show

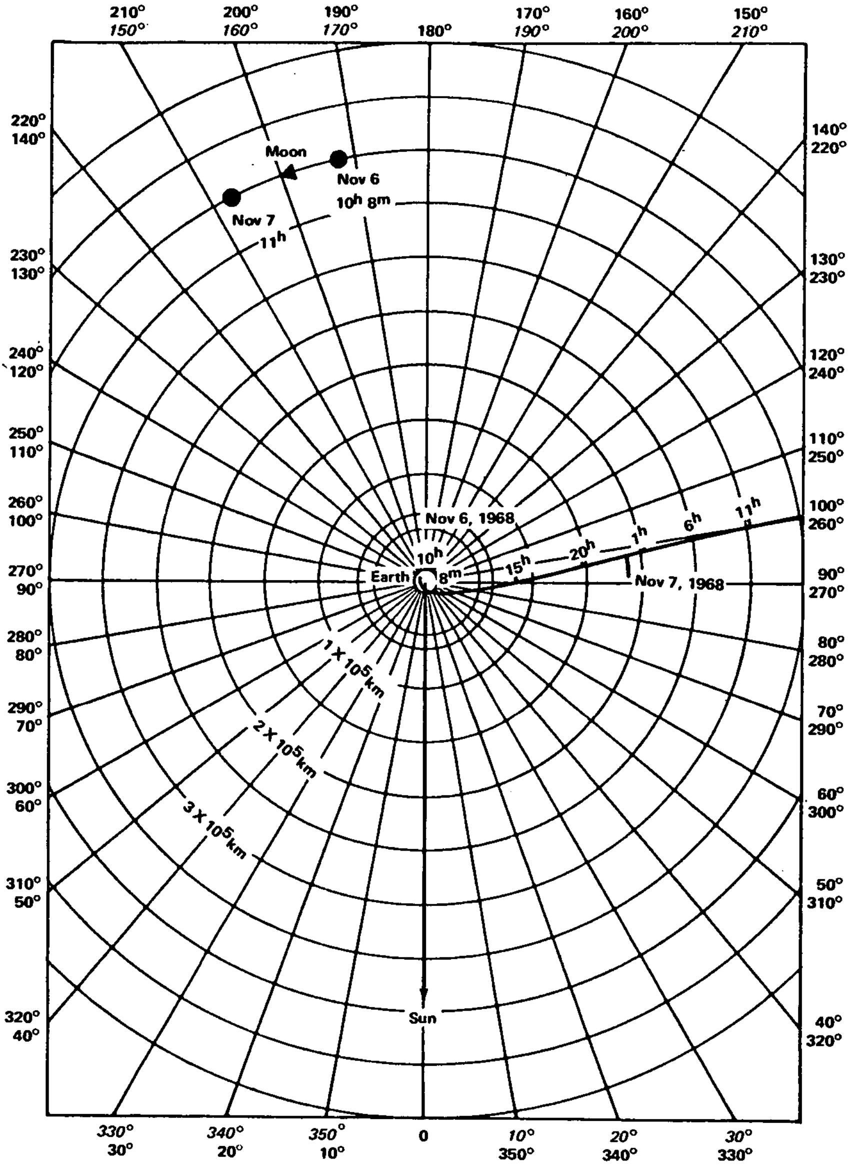

FIGURE 2-7.—Cislunar trajectory for Pioneer 9 (as projected for a Nov. 6 launch).

cusps at perihelion rather than the aphelion loops on figure 2-11. The scientifically important event shown on figure 2-11 occurred when the Sun occulted the Pioneer 9 spacecraft in late 1970. For a month or two on either

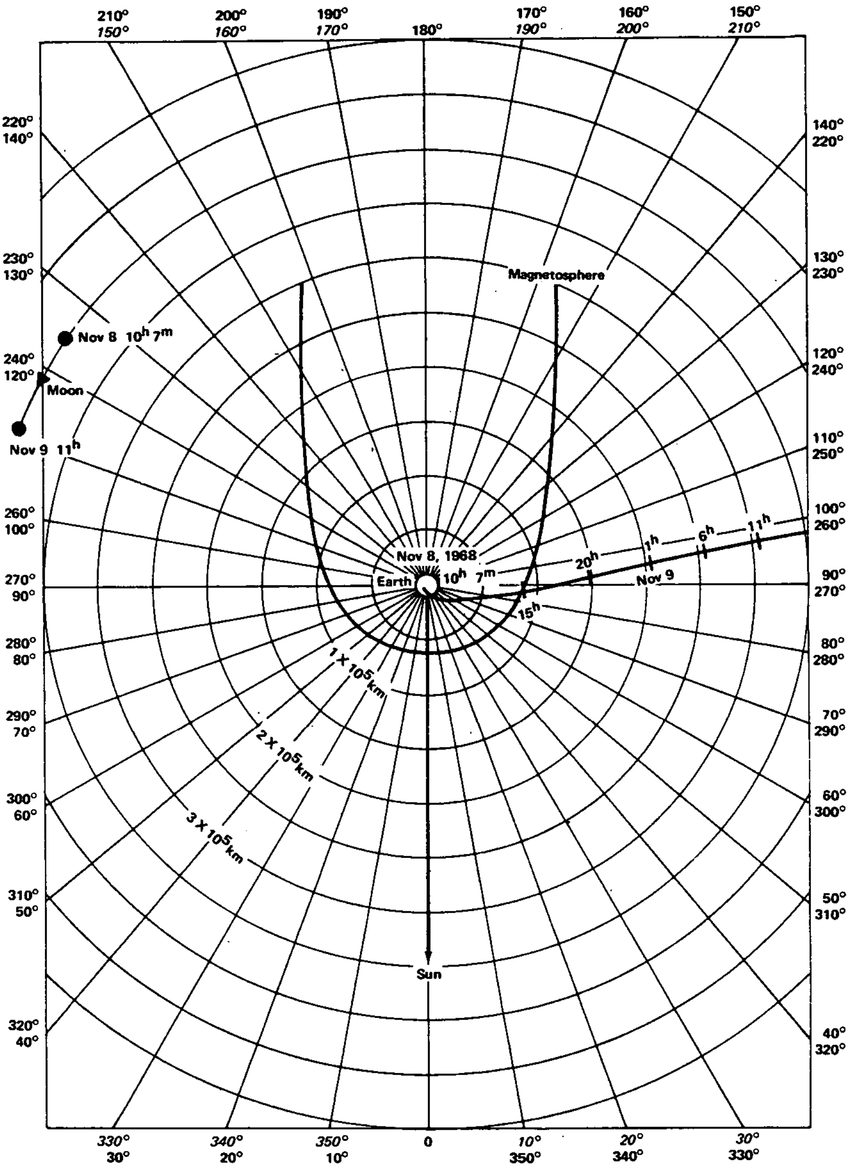

FIGURE 2-8.—Actual cislunar trajectory for delayed Pioneer 9 launch.

side of this date, the DSN 210-ft antenna at Goldstone recorded the effects of the solar corona and atmosphere on the spacecraft radio signals.

The last type of plot considered here shows the relative positions of all four successful Pioneers with respect to one another and the Earth at

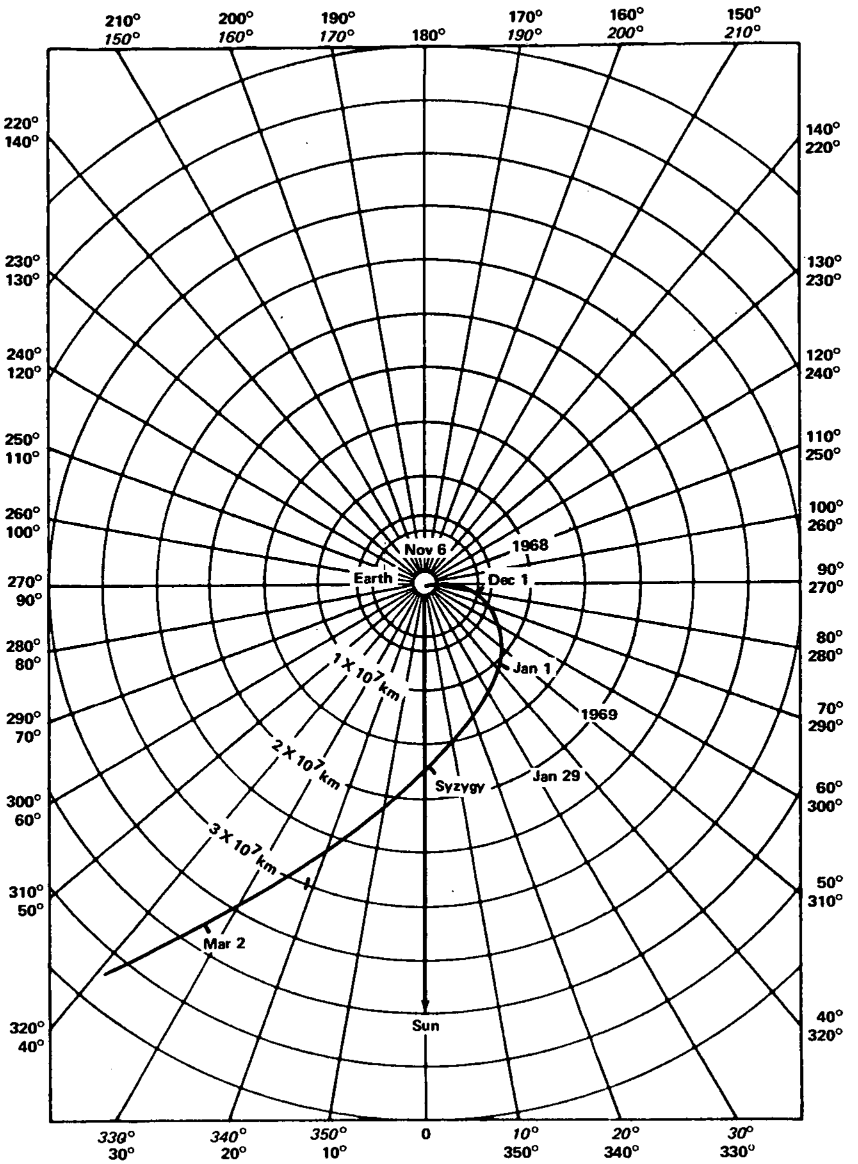

FIGURE 2–9.—Pioneer 9 trajectory through syzygy (as projected for a Nov. 6 launch).

various times. In a sense, figure 2–12 consists of a series of snapshots looking down on the plane of the ecliptic from the north ecliptic pole. The spacecraft and Earth are moving counterclockwise about the Sun, with the inward objects moving faster than the outward objects, as both real and

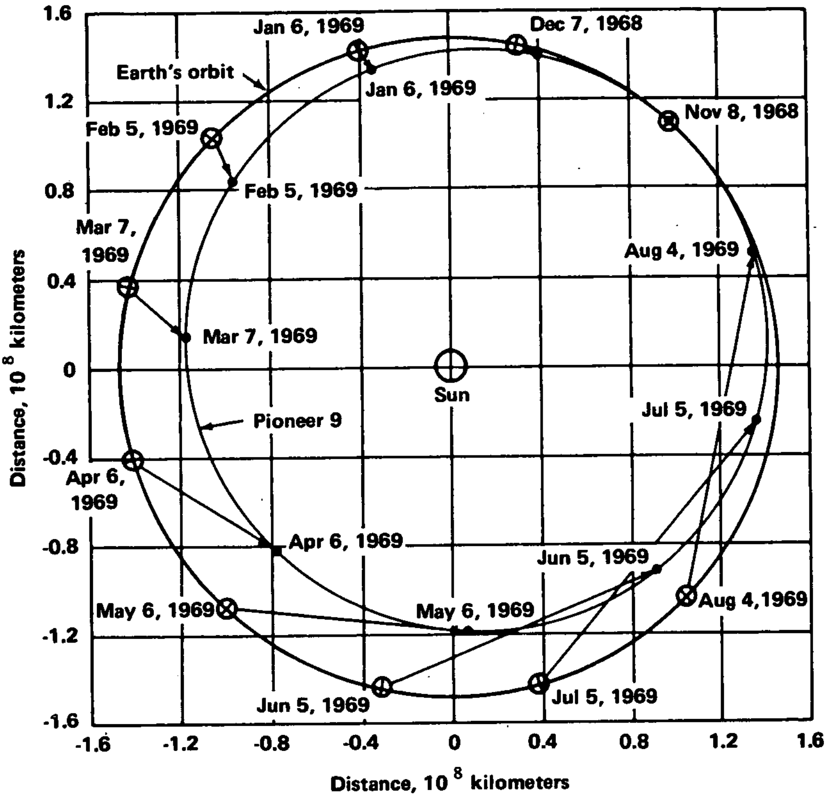

FIGURE 2–10.—Actual heliocentric orbit for Pioneer 9 using vernal equinox reference.

artificial planets should. This view has physical meaning for those attempting to forecast solar weather. The Sun rotates on its axis in the same direction as the Earth and the Pioneers circulate around it. With a 28-day period of rotation, however, the Sun's spiral magnetic field, which rotates with the Sun, turns much faster than the objects in heliocentric orbit. Therefore, the streams of plasma that follow the Sun's magnetic lines of force are always catching up with both the Earth and the probes and spraying them with plasma like a rotary water sprinkler. The lagging Pioneers are thus in a position to forecast solar-related events for the Earth.

PIONEER ORBITAL PARAMETERS

The charts presented above are helpful in visualizing the Pioneer trajectories and orbits. For those interested in more precise information, table 2–2 summarizes Pioneer orbital data as of March 1969.

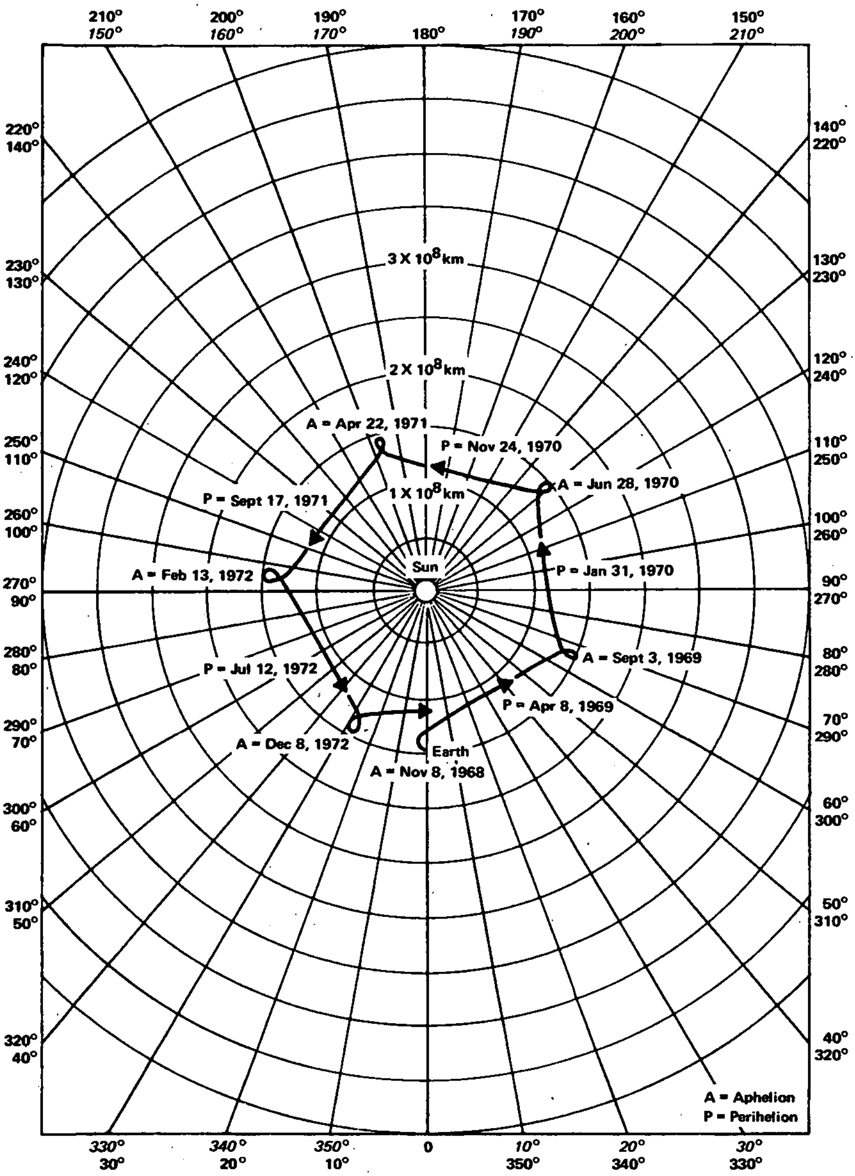

FIGURE 2-11.—Actual heliocentric orbit of Pioneer 9 in Earth-Sun-line reference frame.

SPACECRAFT ORIENTATION

For the Pioneer spacecraft concept to be successful, the spacecraft spin axis had to be oriented so that it was perpendicular to the plane of the ecliptic. Only then would the spacecraft antenna patterns intercept the

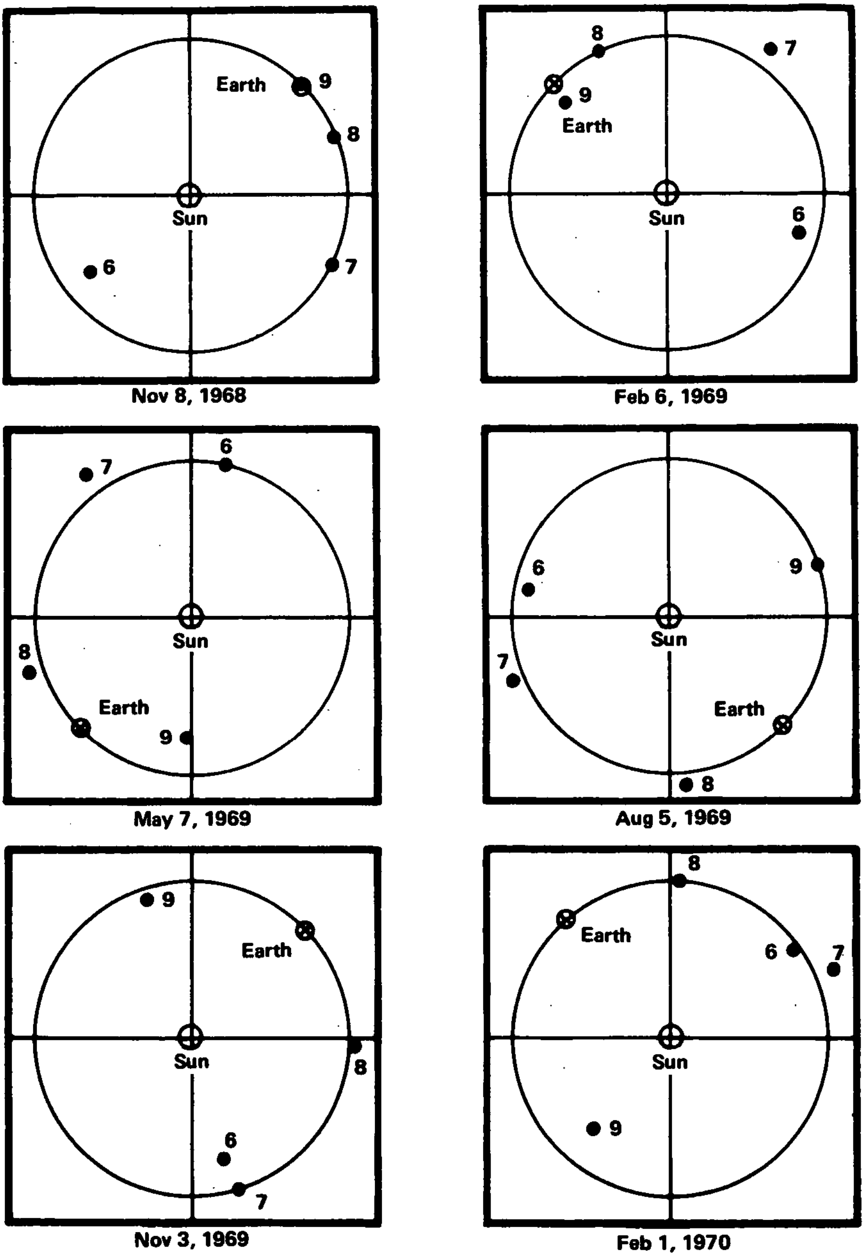

FIGURE 2-12—Relative positions of the four successful Pioneers with respect to the Earth at various times.

TABLE 2-2.—*Pioneer Orbital Parameters*[a]

Orbital injection conditions				
Parameter	6	7	8	9
Date of injection	12–16–65	8–17–66	12–13–67	11–8–68
Time of injection (GMT)	0756:41.1	1545:38.6	1439:32.5	1007:22.4
Injection latitude	7.8° S	14.48° S	22.83° S	3.36° S
Injection longitude	4.6° W	6.8° W	9.385° E	23.26° W
Injection altitude (km)	564.1	378.476	486.02	467.054
Injection velocity (km/sec)	10.8488	10.939	10.7837	11.035674
Flight path angle (deg)	1.7	2.1	−0.364	2.413724
Azimuth angle (deg)	119.3	106.98	129.374	101.04027
Elements of parking orbit				
Semimajor axis (km)	7149.44	7015	6775.1	7049.3
Eccentricity	0.071	0.0549	0.0139	0.0424
Inclination (deg)	30.2	33.0	32.906	32.88
Height of perigee (km)	270.6	330.9	307.6	372.2
Height of apogee (km)	1288.1	1342.5	495.1	970.0
Anomalistic period (min)	100.5	97.4	92.4	98.17
Elements of heliocentric orbit				
Semimajor axis (km)	134 481 910	159 713 300	155 372 610	130 500 710
Eccentricity	0.0942	0.05397	0.0476	0.1354
Inclination to ecliptic plane (deg)	0.1693	0.09767	0.0578	0.0865
Aphelion (AU)	0.936	1.1250	1.0880	0.9905
Perihelion (AU)	0.8143	1.0100	0.9892	0.7542
Period (days)	311.327	402.91	386.60	297.594

[a] Support Instrumentation Requirements Document, Project Pioneer, NASA Headquarters, March 1969.

Earth and only then would the solar-cell arrays generate full power. After injection by the Delta third stage, the typical Pioneer spacecraft was spin-stabilized but its axis was not perpendicular to the ecliptic plane. The spacecraft onboard orientation subsystem automatically began the first or Type-I orientation maneuver. During this maneuver cold-gas jets torqued the spin axis around until it was perpendicular to the Sun-spacecraft line. The second, or Type-II, orientation maneuver was carried out through ground commands from a DSN station, usually Goldstone. Under ground control, the gas jets were fired until the spacecraft transmitter signal received on Earth was maximized. The spacecraft spin axis was then approximately perpendicular to the plane of the ecliptic. A detailed description of these maneuvers and the spacecraft orientation subsystem may be found in chapter 4.

REFERENCE

1. ANON.: Pioneer D(IX) Trajectory and Orientation Analysis. TRW Systems Rept. 3432.8-5, 1968.

Spacecraft Design Approach and Evolution

SPACECRAFT DESIGN APPROACH

SPACECRAFT DESIGN is rarely a series of orderly, obviously logical steps. Instead, spacecraft design is usually iterative; that is, cyclic, with each successive iteration based on the experience of the previous one. The Pioneer spacecraft was no exception. It passed through a conceptual design phase, a feasibility study phase, an iteration on the feasibility study, a final design phase, and, before Pioneer 6 was finally mated to a Delta at Cape Kennedy, it had evolved not into a new species, but rather into a hardier, longer-lived subspecies. Furthermore, spacecraft evolution did not stop with the first launch; each of the five spacecraft and their instrument complements were slightly different, with the largest change occurring between Blocks I and II.

The Pioneer spacecraft evolved because of continuous pressure for improved reliability, telemetry capability, instrument payload, and other measures of performance. The direction of the spacecraft's evolutionary path was determined by the major spacecraft design objectives shown in table 3–1. The width of the path was established by the design constraints, some of which are noted in table 3–2. The achievement of the design objectives within the confines of the design constraints required a design philosophy, based on experience with other spacecraft of the same general type. Table 3–3 presents some of the major elements of Pioneer design philosophy. Quite obviously, design philosophy is really an astute combination of common sense and hard-earned experience.

It is not a foregone conclusion that spacecraft design objectives and constraints are compatible, even with the application of the best design philosophy. This fact is usually discovered during the feasibility study. Fortunately, the Pioneer spacecraft was feasible, and the long lifetimes achieved in space (several times the lifetime objective listed in table 3–1) demonstrate the success of the design philosophy.

Spacecraft Weight

Spacecraft weight is usually an extremely critical parameter during the design history of any spacecraft. Along with reliability and magnetic cleanliness, weight was one of the three spacecraft-wide parameters that

TABLE 3–1.—*Major Spacecraft Design Objectives*

Objective	Remarks
A success probability of 0.75 for a lifetime of six months	Lifetime figure was originally set by expected communication range of DSN 85-ft antennas (about 50 000 000 miles). (From Specification A–6669)
A magnetically clean spacecraft	At 80 in. from the spin axis, on the magnetometer boom, the field perpendicular to the boom axis should not exceed: a. 0.5γ peak at 0–25 Hz b. 16γ due to remanence after magnetization in a 25-G field parallel to the spacecraft axes c. 1.0γ due to remanence after demagnetization in an ac field having an initial magnitude of 50 Oe (A–6669)
Minimum cost	The original cost goal for the spacecraft alone was to be about $20 000 000.
Minimum weight	The upper limit of 111.24 lb was a constraint listed in Specification A–6669. (See table 3–2.)
Wide flexibility in scientific instrument accommodation	This would increase experimental options as more was learned about interplanetary space.
Maximum bit rate	The product of lifetime and bit rate is really the "pay-off function" for an interplanetary probe. Specific NASA objectives: 5 200 000 miles; 512 bits/sec 7 300 000 miles; 256 bits/sec 14 700 000 miles; 64 bits/sec 29 400 000 miles; 16 bits/sec 41 500 000 miles; 8 bits/sec

had to be controlled with great care. These three important factors will now be covered to set the stage for the discussion of specific hardware in chapter 4.

The initial weight estimate is almost always optimistic, with the weight rising alarmingly—10 to 20 percent over the desired value—during the first few months of design. Then a concerted weight-reduction program usually pares off a few pounds until spacecraft weight is once again compatible with the launch vehicle capability and the mission requirements (fig. 3–1). In the case of the Pioneers, weight was very critical for the first flight. Indeed, with the early Delta launch vehicles and the DSN of 1964, the total Pioneer concept was really only marginally feasible; that is, with a 20 percent increase in spacecraft weight or, equivalently, a traveling wave tube (TWT) efficiency of 30 percent instead of 50 percent, the design would not have succeeded. The Delta and DSN, however, were not static systems.

TABLE 3–2.—*Major Spacecraft Design Constraints*

Constraint	Remarks
Delta launch vehicle	The spacecraft/launch vehicle interface is discussed in chapter 7.
Deep Space Network (DSN)	The spacecraft/DSN interface is discussed in chapter 8.
State-of-the-art	Exceptions: TWTs, convolutional coder, and long-distance telemetry represented advances in the state-of-the-art.
Maximum spacecraft weight: 111.24 lb	This is exclusive of instruments, but includes payload penalties from launch-vehicle shroud and third-stage motor case thermal insulation (A–6669). See chapter 7.
Space environment between 0.8 and 1.2 AU	Actually, little was known about this environment in 1963 and 1964. Environmental data were extrapolated from near-Earth measurements.
Ratio of spin-axis moment of inertia to moments of inertia about other axis be greater than one	This would insure dynamic stability of spin-stabilized spacecraft.

The Delta was improved with each launch. Although Pioneer spacecraft weight did generally increase slightly from flight to flight (especially between Blocks I and II) as new experiments and improved equipment were added, the final two flights were launched with up to 30 lb of ballast and a piggyback TETR satellite on the Delta.

Spacecraft Reliability

The Pioneer spacecraft have operated several years beyond their nominal 6-month lifetimes. This extra capability or overdesign has proven of great value to science, extending Pioneer coverage of interplanetary space past the 1969–1970 solar maximum. The original objective of 6-month life was set as the time it would take the spacecraft to forge beyond the 50 000 000-mile limitations of the extant 85-ft DSN antennas. The DSN improved, as explained earlier, and, fortunately, the spacecraft met the challenge, enabling scientific data to be received from spacecraft on the far side of the Sun, a distance of about 200 000 000 miles.

Several proven methods for increasing reliability are listed in table 3–3. Reliability is a somewhat more elusive parameter than weight. Weight may be measured precisely, and budgeted, as program dollars are. Re-

TABLE 3–3.— *Major Elements of Spacecraft Design Philosophy*

Element of philosophy	Remarks
Failure modes of operation will be provided	The single failure of any control system, except the pneumatic assemblies of the orientation subsystem, would not result in the catastrophic failure of the mission (A–6669). This is the classical use of redundancy to attain higher reliability. (See further discussion in text.)
Subsystems will be of modular form and interchangeable	This expedited testing, replacement, and repair (A–6669).
Proven components will be used	Many entries on the Pioneer "approved parts" list came from the Air Force Minuteman ICBM program. (See "State-of-the-art," table 3–2.)
Parts will be rigorously qualified; components will be "burned-in" before use on spacecraft	The rather high percentage of early or incipient failures were eliminated in this way.
"Magnetic guidelines" will be rigorous	Some guidelines established are: (1) use of Pioneer approved parts list, which eliminated the most offensive materials and components; (2) all magnetic leads to be less than 0.25 in. long; (3) all chokes, inductances, and transformers to be carefully designed and screened; (4) alloy 180 to be used for welded wire work and interconnections; (5) extreme caution to be employed with electrodeless nickel plating; (6) leads carrying 10 mA or more to be twisted with the return lead; (7) solar array to be backwired; and (8) ground loops to be avoided.

liability, being a statistical parameter, cannot be measured for any single, specific part with any single, specific measuring instrument. Instead, the spacecraft engineer must rely upon collective experience with parts in applications similar to the intended one. In short, reliability assessment is as much an art as an engineering discipline. The attainment of high reliability in a spacecraft such as Pioneer requires the application of procedures and methods that have proven successful in the past. The two most successful general methods involve the use of carefully selected parts and the judicious application of redundancy. Added to the formal technique was the dedication of the program people to the goal of high reliability.

Mathematical assessments of Pioneer reliability were made by STL

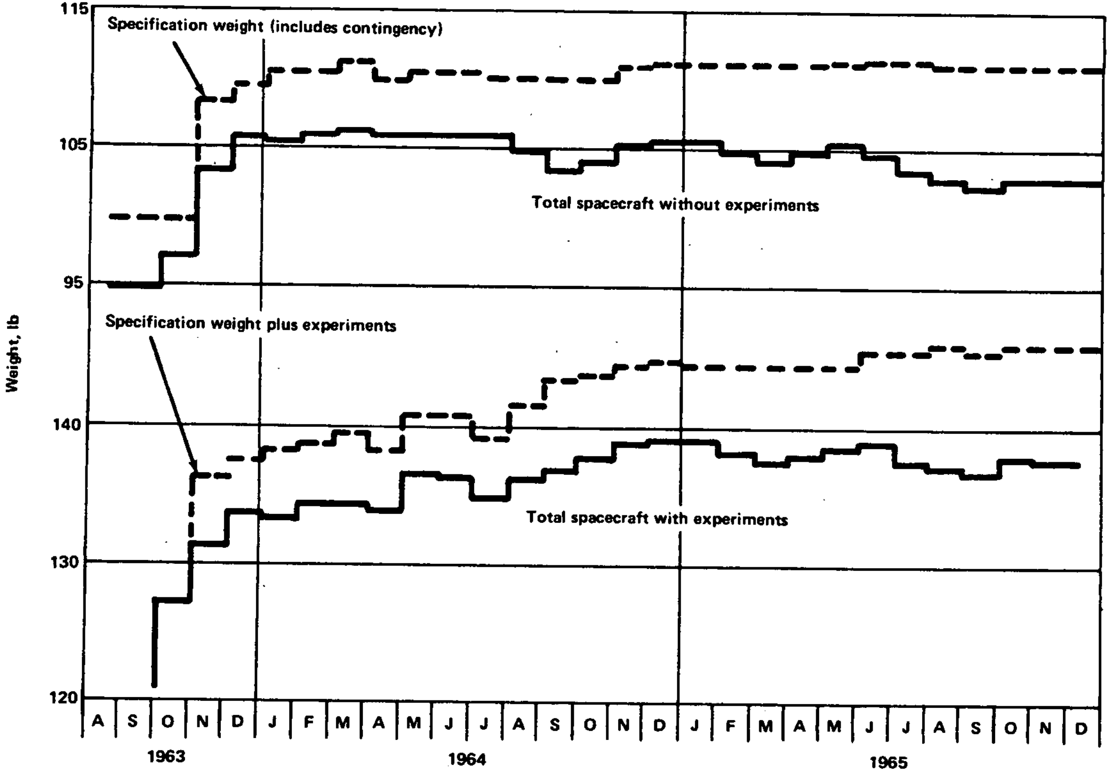

FIGURE 3–1.—Total spacecraft weight (Block I) without contingency as a function of calendar time.

during its feasibility study (ref. 1), and by Walter V. Sterling Co. during the performance of the following hardware design and development contract. Perhaps even more important to Pioneer's long life were the rigorous Quality Assurance and Failure Reporting programs established at STL and the equally severe test program that followed.

The high reliability of Pioneer is attributable to a three-level approach plus those elements of design philosophy listed in table 3–3. The three level approach was:

(1) Mathematical analysis to identify weak points and the value of redundancy (described briefly below)

(2) Selection and qualification of parts and subassemblies using very high standards (See Appendix for details on the STL Reliability and Quality Assurance Programs and Test Failure Reporting approach.[5])

(3) Testing in environments simulating those to be encountered by the spacecraft (covered in ch. 6)

During its 1962 feasibility study, STL estimated that using parts meeting military specifications but without the use of redundancy (except in the solar-cell array), overall system reliability would be an untenable 0.31. The use of the high-reliability parts developed during the Minuteman Intercontinental Ballistic Missile (ICBM) Program would boost system reliability to 0.59. Finally, the application of failure-mode protection (i.e., redundancy) would raise system reliability to an acceptable 0.86. The specific reliability model employed early in the program is portrayed in figure 3–2, while the effects of adding failure-mode protection are listed in table 3–4. It should be emphasized that figure 3–2 and table 3–4 are from the STL Pioneer proposal (ref. 3) and represent an early point in spacecraft design and not the spacecraft launched between 1965 and 1969, although most design features did not change significantly.

Magnetic Cleanliness Campaign

In the Pioneer Program a third spacecraft-wide factor was added to those of weight and reliability control: magnetic cleanliness. Since all spacecraft subsystems might use magnetic materials and might also generate interfering electromagnetic fields, the cleaning of a magnetically "dirty" spacecraft demanded the cooperation of all spacecraft engineers. Magnetic cleanliness, like high reliability, owes more to design experience than any single device or technique.

Magnetic cleanliness becomes essential on scientific deep-space probes that venture out into the weak interplanetary fields which generally measure less than 10γ. An ordinary capacitor, for example, may generate

[5] NASA Pioneer Specification A-6669.01 required the spacecraft contractor to establish these programs in accordance with NASA-wide product assurance guidelines. See A-6669 for details (ref. 2).

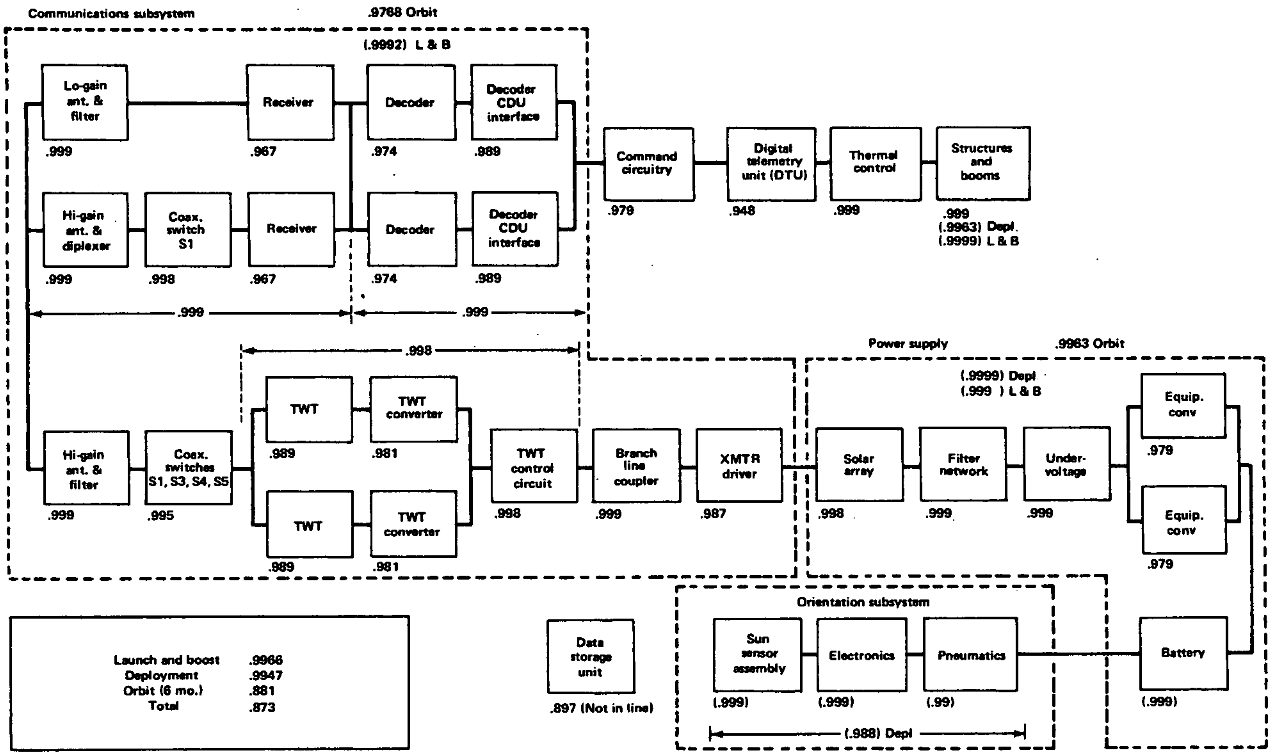

FIGURE 3–2.—Reproduction of the reliability diagram submitted in the STL Pioneer proposal. Redundancy of key components was essential to high reliability.

TABLE 3–4.—*Potential Elements of Failure Mode Protection*[a]

Subsystem	Redundancy	Result	R^*
Decoder	Parallel redundant and cross-strapped to each receiver	Command link is retained if internal failure occurs to one decoder.	12.4
General-purpose converter	General-purpose converter paralleled	Internal failure of one converter allows operation of the nonredundant critical functions.	4.3
Digital telemetry unit programmer and analog-digital converter	Programmer and analog-to-digital converter series parallel redundant; ground command able to select either path	Internal failure of one programmer or converter in a series path does not catastrophically affect mission success.	8.9
Receiver and antenna	Receiver able to be switched from wideband to narrowband only;[b] receiver and general-purpose converter series-parallel redundant; receivers cross-strapped to the high- and low-gain antennas	Removes possibility of inadvertently switching from the high-gain mode during the extended mission (after 6 months). Internal failure of one receiver or associated converter does not catastrophically affect mission success. Ability to switch antennas gives greater reception flexibility.	25.6
Orientation subsystem	Quad-redundancy provided at the part level for valves, valve drivers, sensors, sensor drive and filters, and flip-flops; dual-parallel redundancy provided at the part level for all other assemblies	Two, or in some cases, three of the elements can fail without catastrophic results. Failure of Type-I orientation maneuver to begin at separation can be corrected by ground command.	3.1
Command distribution unit	Enable driver with parallel redundant coils and parallel series cross-strapped contacts	One open enable driver coil or one or two open or shorted contacts and failure of two of the 16 silicon controlled rectifiers can occur without catastrophic results.	7.8
TWT converter and TWT	The TWT converter and TWT series-parallel sequence redundant,	Internal failure of one transmitter converter or TWT in a series	35.3

TABLE 3–4.—*Potential Elements of Failure Mode Protection* (Concluded)

Subsystem	Redundancy	Result	R^*
	ground command able to select either path; 8- and 5-W power modes and high- and low-gain antennas that can be selected by ground command[b] also available	path does not catastrophically affect mission success. The capability for switching antennas and/or power modes gives added flexibility. For example, if the available solar-array power is reduced by meteoroid damage, excessive radiation, etc., the 5-W power mode can be used as a backup.[b]	
Power supply, solar array, battery, and undervoltage control	Oversize solar array incorporating multiple redundancy by means of extensive cross-strapping	The array allows for output degradation and random failures; multiple cell cross-strapping allows for the anticipated failure mode.	2.7[c]

[a] See reliability diagram, fig. 3-1. The spacecraft components and subsystems are described in detail in ch. 4. This table from the STL feasibility study is intended to show the design approaches to high reliability; not all were used.
[b] This stratagem was not employed on the actual spacecraft.
[c] Solar array redundancy not included.
R^* = Reliability improvement, percent (total $R = 0.59$ to 0.86).

a field of 1γ at a distance of 3 in. after the application and removal of a 25-G field. Inductances and relay coils with magnetic cores are even more offensive to the magnetometers on board. Unless some concerted action is taken, the cumulative fields of 10 000-plus parts on a Pioneer-class spacecraft can completely overwhelm the interplanetary field of a few gammas.

Early interplanetary craft, such as the first Mariners, were not very clean magnetically and absolute measurements of the interplanetary field were more difficult. The first intensive efforts to build clean spacecraft came with the Goddard Space Flight Center series of Interplanetary Monitoring Platforms (IMPs); the first three IMPs were Explorers 18, 21, and 28. The IMP techniques were borrowed and extended for the Ames Pioneers—the first spacecraft to be designed magnetically clean from the start (ref. 4).

Comparing the pro-reliability and pro-magnetic cleanliness philosophies in table 3–3, one notes many similarities: approved parts lists, stipulations about components usage, and, of course, the same sorts of rigorous qualification and test programs. (See ch. 6 for magnetic test program.)

The TWTs in the communication subsystem were the "dirtiest" spacecraft components, due to their platinum-cobalt magnet assemblies. Since the TWTs could not operate without the magnetic fields, the only solution lay in magnetic compensation; that is, placing other permanent magnets so that their fields combined to cancel those from the TWT magnets in the vicinity of the magnetometer.

In some cases, it was possible to arrange with manufacturers for a special run of nonmagnetic components for Pioneer. Tantalum capacitors, for example, were procured in this manner by STL and the experimenters.

The combination of all these philosophies—compensation, parts screening, use of twisted leads, avoidance of ground loops, and careful attention to details exemplified in table 3–3—made the Pioneers the cleanest space-

TABLE 3–5.—*Evolution of the Pioneer Spacecraft*

Point in time: STL feasibility study complete		Point in time: Ames Specification A–6669 issued		Point in time: Pioneer 6 complete	
Spacecraft	96.2 lb	Spacecraft	111.24 lb	Spacecraft	102.7 lb
Experiments	20.0			Experiments	34.3
	116.2 lb				137.0 lb
Changes from early STL conceptual study		Changes from STL feasibility study		Changes from first version A–6669	
Antenna supports removed		TWT substituted for amplitron		Ames micrometeoroid experiment deleted	
Battery added		Flat solar-cell modules changed to curved modules		Stanford radio propagation experiment added	
		Proposed experiment complement changed by NASA		Solar sail added to antenna mast	
		Four booms added to top of spacecraft (fig. 3–4)		Three booms located on spacecraft viewing band (fig. 3–5)	
				Solar cells removed from viewing band	
				Thermal insulation added to protect spacecraft from X–258 exhaust plume	
				Magnetometer moved from antenna mast to radial boom	

TABLE 3–5.—*Evolution of the Pioneer Spacecraft* (Concluded)

Point in time: Pioneer 7 complete		Point in time: Pioneer 8 complete		Point in time: Pioneer 9 complete		Point in time: Pioneer E complete	
Spacecraft	103.26 lb	Spacecraft	106.1 lb	Spacecraft	107.13 lb	Spacecraft	106.54 lb
Experiments	35.09	Experiments	38.0	Experiments	41.27	Experiments	41.06
	138.35 lb		144.1 lb		148.40 lb		147.60 lb
Changes from Pioneer 6		Changes from Pioneer 7		Changes from Pioneer 8		Changes from Pioneer 9	
Magnetometer range reduced to $\pm 32\gamma$ Energy windows and angular resolution of cosmic-ray experiment changed		Block-II experiments substituted Telemetry format altered Larger battery added for experiments		Ames magnetometer substituted for Goddard magnetometer Convolutional coder experiment added Texas Instruments solar cells substituted for RCA cells Thick glass covers placed on Sun sensors		Ultraviolet filters substituted for thick glass covers on Sun sensors	

craft built to date. The field due to the spacecraft in the vicinity of the magnetometer was roughly 0.5γ, or an order-of-magnitude below the interplanetary field being measured.

EVOLUTION OF THE SPACECRAFT DESIGN

The Pioneer spacecraft design changed in several minor ways during the seven years between conception and the launch of Pioneer E in 1969. The basic system design—a spin-stabilized spacecraft oriented with its spin axis perpendicular to the plane of the ecliptic, holding the Earth perpetually in its disk-shaped antenna pattern—was absolutely essential to the scientific success of a small, low cost interplanetary probe. Basic system design could not be changed, but spacecraft details could. It is appropriate here to survey the more important of these evolutionary (not revolutionary) changes before a large-scale perspective is lost in the next chapter's welter of details.

At the risk of some oversimplification, table 3–5 divides Pioneer spacecraft evolution into periods and lists the important changes that took place within each time frame. The original STL concept was born of its family of Able spacecraft. The most important changes were made during the feasibility study and NASA's issuance of the basic Pioneer Specifications (fig. 3–3). Understandably, this was the period of greatest flux, as NASA and STL focused on a design that would best meet the engineering, scien-

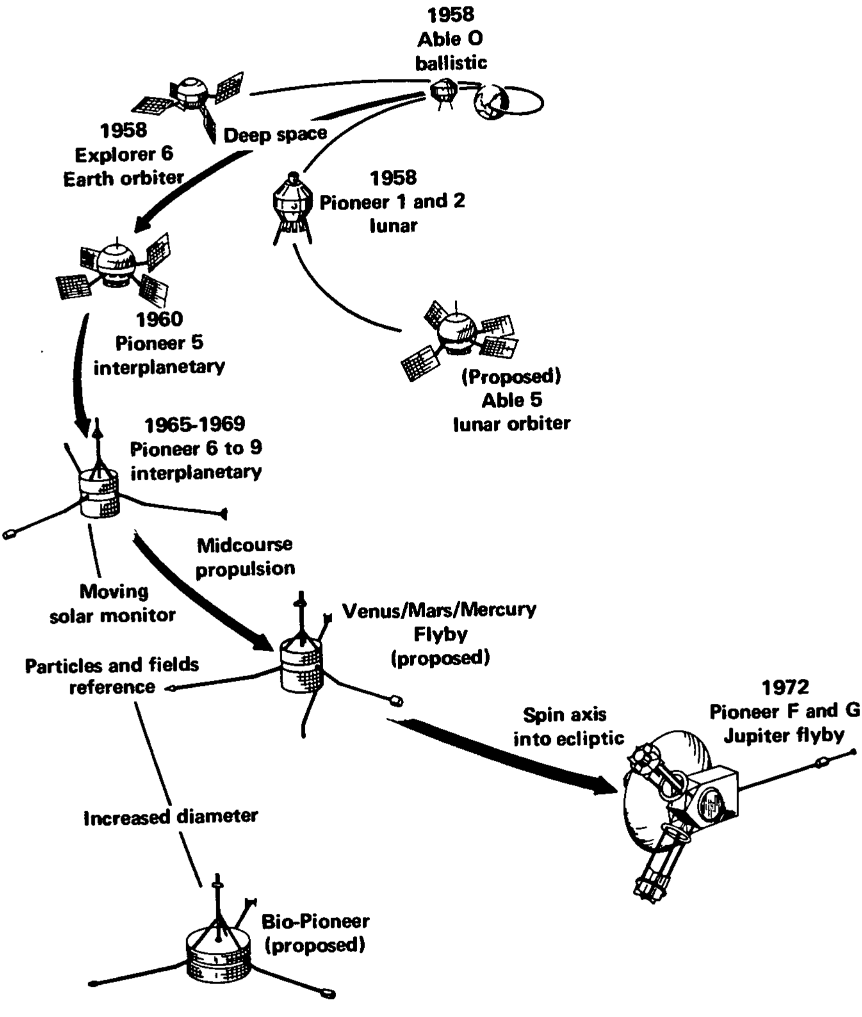

FIGURE 3–3.—Sketch showing the evolution of the STL Pioneer "family." Mainline evolution shows the changes from the spherical to cylindrical to box geometry and the changes from solar-cell paddles to body-mounted cells to radioisotope thermoelectric generators (RTGs). Note that the sketch of Pioneers F and G represents an early version.

tific, and political requirements. The flight of Pioneer 6 in late 1965 did not end spacecraft evolution, though it dampened the magnitude of the changes. There was little change within Block I (Pioneers 6 and 7). In fact, Pioneer 7 was originally intended to be a backup spacecraft for Pioneer 6, but this philosophy was changed in favor of two separate pairs of spacecraft making up Blocks I and II, with enough qualified spares for

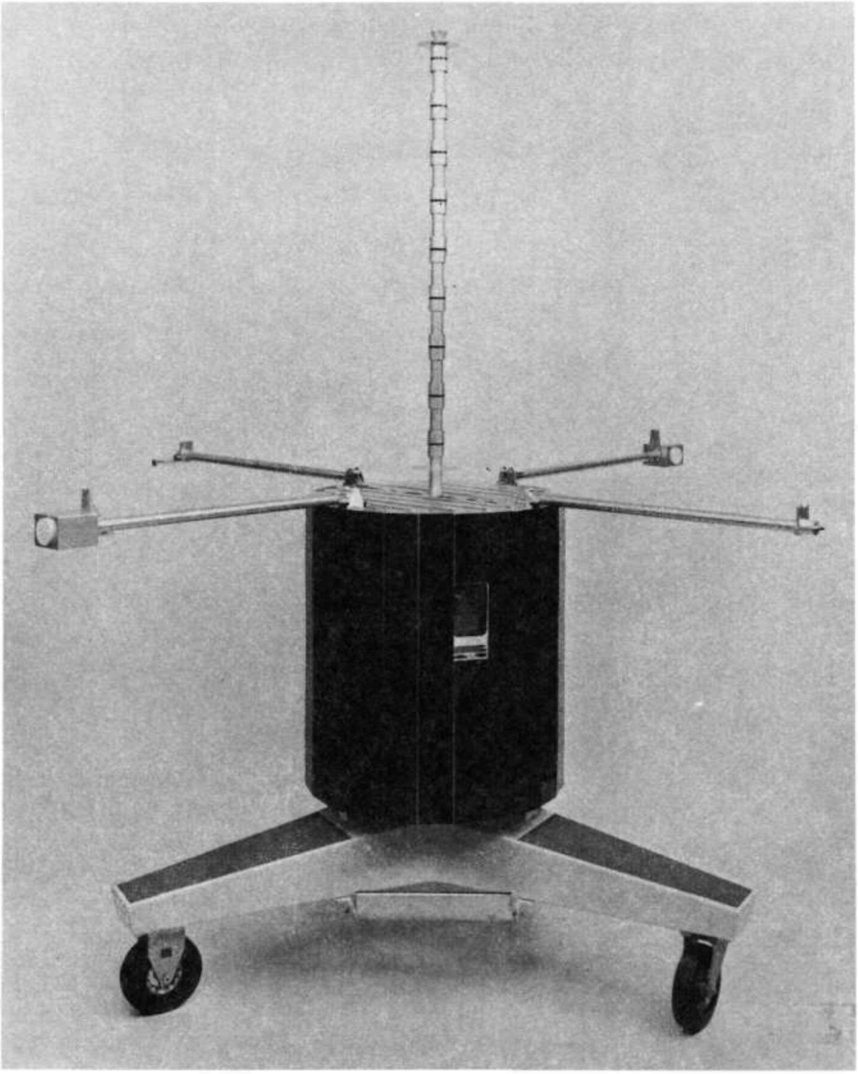

FIGURE 3-4.—Model of the Pioneer configuration proposed by STL in Mar. 1963. Note the four booms mounted at the top of the spacecraft. Earlier designs had no booms at all. (Courtesy of TRW Systems.)

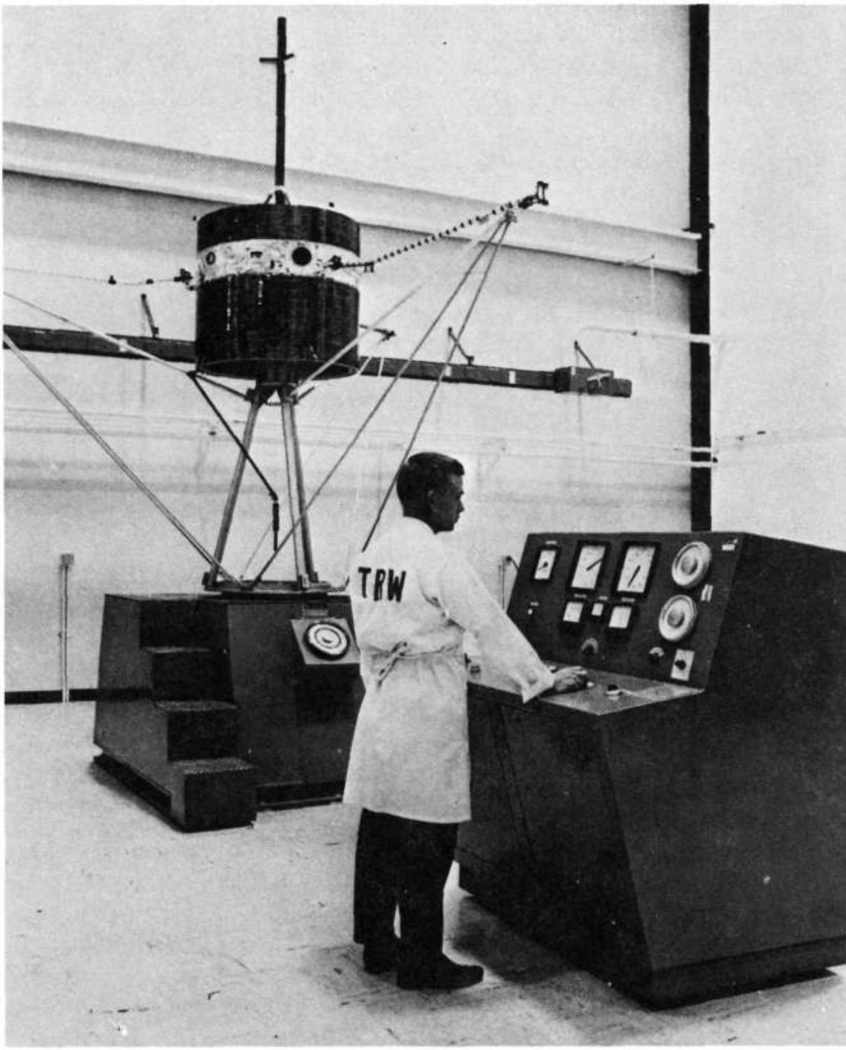

FIGURE 3-5.—The Pioneer-6 spacecraft, showing final boom arrangement and the viewing band void of solar cells. Spacecraft is undergoing a spin test.

a fifth vehicle. For budgetary reasons the fifth spacecraft was dropped early in the program, but it was reinstated as Pioneer E in 1968.

With the substitution of a new array of experiments, several changes had to be made in the communications and electric power subsystems of the Block II spacecraft. During the latter part of the program, the payload capability of the Delta had increased to the point where an engineering experiment, a convolutional coder, could be added to Pioneers 9 and E.

The three Block-II flights carried piggyback TETR satellites as well as ballast.

All five Pioneers (6, 7, 8, 9, and E) were similar with only slight changes from spacecraft to spacecraft—a larger battery on Pioneer 8, an ultraviolet Sun sensor filter on Pioneer 9, etc., as shown in table 3–5. (See also figs. 3–4 and 3–5.) With this overview, the discussion can proceed to detailed descriptions of the spacecraft subsystems.

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CHAPTER 4

The Spacecraft Subsystems

THE SEVEN PIONEER SPACECRAFT SUBSYSTEMS are defined by their various functions, such as communication, data handling, power generation, etc. (See table 1–2.) At the subsystem level in the Pioneer hierarchy of supersystem, system, and subsystem, the first engineering details begin to emerge. It is well known, however, that the subsystem engineer too often visualizes the spacecraft system as a collection of obscure black boxes dominated by the subsystem he is designing. The interface concept, discussed in the preceding chapter, helps dispel this myopia. Interfaces must be matched wherever signals, power, heat, and mechanical forces move from one subsystem to another. The following word portraits of the Pioneer subsystems and their interrelations will quickly dispel any thought that spacecraft subsystems can ever be independent black boxes. The solid lines separating the subsystems in the block diagram of figure 4–1 represent artificially constructed conceptual walls only.

THE COMMUNICATION SUBSYSTEM

Compared with other interplanetary spacecraft—the U.S. Mariners and the Russian Veneras—the IQSY Pioneer spacecraft are factors of 5 and 20 lighter, respectively. Yet the much smaller Pioneers have done more than hold their own in the competition for honors in long-distance communication. It is impressive to be present in the Missions Operations room at Ames Research Center when several of the Pioneers are being worked simultaneously by DSN antennas that have the spacecraft in view from various locations around the world. Taking the Sun's pulse simultaneously from several locations across tens of millions of miles is a *tour de force* in communications engineering.

The basic problems in long-distance communication (ref. 1) are distance and natural radio noise. The inverse-square law cannot be circumvented and there is no way to turn off galactic and solar radio noise. Bigger spacecraft overcome these obstacles with a combination of high-power transmitters and high-gain, onboard paraboloidal antennas pointing directly at the Earth. Compared to most Earth satellites, the Pioneers have high-power transmitters for their weight class, but they cannot afford the added weight and complexity of paraboloids that can be pointed Earthward. Of course, the Pioneer high-gain Franklin-array antenna is pointed, in a sense.

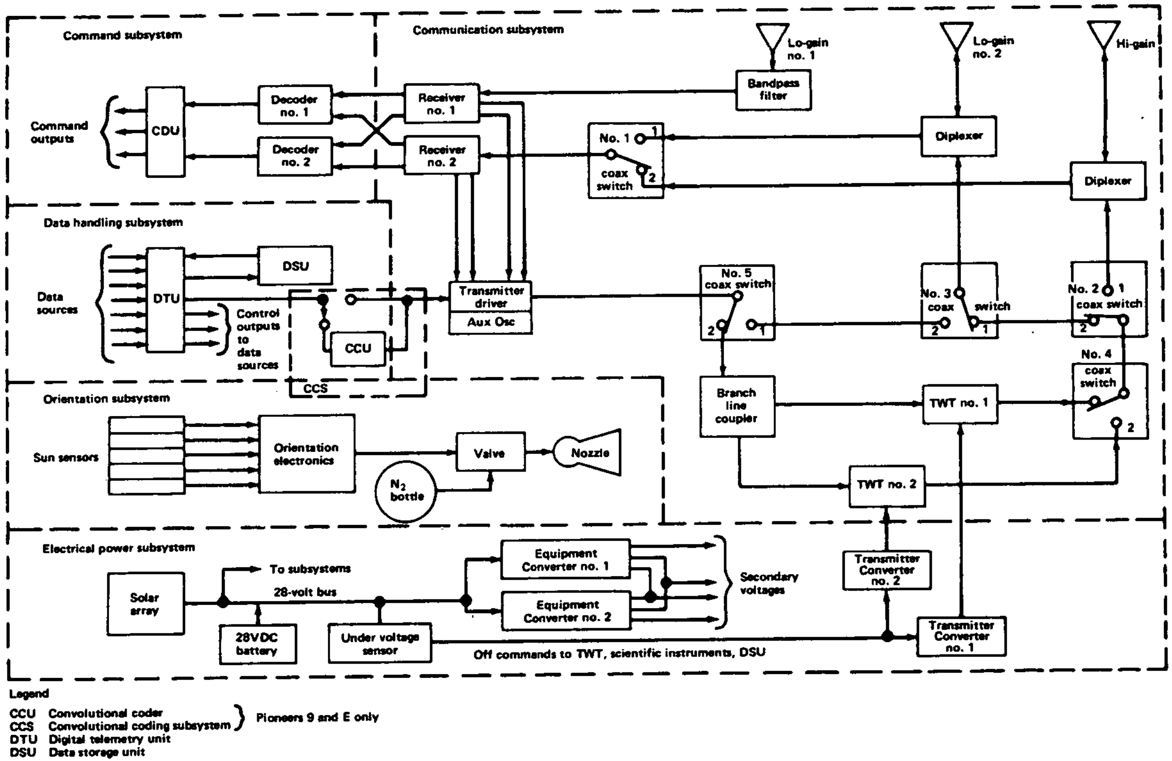

FIGURE 4–1.—Simplified block diagram showing the Pioneer spacecraft subsystems.

When the spacecraft's spin axis is oriented perpendicular to the plane of the ecliptic, the thin, disk-shaped beam intercepts the Earth. The gain of this type of antenna, however, is much lower than that of the pointable paraboloid.

Another factor entering the space communication picture is bandwidth. To transmit information rapidly, that is, attain a high bit rate, a wide-bandwidth is necessary—however, the wider the bandwidth the more power required by the spacecraft transmitter. Thus, transmitter power, bit rate, and antenna gains are all involved in the "communication trade-off." In the Pioneer concept transmitter power is fixed, but bandwidth and bit rate may be reduced by terrestrial command as the spacecraft recedes from Earth. In sum, the Pioneer spacecraft have relatively high transmitter power levels, moderate antenna gains, and variable bit rates; the last can be made very small (8 bits/sec) at extreme distances by command from Earth.

Telemetry of scientific data across the solar system is the most critical of the communication subsystem's functions. Two other communication functions greatly increase the spacecraft's value: the ability to receive commands from the Earth, and the transmission of signals that allow measurements of spacecraft radial velocity from the Doppler effect. The versatilities of the scientific experiments and the spacecraft equipment depend upon the ability to change modes of operation through commands from the Earth.

The final requirement placed upon the communication subsystem is maintaining communication with the spacecraft during the launch sequence, the coast period, and injection. After injection, the spacecraft must respond to commands that initiate the orientation maneuvers during which the entire spacecraft, carrying the rigidly mounted high-gain antenna, is torqued around so that its disk-shaped lobe intercepts the Earth. The high-gain antenna is useless for long distance communication until the completion of the orientation maneuvers. Prior to these maneuvers, communication is maintained between spacecraft and ground through two low-gain antennas, which have nearly isotropic reception patterns.

Once the launch pad umbilical cords are jettisoned, the communication subsystem interfaces first with the DSN 85-foot paraboloid at Johannesburg. As it ascends, other DSN antennas come into view. The DSN has been presented in earlier chapters as more than a communication interface. During the formulation of the Pioneer Program it also acted as a constraint upon the design of the communication subsystem. The JPL approach to interplanetary communication, with its phase-locked loops, phase-shift keying, and Doppler tracking was well-proven by the time the IQSY Pioneers reached the design stage. There was no reason to examine other schemes; the DSN was operational and it would have taken considerable time and money to implement any other communication scheme.

Just as the Delta exerted great influence upon the weight and volume of the spacecraft, the extant DSN dictated answers to the questions the space communications engineer usually asks at the beginning of a new spacecraft design. It should also be remembered that the spacecraft/DSN interface was not static. During the course of the Pioneer Program, the DSN improved its signal detection capability by a factor of about 10 dB, mainly through the introduction of the Goldstone 210-ft antenna (ch. 8). Furthermore, at the beginning of the Pioneer Program, the DSN was not fully operational as an S-band system.

The interfaces between the communication subsystem and the onboard spacecraft subsystems were less momentous. They are dealt with in table 4–1.

The major components of the communication subsystem are: one high-gain and two low-gain antennas, two receivers, a transmitter driver, two TWT power amplifiers, and five coaxial switches that can be activated from the ground to switch in redundant components should failures or anomalous operation occur (fig. 4–2). Although telemetry, commands, and tracking information are all handled by the communication subsystem, one cannot distinguish three separate, corresponding groups of

TABLE 4–1.—*Communication Subsystem Interfaces*

Subsystem	Interface considerations
Data handling	The communication subsystem receives telemetry words from the data handling subsystem and transmits them to Earth.
Command	Uplink commands are received by the communication subsystem and passed on to the command subsystem.
Electric power	The communication subsystem is the largest single user of electric power on the spacecraft.
Orientation	A feedback control loop exists during the orientation maneuvers as the spacecraft is torqued via terrestrial commands into a position where signal strength received at a DSN station is maximum.
Thermal control	The communication subsystem is also the largest producer of waste heat (primarily the TWT).
Structure	This provides mechanical support of electronic packages and three antennas.
Scientific instrument system	The data handling subsystem acts as a buffer here. The spacecraft's end-mounted antennas do not compete for solid angle with instruments scanning the plane of the ecliptic. The magnetic interface with the TWT magnets is controlled by compensating magnets.
Tracking and data acquisition system	The electromagnetic interface and DSN constraints are discussed in chapter 8.
Launch vehicle system	A standard launch vehicle system is used.

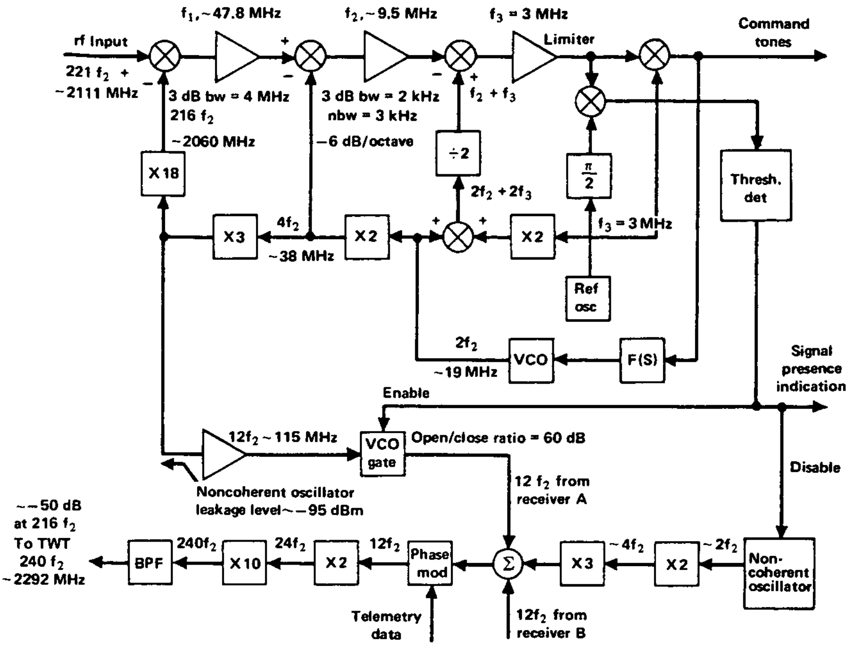

FIGURE 4–2.—Simplified receiver and transmitter driver block diagram.

subsystem components; they are all integrated into the basic subsystem.

To understand phase-lock loop operation, picture a Pioneer spacecraft 100 million miles or so ahead of the Earth in its orbit about the Sun. Assume first that the terrestrial DSN antennas are busy with some other spacecraft. In this situation, both spacecraft receivers are in a state of readiness for further instructions from Earth. The transmitter, however, still transmits any scientific and housekeeping information it receives from the data-handling subsystem even though no terrestrial antenna intercepts it. Thus, even if both spacecraft receivers should fail, DSN antennas can still acquire the spacecraft and record whatever data it transmits. During these periods, when the spacecraft is "on its own," the spacecraft transmitter frequency is controlled by an internal crystal-controlled oscillator. This is called the noncoherent mode of operation. This one-way Doppler tracking can be accomplished by merely listening to the spacecraft. The angular coordinates of the spacecraft can be measured accurately by the DSN antennas but, as explained in chapter 8, Doppler measurements suffer because the spacecraft oscillator frequency may drift slightly and introduce range-rate uncertainties. Only the functions of telemetry transmission and (limited) tracking can be performed during this type of operation.

Suppose, next, that a DSN antenna is swung around to point in the direction where orbital computations predict the spacecraft will be. DSN receivers pick up (acquire) the weak telemetry signal and "lock on" to it. Lock is attained by means of a feedback loop involving a narrow bandpass filter and a voltage-controlled oscillator. A down-link lock exists when the voltage-controlled oscillator generates a signal at precisely the carrier frequency received from the spacecraft but with a 90° phase change. The feedback circuit in essence operates as a servomechanism to force the oscillator to match the spacecraft carrier frequency. Once a down-link lock has been established, the ground transmitter sends its own carrier in the direction of the spacecraft. Since the two spacecraft receivers are tuned to operate at different frequencies, the ground transmitter can select either one by using the proper carrier frequency. The presence of a signal in the spacecraft receiver automatically disconnects the spacecraft crystal-controlled oscillator and switches in a voltage-controlled oscillator that generates a frequency precisely 12/221 times that received from the DSN. This frequency is then multiplied by twenty in the transmitter driver. A phase-coherent transmitter signal with a frequency 240/221 times the frequency received from Earth is amplified in the operational TWT and dispatched to Earth via the high-gain antenna. The waiting DSN antenna locks onto this signal, which may be slightly different from that originally acquired because the spacecraft's crystal-controlled oscillator drifts slightly. Only when the spacecraft and DSN receivers are both locked on the signals received from Earth and spacecraft, respectively, can coherent, highly accurate, two-way tracking measurements be made.

While the spacecraft and DSN station are operating in phase-lock loop modes, telemetry signals are sent to Earth by phase-shift keying (PSK) of a 2048-Hz subcarrier that phase modulates (PM) the main carrier. Commands are sent up-link by using frequency-shift keying (FSK). Both commands and telemetry are pulse-code modulated (PCM). The Pioneer telemetry and command systems are therefore designated PCM/PSK/PM and PCM/FSK/PM, respectively. The information carried on the 2048-Hz subcarrier does not interfere with coherent Doppler measurements being made on the transmitter carrier which is in the 2290- to 2300-MHz range (part of the S-band).

The communication subsystem block diagram (fig. 4–2) shows five coaxial switches that give the subsystem appreciable flexibility should a failure or some anamolous event occur. Seven different coaxial switch commands comprise what is called the rf logic for Pioneer, as shown in table 4–2.

Note that one low-gain antenna can always receive commands regardless of switch positions or operability. Coaxial switches nos. 4, 5, and 2, however, must operate properly for the spacecraft to transmit telemetry. In other words, the coaxial switches are in line and essential to mission success.

TABLE 4–2.—*Switching Logic*

Command number	Function	Switching sequence
046	Driver to low-gain antenna	S3–2, S5–1[a]
047	Operational TWT to high-gain antenna	S2–1
025	Operational TWT to low-gain antenna	S2–2, S3–1
015	TWT no. 1 to operational status	S4–1, S5–2, S3–1
022	TWT no. 2 to operational status	S4–2, S5–2, S3–1
033	Receiver no. 2 to high-gain antenna	S1–2
003	Receiver no. 2 to low-gain antenna	S1–1

[a] S5–1 = Switch no. 5 commanded to position no. 1.

Spacecraft Receiver

The two tasks assigned to the spacecraft receiver are:

(1) To detect, demodulate, and amplify the commands impressed upon the carrier that is received from the DSN station working the spacecraft

(2) To provide to the transmitter driver a phase-coherent signal 12/221 times the frequency of the received DSN carrier.

The up-link signal loss due to inverse-square-law attenuation and absorption is approximately 264.27 dB when a Pioneer is 100 million miles from Earth. To overcome this power loss, the spacecraft receiver can be made extremely sensitive; and the DSN stations, being ground-based, can afford to pump considerable power into a very narrow beam and point it directly at the spacecraft. In fact, the DSN station transmitter power is rated at 10 000 W compared with the spacecraft's 8 W. The seemingly easy up-link communication task is compounded by the necessity for making the command information more error-free than spacecraft telemetry. This is understandable because a spurious command could conceivably turn spacecraft off permanently by accident. Therefore, Pioneer up-link power budgets are calculated assuming the very low bit-error rate of 10^{-5}. The power budget having an appropriate signal-to-noise ratio for this low bit-error rate is presented in table 4–3.

The small, lightweight Pioneer receiver was developed by STL and had already successfully flown on many spacecraft before it was adopted for the Pioneer spacecraft. A block diagram of the receivers and transmitter driver is presented in figure 4–2. Note that the threshold detector disables the on-board, crystal-controlled, noncoherent oscillator whenever an external signal from the DSN is detected. The coherent receiver then generates the phase-coherent signal that ultimately drives the TWT at 240/221 times the received frequency. There are, of course, many ways to build receivers to accomplish the tasks prescribed for the Pioneer receivers. An

TABLE 4–3.—*Uplink Power Budget*

Requirement	Value via high-gain antenna	Value via low-gain antenna
Parameter		
Total ground transmitter power (10 kW)	70.0 dBm	70.0 dBm
Circuit loss (diplexer, switch, waveguide)	0.4 dB	0.4 dB
Ground antenna gain (85-ft paraboloid)	51.0 dB	59.0 dB
Space attenuation (2110 MHz; 30×10^6 n. mi.)	----------	253.83 dB
Space attenuation (2110 MHz; 100×10^6 n. mi.)	264.27 dB	----------
Polarization loss (1.0 ± 0.5 dB ellipticity)	3.00 dB	3.01 dB
Spacecraft antenna gain	10.5 dB	−1.0 dB
Spacecraft circuit loss	1.5 dB	1.0 dB
Net transmission loss	207.68 dB	208.24 dB
Total received power	−137.68 dBm	−138.24 dBm
Receiver noise spectral density (10 dB noise figure)	$-164.0 \frac{\text{dBm}}{\text{Hz}}$	$-164.0 \frac{\text{dBm}}{\text{Hz}}$
Carrier loop performance		
Carrier modulation loss (1.2 ± 5 percent radian peak deviation)	3.46 dB	3.46 dB
Received carrier power	−141.14 dBm	−141.70 dBm
Carrier loop noise bandwidth ($2B_L = 25$ Hz at −141 dBm)	14.0 dB	14.0 dB
Threshold signal-to-noise ratio in $2B_L$	6.0 dB	6.0 dB
Threshold carrier power	−144.74 dBm	−144.74 dBm
Performance margin (low-gain antenna; 30×10^6 n. mi.)	----------	+3.0 dB
Performance margin (high-gain antenna; 100×10^6 n. mi.	+3.6 dB	----------
Command data performance		
Data modulation loss (1.2 ± 5 percent radian peak deviation)	3.04 dB	3.04 dB
Received data power	−140.72 dBm	−141.28 dBm
Data noise bandwidth (2.0 ± 0.5 Hz)	3.0 dB	3.0 dB
Data threshold signal-to-noise ratio (probability of bit error $= 10^{-5}$)	13.4 dB	13.4 dB
Degradation from theoretical	3.0 dB	3.0 dB
Threshold data power (0.25 dB limiter suppression)	−144.31 dBm	−144.31 dBm
Performance margin (low-gain antenna; 30×10^6 n. mi.)	----------	+3.0 dB
Performance margin (high-gain antenna; 100×10^6 n. mi.)	+3.6 dB	----------

important feature of the STL design is that the frequencies of the phase detector and reference oscillator are not related in any simple way to the incoming frequency when divided by 221. This offset technique makes "self-lock" unlikely; that is, the receiver will not lock in error on a subharmonic of some frequency created in the receiver itself.

In constructing the Pioneer receivers, STL employed discrete circuits rather than the integrated circuits so common in more recent spacecraft. Integrated circuits were just coming into their own in 1963–64, and STL and NASA felt that they were not adequately proven to incorporate them in a spacecraft aiming at a 6-month life as a minimum. The circuit construction, like the design, was derived from previous STL space programs, notably the Able series and the "telebit" technology used on Explorer 6.

Perhaps the most critical transmitter and receiver components were the in-line coaxial switches (fig. 4–3). As mentioned earlier, these had to work after launch before all spacecraft functions could be consummated. During the component test program some of the coaxial switches proved unreliable, remaining stuck in the unproductive middle position. The cause was ultimately traced to a procedural problem in the supplier's plant, and corrected. Nevertheless, some concern always remained that when one of the coaxial switches was commanded to change its polarity it would stick in a middle position, disconnecting the antennas from the TWTs. Fortunately, this never happened in space; the Pioneer coaxial switches have perfect performance records.

The telemetry link from the spacecraft to the waiting DSN antenna possesses a power budget analogous to that for the uplink (table 4–4). Here,

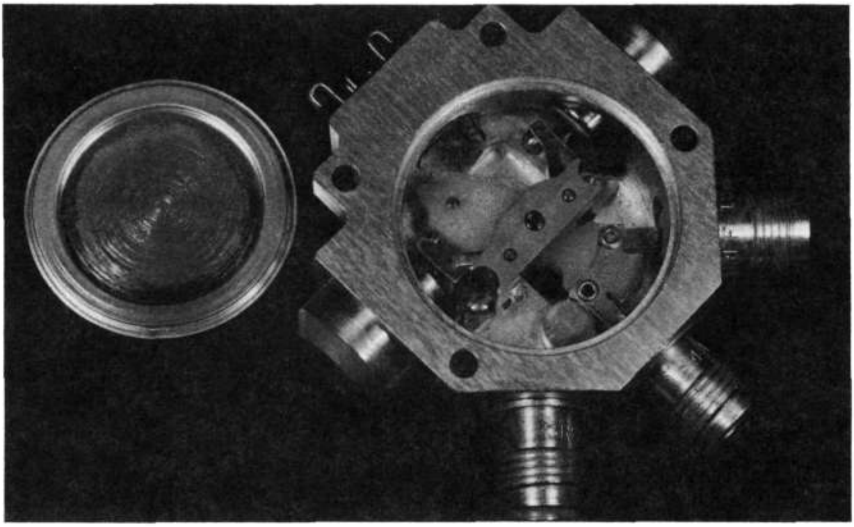

FIGURE 4–3.—One of the Pioneer coaxial switches. (Courtesy of TRW Systems.)

TABLE 4–4.—*Downlink Power Budget*

Requirement	Value
Parameter	
Spacecraft transmitter power (7.7 W)	38.86 dBm
Circuit loss (diplexer, switch, coaxial cable)	1.7 dB
Spacecraft transmitting antenna gain	11.2 dB
Space attenuation (2292 MHz; 41.5×10^6 n. mi.)	257.36 dB
Polarization loss (including antenna pointing loss)	3.0 dB
Ground receiving antenna gain (85-ft paraboloid)	53.0 dB
Ground circuit loss (diplexer, switch, waveguide)	0.18 dB
Net transmission loss	198.04 dB
Total received power	−159.18 dBm
Receiver noise spectral density ($T_s=55\pm10°$ K)	$-181.2\frac{\text{dBm}}{\text{Hz}}$
Carrier loop performance	
Carrier modulation loss (0.9 radian peak deviation, ±5 percent)	4.12 dB
Received carrier power	−163.30 dBm
Carrier loop noise bandwidth ($2B_L=23.5$ Hz)	13.71 dB
Signal-to-noise ratio in $2B_L$	4.2 dB
Threshold signal-to-noise ratio in $2B_{LO}=12$ Hz	6.0 dB
Threshold carrier power	−164.43 dBm
Performance margin	+1.13 dB
Data channel performance	
Data modulation loss (0.9 radian peak deviation, ±5 percent)	2.14 dB
Receiver i.f. and limiter degradation	0.95 dB
Receiver data power	−162.27 dBm
Data noise bandwidth (8 Hz)	9.03 dB
Signal-to-noise ratio in data bandwidth	9.91 dB
Carrier loop degradation	1.03 dB
Sync and subcarrier loop degradation	1.0 dB
Adjusted data signal-to-noise ratio	7.88 dB
Data threshold signal-to-noise ratio [a] (probability of bit error $=10^{-3}$)	7.3 dB
Performance margin	+0.58 dB

[a] Signal-to-noise ratio is defined as the ratio of average signal power to the noise power in a bandwidth equal to the bit rate. Noise power is computed using the single-sided noise spectral density.

again, the real technological burden is placed on the ground equipment instead of the spacecraft. The big high-sensitivity DSN paraboloids with their low-noise amplifiers are essential to long distance communication with the Pioneers. Despite terrestrial sophistication, the spacecraft must still generate considerable radio power, most of which is wasted because the Earth occupies only a small sector of the circular disk-like antenna pattern. The Pioneer transmitters generate about 8 W of radio-frequency

TABLE 4–5.—*Frequencies Assigned to the Pioneer Program*

Link	Channel	Frequency (MHz)
Downlink telemetry	6A [a]	2292.037037
	7A	2292.407407
Uplink commands	6B	2110.584105
	7B	2110.925154

[a] Channel 6A is also the nominal frequency of the on-board, crystal-controlled oscillator, which may drift slightly from the assigned frequency during non-coherent operation. In the coherent mode, either channels A or channels B may be used.

power compared to roughly 3 W radiated from Mariner 2's paraboloidal antenna.

Spacecraft Transmitter (Driver and Power Amplifier)

The block diagram of the transmitter driver (fig. 4–2) shows three possible input signals, and a single output signal that drives the power amplifier connected to either the spacecraft high-gain or the low-gain antenna. The driver frequency, which becomes the spacecraft transmitter carrier frequency, is supplied by the receiver. The driver provides either the noncoherent signal from its crystal-controlled oscillator, or the phase-coherent signal that is 240/221 times the DSN carrier frequency. The signals from the data handling subsystem phase-modulate this carrier when they are present. The frequencies assigned to the Pioneer Program are listed in table 4–5.

The transmitter driver consists of a transistorized amplifier-modulator and a varactor multiplier (factor of 20).[6] The amplified signal of approximately 50 mW is fed next to the power amplifier, one of the most critical of all spacecraft components. The power amplifier must deliver about 8 W rf power to the spacecraft antenna with high reliability and high efficiency.

During STL's early studies of the Pioneer mission, four possible power amplifiers were examined: an all solid-state transmitter, the triode amplifier, the amplitron, and the TWT. The solid-state transmitter and triode amplifier were eliminated from consideration because of their low efficiencies. The amplitron, an rf amplifier tube similar to the magnetron, is very efficient—on the order of 50 percent for the power levels being considered for Pioneer. In fact, before the 1962 STL feasibility study, the amplitron was thought to be the best choice. But confidence in the amplitron waned with further study. A satisfactory operational lifetime had not been demonstrated for the amplitron in 1962. In addition, NASA and

[6] A varactor is a type of parametric amplifier.

STL engineers were concerned over the tendency of the amplitron to switch to a noisy mode of operation after a power supply transient, such as that expected when changing from coherent to noncoherent operation. The amplitron was also dirty from the magnetic standpoint. One test generated 1700γ at 3 ft. TWTs seemed the only reasonable choice. Both Hughes Aircraft and Watkins Johnson had TWTs that very nearly met all Pioneer requirements for efficiency, weight, magnetic field, lifetime, and operating frequency. Ultimately the Hughes 349H TWT was selected for power amplification.

The performance of the Pioneer TWTs has been fairly good, but they have always been a source of concern. Hughes had difficulty in meeting the 30 percent efficiency goal; many TWTs had to be discarded before satisfactory tubes were found. NASA was concerned during the early Pioneer flights about the ruggedness of the TWT's hot filaments during the rigors of launch. For this reason, and to conserve the battery, the TWT filaments were not turned on during rocket ascent. A special automatic filament turn-on switch was installed so that the spacecraft could be acquired early by the Johannesburg tracking station. So far, the TWTs have not aborted any Pioneer mission, but shortly after the launch of Pioneer 7, the operational TWT began operating abnormally (its temperature changed, too), and the redundant TWT was switched in. The operational TWT on Pioneer 6 showed similar but less severe symptoms after over 3½ years of operation; however, 3½ years demonstrate a certain measure of reliability.

Spacecraft Antennas

Three antennas serve the communication subsystem. Two are low-gain, multislot types with broad beamwidths. In the subsystems arrangement of coaxial switches (fig. 4–1) one of the low-gain antennas is permanently connected to one of the receivers. This arrangement guarantees that the spacecraft is always listening for commands, regardless of the operability of the coaxial switches. The other low-gain antenna may be connected to the second receiver by ground command. The low-gain antennas are essential during spacecraft acquisition and during the orientation maneuver when the high-gain antenna is being torqued into position. The high-gain antenna—the prominent mast atop the cylindrical Pioneer spacecraft (Figs. 1–4 and 4–4)—is critical to the whole Pioneer concept. Its gain is roughly 10 dB over an isotropic antenna. It contributes to Pioneer's long-range communication capability. The antenna is a colinear broadside array (a modified Franklin array) consisting of nine driven and nine parasitic elements. Commercial TV antennas use similar arrays because they also must focus rf energy into a flat, disk-like pattern, symmetric around 360° (fig. 4–4). More detailed antenna characteristics are presented in table 4–6.

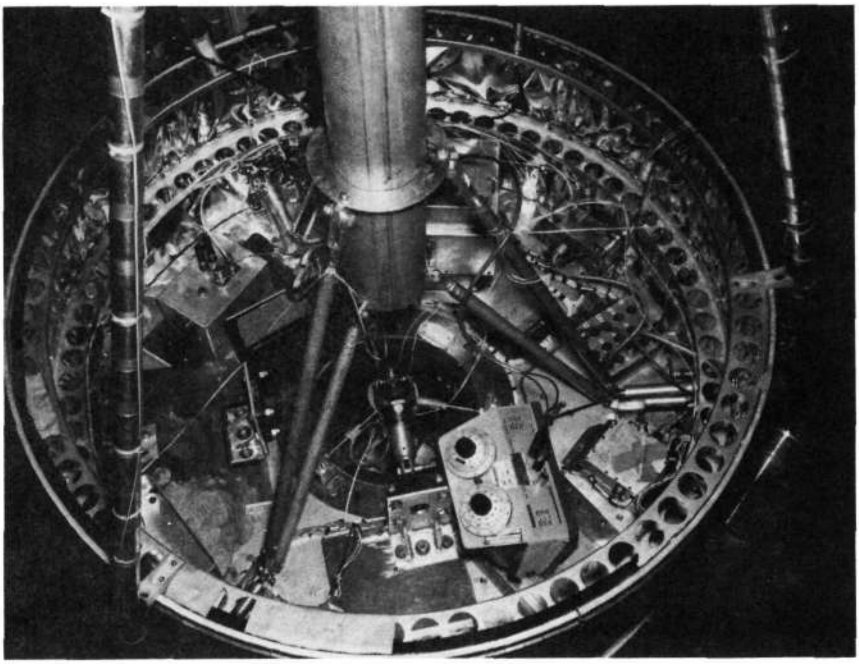

FIGURE 4–4.—Closeup view of the spacecraft with top cover removed. Base of telemetry mast is supported by struts. Some test equipment is shown above the equipment platform.

The spacecraft antennas interface directly with the DSN antennas. All three spacecraft antennas emit linearly polarized electromagnetic waves; the high-gain antenna's plane of polarization is parallel to the spin axis, while the planes of both low-gain antennas are perpendicular to it. Usually, the DSN paraboloids are fitted with an "ultracone" which permits them to receive and transmit circularly polarized waves. The mismatching antenna polarizations result in a 3 dB loss in signal power. Between November 1968 and July 1969, however, "multifrequency" cones were installed in the DSN, enabling the antennas to receive linearly polarized signals without attenuation. In effect, matching the planes of polarization increases the potential communication distance by 40 percent.

A basic weakness in the Pioneer antenna patterns is that they all drop to very low values of gain in the directions viewed by the spacecraft spin axis (fig. 4–5). This causes no trouble when the spacecraft is properly oriented in deep space. However, situations can and do occur, as the spacecraft escapes the Earth's gravitational field and terrestrial antennas must look along the spacecraft axis, where the low-gain antenna sensitivity is so low that the reorientation maneuvers may be compromised by weak com-

TABLE 4–6.—*Spacecraft Antenna Characteristics*

Single-frequency low-gain antenna (multi-slot)	
Uplink (2110 MHz)	
Beamwidth	110° at −3 dB points
Polarization	Linear, perpendicular to spin axis
Gain	−1.5 dB minimum
Downlink—not applicable	
Dual-frequency low-gain antenna (multi-slot)	
Uplink (2110 MHz)	
Beamwidth	110° at −3 dB points
Polarization	Linear, perpendicular to spin axis
Gain	−2.5 dB minimum
Downlink (2292 MHz)	
Beamwidth	85° at −3 dB points
Polarization	Linear, perpendicular to spin axis
Gain	−0.5 dB minimum
Dual-frequency high-gain antenna (collinear broadside array)	
Uplink (2110 MHz)	
Beamwidth	5° at −3 dB points
Polarization	Linear, parallel to spin axis
Gain	+10 dB minimum
Downlink (2292 MHz)	
Beamwidth	5° at −3 dB points
Polarization	Linear, parallel to spin axis
Gain	+10.7 dB minimum

mand signals. The solution to this dilemma was a special maneuver termed "partial orientation," commanded from Johannesburg for Pioneer 6 and from Goldstone (where the normal orientation maneuver was commanded) for Pioneer 9. After partial orientation, the spacecraft is in such an attitude, with respect to terrestrial antennas, that commands dispatched from Goldstone can be heard easily by the spacecraft.

THE DATA HANDLING SUBSYSTEM

The end product of most spacecraft, the Pioneers included, is information. Data flow not only between Earth and spacecraft but also among the various spacecraft subsystems. In the guises of telemetry, commands, and control signals, information is ubiquitous onboard a spacecraft. The data handling subsystem acts as a central clearing house where data are re-

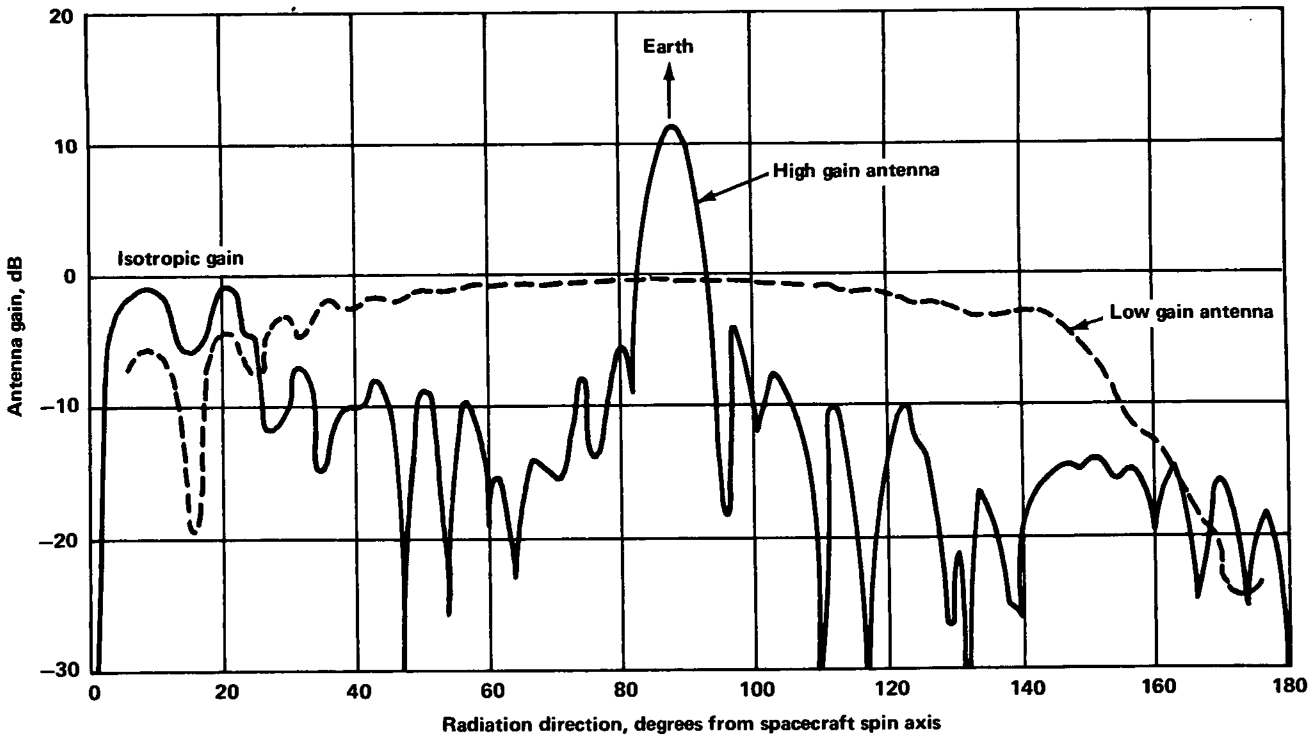

FIGURE 4–5.—Spacecraft antenna radiation patterns.

ceived, formatted, processed, stored, and sent back to Earth or to other Pioneer subsystems.

The functions of the data handling subsystem are:

(1) The sampling and encoding of analog and digital measurements taken by the scientific instruments (In special cases, the encoding is done by the scientific instrument.)

(2) The sampling and encoding of spacecraft engineering or housekeeping measurements

(3) The storage, upon command, of data, when DSN stations are not available to acquire spacecraft data

(4) The storage, upon command, of special data formats, when the spacecraft is communicating with the DSN

(5) The changing, upon command, of data bit rate and/or format as the spacecraft recedes and approaches the Earth (fig. 4–6 shows the impact of this feature on Pioneer-7 communication).

(6) The provision of sundry clock and control signals throughout the spacecraft, in effect forcing all spacecraft experiments and sybsystems to work together in synchronism

Two units, or subsubsystems, make up the data handling subsystems: the digital telemetry unit (DTU), really the data processor, and the data storage unit (DSU), the spacecraft memory (fig. 4–7). A convolutional coder unit (CCU), which could be switched in-line from a standby status or vice versa, was added to Pioneers 9 and E on an experimental basis.

A look at the data handling subsystem as a black box reveals the following inputs:

(1) Scientific and engineering measurements

(2) Commands to change mode of operation

(3) Sun pulses from the Sun sensors to provide spacecraft attitude references

The outputs are only two:

(1) A PCM signal to the transmitter driver

(2) Timing and control signals to the rest of the spacecraft

The input-output view of the data handling subsystem oversimplifies the situation; it does not portray the great flexibility intrinsic in a commandable spacecraft. Commands from Pioneer Mission Control can change operational characteristics of the Pioneer data handling subsystem as follows:

(1) Bit rates available: 8, 16, 64, 256, and 512 bits/sec

(2) Transmission formats available: scientific format A, scientific format B, an engineering format, and a special-purpose format

(3) Modes of operation available: real-time mode, duty-cycle-store mode, telemetry mode, and memory readout mode

On Pioneers 9 and E, the convolutional coder can be switched in and out by ground command to provide another permutation. From the view-

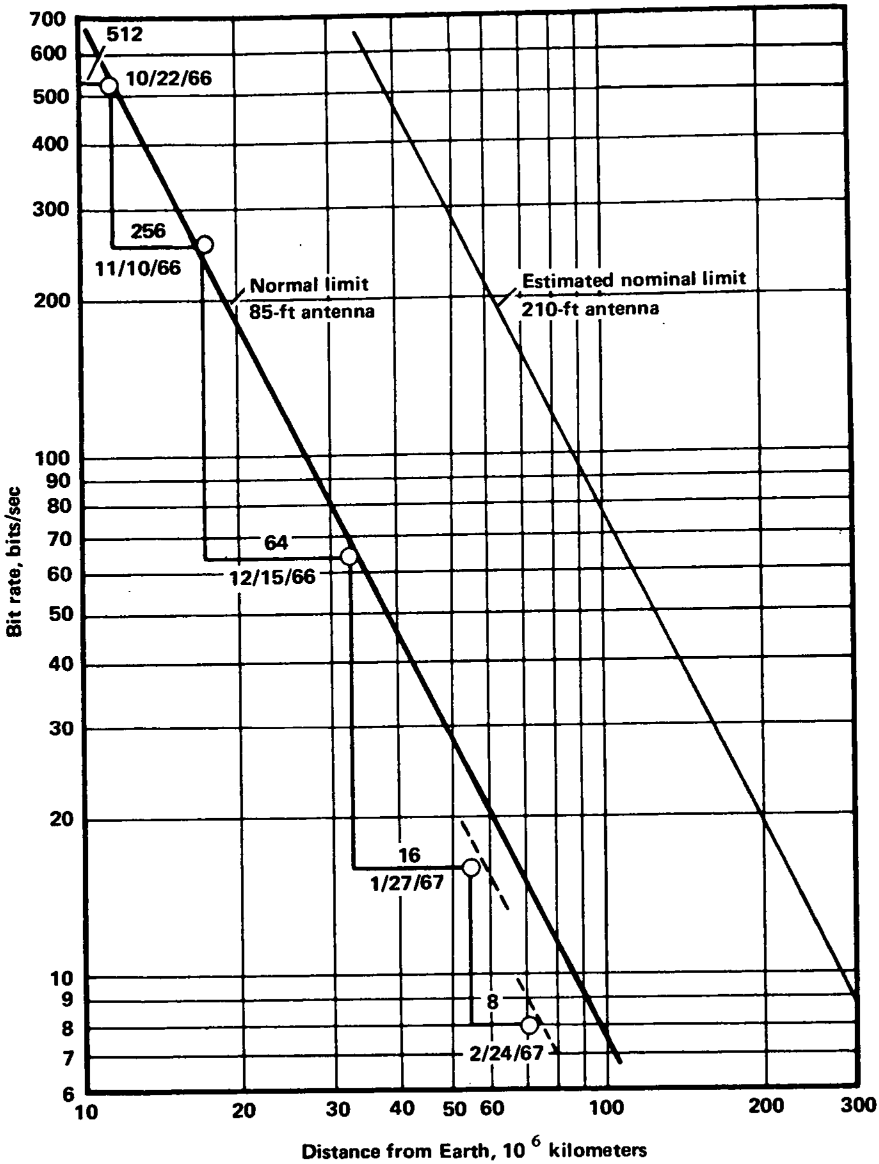

FIGURE 4–6.—Distance limitations for Pioneer 7, showing dates when telemetry bit rate was changed. Note the improved performance with the 210-ft antenna.

point of data acquisition and processing on the ground, Pioneer telemetry may arrive at a DSN antenna in any one of 80 varieties (160 for Pioneer 9), depending upon terrestrial commands. There is a definite need for these different formats and modes as will now be described.

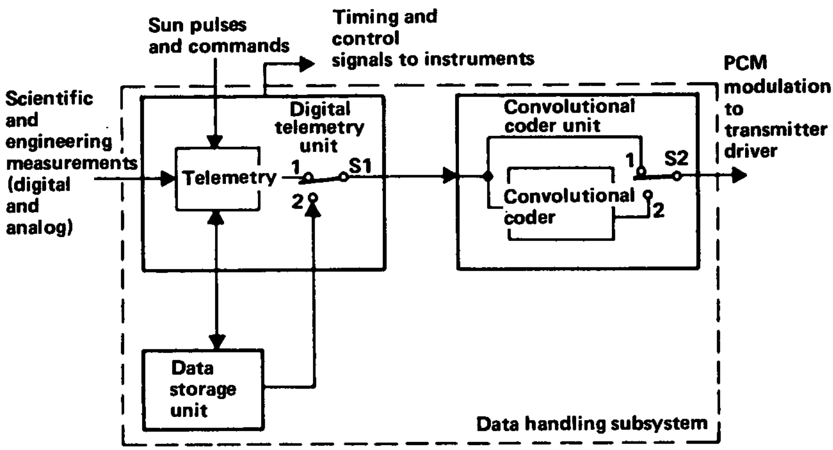

FIGURE 4–7.—Block diagram of the data handling subsystem.

The interfaces of the data handling subsystem are all internal to the spacecraft, since it "sees" the DSN only through the communication and command subsystems. The most important interface separates the data handling subsystem from the several scientific instruments. The design of this interface was controlled by the desire to make the spacecraft as useful as possible to the experimenters. A scientist on Earth looking at telemetry data from his instrument—which is far out in interplanetary space but still commandable at his discretion—wants to know, and be able to do, several things. He might pose the following questions: Where was the Sun when these data were taken? How can I turn my experiment on only when it points directly at the Sun? When the Sun shows signs of unusual activity, how can I record data more often, perhaps at a higher rate than the communication subsystem can handle? He might think his instrument more important during a solar flare than instrument X; however, he would like very much to know what instrument Y recorded at the time he recorded his data. All experimenters cannot be satisfied all of the time; NASA must set priorities and encourage cooperation. The basic reasons for a flexible data handling subsystem with the provision for data storage become apparent when the wishes of the experimenters are considered. And when the situation requires it, NASA mission controllers can alter priorities at will.

Each instrument (and experimenter) is different. Some instruments deliver digital data to the data handling subsystems; others send analog signals. The formats and word structures coming across the interface are also different. Besides being versatile in terms of what it does with the basic data, the data handling subsystem also must be generalized enough to accept a wide range of inputs and convert them all into standardized PCM telemetry for the transmitter driver. It may be considered an "information melting pot."

Reliability Considerations

Only the DTU of the data handling subsystem is permanently in-line; the DTU is absolutely critical to mission success at all times. All spacecraft telemetry must flow through the DTU; high reliability is as essential here as it is with the communication subsystem. According to figure 3–2, the early spacecraft reliability budget chart, the on-line portion of the data handling subsystem must achieve a reliability of 0.948. The DSU, on the other hand, is off-line; that is, it may be off-line if the proper switches are thrown by command from the Earth. Presumably, if the DSU should fail, it can be bypassed completely for the remainder of the mission, although this would reduce versatility by eliminating modes of operation involving data storage. The probability that both DSU and switch would fail together is negligible. The DSU reliability allocation of 0.897 was not figured into the overall reliability assessment for the spacecraft.

The basis for reliable design of both DTU and DSU was much the same as it was for the communication subsystem: employ conservative design practices, use proven techniques, apply redundancy judiciously, and select only well-qualified parts. Modular construction was also required here. In constructing the data handling subsystem, STL was once more able to draw upon many components, circuits, and techniques developed during prior military programs. Almost all subsystem components, for example, had already been qualified for a one-year lifetime in space.

Codes, Words, and Formats

When the Pioneer Program was being formulated in 1962, there existed a general trend in the direction of PCM for space telemetry. The Mariner space probes, NASA's observatory series of satellites, and both the Gemini and Apollo Programs had adopted PCM. PCM has many advantages: unlimited accuracy (in principle), the existence of self-checking and error-correcting codes, and—far from the least—instant compatibility with computers. Because the Pioneers were going to interface with the DSN, with its already strong bias toward digital techniques, there were no overriding technical reasons not to follow the PCM trend.

The bits that constitute each PCM word can be communicated by any one of several two-valued properties of a modulated radio signal. Pioneer PCM bits are impressed upon the transmitter carrier by phase-modulating the 2048-Hz square-wave subcarrier; this follows JPL practice. More technically, the subcarrier is biphase modulated by a time-multiplexed train of bits, using a non-return-to-zero-mark (NRZ–M) format.[7]

[7] On Pioneers 9 and E, the non-return-to-zero-level (NRZ–L) format was introduced.

The basic unit of information in a telemetry message from a Pioneer spacecraft is a seven-bit word. The first six bits represent the instrument reading or datum, with the most significant bit (MSB) appearing first. The last, or seventh, bit is a parity bit based upon the first, third, and fifth bits in the preceding word. If the sum of these bits is odd, the parity bit will also be odd; i.e., one.

The parity bit represents a self-checking feature of the code. Words containing errors introduced during transmission and the many processing steps along the way can be identified and flagged in most instances by recomputing and checking the parity bit for the word that finally arrives at its terrestrial destination. The parity bit, as used in Pioneer telemetry, was worth roughly 2 dB in the sense that transmitted messages could be edited and made more accurate.

PCM words can be made as long as required by the spacecraft instruments. It is often said that PCM words can be made "infinitely accurate." However, the accuracy of much Pioneer scientific data is set by the capabilities of the analog-digital (A/D) converter, which is on the order of 2 percent. The six-bit length of Pioneer words gives the least significant bit (LSB) a value of 1/64—an accuracy greater than that of the A/D converters. The exceptions to the universal adequacy of the six-bit word were the Goddard Space Flight Center fluxgate magnetometers, the Minnesota cosmic-ray instruments, and the Ames plasma probes. The Goddard experiment, for example, needed eight-bit words and fabricated them by combining adjacent pairs of standard-length Pioneer words.

Four special-status words carry no parity bits. These are: (1) the frame-sync word and its complement, (2) the word identifying the telemetry mode being used, (3) the extended-frame counter word, and (4) the spin-rate word.

Just as bits are organized into words, the words themselves are ordered into frames consisting of 32 words each. The frames keep repeating one after the other, but the arrangement of words can be modified by command. This separation of words by interspersing them in the time dimension is called time multiplexing. In effect, each scientific and engineering instrument gets read periodically and the data are strung together in the 32-word frames (fig. 4–8). The flexibility of the formats represents one of the strong points of the Pioneer system design.

There is a problem in terminology. As indicated earlier, four fundamental Pioneer telemetry formats exist. There are, however, five lists of scientific and engineering words that are used in making up the four different telemetry formats that are commandable from the Earth. Pioneer literature often refers to these five lists as formats A through E, implying five Pioneer formats, when only four exist. To avoid semantic confusion, the five lists will be called "lists" A through E in this book. The four *bona fide* formats A, B, C, and D are described as follows:

1* Frame synchronization FS	2 Format identification	3 Scientific subcommutator (16 words)	4
5	6	7	8
9	10	11	12
13	14	15	16
17* $\overline{FS}$	18 Subcommutator identification	19 Engineering subcommutator (64 words)	20
21	22	23	24
25	26	27	28
29	30	31	32

*Fixed Words

$\overline{FS}$ = complement of the frame synch word in position 1 (i.e., ones are replaced by zeros and vice versa).

FIGURE 4–8.—Pioneer main telemetry frame, 32 words long.

(1) Format A, identical to list A (tables 4–7 and 4–8), is used primarily at bit rates of 512 and 256 bits/sec when the spacecraft is close to the Earth.

(2) Format B, identical to list B (tables 4–7 and 4–8), is used primarily at bit rates of 64, 16, and 8 bits/sec when the spacecraft is far from the Earth.

(3) Format C, identical to list C, except that list C has 64 words rather

TABLE 4–7.—*Lists A, B, and D for Pioneers 6 and 7*[a]

Word	Scientific format A	Scientific format B	Special-purpose format D
1	Frame sync, 7 bits: 1110010	Frame sync, 7 bits: 1110010	Frame sync, 7 bits: 1110010
2	Format/mode identification	Format/mode identification	Format/mode identification
3	Scientific subcommutator (16 words)	Scientific subcommutator (16 words)	Scientific subcommutator (16 words)
4	Cosmic ray (Chicago)	Cosmic ray (Chicago)	
5 6 7 8	Magnetometer (Goddard)	Magnetometer (Goddard)	
9 10 11	Cosmic ray (Chicago)	Cosmic ray (Chicago)	Radio propagation (Stanford)
12	Cosmic ray (GRCSW)[b]		
13	Radio propagation (Stanford)		
14 15 16	Plasma (MIT)	Plasma (MIT)	
17	Frame sync complement 7 bits: 0001101	Frame sync complement 7 bits: 0001101	Frame sync complement 7 bits: 0001101
18	Subcom identification 6-bit counter	Subcom identification 6-bit counter	Subcom identification 6-bit counter
19	Engineering subcommutator (64 words)	Engineering subcommutator (64 words)	Engineering subcommutator (64 words)
20	Cosmic ray (Chicago)	Cosmic ray (Chicago)	
21			

TABLE 4-7—*Lists A, B, and D for Pioneers 6 and 7*—Concluded.[a]

Word	Scientific format A	Scientific format B	Special purpose format D
22		Magnetometer (Goddard)	
23			
24			
25			
26	Plasma (Ames)	Cosmic ray (GRCSW)	Radio propagation (Stanford)
27			
28			
29		Radio propagation (Stanford)	
30			
31		Plasma (Ames)	
32			

[a] Identical to formats A, B, and D mentioned in the text. See chapter 5 for experiment details.

[b] GRCSW = Graduate Research Center of the Southwest; later renamed Southwest Center for Advanced Studies (SCAS) and now known as The University of Texas at Dallas.

TABLE 4-8.—*Lists A, B, and D for Pioneers 8 and 9*[a]

Word	Scientific format A	Scientific format B	Special-purpose format D
001	Frame sync, 7 bits: 1110010	Frame sync, 7 bits: 1110010	Frame sync, 7 bits: 1110010
002	Format/mode identification	Format/mode identification	Format/mode identification
003	Scientific subcommutator (16 words)	Scientific subcommutator (16 words)	Scientific subcommutator (16 words)
004	Radio propagation (Stanford)	Radio propagation (Stanford)	
005			
006	Magnetometer	Magnetometer	
007	Goddard on Pioneer 8		
008	Ames on Pioneer 9		

TABLE 4–8—*Lists A, B, and D for Pioneers 8 and 9*—Concluded.[a]

Word	Scientific format A	Scientific format B	Special purpose format D
009 010 011 012 013 014 015 016	Plasma (Ames)	Plasma (Ames)	Radio Propagation (Stanford)
017	Frame sync complement 7 bits: 0001101	Frame sync complement 7 bits: 0001101	Frame sync complement 7 bits: 0001101
018	Subcom identification 6-bit counter	Subcom identification 6-bit counter	Subcom identification 6-bit counter
019	Engineering subcommutator (64 words)	Engineering subcommutator (64 words)	Engineering subcommutator (64 words)
020	Cosmic ray (Minnesota)	Cosmic ray (Minnesota)	Radio propagation (Stanford)
021 022 023 024	Magnetometer Goddard on Pioneer 8 Ames on Pioneer 9	Magnetometer	
025 026 027 028 029	Cosmic ray (SCAS)	Cosmic ray (SCAS)	
030 031 032	Cosmic ray (Minnesota)	Cosmic ray (Minnesota)	

[a] Identical to formats A, B, and D mentioned in the text.

than 32 and takes two frames (tables 4–9 and 4–10), consists mainly of engineering data and is used during special maneuvers (orientation) or when the spacecraft is in trouble.

(4) Format D, identical to list D (tables 4–7 and 4–8), consists of data from Stanford radio propagation experiment only, and is used during lunar occultations and other special events.

TABLE 4–9.—*List C: Subcommutated Engineering Measurements for Pioneers 6 and 7*

Word	Bit [a]	State 0	State 1	Identification [b]
[c] 201	1–7	No	Yes	Frame sync 1110010
[c] 202	8	No	Yes	Format A
	9	No	Yes	Format B
	10	No	Yes	Format C
	11	No	Yes	Format D
	12	No	Yes	Duty cycle store
	13	No	Yes	Telemetry store
	14	Yes	No	First 32 words of format C
203	15–20			
204	22–27			
205	29	No	Yes	Bit rate, 512 bps
	30	No	Yes	Bit rate, 256 bps
	31	No	Yes	Bit rate, 64 bps
	32	No	Yes	Bit rate, 16 bps
	33	No	Yes	Bit rate, 8 bps
	34	Yes	No	Interlock switch to orientation electronics
206	36	Off	On	Battery power
	37	----	----	Orientation pressure switch actuated
	38	Off	On	Orientation power
	39	Yes	No	Undervoltage protection in effect
	40	No	Yes	Voltage below switch-trip level
	41	B	A	DTU redundancy
207	43	2	1	Antennas to TWT number
	44	Low	High	TWT to gain of antenna
	45	Antenna	TWT	Driver to
	46	Driver	TWT	Low-gain antenna to
	47	Off	On	TWT 1 power
	48	Off	On	TWT 2 power
208	50	Off	On	Converter 1, +16 V
	51	Off	On	Converter 1, +10 V
	52	On	Off	Converter 1, −16 V
	53	Off	On	Converter 2, +16 V
	54	Off	On	Converter 2, +10 V
	55	On	Off	Converter 2, −16 V
209	57	No	Yes	Decoder 1 signal present
	58	No	Yes	Decoder 2 signal present
	59	Yes	No	Receiver 1 signal present
	60	Yes	No	Receiver 2 signal present
	61	Low	High	Receiver 2 to gain of antenna
	62	Off	On	Coherent mode
210	64	No	Yes	Ordnance system armed
	65	No	Yes	Third stage separated
	66	Yes	No	Boom 1 (orientation) deployed
	67	Yes	No	Boom 2 (magnetometer) deployed

TABLE 4–9.—*List C: Subcommutated Engineering Measurements for Pioneers 6 and 7* (Continued)

Word	Bit [a]	State 0	State 1	Identification [b]
	68	Yes	No	Boom 3 (wobble damper) deployed
	69	Yes	No	Stanford antenna deployed
211	71	On	Off	Experiment B power
	72	---------	-------	Not assigned
	73	---------	-------	Not assigned
	74	On	Off	Experiment C power
	75	No	Yes	Experiment C acquiring data
	76	2	1	Experiment C mode
212	78	On	Off	Experiment A power
	79	No	Yes	Telemetry store mode signal to experiment A
	80	---------	-------	Not assigned
	81	On	Off	Experiment G power
	82	---------	-------	Not assigned
	83	---------	-------	Not assigned
213	85	On	Off	Experiment D power, 28 V
	86	No	Yes	Experiment D calibrate mode
	87	Off	On	Experiment D dynamic range
	88	No	Yes	Experiment D data overflow
	89	Off	On	Experiment D power, 12 V
	90	On	Off	Experiment E power
214	92–97	---------	-------	Not assigned
215	99–105	---------	-------	Spacecraft spin, rev/64 sec
216	106–111	---------	-------	TWT 1 anode voltage
[c] 217	113–119	---------	-------	Frame sync complement 0001101
218	120–125	---------	-------	Receiver 1 static phase error
219	127–132	---------	-------	Receiver 2 static phase error
220	134–139	---------	-------	Receiver 1 signal strength
221	141–146	---------	-------	Receiver 2 signal strength
222	148–153	---------	-------	Receiver 1 and 2 temperature
223	155–160	---------	-------	TWT 1 helix current
224	162–167	---------	-------	TWT 1 cathode current
225	169–174	---------	-------	TWT 2 helix current
226	176–181	---------	-------	TWT 2 cathode current
227	183–188	---------	-------	TWT 1 temperature
228	190–195	---------	-------	TWT 2 temperature
229	197–202	---------	-------	TWT converter temperature
230	204–209	---------	-------	Driver temperature
231	211–216	---------	-------	DTU temperature
232	218–223	---------	-------	DSU temperature
[c] 233	225–231	---------	-------	Frame sync 1110010
[c] 234	232	No	Yes	Format A
	233	No	Yes	Format B
	234	No	Yes	Format C
	235	No	Yes	Format D
	236	No	Yes	Duty cycle store

TABLE 4–9.—*List C: Subcommutated Engineering Measurements for Pioneers 6 and 7* (Concluded)

Word	Bit [a]	State 0	State 1	Identification [b]
	237	No	Yes	Telemetry store
	238	No	Yes	Last 32 words of format C
235	239–244	---------	-------	Experiment A, D[4] voltage
236	246–251	---------	-------	Experiment A temperature
237	253–258	---------	-------	DTU A/D converter calibrate 1
238	260–265	---------	-------	DTU A/D converter calibrate 2
239	267–272	---------	-------	DTU A/D converter calibrate 1
240	274–279	---------	-------	Equipment converter +16-V bus
241	281–286	---------	-------	Equipment converter +10-V bus
242	288–293	---------	-------	Equipment converter −16-V bus
243	295–300	---------	-------	Equipment converter 1 and 2 temp
244	302–307	---------	-------	Bus voltage
245	309–314	---------	-------	Bus current
246	316–321	---------	-------	Battery temperature
247	323–328	---------	-------	Battery current
248	330–335	---------	-------	TWT 2 anode voltage
[c] 249	337–343	---------	-------	Frame sync complement 0001101
250	344–349	---------	-------	Forward solar panel temperature
251	351–356	---------	-------	Aft solar panel temperature
252	358–363	---------	-------	Platform temperature (no. 2)
253	365–370	---------	-------	Boom bracket temperature
254	372–377	---------	-------	High-gain antenna mounting bracket temperature
255	379–384	---------	-------	Louver actuator housing temperature
256	386–391	---------	-------	Sun sensor A temperature
257	393–398	---------	-------	Platform temperature (no. 1)
258	400–405	---------	-------	Nitrogen bottle pressure
259	407–412	---------	-------	Nitrogen bottle temperature
260	414–419	---------	-------	Not assigned
261	421–426	---------	-------	Not assigned
262	428–433	---------	-------	Sun sensor C temperature
263	435–440	---------	-------	Platform temperature (no. 3)
264	442–447	---------	-------	Experiment B temperature

[a] There are seven telemetry bits in each telemetry channel. Bits are numbered from 1 to 448 in the time order they are received from the spacecraft. Bit numbers missing in sequence refer to bits used to indicate parity (odd) for the first, third, and fifth bits. For analog and digital words, the first bit received is the MSB and assumes the largest weighted value in a word.

[b] Pioneer-6 experiments are identified by letters as follows:

A = Chicago cosmic-ray experiment
B = Goddard magnetometer
C = MIT plasma experiment
D = GRCSW cosmic-ray experiment
E = Stanford University radio propagation experiment
G = Ames plasma experiment

[c] Word not assigned when subcommutated.

TABLE 4–10.—*List C; Subcommutated Engineering Measurements for Pioneers 8 and 9*

Word	Measurement[a]	
001	Frame sync, 7 bits: 1110010	
201	Exp H frequency count	
002	Format/mode identification	
	Binary word	Indication
	001000X	RT[b]
	001001X	MRO from TS[b]
	101010X	MRO from DCS with format A[b]
	011010X	MRO from DCS with format B[b]
202	Exp B overscale indicator	
Y03	Exp A internal temperature	
Y04	Digital	
	Bit 1	Exp A power not on
	Bit 2	Exp A not in flare mode
	Bit 3	Exp A not in flare mode sector
	Bit 4	Exp A detector B_2 not suppressed
	Bit 5	Exp A detector B_3 not suppressed
	Bit 6	Exp A detector D not suppressed
	Bit 7	Parity bit
Y05	Digital	
	Bit 1	512 bps
	Bit 2	256 bps
	Bit 3	64 bps
	Bit 4	16 bps
	Bit 5	8 bps
	Bit 6	Orientation power on and spacecraft not separated
	Bit 7	Parity bit
Y06	Digital	
	Bit 1	Battery on
	Bit 2	Change indicates orientation pulse
	Bit 3	Orientation power on and spacecraft separated
	Bit 4	Undervoltage protection off
	Bit 5	CCU power on
	Bit 6	DTU redundancy A on
	Bit 7	Parity bit
Y07	Digital	
	Bit 1	TWT 1 to antenna (S4–1)
	Bit 2	TWT to high-gain antenna (S2–1)
	Bit 3	Driver toTWT (S5–2)
	Bit 4	TWT to low-gain antenna (S3–1)
	Bit 5	TWT 1 power on
	Bit 6	TWT 2 power on
	Bit 7	Parity bit
Y08	Digital	
	Bit 1	Equip. conv. 1, +16 V on
	Bit 2	Equip. conv. 1, +10 V on
	Bit 3	Equip. conv. 1, −16 V not on
	Bit 4	Equip. conv. 2, +16 V on
	Bit 5	Equip. conv. 2, +10 V on
	Bit 6	Equip. conv. 2, −16 V not on
	Bit 7	Parity bit
Y09	Digital	
	Bit 1	Decoder 1 signal present
	Bit 2	Decoder 2 signal present

TABLE 4–10.—*List C: Subcommutated Engineering Measurements for Pioneers 8 and 9* (Continued)

Word	Measurement [a]	
Y09	Digital—Continu d	
	Bit 3	Receiver 1 signal not present
	Bit 4	Receiver 2 signal not present
	Bit 5	Receiver 2 to high-gain antenna (S1–2)
	Bit 6	Coherent mode enabled
	Bit 7	Parity bit
Y10	Digital	
	Bit 1	Ordnance system armed
	Bit 2	Spacecraft separated from third stage
	Bit 3	Boom 1 not deployed (orientation)
	Bit 4	Boom 2 not deployed (magnetometer)
	Bit 5	Boom 3 not deployed (wobble damper)
	Bit 6	Stanford antenna not deployed
	Bit 7	Parity bit
Y11	Digital	
	Bit 1	Exp B power not on
	Bit 2	Exp B calibrate on
	Bit 3	Exp B channel switch flag (0 = normal, 1 = flipped)
	Bit 4 / Bit 4 / Bit 6	Exp. B Filter Freq: 001 = 512 bps; 010 = 256 bps; 011 = 64 bps; 100 = 16 bps; 101 = 8 bps
	Bit 7	Parity bit
Y12	Digital	
	Bit 1	Exp F power not on
	Bit 2	Telemetry store mode on
Y12	Digital—Continued	
	Bit 3	Exp H power not on
	Bit 4	Exp G power not on
	Bit 5	Exp B sensor position indicator (0 = normal, 1 = flip command)
	Bit 6	Exp B change indicates flip command verification (0 = normal, 1 = flip command)
	Bit 7	Parity bit
Y13	Digital	
	Bit 1	Exp D power not on
	Bit 2	Exp D calibrate on
	Bit 3	Exp D low power mode on
	Bit 4	Exp D slip mode on
	Bit 5	Exp D aspect clock free running
	Bit 6	Exp E power not on
	Bit 7	Parity bit
Y14	Digital	
	Bits 1–4	Exp D measurements counter number
	Bits 5–6	Exp D supercommutation number
Y15	Digital	
	Bits 1–7	Spin rate counter
Y16	TWT 1 anode voltage	
017	Frame sync complement, 7 bits: 0001101	
217	Exp H frequency count	
Y18	Receiver 1 loop stress	
Y19	Receiver 2 loop stress	
Y20	Receiver 1 signal strength	
Y21	Receiver 2 signal strength	
Y22	Receiver 1 temperature	
Y23	TWT 1 helix current	

TABLE 4–10.—*List C: Subcommutated Engineering Measurements for Pioneers 8 and 9* (Concluded)

Word	Measurement [a]	Word	Measurement [a]
Y24	TWT 1 cathode current	Y44	Primary bus voltage
Y25	TWT 2 helix current	Y45	Primary bus current
Y26	TWT 2 cathode current	Y46	Battery temperature
Y27	TWT 1 temperature	Y47	Battery current
Y28	TWT 2 temperature	Y48	TWT 2 anode voltage
Y29	TWT converter temperature	049	Frame sync complement, 7 bits: 0001101
Y30	Transmitter driver temperature	249	Exp H freq count
Y31	DTU temperature	Y50	Solar panel 1 (upper) temperature
Y32	DSU temperature	Y51	Solar panel 2 (lower) temperature
033	Frame sync, 7 bits: 1110010	Y52	Mounting platform 2 temperature
233	Exp H frequency count	Y53	Exp H ramp-generator voltage level
034	Format/mode identification Binary word — Indication 001000X — RT [b] 001001X — MRO from TS [b] 101010X — MRO from DCS with format A [b] 011010X — MRO from DCS with format B [b]	Y54	Antenna mtg bracket (high gain) temperature
234	Exp D detector B temperature	Y55	Louver actuator housing temperature
Y35	Exp E 49-Hz signal amplitude	Y56	Sun sensor "A" temperature
Y36	Receiver 2 temperature	Y57	Mounting platform 1 temperature
Y37	DTU inflight calibrate 1	Y58	Nitrogen bottle pressure
Y38	DTU inflight calibrate 2	Y59	Nitrogen bottle temperature
Y39	DTU inflight calibrate 3	Y60	Exp G electronics temperature
Y40	Equip. conv, +16 V bus	Y61	Exp B sensor temperature
Y41	Equip. conv, +10 V bus	Y62	Sun sensor "C" temperature
Y42	Equip. conv, −16 V bus	Y63	Mounting platform 3 temperature
Y43	Equip. conv 1 and 2 temperature	Y64	Not used (ground)

[a] Pioneer-9 experiments are identified by letters as follows:
A = Minnesota cosmic ray
B = Ames magnetometer
D = SCAS cosmic ray
E = Stanford radio propagation
F = Goddard cosmic dust
G = Ames plasma
H = TRW electric field

[b] Telemetry modes are as follows:
RT = Real time

Formats A, B, and D each possess spaces for two subcommutated words. Words from the 64-word list C of engineering data are repeated one after the other, always in position 19, in successive frames of formats A, B, and D. When subcommutated in these formats, list C repeats every 64 frames compared to its repetition every two frames when format C is selected. This option permits the mission controllers to emphasize or de-emphasize engineering data as the situation requires.

List E (or format E) is always subcommutated in position 3 of formats A, B, and D. List E is only 16 words long (table 4–11) and consists primarily of lower-priority scientific data.

During the launch and reorientation maneuver, the spacecraft normally transmitted format C. While the spacecraft was still near the Earth, format A was usually employed. As the spacecraft receded from Earth, format B was adopted. If the trajectory of a Pioneer was favorable for lunar occultation, a command from Earth switched to format D. Out in the relatively calm reaches of deep space, the spacecraft transmits format B most of the time.

Four Modes of Operation

Although variable bit-rate and telemetry format confer considerable flexibility, provision is needed for storing and thus delaying data transmission back to Earth. An important solar event could occur when one or more of the Pioneers is too far away to telemeter plasma-probe data rapidly enough to catch the details of the fast-breaking action. It would be like trying to make a movie of a high jumper with a movie camera taking only a frame or two per second; many details would be missed. The initial STL studies recognized the advantages of a small memory device in such situations. Data could be recorded at a high rate during the event and then retransmitted later at a bit rate compatible with the spacecraft's transmitter power and distance from the Earth.

The illustration above justifies three of the four Pioneer telemetry modes: (1) real-time operation, (2) telemetry store, and (3) memory readout. The fourth mode, the duty-cycle store mode, simply stores data in the memory when the spacecraft is not being worked by a DSN station.

MRO = Memory readout
TS = Telemetry store
DCS = Duty-cycle store

Notes:

For bit 7 of format/mode ID, X = 0 (zero) for the first 32 words of engineering subcommutator and X = 1 (one) for the last 32 words.

Word numbering system: 017 indicates main frame word 17.
217 indicates engr subcom word 17.
Y18 indicates can be either 018 or 218.

Statements indicated are for the true (one) state on digital words Y04 through Y13.

TABLE 4–11.—*Formats E (Lists E) for Pioneer Spacecraft*[a]

Pioneers 6 and 7			Pioneers 8 and 9		
Word	Type	Identification	Word	Type	Identification
101	Digital	Cosmic ray (Chicago)	101	Digital	Cosmic ray (Minnesota)
102	Analog	Radio propagation (Stanford)	102	Analog	Radio propagation (Stanford)
103	Digital	Unassigned	103	Digital	Cosmic dust (Goddard)
104	Digital		104	Digital	
105	Digital		105	Digital	
106	Digital		106	Digital	
107	Analog	Radio propagation (Stanford)	107	Analog	Radio propagation (Stanford)
108	Analog	Plasma (Ames)	108	Analog	Electric field (Stanford/TRW)
109	Digital	Cosmic ray (Chicago)	109	Digital	Cosmic ray (Minnesota)
110	Digital	Plasma (MIT)	110	Digital	
111	Analog	Radio propagation (Stanford)	111	Analog	Radio propagation (Stanford)
112	Analog	Plasma (Ames)	112	Analog	Electric field (Stanford/TRW)
113	Digital	Magnetometer (Goddard)	113	Digital	Cosmic ray (Minnesota)
114	Digital		114	Digital	
115	Analog	Radio propagation (Stanford)	115	Analog	Radio propagation (Stanford)
116	Digital	Bit rate code and extended frame count	116	Digital	Bit rate code and extended frame count

[a] Also called the "scientific subcommutator formats."

Any one of the four modes can be started with a specific command from Earth. Regardless of when the spacecraft receives the command, actual execution is delayed until the beginning of the next 32-word frame. Switching modes by command has its advantages, but if the data handling subsystem got stuck ("hung up") in the memory readout mode, when the 15 232 bits, (68 frames) worth of information in the memory came to an end, so would the mission. Consequently, all spacecraft modes automatically revert to the real-time mode whenever the DSU is filled or emptied, because the real-time mode is the most useful of the four.

There are some data-handling subtleties and restrictions that are best explained with a table (see table 4–12).

TABLE 4–12.—*More Details on Pioneer Telemetry Modes*

Mode	Characteristics
Real-time mode	This mode transmits any one of the four commandable formats continuously at any selected bit rate except for format D. When format D is selected, only 68 frames are transmitted before the subsystem automatically reverts to format B at 16 bits/sec. This precludes getting hung up on the Stanford radio propagation experiment. The 68 frames are stored in the memory as they are transmitted. This mode is employed during lunar occultation and other special scientific events.
Telemetry-store mode	The DSU is filled to 68-frame capacity (32-word frames) with format A, B, or C, whichever is commanded. The data are also transmitted in real time. When the DSU is filled, the subsystem reverts to format B at 16 bits/sec. This mode is useful for sampling data faster than real-time transmission permits.
Duty-cycle mode	Frames in format A, B, or (rarely) C are filled at the 512 bits/sec rate and are stored in the DSU, one at a time, at one of four commandable rates given below: Interval between frames, min — Time to fill DSU, hr 17 — 19 8.5 — 9.5 4.25 — 4.75 2.125 — 2.3 In other words, frames of data can be selected and stored for up to 19 hours. This mode is used primarily when the spacecraft is not being worked by the DSN. Following each eight frames of scientific data, format C (list C) is also stored. Data are also transmitted in real time but are, of course, not recoverable if the DSN is not working the spacecraft. When the DSU is filled, the subsystem reverts to format B at 16 bits/sec.
Memory-readout mode	The contents of the DSU are transmitted at any selected bit rate. Data are read out only once because the process destroys the memory contents. Unfilled portions of the memory appear as is. At the end of memory readout, the subsystem reverts to the format and bit rate in use prior to the receipt of the readout command.

In summary, the variable bit rate, variable format, and variable mode permit the mission controller to tune the spacecraft and supporting systems to changing scientific requirements, emergency situations aboard the spacecraft, and the lengthening distance between the Earth and its automated outpost far out along the plane of the ecliptic.

The Digital Telemetry Unit

The digital telemetry unit is not only the central clearing house for all spacecraft-generated data, it is also the spacecraft timer or synchronizer that keeps all spacecraft components operating in step. To do this and impose order upon the varied data requires a rather complex array of logic circuits, counters, and A/D converters (fig. 4–9).

The timing function is performed by a crystal-controlled-oscillator clock producing a 16 384-Hz output signal. This signal is then divided by 32, 64, 256, 1024, and 2048 to establish the five standard bit rates. (Note that these numbers are all powers of 2.) Armed with timing signals, the multiplexers and submultiplexers sample the various analog and digital outputs of the scientific and engineering instruments. All instruments are usually on at all times; and the only stimulus needed to make them provide a reading is an electronic "gate." (An exception is the Stanford radio propagation experiment, which is usually turned off at great ranges.) The multiplexers simply open and close gates leading to the instruments in the order specified by the last command from Earth. Electronic switches or gates are the mainstays of computers and other logic circuits. It is the spacecraft clock, of course, that ultimately drives all subsystem circuits.

Many instruments deliver digital information (bits) when the multiplexers open the gates. Others, particularly the engineering instruments, yield analog data. Such analog signals, really voltage levels between 0 and +3 V, must be converted into digital signals. This task falls to the A/D converter, another basic type of circuit in the data-handling subsystem.

Figure 4–9 also indicates that signals from the Sun sensors are counted in the DTU to establish a spin rate. Gas pulses from the orientation system are indicated by a one-bit toggle which switches between 0 and 1 or 1 and 0 for each pulse. Spin rate and orientation pulse data are dubbed into the telemetry stream as engineering data. The Sun pulses are sent to some experiments in order to turn them on to record data at certain azimuths, notably when pointed in the direction of the Sun.

All DTU components were essentially off-the-shelf, having been developed and qualified in commercial and military programs.

The Data Storage Unit

The memory of the DSU is not large by terrestrial standards—only 15 232 bits—but this is sufficient for Pioneer's purposes in view of the very low data rates possible for transmission back to Earth. It takes over half an hour to read out a 15 232-bit memory at 8 bits/sec. Earth satellites, which generally have much larger memories, commonly carry tape recorders to store data until the memory can be read out over the next data-acquisition station. Tape recorders possess a high storage capacity but do not have the requisite reliability for a Pioneer mission. A solid-state mem-

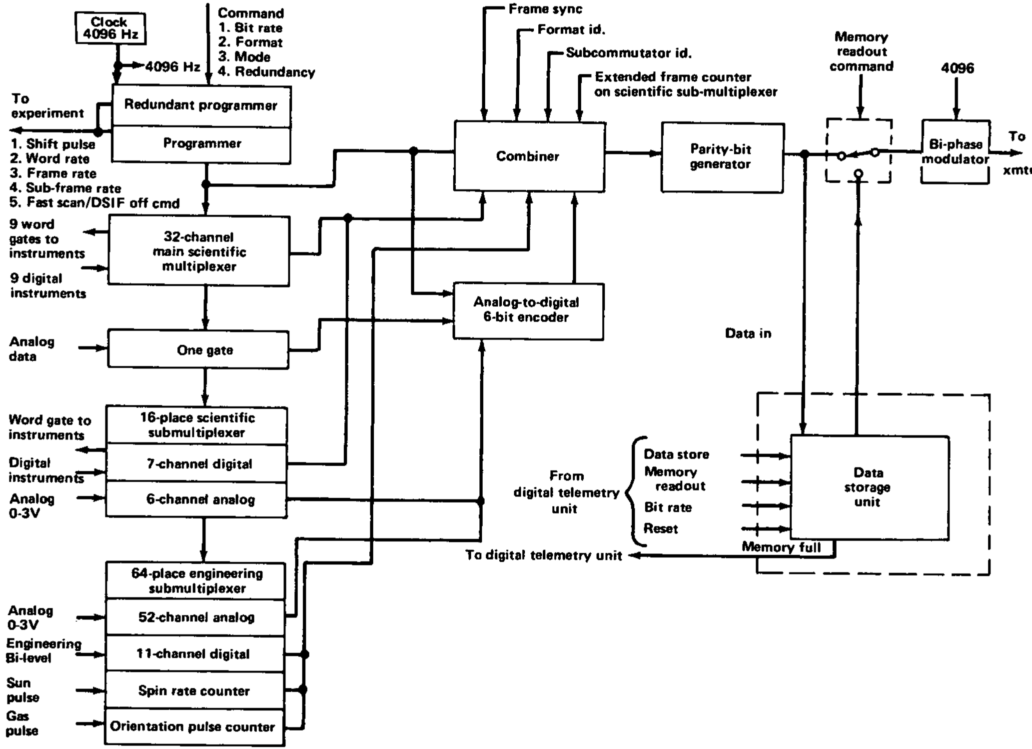

FIGURE 4–9.—Simplified block diagram of the digital telemetry unit.

ory, employing magnetic cores similar to those found in digital computers, was selected (fig. 4–10). The STL feasibility study and proposal had been based upon non-destructive memory readout, where each bit is reinserted into the memory immediately after being read out. But the power and weight penalties of this approach were too high, and destructive readout was chosen. The inevitable engineering tradeoff in this instance was that the DSN must be in downlink lock with the spacecraft and ready to accept data before the memory readout command can be transmitted. Memory readout can be terminated at any time by commanding a switch to the real-time mode, but the remaining data in storage are destroyed. In fact, if the data in the memory are not desired, the memory readout command followed immediately by a real-time mode command are collectively equivalent to a "memory clear" command. However, the real-time mode command can interrupt the telemetry-store mode, without the stored data's being destroyed. The DSU was built by Electronic Memories, Inc.

The Convolutional Coder

Adding the parity bit in the standard Pioneer telemetry code is worth 2 dB of added gain; the parity bit enables the DSN to gather good data from greater distances than otherwise possible. The price paid for reducing the error rate, however, is additional spacecraft circuitry and the transmission time taken by the parity bits.

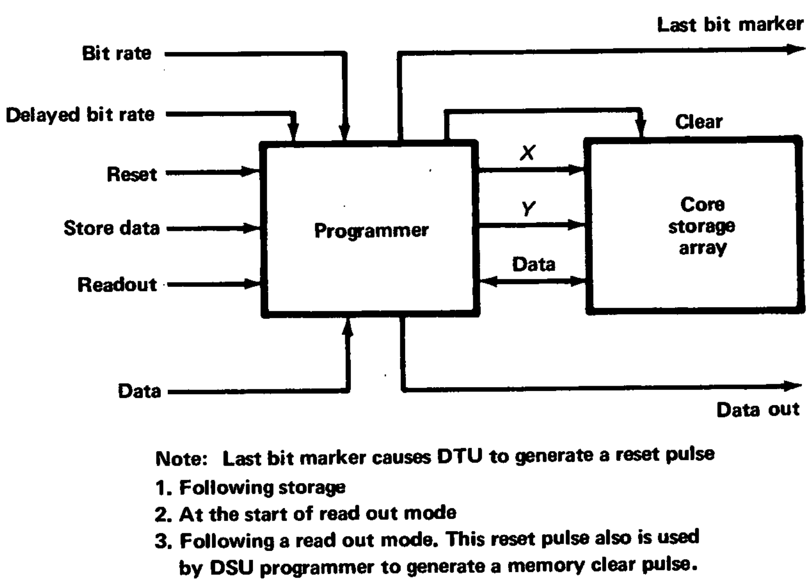

FIGURE 4–10.—Simplified block diagram of the data storage unit.

The CCUs installed on Pioneers 9 and E, through more elaborate coding, reduce telemetry error rates by an amount equivalent to about 3 dB in overall system gain, perhaps as much as 3.9 dB (ref. 2). The price paid is 1.3 lb in the weight of the coder, plus the 1.3 W of power it draws when operating in-line, plus the bits added to the telemetry stream. With the Pioneer convolutional coder, every bit of telemetry information is matched by an extra bit from the coder, which in effect carries "information about information." The doubled bit stream created by the convolutional coder represents redundancy, which increases the accuracy of telemetry communication from a distant spacecraft.

The parity bit in the standard Pioneer code is computed from the three odd-numbered bits in the preceding telemetry word. If the sum of these three bits is odd, the parity bit is also odd, i.e., one. The parity bits in the convolutional coder are computed by directing the bit stream from the DTU into a 25-bit register. Each bit leaving the DTU moves into position 1 in the register, shifting the 25th bit out the other end. The bits in positions 1, 2, 4, 6, 8, 9, 12, 14, 15, 16, 20, 21, 22, 24, and 25 are then added; if the sum is odd, the parity bit associated with the first bit is also odd, i.e., one.[8] After the telemetry bit and its newly computed parity bit have been sent on, the entire group of bits moves up a position and a new one is added from the DTU. The process is repeated for each telemetry bit in the 7-bit Pioneer word, including the regular parity bit, so that the CCU gain of 3 dB is added to the 2 dB picked up from the normal parity bit. Each parity bit thus contains some intelligence regarding 15 bits in the 25-bit register (fig. 4–11). The register is reset to zero once each 32-word frame so that decoding can be done on a frame-for-frame basis.

The telemetry bit and its companion parity bit are phase-modulated onto the square wave subcarrier and sent on to the transmitter driver for relay to the Earth. On Pioneers 9 and E, the only spacecraft carrying the convolutional coder, the NRZ–L method of modulation was employed rather than the NRZ–M approach used in the earlier Pioneers.

Figure 4–7 illustrates how the convolutional coder was installed in the Pioneers 9 and E data handling subsystems on an experimental basis. It can be switched in or out, being an off-line element like the data storage unit. Flight experience on Pioneer 9 has been good. The 3-dB coding gain extended the maximum communication range of Pioneer 9 about 40 percent.

THE COMMAND SUBSYSTEM

None of the flexibility and reliability gained through alternate modes of operation and redundancy can be realized without switches commandable from the Earth. To substitute a new TWT for one that falters,

[8] Technically, this parity bit is the "modulo-2 sum" of the 15 bits indicated.

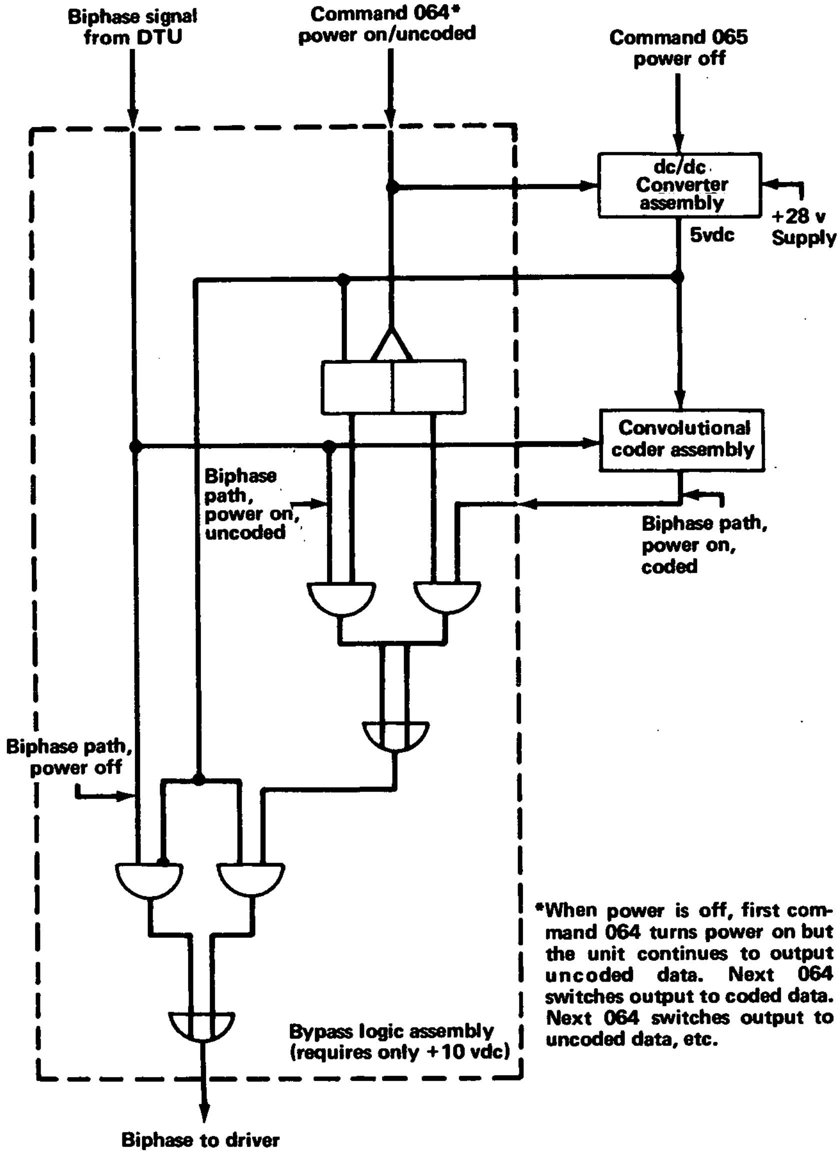

FIGURE 4–11.—Block diagram of the Pioneer convolutional coder unit.

or to change bit rate, the mission controller dispatches a command to the spacecraft directing a specific switch to open or close. The Pioneers employ between 57 and 67 commands (each spacecraft is slightly different) to activate the same numbers of spacecraft switches. About two-thirds of the commands pertain to spacecraft functions and the rest to experiments. The number of Pioneer on-off switches corresponds roughly to the number of electric switches in a modern home, considering all the appliances. Thus while Pioneer has been presented as a relatively simple spacecraft in the preceding pages, it has almost 2^{57} different operating permutations and can hardly be called primitive.[9]

Let us say that the mission controller at Ames Research Center wishes to change Pioneer 6's bit rate from 16 bits/sec to 8 bits/sec because the spacecraft is too far from Earth for the higher bit rate to be received without an excessive error rate. He constructs a 23-bit command word that is sent through JPL along NASA Communications Network (NASCOM) lines to the DSN station working Pioneer 6. The command is modulated onto the uplink carrier by frequency-shift keying. If a digital one is to be sent, a 240-Hz tone is phase-modulated on the DSN carrier; a 150-Hz tone represents a digital zero. The bit stream representing the command is thus a series of 23 beeps (in two pitches) on the DSN carrier.

The spacecraft communication subsystem possesses two frequency-addressable receivers; the carrier frequency selects the receiver once the PCM/PM/FSK signal reaches the spacecraft. The addressed spacecraft receiver demodulates the incoming signal and passes the series of tones on to two decoders. The command carries an address specifying only one of the redundant decoders; that decoder converts the tones into the 23-bit command and stores it in registers.

After checking the command for errors, the addressed decoder sends the command to the command distribution unit (CDU). The CDU selects the wire leading to the proper electronic switch, and the command is executed once the switch is thrown. If the switch is already in the commanded position no switching is changed.

Command Format

The standard telemetry word is seven bits; the Pioneer command's 23-bit word is much longer. If only the command number were sent, seven bits would be sufficient. Pioneer 9, which used the most commands (67), barely needed seven bits. As figure 4–12 indicates, the basic Pioneer command number was actually seven bits long. Preceding the seven-bit segment, however, was a seven-bit complement of the command, in which the ones in the command number were replaced by zeros and vice versa. It is common spacecraft practice to promote high command accuracy by sending a considerable amount of redundant information. The consequences of a garbled command are too serious to settle for simple parity checks or even the more elaborate coding adopted in the convolutional coder.[10] While the

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[9] Not all commands are mutually exclusive, so that 2^{57} is indicative only.

[10] In some satellites, where transmission times are negligible, the spacecraft repeats the command it has received to the tracking station before executing it.

2

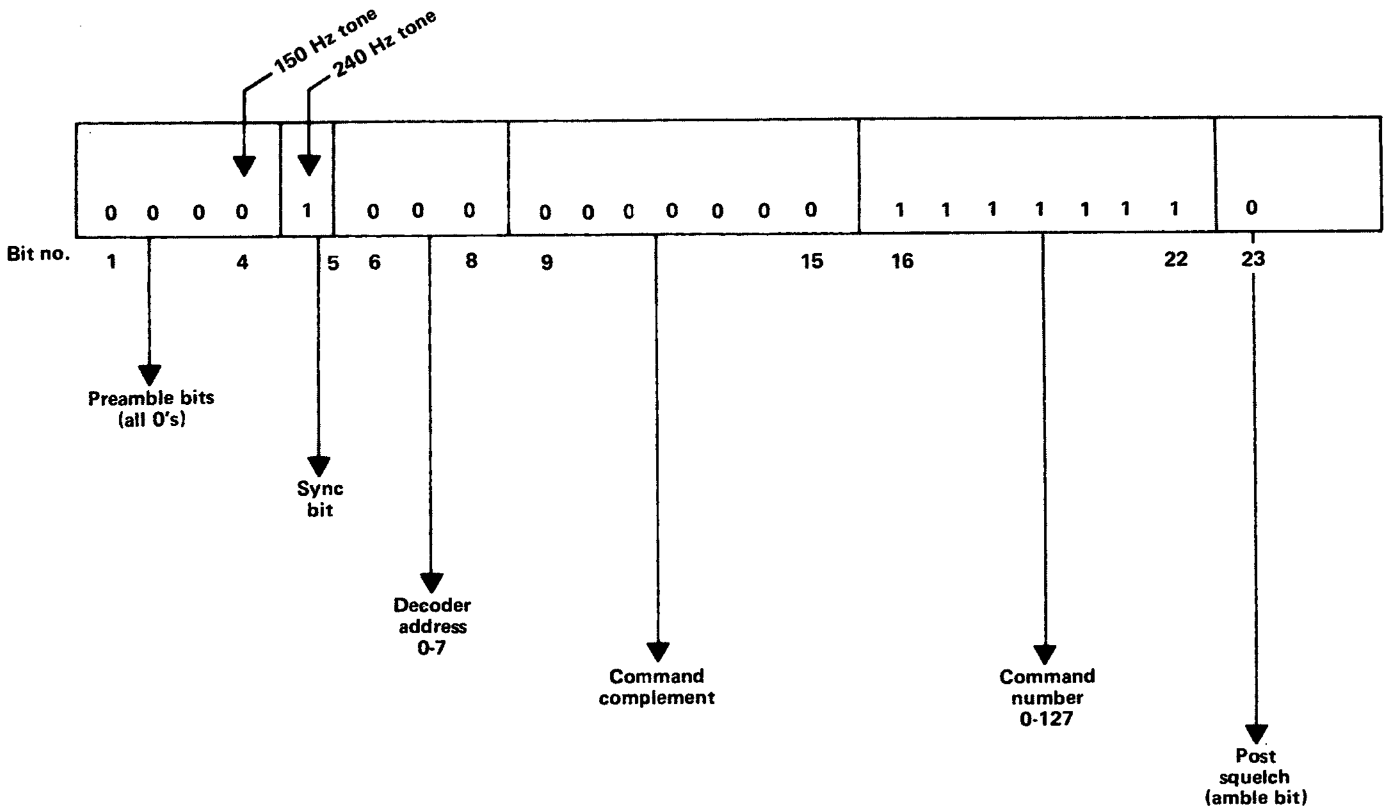

FIGURE 4-12.—Structure of the Pioneer command word.

23-bit command is in the decoder register, it is compared bit-by-bit with its complement. Complete correspondence is required before the command is released for execution. Incomplete or distorted commands are not executed. Loss of receiver lock also inhibits command execution.

The command number and its complement are preceded by the address that selects: (1) a specific Pioneer spacecraft, and (2) one of the two decoders on that spacecraft. Preceding the address is a sync bit and a series of four zeros at the beginning of the command word. The zeros, called a "preamble," aid command acquisition by the spacecraft decoder—in a sense, the zeros tell the decoder to prepare to receive a command. A post-squelch bit (or "amble" bit) follows at the end of the command word. The amble bit is always zero, and signals the end of the command. Physically, the function of the amble bit is to keep the decoder in operation; that is, it keeps the appropriate gates open until the command has been executed.

Command tones are modulated on the DSN carrier at the rate of only 1 bit/sec. It takes 27 sec to receive and execute a command aboard the spacecraft; this includes the time required for processing the command in the decoder and executing it. Pioneers are generally several light minutes away from Earth and are always "out of touch" to some degree, regardless of the low command rate.

The command numbers and their functions are listed in tables 4–13 and 4–14.

The Command Decoder and Command Distribution Unit

The assigned task of the decoder is the delivery of a verified bit train to the CDU. The decoder block diagram shown in figure 4–13 illustrates how the incoming series of tones is detected by a filter-detector circuit. Once the filters sort out the tones by frequency and turn them into pulses, the bits move into the shift register described earlier. After checking the complement, the decoder transmits a series of pulses to a diode matrix that makes up the gating circuits within the CDU. The diode matrix sends an execute signal to the proper address within the spacecraft (fig. 4–14). Four different kinds of signals flow out of the CDU, each tailored for triggering a specific action—the end result of the command transmitted from Earth:

(1) Most command pulses are short (10 μsec), low current (about 10 mA), at 10 V. These signals are sufficient to drive most Pioneer electronic circuits.

(2) Some devices, such as the coaxial switches, require somewhat longer pulses; the CDU provides a 160-msec, 28-V pulse for such devices.

(3) Where solid-state switches are inadequate because of the high currents involved, as in the case of the battery switch, the CDU activates relays.

TABLE 4–13.—*Command List for Pioneers 6 and 7, Grouped by Function*

Command number	End function	Command number	End function
	Communications		Telemetry (Continued)
001	TWT 1 on	037	Format C
003	Receiver 2 to low-gain antenna	044	DTU redundancy A
010	TWTs off	050	Format D
011	TWT 2 on	051	Telemetry store
015	TWT 1 to antenna/driver to TWT	052	Memory readout
022	TWT 2 to antenna/driver to TWT	053	Duty-cycle store
025	TWTs to low-gain antenna	060	Real time
030	Coherent mode enabled		Experiments
033	Receiver 2 to high-gain antenna	020	All experiments off
043	Non-coherent mode enabled		Chicago cosmic ray
046	Driver to low-gain antenna	063	Calibrate
047	TWTs to high-gain antenna	070	Normal mode
	Electrical	076	Power on
000	Undervoltage simulate		Goddard magnetometer
017	Battery on	055	Power on
036	Battery off	061	Calibrate
107	Undervoltage protection off	062	Flip sensor
110	Undervoltage protection on		MIT plasma
	Ordnance	013	Power on
045	Boom deploment (backup)	111	Mode change no. 1
	Orientation	112	Mode change no. 2
021	Type-I restart		GRCSW cosmic ray
031	Type-II clockwise	073	Dynamic range on
040	Type-II counterclockwise	100	Dynamic range off
041	Power on	101	Calibrate
042	Power off	116	Power on
	Telemetry		Stanford radio propagation
004	512 bit rate	071	Calibrate
005	256 bit rate	077	Power on
006	64 bit rate		Ames plasma
016	16 bit rate	054	Power on
024	DTU redundancy B	072	Calibrate
027	8-bit rate	113	Mode change
034	Format A		Spacecraft commands: 38
035	Format B		Experiment commands: 19
			57

TABLE 4–14.—*Command List for Pioneers 6 and 7, Grouped by Function*

Command number	End function
	Electrical
000	Undervoltage simulate
017	Battery on
036	Battery off
107	Undervoltage protection off
110	Undervoltage protection on
	Ordnance
045	Boom deployment (backup)
	Orientation
021	Type-I restart
031	Type-II clockwise
040	Type-II counterclockwise
041	Power on
	Telemetry
004	512 bit rate
005	256 bit rate
006	64 bit rate
016	16 bit rate
024	DTU redundancy B
027	8 bit rate
034	Format A
035	Format B
037	Format C
044	DTU redundancy A
050	Format D
051	Telemetry store
052	Memory readout
053	Duty-cycle store
060	Real time
[a] 064	CCU power on uncoded
[a] 065	CCU power off
	Communications
001	TWT 1 on
003	Receiver 2 to low-gain antenna
010	TWTs off
011	TWT 2 on
015	TWT 1 to antenna/driver to TWT
022	TWT 2 to antenna/driver to TWT
025	TWTs to low-gain antenna
030	Coherent mode enabled
033	Receiver 2 to high-gain antenna
043	Noncoherent mode enabled
046	Driver to low-gain antenna
047	TWTs to high-gain antenna
	Experiments
020	All experiments off
	Minnesota cosmic ray
076	Power on
102	Arm
103	Code
104	Flare mode
105	Sector flare mode
111	Execute
114	Disable detector D
115	Select telescope T1
	Ames magnetometer
013	Power on
061	Calibrate and flip (if enabled)
062	Flip enable
063	Spin demodulator select
070	Bandwidth change
	SCAS cosmic ray
073	High power mode on
101	Calibrate
113	Low power mode on
116	Power on
	Stanford radio propagation
077	Power on
071	Calibrate
	Goddard cosmic dust
055	Power on
112	Calibrate
	Ames plasma probe
054	Power on
072	Calibrate/sector delay mode select
[a] 074	Energy range select
100	Suppression mode change
	TRW electric field detector
106	Power on

Pioneer	8	9
Spacecraft commands	38	38
Experiment commands	26	29
	—	—
	64	67

[a] Pioneer 9 only.

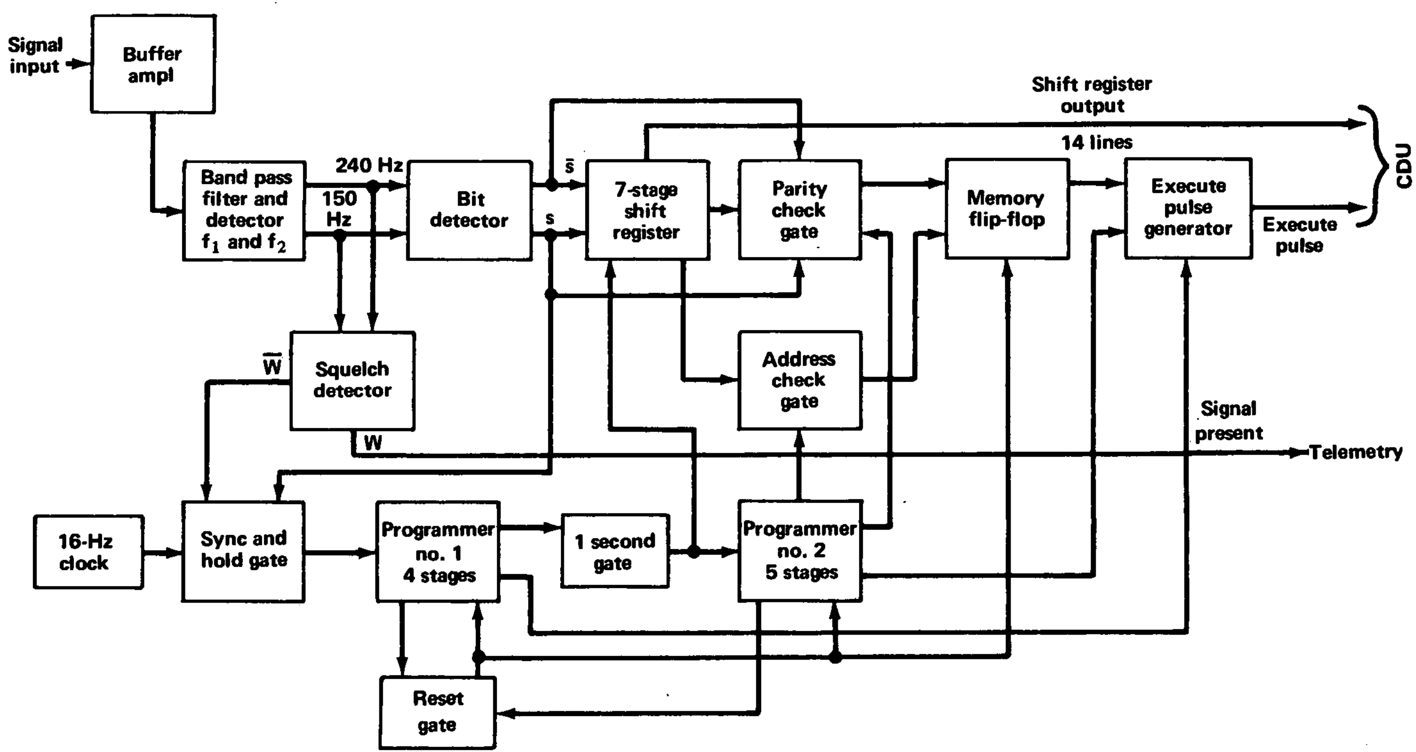

FIGURE 4–13.—Block diagram of the Pioneer command decoder.

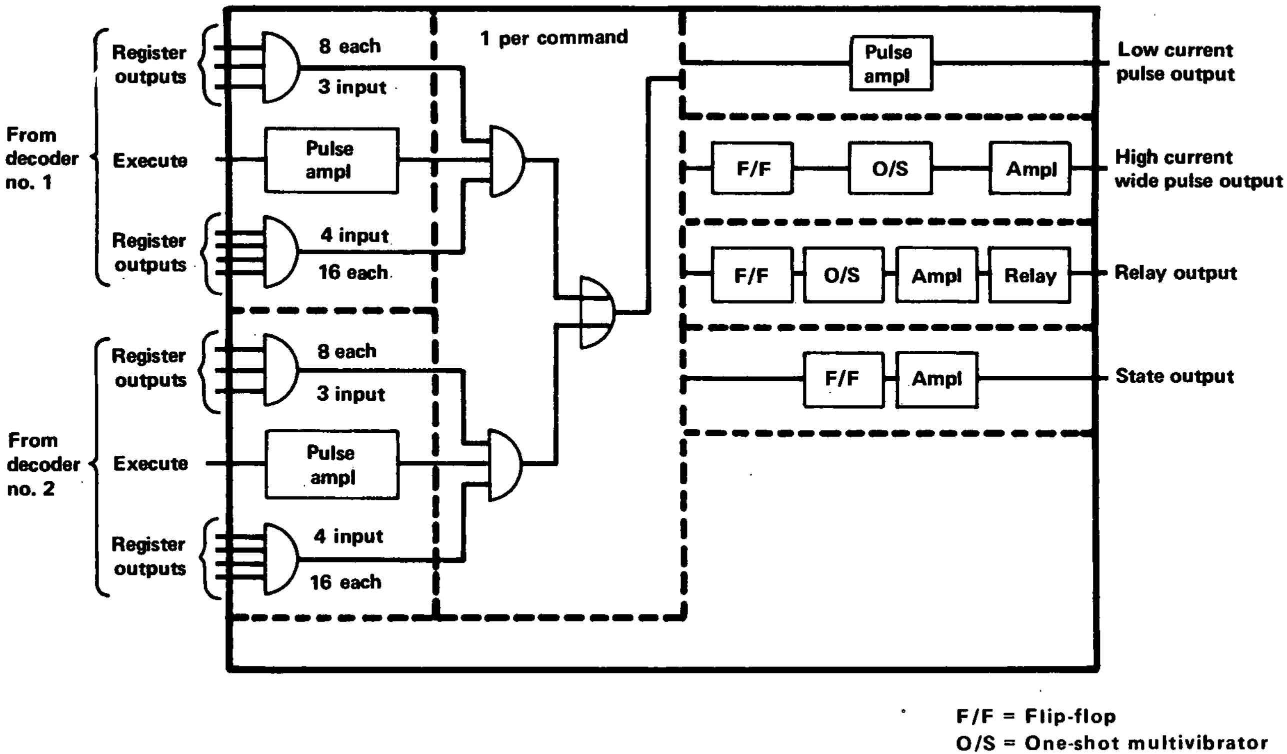

FIGURE 4–14.—Functional diagram of the command distribution unit.

(4) A "state" output (one of two voltage levels) is available for instrumentation. On the Pioneers, state commands were simply "voltage on" or "voltage off" commands.

Like the designs of much of Pioneer's electronic hardware, the command-decoder and CDU designs were derived from STL experience with Air Force aerospace programs. Solid-state components were employed throughout the command subsystem. In fabricating command subsystem hardware, STL employed welded modules, a reliable technique that has also proven to be very efficient in volume utilization.

THE ELECTRIC POWER SUBSYSTEM

Once it leaves the Earth far behind, the Pioneer spacecraft is in full sunlight. The spacecraft can then convert solar energy into electricity to operate its scientific instruments and also to drive the subsystems that enable the vehicle to survive in outer space and maintain a communication link with the Earth. Without power, there can be no deep space mission. Only the conversion of solar energy into electromagnetic waves of a specific frequency makes the Pioneer stand out against the background of stars, planets, and other radio emitters on the celestial sphere.

The power picture is more complicated, however. A basic program ground rule states that the spacecraft must be flexible enough to operate between 0.8 and 1.2 AU without modification. Also, for purposes of acquisition, the spacecraft must be operable prior to escaping the Earth and breaking into full perpetual sunlight. The Pioneer shadow problem is a one-time affair, not repeating every few hours like that of an Earth satellite. Yet, the problem can be solved in the same way—with a battery serving as a reservoir of energy. In a satellite the battery is discharged and charged through several cycles each day; but with Pioneer, the battery becomes largely excess baggage once the Earth's shadow is traversed. Even in full sunlight, however, the spacecraft depends upon the battery for an assist in meeting sudden, brief surges in power demands during normal operation, due in particular to pneumatic valve pulses and, on Pioneers 6 and 7, the MIT experiment (fig. 4–15). The solar-cell array keeps the battery charged at a low level for this purpose.

The total electric power subsystem consists of (1) the solar array, the only source of new energy after launch; (2) the battery, which acts as a temporary source of power during the shadow period and as a reservoir to supply peak demands in space; (3) converters that change bus power into the voltages and current levels required by the TWTs and other spacecraft equipments;[11] (4) current and voltage sensors and protective devices; and (5) power switching and distribution equipment. The block diagram

[11] Individual experiments are supplied with converters to convert bus power to meet their specific requirements.

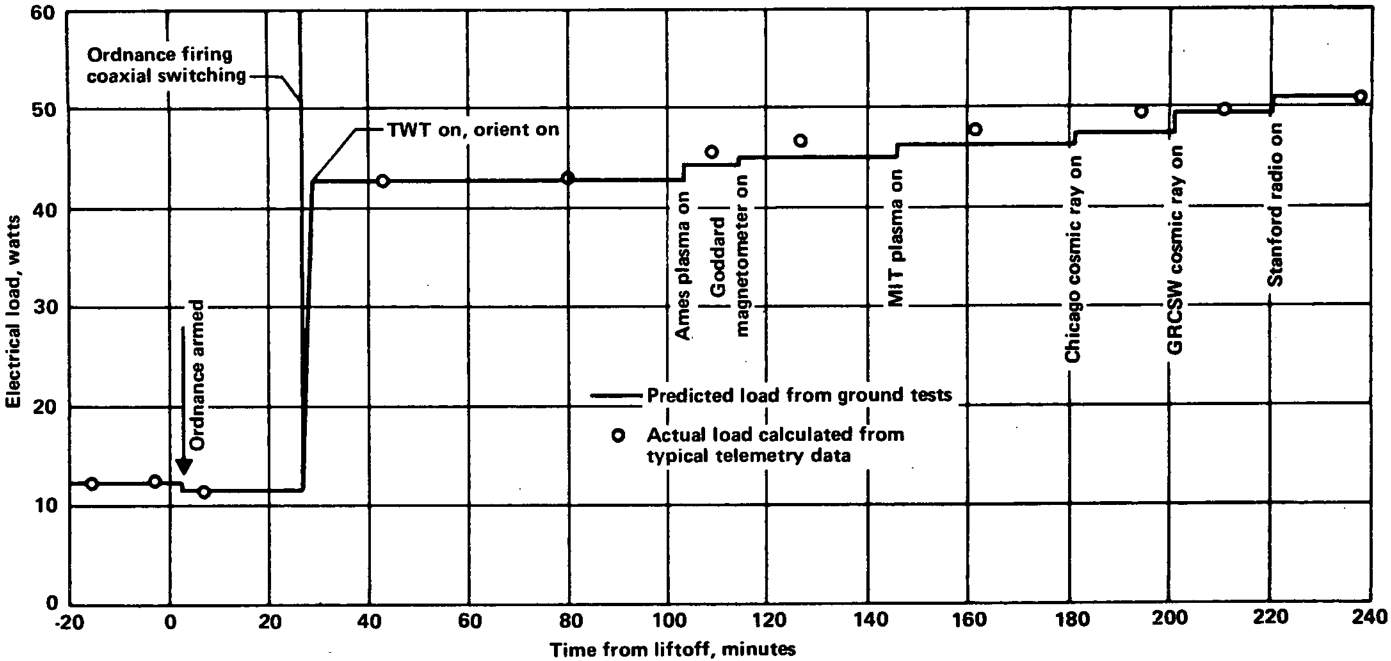

FIGURE 4–15.—Pioneer-6 power profile during the first 4 hr of flight. Superimposed upon the average power level are peak values up to 10 W from the MIT experiment and 7 and 8 W from the pneumatic valve solenoid in the orientation subsystem.

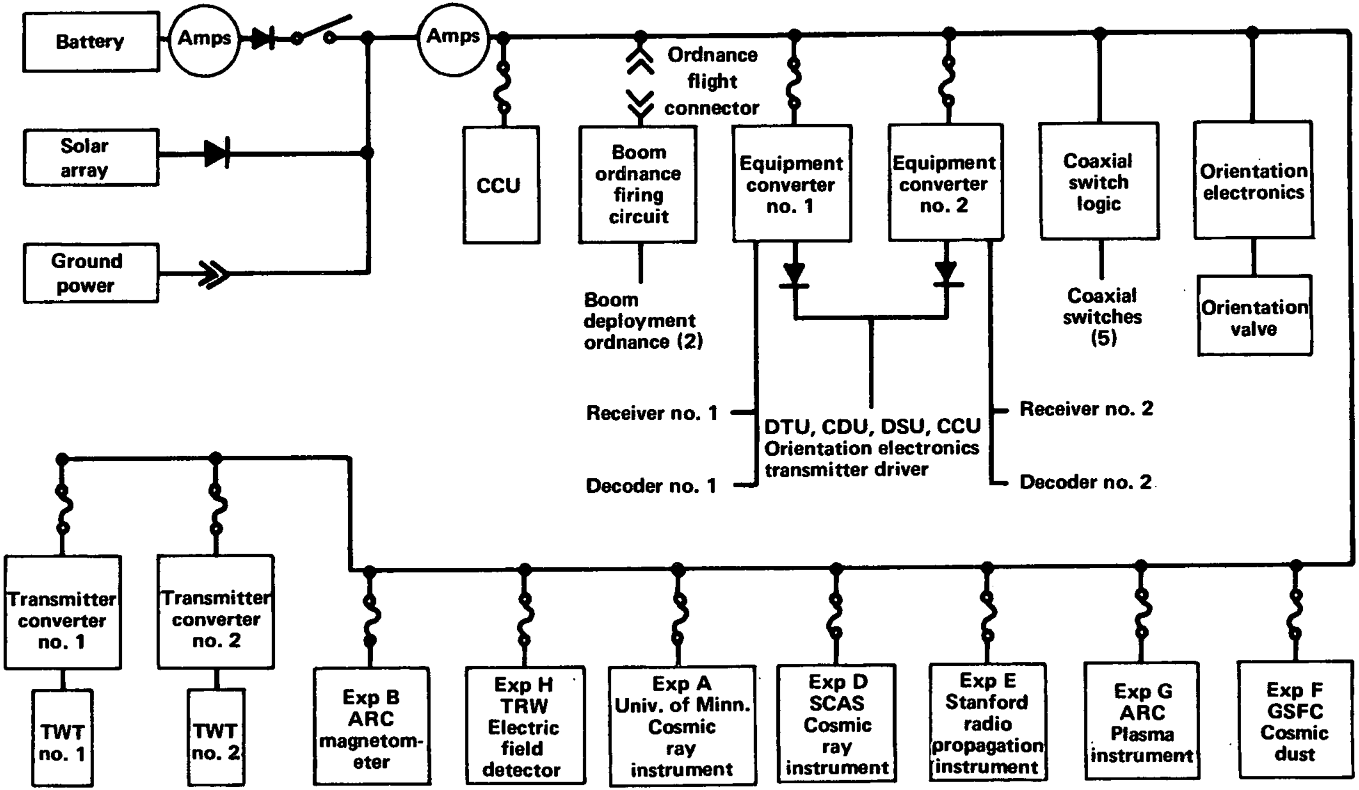

FIGURE 4–16.—Functional diagram of the Pioneer-9 electric power subsystem.

in figure 4–16 reflects the complexity added by items (3), (4), and (5); the power-conditioning, power-distribution, and protective equipment.

The Electric Power Subsystem Interfaces

The preceding sections have dealt almost exclusively with electromagnetic and information interfaces—those associated with the brain and nervous system of the Pioneer man-machine system. In a biological analogy, the power subsystem must represent the heart and blood vessels of the spacecraft. Subsystems are completely dependent upon electrical power to do things, even the pyrotechnic stored-energy devices that effect boom deployment are detonated electrically. To illustrate the close relationship between action and power on Pioneer, one has only to examine the STL feasibility study and proposal. Early in the program, the CDU was considered part of the power subsystem rather than the command subsystem, because of the relationships between commands, energy, and physical action. The CDU was later consigned to the command subsystem because in reality it is a complicated control valve that permits pulses of power to flow to command-selected spacecraft equipment. The pulses, in turn, operate switches and fire ordnance. Thus, pulses of power animate the spacecraft while the steady bus power keeps the vital functions going.

Other interfaces are more straightforward. Because the power subsystem must sustain the spacecraft electrical load continuously and cannot depend upon the battery for anything but short bursts of power, the solar array must be kept directed toward the Sun as accurately as possible. Thus, the power subsystem imposes on the orientation subsystem the requirement that the spin axis be perpendicular to the Sun line within 2°. The shadowing or solid-angle interface with the spacecraft booms is the cause for the solar-cell-viewing band or bellyband around the girth of the cylindrical portion of the spacecraft. A thermal radiation interface exists between the solar array and the exhaust plume of the solid rocket motor comprising the final stage of the Delta launch vehicle. The plume fans out behind the motor to such an extent in a vacuum that the solar cells can obliquely "see" the hot gases. The thermal radiation can be particularly serious when the exhaust carries metal particles, as it does with newer high performance solid fuels. Fortunately, the solar cells on Pioneer were not compromised by the thermal plume.

The solar cells also interface directly with the solar electromagnetic and particulate radiation as well as the micrometeoroid flux prevailing between 0.8 and 1.2 AU. Solar thermal radiation raises the cell temperatures as the spacecraft swings in toward the Sun; this results in a drop of energy-conversion efficiency. The particulate radiation and hard electromagnetic radiation emitted by the Sun can damage the cells over a long period of time. Glass covers are applied to reduce this effect. Figure 4–17 illustrates two of these considerations:

(1) If the spacecraft ventures closer than 0.8 AU to the Sun, the solar array becomes "voltage-limited." Increases in solar power are more than offset by voltage losses due to overheating. Outward from 0.8 AU, the power subsystem is power-limited by the dwindling solar flux.

(2) The predicted useful power generated drops about 10 W between 6 months and 3 years due to radiation damage of the solar cells.

Pioneer 9, an inward Pioneer, could not operate at 1.2 AU due to increased loads over the nominal Pioneer. The increases in electrical load came primarily from the instruments and the convolutional coder.

These interface forces obviously play a leading role in power subsystem design.

The Design Approach

Flexibility and reliability were two critical design goals. Flexibility applied not only to the spacecraft's capacity to handle various scientific payloads, but also the ability to operate between 0.8 and 1.2 AU without the basic spacecraft design's being altered. The problem of the scientific instruments' differing from spacecraft to spacecraft was handled by the provision of a convenient bus voltage and placement of the burden of making further modifications upon experiment power converters.

By design, the bus voltage varied with distance from the Sun (fig. 4–17). The entire Pioneer power subsystem "floated" at a voltage determined by the solar-cell temperature. The spacecraft was overpowered intentionally on inward missions. The battery was provided with taps at lower voltages for use during the inward missions. Enough solar cells were added to the nominal spacecraft that it could operate at 1.2 AU; thus there were too many at 0.8 AU.

The pursuit of high subsystem reliability led to extensive paralleling or cross-strapping of critical components. The TWTs are fed by separate, independent converters, but much of the remainder of the spacecraft equipment receives power from two cross-strapped converters (fig. 4–16). Redundancy in the solar-cell array groups the 10 368 cells into 48 strings, each consisting of 216 cells. Each string is a series-parallel arrangement of four parallel groups of 54 cells in series. Fortunately, the Pioneer cells do not undergo the repetitive thermal cycling characteristic of cells on Earth satellites. Yet, one may expect a certain number of failures due to long-term thermal cycling as the spacecraft approaches and recedes from the Sun. The blocking diodes (needed to prevent cells on the sunlit side from sending current through those on the dark side) are also fallible elements. The impact of micrometeoroids—of large concern during the early days of the space effort—assumes negligible importance away from Earth. With

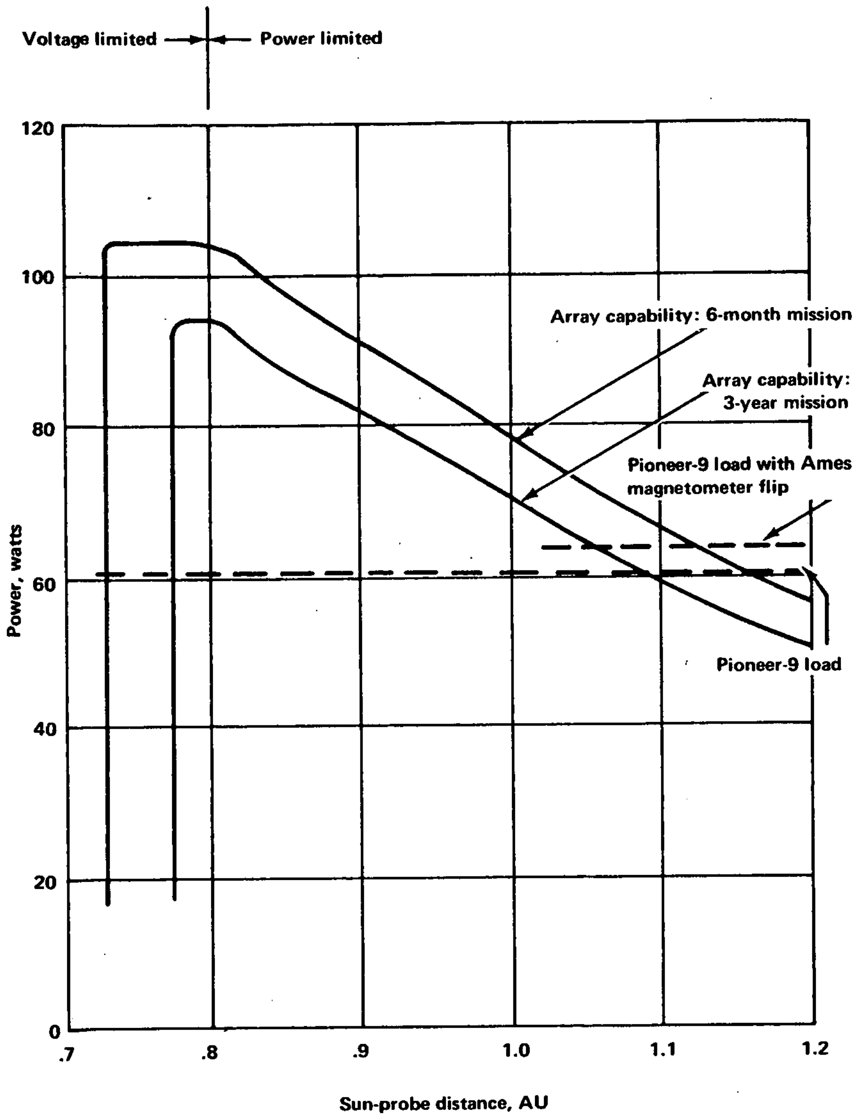

FIGURE 4–17.—Solar array power output vs. distance from the Sun, showing inverse-square-law and cell-heating effects.

these factors in mind, STL computed the reliability for the entire power subsystem to be a very high 0.9963 for a 6-month lifetime (fig. 3–1).

Pioneer Power Budgets

Power requirements changed slightly from mission to mission. The largest change took place between the Pioneer 8 and 9 missions, when the convolutional coder was added and the Goddard magnetometer was replaced by one from Ames. These changes are summarized in table 4–15; these are, of course, average power levels, and the switching among the many spacecraft and experiment modes always created a varying power profile (fig. 4–15).

The Solar Array

The Pioneer solar cell is a high efficiency, solderless, n-on-p type, with 1 to 3 ohm-cm base resistivity. Each cell is 1×2 cm and is covered by a 0.15-mm glass slide for radiation protection. Early in the program, the average cell efficiency target was 12 percent; this was never achieved and the cells on the spacecraft averaged about 10.5 percent. Both suppliers, RCA and Texas Instruments, had considerable difficulty manufacturing cells to the demanding Pioneer specifications.

The individual cells were fabricated into two types of modules. In the first type, 12 cells were interconnected so that 3 were in series and 4 in parallel; in the second, there were 6 in series and 4 in parallel (figs. 4–18 and 4–19). A close look at figure 4–19 seems to show the cells "shingled" together along the long edges according to conventional practice. Actually each cell is soldered to metal connectors; this makes the modules both self-supporting and flexible. It was this flexibility that allowed the modules to be affixed with silicone rubber adhesive to a curved substrate conforming to the cylindrical spacecraft surface. The rather awkward faceted construction comprised of many small flat solar-cell modules originally proposed

TABLE 4–15.—*Pioneer Power Budgets*

Average electrical loads (W)	Pioneer spacecraft 6	7	8	9	E
Spacecraft system [a]	43.4	43.6	43.1	43.66	41.86
Experiments	9.2	8.2	12.3	17.57	17.80
Total	52.6	51.8	55.4	61.23	59.66

[a] Includes 30 W for the TWTs.

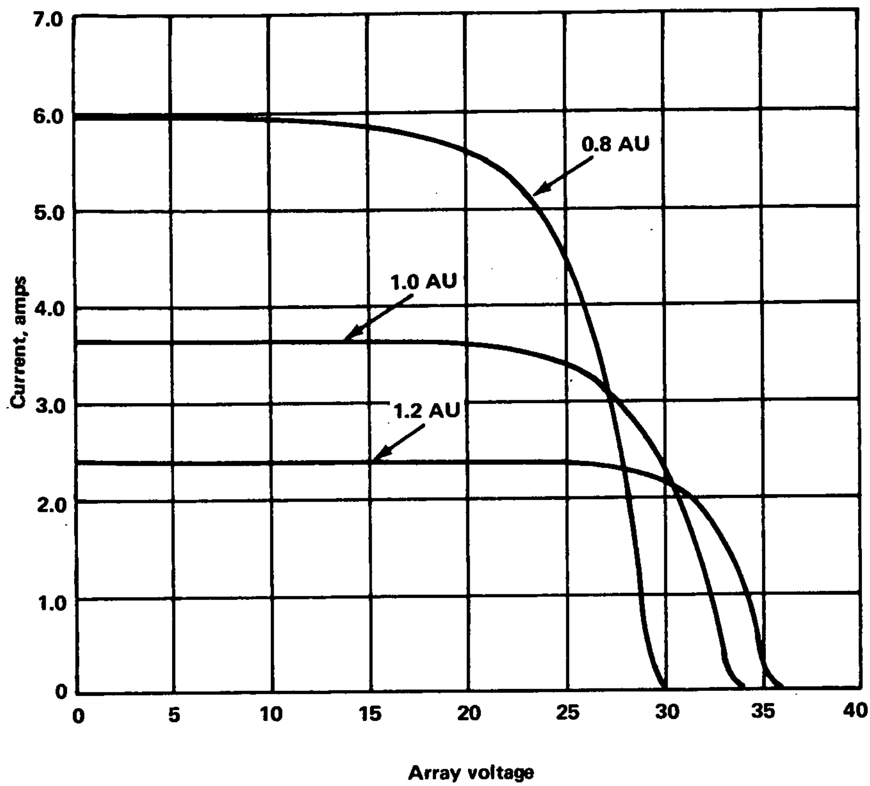

FIGURE 4–18.—Pioneer solar array output characteristics.

was thus eliminated. The self-supporting property did away with the usual module substrate (assumed in the STL proposal), reducing solar-cell-array weight. Instead, the modules were bonded to fiberglass face sheets separated by and bonded to an aluminum honeycomb core. The large curved panels created in this way were then attached to the spacecraft structure. These advances in array design and fabrication cut the array weight from 30 to 15 pounds.

Each of the 48 solar-cell strings was made from interconnected modules and a blocking diode. The diodes, in effect, permit power to flow out of, but not into, the strings. The strings cover a total area of 22.8 ft^2; essentially this is all of the spacecraft's cylindrical surface except for the 7.5-inch viewing band—the locus of the heaviest boom shadowing. Solar cells along the edge of the bellyband are provided with shunt diodes arranged so that, even if they are shadowed, other cells in the string can still provide useful power to the spacecraft. The capability of the solar array is summarized in table 4–16.

The solar-cell strings are paralleled and attached to the bus feeding the

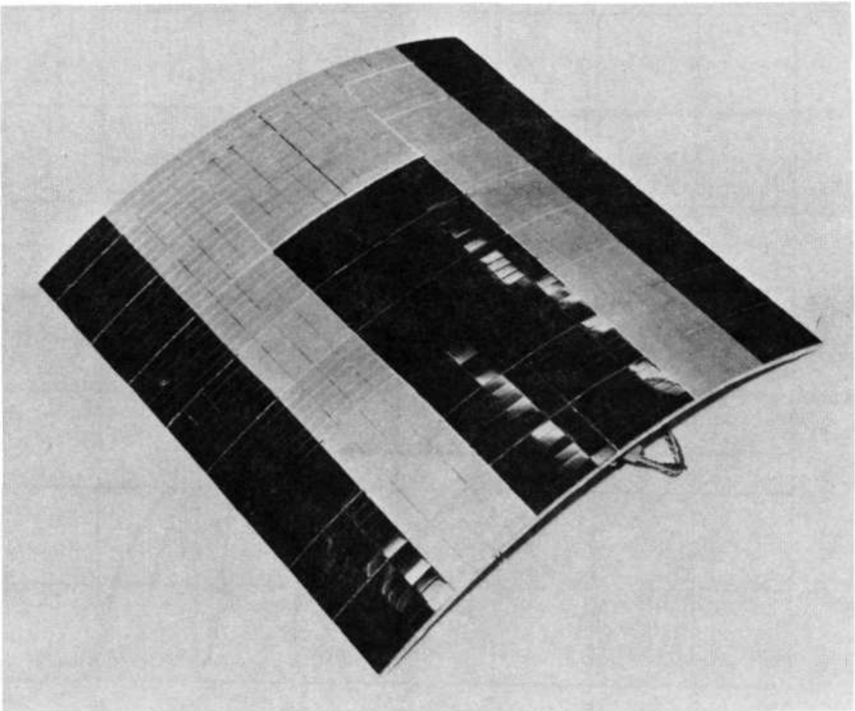

FIGURE 4–19.—One of the Pioneer solar panels, showing both 12- and 24-cell modules mounted on a curved substrate. (Courtesy of TRW Systems.)

spacecraft equipment and experiments. The bus voltage "floats" at the solar-cell array voltage. NASA specifications restricted the voltage swing of this bus to 28^{+5}_{-4} volts for any load between 15.3 and 55.6 W in interplanetary space between 0.8 and 1.2 AU over the nominal lifetime of 6 months. These specifications were met satisfactorily.

The Battery

A Pioneer is completely dependent upon its battery from the time ground power is severed on the launch pad until the fairing is jettisoned, and while the spacecraft is in the shadow cast by the Earth. During the latter period, the battery must supply about 12 W. After orientation, at the discretion of the mission controller back on Earth, the battery is left connected across the bus bar dominated by the solar-cell array voltage. The mission controller disconnects the battery by command if it begins to compromise the mission for some reason. Normally, the battery is left on for 6 to 12 months to accommodate any temporary power shortages or overloads. No power shortages have been known to occur in practice.

TABLE 4–16.—*Solar Array Capability*

Type of cell	n/p
Efficiency of bare cell	10.7 percent min.
Cell internal resistance	0.46 ohm/cell
Array configuration	48 strings in parallel (each string: 4 parallel, 54 series cells; total cells/array: 10 368)
Assembly losses	1 percent
Glass losses	5.4 percent
Losses due to proton damage	0
Diode losses	1.0 V max.
Temperatures	+116° F at 0.8 AU +52° F at 1.0 AU +4° F at 1.2 AU

Calc min. net power	Range										
	0.8 AU			1.0 AU				1.2 AU			
Bus volts	24	25	26	24	26	27	29	27	28	29	30
Watts	103	89.5	63	79.6	81.4	79.7	62.3	59.7	60.5	60.6	57.3

Typical peak loads include instrument-power peaks, fault clearing, coaxial-switch operation, and pneumatic-valve operation. The battery is eventually disconnected when its age begins to make it a poor risk, sometimes as late as 18 months after launch.

Originally, a non-rechargable battery was proposed for the spacecraft, but a study of the MIT plasma probe power requirements showed the need for a rechargeable battery that could meet the experiments' peak demands.

The battery finally chosen for Pioneer was of the sealed, silver-zinc type, which lends itself well to operation in the floating mode. The sealed case was made from fiberglass, a non-magnetic material. As already mentioned, the battery can be wired for inward and outward missions. Although it was built with 18 cells, taps were provided at 16, 17, and 18 cells, for the sake of mission flexibility. On the Block-I spacecraft, the usable battery capacity was about 1 A-h; on Block-II, it was increased to roughly 2 A-h. During normal mission operation, the battery was recharged only when the solar-cell-array voltage exceeded the battery voltage. However, the blocking diodes in the solar-cell array prevent battery current from flowing through the cells should the array voltage drop below that of the battery. In the event of a transient demand for more power than the solar cells can provide, the bus voltage drops until the battery level takes over. No battery charge-control devices exist on Pioneer, but the battery usually remained only partially charged. Battery volume was 44 in.^{3}; weight, 2 lb; reliability, 0.99975 for 6 months' operation between 40° F and 80° F.

The Converters

Three classes of power converters transform bus power into usable form for: (1) the TWTs, (2) the scientific instruments, and (3) the rest of the spacecraft equipment.

Each TWT has its individual converter. The TWT converters are similar to those used for the spacecraft equipment except for special output voltages, including 1000 V for the TWTs. The 1000-V line also must be regulated rather precisely: ± 0.5 percent over the $28\,^{+5}_{-4}$ voltage swing of the bus bar. The TWT converters may be commanded on and off separately; but the switching logic is such that the TWTs cannot operate simultaneously. Furthermore, if the bus voltage falls below 23.5 V for more than 0.4 sec, an undervoltage command automatically shuts the TWTs off to preclude defocusing them or burning them out. Removal of the TWT load of approximately 30 W causes an immediate rise in bus voltage under normal operating conditions. Minimum efficiency specified for the TWT converters was 80 percent. Each converter weighs 2.35 lb and has a volume of 64.12 in.^{3}.

As mentioned earlier, each scientific instrument possesses its own converter tied directly to the bus. All experiments are turned off simultaneously via a ground command or an undervoltage condition, but they may be turned on one at a time. Simultaneous turn-off permits quick diversion of power to spacecraft equipment in the event of an emergency. The instruments are also turned off by the same automatic undervoltage command that shuts down the TWTs.

The two equipment converters are packaged together, weigh 3.2 lb, and occupy a volume of 111.3 in.^{3}. The units are identical, but their outputs are partially cross-strapped. Converter no. 1 supplies receiver no. 1 and decoder no. 1, while converter no. 2 provides power for receiver no. 2 and decoder no. 2; these power taps are not cross-strapped. All other outputs are cross-strapped and supply the CDU, DTU, DSU, the orientation subsystem, the transmitter driver, and the signal conditioner. The individual converters are fused separately and may thus be automatically removed from the circuit in the event of a short or some other fault that draws high current. However, the equipment converters cannot be turned on and off from the ground. The reason for this restriction is obvious; if both were turned off inadvertently, the spacecraft would be dead; without receivers it would no longer respond to commands.

Power Control and Distribution

Power distribution within the spacecraft is commandable and automatic, with some provision for commandable override of the automatic. The solid-state logic for all power switching, and the switches themselves reside in the CDU (fig. 4–14). The power-distribution portion of the CDU is illustrated in figure 4–20. Tables 4–13 and 4–14 show a large number of on-off commands that are really power-on/power-off commands that connect or remove components from the power source. With command 107, the mission controller can override the automatic undervoltage switch that disconnects the TWTs and experiments. Command 000, labelled "undervoltage simulate," is used if the TWTs and experiments must be disconnected all at once. Command 000 is then OR-gated with the undervoltage signal (which has not yet disabled the TWTs and experiments because the voltage is still within bounds), and the TWTs and experiments are turned off. Command 107 may also be employed to lock out command 000.

The undervoltage control senses the bus voltage from a voltage divider with a half-volt resolution. The trip point is adjustable and is usually set for 23.5 V. To prevent the inadvertent shutdown of the TWTs and experiments due to transients, the undervoltage control has a half-second time constant. Like all electronic circuits, the undervoltage control is fallible and might fail in a way that would shut down the spacecraft. The undervoltage override command was introduced primarily to prevent such an occurrence.

A number of current and voltage monitors report the operational condition of the electric-power subsystem to the mission controller back on Earth. These are listed in tables 4–9 and 4–10 with the other housekeeping telemetry words.

THE ORIENTATION SUBSYSTEM

Only a small, spin-stabilized spacecraft could meet the cost, reliability, and launch-vehicle constraints of the Pioneer. For maximum utility, the Pioneers had to be oriented after launch so that their spin axes were perpendicular to the plane of the ecliptic. Only in this orientation would:

(1) The scientific instruments be able to scan along the plane of the ecliptic

(2) The disk-shaped antenna beam intercept the Earth, permitting greater communication range

(3) The solar array power be maximized, eliminating the necessity of cumbersome, failure-prone solar paddles

(4) The spacecraft's thermal control subsystem be able to radiate waste heat out the bottom of the spacecraft away from the Sun into cold space easily

The success of the Pioneer mission depended completely upon twisting the spacecraft's spin axis around after injection until its high-gain antenna pointed within 2° of the north ecliptic pole. The same orientation equipment performing this maneuver could also be used later in the mission to

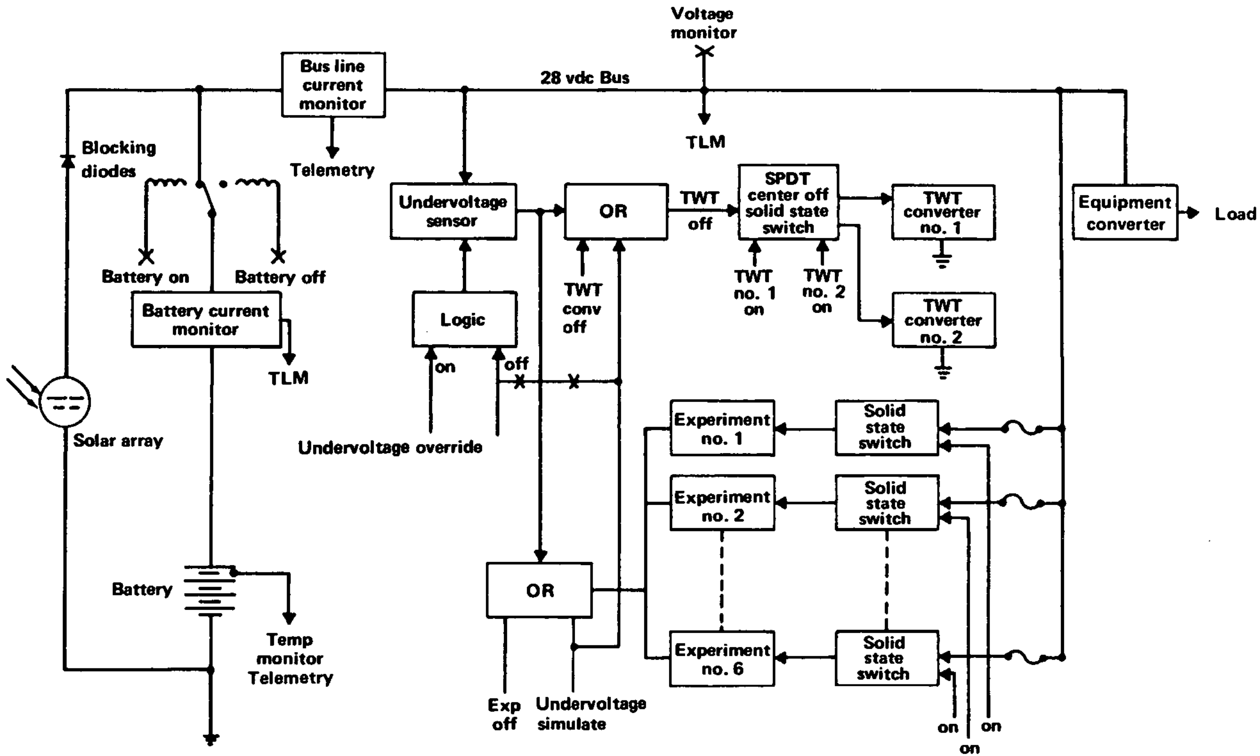

FIGURE 4–20.—A portion of the CDU showing how spacecraft power is controlled.

adjust spacecraft orientation if the axis drifted out of the $90° \pm 2°$ attitude with respect to the plane of the ecliptic.

The most important components needed in such an orientation maneuver are: (1) a device to torque the angular momentum vector of the spacecraft, (2) sensors to control the direction of axis motion, (3) sensors to signal the status and, hopefully, the success of the orientation maneuver, and (4) a nutation wobble damper to dissipate nutation energy induced during the orientation. The small solar sail added at the tip of the high-gain antenna mast to offset any residual torque due to solar pressure[12] was not part of the original design.

The components will be covered in more detail later; first the orientation concept will be sketched out completely. After the spacecraft is injected into the plane of the ecliptic, two pairs of Sun sensors determine the attitude of the spacecraft with respect to a line joining Sun and spacecraft. The Type-I orientation maneuver commences automatically. The Sun sensors cause the nitrogen gas jet to fire and torque the spacecraft spin axis through the smallest angle until it is perpendicular (within ± 0.5 percent) to the spacecraft-Sun line.[13] At this point, thermal control is possible and the solar array generates full power. The Type-II orientation is commanded from the ground and is controlled by monitoring the strength of the spacecraft transmitter's signal strength. When it is maximized, the Pioneer spin axis is also perpendicular to the spacecraft-Earth line; the desired accuracy is ± 1.0 percent. If the spacecraft is perpendicular to both the spacecraft-Sun and spacecraft-Earth lines, it is also approximately perpendicular to the plane of the ecliptic. Orientation is now complete. Spin-axis orientation is maintained through spin stabilization at roughly 60 rpm (ref. 4).

Pioneer Specification A-6669 stipulated the performance of the orientation subsystem more precisely:

(1) It had to function properly whenever the angle between the spacecraft spin axis and spacecraft-Sun line was 10° or greater.

(2) It had to orient the spin axis to $90° \pm 1°$ (changed later to $90° \pm 2°$) from the spacecraft-Sun line.

(3) It had to be able to turn the spin axis around the spacecraft-Sun line for up to 90 days after the orientation maneuvers.

(4) It had to provide enough gas to turn the spin axis a total of 225°.

(5) It had to provide a Sun reference pulse and indicate the orientation relative to the spacecraft-Sun line with a jitter of less than 0.3°.

(6) It could not be deceived by sources of light other than the Sun.

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[12] A net solar torque exists only when the center of pressure does not coincide with the center of mass. The addition of the Stanford antenna was the most significant change introducing asymmetry. The solar sail, of course, had nothing to do with the orientation maneuvers.

[13] The basic orientation concept was first proposed by T. G. Windeknecht in 1961 (ref. 3).

The Sun Sensors

The sensitive elements of the Sun sensors were quad-redundant, photosensitive silicon-controlled rectifier (PSCR) chips, manufactured by Solid State Products, Inc. The chips were developed especially for the Pioneer Program. They delivered a signal to the orientation-control circuitry whenever the Sun was in view. However, the view of each Sun sensor was restricted by aluminum shades (fig. 4–21). On Pioneers 6 and 7 the light-sensitive chips were protected against space radiation damage by 20-mil quartz covers. Several months after launch, however, it was discovered that the Sun-sensor thresholds had changed. Laboratory testing implied that radiation damage was the primary cause, and the quartz covers on Pioneer 8 were made 100 mils thick. The trouble persisted. The real problem was discovered inadvertently at TRW Systems when the sensors were tested under ultraviolet light to see if it degraded the adhesives used in sensor construction. During these tests, it was discovered that the sensors were ultraviolet-sensitive. In space, the ultraviolet light from the Sun had caused the change in the sensor thresholds. Simple ultraviolet filters were added to 60-mil quartz covers; this cured the situation on Pioneers 9 and E.

The five Pioneer Sun sensors are mounted on the spacecraft with the fields of view specified in figure 4–22. Sensors A and C, located on the spacecraft bellyband, looking up and down respectively, help position the spacecraft during the Type-I orientation. As long as the spin axis does not point within 10° of the Sun, except for a small overlap of the field of view, sensors A and C will see the Sun once each revolution as the spacecraft spins. The Type-I orientation proceeds as sensor A or C, whichever one is illuminated, stimulates a succession of gas pulses from the jet on the end of the orientation boom. Each pulse lasts for 45° of spacecraft rotation and torques the spin axis around about 0.15° in the direction of the smallest angular displacement toward maneuver completion. The pulses cease when the other sensor finally sees the Sun. When both sensors see the Sun at the same time, the spin axis will be perpendicular to the spacecraft-Sun line within about $\pm 0.5°$. The original design of the orientation subsystem provided a deadband, rather than overlapping fields of view for sensors A and C. Presumably, the gas pulses would cease when neither sensor saw the Sun. Analog simulation, however, demonstrated that this arrangement was unstable, due mainly to the presence of residual wobble. The simple changes in logic and sensor fields of view solved the problem.

The Type-II orientation employs sensors B and D, also located on the spacecraft bellyband, but with 20° fields of view centered on the spacecraft meridian plane. These sensors do not exercise complete control over the gas pulses that torque the spin axis during Type-II orientations; they only time the pulses. Sensor B, for example, triggers the gas pulse at just the right time for clockwise rotation of the spin axis around the spacecraft-Sun

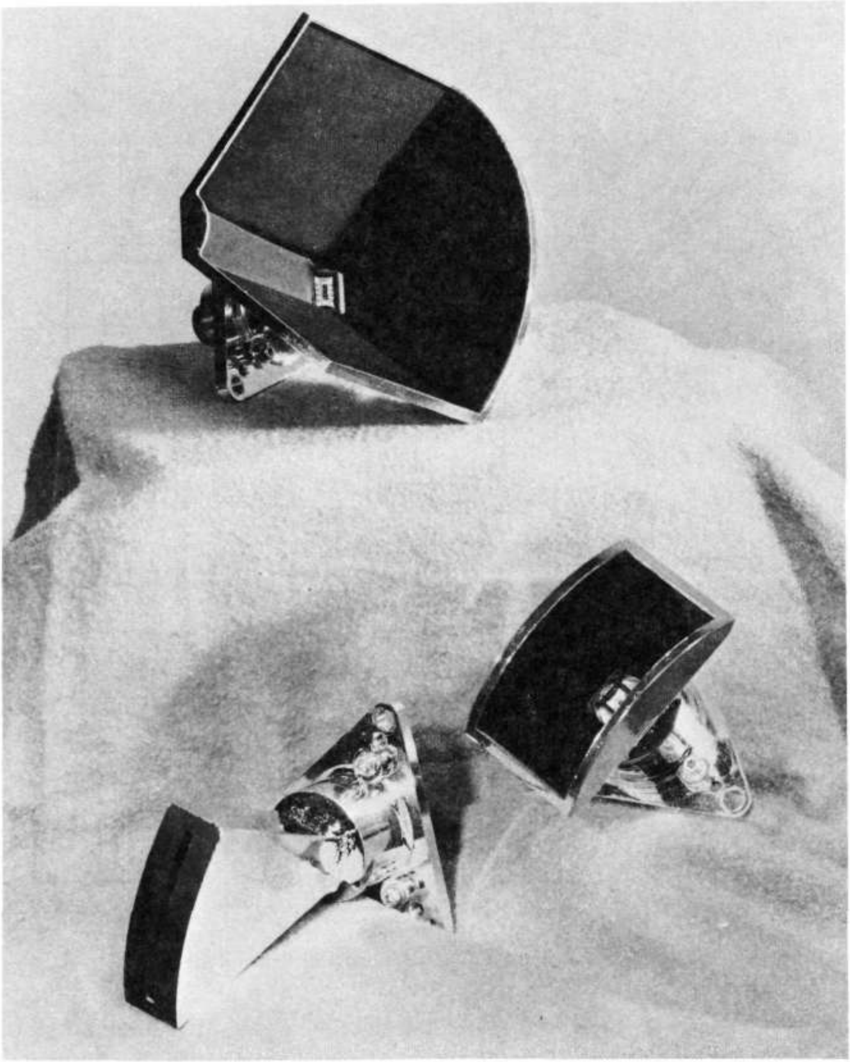

FIGURE 4–21.—The Pioneer Sun sensors: A or C (top); B or D (middle); and E (bottom).

line. (Note that through Type-I orientation, the spacecraft is already in a position perpendicular to the spacecraft-Sun line; it retains this attitude during Type-II orientation.) Sensor D times the gas pulses for counterclockwise torquing of the spin axis. Thus, sensors B and D control the direction and pulse duration but not the extent of the rotation about the spacecraft-Sun line.

The magnitude of the angle of rotation is determined solely by carrier strength of the spacecraft at a DSN station (usually Goldstone); when the

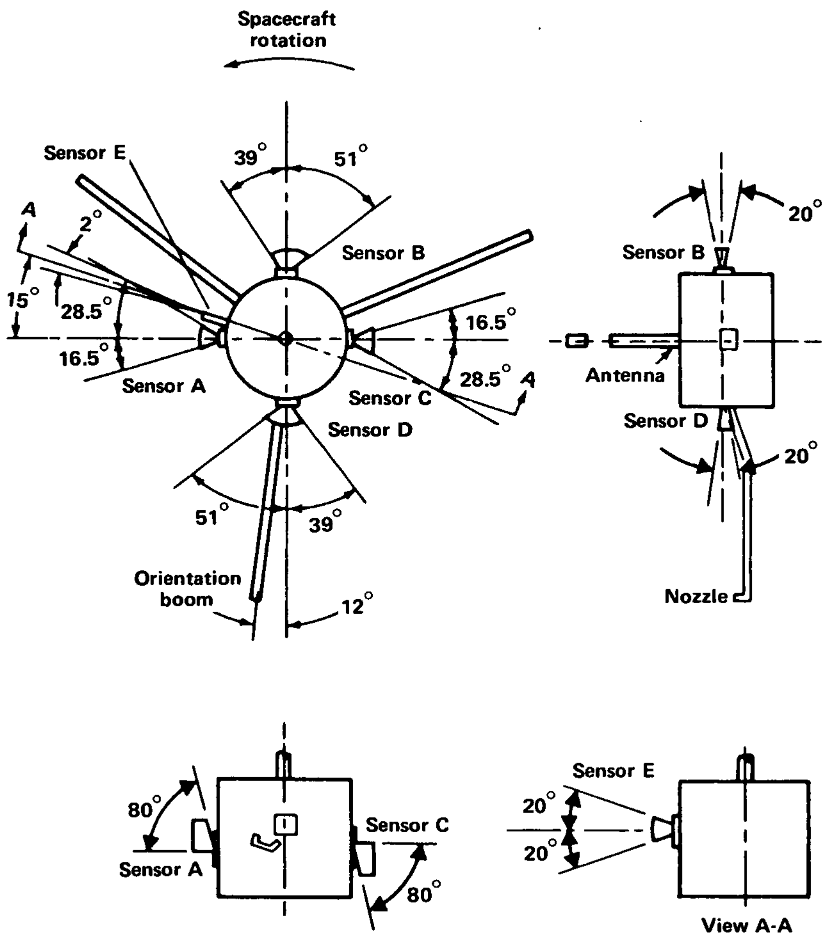

FIGURE 4–22.—Sun-sensor locations and fields of view.

carrier strength is "loudest" (after taking the spacecraft's inclination to the plane of the ecliptic into account), Type-II orientation is complete. During this maneuver, the side lobes of the spacecraft antenna pattern (fig. 4–4) give the mission controller clues about the spacecraft attitude. In practice the maximum is usually overshot a few degrees intentionally to insure that a true maximum has been found and to help calibrate the effectiveness of the gas pulses. Backtracking to the maximum then occurs. (The potential for success of this maneuver was a controversial subject early in the program.) The DSN station senses the relative orientation of the spacecraft antenna mast and then sends signals initiating gas pulses. The seat of control is on the spacecraft during Type-I orientation and at the DSN station during Type-II orientation.

Sensor E establishes the reference position of the spacecraft with respect to the Sun and sends signals to the scientific experiments. Sensor E is also mounted on the viewing band of the spacecraft. It possesses only a 2° field of view that provides short, sharp pulses, as it sees the Sun roughly once each second. Because the field of view is only 40° in the other direction (fig. 4–22), Sun pulses appear only when the spin axis is within 20° of being perpendicular to the spacecraft-Sun line. The appearance of Sun pulses also indicates that the Type-I orientation is proceeding successfully and near its end.

The Pneumatics Assembly

Short bursts of cold nitrogen gas from the pneumatics assembly change the spacecraft angular-momentum vector. Gyroscopes, hot-gas jets, small pyrotechnic devices, and miniature rockets have all been used on Earth satellites for purposes of attitude control. The cold-gas system chosen for Pioneer is simple and extremely reliable. It had already been well proven on other space missions when Pioneer was being designed.

The pneumatic assembly is a titanium alloy pressure vessel containing about 0.9 lb of nitrogen at 3250 psi (fig. 4–23), a pressure regulator, a solenoid valve, a pressure switch, and a nozzle. The nitrogen had to be very dry to preclude the valve's icing at low temperatures. An electrical signal opens the solenoid valve for a moment, releasing a burst of gas at about 50 psi which provides the desired impulse. The solenoid valve and nozzle are located on the end of a 62-in. boom to increase the angular impulse and isolate the iron core in the valve solenoid from the magnetometer (fig. 4–24). Originally the nozzle was on the tip of the high-gain-antenna mast, but it was displaced by the magnetometer during the evolution of the spacecraft. Finally, both were placed on booms.

Simple as the pneumatic assembly is, it was the source of concern on Pioneers 6 and 7. Although the basic missions were not compromised, gas leaks did reduce the amount of spin-axis torquing possible. After a study of the declining gas supply on Pioneer 6 and simulations of the failure in the laboratory, a tuning fork spring was installed under the pressure-vessel regular assembly to cushion it against the shock and vibration of the launch. The problem persisted on Pioneer 7, though greatly reduced in magnitude, and pressure-system seals were tightened beyond the supplier's recommendations on subsequent spacecraft. No important leaks occurred on Pioneers 8 and 9.

Orientation Subsystem Electronics

The function of the orientation subsystem electronics is to deliver the pulses that activate the solenoid valve upon signal from the Sun sensors and command from the Earth. The electronic block diagram is shown in figure 4–25.

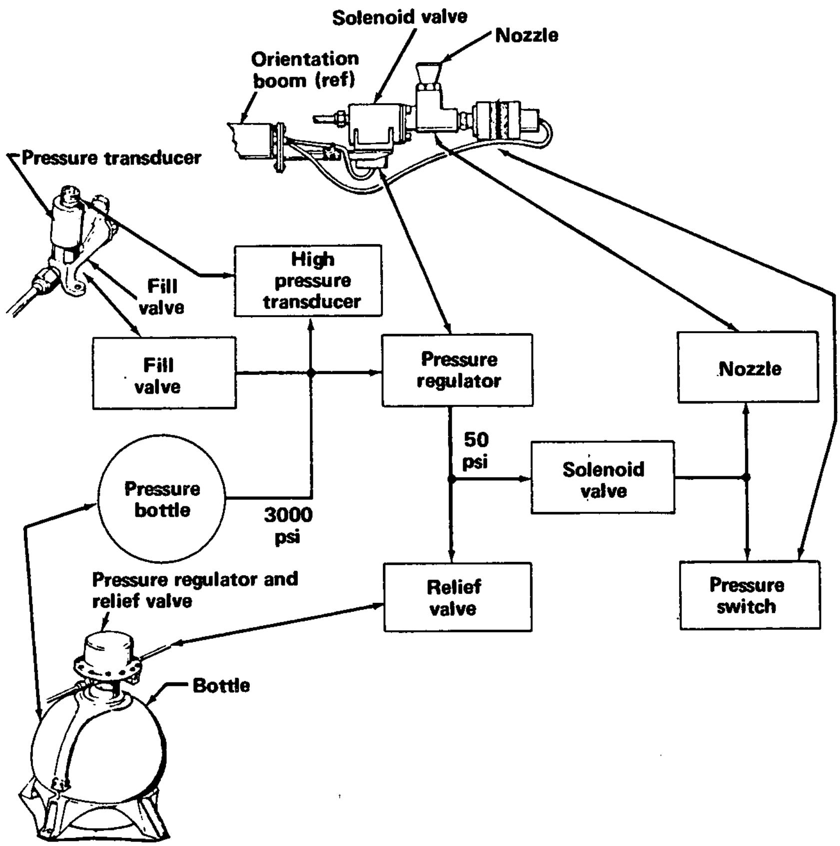

FIGURE 4-23.—Components and block diagram of Pioneer pneumatic equipment.

Whenever the orientation subsystem electronics are switched on by terrestrial command, a Type-I orientation is automatically set into motion. The first Type-I orientation, however, begins automatically when any one of the booms is properly deployed and locked into position, closing a micro-switch. In other words, the first Type-I orientation transpires without ground command and without intervention from any other spacecraft subsystem—it is that important a maneuver. The pneumatic valve is pulsed until Sun sensors A and C are both illuminated. A subsequent Type-I orientation begins whenever the electronics are turned on and always precedes a Type-II orientation. But, once the electronics are on, Type-II commands can be given indefinitely.

During a Type-II orientation, a command from the Earth enables either sensor B or D to signal a respective clockwise or counterclockwise angular impulse. The Sun sensor provides only the precise timing necessary; the

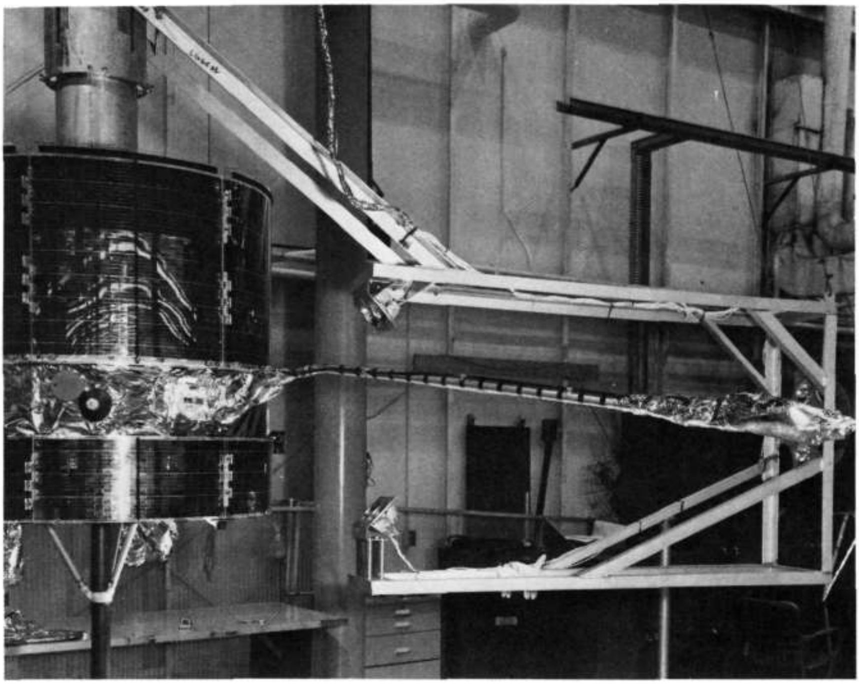

FIGURE 4-24.—The Pioneer orientation boom, shown on Pioneer C during thermal-vacuum test. (Spacecraft is upside down.)

terrestrial command starts the chain of events leading up to the gas pulse. A command from Earth is needed for each Type-II gas pulse; there may be hundreds during a complete maneuver.

The Wobble Damper

Spin-axis nutation may result from the orientation maneuver, from injection and third-stage separation forces, or possibly from an external cause, such as a meteoroid impact (an unlikely event). If the wobble is excessive, it can compromise the scientific experiments and interfere with the Type-I orientation maneuver. A Type-I orientation maneuver usually begins before the wobble damper can remove all of the wobble. The maneuver proceeds until Sun sensors A and C both are illumined. Suppose that the maneuver is moving along satisfactorily with gas pulses slowly torquing the spin axis into position. If the spacecraft is wobbling excessively, however, the unillumined Sun sensor (A or C) will see the Sun momentarily during one of the wobbles before it should; the maneuver is then terminated automatically, prior to actual completion. The average

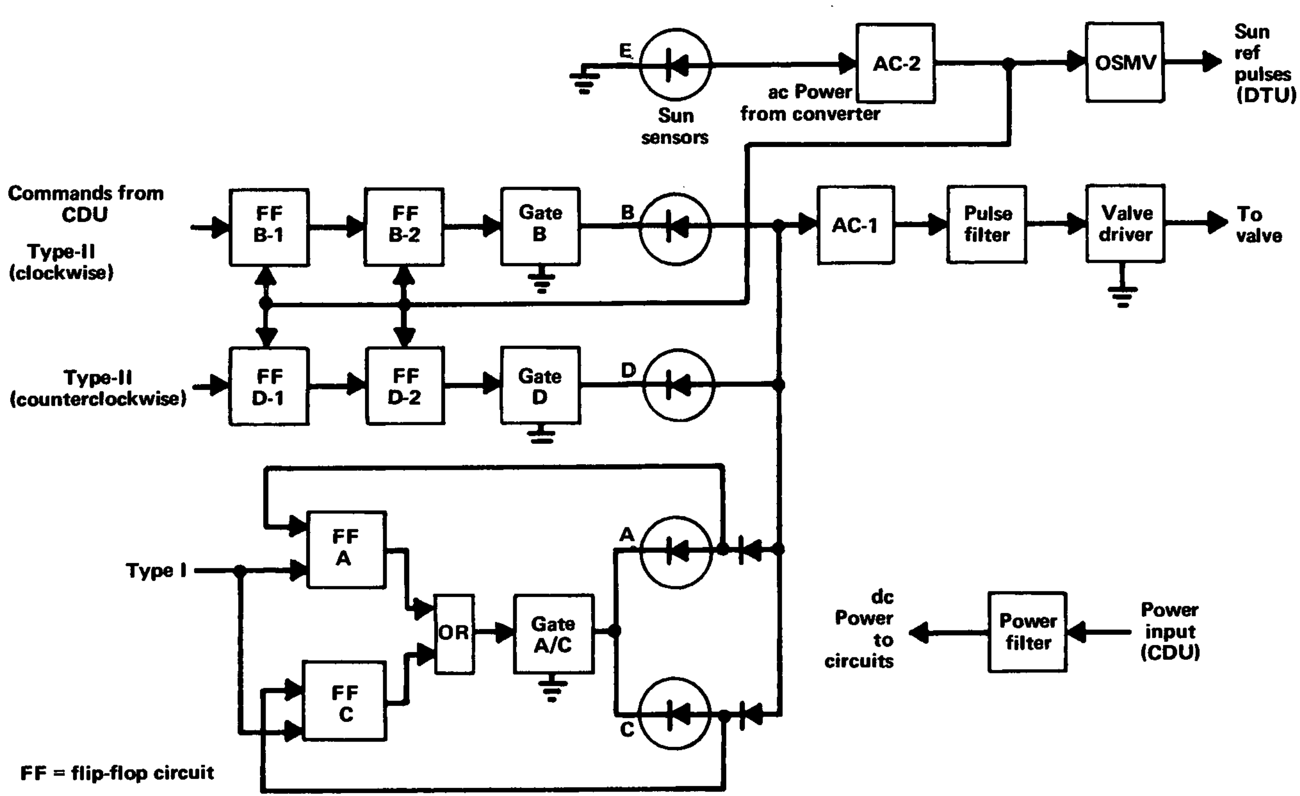

FIGURE 4–25.—Block diagram of the orientation subsystem electronics.

pointing error will equal the peak amplitude of the wobble. Even with a severe wobble, this degree of orientation achieved will probably be sufficient for thermal control and nearly full power production. The spacecraft will be self-sustaining and not dependent upon the battery. Thus there will be time for a wobble damper to suppress the residual wobble. After this occurs, the orientation electronics can be turned on again, automatically initiating another Type-I orientation to trim the spacecraft attitude more finely. In fact, the accuracy of the orientation can be checked by noting whether the gas jet fires upon the initiation of a Type-I maneuver. If nothing happens or if only one pulse is detected, the spacecraft is oriented precisely enough for sensors A and C both to see the Sun.

Most wobble dampers in use on satellites and other spacecraft remove wobble energy by dissipating it as friction-generated heat. On the Pioneer spacecraft, the energy of nutation was dissipated by beryllium-copper balls rolling inside and impacting at the ends of a pair of tubes located at the end of the 62-in. boom. Rolling friction and inelastic collisions at the ends of the tubes extracted the energy of nutation, converting it to heat. Originally, the tubes were to be filled with gas to provide hydrodynamic friction, but it was found that the gas was unnecessary. The damper was built by STL.

Weight, Reliability, and Power Drain

The entire orientation subsystem weighs only 6.5 lb, including about 0.9 lb of nitrogen. This figure includes almost completely redundant parts (with voting circuits) in the electronics and Sun sensor assemblies. The pneumatic equipment is not redundant, although this was seriously considered early in the program. Even so, the reliability of the entire subsystem was calculated as 0.980 for a 6-month lifetime in space. When the orientation subsystem is in a standby mode (as it is most of the time), it consumes roughly 0.6 W. Maximum power is drawn when the gas valve is firing: about 6.3 W.

THE THERMAL CONTROL SUBSYSTEM

The task of the thermal-control subsystem is keeping the spacecraft cool enough (under 90° F) on the inward missions and warm enough (over 30° F) on those swinging away from the Sun to 1.2 AU. The solar heat flux varies from 690 to 307 Btu/hr-ft^2 between 0.8 and 1.2 AU; and Pioneer ground rules stipulated that these conditions be handled without spacecraft design changes for inward and outward missions. The internal heat loads also vary as electrical equipment is switched on and off. These load changes are small, however, roughly a swing of 12 W or about 20 percent compared to the greater than 2:1 fluctuation in solar flux.

NASA and STL engineers also had to examine several transient events or situations that occurred before the spacecraft broke into full sunlight following launch:

(1) The launch-pad environment—the spacecraft-determined air-conditioning requirements had to be examined.

(2) Aerodynamic heating of the shroud during launch and the consequent transfer of heat to the spacecraft—this was controlled by adding thermal insulation (chargeable to spacecraft payload) to the shroud in quantities dependent upon the specific trajectory selected (ch. 7).

(3) Aerodynamic heating of the spacecraft at very high altitudes after shroud ejection—analysis showed that no problem existed here.

(4) Radiant heating of the bottom of the spacecraft by the third-stage rocket plume—the switch to the X-258 third stage, which used aluminum oxide additives in the rocket grain, stimulated concern over excess radiation; a special STL study determined that a thermal shield was needed to block the solar array's view of the plume.[14]

(5) Cooling during eclipse of the Sun by the Earth during ascent—this period, which would last at the most 30 min, would not be long enough to allow the spacecraft to cool excessively. (Actually, the dark side of the Earth contributes considerable thermal radiation to the spacecraft during eclipse.)

In summary, analysis of the transient events from launch pad to solar orbit resulted only in the addition of a radiation shield for the thermal louver actuators and varying amounts of thermal insulation to the shroud. The long cruise around the Sun controlled the major aspects of spacecraft thermal design.

So far, only the thermal control of the spacecraft interior has been mentioned. The solar cells, Sun sensors, antenna mast, and booms must be maintained within operating limits, too. Because they are spacecraft extremities, the thermal control techniques applying to the interior of heat-generating spacecraft may not apply to them.

Coping With Variability

Passive thermal control, employing no moving parts, would have been the simplest and most reliable approach in the Pioneer program. However, the more than 2:1 variation in solar flux and changing internal heat loads ruled out passive control. To illustrate, a passive thermal control subsystem that provided a 60° F spacecraft interior at the Earth's orbit would have permitted the temperature to rise to 142° F and fall to 40° F at 0.8 and 1.2 AU, respectively. The changing internal loads and varying heat leakage through the solar cells and down the antenna mast swung the

[14] This contract (NAS2–1642) was let July 23, 1963, while the main spacecraft contract was being written. The final report, issued in 1964, was entitled "Study of the Effects of X-258 on Pioneer Spacecraft."

internal heat load by 50 percent at these extreme points in the mission spectrum. Active thermal control was the best solution, even though the addition of moving parts would detract from overall spacecraft reliability.

The whole Pioneer mission concept depended upon the concept of a spin-stabilized spacecraft with a spin axis normal to the plane of the ecliptic. The curved sides of the cylinder receive essentially all solar radiation, while the ends point toward cold space. This situation is ideal for a thermally insulated spacecraft with active thermal control. Insulation around the sides of the structure allows only a small portion of the solar heat load to reach the inside of the spacecraft. Insulation on the top leaves the bottom as the only possible exit for heat (fig. 4–26). This heat leakage, which varies depending on the distance from the Sun, can be radiated out the spacecraft bottom along with the variable internally generated heat load. The variability is handled by changing the effective radiating area of the bottom of the spacecraft. Mechanization of the concept consists of a set of Venetian-blind-like louvers that varies the effective radiating area, increasing it as the internal temperature rises and reducing it when the inside of the spacecraft becomes too cool (fig. 4–27). The setting of the louvers is controlled by bimetallic thermal actuators sensitive to the internal temperature. When the Pioneer program began, STL was also applying this basic concept to the OGO, which, though much larger than Pioneer, was fully stabilized in orbit and had many of the same thermal problems. The louvers used on Pioneer came directly from OGO technology. (STL was also the OGO spacecraft prime contractor.) Other space probes and stabilized Earth satellites have also used the same approach as Pioneer.

The thermal insulation covering the spacecraft sides and top thermally isolate the antenna mast, the booms, the Sun sensors, and the solar array from the volume that is temperature-controlled by the louvers. As we shall see below, the components just listed can tolerate the harsher outside environment. Their temperatures can be regulated adequately by passive thermal coatings applied specifically for conditions anticipated on each mission. In reality, then, two thermal control schemes were applied to the Pioneer spacecraft: active control inside, and passive control for the extremities.

In the discussions of the other Pioneer subsystems, the subject of interfaces, particularly information interfaces, has always been high on the list of priorities. The Pioneer thermal control subsystem, however, cannot be commanded from Earth; it is completely automatic. Temperature readings at various locations around the spacecraft are monitored and telemetered, (table 4–17) but if they prove anomalous the only solution is to disconnect electrical loads. Electrical power is not required by the thermal control subsystem. It is a simple subsystem, but just as critical to mission success as subsystems with thousands of electronic parts. STL engineers believed that

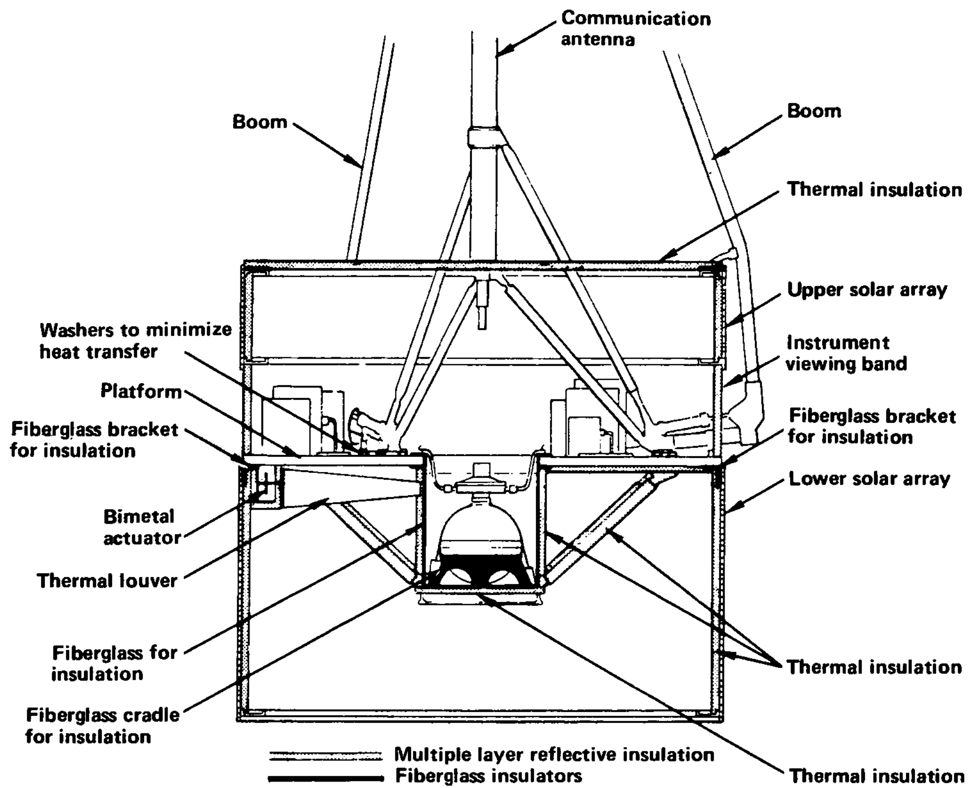

FIGURE 4–26.—Diagram of the components affecting thermal control.

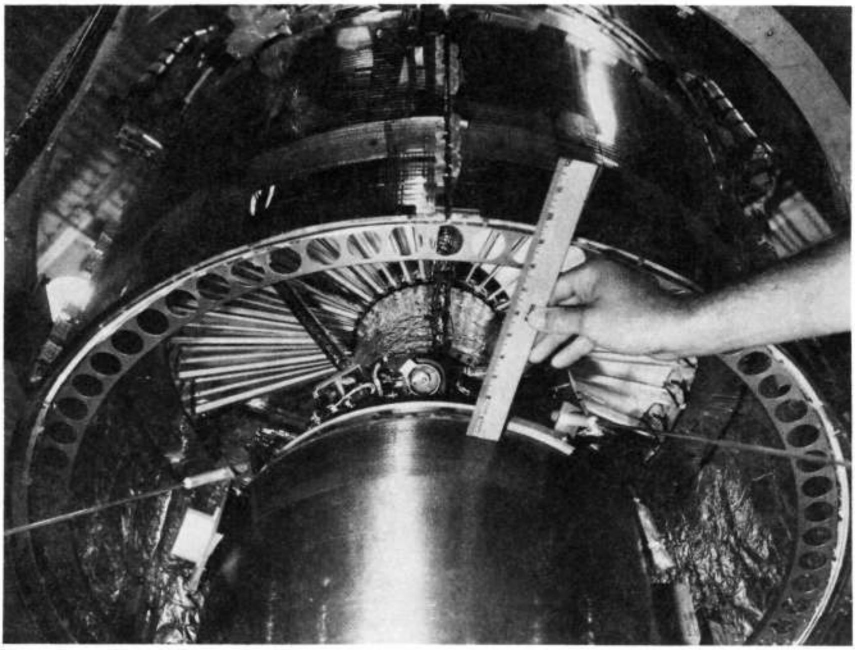

FIGURE 4–27.—Bottom view of the Pioneer spacecraft with the third-stage motor in place. The thermal louvers, covering two-thirds of the equipment platform, are shown in an open position.

the basic spacecraft thus protected could venture between 0.55 and 2.0 AU without thermal control modifications, although contractually they were committed to only 0.8 and 1.2 AU.

Spacecraft Thermal Analysis

Spacecraft thermal equilibrium results if the sum of the internally generated heat power and the heat power leaking in through the insulation and the structures that pierce it exactly equals the heat power radiated through the louver assembly. STL used two analytical thermal models in its computations. The first model assumed that the spacecraft was located 1.0 AU from the Sun. The temperatures at various locations within the spacecraft were then computed. In the second model, the spatial temperature distribution computed in the first model was assumed fixed; that is, relative temperatures remained the same, but absolute temperatures would rise or fall (by the same amount) throughout the spacecraft interior. With this greatly simplified model, the effects of distance from the Sun and internal power loads were calculated. The actual analysis was rather involved and cannot be pursued here.

TABLE 4–17.—*Thermal-Sensor Locations*

Thermal-sensored equipment	Thermal-sensor location
Receivers 1 and 2	On receivers, near voltage-controlled oscillator
TWTs 1 and 2	At juncture of mounting screw and platform
TWT converter	Exterior of converter top cover
Transmitter driver	On platform close to driver
Digital telemetry unit	At juncture of mounting screw and platform
Data storage unit	At base of data storage unit
Equipment converters 1 and 2	At base of equipment converter
Battery	Internal to battery
Upper solar panel and lower solar panel	Approximately 30° to right (top view) of orientation boom. Between substrate and insulation (insulated from compartment).
Platform 2	On platform position 2
Antenna mounting bracket	Midway on bracket, within compartment
Louver actuator housing	Between insulation and housing
Sun sensor A	In head of sensor
Platform 1	On platform position 1
Nitrogen bottle	Epoxied directly to bottle
Sun sensor C	In head of sensor
Platform 3	On platform position 3
Magnetometer sensor (Ames) 4	Internal to boom-mounted magnetometer sensor
Magnetometer, electronics (Ames) 4	Internal to instrument
Plasma, electronics (Ames) 5	Internal to instrument
Cosmic ray (SCAS) 6	Internal to instrument
Cosmic ray (Minn.) 1	Internal to instrument

Because complex geometry and the manifold heat paths make spacecraft thermal analyses so difficult, it is customary to build a thermal mockup or model of the spacecraft. Heat sources and sinks as well as all significant spacecraft structures are usually simulated physically rather than mathematically. Temperatures are measured and compared with those computed. In the Pioneer analysis STL built a thermal model and simulated space conditions between 0.8 and 1.2 AU using their cryogenic vacuum chambers. Different internal loads and solar fluxes were approximated. Agreement between computations and thermal model measurements was good. Inflight performance has also verified the accuracy of the original analysis.

Spacecraft Thermal Design

The back of the solar array, the interstage structure, and the top of the spacecraft (fig. 4–27) are all covered with multiple-layer, aluminized-Mylar thermal insulation. All interruptions in the layers of insulation, the places where antenna, boom, and solar-array supports pierce it, are made high heat impedances with fiberglass mountings to minimize heat leaks into the interior. On the other hand, heat paths from internal components to the bottom of the equipment platform are designed with high conductivity in mind. The spacecraft's instrument platform and the boxes mounted on it were made thermally "black" to encourage temperature equalization. Other "inside" surfaces, such as the top cover and spacecraft sides, were either aluminum or aluminized Mylar. The equipment platform is an aluminum honeycomb panel constructed with the "starved" bonding technique to insure good thermal conductance through it to the radiating surface on the bottom.

The louver system (fig. 4–27) consists of 20 individual louvers, each actuated independently by a spiral-wound, bimetallic spring. Springs are insulated so that they are responsive only to local temperatures. The open radiating area was approximately 3 ft^2. One-third of the platform area, the portion directly under the magnetometer electronics, does not require thermal louvers. Instead, it is covered with aluminized-Mylar insulation. The louver blades themselves were made highly reflective and specular to infrared radiation to minimize radiation from them back to the equipment platform when they were in the full open position. They are also good thermal insulators, so that when closed they help retain heat within the spacecraft. The bottom of the equipment platform is the emitting surface for all waste heat.

Protection of the spacecraft from thermal-plume heating during injection consisted of applying aluminum foil around the top of the Delta third stage and aluminized-Mylar insulation around the interstage ring. The plume heating, however, was not so severe as expected.[15]

Controlling Extremity Temperatures

The spacecraft extremities, including the solar cells, possess no internal energy sources which receive electrical power except the boom-mounted magnetometer and the pneumatic valve. The solar array and each boom and mast were subjected to thorough thermal analysis to determine their approximate temperatures at various distances from the Sun. Thermal coatings were applied to the booms and Sun-sensor shades. If the same thermal coatings are applied for both inward and outward missions, the temperatures of externally exposed components at 0.8 AU will be 1.118 times those at 1.0 AU and 0.913 times lower at 1.2 AU. The application of a different thermal coating may raise or lower the absolute values of the temperatures, but the ratios remain fixed. However, it is little trouble to change the coatings for inward and outward missions, and this was done to a limited extent for the various Pioneer flights.

[15] In 1964, an Argo D-4 sounding rocket was fired from Wallops Island carrying an experiment to measure plume heating. Unfortunately the flight was a failure.

Thermal Control Subsystem Reliability

The thermal coatings, thermal insulation, and thermal conduction paths in the Pioneer spacecraft present no reliability problems. The only moving parts are the individually actuated thermal louvers. Catastrophic failure of several louvers in the neighborhood of a large source of thermal energy is highly unlikely. In fact, the use of individual actuators for the louvers makes the probability of acceptable operation over a 6-month lifetime very high, roughly 0.999.

THE STRUCTURE SUBSYSTEM

The structure subsystem, like the thermal control subsystem, is a largely passive, but critical, subsystem. Spacecraft have rather complex structure subsystems which must be analyzed as painstakingly as the communications subsystem or any other subsystem. The Pioneer structure (figs. 4–28 through 4–30) consists of the following major sections:

(1) The interstage ring and cylinder
(2) The equipment platform and struts
(3) High-gain antenna mast supports
(4) Solar-array substrate and supports
(5) Boom dampers
(6) The booms and associated deployment and locking equipment
(7) The Stanford experiment antenna

Overall Configuration

Spacecraft structure is highly variable. In orbit about the Earth are cylinders, spheres, boxes, even tetrahedrons and other polyhedrons. Subsystem functions and program ground rules determine spacecraft geometry. In the case of Pioneer, axial symmetry was the direct result of the choice of spin stabilization—an essential ingredient of the whole Pioneer concept.

Spin-stabilized spacecraft need not be cylindrical in shape; only symmetry about a spin axis is required. Spheres, for example, also lend themselves to spin stabilization. With Pioneer, however, there was good reason to choose a cylinder. The spacecraft was to be oriented with its spin axis perpendicular to the plane of the ecliptic. Thus, body-mounted solar cells would always be perpendicular to sunlight once each revolution (roughly once per second). Axis perpendicularity was a condition for maximum power generation and obviously a factor enhancing the whole Pioneer concept. Pioneer depended upon several highly dependent, interlocking ideas (ch. 3). The cylindrical body of Pioneer, with the long high-gain antenna mast atop it, is the logical consequence of the Pioneer ground rules of simplicity and low cost, and the Delta payload capability.

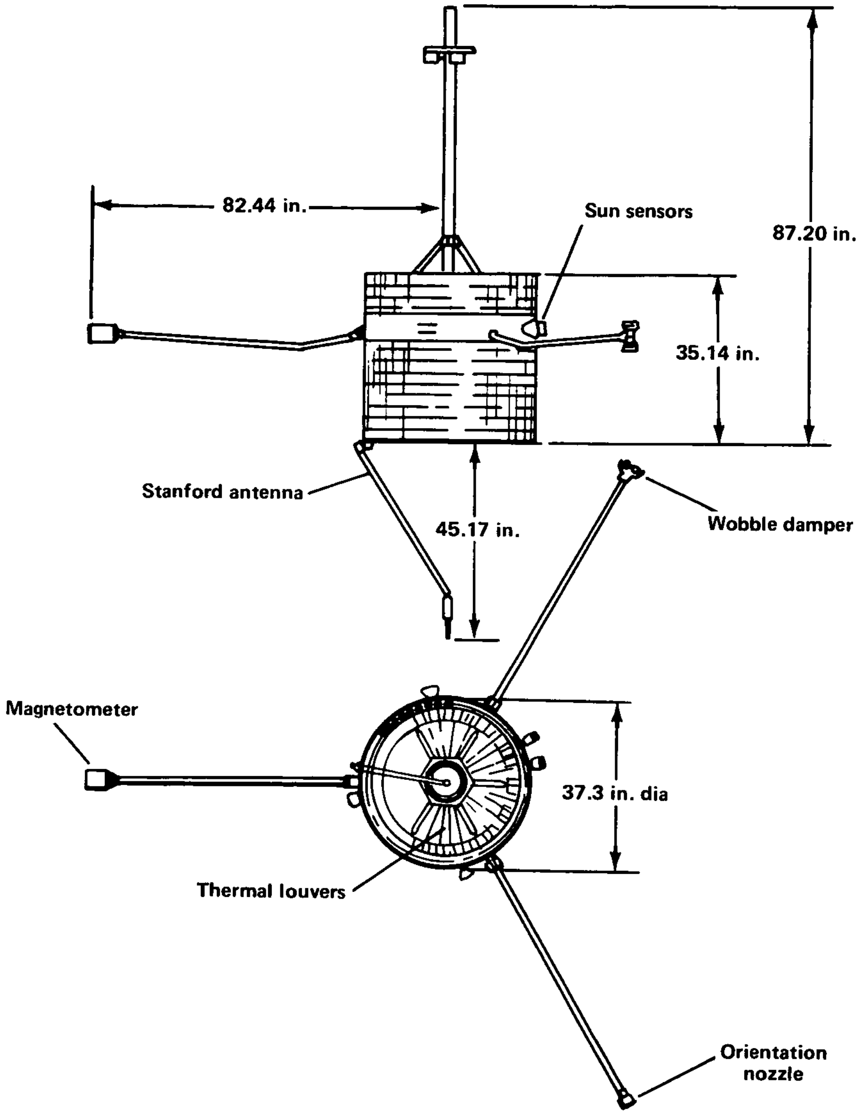

FIGURE 4–28.—Spacecraft external dimensions.

The Pioneer spacecraft sketched out in the feasibility study had no booms at all. Booms were added for three reasons:

(1) With the magnetometer at the top of the antenna mast, as it was in the original concept, spin stability was marginal. The addition of booms, one with the magnetometer at its end, assured stability. Spin stability depends upon a moment of inertia along the spin axis that is greater than those moments along the other axes.)

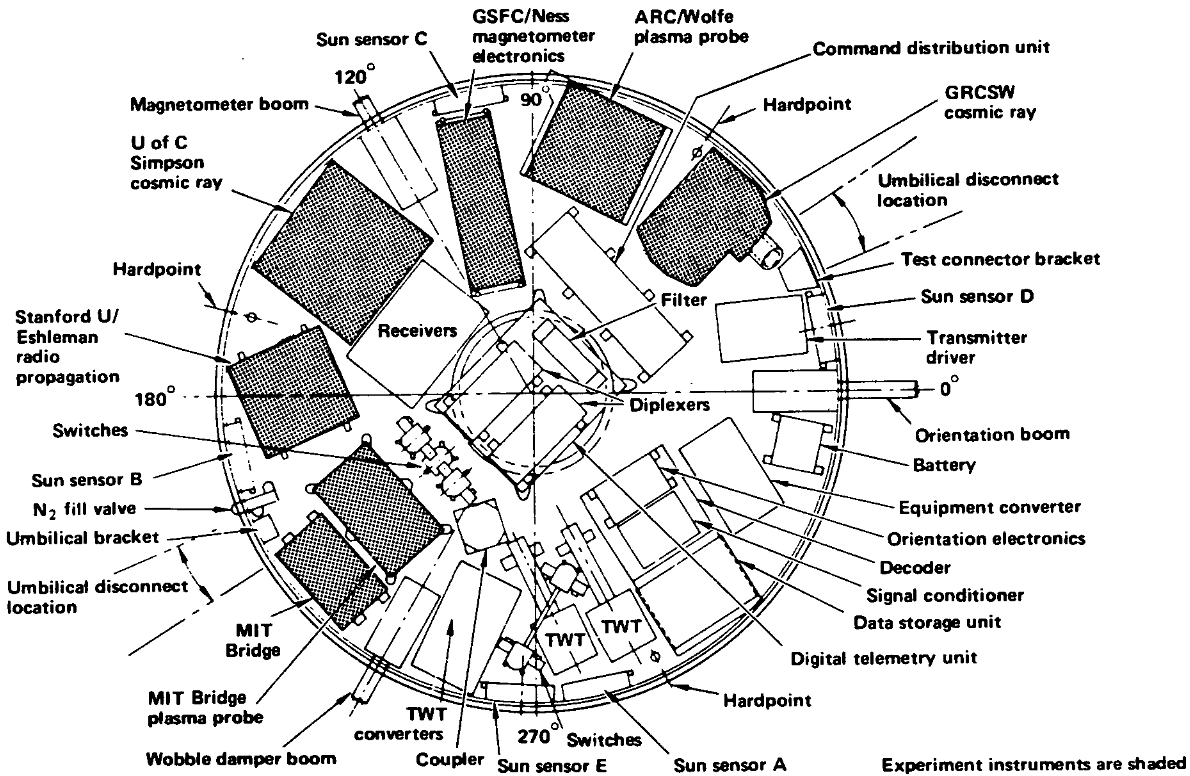

FIGURE 4–29.—Equipment platform layout for Pioneers 6 and 7. Layouts for the Block-II Pioneer were very similar.

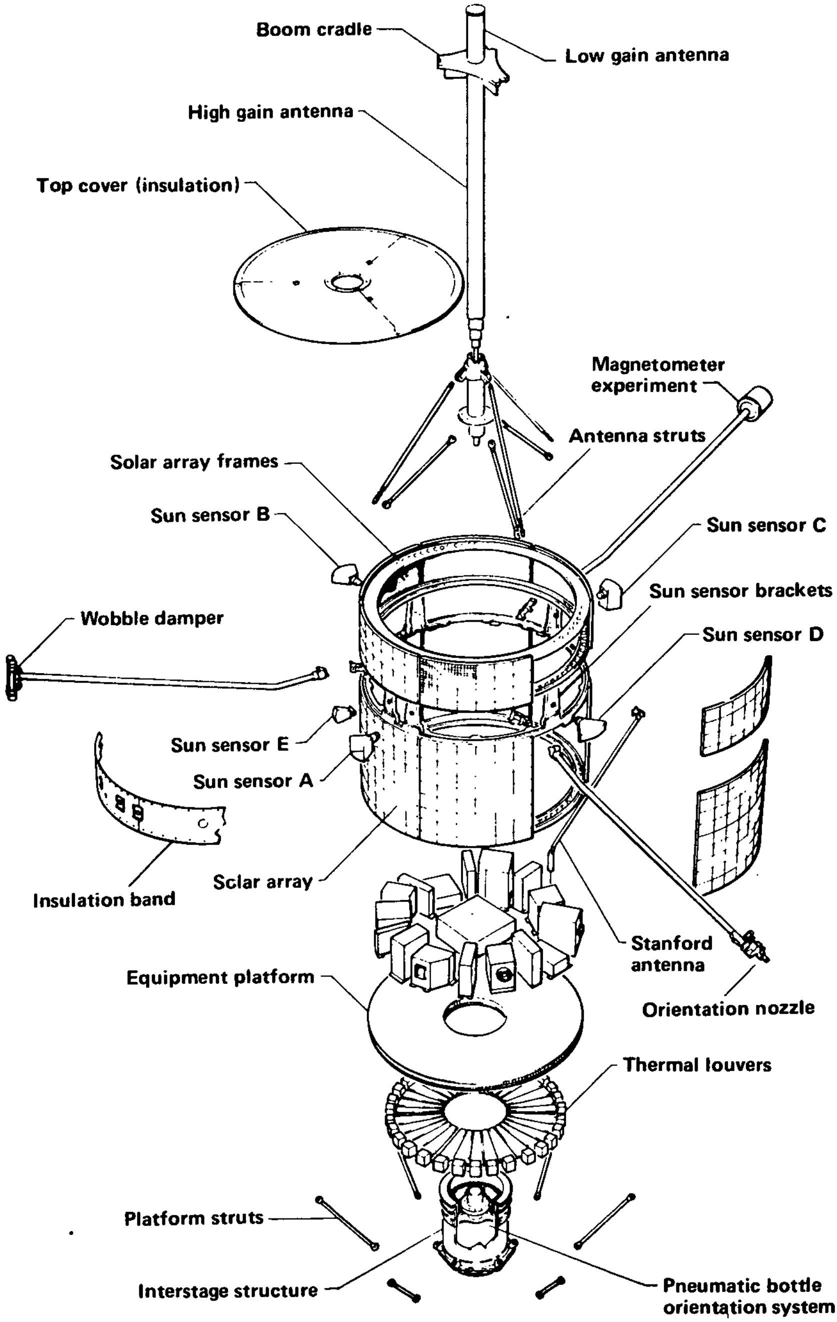

FIGURE 4–30.—Exploded view of the Pioneer spacecraft.

(2) The booms provide magnetic isolation for the magnetometer and exile the orientation nozzle solenoid and the wobble damper, both of which pose magnetic cleanliness problems.

(3) The effectivenesses of the orientation nozzle and wobble damper are increased by placing them on the ends of booms.

The decision to add deployable booms to the spacecraft was most critical from the structures standpoint. Booms are moving parts that must be stowed in a launch configuration and then unfolded and locked in position after the launch vehicle fairing has been jettisoned. Other scientific satellites, such as OGO 1, have been compromised by boom failures. In the case of Pioneer, the advantages of using booms far outweigh their potential liability.

Externally, the Pioneers are cylinders 37.3 in. in diameter and 35.14 in. long, with three booms 120° apart extending 82.44 in. from the spin axis (fig. 4–28). The Stanford experiment antenna projects downward when deployed, and in appearance and complexity is a fourth boom. The high-gain antenna mast projects roughly 53 in. above the top edge of the cylinder. Pioneer, therefore, presents appendages in all directions, in contrast to the relatively clean configuration first suggested by STL (ref. 5).

Internally, the major requirements were support for scientific instrumentation and spacecraft subsystems and, once again, spin-axis symmetry. Symmetry must be taken here to mean the judicious placement of mass around the spin axis to preclude the spacecraft's wobbling. The farther components were located from the spin axis, the greater the spin stability; that is, the better the spinning spacecraft could resist destabilizing influences. The internal configuration (fig. 4–29) follows general spacecraft practice—the major structural element is a strong equipment platform. This platform supports all internal components, the three radial booms, and the high-gain antenna mast. The equipment platform is the internal skeleton. The cylindrical shell, which is rigidly attached to the equipment platform, is constructed of aluminum honeycomb with fiberglass face sheets; it is the structural skin that forms the base of the solar array. Sun sensors and the Stanford antenna are attached to the equipment platform. The major structural elements are the equipment platform, appendages, and cylindrical shell.

Structural Details

The most important structural loads are impressed during the launch process. The acceleration, vibration, and shock loads that dictate much of the spacecraft structure are stipulated in some detail in chapter 7. Once the spacecraft is injected into solar orbit, applied loads fall far below those imposed by the Delta launch vehicle.

An 8.75-in. fiberglass thrust cylinder carries the launch vehicle forces from the aluminum interstage ring to another ring attached to the bottom of the equipment platform. The nitrogen pressure vessel supplying the orientation subsystem nests within this cylinder. Six struts link the platform with the bottom of the thrust cylinder, providing additional rigidity. The equipment platform supports the rest of the spacecraft (fig. 4–30). Three aluminum-mast struts absorb side loads transmitted by the antenna mast. The top cover, which is made of an aluminized-Mylar blanket, bears no loads. The original cover was an aluminum sheet, but it was discarded to save weight. The bottom of the "can" is really the equipment platform, although the solar-array skirt continues downward for another 20 in.

The equipment platform is an aluminum honeycomb sandwich, 0.75-in. thick with 0.016-in. aluminum face sheets. The material weighs 3.1 lb/ft^3. It carries the loads transferred from interstage cylinder and struts to the booms, antenna mast, solar array, etc. The thermal louvers are mounted on two-thirds of the lower surface of the platform. An insulation blanket covers the remaining one-third of the surface. On the top surface, the spaces between mounted equipment are covered with black paint.

Aluminum-honeycomb sandwich material was also used for the solar-array substrates. The inner and outer face sheets are "prepreg" fiberglass sheets. The substrate panels are attached to the equipment platform by fiberglass brackets around the lower ring of solar panels and through the Sun sensor brackets around the upper ring. Support rings at the top and bottom ends complete this part of the structure. Upper panels are interchangeable among themselves, as are those in the lower ring. The 6.75-in. band between upper and lower rings was left bare because the boom shadows would have rendered solar cells useless in that area anyway. This band is closed thermally with an aluminized Mylar blanket. The booms are hinged in this "bellyband."

The 62-in. radial booms, which isolate the orientation nozzle, wobble damper, and magnetometer from the spacecraft proper, are made from thin-walled aluminum tubing. During launch, a reefing line holds these booms in a stowed position around the antenna mast. Immediately after third-stage separation, redundant pyrotechnic cable cutters free the booms, allowing centrifugal force to spread them outward. Piston-type boom dampers, somewhat like those on heavy doors, prevent them from snapping into position too rapidly. A leaf-type spring and pawl lock the booms into position permanently.

The Stanford experiment antenna is similar to the radial booms in construction. However, it has two hinges: one controls the kinematics of the entire assembly, and the other permits the unfolding of the high frequency element of the antenna until it lies along the spin axis. Brackets on the magnetometer boom hold the Stanford antenna in its stowed position until the magnetometer boom has deployed about 40°. Microswitches on each boom indicate successful deployment and locking.

Other Structures Tests and Analysis

STL performed the stress analysis of the Pioneer spacecraft. The studies involved static analysis and examination of such dynamic factors as balance, moments of inertia, rigidities, limitations of vibratory response, spacecraft spinup and separation, and attitude stability and damping. The dynamics of appendage deployment were of particular concern because of past difficulties with booms. This concern led to a special test apparatus which was built to check deployment under close-to-actual conditions (see ch. 7). Spin tests, vibration tests, and the other related tests described in chapter 7 were extensive. They required the construction of a special "structural model" of the spacecraft, wherein the major structures either duplicated those in the intended flight model or, in the case of electronic equipment, simulated them in weight.

Structural reliability analysis is not as advanced as it is for electronics equipment; nevertheless, some estimates can be made. STL calculated that the overall structure reliability would be 0.998 for launch, boost, injection, and free flight. This estimate was based upon tests performed upon cable cutters and deployable booms built for OGO and other space programs. Of course, such estimates based on moving parts assume that no failures of static structural members occur. The use of factors of safety during the stress analysis gives this assumption some foundation. Pioneer structural studies assumed a yield factor of 1.35 and an ultimate safety factor of 1.50.

The possible effects of the space environment upon the spacecraft were also analyzed carefully. The analysis showed:

(1) Solar heat flux was controlled by the louvers and thermal coatings discussed previously under thermal control.

(2) Solar particulate radiation, which has a potential for degrading material structures, was several orders of magnitude below damage thresholds.

(3) Micrometeoroids in deep space (many times less prevalent than in Earth orbit) were deemed to be of negligible structural import to vehicles this side of the asteroid belt.

(4) Space vacuum may result in the sublimation of materials with high vapor pressures and the cold-welding of moving parts. Of the Pioneer structural materials, magnesium has the highest vapor pressure, but the loss over a year's time was computed to be negligible. The only spacecraft moving parts that must operate successfully throughout the mission are the thermal louvers. These are lubricated with a low-vapor-pressure solid grease developed for the OGO program.

OVERALL WEIGHT BREAKDOWN

The subsystem weight breakdown for the entire spacecraft is presented in table 4–18 for the two blocks of IQSY Pioneer spacecraft.

TABLE 4–18.—*Block-I and Block-II Spacecraft Weight Breakdowns*

Equipment	Block I [a]	Block II [b]
Communication subsystem	14.35 lb	14.33 lb
Receivers (2)	6.14	6.14
Transmitter driver	1.31	1.29
TWTs (2)	1.89	1.89
Attenuators and supports	0.12	0.12
Branch line coupler	0.24	0.25
Diplexers (2)	1.39	1.38
Bandpass filter	0.25	0.25
Coaxial switches (5)	1.00	1.00
High-gain antenna	2.01	2.01
Data handling subsystem	10.65	10.78
Digital telemetry unit	8.57	8.64
Data storage unit	1.73	1.75
Signal conditioner	0.35	0.39
Command subsystem	10.72	11.47
Command decoder	5.60	6.15
Command distribution unit	5.12	5.32
Electric power subsystem	36.54	38.03
Solar cells, substrate, glass, etc.	13.98	13.41
Support rings and brackets	2.96	2.96
Battery	2.19	3.16
Equipment converter	3.02	2.99
TWT converters (2)	4.52	4.49
Cabling and connectors	9.87	11.02
Orientation subsystem	6.68	6.95
Nitrogen bottles and supports	1.54	1.75
Nitrogen gas	0.87	0.93
Solenoid valve	0.44	0.40
Regulator	0.99	1.00
Nozzle	0.01	0.01
Pressure transducer	0.21	0.21
Sun sensors	0.86	0.87
Pressure switch	0.12	0.12
Plumbing and supports	0.46	0.46
Logic circuits	0.98	1.00
Fill valve	0.20	0.20
Thermal control subsystem	6.80	7.00
Louvers	0.48	0.48
Structure and actuators	1.64	1.64
Thermal insulation	4.68	4.88
Structure subsystem	17.46	18.10
Equipment platform	7.01	7.03
Interstage structure	0.99	0.99
Interstage support ring	0.18	0.18
Payload fitting	0.97	0.97
Antenna supports and fittings	1.03	1.03
Wobble damper	0.46	0.46
Booms (3)	1.47	1.57

TABLE 4–18.—*Block-I and Block-II Spacecraft Weight Breakdowns* (Continued)

Equipment	Block I [a]	Block II [b]
Magnetometer boom flange	0.05	0.07
Hinge fitting structure	1.50	1.50
Boom dampers (3)	0.57	0.57
Boom tie-down	0.79	0.79
Solar sail	0.06	0.06
Hardware	1.18	1.25
Inertia weights (on boom)	0.12	0.55
Platform struts	1.08	1.08
Total spacecraft weight without experiments	103.20	106.66
Experiments	34.74	40.54
Magnetometer (Goddard/Ames)	5.81	7.74
Cosmic ray detector (Chicago)	4.71	----------
Cosmic ray detector (GRCSW)	4.39	5.55
Plasma probe (MIT)	6.13	----------
Plasma probe (Ames)	6.33	5.92
Stanford radio propagation experiment [c]	7.37	7.01
Cosmic dust (Goddard)	----------	4.29
Cosmic ray (Minnesota)	----------	7.91
Electric field (TRW)	----------	0.87
Convolutional coder	----------	1.25
Total spacecraft weight with experiments	137.94	147.20 [d]

[a] Reported spacecraft weights vary slightly for each spacecraft depending upon the data source. The weights shown in this column are for Pioneer 6; taken from "Monthly Informal Technical Progress Report, Pioneer Spacecraft Program." Period 1 December to 31 December 1965, TRW Systems Report 8400.3–247, January 10, 1966.

[b] For Pioneer 9. Taken from "Monthly Informal Technical Progress Report, Pioneer Spacecraft Project." Period 1 November to 30 November 1968, TRW Systems Report, December 9, 1968.

[c] Includes balance compensation weights of 1.33 and 0.63 lb, respectively.

[d] Actual weight 146.82 lb when the assembled parts were weighed en masse.

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CHAPTER 5

Scientific Instruments

SCIENTIFIC OBJECTIVES

THE PIONEERS are multidisciplinary spacecraft. From the scientific standpoint they are very close relatives of the Interplanetary Monitoring Platforms (IMPs) orbited around the Earth and Moon from 1963 on. In fact, many IMP experimenters are also Pioneer experimenters, and their instruments are similar on both series of spacecraft. This is not surprising, as both types of spacecraft were designed to measure the same important facets of the interplanetary medium: the plasma, cosmic rays, magnetic fields, cosmic dust, electric fields, and space propagation properties. The IMPs, however, center on the Earth-Moon system, while the Pioneers are Sun-centered.

The scientific objectives of the Pioneer spacecraft are to measure the above-named facets of the interplanetary field. In 1962 virtually nothing was known of what transpired in interplanetary space. In particular, Earth-bound scientists had little feel for how plasma, cosmic rays, etc., varied spatially and in the time and energy dimensions. The Pioneer scientific objectives were sharpened in three ways:

(1) The spacecraft were launched at intervals that permitted the solar cycle to be covered from minimum to maximum. (The long lifetimes of the Pioneers has extended this coverage well beyond the 1969–1970 maximum.)

(2) Pioneers were launched on inward and outward missions so that some precede and some lag the Earth, giving scientists synoptic coverage over much of the plane of the ecliptic.

(3) The outward launches (Pioneers 7 and 8) were sent in backward-curving arcs that took them far out into the Earth's "tail" or geomagnetic wake (fig. 5–1). Thus, measurements were acquired in this poorly understood shadow zone; the zone is the subject of considerable controversy concerning its length and structure.

It is this extensive spatial and temporal coverage of the interplanetary medium that makes the Pioneers especially valuable scientifically. The scientific results and their interpretations are presented in Volume III. Only experimental hardware is covered in this chapter.

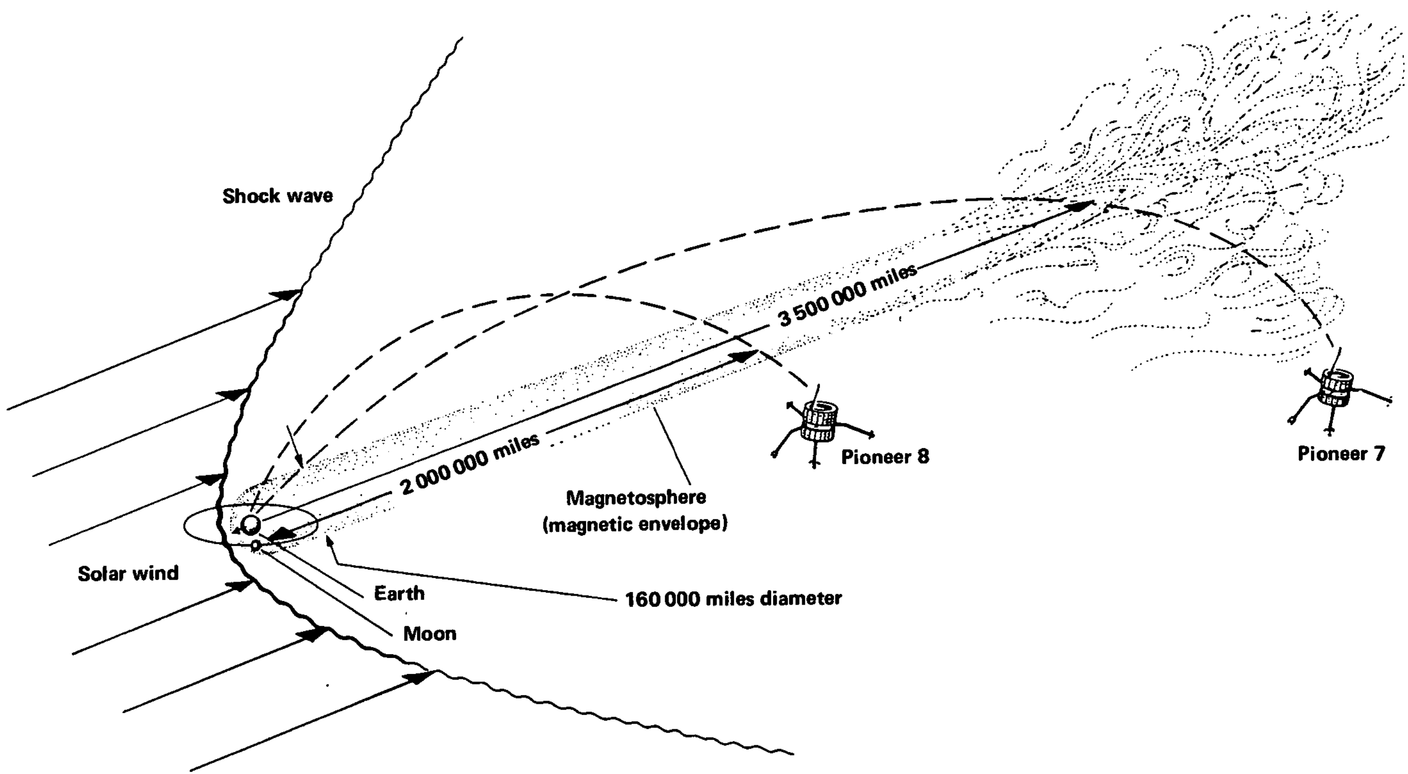


FIGURE 5-1.—Trajectories of Pioneers 7 and 8 through the Earth's magnetic tail.

APPLICATIONS OF PIONEER DATA

In 1962 and 1963, the Pioneers would hardly have been called "applications" spacecraft, so firmly were they directed toward satisfying scientific curiosity. Solar events, however, have wide repercussions, jiggling magnetometers on Earth, disrupting long-distance communication, and the like. Pioneers 8 and 9, cruising well behind the Earth in its path around the Sun, can radio warnings to the Earth of solar radiation storms which will soon catch up with the Earth. Since the Sun rotates once each 28 days and drags its plasma and radiation streams around with it, the Pioneers lagging the Earth are well-situated to forecast interplanetary weather for the Earth several days in advance. These data are now sent to the Environmental Science Services Administration, which then distributes them to about a thousand users (see Vol. III).

Thus, Pioneer instrumentation has practical applications not foreseen at the beginning of the program. On the later Pioneers, instrument selection and design were affected to some extent by this new dimension of the program.

INSTRUMENT INTERFACES AND SPECIFICATIONS

In Pioneer terminology, the scientific instruments are considered a separate system rather than a subsystem of the spacecraft. The forces exerted on the scientific instruments are considered to be similar to those encountered by the spacecraft subsystems. The most important are the mechanical loads imposed during launch, the heat from the Sun, the magnetic and electromagnetic environments extending from the other subsystems, and the information interface enforced by the data handling subsystem. These spacecraft subsystems are defined in detail in chapter 4.

To provide the experimenters with a view of the interfaces as seen by their instruments, Ames Research Center prepared a series of specifications and interface documents. The first, "Scientific Instrument Specification," No. A–7769, was issued December 31, 1963, for the purpose of acquainting experimenters with the spacecraft test requirements, the ground-support equipment, documentation requirements, and the responsibilities levied on the experimenters. A series of more detailed documents describing the spacecraft/scientific instrument interfaces followed. The reader should consult the references at the end of this volume for a complete list of Pioneer specifications. The interface documents were double-edged—both engineers and scientists working on the spacecraft could use them as definitive descriptions of the scientific instruments written in hardware language with dimensions, weights, electrical-connector-pin assignments, and the like, spelled out.

It is worth while to review the main points covered in the instrument interface specifications to reemphasize the extra dimensions involved in instrument design for space research. The following considerations were necessary:

(1) Mechanical interfaces—dimensions, weights, mounting orientations and view angles

(2) Electrical interfaces—power levels, voltages, transients, connectors and cabling

(3) Information interfaces—word lengths, bit rates, formats, and timing signals

(4) Thermal interfaces—operating temperatures and surface coatings

(5) Electromagnetic interfaces—interference, shielding and grounding

The factors mentioned above are discussed in chapter 4.

In addition to matching spacecraft interfaces, each scientific instrument had to mesh with the interfaces presented by ground-support equipment. Before even reaching the launch pad, instruments had to be qualified (usually through prior flights on sounding rockets or satellites) and then tested according to the standards described in chapter 6.

A number of military specifications were also applied to the Pioneer spacecraft and its cargo of instruments. One of the most critical was MIL–I–26600, "Interference Control Requirements, Aeronautical Equipment," highlighting the fact that the electromagnetic environment had to be shared with other spacecraft as well as a host of other aerospace equipment at Cape Kennedy prior to and during launch. Finally, the presence of radioactive sources for instrument calibration meant that federal and state laws governing the use and transport of radioactive materials also applied.

The scientists flying instruments on Pioneer (or any spacecraft, for that matter) had to deal with managerial controls, with specifications more restrictive than those encountered in the terrestrial laboratory, and with rather rigorous testing and qualification regimens. The specific details may be found in the other chapters referenced above and in the documents listed at the end of this chapter (refs. 1 and 2).

INSTRUMENT SELECTION

The instruments selected for the Pioneer flights had to promote the mission's scientific objectives, as well as match spacecraft interfaces and meet management criteria such as the cost and schedule limitations set forth in chapter 1. NASA has a well-defined procedure for choosing experiments and experimenters. When a mission has been delineated well enough to permit some hard thinking about experiments, NASA solicits the scientific community by letter, telegram, or (more commonly today) a solicitation entitled "Opportunities to Participate in Space Flight Experiments." Experiments for Pioneers A and B were solicited by letter in early 1963; for Pioneers C and D, in late 1963 and early 1964.

When experiment proposals have been received, they are evaluated at NASA Headquarters with the assistance of the Space Sciences Steering Committee. The members of the Committee and its several subcommittees are appointed from scientists in NASA, other Government agencies, universities, and non-profit organizations. In the case of Pioneer, the following four subcommittees were involved: Astronomy, Solar Physics, Ionospheres and Radio Physics, and Particles and Fields. The Pioneer Project Office also reviewed experiment proposals from the standpoints of engineering feasibility, cost, and compatibility with the spacecraft.

Usually NASA receives more proposals for experiments than the spacecraft can carry. Therefore, the Space Sciences Steering Committee must choose those experiments that meet the minimum requirements and then assign priorities. For example, 18 proposals were evaluated in depth for Pioneers A and B, 15 for Pioneers C and D; but only 7 and 8 experiment flew on these two blocks of spacecraft, respectively. The general criteria employed in the selection process are: (1) scientific merit, (2) ability of the instrument to make the desired measurement, (3) development status of the instrument, and (4) understanding and experience of the experimenters. The criteria specific to Pioneer that were employed are: (1) pertinence to Pioneer mission, weight, data rate, power, etc.; and (2) pertinence with respect to the solar minimum.

The Summary Minutes of the meeting of the Space Sciences Steering Committee, dated July 22, 1963, typify the selection procedure. Pioneer A and B experiments were divided into two categories as follows:

(1) Firm payload, including magnetic fields, plasma, cosmic-ray gradients, and radio propagation

(2) Tentative or backup experiments, including cosmic-ray anisotropy and plasma

It was also recommended that the cosmic-ray anisotropy experiment be rocket-tested prior to the Pioneer launch. Within the "firm payload" group, the radio propagation experiment was given the lowest priority should subsequent spacecraft and instrument developments require weight reduction.

The above list for Pioneers A and B includes no cosmic dust or micrometeoroid experiment. A cosmic dust experiment was proposed for the Block-I Pioneers by Ames Research Center, but development problems precluded its inclusion. Thus, one of the major parameters of interplanetary space had to be neglected on the early Block-I flights. When experiments were solicited for Pioneers C and D, no cosmic dust proposals were received. Trying to make up for this deficiency, NASA specifically solicited several scientists by telegram, asking if they would be interested in building cosmic-dust experiments for the Pioneer interplanetary mission. Three proposals were received; and ultimately the one submitted by Goddard Space Flight Center was selected for the Pioneer C and D payloads.

When it was decided to combine the parts left over from Pioneers A through D and assemble Pioneer E, the question of experiment selection was revived. During the fall of 1965, however, NASA decided to retain the Pioneer C and D payload rather than making extensive modifications to the spacecraft parts on hand.

Some of the proposed experiments did not fall within the mission scope suggested by NASA. For example, a proposal submitted by Space/Defense Corporation was aimed at investigating the influence of electromagnetic and gravitational fields on diurnal rhythm. Interesting as such an experiment would have been, it would not have measured parameters related to the other investigations.

The experiments finally selected for the five Pioneer spacecraft are presented in table 5–1.

THE GODDARD MAGNETOMETER (PIONEERS 6, 7, AND 8)

The interplanetary magnetic field is created by the Sun and modulated by the streams of plasma that spiral out into the space between the planets. Magnetic field measurements, particularly those that record transients following solar activity, are critical to our understanding of the space surrounding the Sun.

The spin-stabilized Pioneers permitted the use of a unique magnetometer design whereby all three components of the magnetic field could be measured with a single-axis sensor (ref. 3). If the sensor axis is mounted at an angle of 54°45′ to the spin axis, and if the sensor is sampled at three equally spaced intervals during the rotation of the spacecraft, the experimenter receives three independent measurements of three orthogonal components of the magnetic field.

The sensor of the single-axis fluxgate magnetometer employed in the Goddard experiment is a saturable inductance driven by a gating magnetic field applied in a winding. The flux induced in the saturable core is modified by the presence of the external magnetic field in such a way that the contribution of the external field can easily be extracted and quantified.[16] The Pioneer magnetometers were developed and manufactured for Goddard by the Schonstedt Instrument Company.

The fluxgate sensor is mounted on one of Pioneer's three booms, at a distance of 2.1 m from the spin axis, in a canister employing passive thermal control. An unusual feature of this experiment is the explosive-actuated indexing device, which permits the experimenter back on Earth to flip the sensor over by 180° so that magnetic fields created by the space-

[16] For a more complete description of how the various instruments used in space science work, see: W. R. Corliss, Scientific Satellites, NASA SP–133, 1967.

TABLE 5–1.—*Experiments Aboard the Pioneers*

Instrument	Pioneer spacecraft				
	6	7	8	9	E
Single-axis fluxgate magnetometer [a]	X	X	X		
Triaxial fluxgate magnetometer [a]				X	X
Faraday-cup plasma probe	X	X			
Plasma analyzer	X	X	X	X	X
Cosmic ray telescope	X	X			
Cosmic-ray anisotropy detector	X	X	X	X	X
Cosmic-ray gradient detector			X	X	X
Radio propagation experiment	X	X	X	X	X
Electric-field detector			X	X	X
Cosmic dust detector			X	X	X
Celestial mechanics	X	X	X	X	X

[a] The triaxial fluxgate magnetometer was originally scheduled to fly on Pioneers C and D, but because it could not make the launch date, the single-axis fluxgate magnetometer was substituted on Pioneer C.

craft can be taken into account. The Goddard sensor is rotated by a spring-driven escapement mechanism. Because of their high reliability, eleven pairs of small explosive charges were used to activate the escapement mechanism. Thus, eleven sensor flip-overs are possible by remote control.

The spacecraft Sun sensor triggered the Goddard experiment once each rotation of the spacecraft. Beginning with this signal, the experiment electronics generated three equally spaced sampling gates which permitted analog readings from the fluxgates to enter analog/digital converters. These data were then converted into digital words. Each magnetometer measurement required eight bits. Measurements were stored in a 24-bit data buffer until all three measurements were completed; then, they were transferred to the spacecraft data handling subsystem (fig. 5–2). As noted in chapter 4, the Goddard telemetry word is longer than the standard Pioneer 6-bit word. Consequently, four spacecraft words were required to transmit three Goddard experiment words.

Another important component of the experiment was the analog/digital converter, which converted the analog voltage measurements provided by the sensor into eight-bit words. The time-average computer, also shown in figure 5–2, averaged sensor readings during those periods of the mission when the spacecraft was far from Earth and the telemetry rates were less than the rate at which data accumulated from the experiment.

Magnetic interference is a critical problem for the experimenter flying a magnetometer in interplanetary space. The fields are usually less than 10γ and may be overwhelmed by the fields generated by the spacecraft.

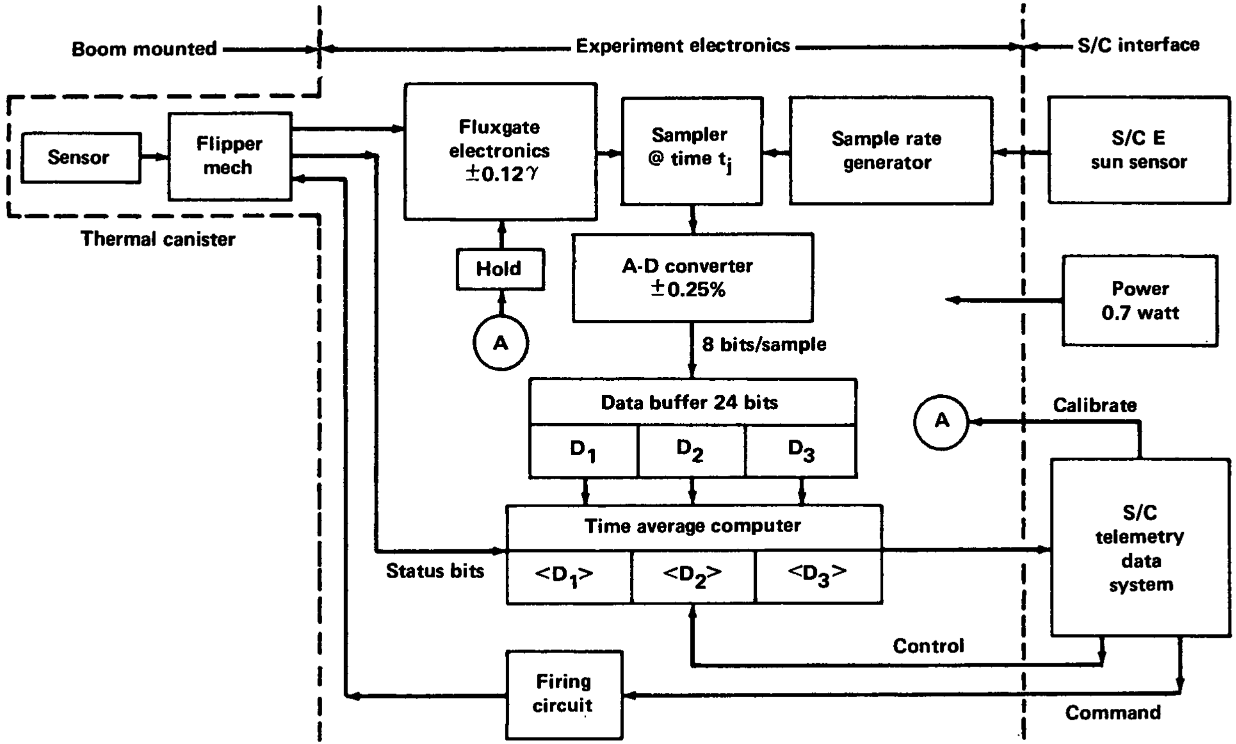


FIGURE 5-2.—Block diagram of the Goddard magnetic field experiment.

For this reason, the Pioneers were made as magnetically clean as possible (ch. 4), and the magnetometer sensor was located on a spacecraft boom 2.1 m from the spacecraft spin axis. Detailed mapping indicated that the magnetic interference from the spacecraft was less than 0.125γ, 0.35γ, and 0.2γ on Pioneers 6, 7, and 8, respectively. The Pioneers were among the magnetically cleanest spacecraft ever built. The data telemetered to Earth have, therefore, been of great utility in mapping the magnetic structure of solar disturbances and, during the first few hours of flight, the Earth's magnetic tail.

The overall characteristics of the Goddard magnetometer are tabulated in table 5-2. Originally, the magnetometer was to be located at the end of the axial high-gain telemetry antenna, but this proved impractical and it was mounted on a boom (fig. 1-4). The location of the experiment electronics on the spacecraft equipment platform is shown in figure 4-29.

THE AMES MAGNETOMETER (PIONEERS 9 AND E)

The Ames magnetometer instrumentation consists of a fluxgate-sensor package located at the end of one of the 62-in. spacecraft booms and an electronics package mounted on the spacecraft equipment platform. Like the Goddard magnetometer, the Ames instrument is based on the fluxgate saturable inductance sensor; but it employed three sensors mounted along

TABLE 5.2.—*Characteristics of the Goddard Magnetometers*

Characteristic	Pioneers		
	6	7	8
Weight			
Electronic assembly	4.5 lb	4.5 lb	5.0 lb
Boom assembly	0.7 lb	0.7 lb	0.7 lb
Power required	0.7 W	0.7 W	0.9 W
Input voltage (spacecraft bus)	28^{+5}_{-4} V dc	28^{+5}_{-4} V dc	28^{+5}_{-4} V dc
Range	±64γ	±32γ	±32γ [a] ±96γ
Thermal calibration			
Electronics		−25° C to +55° C	
Sensor		−75° C to +75° C	
Resolution (sensitivity)	±0.25γ	±0.125γ	±0.125γ ±0.375γ

[a] An automatic switch was added on Pioneer 8. High magnetic fields switched the magnetometer to the higher range.

mutually orthogonal axes rather than a single sensor like the Goddard instrument. One fluxgate axis is parallel to the spacecraft spin axis and a second oriented radially. The Ames experimenters hoped that their three-axis magnetometer would provide a better measure of the interplanetary magnetic field during disturbances involving large, rapid magnetic fluctuations.

The permalloy-core fluxgate sensors were built by Honeywell, Inc., and supplied to Philco-Ford, the magnetometer prime contractor, as Government-furnished equipment (ref. 4). The general construction of the basic sensor is portrayed in figure 5–3. A 6144-Hz drive signal of approximately 1.5 V rms applied to the toroidally wound drive winding modulates the permeability of the sense-winding core. The sense winding provides the signal that indicates the direction and strength of the ambient magnetic field. The feedback winding generates a signal that helps to minimize nonlinearities. The three sensors comprise two packages: one single-axis fluxgate is located in a package mounted so that the sensor axis is parallel to the spacecraft boom axis; the second package contains two orthogonally mounted fluxgates with both axes perpendicular to the boom axis. The Ames instrument includes a flipping mechanism, but it is powered by two resistance-heated bimetallic motors rather than a pyrotechnic device, such as that used by the Goddard magnetometer. The motors on the Ames instrument flip the dual sensor assembly 90° upon command from Earth. One motor flips the sensors clockwise; the other counterclockwise. The Ames magnetometer sensors can be flipped again and again and are not limited to the number of pyrotechnic charges launched with the spacecraft (the Goddard instrument has 11 flips); however, an additional burden is placed upon the spacecraft power supply by the resistance heaters in the motors.

The electronics package (fig. 5–4) has these major functions:

(1) Provision of the 6144-Hz drive signals, which must have a negligible second-harmonic content

(2) Selection, amplification, and demodulation of the second harmonic signal obtained from the sense windings

(3) Analog-to-digital conversion of the analog signals from the sense winding

(4) Spin demodulation of the signals received from the two sensors in the spacecraft spin plane

(5) Digital filtering to match the five allowable spacecraft bit rates

(6) Periodic sampling of data in the buffer storage

Once safely in space, the Ames instrument is commanded each day into a self-calibrate sequence. Sinusoidal currents are injected into the feedback windings to establish calibration fields. The calibration sequence is often repeated after the dual-sensor package has been commanded into the flipped position. This interchange of sensor positions, of course, permits

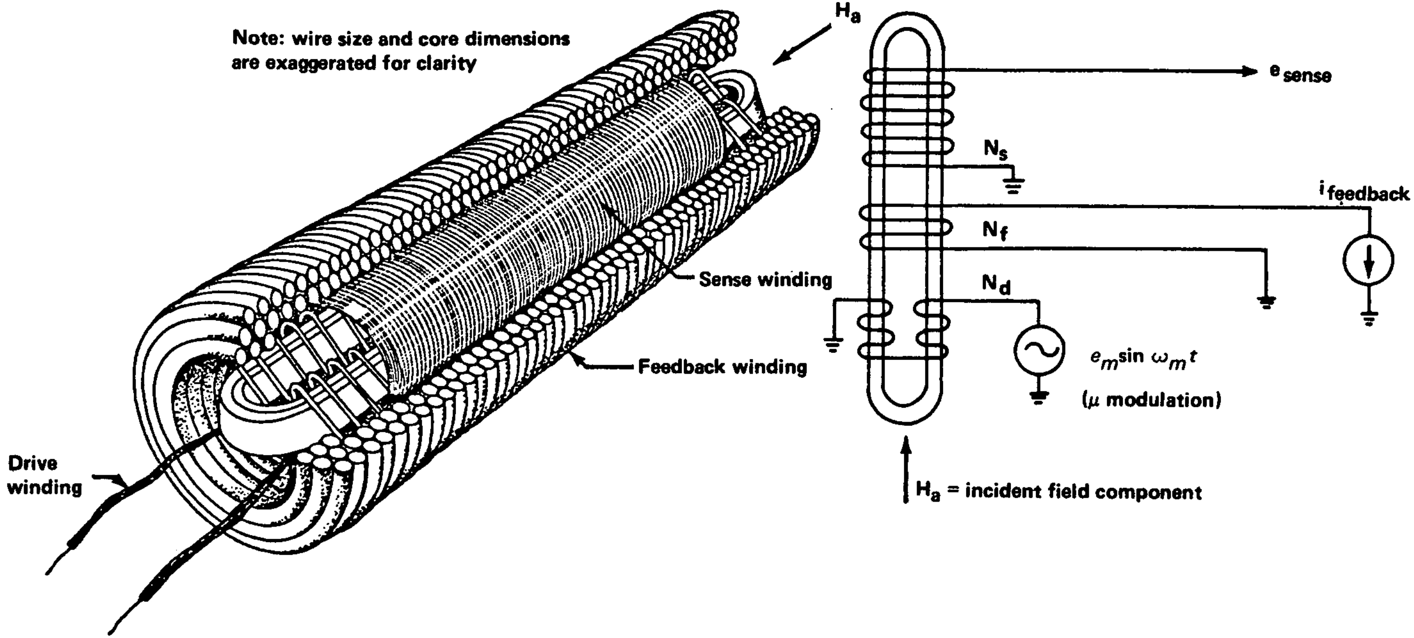


FIGURE 5-3.—Construction of the Ames fluxgate sensor.

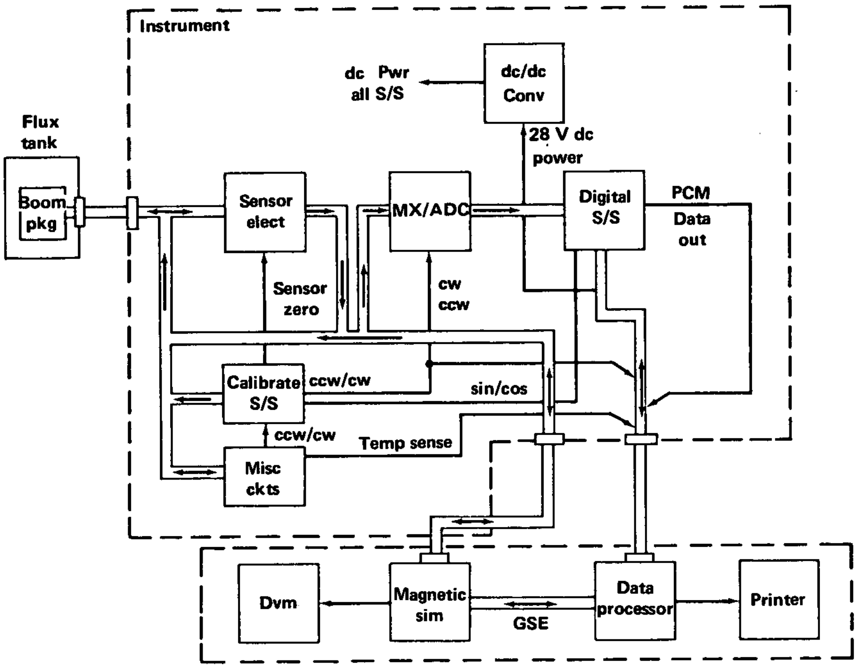


FIGURE 5-4.—Simplified block diagram of the Ames fluxgate magnetometer.

periodic measurement of the zero level of the sensor aligned with the spacecraft spin axis.

The overall instrument parameters—as originally specified and ultimately attained—are presented in table 5-3. The table indicates a change in dynamic range from $\pm 200\gamma$ to $\pm 50\gamma$ after it became apparent from deep space measurements from the Mariners and Block-I Pioneers that $\pm 50\gamma$ was more than adequate.

The Ames magnetometer was designed and fabricated by Philco-Ford's Space and Reentry Systems Division. The contract was awarded by NASA in December 1965. The memory system was procured from Electronic Memories, Inc.; the fluxgate sensors were supplied through NASA from Honeywell, Inc. The initial program goal was the provision of a flight-qualified instrument for Pioneer 8 but this date could not be met. As a consequence, the Goddard magnetometer flew on Pioneer 8, and the Ames instrument was deferred to Pioneer 9.

MIT FARADAY-CUP PLASMA PROBE (PIONEERS 6 AND 7)

By 1965, plasma probes flown on several Earth satellites and planetary probes had confirmed that the interplanetary plasma originates in the Sun's corona and flows outward toward the planets at about 300 km/sec,

TABLE 5–3.—*Ames Magnetometer Specifications*

Parameter	Specification	Design value	Realized parameters: Prototype	Realized parameters: Flight unit no. 1	Realized parameters: Flight unit no. 2
Weight (lb)					
Boom package	0.7	0.85 (±0.1)	0.815	-------	0.837
Instrument	4.6	6.8 (±0.3)	5.85	5.79	5.87
Power requirement (W)					
Normal	3.0	5.6	5.72	5.5	5.3
Calibrate	----------	8.0	9.16	8.7	8.65
Flip calibrate	6.36	8.3	8.24	8.3	7.8
Reliability prediction	0.75 @10 000 hr				
Instrument dynamic range					
Prototype and flight no. 2	± 50γ	±50γ			
Flight no. 1	±200γ	±200γ	±50γ	±200γ	±50γ
Resolution					
Prototype and flight no. 2	±0.05γ	±0.05γ	±0.05γ	±0.2γ	±0.05γ
Flight no. 1	±0.02γ	±0.2γ			
Repeatability	±0.2γ	±0.1γ	±0.2γ	±0.2γ	±0.1γ
Accuracy (percent)	±0.35	±0.35	±0.5	±0.5	±0.35
DC offset	±0.2γ	----------	0.3γ	±0.4γ	±0.4γ
Step response (percent)	<1	----------	1	1	<1
Cross coupling	<2γ	----------	<2γ	<2γ	<2γ

remaining ionized out to several AU. Further, this plasma is electrically conducting and interacts in complex ways with solar and planetary magnetic fields. The scientific objective of the MIT plasma probes was to measure the following characteristics of this interplanetary plasma:

(1) Positive ion flux as a function of energy and direction
(2) Electron flux as a function of energy and direction
(3) The temporal and spatial variations of the above physical quantities
(4) Correlation of plasma measurements with magnetic field measurements

MIT scientists had flown Faraday-cup plasma probes on the IMP and OGO series of Earth satellites prior to the Pioneer 6 and 7 flights. The Pioneer instruments were basically similar to these flight-proven plasma probes. The Pioneer sensors, the Faraday cups, are 6 in. in diameter, with the open sides normal to the spacecraft spin axis so that they sweep out the plane of the ecliptic as the spacecraft spin (ref. 5). At the bottom of the cup, two halves of a split collector intercept those electrons and positive ions that are able to pass through a modulator grid which electrically sorts out the particles in the external plasma according to species and energy. The split in the collector is parallel to the spacecraft equatorial plane to provide directional information about the plasma fluxes in the meridian plane.

The energy spectra of the plasma ions and electrons are measured by applying square waves at different voltage amplitudes to the modulator grid directly in front of the split collector. For example, an 1800-Hz square wave varying between V_1 and V_2 admits only those particles in the plasma with energies between V_1 and V_2 electron volts. Further, the square wave modulates the stream of particles impinging on the collectors so that the currents collected and resultant signals delivered to the electronics section of the experiment varies at 1800 Hz, a signal that can be amplified and filtered conveniently. The amplitude of the square wave is varied between 100 and 10 000 V in 14 contiguous intervals to scan the positive ion spectrum and between 100 and 2000 V in four intervals for the electron spectrum.

The instrument's sensor is sampled once during each of the 32 equal 11.25° angular increments that the Faraday-cup sees in one complete rotation of the spacecraft. Since the interplanetary plasma flows outward from the Sun, samples from the eight segments within ±45° of the Sun line are always used to make up a data frame. Five additional samples taken during a spacecraft revolution complete the 13 data samples that comprise the "fine" measurements. Each of these represents the highest flux measurement from the four (11.25°) segments in each of the five (45°) sectors comprising the remainder of the complete rotation. Although all 32 (11.25°) segments are examined during each rotation, only 13 plasma measurements are recorded. The 13 samples are then digitized as six-bit words and stored temporarily in a core memory with a 256-word capacity.

During each complete spacecraft revolution, the square-wave amplitude is held constant. Then, the entire sampling procedure is repeated—on alternate revolutions of the spacecraft—for another square-wave amplitude, until all of the 14 positive ion and one of the four electron energy groups have been scanned at all azimuths. Instrument calibration and engineering data are placed in the core during a sixteenth revolution. During the other sixteen interlaced revolutions, no data are taken; rather, the square-wave amplitude is changed during these revolutions. Thus, a complete data "mode" requires 32 spacecraft revolutions. All 14 positive ion-energy groups are scanned each revolution, but the four electron-energy groups are subcommutated, with a different group being scanned during each data-taking revolution (fig. 5–5).

During the sampling procedure described above, the currents measured by the two halves of the collector are summed to make the basic data words. These words represent the instrument's "fine" data, but, because

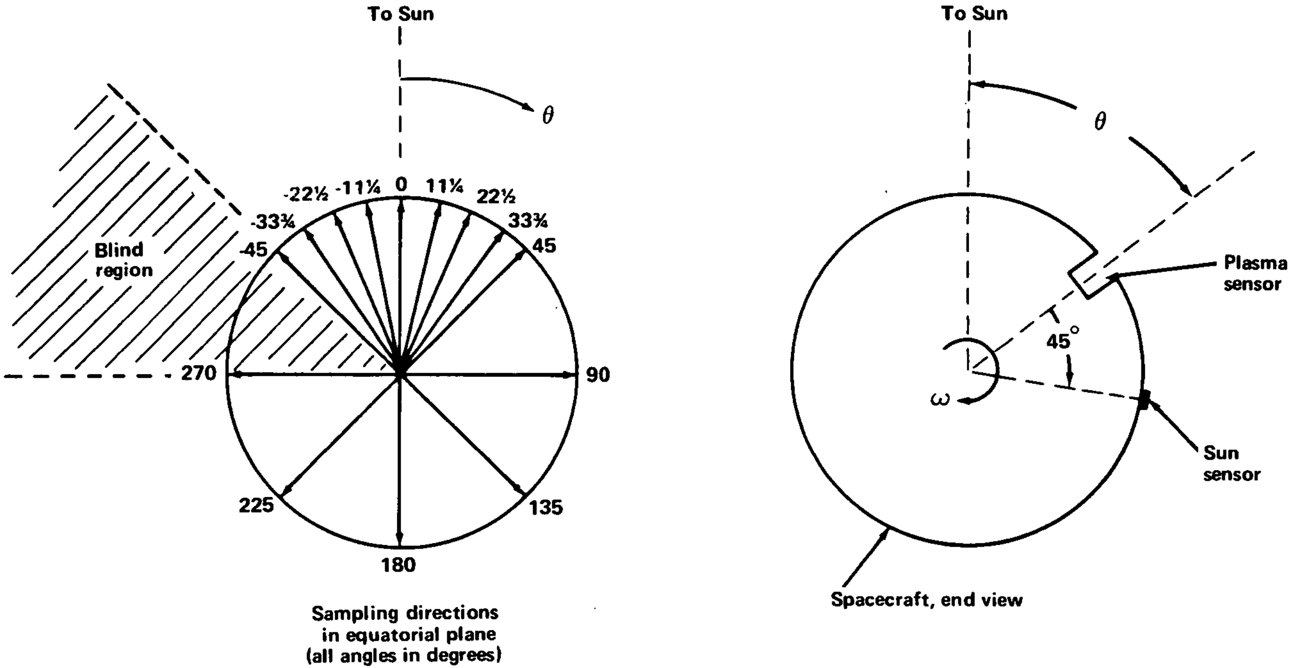


FIGURE 5–5.—Sampling interval of the MIT plasma probe.

of the summing operation, fine data indicate plasma characteristics only in the spacecraft equatorial plane. "Coarse" data are obtained by processing the current collected by a single half of the split collector. The largest of the 32 measurements taken from this collector during a spacecraft revolution comprises the coarse data word. Comparison of coarse and fine data words yields a measure of plasma direction in the spacecraft meridian plane; that is, the shadowing effect of the Faraday-cup walls combined with the view angles of the split collectors permits some coarse resolution of plasma fluxes arriving from above and below the plane of the ecliptic.

To summarize, a complete data mode of 256 words consists of 15 frames made up of 13 fine data words, 1 coarse data word, 1 coarse angle word (indicating which of the 32 angular segments provided the coarse data word) and 1 high voltage calibration word. The sixteenth frame comprises 16 words of engineering data. When the 256-word memory is completely filled—as it is at the end of a data mode—power is shut off to portions of the electronics, and the entire memory is read out to the spacecraft data handling subsystem. Then a new mode begins. The power drain of the MIT plasma probe therefore varies in a cyclic fashion, as mentioned in the last chapter.

The block diagram of the electronic circuits required to accomplish the experiment's formidable switching tasks is shown in figure 5–6.

Angular resolution of the MIT instrument is better than 5° in the equatorial plane of the spacecraft, which is approximately parallel to the plane of the ecliptic. The useful flux range is between 10^6 and 4×10^9 singly charged particles per cm^2 per sec.

AMES PLASMA PROBE (PIONEERS 6, 7, 8, 9, AND E)

When the angular distributions of the ions and electrons comprising the interplanetary plasma are not well known, the response of the Faraday-cup probe is often hard to interpret. The so-called curved-surface electrostatic plasma analyzers provide more detail, but they are correspondingly more complex. Plasma analyzers work on a different principle. They separate the plasma components into different energy-per-unit-charge (E/q) groups and also into much smaller solid angles. In other words, their E/q and solid-angle discriminations are better.

The theory of operation of the curved-surface plasma analyzers has been described in detail in other publications. They work by the application of stepped voltages to a pair of curved plates (fig. 5–7). Positively charged particles in the plasma are deflected toward one plate, negatively charged particles toward the other. Depending upon the voltage difference across the plates, only those particles within a narrow range of energy-to-charge ratio and within a narrow solid angle will reach the particle collector at the end of the curved plates. In effect, the curved plates form a filter that

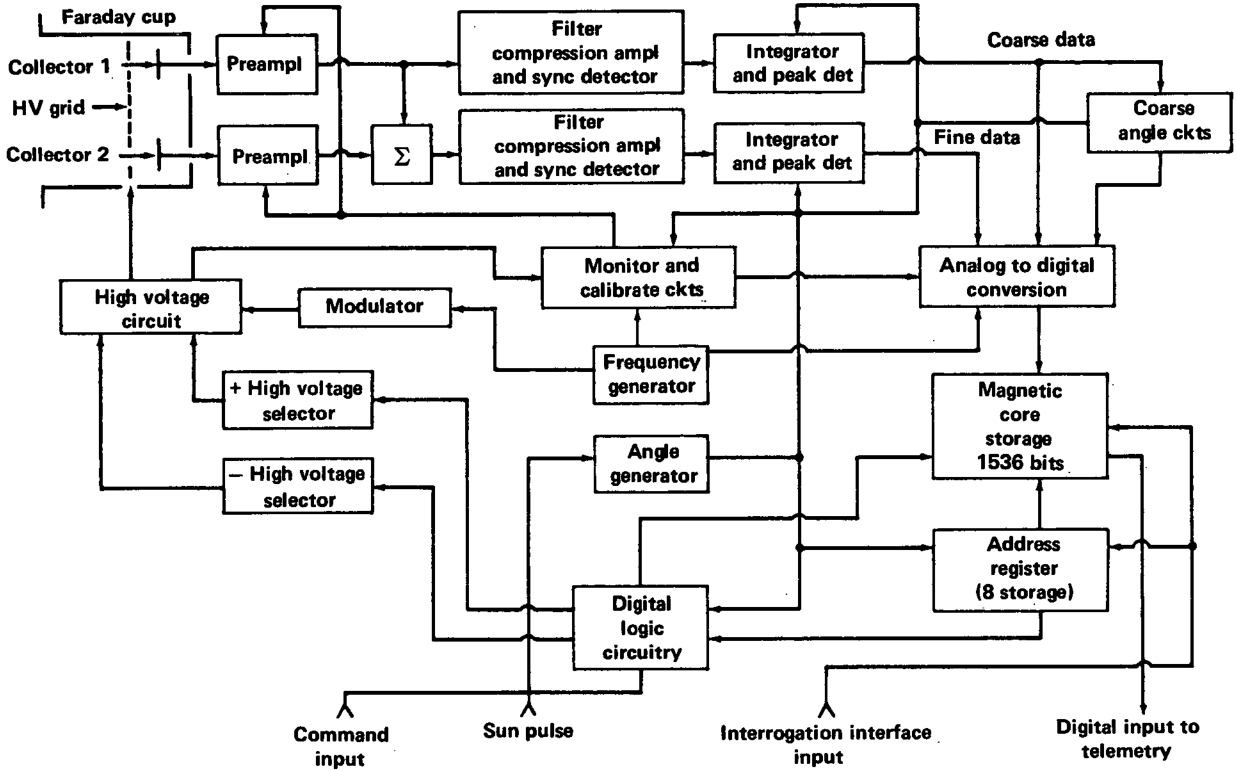


FIGURE 5–6.—Block diagram of the MIT plasma probe.

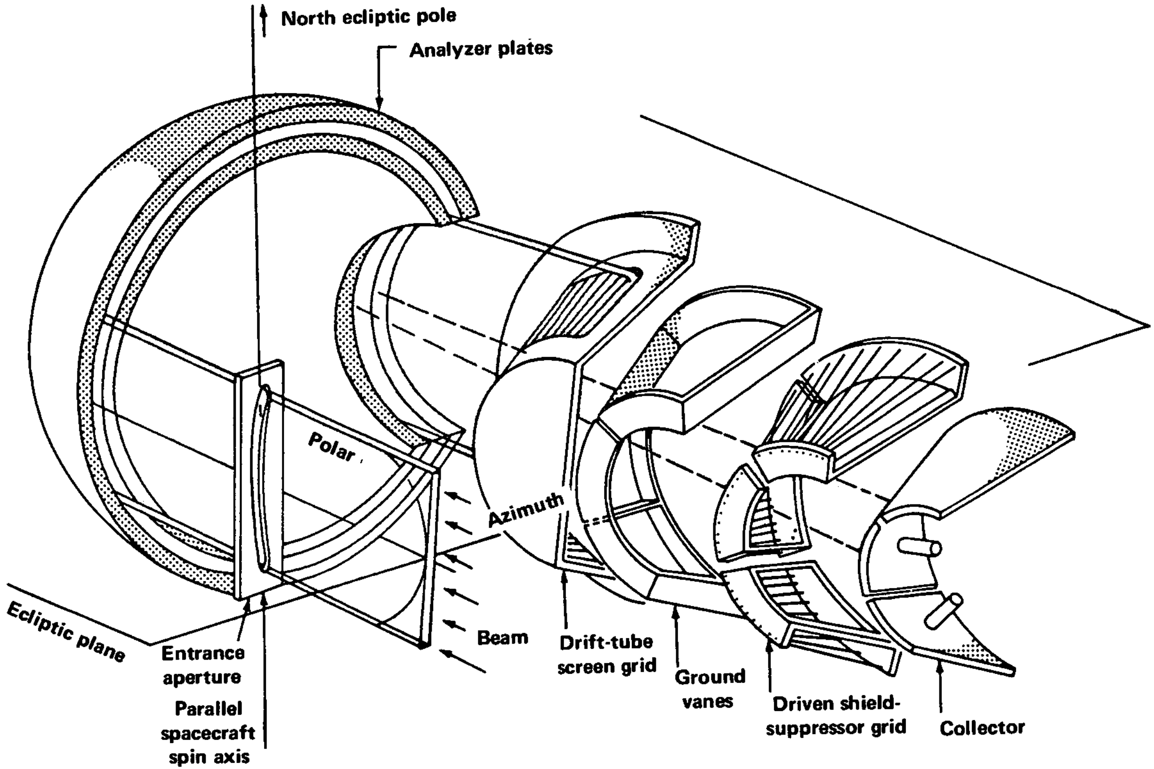


FIGURE 5–7.—Arrangement of the electron optics in the Block-II Ames plasma probe.

passes only a certain range of energy-to-charge ratios. If the plates are made portions of spherical surfaces and the collectors are segmented, the plasma flux arriving from different directions can be analyzed. If the applied voltages are stepped, energy-to-charge spectrum scanning is possible. Through the use of a mass spectrometer as the particle detector, particles with different masses, but within the same energy-to-charge-ratio group, can be extracted, but this was not part of the Pioneer experiment.

When the Pioneer payload was being formulated, the Ames Research Center group had flown quadrispherical plasma analyzers on OGO 1 and the IMP Earth satellites and was therefore in a good position to propose the similar instruments for the Pioneer series.

Although their basic principles of operation were the same, the plasma analyzers flown on the Block-I Pioneer spacecraft were significantly different from those on Block-II spacecraft. These differences are summarized below:

Block-I Instruments	Block-II Instruments
Quadrispherical plates	Truncated hemispherical plates
8 current collectors	3 current collectors
16 positive ion groups between 200 and 10 000 eV	30 positive ion groups between 150 and 15 000 eV
8 electron groups between 0 and 500 eV	14 electron groups between 12 and 1000 eV

Block-I Instruments

An entrance slit on the instrument face, figure 5–8, permits electrons and positive ions in the interplanetary plasma to pass into the space between the quadrispherical plates. The amplitude and phase of the voltage applied to these plates are varied through 24 steps, dividing the plasma into the 16 positive ion and eight electron groups mentioned above. The ions and electrons that pass all the way around the curved plates are collected by eight identical targets located in a semicircle at the lower end of the quadrisphere. Each of the targets is connected to an electrometer amplifier. Currents detected by the instrument vary between 10^{-14} and 10^{-10} amperes. The magnitudes of the currents measured, when combined with the knowledge of the instantaneous position of the entrance slit with respect to the Sun line as the spacecraft spins, determine the angular distribution of the interplanetary plasma flux.

The basic timing signals for the instrument are the Sun pulses, which arrive approximately once per sec. A sector programmer locks onto the Sun pulse frequency and divides the interval between the pulses into 1024 equal segments. Sector pulses are then generated that divide the azimuthal

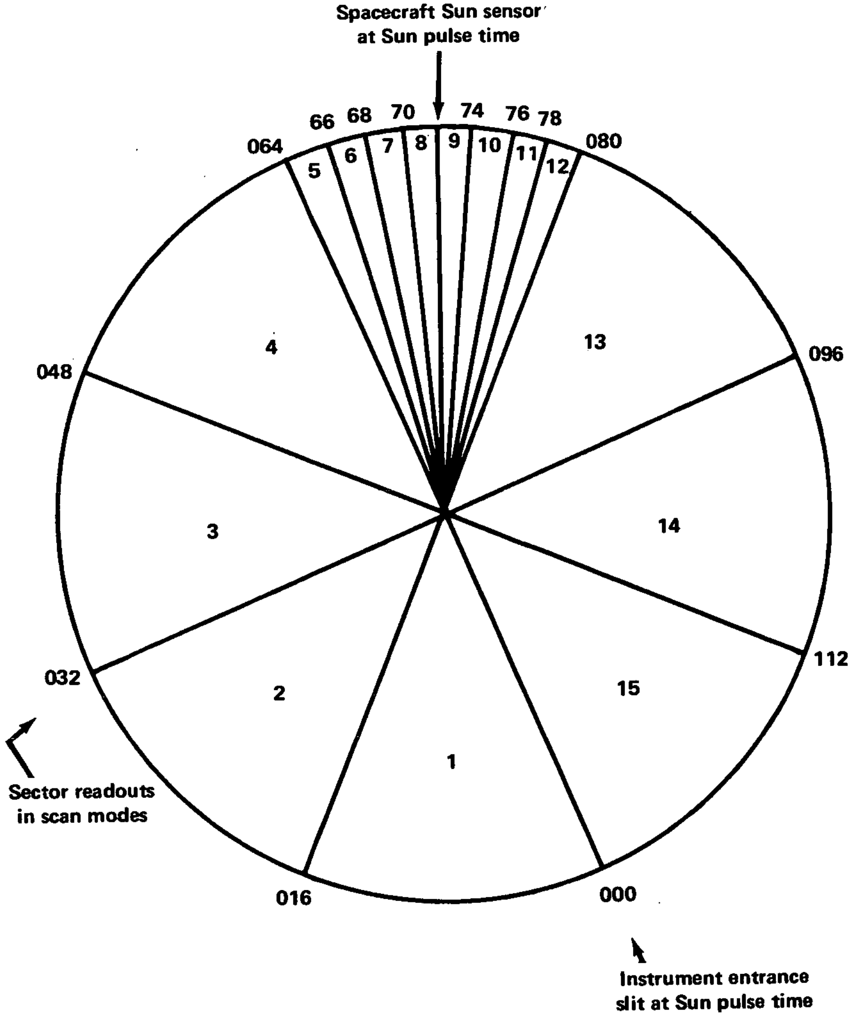


FIGURE 5-8.—Sector division and orientation for the Block-I Ames plasma probe.

plane into 128 equal segments and ultimately into the 15 sectors diagrammed in figure 5-8.

The Block-I instrument is capable of three separate modes of operation:

(1) Full Scan (FSM)—All 15 sectors are scanned during one spacecraft revolution at a specific energy step, for one specific collector. The second scan is at the same energy step and same collector but in the MFM mode described below. For the third scan the energy step is changed. The fourth scan is another MFM scan. This is repeated until all 24 energy steps have been utilized. Then, the process begins over again but for a different one of the eight collectors. This continues until all eight collectors (or channels)

have been measured for all 24 energy steps, in all 15 sectors. This mode is employed when the bit rate is 512 bits/sec. Spacecraft format A is used.

(2) Short Scan (SSM)—This mode is identical to the FSM mode except that only the eight small selectors clustered around the Sun line, as shown in figure 5–8, are read out. SSM is used at 256 bits/sec, format A.

(3) Maximum Flux (MFM)—In this mode, the eight collectors or channels are not scanned in sequence. The measurement read out in this mode is the maximum flux measured during each spacecraft revolution. The specific sector and channel where this measurement was made are identified in the telemetry. As with the other modes the plate voltage remains the same during each spacecraft revolution and is stepped when a new revolution commences. MFM is used at the 8, 16, and 64 bits/sec rates, format B.

Block-II Instruments

The Block-II instrument configuration is basically hemispherical, with the entrance aperture defined by a slit, as shown in figure 5–7. A series of grids and ground vanes are interposed between the analyzer plates and the three current-collectors, which are located so that they can monitor particles arriving from above and below as well as from along the plane of the ecliptic. Fluxes within $\pm 80°$ of the plane of the ecliptic can be monitored. The detectable range of current is 10^{-14} to 10^{-9} amperes.

A sector programmer once again is employed to divide the time between Sun pulses into equal parts; 2048 timing intervals are used to divide the azimuthal plane into 128 equal sectors. During one of the scan modes, the 23 shaded sectors shown in figure 5–9 are selected for current measurements. Most of these favored sectors lie near the Sun line. A comparison of figure 5–8 with figure 5–9 shows the differences in sector widths, especially away from the Sun line.

The types of operation for the Block-II instrument differed markedly from those of Block I:

(1) Polar Scan (PS)—The instrument identifies the sector at which the flux amplitude is maximum during each spacecraft rotation. The fluxes are measured for each of the three collectors at this sector.

(2) Azimuthal Scan (AS)—Only the flux reaching the center collector is measured and, then, only at the 23 sectors defined in figure 5–9.

(3) Maximum Flux Scan (MFS)—Only the maximum flux seen by the center collector and the sector during which it occurs is measured.

When the spacecraft is close to Earth and the two highest bit rates can be used (512 and 256 bits/sec), polar and azimuthal scans are made alternately. One PS and one AS is made for each of the 30 high-voltage steps in each positive ion subcycle; 15 steps are used in the electron subcycle. A complete high-bit-rate cycle consists of seven ion subcycles followed by one electron subcycle.

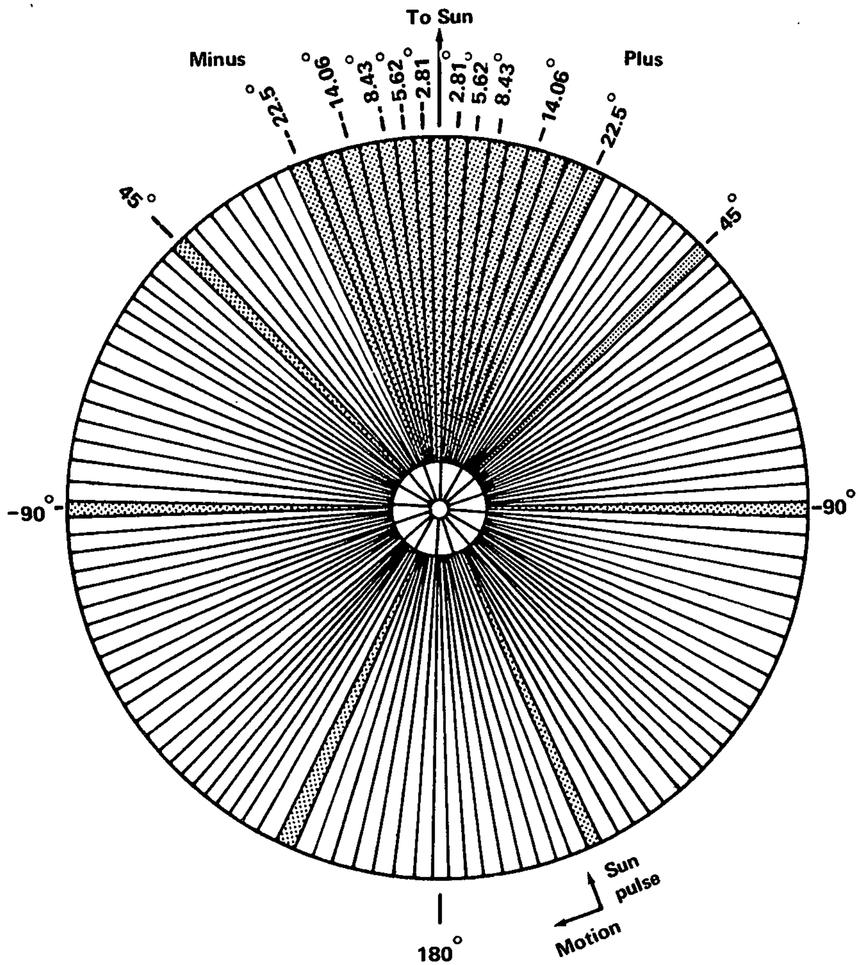


FIGURE 5-9.—Sector division and orientation for the Block-II Ames plasma probe.

At low bit rates (8, 16, and 64 bits/sec), an MFS is made for each of the 30 positive ion steps. Then, the voltage is set at the step where the maximum flux was detected, and one PS and one AS are made. In the electron mode, an MFS is made for each of the 15 levels followed by a PS and an AS as the voltage level of the seventh step. A complete low-bit-rate cycle also consists of seven ion subcycles followed by one electron subcycle.

A functional block diagram of the Block-II plasma probe is presented in figure 5-10 and the external view is shown in figure 5-11.

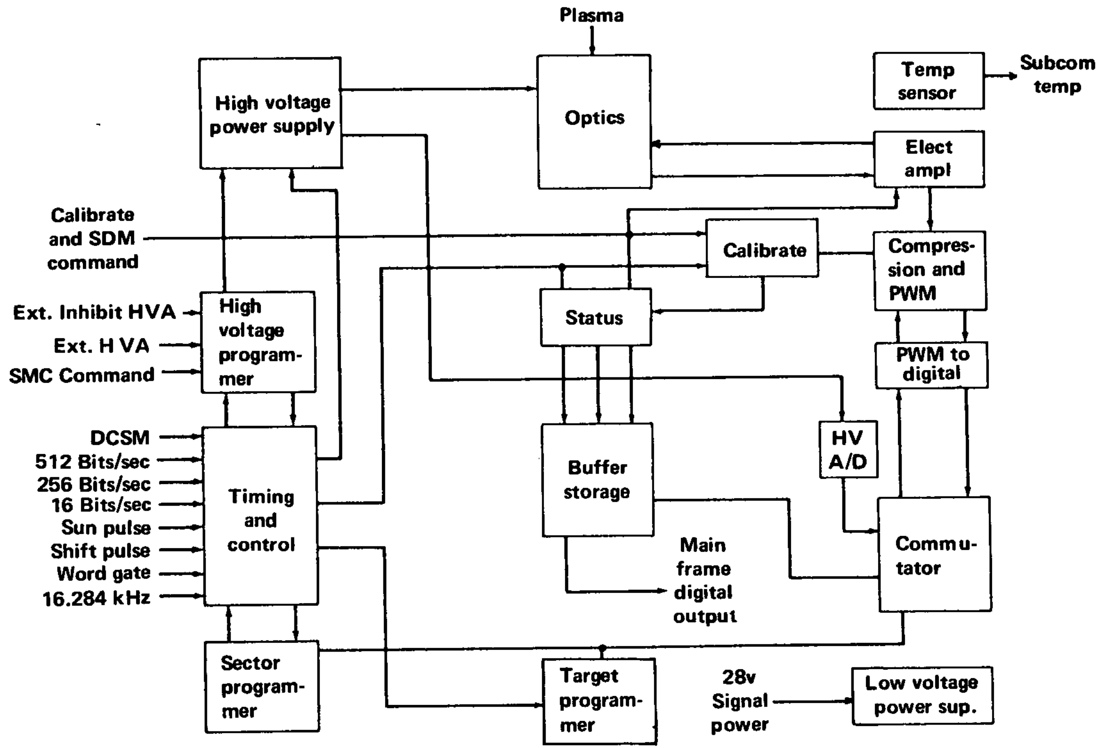


FIGURE 5-10.—Functional block diagram of the Block-II Ames plasma probe.

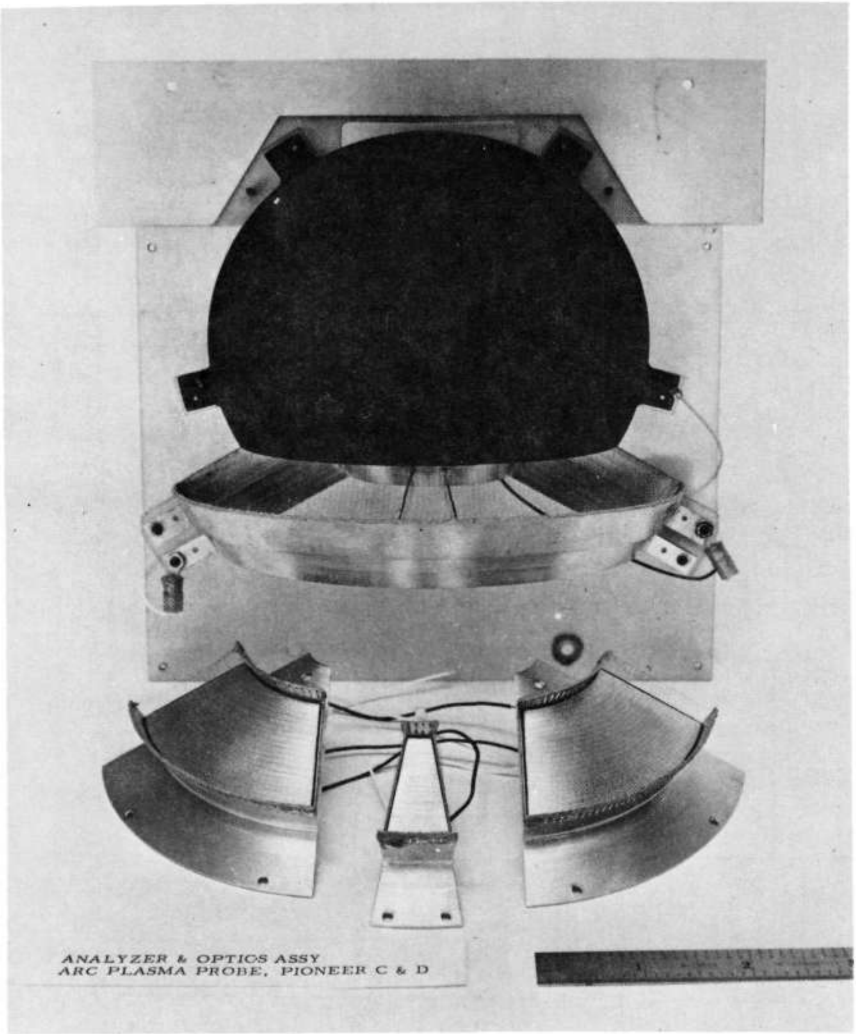
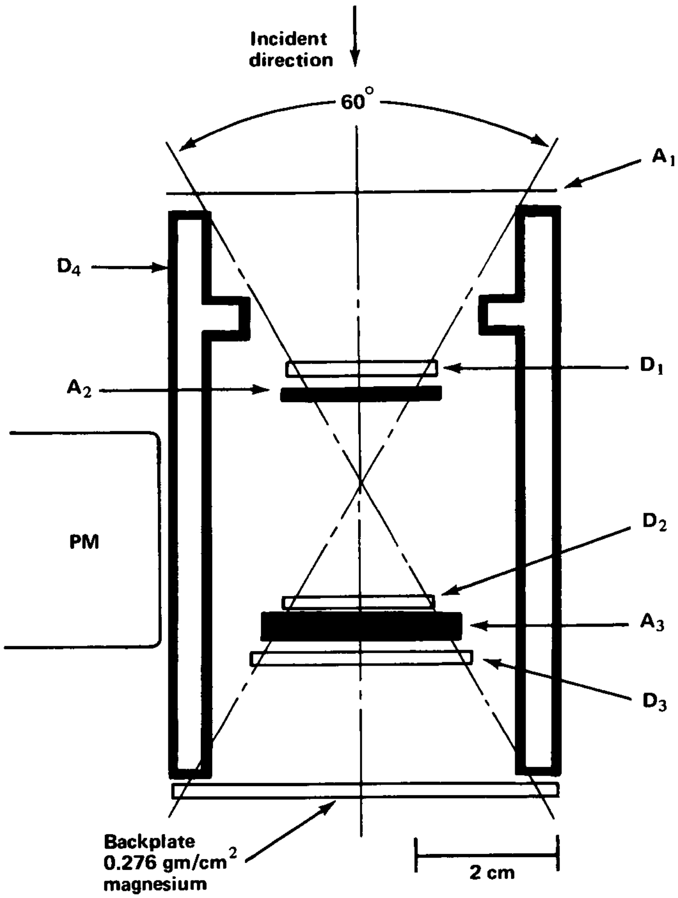


FIGURE 5-11.—External view of the Block-II Ames plasma probe.

THE CHICAGO COSMIC RAY EXPERIMENT (PIONEERS 6 AND 7)

The scientific objective of the Chicago cosmic ray experiment was the measurement of the heliocentric, radial gradient of the proton and alpha-particle fluxes in various energy ranges. Such information is useful in evaluating various models of the interplanetary magnetic field that modulates solar cosmic rays.

The basic instrument is a four element, solid-state, cosmic ray telescope



Legend

A_1	Aluminized Mylar window 0.5 mg/cm^2	D_2	Lithium drift silicon detector 0.230 gm/cm^2
D_1	Lithium drift silicon detector 0.122 gm/cm^2	A_3	Platinum absorber 8.46 gm/cm^2
A_2	Aluminum absorber 0.103 gm/cm^2	D_3	Lithium drift silicon detector 0.22 gm/cm^2
		D_4	Plastic scintillator
		PM	Photomultiplier tube

FIGURE 5-12.—Arrangement of detectors and absorbers in the Chicago cosmic-ray telescope.

(fig. 5-12) (ref. 6). Three telescope elements (D1, D2, and D3) are lithium-drifted silicon-semiconductor wafers. These detectors are surrounded by a plastic scintillator (D4), which defines the 60° acceptance cone for incident charged particles. A photomultiplier tube monitors the plastic scintillator. The silicon wafers, and of course the photomultiplier tube, are all sensitive to sunlight; this makes a light-tight enclosure a necessity. Particle absorbers between the telescope elements define the response of the elements to various particles at various energies.

The analysis of the pulses generated in the four telescope elements is complex, as indicated by the supporting electronic circuitry (figure 5–13). Ignoring for the moment the significance of pulses indicating the passage of particles from the four detectors, let us look at the 6 six-bit words that the experiment feeds into the spacecraft data handling subsystem. Five of these words are displayed on the main scientific data frames while the sixth appears twice in subcommutated scientific data (ch. 5). The Chicago experiment constructs six "spacecraft" words from five "experiment" words, which are labeled Aa, Ab, Ac, Ad, and Ae. The 18-bit word Ab is composed of three contiguous six-bit spacecraft words that are derived from the following experiment components: seven bits from the pulse-height analyzer associated with D1 and five bits from the D3 pulse-height analyzer; four more bits of information concerning the quadrant in which arriving particles were detected; one bit indicating the counting rate of coincidences from all four detectors (D1D2D3D4); and one bit showing whether the experiment is in a normal or calibrate mode.

Words Aa and Ac each contain three bits from each of the two counting-rate scalers that indicate counting rates for the following coincidence-anticoincidence situations: $\mathrm{D1\overline{D2D4}}$; $\mathrm{D1D2D3\overline{D4}}$; $\mathrm{\overline{D1}D2D3\overline{D4}}$; and D1D2D3D4.[17] The sixth (six-bit) word is subcommutated twice and is labelled Ad and Ae. This word contains five bits of rate information for the four quadrants of spacecraft rotation for the $\mathrm{D1D2\overline{D3D4}}$ logic, plus a quadrant flag-indicator bit.

Now, consider particles entering the instrument through the solid angle defined by the plastic scintillator. The particles pass through D1, producing pulses with heights proportional to the amount of energy lost in transit through the silicon wafer. The energy-loss response of D1 is plotted in figure 5–14. The detectors D2 and D3 have the same general characteristics. Armed with knowledge of the energy-loss characteristics of the absorbers placed between D1, D2, and D3, and pulse-height analysis, the experimenters can deduce considerable information about the cosmic-ray environment seen by the instrument as it scans the plane of the ecliptic.

The energy-discriminating capabilities of the experiment (when pulse-height analysis is employed) are summarized below:

For protons:	6 to 8 MeV and 80 to 190 MeV
For alpha particles:	8 to 80 MeV per nucleon and 80 MeV per nucleon to relativistic energies
For electrons:	1 to 20 MeV in the mode $\mathrm{D1D2\overline{D3D4}}$ and in excess

[17] A bar over a detector designation signifies anticoincidence. For example $\mathrm{D1\overline{D2}}$ logic means that detector D1 detects a particle at a given instant in time but D2 does not.

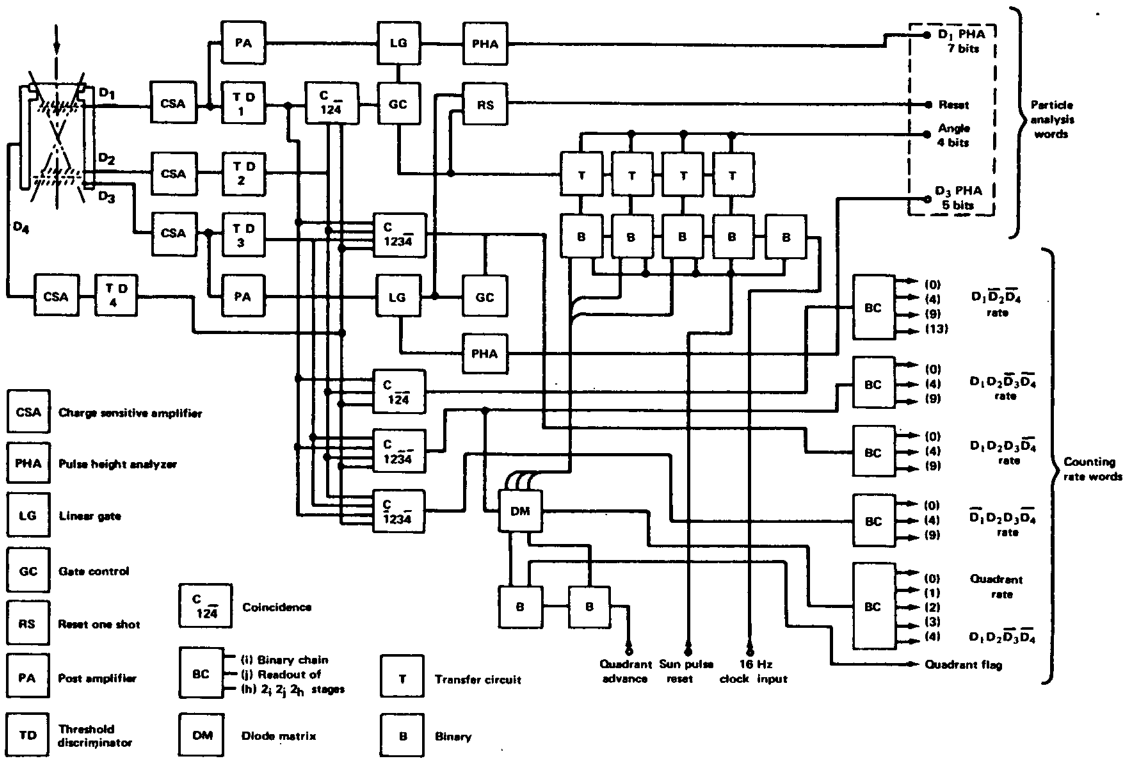


FIGURE 5–13.—Block diagram of the Chicago cosmic-ray telescope electronics.

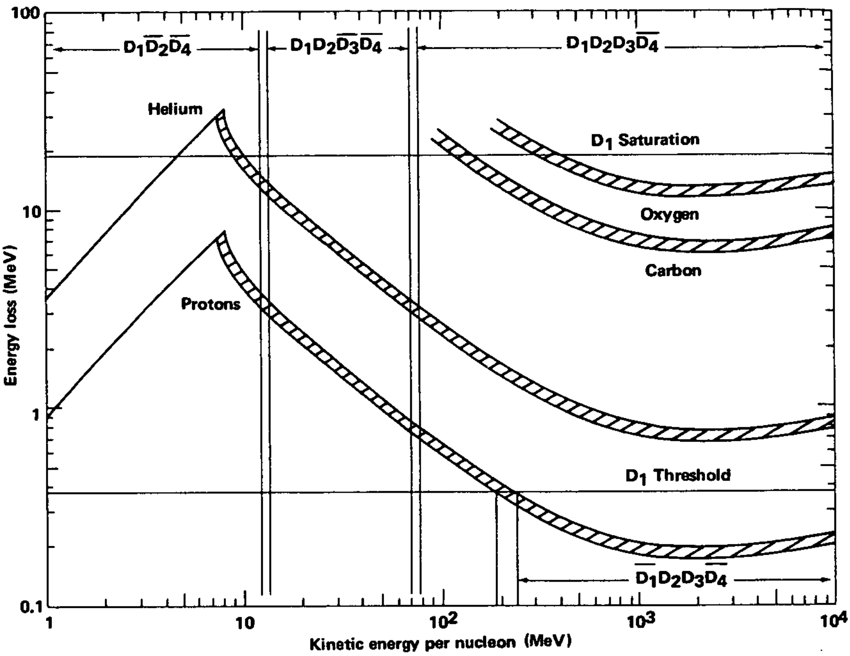


FIGURE 5–14.—D1 detector energy loss vs. particle energy, Chicago cosmic-ray telescope.

of 160 keV when D1 counts are considered alone. Electrons can be distinguished in the pulse-height analysis of D1 signals because they cause mainly low-amplitude pulses. Counting rates alone—that is, without pulse-height analysis—can also provide significant energy-and-particle discrimination in themselves. Two examples follow:

For protons plus alphas: D1$\overline{\text{D2D4}}$ logic provides counts in the 0.8 to 8 MeV per nucleon range.

For protons and alphas: D1D2$\overline{\text{D3D4}}$ logic yields counts between 8 and 80 MeV per nucleon.

The direction of particle arrival can be determined to within less than 60° with respect to the Sun line. As figure 5–9 indicates, the spacecraft Sun pulse signals are employed as references to establish the quadrant data mentioned above.

The final layout of the spacecraft instrument platform was such that the Chicago instrument was shadowed by the magnetometer boom. To compensate, the entire telescope assembly was rotated 10° with respect to the housing. The first instrument supplied to Ames was too heavy (by 300 g) and also violated magnetic cleanliness specifications. Consequently, major redesign work was carried out at the University of Chicago in July 1964 to bring the experiment within specifications. Weight was shaved off the final design by such stratagems as the use of hollow mounting screws, the elimination of unused pins on plugs, and the milling of the magnesium structures.

An operational problem cropped up at Cape Kennedy because the Chicago experiment used—for the first time—large-area lithium-drifted silicon wafers, which required stringent humidity control. The experiment had to be flushed with dry nitrogen while it was on the launch pad at Cape Kennedy until the last possible moment. Ames Research Center provided the control system for the nitrogen flushing apparatus.

THE GRCSW COSMIC RAY EXPERIMENTS (PIONEERS 6, 7, 8, 9, AND E)[18]

The Earth-based study of cosmic-ray anisotropy has always been hampered by the presence of the Earth's magnetic field and atmosphere. Even scientific satellites do not get far enough away from the Earth to avoid its magnetic field completely. The crucial test of one theory that describes the motion of cosmic rays within the solar system depends upon the careful measurement of cosmic-ray anisotropy at energies below 1000 MeV. For such measurements, the instruments must be carried well away from the Earth. The Pioneer probes were ideal for this purpose.

GRCSW instruments were part of all five Pioneer payloads, but those on Pioneers 8, 9, and E (Block II) represented a second generation of equipment (refs. 7 and 8). The later equipment was more sophisticated because additional low-energy measurements were made in, above, and below the plane of the ecliptic.

In both Block-I and Block-II generations of equipment, the principal cosmic-ray detector consisted of a flat cylindrical CsI(Tl)-scintillator crystal (detector C) contained within a cup-like cylindrical container of scintillating polytoluene (detector D), which functioned as a guard detector. On all five Pioneers the CsI(Tl) and plastic scintillators were connected in anticoincidence so that the detector was directional with an acceptance cone of about 107°. Particles with energies greater than 90 MeV/nucleon were also eliminated because, even if they entered the instrument's aperture, they passed right through the CsI(Tl) scintillator and activated the guard scintillator. Separate photomultiplier tubes watched the two scintillators (fig. 5–15). A 10-nanocurie americium-241 radioactive source was attached to each of the CsI(Tl) scintillators for purposes of

[18] GRCSW = Graduate Research Center of the Southwest; later renamed Southwest Center for Advanced Studies (SCAS) and now known as The University of Texas at Dallas.

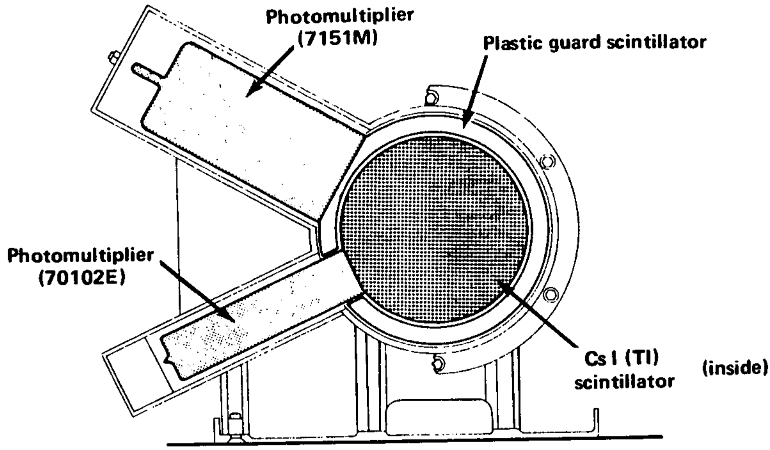


FIGURE 5–15.—Axial view of the GRCSW cosmic-ray telescope, Block-I Pioneers. The detector dimensions and positions were changed for the Block-II flights; see text.

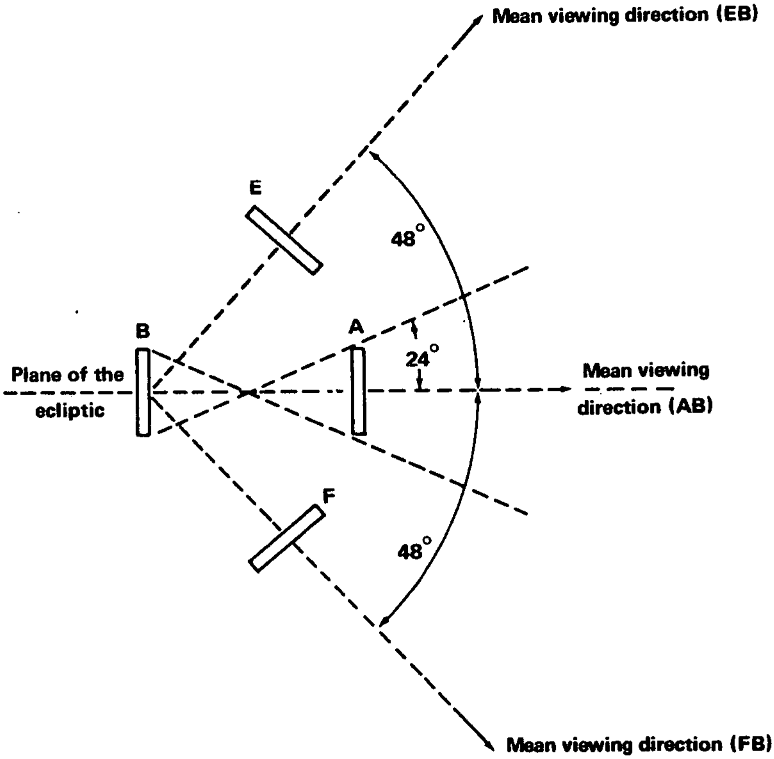


FIGURE 5–16.—Viewing angles of the Block-II GRCSW cosmic-ray experiment's solid-state detectors.

instrument calibration. Two sheets of 0.0005-in. aluminized-Mylar covered each plastic cup and protected the detectors from light, heat, and micrometeoroids.

The same basic scintillator arrangement was employed for the Block-II flights, but it was supplemented with a three-way coincidence telescope consisting of four 100-micron, totally depleted silicon, surface-barrier detectors arranged and labeled as shown in figure 5–16. EB-coincidence logic counts particles within the energy range 0.5 to 5.0 MeV/nucleon arriving from 48° above the plane of the ecliptic. AB logic permits monitoring the plane of the ecliptic, while BF logic keeps track of particles arriving from 48° below the plane of the ecliptic.

The detectors scan the plane of the ecliptic as the spacecraft spins. In the Block-I instruments, additional directional discrimination is provided by four electronic quadrant gates (Q_1, Q_2, Q_3, and Q_4 in fig. 5–17) which are opened by electronic gates in sequence during each revolution following a Sun pulse. Quadrants 1 and 3 look away from and toward the Sun, respectively; quadrants 4 and 2 look fore and aft along the spacecraft's orbit, respectively.

The goal of the experiments was the study of cosmic ray anisotropies as small as 10^{-3} of the mean cosmic-ray flux. Consequently, the count-accumulation times for the four quadrant-registers had to be identical to at least one part in 10^4 to provide meaningful experimental results. A unique and critical part of the experiment, therefore, was the precision, crystal-controlled aspect clock that controlled the gating pulses.

The primary modes of operation of the Block-I experiment are listed below:

(1) Dynamic range off—The length of each of the four time periods is almost one-fourth of a spacecraft revolution. This is used when the Sun is relatively quiet.

(2) Dynamic range on—Each of the four periods is approximately $\frac{1}{32}$ of a revolution. This is used when the Sun is active.

(3) Slip mode—The quadrant time periods are shifted by 45° to help obtain better angular resolution with only four basic time sectors. The slip mode was used only on Pioneer 7.

(4) Calibrate mode—The built-in americium-241 source was used to calibrate the instrument.

The desire for finer directional discrimination in the Block-II instruments led to the development of a more sophisticated method of scanning the azimuthal plane as well as the four surface-barrier detectors already mentioned. The electronic equipment supporting the Block-II instruments is correspondingly more complex (see fig. 5–18). Signals generated by the passage of cosmic rays in the six detectors are selected for pulse-height analysis by an array of linear switches. The pulse-height analyzers establish energy windows for the particle-produced signals. In the case of the ani-

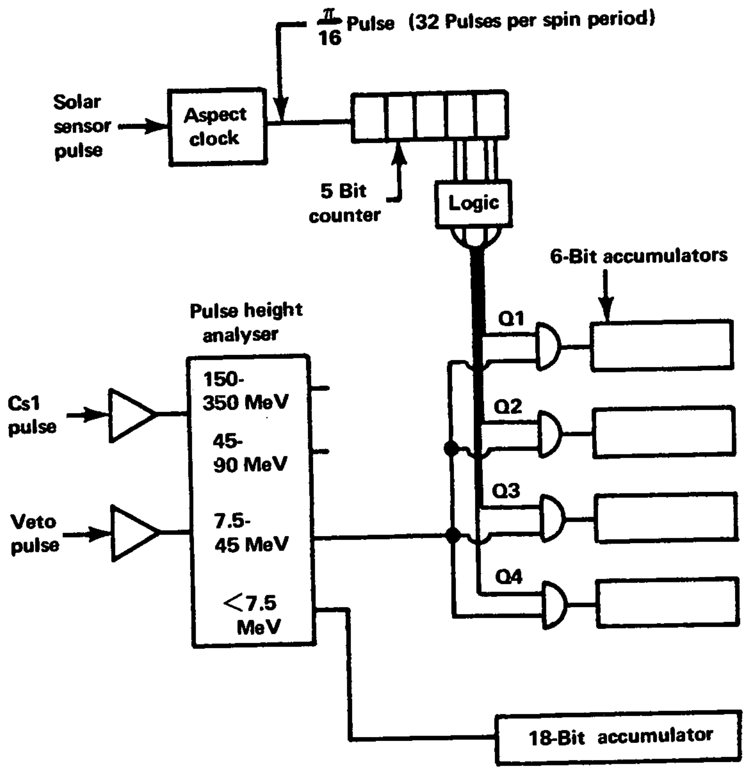


FIGURE 5–17.—Greatly simplified block diagram of the electronics associated with a single channel of the GRCSW cosmic-ray experiment, Block-I instruments only.

sotropy pulse-height analyzer, a pulse from detector B must fall within the energy window and also be coincident with a signal from A, E, or F. In this way, particles are divided into three groups that accumulate counts from particles arriving above, below, and within the plane of the ecliptic. The logic for detectors C and D is the same as it was for the Block-I instruments.

Rather than the Block-I approach to quadrant division—use of the Sun pulses and aspect clock—the Block-II experiments employ octant-division circuitry. An azimuth computer accurately divides the plane of the ecliptic into 32 equal parts plus a short "dead time" to allow for variations in the spacecraft spin rate. The slip mode used on Block-I Pioneers was not necessary because of the finer spatial discrimination of the Block-II electronics.

Five spacecraft telemetry words (a total of 30 bits) were assigned to the Block-II instruments. Nine bits were assigned to each of three experiment data accumulators. The remaining three bits were associated with the

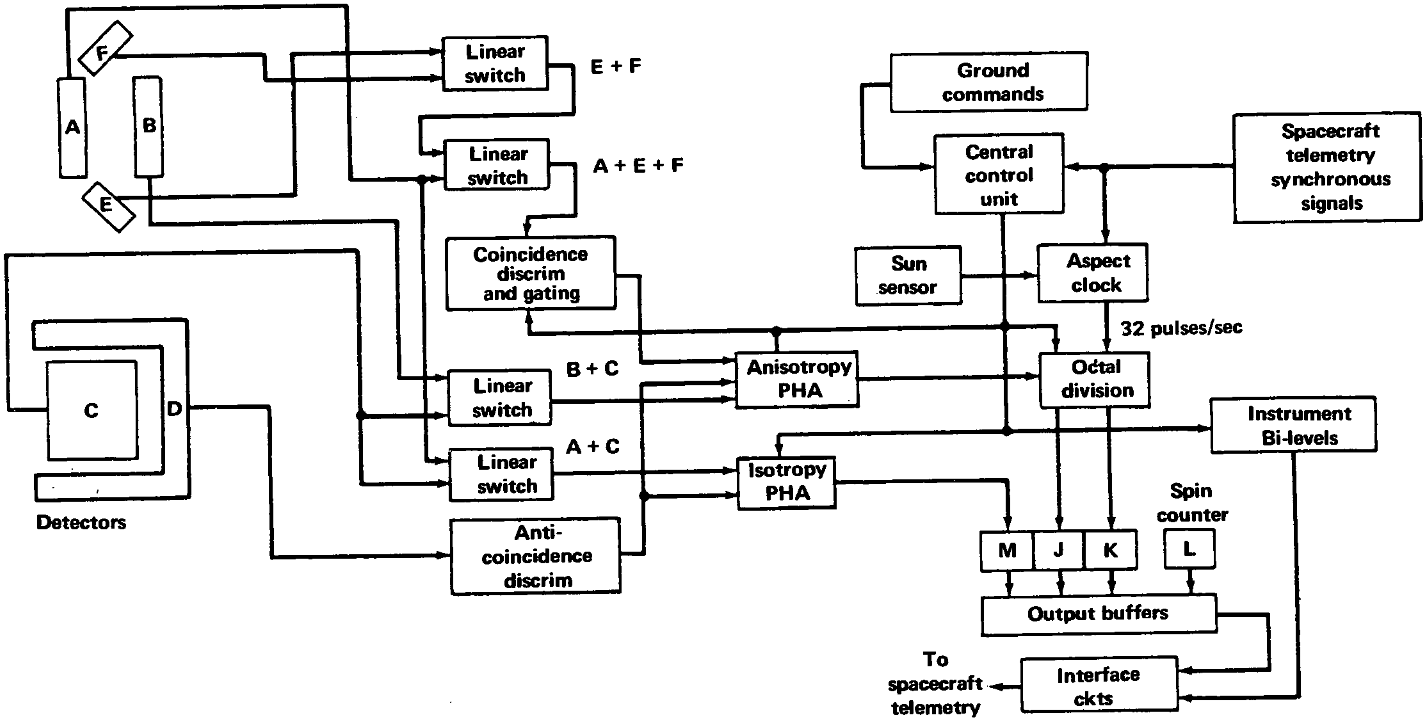


FIGURE 5-18.—Block diagram of the electronics associated with the Block-II GRCSW cosmic-ray experiment.

experiment's spin counter. To conserve power, the output buffer—a serial shift register—was limited to only 21 bits. During readout of the experimental data, as soon as 13 bits are shifted out of the buffer into the telemetry stream, nine bits from the isotropy data accumulator are dumped into the buffer to complete the 30-bit message. Because the experiment counting rate often exceeds the storage capacity of the nine-bit accumulators, a form of logarithmic storage was employed which allowed 2752 counts to be stored in the nine-bit format.

The additional sophistication of the Block-II instruments increased experiment weight from 4.4 to 5.6 lb; the average power drain went from 1.6 to 1.8 W. During the hardware development of the experiment, it was discovered that commercially supplied photomultiplier tubes normally contained so much magnetic Kovar alloy that the magnetic cleanliness standards could not be met. Tubes with the Kovar replaced by a nonmagnetic nickel alloy (Alloy 180) were built especially for the experiment.
standards could not be met. Special tubes with the Kovar replaced by a nonmagnetic nickel alloy (Alloy 180) were built especially for the experiment.

MINNESOTA COSMIC RAY DETECTOR (PIONEERS 8, 9, AND E)

The Minnesota cosmic ray experiment had a purpose entirely different from that of the GRCSW instrument. The experiment objectives listed below reflect the lack of high precision cosmic ray experiments flown on spacecraft prior to the spring of 1964:

(1) To measure the quiet-time energy spectrum of protons, alphas, and heavier nuclei up to a charge of 14 over a wide energy range with better energy and background discrimination than previously obtained

(2) To measure the variations in these spectra, including the features of Forbush decreases as well as the 11-year variation during the solar cycle

(3) To measure the radial and azimuthal cosmic-ray gradients existing in interplanetary space during quiet and disturbed periods on the Sun

(4) To measure comprehensively the charge, isotropic composition, and energy spectrum of solar cosmic rays

The Minnesota instrument incorporates seven separate detectors (fig. 5–19), which are, in effect, electronically arranged into five different telescopes via commands from the Earth (ref. 9). Detector G is a two-piece guard counter made of Pilot B plastic; it is viewed by a photomultiplier tube. Detector D, at the bottom of the telescope, is a 1-cm-thick piece of synthetic sapphire and functions as a Cerenkov counter. Another photomultiplier tube views this detector. The remaining five detectors—B_{1A}, B_{1B}, B_2, B_3, and C—are all of the semiconductor type. The coincidence-anticoincidence conditions that electronically create five different telescopic arrangements are listed in table 5–4 along with the ranges and particles which they can detect.

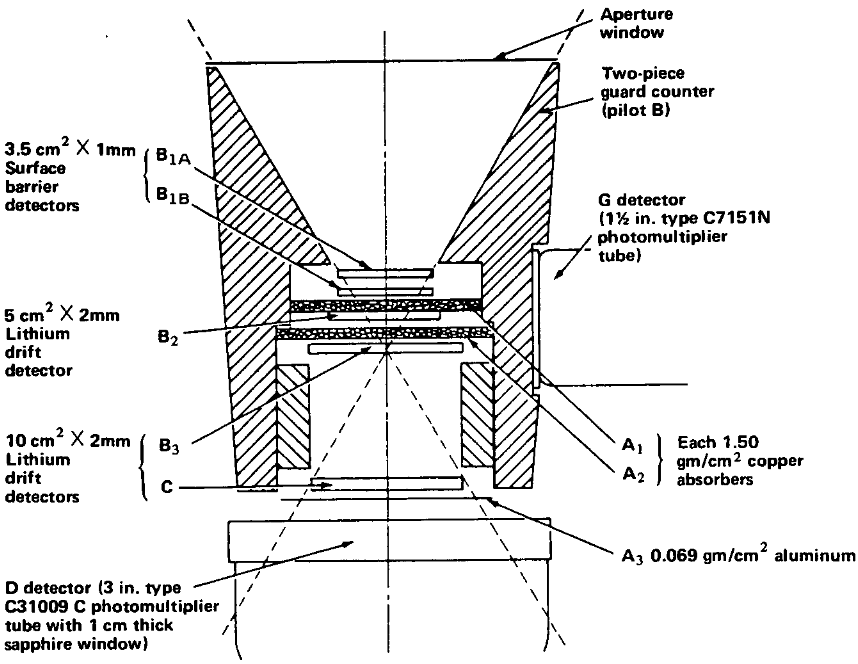


FIGURE 5–19.—Arrangement of detectors and absorbers in the Minnesota cosmic-ray telescope.

Telescope T5 is essentially omnidirectional and samples the particles in the cosmic-ray flux with energies greater than 14 MeV per nucleon. Electrons greater than 0.6 MeV are also detected. In telescope T4, however, the guard scintillator G is in anticoincidence and only particles entering the aperture are counted. Detector B_2 sets the range energy. The pulses from T4 are pulse-height-analyzed into nine energy groups. In telescope T3, the geometry is established by B_{1A}, B_{1B}, B_2, and G. The energy range is set by detectors G and C. Here, the signals from B_2 and B_3 are summed and pulse-height-analyzed into three energy groups. Telescopes T1 and T2 are differentiated by the following condition: If $D > B_3 - B_1$, the event is defined as a T1 event; if $D < B_3 - B_1$, it is a T2 event. The reader is referred to Lezniak's thesis for details about all five telescopes (ref. 10).

Figure 5–20 presents a functional block diagram of the electronic circuitry supporting the experiment. The entire experiment weighed approximately 8.0 lb and required 3.1 W of spacecraft power.

THE STANFORD RADIO PROPAGATION EXPERIMENT (PIONEERS 6, 7, 8, 9, AND E)

The Stanford experiment[19] measured the integrated electron density

[19] Actually a joint project of Stanford University and Stanford Research Institute.

TABLE 5–4.—*Minnesota Cosmic-Ray Telescope Arrangements*

Telescope	Coincidence-anticoincidence requirements	Charge and energy ranges of the particles detected	
T1, T2	$B_{1A} \cdot B_{1B} \cdot B_2 \cdot B_3 \cdot C$	$Z \geq 1$	$E \gtrsim 64$ MeV per nucleon
		$e^{\pm}$	$E \gtrsim 8.4$ MeV
T3	$B_{1A} \cdot B_{1B} \cdot B_2 \cdot \overline{C} \cdot \overline{G}$	$e^{\pm}$	4.2 MeV $\leq E \leq$ 8.4 MeV
		$_1H^1$	39.6 MeV $\leq E \leq$ 64.3 MeV
		$_2He^4$	39.4 MeV per nucleon $\leq E \leq$ 64.1 MeV per nucleon
T4	$B_1 \cdot \overline{B}_2 \cdot \overline{G}$ $(B_1 = B_{1A} + B_{1B})$	$e^{\pm}$	0.34 MeV $\leq E \leq$ 4.3 MeV
		$_1H^1$	3.5 MeV $\leq E \leq$ 39.7 MeV
		$_2He^4$	6.6 MeV per nucleon $\leq E \leq$ 39.7 MeV per nucleon
T5	$B_{1A} \cdot B_{1B}$	$Z \geq 1$	$E \gtrsim 14$ MeV per nucleon
		$e^{\pm}$	$E \gtrsim 0.6$ MeV

along the radio transmission path between the Earth and spacecraft (ref. 11). For successful operation the experiment required that a dual-channel, phase-locked-loop receiver in the spacecraft lock onto signals transmitted from the 150-ft parabolic antenna located on the Stanford campus. When the experiment is in progress, two modulated coherent carriers of approximately 49.8 and 423.3 MHz are sent to the spacecraft from the 150-ft Stanford antenna. The spacecraft receiver measures the relative phase change between the modulation envelopes. Since the higher frequency is relatively unaffected by the presence of ionization, the comparison provides the information needed to compute the integrated electron-number density, or the total number of electrons per square meter between Earth and spacecraft. The rate of phase change of one signal with respect to the other is also measured to very high precision to determine the time variation of the integrated electron number density. The experiment also measures the strength of the signals sent from Earth.

Both the 49.8- and 423.3-MHz transmissions to the spacecraft originate at the Stanford computer-controlled "Big Dish." The 49.8-MHz signal is fed to a crossed, folded dipole and a reflector that are located just below the focal point of the 150-ft dish (fig. 5–21). This signal is generated in a 300-kW linear amplifier transmitter. The high frequency signal, 423.3 MHz, is radiated directly from the horn of the dish. A 30-kW klystron transmitter generates the signal. Some additional transmitter information is listed in table 5–5.

The transmitting system block diagram (fig. 5–22) begins with the 1-MHz crystal oscillator common to both transmitted frequencies. The circuits

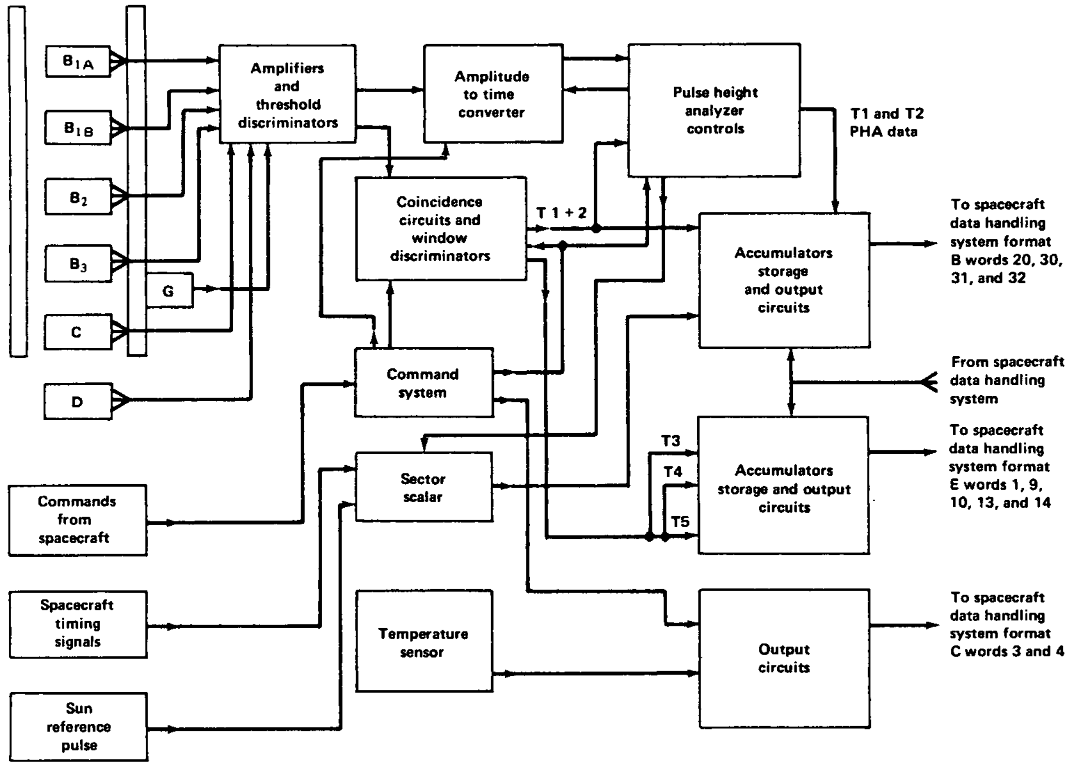


FIGURE 5–20.—Data flow diagram for normal-mode operation, Minnesota cosmic-ray experiment.

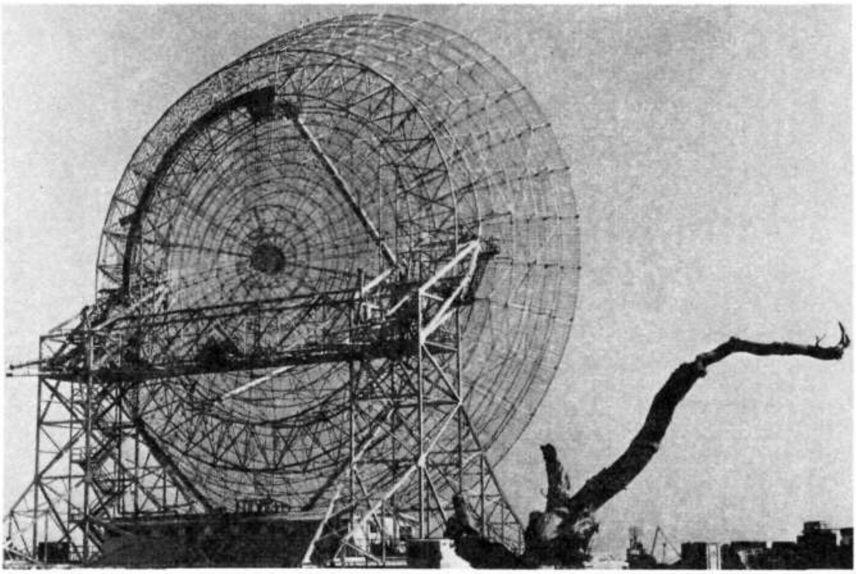


FIGURE 5-21.—The 150-ft dish at Stanford University, Palo Alto, Calif. This antenna is used in the radio propagation experiment.

TABLE 5-5.—*Stanford Experiment Transmitter Characteristics*

Characteristic	Channel 1	Channel 2
Transmitter frequency	49.8 MHz	423.3 MHz
Type of transmitter	Triode linear amplifier	Klystron
Power output for Pioneer experiment	300 kW 84.0 dBm	30 kW 74.7 dBm
Antenna type	150-ft-diameter parabolic dish	
Antenna gain, $\eta = 1$	25 dB	42 dB
3-dB beamwidth	8.0°	1.2°
Antenna efficiency, η (est.)	0.50	0.50
Polarization	Left-hand elliptical	Right-hand circular
Polarization loss	Varying with Faraday rotation	3 dB
Area of receiving dipole on spacecraft	4.7 m^2	0.65 m^2
Modulation frequency	7.692 or 8.692 kHz	
Modulation phase	Continuously adjustable through 360°	Fixed
Fraction of power in modulation sidebands	0.5	0.5

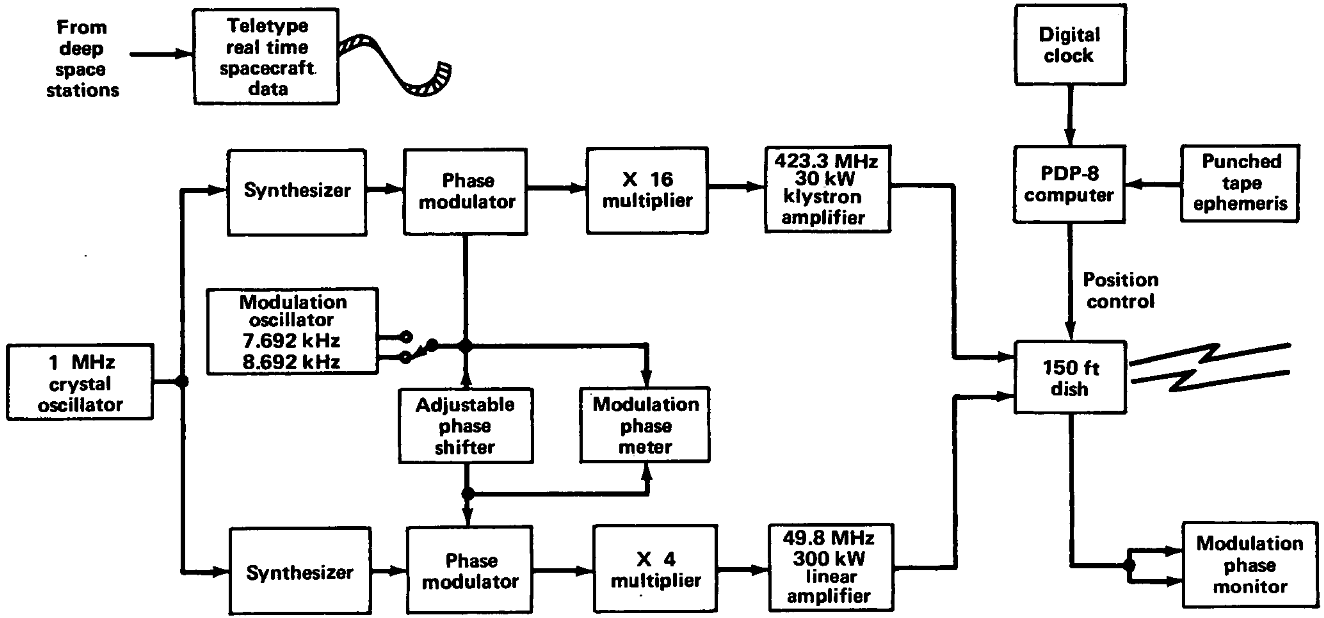


FIGURE 5-22.—Block diagram of the Stanford Earth-based transmitter used in the Pioneer radio-propagation experiment.

following then set the 423.3-MHz carrier at exactly 17/2 times the 49.8-MHz carrier. Both carriers are modulated at either 7.692 or 8.692 kHz for differential group path measurement. Real-time teletype data are sent to Stanford from the DSN showing the operating point of the spacecraft phase meter. This information is used to keep the spacecraft phase meter on its positive slope.

Both carriers from the Earth are received by the Stanford spacecraft antenna (fig. 5–23) and sent to the dual-channel receiver, which consists of two separate coherent phase-locked receivers (fig. 5–24). The main reasons for the phase-lock design are: (1) to increase the sensitivity of the receiver, and (2) to detect the difference in radio frequency cycles between the 49.8 MHz and the 2/17 harmonic of the 423.3-MHz carrier. Additional receiver data are presented in table 5–6.

Because the Stanford experiment must have transmitter operators at Stanford in the loop during its operation, real-time teletype data are relayed directly from JPL's SFOF to Stanford (ch. 8). Teletyped parameters include the modulation phase-difference measurements and the rf-difference counts. The Stanford operator uses this information to adjust the transmitter frequencies, powers, and modulation phase offset for best operation. At the experiment design range of 300 000 000 km, it takes about 33 min for the effects of transmitted changes to be seen in the teletype messages from JPL.

Minor changes were made in the experiment during the program and these are reflected in the slight weight change between Block-I and Block-II Pioneers. The spacecraft portion of the Stanford experiment weighed 6.0 and 6.3 lb for Blocks I and II, respectively. Power consumption was approximately 1.6 W for all flights.

THE TRW SYSTEMS ELECTRIC FIELD DETECTOR (PIONEERS 8, 9, AND E)

The Stanford and TRW Systems experiments are closely related. In fact, the TRW Systems experiment makes direct use of the Stanford antenna (ref. 12). The purpose of the radio propagation experiment—measuring integrated electron density over long distances—is essentially macroscopic in nature, whereas the TRW Systems experiment is microscopic in design. Its purpose is the detection of charge differences over small distances in interplanetary space through the electric fields they create along the Stanford antenna. Plasma waves and other cooperative actions in the 100- to 100 000-Hz vlf range of charged particles in collisionless interplanetary space can be detected with the instrument.

The decision to add the Electric Field Detector was made well after the Block-II payload was selected. Six spare words from the Pioneer telemetry format were made available. The weight, 0.9 lb, and power drain, 0.5 W,

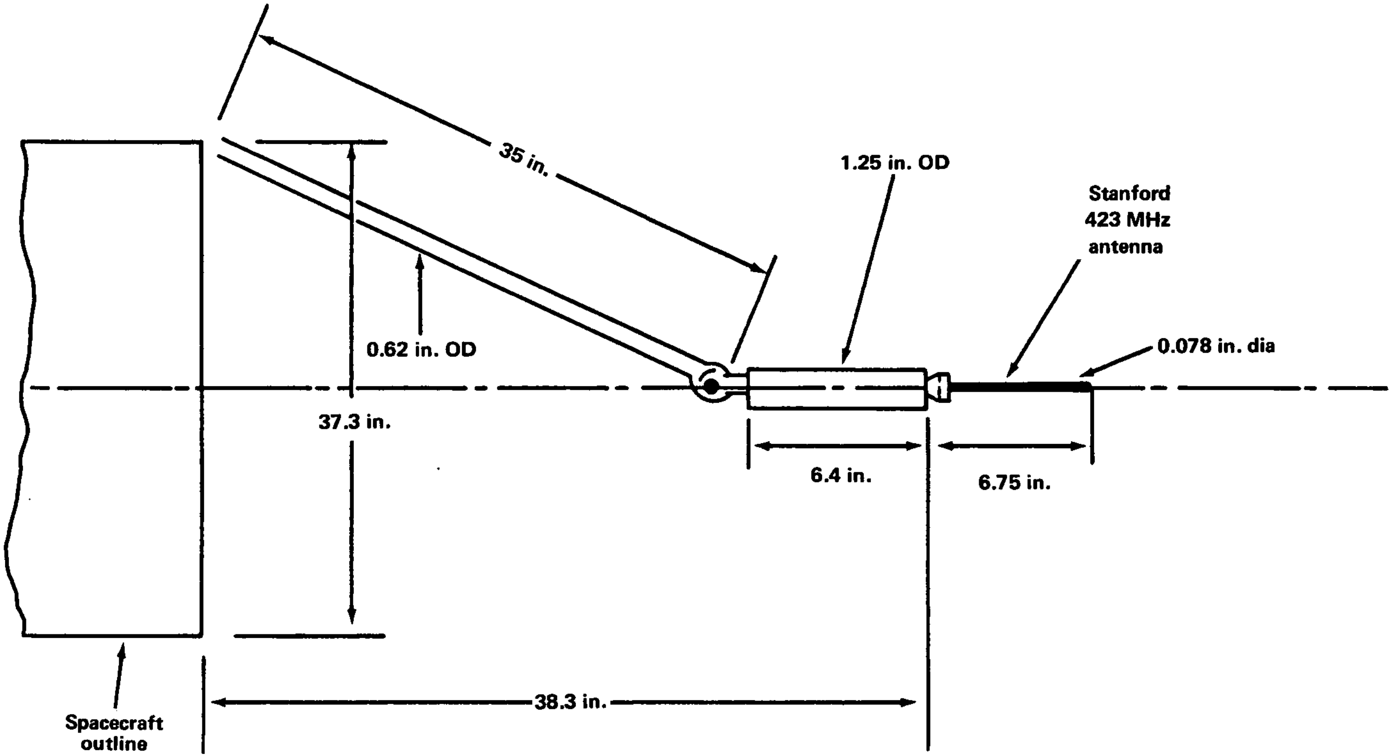


FIGURE 5-23.—Dimensions of the spacecraft antenna used in the Stanford radio-propagation experiment (not to scale).

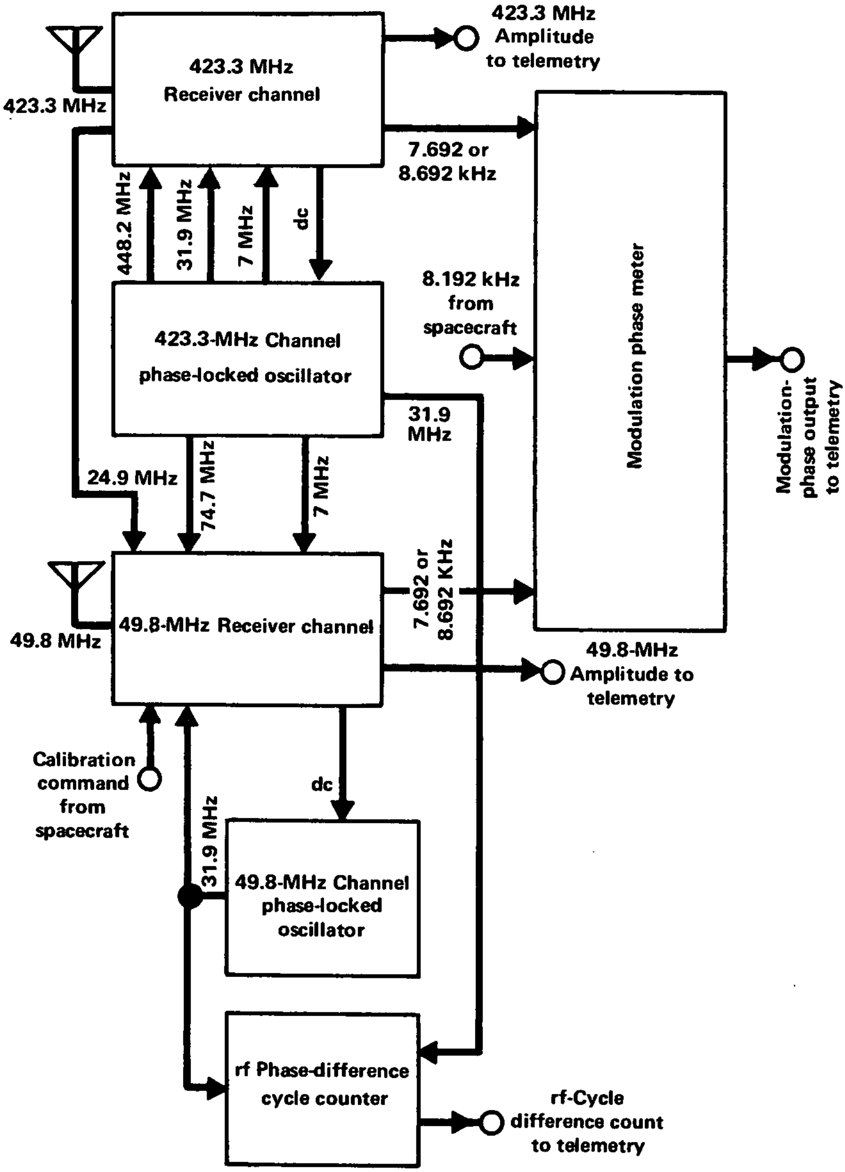


FIGURE 5-24.—Block diagram of the Stanford dual-channel spacecraft receiver.

made it possible to squeeze this experiment onto the spacecraft without major changes (particularly since it could use the Stanford antenna). In a sense, it is an addendum to the Stanford experiment, and it is often treated thus in the literature.

The electric field experiment makes use of the short (6.4 in.) 423.3-MHz

TABLE 5–6.—*Stanford Experiment, Receiver Parameters*

Frequency	49.8 MHz	423.3 MHz
I.F. bandwidths		
3-dB bandwidth	40 kHz	40 kHz
Noise bandwidth	45 kHz	45 kHz
Receiver input noise (including image)		
Noise figure	3 dB	3 dB
Noise temperature	300° K	290° K
Cosmic noise (in a dipole with axis parallel to galactic axis)	8000° K	100° K
Spacecraft noise plus cosmic noise for Pioneer 7		
Noise temperature	17 000° K	400° K
Modulation frequencies for phase meter	7.692 or 8.692 kHz	
Modulation phase accuracy	Digitized to approximately 3.15° per digit; additional error estimated to be +2°	
Differential phase path	Quantized to 1 Hz of rf phase difference at 49.8 MHz	

segment of the Stanford antenna (fig. 5–23) as a capacitively coupled sensor with which local plasma waves can be detected. The sensor is relatively insensitive, but adequate for the purposes of the experiment. A number of Earth satellites have carried similar vlf radio receivers for the same purpose.

The availability of only six subcommutated telemetry words restricted the experiment's capability to survey a wide range of plasma waves in interplanetary space. Even when the spacecraft transmits at the highest bit rate of 512 bits/sec, the TRW Systems experiment sends only two words every 7 sec, and the four others every 28 sec. The portion of the wave spectrum to be studied was selected carefully in advance on the basis of limited knowledge of plasma waves in space. The high frequency channel selected was at 22 kHz for Pioneer 8 and 30 kHz for Pioneers 9 and E. The low frequency channels were at 400 Hz and 100 to 100 000 Hz (for the broadband survey) on all Block-II spacecraft.

Referring to the experiment block diagram, figure 5–25, the two bandpass channels are sampled every seven seconds when the spacecraft transmits at 512 bits/sec. The remaining four telemetry words carry data from the broadband portion of the broadband pulse-height experiment. As analysis proceeds, positive pulses per unit time that exceed preset trigger levels are counted. The trigger level is changed in a programmed sequence of 16 or 8 steps. (See table 5–7 for differences between Pioneer 8 and 9 instrumentation.) The pulse-frequency count is read out before the trigger level is changed to the next step. At the 512 bits/sec rate, the entire broad-

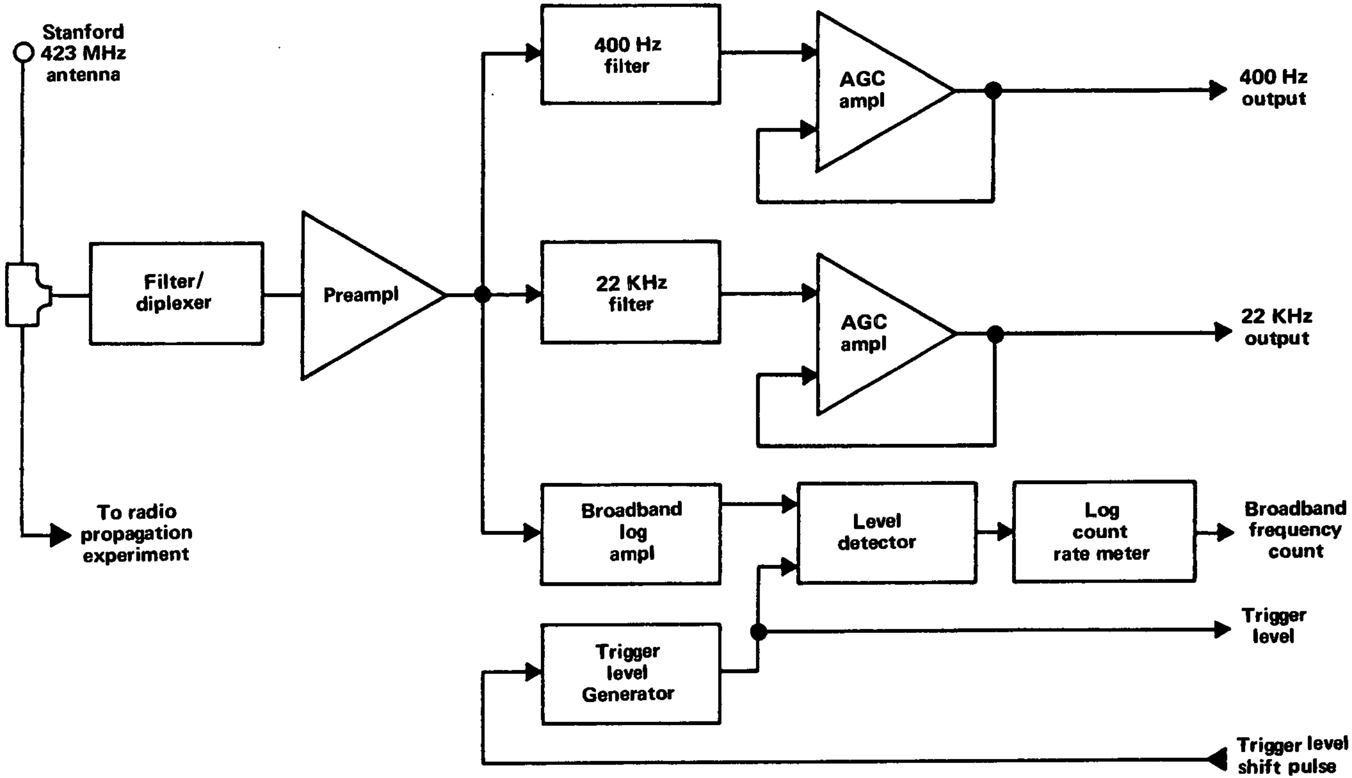


FIGURE 5–25.—Block diagram of the TRW Systems electric-field instrumentation, showing the use of the Stanford antenna.

TABLE 5-7.—*Differences Between TRW Experiments*

Difference	Pioneer 8	Pioneers 9 and E
Engineering words for frequency count	2	4
Engineering words for step number	2	1
Number of steps	16	8
Frequency readings before step is changed	2	1
Time for broadband scan	7.47 min for 16 channels	56 sec for 8 channels

band scan is repeated every 7.47 min on Pioneer 8 and every 56 sec on Pioneer 9.

THE GODDARD COSMIC DUST EXPERIMENT (PIONEERS 8, 9, AND E)

As related at the beginning of this chapter, no cosmic dust experiments were initially proposed for the Block-II Pioneers, and the Block-I experiment proposed by Ames Research Center was not far enough along in development to make the Block-I flights. The Block-II Goddard cosmic dust experiment described below is the result of a specific solicitation of likely experimenters in this field by NASA Headquarters. The following discussion is adapted from Berg and Richardson (ref. 13).

The experiment objectives were four in number:

(1) To measure the cosmic-dust density in the solar system well away from the Earth

(2) To determine the distribution of cosmic-dust concentrations (if any) in the Earth's orbit

(3) To determine the radiant flux density and speeds of particles in meteor streams

(4) To perform an in-flight determination of the reliability of the microphone as a cosmic-dust detector

The last objective reflected the growing disenchantment with microphone micrometeoroid detectors due to the possibility of spurious data arising from thermal effects in Earth orbit.

The instrument consists of two film-grid sensor arrays spaced 5 cm apart, followed by an acoustical impact plate (microphone) upon which the last film is mounted. Three types of cosmic dust particles were considered in the design of the experiment:

(1) High-energy, hypervelocity particles ($>$ 1.0 erg)

(2) Low-energy, hypervelocity particles ($<$1.0 erg)

(3) Relatively large high-velocity particles ($> 10^{-10}$ grams).

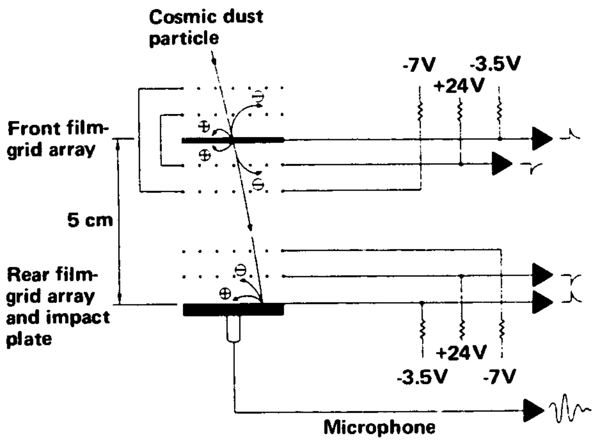


FIGURE 5–26.—Schematic diagram of the Goddard micrometeoroid sensor.

As a *high-energy, hypervelocity particle* pierces the front film sensor (fig. 5–26), some of its kinetic energy generates ionized plasma at the front, or "A" film. The electrons in the plasma are collected on the positively biased grid (+24 V); this creates negative pulses, as shown. The positive ions in the plasma are collected on the negatively biased film (−3.5 V); this produces a positive pulse that is pulse-height-analyzed to measure the particle's kinetic energy. The same thing occurs at the rear sensor, or "B" film; this generates a second set of plasma pulses. Impact on the plate produces an acoustical pulse. A peak-pulse-height analysis is performed on the acoustical sensor output as a measure of the particle's remaining momentum.

A *low-energy, hypervelocity particle* will yield all of its kinetic energy at the "A" film. A pulse-height analysis measures the particle's kinetic energy. A high-energy hypervelocity particle may be erroneously registered as a low-energy hypervelocity particle if, because of its angle of entry, it fails to hit the "B" film. If a *relatively large, high-velocity particle* enters, it may pass through the front and rear film arrays without generating detectable plasma because of its relatively low velocity; but it may still impart a measurable impulse to the acoustical sensor. An electronic "clock" registers the times of flight of particles. The time lapses between positive pulses from the "A" and "B" films are used to derive particle speeds.

The time-of-flight sensor is one of 256 similar sensors that comprise the portion of the Pioneer instrument measuring particle speed and direction. Figure 5–27 is an exploded schematic view of the overall experiment. It shows the four vertical film strips crossed by four horizontal grid strips that create 16 front and 16 rear film sensor arrays (each 2.5 × 2.5 cm) or 256 total combinations. Each grid strip and film strip connects to a

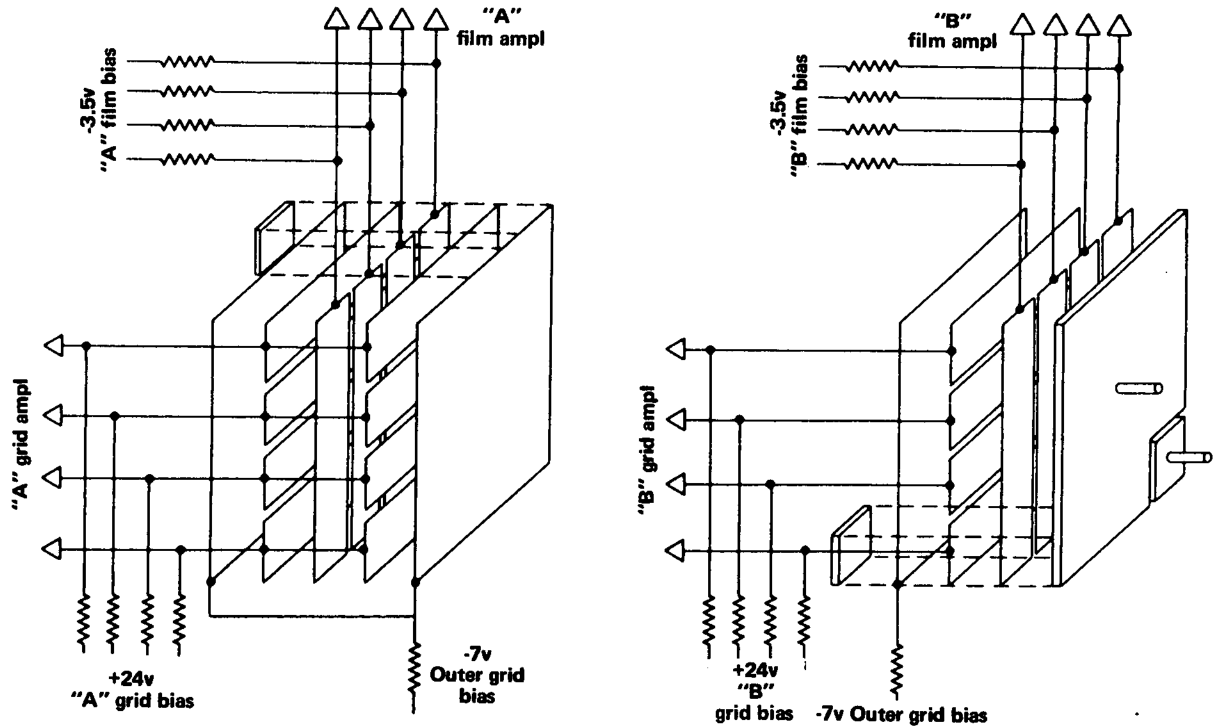


FIGURE 5–27.—Geometry of the sensor arrays in the Goddard cosmic-dust experiment.

separate output amplifier. The output signals from these amplifiers are used to determine the segment in which an impact occurred. Thus, by knowing the front film-grid segments penetrated and the rear film-grid segment affected by the impact, one can determine the direction of the incoming particle with respect to the sensor axis and the spacecraft attitude. The solar-aspect sensor determines the Sun line at the time of an impact.

Each of the four vertical films of the front sensor array, as shown in figure 5–27 is a composite of the eight layers shown in the exploded view (fig. 5–28). Ideally, a thin copper foil (500 Å) could be used alone for the vertical strips of the front sensor array, but the foil is very fragile and subject to corrosion. Therefore, the nickel grid, the parylene substrate, and the parylene encapsulation are used as supports and anti-corrosion covering for the metal film deposits. The aluminum layers, which serve only as fabrication aids during the preparation of the composite film, reflect the intense heat generated by copper evaporation upon the parylene substrate. Each of the rear sensor array film strips is a 60-μ molybdenum sheet cemented to a quartz acoustical sensor plate.

Extensive calibrations were made using a 2-MeV electrostatic accelerator. Unfortunately the particles used for calibration have been limited to high-density, hard spheres of iron (10^{-13} g < mass < 10^{-9} g) with velocities merely approaching the low end of the meteoroid velocity spectrum (2 to 10 km/sec).

The plasma sensors respond nearly linearly to the particle's kinetic energy over the limited particle parameter range specified above for the

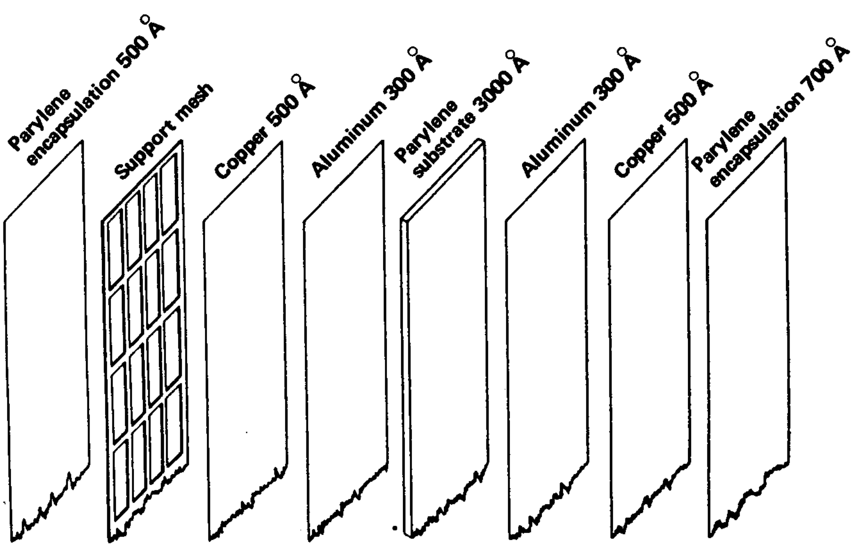


FIGURE 5–28.—Construction of a front film sensor in the Goddard cosmic-dust experiment.

laboratory simulator. The acoustical sensors respond to the particle's momentum for that same particle range.

The threshold sensitivity of the front film sensor array to laboratory particles is 0.6 erg. Time-of-flight is registered for laboratory particles having kinetic energies of 1.0 erg or greater. The electronics of the time-of-flight sensor are limited to particles with velocities ranging from 2 to 72 km/sec.

Figure 5–29 shows a block diagram of the experiment. A summing amplifier receives the positive pulse from each "A" film strip. After a gain of unity, the pulse travels two separate paths. On one path it is amplified 15 times; its pulse height is analyzed; and its amplitude is recorded in the storage register. On the other path it is amplified 1000 times and fed into a threshold one-shot multivibrator. The output pulse performs three functions:

(1) Its origin identification is impressed directly upon the storage register.

(2) It passes through the NOR gate and initiates a time-of-flight measurement.

(3) It is gated back to the threshold one-shot multivibrator to inhibit any other "A" film pulse until the measurement has been completed.

An inhibit signal to the other three films is necessary to avoid capacitative crosstalk for high-energy impact signals. The "A" film pulse is pulse-height-analyzed and the results are stored in the register to await readout.

Positive pulses from the "B" film follow similar, but separate, electronic paths with the following two exceptions: (1) no pulse-height analysis is performed on the "B" film pulses, and (2) the pulse is used to stop the time-of-flight clock. If no "B" film pulse follows an "A" film pulse, the time-of-flight register goes to the full (63-bit) state and remains full until another event occurs.

Negative pulses from each of the "A" and "B" grids are amplified via separate units and are registered by identification (ID) as shown. For simplicity, only one set of collector amplifiers is shown in figure 5–29.

The output signal from the crystal sensor on the impact plate is a ringing sinusoidal wave that increases to a maximum and then decays. After amplification in a tuned amplifier, the peak signal amplitude is used to advance the microphone accumulator, start the register reset (readout of register data), and record the amplitude of the impulse imparted to the microphone sensor plate. The one-shot multivibrator and inhibit block shown in the microphone circuit inhibit further processing of subsequent microphone pulses until after the final pulse is placed in the storage register.

Pulses from the control microphone (not shown in the block diagram) follow a similar, but separate, electronic course with the exceptions that no pulse-height analysis is performed and they do not trigger the register reset.

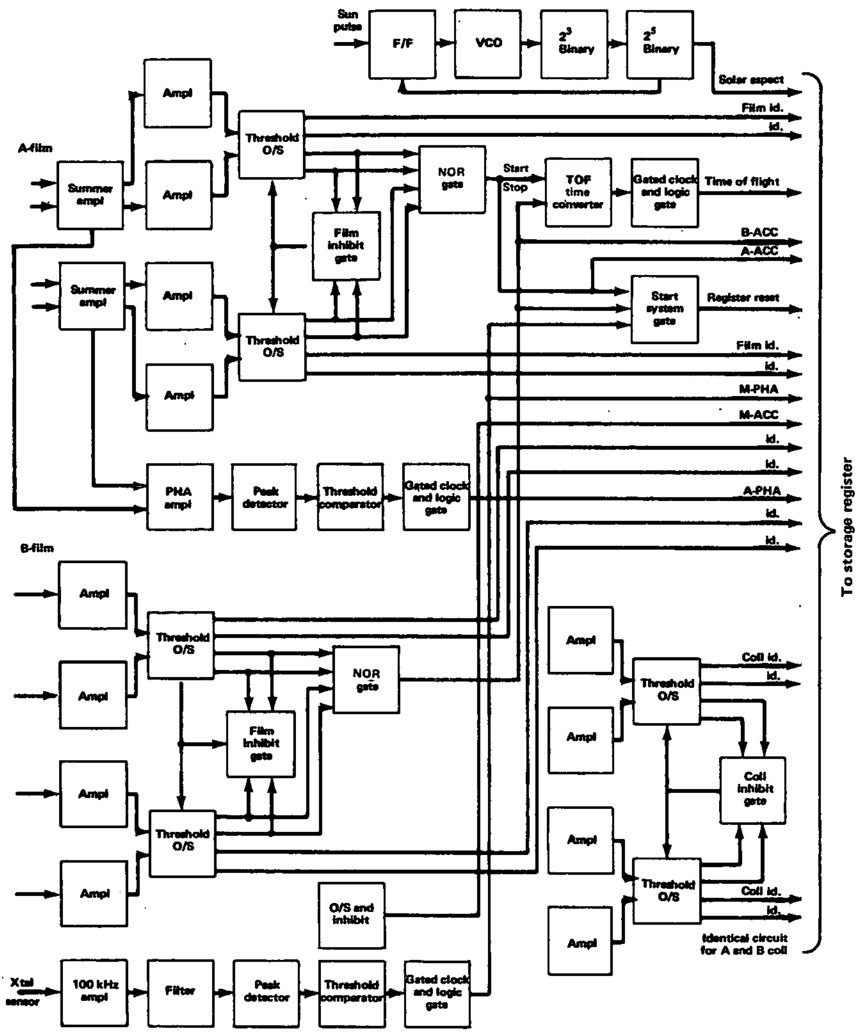


FIGURE 5–29.—Block diagram of the Goddard cosmic-dust experiment.

The data are displayed as 48 bits in four (six-bit) words. This is accomplished by alternately displaying the data in the two formats known as the "0" frame and "1" frame. The first bit in each frame identifies the frame. The next eight bits in the "0" frame identify the "A" film strip and "B" grid column affected by a cosmic dust particle impact. Bits 10 and 11 record the number of events measured by the control microphone. Six bits are assigned to time-of-flight for projectiles in the velocity range of

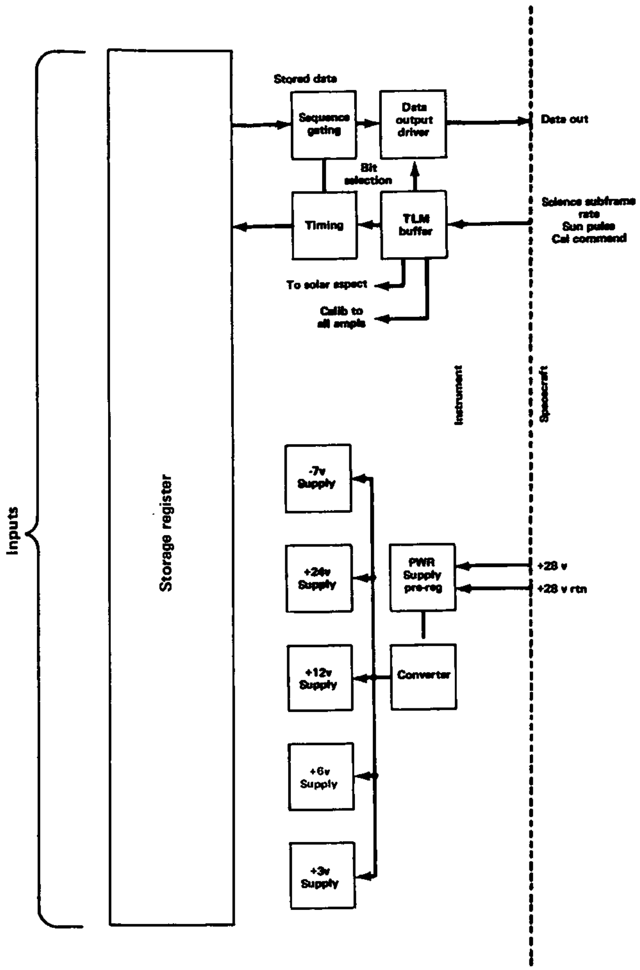


FIGURE 5-29.—Concluded.—Block diagram of the Goddard cosmic-dust experiment.

2 to 72 km/sec, which corresponds to a time-of-flight range of 2.5×10^{-5} to 7×10^{-7} sec. Any "A" film event initiates the start of a 4-MHz clock that is stopped by either a "B" film event or a filled register of 63 bits. A solar-aspect counter uses the next six bits of frame "0". This device starts its count upon each revolution of the spacecraft at a time when the Sun sensor sees the Sun. The last bit in frame "0" provides an experiment parity check.

The next eight bits following frame identification in frame "1" are used for "B" film strip and "B" grid column identification for the rear sensor array. A single bit is used to indicate signal noise that may have occurred during pulse-height analysis of any "A" film event or microphone event. Bits 11 and 12 of frame "1" register the total number of main-microphone events; bits 13 and 14 register the accumulated number of "B" film events. The "A" film pulse-height analysis and microphone pulse-height analysis are registered on the next six bits. The remaining four bits are assigned to the display of accumulated "A" film hits. All of the data on both formats remain, and are repetitively displayed, until an event occurs involving the "A" film, the "B" film, or the microphones.

THE JPL CELESTIAL MECHANICS EXPERIMENT (PIONEERS 6, 7, 8, 9, AND E)

The celestial mechanics experiment required no special equipment on the spacecraft or at the tracking stations (ref. 14). The tracking data provided by the DSN (ch. 8) were sufficiently accurate to support the following primary objectives:

(1) To obtain better measurements of the masses of the Earth and Moon and of the Astronomical Unit

(2) To improve the ephemeris of the Earth

(3) To investigate the possibility of testing the Theory of Relativity using Pioneer tracking data

The methods employed in the analysis of the tracking data are discussed in Volume III, where the results from all experiments are presented.

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CHAPTER 6

The Pioneer Test and Ground Support Program

THE SCIENTIFIC UTILITY of the Pioneer spacecraft rested ultimately upon: (1) their high intrinsic reliability, (2) their ability to withstand the rigors of the space environment, and (3) their "clean" experimental environments—that is, low magnetic field, lack of electromagnetic interference, and favorable temperatures. These attributes did not arise automatically. Experience and good design were vital; so was a deep-searching, comprehensive test program.

The Pioneer test program began at the material and component level and continued until the final moments before launch. The magnitude of the test program, as blocked out in figure 6–1,* is impressive, particularly considering that the Pioneers are relatively small spacecraft. The tests that logically fall within the scope of this volume are those that begin with the component manufacturer, continue at TRW Systems, and end with the preship review and final dispatch of the spacecraft and instruments to Cape Kennedy (far right of fig. 6–1). The prelaunch activities and countdown at the Cape are mainly relegated to Volume III, except the electrical ground support equipment (EGSE) tests and the integrated system tests (ISTs), which are essentially the same as the EGSE tests and ISTs at TRW Systems, the spacecraft contractor.[20]

While the test program delineated in figure 6–1 shows the spacecraft and instrument systems being tested in parallel, Pioneer-unique equipment at Cape Kennedy—the EGSE; and at the tracking stations—the ground operational equipment (GOE) also undergo their own battery of tests before they are committed to a mission. Similarly the DSIF and the Delta launch vehicle systems undergo their own series of tests and checkout.

The highlights of the Pioneer test program—indeed, of almost every spacecraft test program—are the qualification and acceptance tests, and the integrated system tests. Along the way, many special tests are the rule rather than the exception. For example, magnetic testing was much more important in the Pioneer program than in most other spacecraft programs. Boom deployment in space was considered a potential source of difficulty;

* See foldout at end of book.

[20] For a detailed description of the tests performed by TRW Systems—the great bulk of all Pioneer spacecraft tests—see the TRW Final report: Pioneer Spacecraft Project, Final Project Report. TRW Systems, 8830–28, 1969.

so a special ground test was devised to simulate conditions during deployment. Thus, the primary testing sequence is embellished by many special checks not portrayed in figure 6–1. Some of the more critical tests of this type are covered below.

TEST SPECIFICATIONS

Just what constitutes a successful test of a Pioneer component or even the entire spacecraft? Test specifications constitute the standards against which all tests are measured. In view of all of the forces exerted on a Pioneer spacecraft and the abundant interfaces, it is not surprising to find the test specifications rather voluminous (ref. 1). The Pioneer test specifications delineate the 11 classes of tests defined in table 6–1. Test specifications include requirements placed upon the test facilities employed and stipulate exactly how the tests will be made and witnessed, and the documentation required of the contractor. A typical development test matrix is given in table 6–2.

SPACECRAFT MODELS

Throughout tables 6–3 through 6–6, certain spacecraft "models" are mentioned; for example, the "prototype model" of the spacecraft. Likewise, figure 6–1 shows the parallel paths of acceptance and qualification spacecraft models. The flight models are the spacecraft actually intended for flight. They are identical in almost every respect to the prototype model. The flight models are subjected to the milder acceptance tests, while the prototype must survive the stiffer qualification tests. It is proper to look upon the prototype model as a machine the engineers could work with, a machine much like the prototype automobiles the car manufacturers subject to grueling tests and design modifications before they commit a design to the production line.

During the entire Pioneer program, TRW Systems built one prototype and five flight models. Originally, Ames Research Center had adopted the philosophy that a backup spacecraft would be prepared for each flight, but this was soon dropped as too costly. In fact, the Pioneer program could be characterized as austere from the few spares and minimum extra hardware ordered from manufacturers. Pioneer E, which was not part of the original Block-II procurement under which TRW Systems built Pioneers C and D, was assembled from spares and other extra hardware.

A spacecraft program rarely moves directly from the design stage into a completely instrumented prototype. Instead, a succession of cruder models precedes the prototype. During the development stage, when some of the critical engineering questions have not been answered, it is desirable to have various engineering models available to test ideas and to try different arrangements of components and different kinds of materials. These engi-

TABLE 6–1.—*Classes of Tests Used in the Pioneer Program*

Type of test	Description
Parts, materials, and processes	Outgassing tests and magnetic properties tests were used.
Development	Performance verification of breadboards, engineering models, subsystems, etc.—see table 6–2 for a typical test matrix.
Life	Tests conducted to establish failure modes, wearout characteristics, and their effect on spacecraft reliability—many spacecraft parts were subjected to thermal-vacuum tests for over 6 months.
Fabrication	Assemblies and subassemblies were checked during fabrication to assure functional integrity.
Integration	During spacecraft assembly, compatibility was established by electrical continuity tests, rf interference tests, etc.
Assembly qualification	Tests conducted on all spacecraft assemblies [a] under forces usually more severe than those anticipated during launch and interplanetary flight—generally qualification tests were 1.5 times more severe than expected conditions. As indicated in figure 6–1, equipment subjected to qualification testing did not fly. (See table 6–3 for details.)
Spacecraft qualification	Similar to assembly qualification tests, the purpose of these tests was to demonstrate the ability of the spacecraft to meet all performance requirements under conditions much more stringent than those expected during flight. (See table 6–4 for details.) They were conducted on a prototype spacecraft model not intended for flight.
Assembly flight acceptance	Similar to but less severe than the qualification tests, these tests had conditions closely duplicating those expected on the mission; their purpose was to locate latent defects in material and workmanship; assemblies passing these tests might be employed on the flight spacecraft. (See table 6–5 for details.)
Spacecraft flight acceptance	Flight models of the spacecraft were subjected to forces actually expected during mission. (See table 6–6 for details.)
Preflight	Conducted at Cape Kennedy, this test included spacecraft functional tests and spacecraft launch vehicle electrical interface tests. (See Vol. III for details.)
Launch vehicle compatibility	Fit and interface checks were made at the Cape. (See Vol. III for details.)

[a] A spacecraft assembly occupies the level of complexity immediately below the subsystem. An assembly performs some distinctive function in the operation of the overall spacecraft system.

TABLE 6-2.—*Development Test Matrix, Electric Power Subsystem*

Characteristics	System	Solar array	Battery	Battery-current monitor	Battery-temperature monitor	Load-current monitor	Bus-voltage sensor	Battery-voltage sensor	Filter network	Undervoltage circuitry	TWT converter	Equipment converter
Voltage-current		X										
Power output		X	X									
Impedance	X	X	X									
Magnetic effects	X	X	X	X	X	X	X	X	X	X	X	X
Charge-discharge			X									
Ampere-hour capability			X									
Floating mode			X									
Switching capability	X		X									
Dynamic range				X	X	X	X	X				
Resolution				X	X	X	X	X				
Power input				X	X	X	X	X		X	X	X
Noise generation	X	X	X	X	X	X	X	X	X		X	X
Charge requirements			X						X			
Dropout voltage										X		
Sensitivity				X	X	X	X	X		X		
Time delay										X		
Operation after undervoltage	X									X		
General operation	X									X		
Noise susceptibility	X									X		
Solar array operating point	X	X										
Loads	X											
Beat notes	X											
Ground loops	X	X	X	X	X	X	X	X	X	X	X	X

neering models vary in sophistication depending upon their purpose. During the Pioneer program the following types of engineering models were fabricated: (1) a structural model, (2) a thermal model, (3) an electrical development model,[21] and (4) an antenna model. These models were, in effect, specialized mockups of the spacecraft. Instruments and other components were simulated (where necessary) by inert pieces of

[21] The electrical development model was commonly called the "engineering model" during the program. Its accomplishments included the establishment of overall spacecraft electrical compatibility, spacecraft/EGSE compatibility, DSIF compatibility, spacecraft/computer program evaluation, and others.

TABLE 6–3.—*Assembly Qualification Test Details*

Type of test	Description
Humidity	This test aimed at preventing damage from humidity during shipping and storage; performance of assemblies was not to be degraded by 24 hr in humidity chamber at: 86 ± 5° F; humidity, 95^{+3}_{-5} percent.
Vibration	Assemblies were vibrated in each of the three orthogonal axes. The specific frequencies, durations, etc., are too lengthy to list here; they included both sinusoidal and random-vibration test schedules.
Acceleration	Thrust was 1.5 times the maximum acceleration expected in powered flight. (See ch. 7 for Delta characteristics.) Spin was 185 rpm, as compared to the 60 rpm expected during normal cruise.
Thermal-vacuum	Pressure was less than 10^{-5} torr; temperature was 25° F above the predicted maximum and 25° F below the predicted minimum; during the 24-hr exposure, cold-start capability had to be demonstrated at least three times for cyclically operated components.
Shock	Three shocks of 50 ±5 g peak for 6^{+1}_{-0} msec were applied in each of the three orthogonal directions.
Magnetic	See chapter 3.
Solar array	Calculated performance figures were verified in sunlight at the JPL Table Mountain facility and facilities at Palm Springs.

TABLE 6–4.—*Spacecraft Qualification Test Details*

Type of test	Description
Balance	Prototype spacecraft in spinup configuration was dynamically balanced at 150 ±5 rpm; spin balance weights were added to meet balance specifications.
Spin	Spacecraft was: (1) spun at rate varying linearly between 150 and 190 rpm for period of 20 sec, (2) spun at 80^{+5}_{-0} rpm for sufficient time so that at least one frame of telemetry was received for each ground command mode.
Weight, center of gravity, moment of inertia	These factors were measured and compared with design requirements.
Humidity	This test was the same as the assembly qualification humidity test. (See table 6–3.)
Vibration	This test was similar to the assembly qualification vibration test. (See table 6–3.)
Acceleration	The objective was to test for 3 minutes at 1.5 times the maximum acceleration expected during powered flight (see ch. 7), but this value was never attained during the acceleration tests. Critical stresses on booms were simulated by the addition of weights.

TABLE 6–4.—*Spacecraft Qualification Test Details* (Concluded)

Type of test	Description
Thermal-vacuum	The spacecraft was tested in thermal-vacuum chamber at pressures less than 5×10^{-5} torr, with simulated solar radiation and with chamber walls cooled to $-305 \pm 15°$ F; insolation was simulated between the values expected at 0.8 and 1.2 AU; test duration was at least 9 days; spacecraft was spun fast enough to stabilize temperatures of solar array; and the solar wind was simulated. (These thermal-vacuum tests were not carried out on the prototype model of the spacecraft.)
Magnetic properties	Magnetic measurements were made at 34 specified operating modes. (See ch. 3 for magnetic cleanliness philosophy.) Four types of magnetic tests were conducted on the prototype spacecraft: Type I—spacecraft magnetic field mapping; Type II—spacecraft stray field; Type III—solar array mapping; and Type IV—solar-array stray fields.
Electromagnetic compatibility	Systems performance tests were to assure that no subsystems were adversely affected by the operation of the rest of the subsystems.
Boom deployment	These tests under near-zero-g conditions at expected spin rates were to see if all booms deployed satisfactorily; a structural model was used rather than the prototype.
Subsystem qualification	Some of the more critical subsystems were qualification-tested separately.
Spacecraft/solar-array electrical compatibility	Compatibility of complete spacecraft with flight power supply was tested during simulated operational conditions.

metal or other material. In a sense, these models paralleled the customary "breadboarding" of electrical assemblies, but at the spacecraft level.

TEST FACILITIES

Extensive test facilities are essential to the success of any spacecraft development program. In the case of Pioneer, most of the requisite facilities were located at TRW Systems. A few of the more important facilities are described briefly below.

To duplicate the interplanetary environment accurately, the TRW Systems 30-ft thermal-vacuum test chamber (fig. 6–2) was used for Pioneers A through C and their 22 × 45-ft chamber (fig. 6–3) for Pioneers D and E. The 30-ft chamber's inside diameter is 28 ft. The temperature limits are -320 to $+440°$ F, easily meeting the Pioneer qualification test requirements (table 6–3). The pressure within this chamber can be pumped down to about 10^{-6} torr, again exceeding the Pioneer test requirements. The

TABLE 6–5.—*Assembly Acceptance Test Details*

Type of test	Description
Vibration	Similar to qualification tests except that only sinusoidal vibration schedule used; levels not exceeding expected flight levels
Thermal-vacuum	Tests conducted within maximum and minimum expected temperatures only for 12 hr; otherwise similar to qualification tests
Magnetic properties	Same as qualification tests (table 6–3)
Solar array	Same as qualification tests (table 6–3)

TABLE 6–6.—*Spacecraft Acceptance Test Details*

Type of test	Description
Initial balance	Similar to qualification tests, except that degree of balance had to be within 1.5 times the values specified
Vibration	Similar to qualification tests; random and sinusoidal vibration schedules used
Thermal-vacuum	Similar to qualification tests, except test lasted only 7 days (table 6–4)
Final balance	Balanced prior to shipment to Cape Kennedy and (for Pioneer 6 only) again before mating with live third-stage motor; balance weights added to bring degree of balance within stipulated values

chamber walls are cooled cryogenically to simulate the blackness of space away from the Sun. In the 22 × 45-ft chamber the Sun was simulated by a large reflector and a high-power xenon arc.

Because the Pioneers each carried plasma probes to measure the plasma stream outward from the Sun, it was thought advisable to bombard the spacecraft with an artificial plasma beam to see how the probes would respond. These tests were carried out while the spacecraft was spinning in the thermal-vacuum chamber. A Kaufman ion source was used to generate positive ions which were then mixed with electrons to form a neutral plasma (ref. 2).

The magnetic cleanliness campaign required special test equipment (ref. 3). Fortunately, the Pioneer spacecraft is rather small and it was possible to employ the 6.5-m-diameter Fanselau coil system located at Malibu, California (fig. 6–4). With the Fanselau coils, the ambient field can be nulled out completely so that the permanent field of the spacecraft can be measured directly. Once the permanent field has been measured,

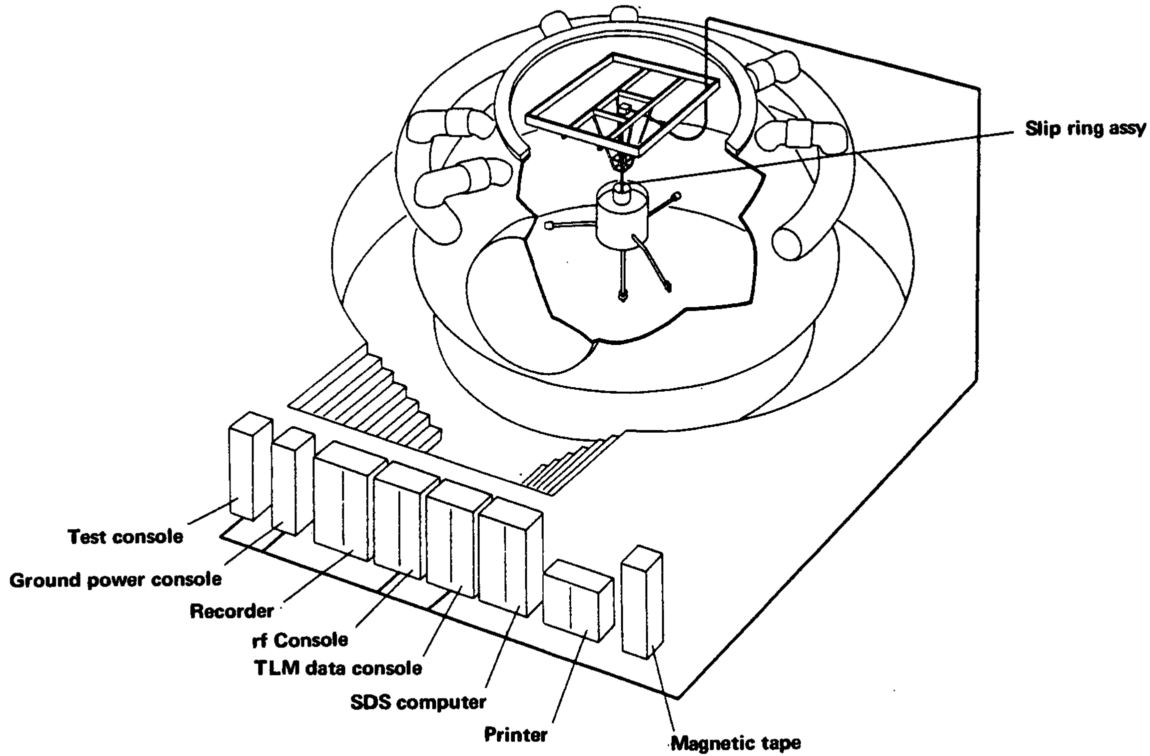

FIGURE 6–2.—Configuration of equipment around the TRW Systems 30-ft thermal-vacuum chamber during tests of Pioneers A, B, and C.

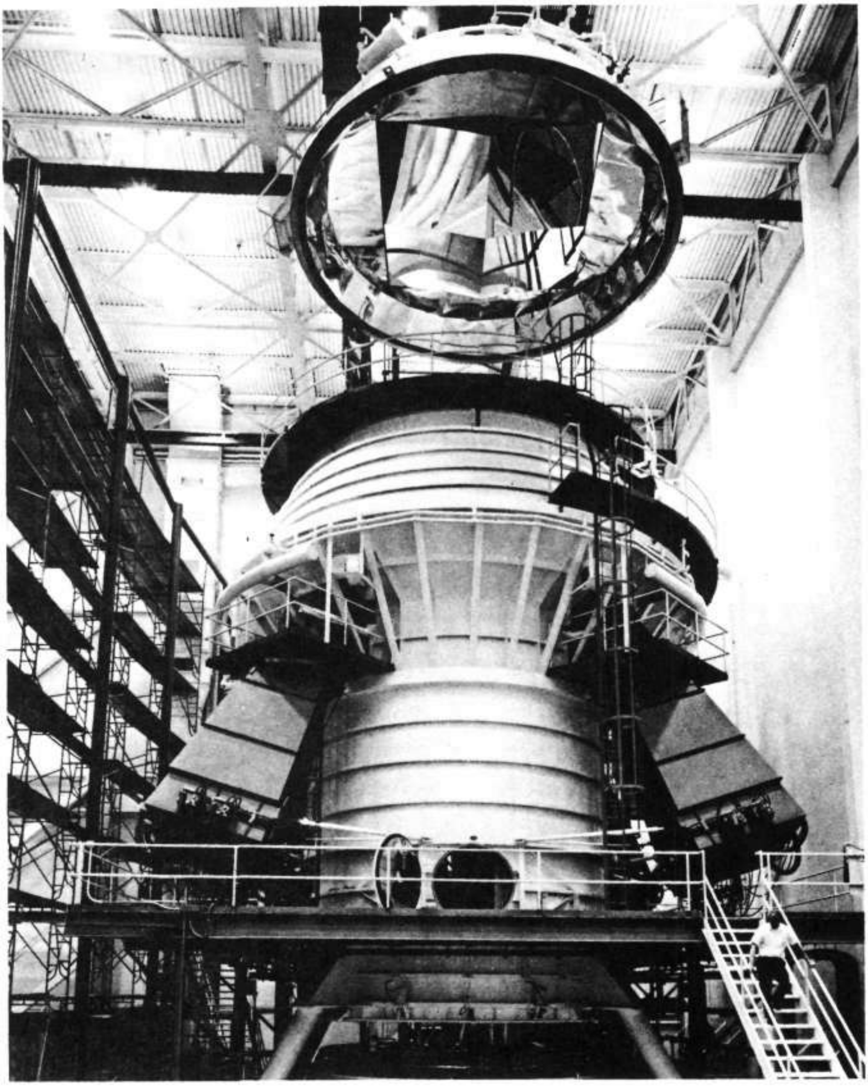

FIGURE 6-3.—The TRW Systems 22 × 45-ft thermal-vacuum chamber and solar simulator used during the tests of Pioneers D and E. (Courtesy of TRW Systems.)

known fields can be applied along each axis to determine induced fields. (See the discussion of magnetic cleanliness in ch. 3.) Testing was done at night to avoid the daytime variations in the Earth magnetic field.

The Pioneer vibration test configuration is shown in figure 6-5. These tests employed a standard shake table made available at the TRW Systems Structural Test Laboratory. Another piece of pertinent equipment at this

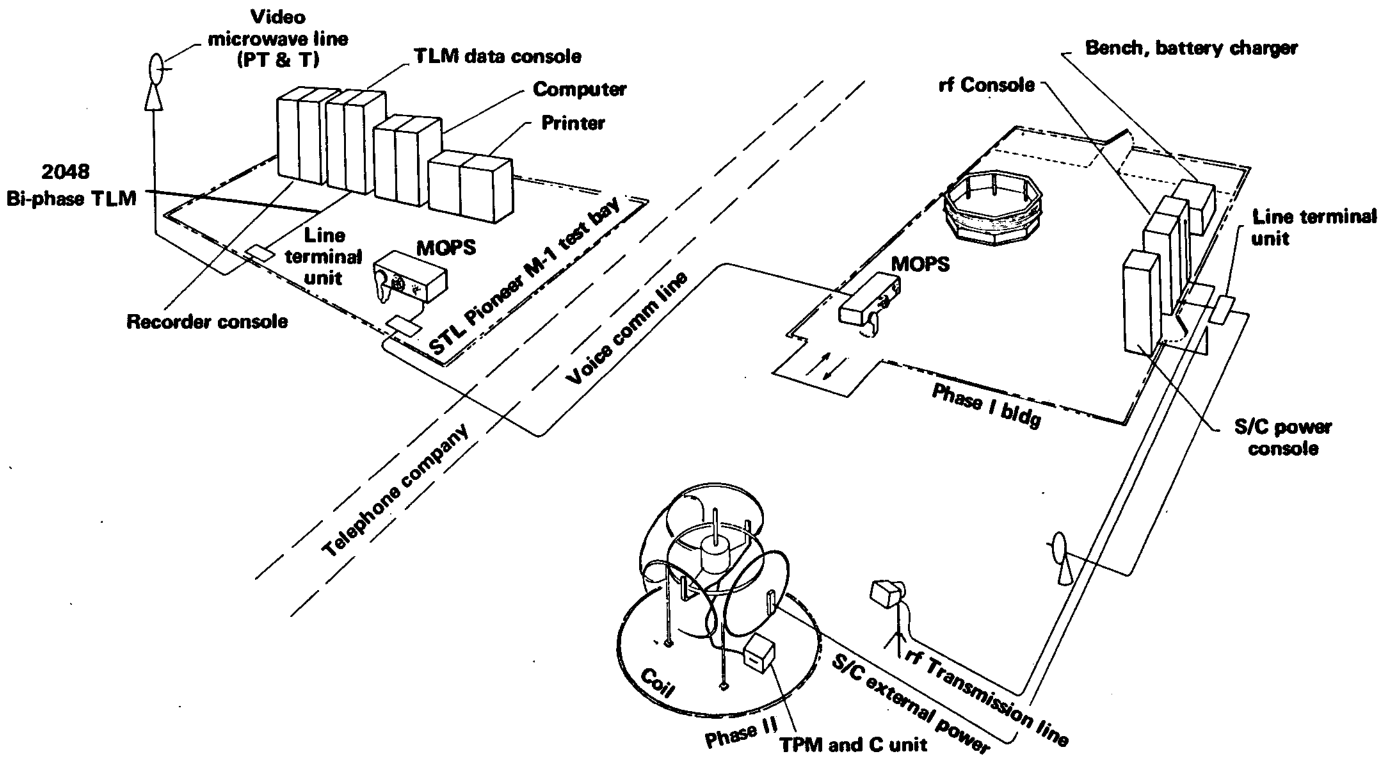

FIGURE 6–4.—Configuration of equipment during Pioneer-A magnetic-field tests at Malibu, Calif. The microwave and telephone links were not used on subsequent spacecraft tests.

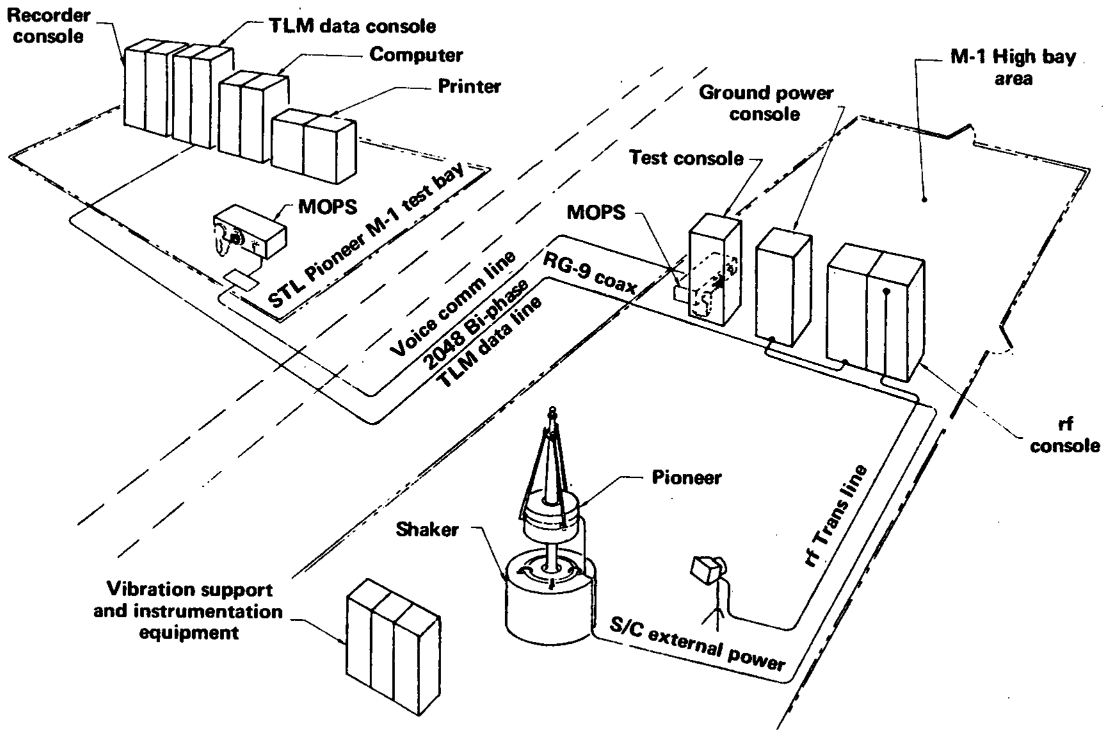

FIGURE 6-5.—Configuration of equipment during Pioneer spacecraft vibration tests.

laboratory was a dynamic balancing machine upon which degree of balance and moments of inertia could be measured.

To test the boom deployment sequence under simulated space conditions, the spacecraft was first spun up to about 110 rpm on a spin table, as shown in figure 6–6. Zero-g conditions were then simulated by a vertical lift of the entire spinning spacecraft (ref. 4). During the coast phase, the booms were deployed. Deployment was observed visually to check the smoothness of the operation, and the possible introduction of spacecraft wobble. A nine-channel telemetry system transmitted additional information on joint angles and stresses during deployment. The entire deployment scheme was a worrisome point during the program. Fortunately, these tests showed that the design was sound.

The solar array was also subjected to a special series of tests (ref. 5). A special outdoor test fixture was constructed that could accept either engineering model panels or the complete solar array. The test fixture consisted of a rotating dummy spacecraft, a simulated electronic load, temperature and power output monitoring devices with sliprings, and a large collimating tube plus standard-cell mount that could be pointed at the Sun (fig. 6–7). The initial test was set up at the JPL Table Mountain site in California; the rest of the panels were tested at Palm Springs.

SPACECRAFT INTEGRATION AND TEST PROCEDURES

The test cycle begins with spacecraft component tests and continues through a graduated series of production tests to final assembly. Each assembly undergoes the complete regimen of environmental and functional tests described earlier. Figure 6–1 shows the overall plan on a broad scale. Upon completion of the assembly tests, subsystems can begin to be integrated into the spacecraft. The spacecraft-integration-and-test phase is that portion of the test program that begins with the receipt of accepted hardware from bonded store and goes through all spacecraft tests to launch. (See sheet 2 of fig. 6–1.) A summary of the tests performed during this phase is presented in table 6–7. Figure 6–8 presents a typical test history from spacecraft integration to the launch pad.

A "building block" philosophy is utilized during subsystem integration to ensure mutual compatibility and proper operation of each subsystem (ref. 6). Basically, there are two kinds of tests: (1) those that evaluate the assembly or subsystem as a unit, and (2) those that explore its interactions with other assemblies and subsystems. As the spacecraft was built up piece by piece, tests were repeated to verify that newly added subsystems (including the instruments) did not interfere with or degrade the performance of those already installed.

The Integrated System Tests are so important that a separate discussion is warranted. The ISTs provide an overall spacecraft performance baseline

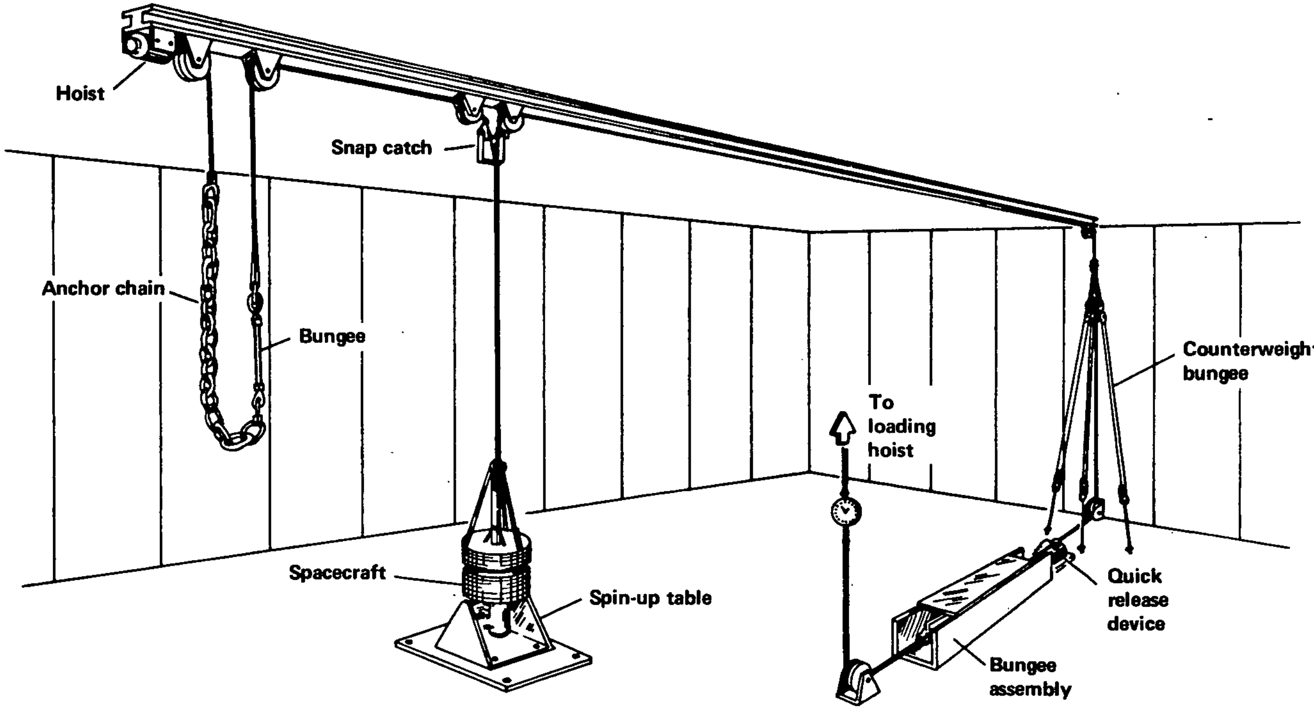

FIGURE 6–6.—The vertical-lift equipment and spin table used during boom deployment test.

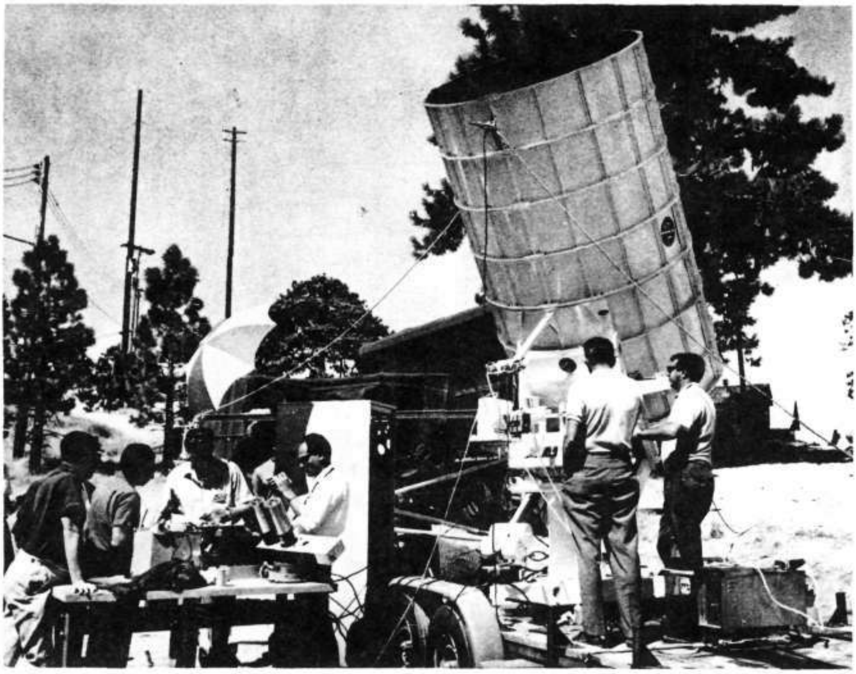

FIGURE 6-7.—Solar-array outdoor test setup.

TABLE 6-7.—*Tests Performed During Spacecraft Integration*

Type of test	Description
Electromagnetic compatibility	The spacecraft, including the instruments, was operated in all proper operating modes within an rf-shielded enclosure; interference and susceptibility tests were performed on the integrated spacecraft.
Telemetry calibration	Projected telemetry outputs were compared with actual signal measurements.
Power profile	Spacecraft load currents were measured under varying input voltages for different spacecraft operating modes.
Subsystem performance	These tests made various parameter checks that could not be ascertained during the Integrated System Test, such as receiver sensitivity, solar-array back emf, and Sun sensor and DTU threshold checks.
Integrated system test	See body of text.
Physical and magnetic properties	These tests verified weight, center of gravity, moments of inertia, and magnetic cleanliness, as described earlier.
Detailed equipment test	Each scientific instrument was tested in detail, with all other instruments and all spacecraft subsystems operating.

whereby changes in performance throughout the test program can be detected. Deviations between the prototype and various flight models can also be checked. The IST was repeated frequently so that engineers could see trends that might develop due to subsystem degradation. The ISTs measured the most important system-wide parameters with the spacecraft as near flight configuration as possible. Some of the parameters measured during the IST are listed below:

(1) Each receiver was frequency-addressed and its performance verified. The quality of the demodulated command signals fed to the decoder was verified.

(2) The rf power radiated from the spacecraft was verified in various spacecraft modes.

(3) The CDU operation was verified in all spacecraft modes, including ordnance-control circuitry and undervoltage output signals.

(4) Simulated Sun pulses were inserted and the pneumatic pressure switch was monitored to check the operation of the pneumatic equipment.

(5) Bit rates, bit formats, and the quality of the data transmitted were monitored to check the DTU and the DSU.

(6) Spacecraft bus current was monitored continuously and compared with the nominal power profile.

(7) Each scientific instrument was tested to verify performance during typical operation.

In a sense, the IST was a thorough physical examination for the complete operating spacecraft. The IST was repeated at least twice for each spacecraft when it reached the launch pad.

ELECTRICAL GROUND SUPPORT EQUIPMENT

Although the operations at Cape Kennedy prior to launch are covered in Volume III, it is pertinent here to describe the EGSE, installed specifically to carry out prelaunch spacecraft tests, especially the "on stand" IST.

The EGSE contains the command generators, the telemetry acquisition equipment, and the necessary data processing, display, and recording equipment to carry out ISTs. A block diagram of the EGSE is shown in figure 6–9.

The prime communication path between the EGSE and the spacecraft was an rf link; this made the tests before launch as realistic as possible. A few hardlines were employed to transmit certain simulation and fault-isolation signals—for example, Sun-sensor signals; but these did not compromise the validity of the tests.

The system test station at the Cape consisted of the equipment racks and associated peripheral equipment shown in figure 6–10. The functions of the various consoles were as follows:

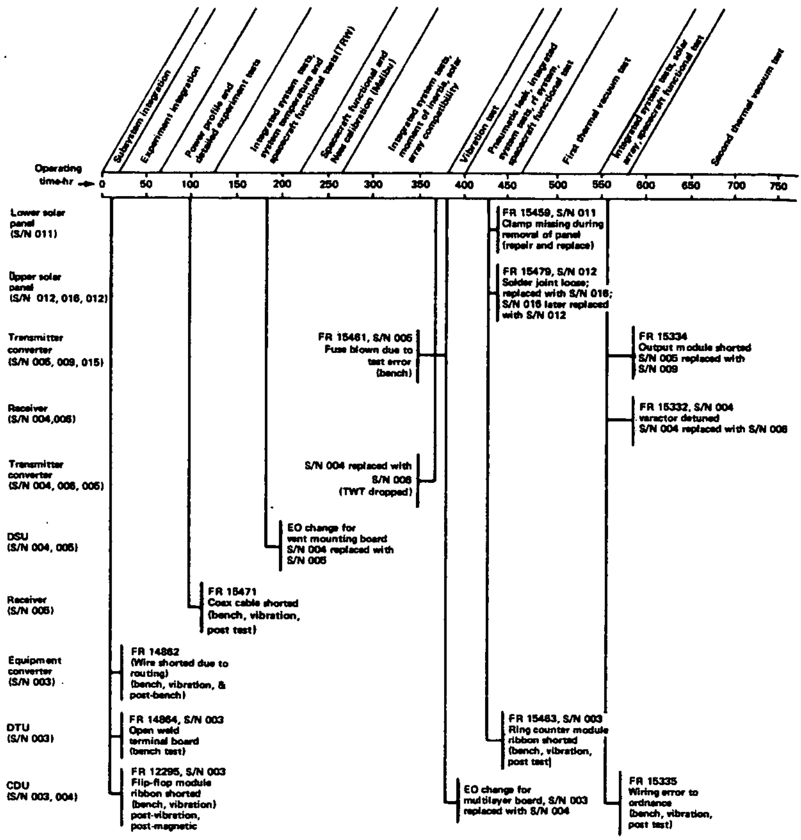

FIGURE 6–8.—Pioneer A spacecraft test history.

(1) Radio frequency console contained command transmitting and data receiving equipment; viz, command encoder, command transmitter, ramp generator, antenna, and telemetry receiver.

(2) Telemetry data console contained equipment which collected, processed, displayed, and recorded the telemetry data received. An SDS–910 computer controlled the data handling, and established frame synchronization and other similar functions.

(3) Recorder console consisted of an instrument patch panel, a strip chart recorder, and a magnetic tape recorder.

(4) Test console was used to support detailed subsystem performance tests. The three major assemblies were a test-point monitor and control assembly, a Sun sensor simulator, and a Sun sensor stimulator.

(5) Ground power console provided primary power to the spacecraft during tests.

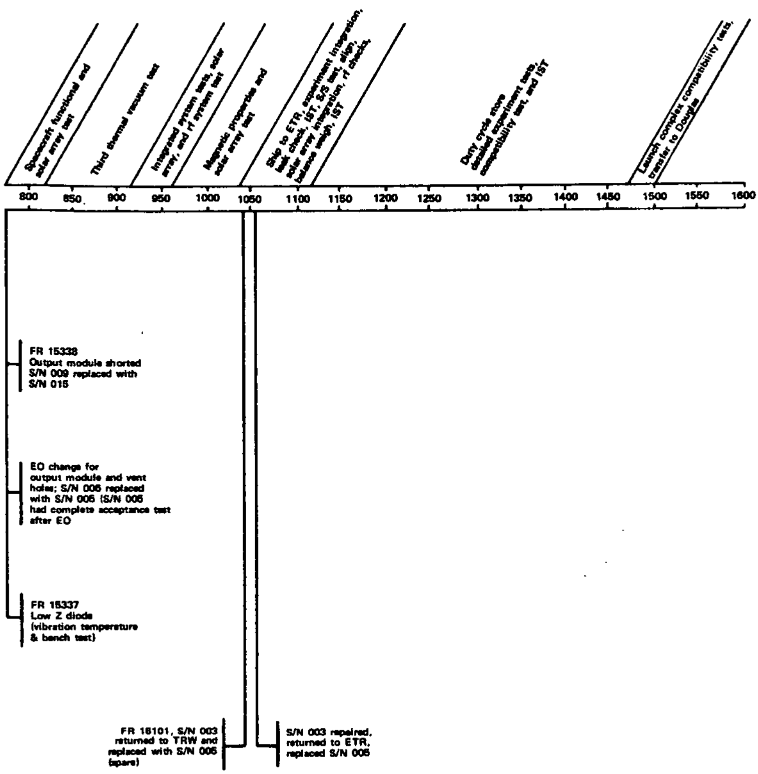

FIGURE 6–8.—Concluded.—Pioneer-A spacecraft test history.

TYPICAL TEST RESULTS

The purpose of any test program is to improve the probability that the spacecraft will perform satisfactorily in space for the desired design lifetime. Since the four Pioneers that were successfully launched have greatly surpassed their nominal 6-month design life, the test program must have been singularly successful in weeding out incipient failures and in pinpointing design weaknesses.

Some of the feedback from the test program into spacecraft engineering is shown in figures 6–11 and 6–12. The electronic subsystems and assemblies required the greatest rework and redesign. The coaxial switches were of particular concern to the engineers.

Another view of the overall test program is shown in figure 6–11. The acceptance portion of the program is shown to be the most important in

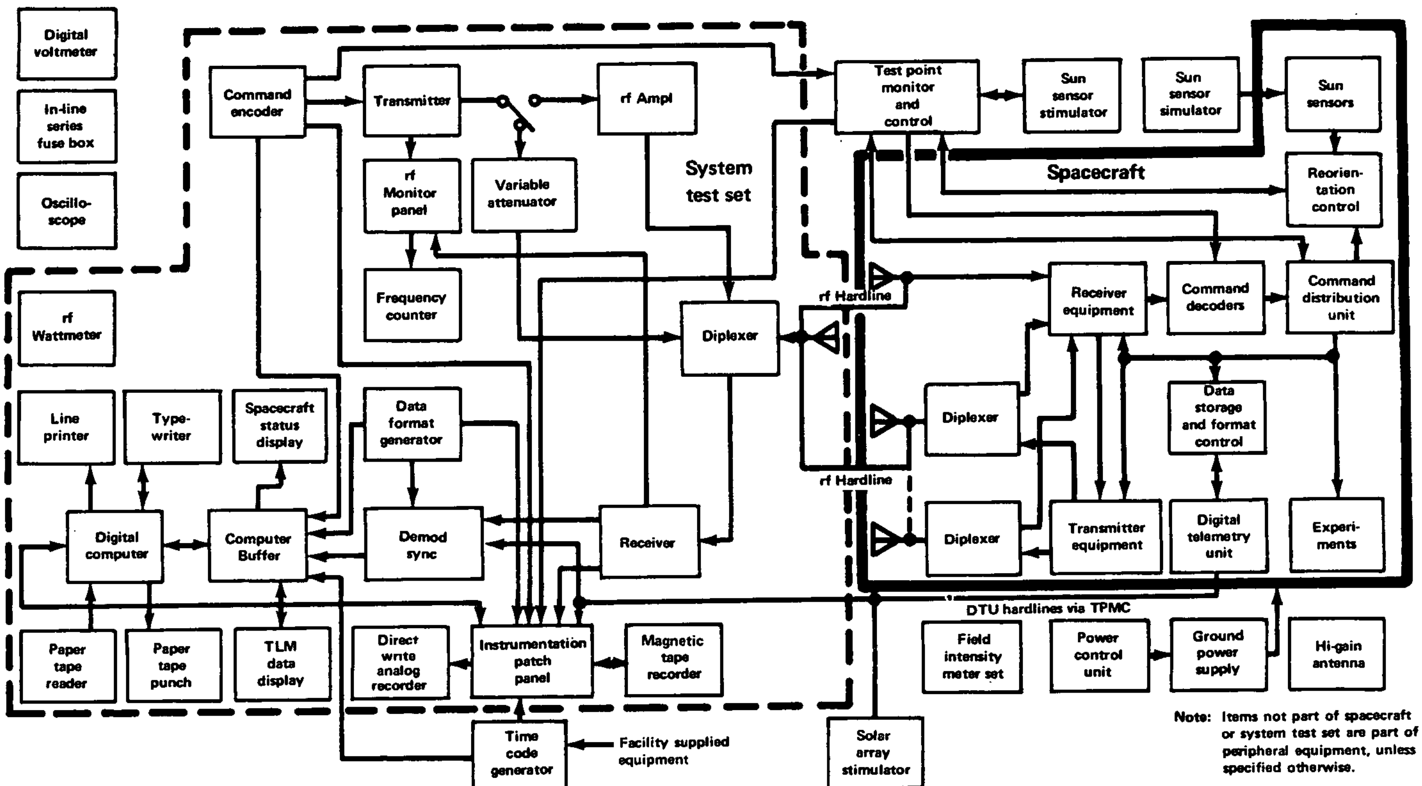

FIGURE 6–9.—Block diagram of the EGSE.

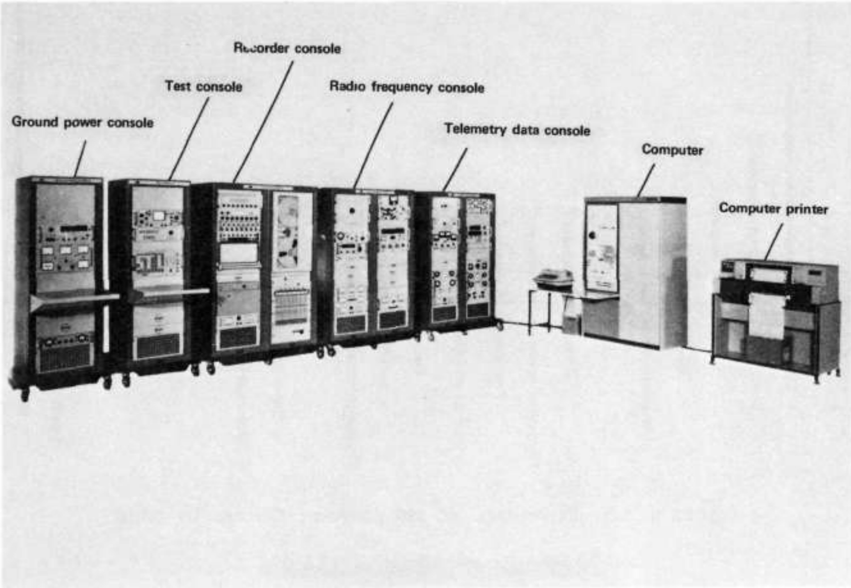

FIGURE 6-10.—Electrical ground support equipment (EGSE) consoles.

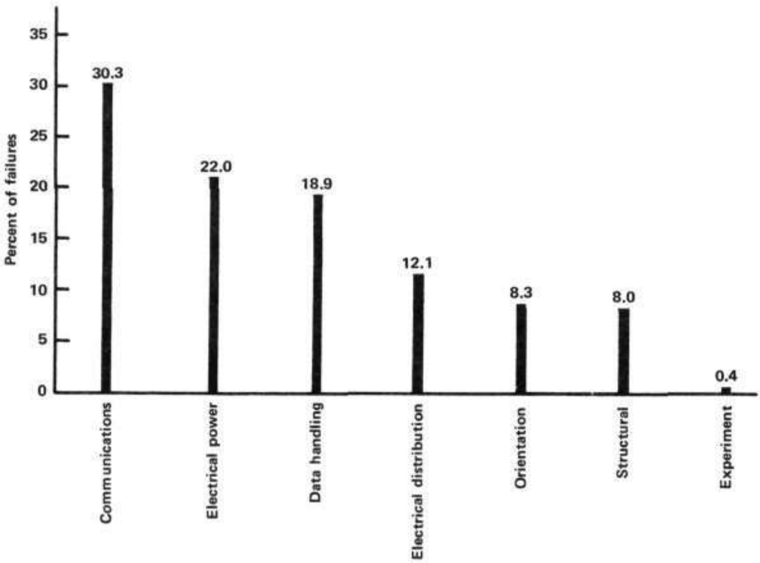

FIGURE 6-11.—Percentage of test-program failures by subsystem.

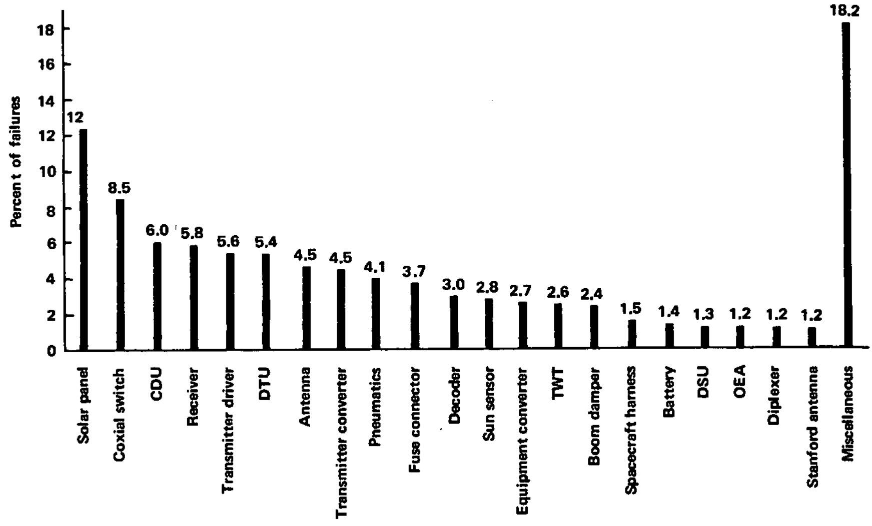

FIGURE 6-12.—Percentage of test-program failures by unit.

terms of detecting failures. It is rather surprising that the qualification tests, which were more severe than the acceptance tests, did not encounter more failures. A possible explanation of the difference lies in the fact that the qualification tests were made only on qualification units, while all flight units, plus the spares, had to pass the acceptance tests.

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CHAPTER 7

The Delta Launch Vehicle

WHY THE DELTA?

THE DELTA LAUNCH VEHICLE, or Thor-Delta, has been one of NASA's most successful launch vehicles (fig. 7–1). As of July 1, 1970, 71 Delta launches have succeeded, with only 7 failures noted on the record books. This remarkable dossier was not available to NASA mission planners in 1961, when the IQSY Pioneers were first brought under discussion. Up to July 1, 1961, the Delta had successfully launched Echo 1, Tiros 2, and Explorer 10, while failing only on its first try (Echo A–1, May 13, 1960). A 75 percent success record was extremely good in 1961. Thus, the planners of Pioneer selected a launch vehicle of high promise.

The Delta had several other points in its favor. It was a low-cost launch vehicle derived largely from previously developed military and Vanguard Program hardware. Its payload capability for the escape mission was something over 100 lb, roughly what NASA wanted for its "precursor" Pioneer. The Delta was also considered NASA's very own launch vehicle, because it had not been obtained through military channels. However, NASA had procured the earlier Thor first stages directly from the U.S. Air Force and then turned them over to the Delta prime contractor, McDonnell-Douglas, who had built them in the first place. NASA was anxious to create its own "stable" of launch vehicles at this point—another plus favoring the choice of the Delta. Further, if history is a valid measure of the future, the Delta was an auspicious choice because Pioneers 1, 2, and 5 had been launched by the Thor-Able launch vehicle, the progenitor of Delta. In particular, the highly successful Pioneer 5 probe had been placed in a solar orbit like those planned for the new series of Pioneers by the Delta-like Thor Able II.

The use of the Delta was thus one of the basic ground rules established for the Pioneer Program in 1961. In the final analysis, it was the only reliable all-NASA launch vehicle capable of doing the job, that would be available for the projected 1964 launch date.

THE EVOLUTION OF THE DELTA

NASA did not develop the Delta as an entirely new launch vehicle; rather, the first Deltas were much-modified Able-II Space Test Vehicles

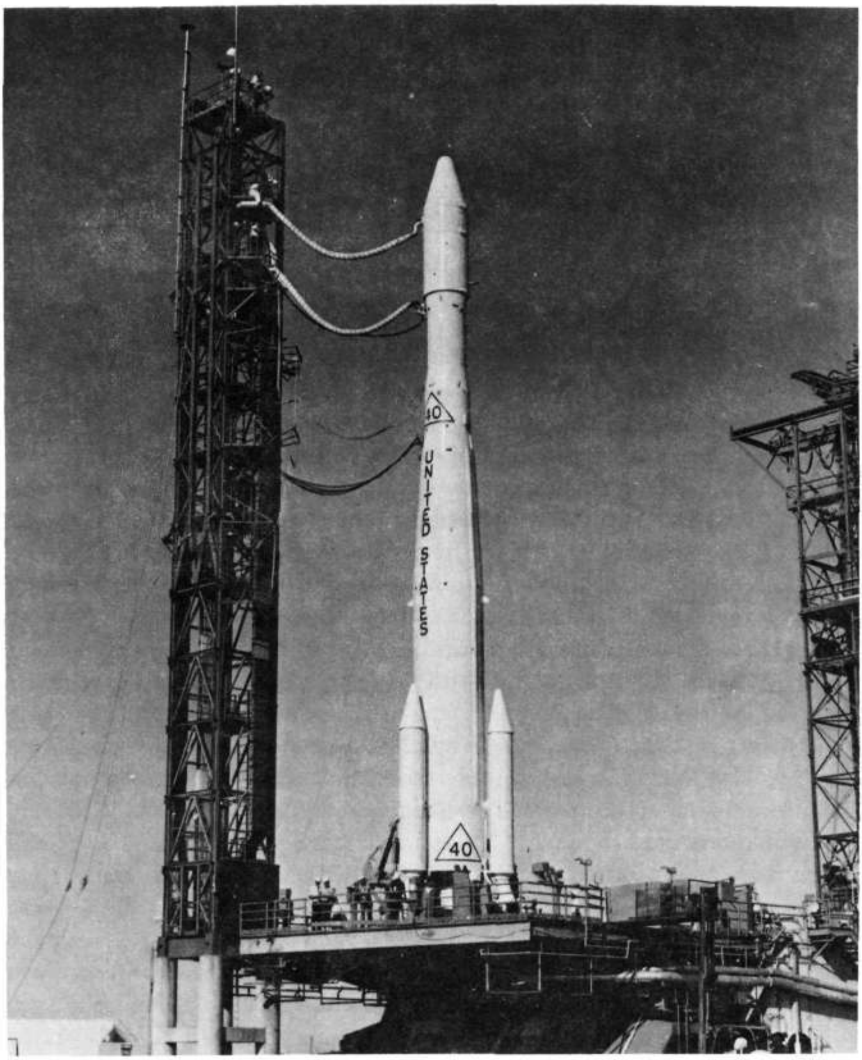

FIGURE 7-1.—The Pioneer-7 launch from Complex 17A at Cape Kennedy. Delta 40 was a thrust-augmented improved Model E launch vehicle.

(STVs), which, in turn, grew out of the Able-II Precisely Guided Reentry Test Vehicle (PGRTV) Program. The precise genealogy becomes confusing here because there were Ables I, II, and IV as well as many other combinations of similar space hardware on the scene. Basically, the Delta rocket owes its first stage to the Thor IRBM;[22] while its second and third stages were based on the Vanguard second and third stages.

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[22] IRBM = Intermediate range ballistic missile

Almost as soon as it was created on October 1, 1958, NASA began to plan its stable of launch vehicles. The favorable record of the Able-II STV led NASA to sign a $24 million contract with Douglas Aircraft on April 29, 1959, for the design and manufacture of 12 Able-based launch vehicles. Originally, the Deltas were intended only as interim launch vehicles for the 1960–1961 period—something to fill the gap while bigger boosters were being developed. As it turned out, the later Deltas could easily and very reliably orbit satellites weighing up to almost 500 lb. This payload was more than ample for most NASA scientific missions. The "interim Delta" did not fade away but became instead a workhorse that has propelled more than three-score spacecraft into orbit around the Earth or Sun.

The Delta is basically a three-stage rocket. The liquid first and second stages are topped by a small solid-propellant third stage (fig. 7–2). The first-stage core is the venerable Thor military rocket, burning a hydrocarbon fuel similar to kerosene (RP–1, RJ–1, etc.) with liquid oxygen. This stage is manufactured by the McDonnell-Douglas Astronautics Company. The three first-stage engines are made by the Rocketdyne Division of North American Rockwell. The solid, thrust-augmentation rockets strapped on the first stages of later models are Castor rockets, usually produced by the Thiokol Chemical Corporation. The fuel for the much smaller second stage is unsymmetrical dimethyl hydrazine (UDMH), which is oxidized by inhibited red fuming nitric acid (IRFNA). The second stage is also a product of McDonnell-Douglas Corporation. It employs an Aerojet-General engine. The third-stage solid rockets have been manufactured by various concerns during the evolution of the Delta: Allegheny Ballistics Laboratory (ABL), United Technology Center (UTC), and Thiokol Chemical Corporation (see table 7–1). The Delta is one of NASA's smaller launch vehicles (first-stage thrust, about 175 000 lb; plus about 160 000 lb from solid strap-ons on later models).

No launch vehicle that has seen as much use as the Delta remains fixed or inflexible for very long. Almost every launch vehicle is different, at least in some minor detail, because the interface with each payload is different. More significant changes arise when rocket motors are uprated, propellant tank sizes are changed, and solid-fuel rockets are strapped on for first-stage augmentation. The Delta has gone through over a dozen of these upratings and improvements as described by the different model numbers in table 7–1.

The Delta variations in physical configuration and terminology are rather confusing to the uninitiated. The following list should relieve this semantic problem:

(1) *Delta* is the basic name for this series of launch vehicles; it is used interchangeably with Thor-Delta. In 1970, the name Delta, used without qualification, meant a TAID. (See below.)

(2) *Thor-Delta* is used interchangeably with Delta.

(3) *Thrust-Augmented Delta* (*TAD*) employs three or more strapped-on

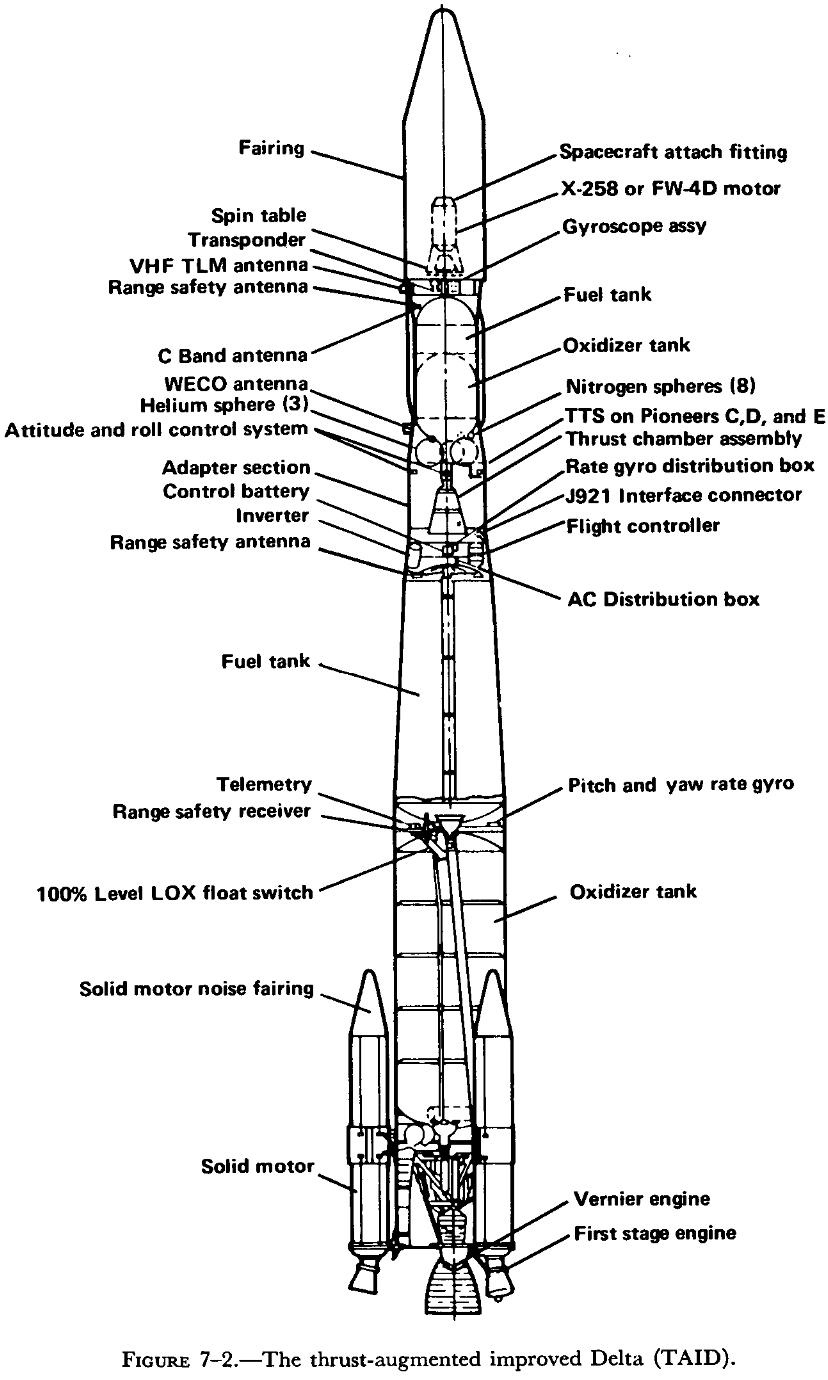


FIGURE 7-2.—The thrust-augmented improved Delta (TAID).

TABLE 7–1.—*Nomenclature of Various Delta Configurations*

Vehicle model number		MDC designation	First-stage propulsion		Second-stage propulsion (Aerojet-General)	Third-stage propulsion	Popular name	Pioneer flights
MDC	NASA		Engine (Rocketdyne)	Augmentation (Thiokol)				
DM–19	Delta	DM–19	MB–3, Block I	None	AJ10–118	NPP X–238	Delta	
DSV–3A	Delta A	DM–21	MB–3, Block II	None	AJ10–118	NPP X–238	Delta	
DSV–3B	Delta B	DSV–2A	MB–3, Block II	None	AJ10–118A	NPP X–248	Delta	
DSV–3C	Delta C	DSV–2A	MB–3, Block II	None	AJ10–118D	ABL X–258	Delta	
						UTC FW–4S	Delta	
DSV–3D	Delta D	DSV–2C	MB–3, Block III	3 Castor I	AJ10–118D	ABL X–258	TAD	
DSV–3E	Delta E	DSV–2C	MB–3, Block III	3 Castor I/II	AJ10–118E	ABL X–258	TAID	Pioneer 6
						UTC FW–4D	TAID	Pioneer 7, 8, 9
DSV–3F	Delta F	DSV–2C	MB–3, Block III	None	AJ10–118E	UTC FW–4D	Imp. Delta	
DSV–3G	Delta G	DSV–2C	MB–3, Block III	3 Castor I/II	AJ10–118E	None	TAID	
DSV–3H	Delta H	DSV–2C	MB–3, Block III	None	AJ10–118E	None	Imp. Delta	
DSV–3J	Delta J	DSV–2C	MB–3, Block III	3 Castor I/II	AJ10–118E	TCC TE–364–3	TAID	
DSV–3L	Delta L	DSV–2L–1B	MB–3, Block III	3 Castor II	AJ10–118E	TCC FW–4D	Long tank Delta	Pioneer E
DSV–3L	Delta M	DSV–2L–1B	MB–3, Block III	3 Castor II	AJ10–118E	TCC TE–364–3	Long tank Delta	
DSV–3L	Delta N	DSV–2L–1B	MB–3, Block III	3 Castor II	AJ10–118E	None	Long tank Delta	

Notes:

MDC = McDonnell-Douglas Astronautics Company
NPP = Naval Propellant Plant
ABL = Allegheny Ballistics Laboratory
UTC = United Technology Center
TCC = Thiokol Chemical Corporation
TAD = Thrust-Augmented Delta
TAID = Thrust-Augmented Improved Delta

solid rockets for first-stage augmentation; most of the later models were TADs.

(4) *Improved Delta* is a delta with the "fat-tank" second stage but no first-stage augmentation.

(5) *Thrust-Augmented Improved Delta* (*ITAD or TAID*) is a Delta "improved" in the mid-1960s by increasing the diameter of the second-stage tank from 32 to 54 in. The burning time of the second stage "fat-tank" Delta was increased from about 160 to about 400 sec.

(6) *Long-Tank Delta* consisted of a Delta with the first-stage tank lengthened by 14.5 ft (fig. 7–3). The first-stage burn time was increased from 150 sec (Pioneer 8) to 221 sec (Pioneer E).

Future changes may involve the adoption of the Titan-III transstage engine and introduction of the Delta Inertial Guidance System (DIGS). The latter would improve the injection accuracy of the first and second stages.

Changes in the third-stage solid rocket motor did not lead to overall name changes, but the model designations did change as the original

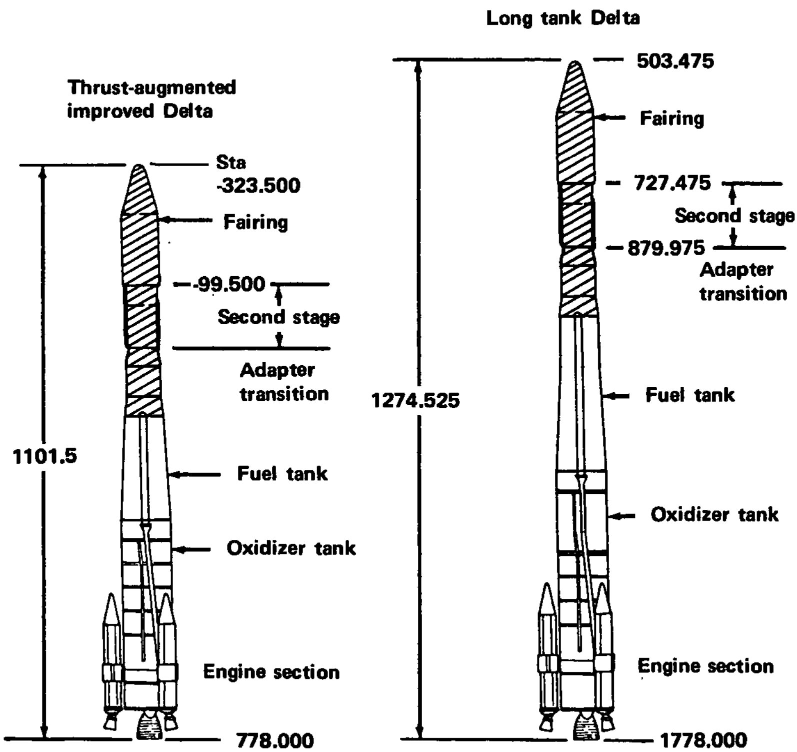


FIGURE 7–3.—Comparative outboard profiles of the TAID and long-tank Delta. (Dimensions are in inches.)

X–248 was replaced by the X–258, which, in turn, was displaced by the FW–4D, etc., as indicated in table 7–1.

The Delta models applied to the Pioneer Program are specified and described in more detail in table 7–2. The Pioneer Deltas came from five points along the evolutionary history of the Deltas, but only four significantly different models were employed:

(1) Pioneer 6: Delta E, TAID with X–258 and Castor I augmentation

(2) Pioneers 7 and 8: Delta E, TAID with FW–4D and Castor I augmentation

(3) Pioneer 9: Delta E, TAID with FW–4D and Castor II augmentation

(4) Pioneer E: Delta L, a long-tank Delta with FW–4D and Castor II augmentation.

THE DELTA-SPACECRAFT INTERFACE

The spacecraft-to-launch-vehicle interface is more subtle than one might suppose. Voluminous documents detail the design restraints that affect spacecraft design. These design restraints are essentially detailed descriptions of the mechanical, spatial, electrical, and thermal interfaces that the spacecraft designer must match if his spacecraft is to fit on the rocket and survive the heat and other forces applied during the launch process.

The hardware manifestations of interface matching are attach fittings, fairings, thermal insulation, and shrouds. Much of the interface matching between spacecraft and launch vehicle occurs where the bottom of the spacecraft physically meets the top of the third stage. Pioneer 6 was launched by a Delta with an X–258 third stage, but the remaining four in the series had to be matched to the FW–4D stage.

A brief description of the FW–4D defines the general physical environment at the top of the Delta. The FW–4D is essentially an encased solid-propellant grain engine with a nozzle at the bottom. The spacecraft attach-fitting (not considered part of the motor) is at the top. The motor proper weighs 663.5 lb before firing and only 58.5 lb afterward. Length and diameter are 59.25 and 19.6 in. respectively. The solid propellant consists of polybutadiene acrylic acid/acrylonitrile (PBAN) binder, ammonium perchlorate oxidizer, and aluminum. From the top down: the attach flange is aluminum; the motor case is made from fiberglass and epoxy resin; a composite material comprises the nozzle; the interstage fittings are aluminum. At an ambient temperature of 75° F, the motor burns for 30.8 sec, with a maximum thrust of 6800 lb, producing a total impulse of 172 700 lb-sec.

Attach Fittings

The Delta Project at Goddard Space Flight Center has developed a wide inventory of attach fittings of various sizes to accommodate different space-

TABLE 7–2.—*Physical Characteristics of the Pioneer Deltas*[a]

Characteristic		Pioneer 6	Pioneer 7	Pioneer 8	Pioneer 9	Pioneer E
Delta popular name		TAID	TAID	TAID	TAID	Long-tank Delta
Delta model number		E	E	E	E	L
Delta launch number		35	40	55	60	73
First stage	Model	DSV–2C	DSV–2C	DSV–2C	DSV–2C	DSV–2L–1B
	Height with adapter (ft)	60.4	60.4	60.4	60.4	70.3
	Diameter (ft)	8	8	8	8	8
	Weight (lb)					186 000
	Sea-level thrust (lb)	175 600	175 600	175 600	175 600	172 000
First stage augmentation	Model	Castor I	Castor I	Castor I	Castor II	Castor II
	Height with nozzle (ft)	24.3	24.3	24.3	24.3	24.3
	Diameter (ft)	2.6	2.6	2.6	2.6	2.6
	Weight (lb)	27 600	27 600	27 600	29 600	29 600
	Sea-level thrust (lb)	161 700	161 700	161 700	156 450	156 450
Second stage	Height (ft)	13	13	13	13	13
	Diameter (ft)	5.8	5.8	5.8	5.8	5.8
	Weight (lb)	14 000	14 000	14 000	14 000	14 000
	Sea-level thrust (lb)	7400	7400	7400	7400	7400
Third stage	Model	X–258	FW–4D	FW–4D	FW–4D	FW–4D
	Height (ft)	5	5	5	5	5
	Diameter (ft)	1.7	1.7	1.7	1.7	1.7
	Weight (lb)	735	663	663	663	663
	Vacuum thrust (lb)	6200	5600	5600	5600	5600
Total launch vehicle	Height with shroud (ft)	92	92	92	92	106
	Weight (lb)	150 000	150 000	150 000	150 000	200 000
	Date	12–16–65	8–17–66	12–13–67	11–8–68	8–27–69
Spacecraft	Nominal weight (lb)	138	138	147	147	147

[a] Thrust and weight figures are approximate.

craft, particularly the larger satellites that can be launched with the more powerful versions of the Delta. The relatively small Pioneer spacecraft however, were attached to the FW–4D and X–258 by a small 9 × 8 in, conical attach fitting (fig. 7–4). A two-piece marmon-type clamp secured by two bolts held the spacecraft in the attach fitting. In flight, the spacecraft were separated from the attach fitting by ordnance cutters that severed both bolts (the severing of one bolt is actually sufficient). Separation springs imparted relative velocities of 6 to 8 ft/sec to the spacecraft.

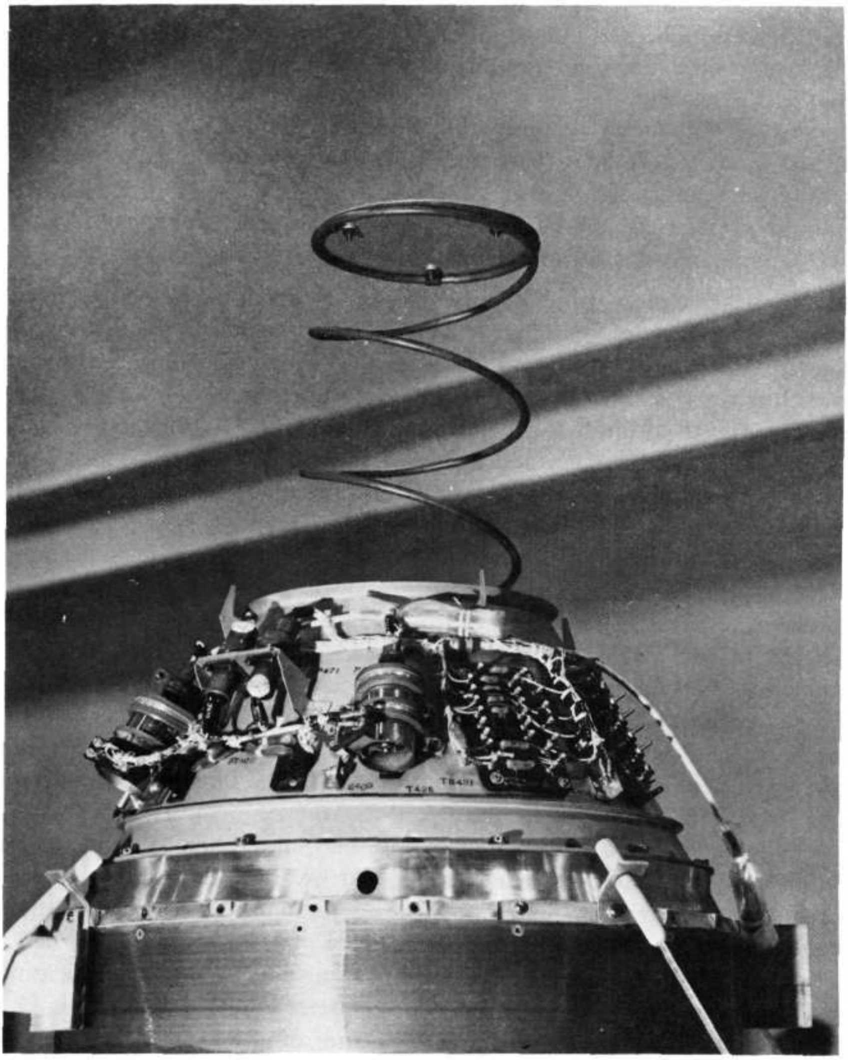


FIGURE 7–4.—The 9 × 8-in. conical attach fitting used on Delta Pioneer launches.

Fairings and Payload Envelopes

During launch, spacecraft were protected from buffeting and aerodynamic heating by a thin-walled fiberglass fairing or shroud. All five Pioneers used the so-called "standard" Delta fairing (fig. 7–5). This protective shell is 224 in. long, 65 in. in diameter, and weighs roughly 535 lb exclusive of any thermal insulation the spacecraft may need.

Once the sensible atmosphere had been breached, the two halves of the fairing were freed by firing explosive bolts. Spring-loaded latches then pushed the fairing halves aside and the spacecraft, plus the second and third stages, proceeded unencumbered.

The interface between the fairing and spacecraft is primarily spatial; that is, the spacecraft must fit within the envelope defined in figure 7–6. Some of the Pioneers required thin layers of thermal insulation on the fairing nose; the amount depended upon the trajectory selected and the resultant heating.

Spacecraft Mechanical Loads During Launch

The rocket motors propelling the Pioneer spacecraft into escape trajectories generate what is termed the "launch environment" for the spacecraft; that is, the sinusoidal, random, acoustic, and shock loads induced during launch affected spacecraft design. The low-frequency sinusoidal excitations occurred mostly at liftoff, during transonic flight, and just prior to first-stage cutoff. Maximum random and acoustic excitations occurred at liftoff and during transonic flight. Shocks were transmitted to the spacecraft when explosive bolts and other pyrotechnic devices detonated. Finally, acceleration or g-loads stressed the spacecraft structure during all phases of powered flight.

The time histories of these mechanical forces vary for each Delta model. The Delta restraint documents specify the payload mechanical environments in detail for each model (ref. 1). A few representative curves presented here should give the reader a feeling for the "dynamic" mechanical interface (figs. 7–6 to 7–9). The Delta restraint graphs are translated into spacecraft design and test specifications (ref. 2). The Pioneer test program is covered in ch. 6 in this volume.

The tests employed shake tables, spin tables, and other equipment that simulates the dynamic environment created by the Deltas. The dynamic forces impressed on the spacecraft during test normally exceeded those stipulated in the Delta restraint publications.

Several of the general design specifications in Pioneer Specification A–6669 were derived from Delta-produced forces. For example:

3.1.10 *Static Balance.* The spacecraft center of gravity shall not be displaced from the spacecraft centerline by a distance greater than 0.015 in.

3.1.11 *Inertial Axes.* The spacecraft principal axes of inertia shall be perpendicular and parallel to the spacecraft centerline within an angle of 0.001 rad.

3.1.12 *Rigidity.* Rigidities of the spacecraft in the launch configuration shall be such as to make all resonant frequencies of the entire structure and/or assemblies greater than 5 cycles per second.

3.2 *Load Factors.* The following critical load factors expressed in gravity units will exist at the spacecraft center of gravity during launch. The side loads are caused by both translational and pitching accelerations.

Condition	Axial load factor (g)	Side load factor (g)
120 sec after ignition of first stage	6.20	1.05
First-stage burnout	14.00	0.70
Third-stage spinup	0	0
Third-stage ignition	$\frac{6485}{587 + (\text{spacecraft weight in lb})}$	0
Third-stage burnout	$\frac{6770}{77 + (\text{spacecraft weight in lb})}$	

The specifications above are taken from NASA–ARC Specification A–6669.07, dated November 13, 1964. Both specifications and launch vehicle restraints changed frequently during the history of the spacecraft program.

Specifications numbered 3.1.10 and 3.1.11 introduce another mechanical force felt by the spacecraft as it ascends from the Earth—centrifugal force. The Pioneer spacecraft were spin-stabilized and, each in the company of a Delta third stage, were "spun up" during third-stage burn to provide dynamic stability, much as rifle bullets are stabilized by spinning. The entire third stage and its spacecraft payload were mounted on a spin table located on the top of the second stage (fig. 7–10). After the fairing had been jettisoned, and prior to third-stage ignition, small rockets mounted with thrust axes tangential to the circular spin table ignited and started the spin table spinning after the fashion of a Fourth-of-July cartwheel. Spinup required dynamic symmetry of the payload around the spin axis, as expressed in specifications 3.1.10 and 3.1.11.

Fairing Heating

As mentioned earlier, the Delta fairing heated up as it ascended through the sensible atmosphere at high velocities. The temperature history of the

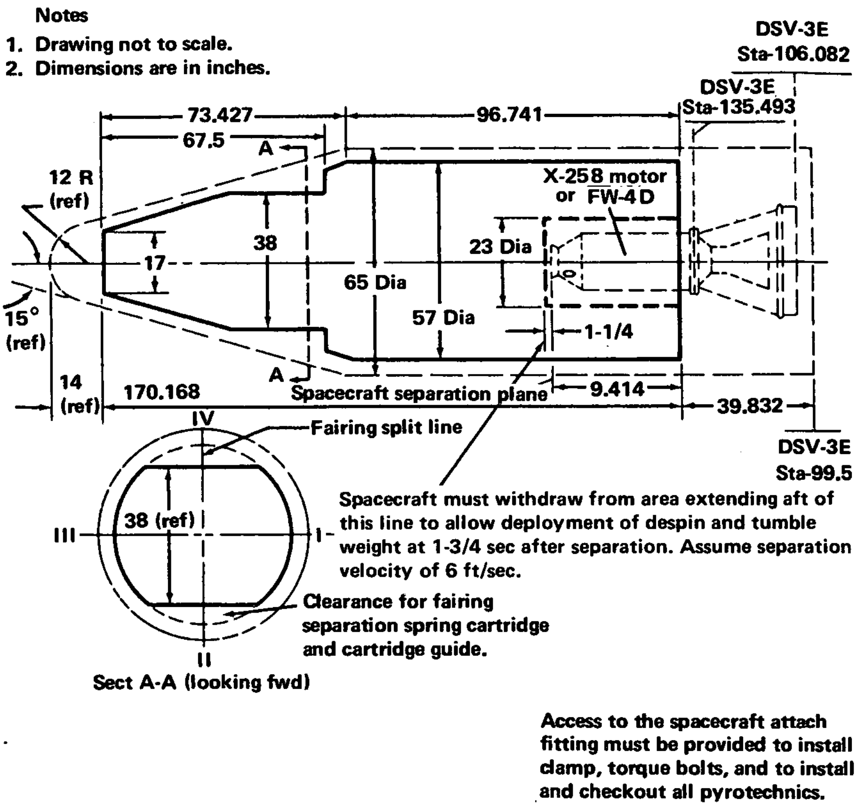


FIGURE 7–5.—Payload envelope of the improved Delta shroud. From: Specification A–6669.07, fig. 3.1.2.

inside of the fairing shows that the temperature varied from point to point and depended upon the actual launch vehicle trajectory. If the inside of the fairing exceeded 500° F, outgassing might have occurred to the detriment of both fairing and payload. Lower temperatures could damage the spacecraft components. It was the responsibility of the launch-vehicle contractor to apply enough insulation to the outside of the fairing to prevent damage. Fairing insulation was a critical matter because some fraction of each pound added—usually 2 to 3 percent—had to be subtracted from the spacecraft weight.

The general approach to this problem in the Pioneer Program was the specification of a reference trajectory and the temperature history of the inside of the fairing without insulation. Figures 7–11 and 7–12 present the trajectory and temperatures stipulated for Pioneer spacecraft. Insulation was applied on the Delta third stage and fairing where needed.

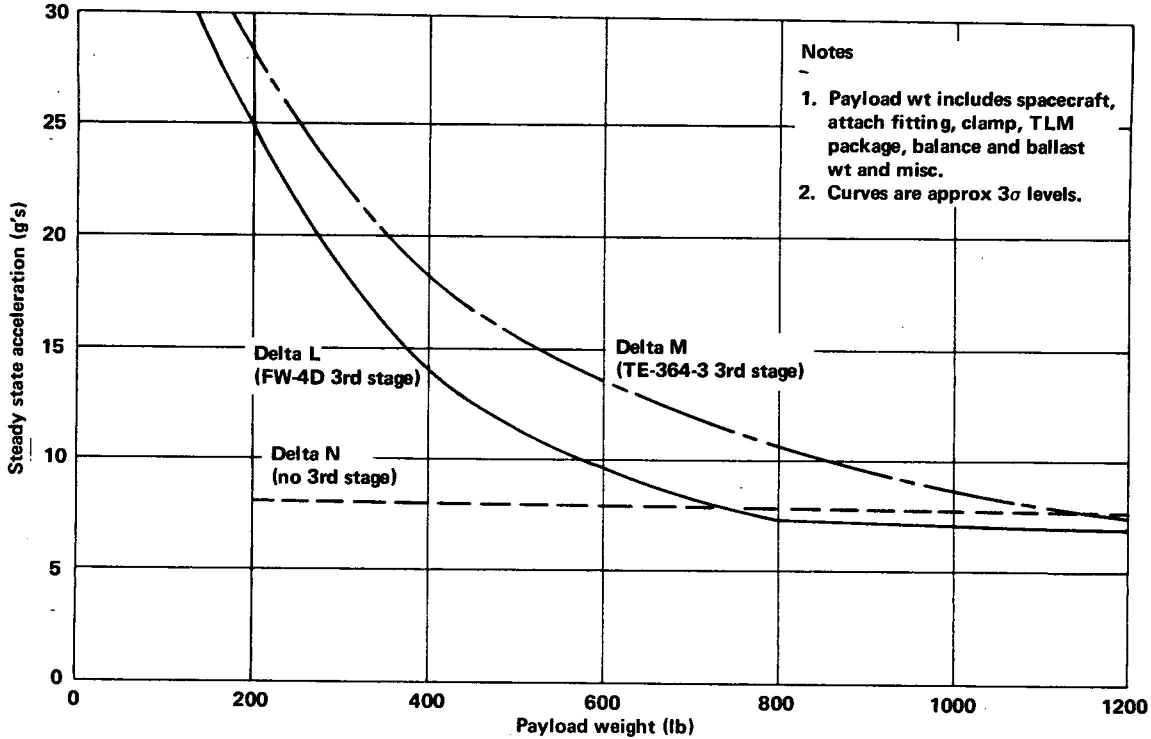


FIGURE 7–6.—Acceleration vs. payload weight for long-tank Deltas.

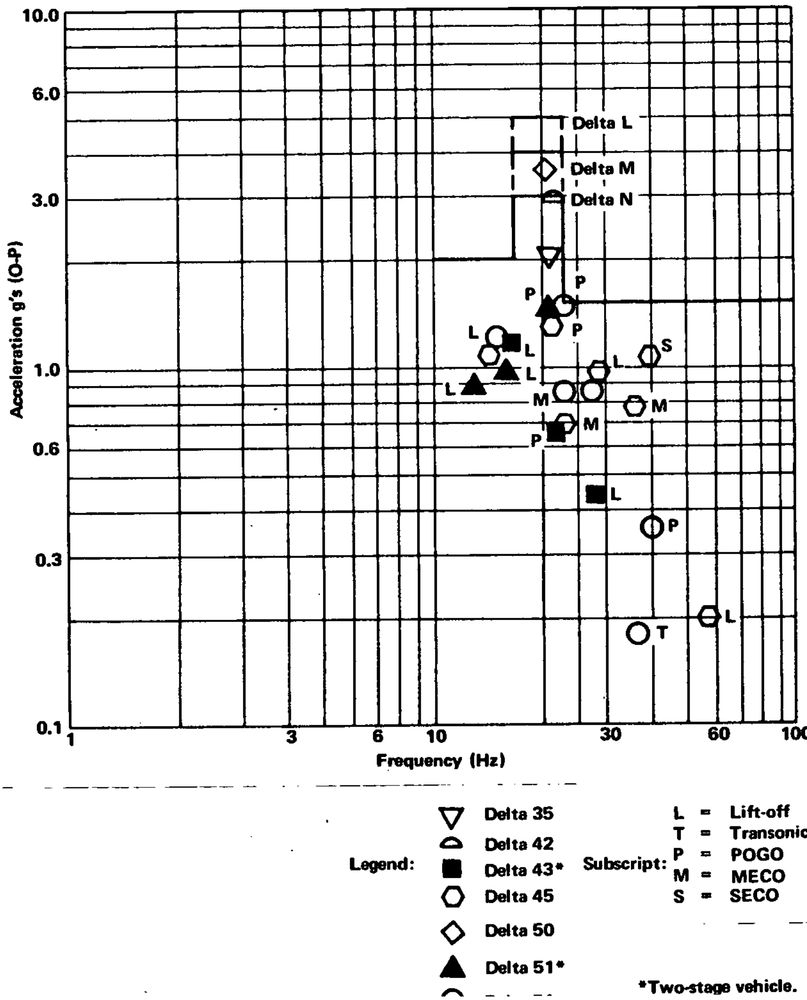


FIGURE 7–7.—Sinusoidal vibrations measured along thrust axis on Models L, M, and N.

OTHER INTERFACES

The other category of interfaces included all the electrical connections that had to be made between the consoles in the blockhouse, through Delta umbilical wiring, to the spacecraft. The spacecraft was checked out and provided with electrical power while on the launch pad, through these wires.

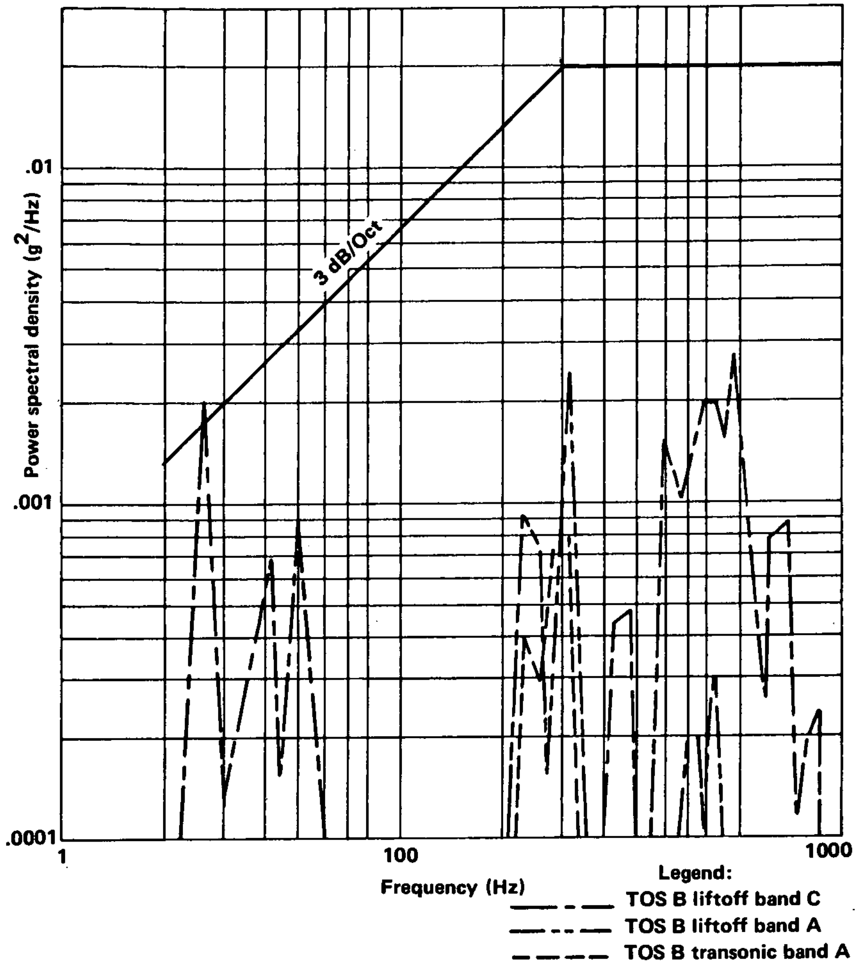


FIGURE 7-8.—Random vibration levels measured along the thrust and lateral axes of Delta L and M vehicles.

TRAJECTORY DESIGN

Each Pioneer launch trajectory was different. The following factors precluded identical trajectories:

(1) The Delta launch vehicle has evolved with the later versions capable of placing much larger payloads into orbit.

(2) Pioneers 6 and 9 were inward-bound (toward the Sun), while Pioneers 7 and 8 were outward-bound.

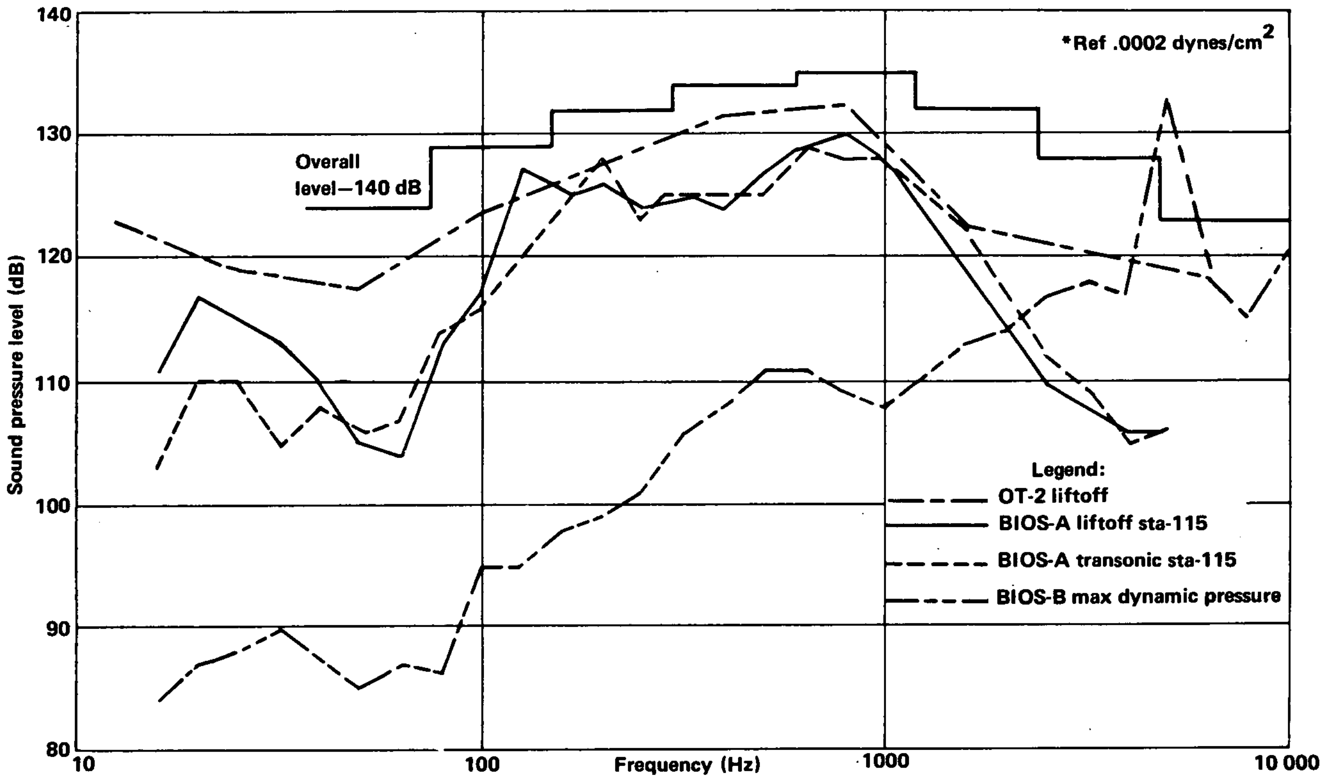


FIGURE 7–9.—Acoustic noise flight levels predicted for the Delta.

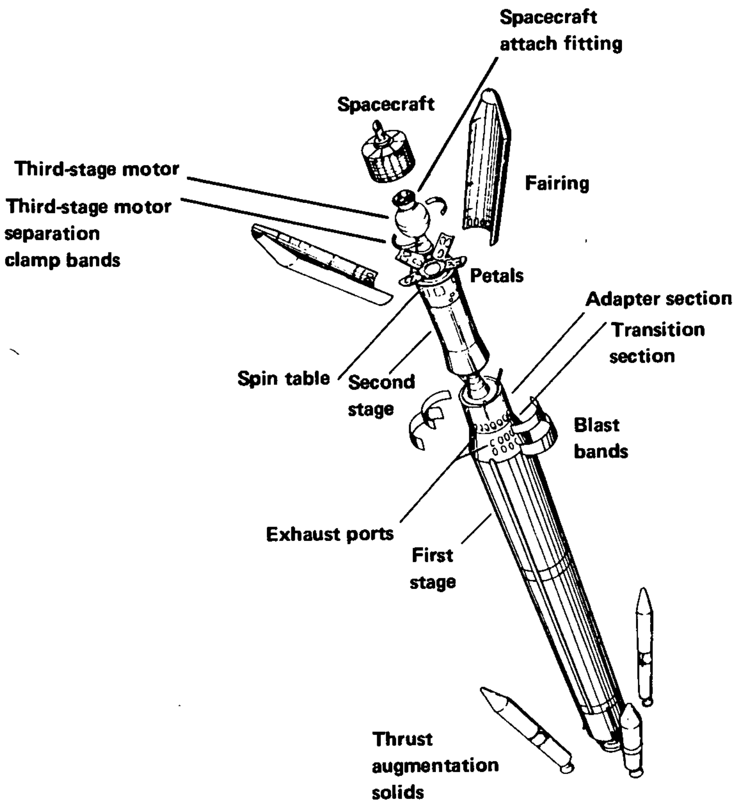


FIGURE 7–10.—Delta staging and separation events, shown here for an Applications Technology Satellite launch.

(3) TETRs were carried "piggyback" on the second stage on the Pioneer 8, 9, and E launches.

In addition to these major factors, the Pioneer payload weights varied slightly.

In very general terms, the Pioneer-carrying Deltas were all launched southeastward from Cape Kennedy along the Eastern Test Range (fig. 7–13). During the coast phase, the Delta passed over Ascension, and the NASA tracking stations in the vicinity of Johannesburg, Republic of South Africa. Roughly 520 sec after liftoff, the second stage cut off and the Delta third stage plus the spacecraft were in orbit over the Indian Ocean during the coast phase. Here, the TETR piggyback satellites were ejected from the second stage on flights 8 and 9.[23] Some several hundred seconds after

[23] The Pioneer Flight E carried a TETR but it was destroyed by the Range Safety Officer soon after liftoff on Aug. 27, 1969.

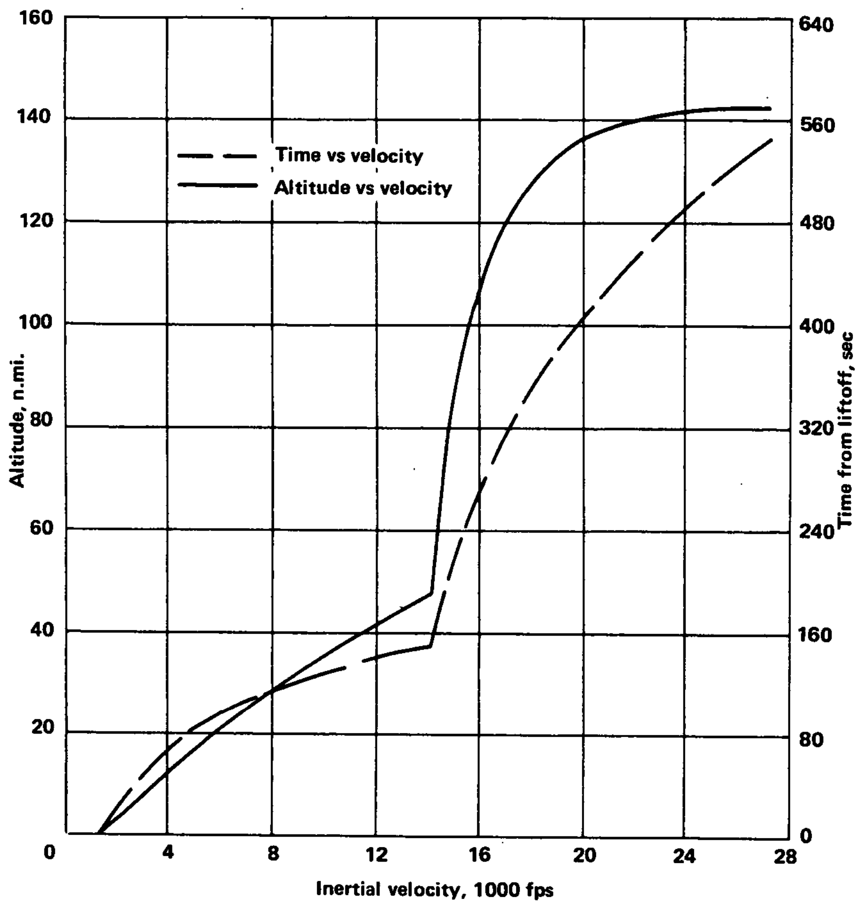


FIGURE 7-11.—The reference trajectory specified by NASA for Pioneer launches.

orbital injection, when the Delta third stage plus spacecraft drifted close to the plane of the ecliptic, the third stage fired, propelling the spacecraft out of Earth orbit into orbit around the Sun. For inward solar orbits, the spacecraft had to be injected antiparallel to the Earth's direction of motion about the Sun; that is, the spacecraft was given an orbital velocity around the Sun smaller than the Earth's. (See ch. 2.) As the inward Pioneers (6 and 9) fall toward the Sun, they pick up speed and eventually pull farther and farther ahead of the Earth. Conversely, Pioneers 7 and 8, the outward Pioneers, were injected parallel to the Earth's direction of motion and now lag the Earth by ever-greater distances.

The actual trajectories flown by the Deltas during the Pioneer launches are described in Volume III. Here, the objective is to show the impact of

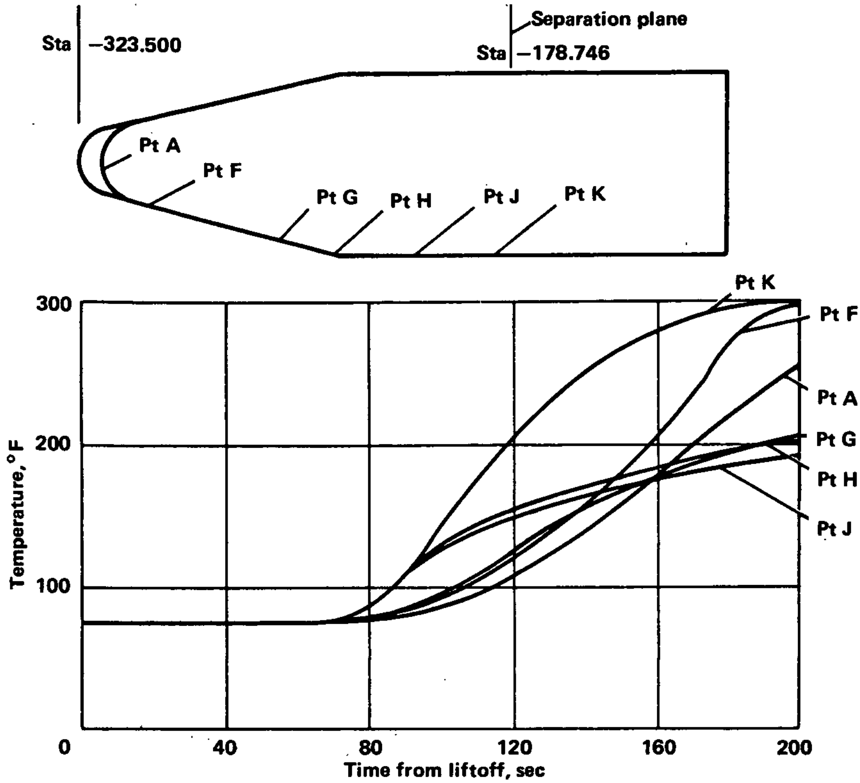


FIGURE 7–12.—Internal temperature histories specified for Delta shroud. Insulation had to be added to meet these specifications on several Pioneer launches.

mission objectives on the launch vehicle and trajectory and the interfaces between the several systems that must work in harmony for a successful launch.

PRIMARY LAUNCH OBJECTIVES

The primary Pioneer launch objective was the successful injection of the spacecraft into an orbit around the Sun. However, the orbital parameters and the shape of the trajectory taking the spacecraft into orbit had to be carefully designed to meet conditions arising from the scientific objectives, tracking requirements, secondary payload objectives, etc. These factors are discussed in more detail in chapter 2.

In addition to meeting the mission objectives, launch trajectories also had to meet stringent range safety requirements at Cape Kennedy, as well as a set of conditions called (in Cape Kennedy jargon) "WECO look-angle

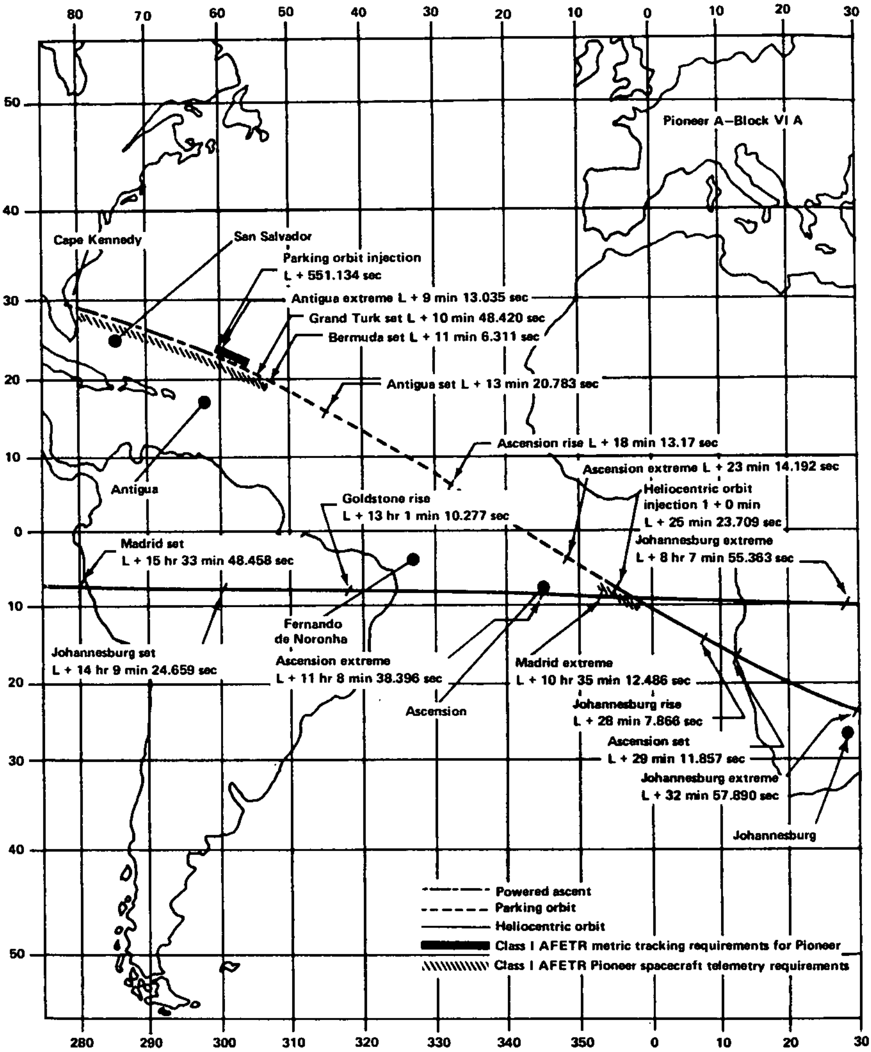
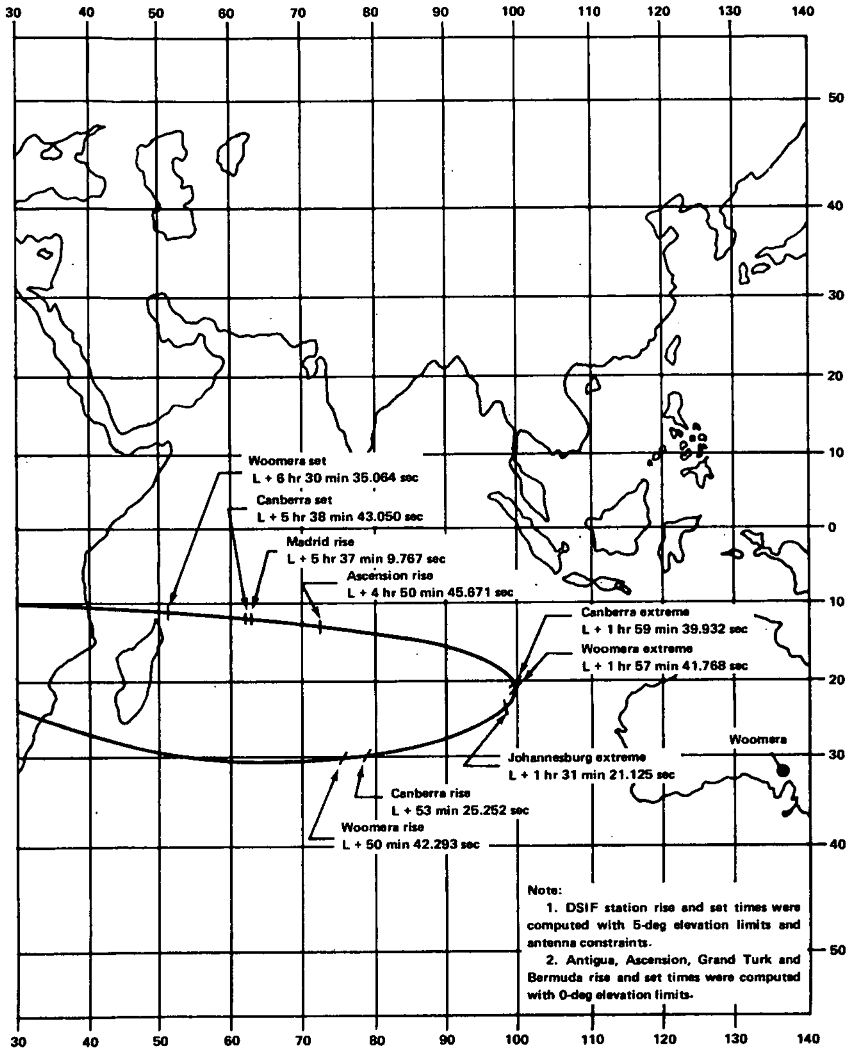


FIGURE 7–13.—Earth track of Pioneer 6 showing DSIF involvement during flight. As the spacecraft escaped from Earth, the Earth turned under it, giving rise to apparent retrograde motion.

constraints." (WECO refers to Western Electric Company, the manufacturer of the radar-controlled guidance equipment employed on all Delta Pioneer launches.) For the guidance equipment to function properly, the launch trajectory had to remain within the radar's field of view (look angle) for a stipulated period of time.



THE LAUNCH SEQUENCE

For the Pioneer-9 launch, which will be used to illustrate the entire series, two launch blocks of time were selected, and within each block, a prime launch day. The Pioneer-9 blocks were November 1 through November 22, 1968, and November 27 through December 22, 1968; the prime launch days were November 6 and December 18, respectively.

The length of time that the third-stage-spacecraft combination has to coast in Earth orbit to reach the plane of the ecliptic varies slightly from

TABLE 7–3.—*Events Planned in Pioneer-9 Trajectory Sequence*

Time (sec)		Event
0.000		Stage-1 liftoff
2.000		Begin stage-1 roll program
3.670		End stage-1 roll program
4.000		Begin stage-1 pitch program
9.670		End first pitch rate—stage 1
10.000		Begin second pitch rate—stage 1
38.190		Solid motors burnout
64.670		End second pitch rate—stage 1
65.000		Begin third pitch rate—stage 1
70.000		Jettison solid motor casings
89.670		End third pitch rate—stage 1
90.000		Begin fourth pitch rate—stage 1
130.000		End stage-1 pitch program
150.531		Main engine cutoff
154.531		Stage-2 ignition signal
154.531		Start VCS [a] channel 1
155.581		Stage-1 separation
156.331		Stage-2 90 percent chamber pressure
159.531		Begin stage-2 pitch program
166.531		End first pitch rate—stage 2
167.531		Begin second pitch rate—stage 2
169.531		Jettison fairing
460.000		End VCS channel 1
460.000		Start VCS channel 2
534.352		End VCS channel 2
534.352		Second-stage engine cutoff command
534.352		End stage-2 pitch program
534.721		Final cutoff—stage 2
561.531		Begin coast-phase pitch program
First firing block	Second firing block	
696.531	734.531	End coast-phase pitch program
735.531	735.531	Begin coast-phase yaw program
760.531	760.531	End coast-phase yaw program
1201.531	1349.531	Start stage-3 ignition time-delay relay
		Fire spin rockets
1203.531	1351.531	Jettison stage 2, activate retro system
1216.531	1364.531	Stage-3 ignition
1247.331	1395.331	Stage-3 burnout
1263.531	1411.531	Pioneer separation
1300.531	1448.531	TETR separation from stage 2

[a] VCS = Velocity cutoff system.

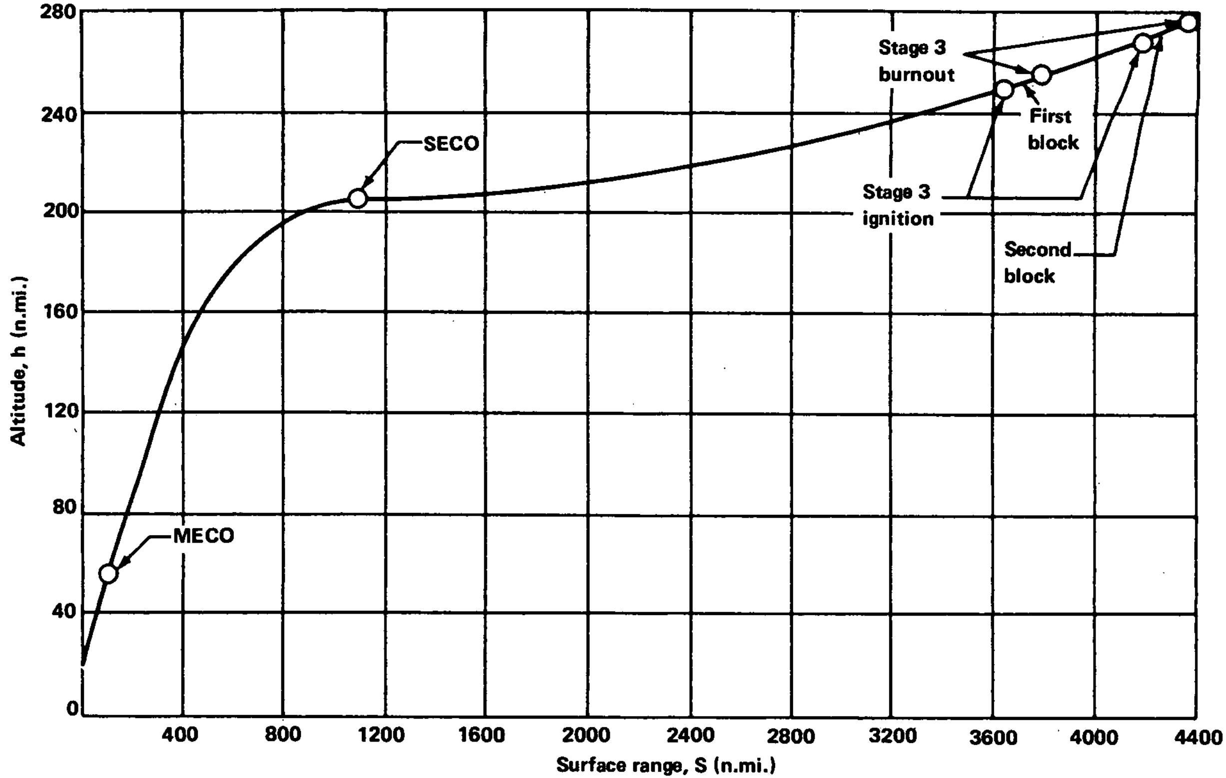


FIGURE 7–14.—Altitude vs. surface-range history planned for Pioneer-D launch.

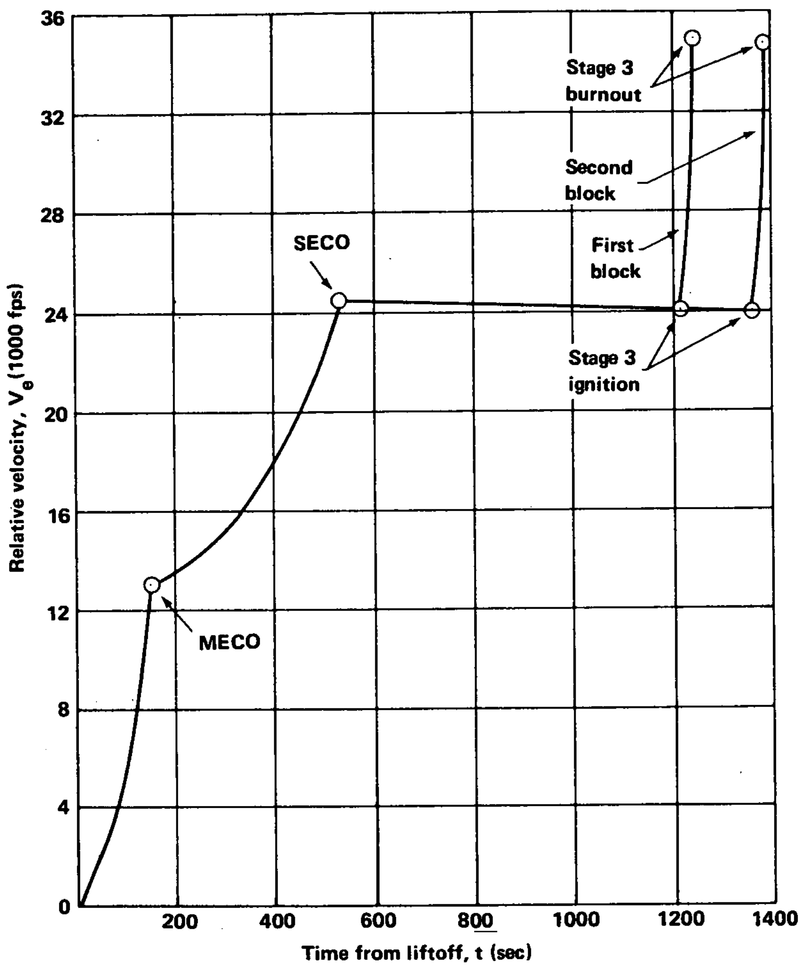


FIGURE 7–15.—Relative-velocity history planned for Pioneer-D launch.

day to day and even more from block to block. From the launch of Pioneer 9 to a planned orbit 502 × 203 n. mi., the trajectories were identical regardless of day of launch within each block of days, as noted in the launch sequence listed in table 7–3. The coast periods, however, are 682 and 830 sec, respectively, for the November and December launch blocks.

The trajectory planned for Pioneer 9 is illustrated crudely by figures 7–14 and 7–15. In this computer-assisted age, trajectory and orbit details are customarily presented at great length by computer printouts. From the moment of liftoff to 1000 days later, the critical rocket and spacecraft parameters for the Pioneer 9 flight were printed out (in some 330 pages) by the Delta contractor (ref. 3). Printouts such as this were prepared for each Pioneer flight. Detailed scenarios were also prepared for each flight telling each member of the launch team what to do and when to do it. (See Vol. III.)

A TYPICAL WEIGHT BREAKDOWN

The object of all these tables and computer printouts, of course, is the injection of the small spacecraft, weighing only about 1 percent as much as the launch vehicle on the pad, into orbit around the Sun. Since this chapter focuses on the Delta, it will be instructive to see how 99 percent of the launch vehicle weight is applied to the 1 percent payload (table 7–4).

TABLE 7–4.—*Typical Pioneer Launch-Vehicle Weight Breakdown*

Item	Weight (lb)
Launch vehicle at liftoff	152 153
Vented liquids and gases	−39
Fuel and oxygen burned, stage 1	−26 316
Solid propellants burned, augmentation	−24 786
Launch vehicle at solid motor burnout	101 012
Vented liquids and gases	−49
Fuel and oxygen burned, stage 1	−20 898
Burned-out solid motor	−4 803
Launch vehicle after jettison of solid motor	75 262
Vented liquids and gases	−129
Fuel and oxygen burned, stage 1	−52 791
Launch vehicle at MECO	22 342
MECO (151.56 sec)	
Main-engine stop losses	−66
Fuel and oxygen burned, vernier engine	−42
Launch vehicle before stage-1 separation	22 234
Vernier engine fuel and oxygen available for impulse	−63
Residual propellants, stage 1	−200
Trapped liquids and gases	−846
Dry stage 1 (DSV–3E–1) jettisoned	−7001
Launch vehicle after stage-1 separation	14 124.10
Interstage structure jettisoned	−180.55
Launch vehicle at stage-2 ignition	13 943.55
Fuel and oxidizer lost during startup transients	−19.60
Usable pressurized nitrogen	−3.15

TABLE 7–4.—*Typical Pioneer Launch-Vehicle Weight Breakdown* (Continued)

Item	Weight (lb)
Fairing jettisoned	−539.45
Fuel and oxidizer consumed	−10 313.33
Launch vehicle at second-stage engine cutoff command (SECOM)	3068.02
Fuel and oxidizer consumed and lost during engine stop transients	−26.50
Residual propellants	−294.91
Trapped propellants and gases	−65.15
Spin table jettisoned	−82.06
Dry stage 2 (DSV–3E–3) jettisoned	−1665.51
TETR satellite released	−58.00
Launch vehicle at stage-3 ignition	875.89
Start losses, stage 3	−0.05
Inert loss during burning, stage 3	−4.30
Propellant consumed, stage 3	−606.90
Launch-vehicle and third-stage burnout	264.64
Stage-3 motor, ballast, balance weights, support rings, beacon, and telemetry kit jettisoned	−46.62
Stage-3 motor case jettisoned	−54.45
Spacecraft attach fitting jettisoned	−16.01
Spacecraft injected into solar orbit	147.56

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1. ANON.: Delta Spacecraft Design Restraints. McDonnell-Douglas Astronautics Co. DAC–61687, Oct. 1968 (one of series of updated Delta restraint documents).
2. ANON.: Pioneer Program, Specification A–6669.00. NASA, Ames Research Center, Rev. no. 7, Jul. 28, 1966.
3. ANON.: Detailed Test Objectives for Improved Delta Launch Vehicle. Spacecraft: Pioneer D, McDonnell-Douglas Astronautics Co. DAC–61696, Oct. 1968. (A similar report exists for each Pioneer flight.)

CHAPTER 8

Tracking and Communicating with Pioneer Spacecraft

TRACKING THE FIRST PIONEERS

THE FIRST GROUP OF PIONEER SPACE PROBES (Pioneers 1 through 5) were launched in the direction of the Moon between 1958 and 1960. The tracking and data acquisition theories and hardware developed by JPL to support these flights ultimately developed into the present DSN. The DSN, managed by JPL for NASA, tracks NASA's unmanned spacecraft launched toward the Moon, the planets, and deep space.

The basic problem that JPL had to solve in tracking and acquiring data from spacecraft beyond Earth orbit involved the immense distances of interplanetary flight. Ten thousand miles posed little difficulty, but at tens of millions of miles spacecraft signals faded away amid the radio noise of interplanetary space. The Minitrack radio interferometer stations that the Naval Research Laboratory (NRL) had installed around the world during 1956 and 1957 for the IGY could work near-Earth satellites, but they could not detect faint signals from deep space with their low-gain antennas. Conventional radars could not track spacecraft much beyond 1000 miles. Therefore, new techniques were needed for deep space tracking.

Three fundamental concepts permit the successful tracking of very distant spacecraft by the DSN:

(1) The concept of a high-gain, highly directional, paraboloidal antenna with a large diameter—high gain permits reception of very weak spacecraft signals; high directionality provides the accurate angular bearings needed for tracking. Big-dish antennas have been in the radio astronomer's repertoire since a radio amateur named Grote Reber built a small one in his backyard in 1937.

(2) A radio ranging technique, utilizing pseudorandom noise, allowed ground observers to measure the transit time and Doppler shifts of radio signals between Earth and spacecraft and back again. Spacecraft distance and radial velocity come from these measurements.

(3) The JPL phase-lock-loop, conceived by JPL's Eberhardt Rechtin during the 1950s, was adopted by the DSN and later by the Manned Space Flight Network (MSFN) for its Unified S-Band tracking during the Apollo Program. The phase-lock-loop concept is fundamental to the

detection of signals by the DSN, but it is independent of the pseudorandom noise approach to tracking.

JPL put its tracking and data acquisition concepts into practice prior to October 1958, while it was still under U.S. Army sponsorship. Explorers 1 through 5 were tracked by a handful of JPL phase-lock-loop, Microlock stations as well as the NRL Minitrack Network. By late 1958, as the first Pioneers were launched, JPL had established tracking stations at Cape Canaveral, Puerto Rico, and Goldstone Lake, California. The biggest dish in the embryonic network was the 85-foot paraboloid at the Goldstone Pioneer Site (fig. 8–1). Smaller dishes were located at Cape Canaveral, Florida, and near Mayaguez, Puerto Rico. A 60-ft, Department of Defense dish in Hawaii and the famous 250-ft antenna at Jodrell Bank, England, cooperated with the JPL stations during the early Pioneer launches.

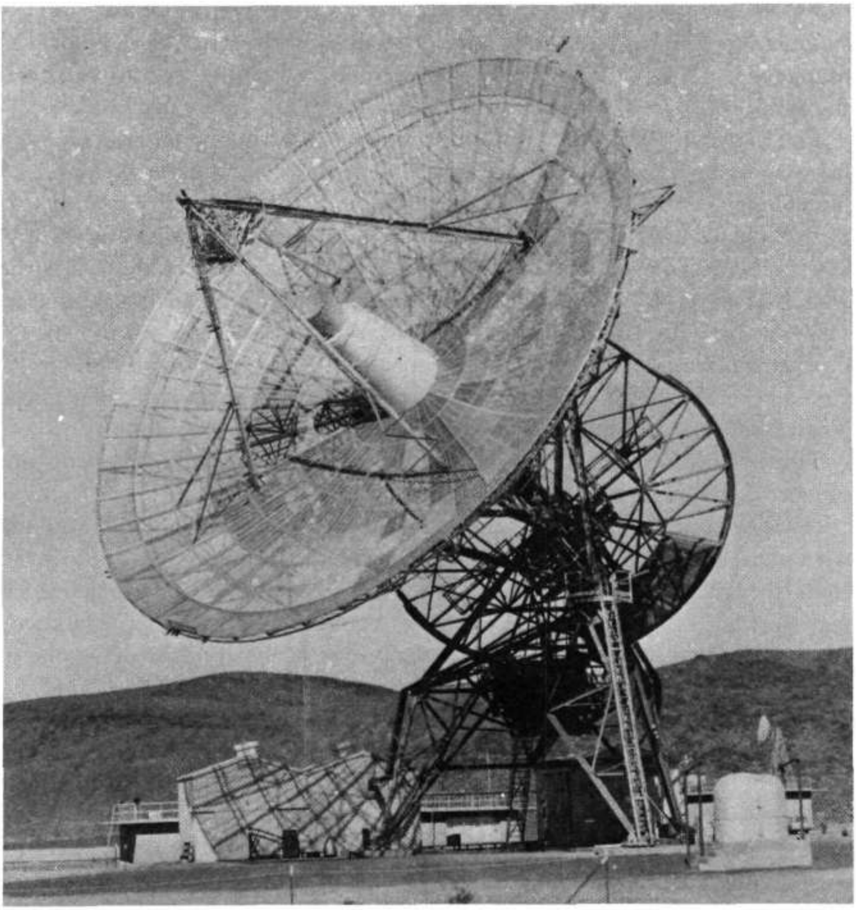

FIGURE 8–1.—The first 85-ft paraboloidal antenna installed at Goldstone (Pioneer site).

DSN evolution since 1960 has been expressed primarily in terms of physical size (antenna diameter and new stations), electronic sophistication (masers, lower antenna noise temperatures, etc.). The 85-ft dishes, the hallmark of the DSN, are now found near Madrid, Johannesburg, Woomera, and Canberra, as well as Goldstone. A 210-ft paraboloid was added at Goldstone in 1966; others are under construction at Madrid and Canberra.

When the Pioneer Program began in late 1961, there was no question about network choice. The DSN was the only one of NASA's three networks that could track and communicate with a deep-space probe. Like the Delta launch vehicle, the DSN became a basic, general-purpose pillar of the Pioneer Program—but a pillar already in place that could be altered very little for any specific mission. Even more than the Delta's, the basic capabilities of the DSN helped shape the Pioneer spacecraft design.

SOME GENERALITIES ABOUT TRACKING AND DATA ACQUISITION

The three basic functions performed by terrestrial ground-support equipment during the Pioneer missions were:

(1) Tracking—Spacecraft position was measured with high precision from liftoff at the launch pad to injection into parking orbit, through the coast phase, to injection into heliocentric orbit, and as far out in deep space as possible—several hundred million miles and more if possible. Out to about 10 000 miles this function was accomplished by the Near Earth Phase Network, which consists of MSFN and U.S. Air Force precision radars; beyond 10 000 miles, the DSN performed this function.

(2) Communication or data acquisition—Scientific and housekeeping data were detected and acquired from the spacecraft, and routed from the worldwide network stations to a central location for evaluation and processing.

(3) Command—Commands were dispatched from a centralized control center to the network station working the spacecraft and, then, to the spacecraft itself.

Obviously the Pioneer spacecraft could not be designed independently of the DSN and its relatively fixed roster of equipment. As described in chapter 4, the spacecraft communication subsystem had to be matched in terms of power level and frequency to the specific DSN receiving equipment expected to be operational at the time of launch. The same was true for the uplink that carried commands to the spacecraft.

The DSN was not a static facility. Its capabilities improved markedly over the 5-year Pioneer launching schedule. These improvements were not due to fundamental changes in the DSN but rather to continual upgrading and improvement, much like the collective changes that so greatly increased the Delta's payload capacity during the same period. In addition to the evolutionary improvements, some of the 85-ft MSFN antennas adjacent to the DSN antennas for redundancy during Apollo flights were pressed into service tracking Pioneers while they were still relatively close to the Earth. With the Apollo, Mariner, and Pioneer Programs, NASA had so many active spacecraft in deep space that it pooled its big antennas to achieve optimum coverage.

In general terms, the DSN carries out its three basic functions using three distinct facilities (ref. 1):

(1) The Deep Space Instrumentation Facility (DSIF) consists of the DSN tracking and data acquisition stations shown in table 8–1.

(2) The Space Flight Operations Facility (SFOF) is located at JPL, in Pasadena, California; it monitors all spacecraft data, issues commands, and performs all necessary mission calculations.

(3) The Ground Communication Facility (GCF) ties all DSIF stations to the SFOF with high-speed, real-time communications. The bulk of DSN communication traffic is carried via NASA's global communication system, NASCOM, which contributes circuits to the GCF.

One other network of equipment crucial to the Pioneer mission is the Eastern Test Range (ETR) run by the U.S. Air Force. ETR radars, optical instruments, and other tracking equipment follow all launches from Cape Kennedy down range past Ascension Island, over Africa, into orbit, where NASA networks assume the full tracking load. They are considered

TABLE 8–1.—*The DSN Stations*

DSS no.	Location	Dish size	Primary during Pioneer flight				
			6	7	8	9	E
11	Goldstone, Cal. (Pioneer) [a]	85-ft			X	X	
12	Goldstone, Cal. (Echo)	85-ft	X	X	X	X	
13	Goldstone, Cal. (Venus) [b]	85-ft					
14	Goldstone, Cal. (Mars) [c]	210-ft	X	X	X	X	
41	Woomera, Australia	85-ft			X		
42	Canberra, Australia [a] [d]	85-ft	X	X		X	
51	Johannesburg, South Africa	85-ft	X	X	X	X	
61	Madrid, Spain (Robledo) [a]	85-ft			X	X	
62	Madrid, Spain (Cebreros)	85-ft			X	X	
71	Cape Kennedy, Fla.	4-ft	X	X	X	X	X

[a] MSFN Apollo Wing located here was used during some Pioneer flights.
[b] Used primarily for research and development.
[c] Used on "extended" Pioneer missions.
[d] Also called Tidbinbilla.

part of the Near Earth Phase Network during the early portions of the Pioneer missions.

To complete the picture of the DSN, JPL engineers often visualize the three facilities just described as vertical sinews interwoven with six horizontal sinews representing the groups of equipment that accomplish the tracking, data acquisition, and command functions as well as those of simulation, monitoring, and operational control (fig. 8–2).

When Pioneers 1 through 5 headed for deep space, they were the only active spacecraft beyond Earth orbit. The few trackers in existence could find and follow these space probes readily, though without great precision and not very far. It was a "simple" picture in 1960. Today, however, NASA has stationed almost a score of 85-ft dishes and one 210-ft dish at various spots on the globe and filled the adjacent buildings with advanced electronic gear. It is possible to listen to, track and command spacecraft 200 million miles away from Earth. Because of many currently active spacecraft, DSN priorities have to be assigned to each spacecraft; and each tracking station, being only a part of a tremendously complex machine, operates on a rigorous schedule. The DSN is a world-wide data collector for scientists.

GENERAL DEEP SPACE NETWORK CAPABILITIES

After the Pioneer Program was officially approved by NASA Headquarters on November 9, 1962, spacecraft design and mission planning commenced at Ames Research Center. The capabilities of the Delta launch vehicle helped fix the weight and volume of the spacecraft, while the DSN—as it was projected for the 1965–1969 period—had considerable influence over spacecraft antenna design, frequency selection, telemetry bit rates, type of telemetry, and many other facets of spacecraft communication and command.

The Deep Space Instrumentation Facility

In tracking language, the DSIF is the Earth-based portion of a two-way, phase-coherent,[24] precision tracking and communication system capable of providing position tracking, telemetry, and command for spacecraft more than 10 000 miles from the Earth. Each DSN station feeds acquired tracking measurements (two angles and range rate) to the Ground Communications Facility (GCF) which relays it to the SFOF in near real time (i.e., almost instantaneously). Pioneer telemetry data are partially processed in real time by on-site computers. Data are then teletyped or airmailed to the SFOF at

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[24] "Phase-coherent" signifies fixed frequency and phase relationships between transmitted and received signals.

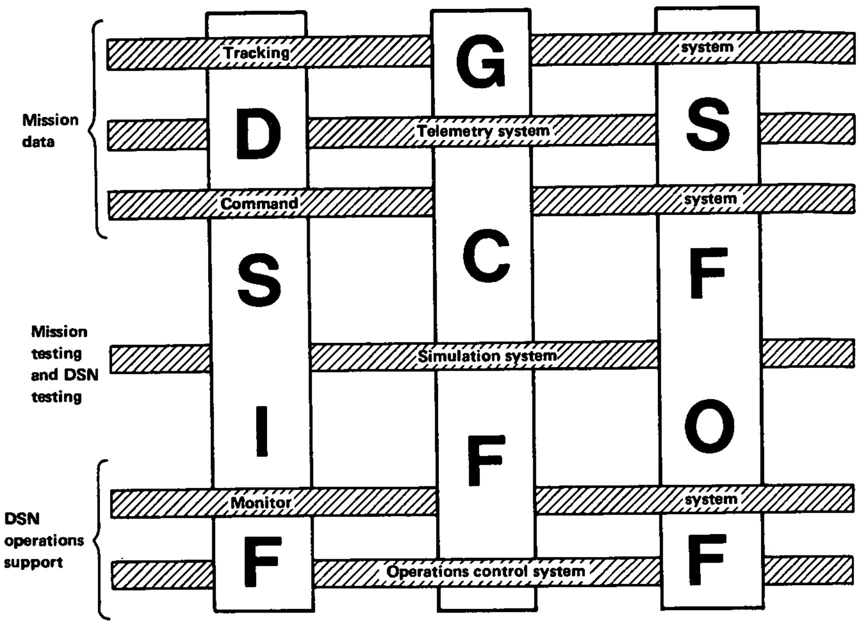


FIGURE 8–2.—The Deep Space Network can be visualized as three facilities (DSIF, GCF, and SFOF) interwoven with six systems.

JPL. The spacecraft must carry a phase-coherent transponder for the DSN to track the spacecraft satisfactorily.

The DSN stations listed in table 8–1 were deliberately placed about 120° apart in longitude in a band between 40° north and 40° south latitude. Overlapping sky coverages result with the 85-ft dishes (figs. 8–3 and 8–4). Although local conditions cause slight variations in building arrangements, the DSN stations appear essentially identical to a spacecraft across the electromagnetic and information interfaces.

Pioneer Earth-to-spacecraft transmissions occur at 2110 MHz; spacecraft-to-Earth at 2292 MHz. For coherent two-way Doppler tracking measurements, several pairs of channels are selected with a frequency ratio of 221/240. (See ch. 4 for a discussion of phase-lock receivers and their use in the Pioneer Program.)

Two-way Doppler measurements are made by first transmitting an S-band signal from a DSN site to the spacecraft. The spacecraft, using phase-coherent frequency multiplication, converts the received signal into one of higher frequency and transmits it back to Earth. Measurements of time and Doppler shift provide range and range rate. In the Pioneer Program, only Doppler range is used in pinpointing spacecraft locations. DSN precision Doppler measurements are usually made with this closed,

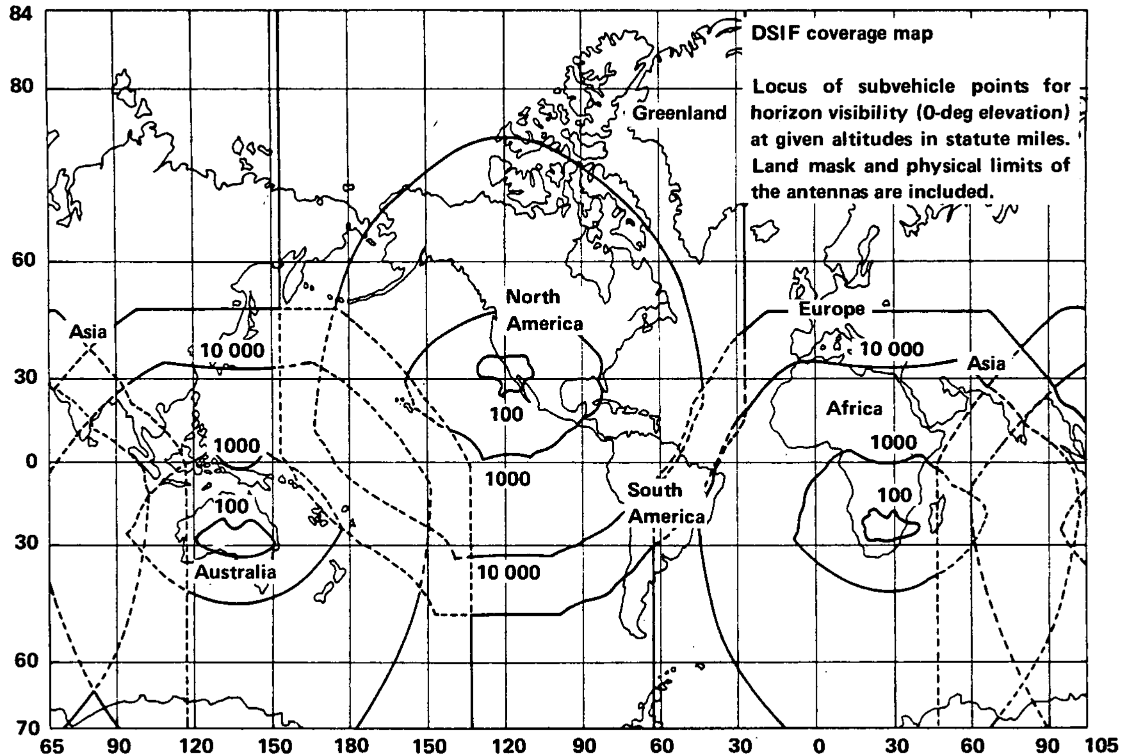


FIGURE 8–3.—Station coverages for the three polar-mounted 85-ft DSIF antennas at Goldstone, Woomera, and Johannesburg.

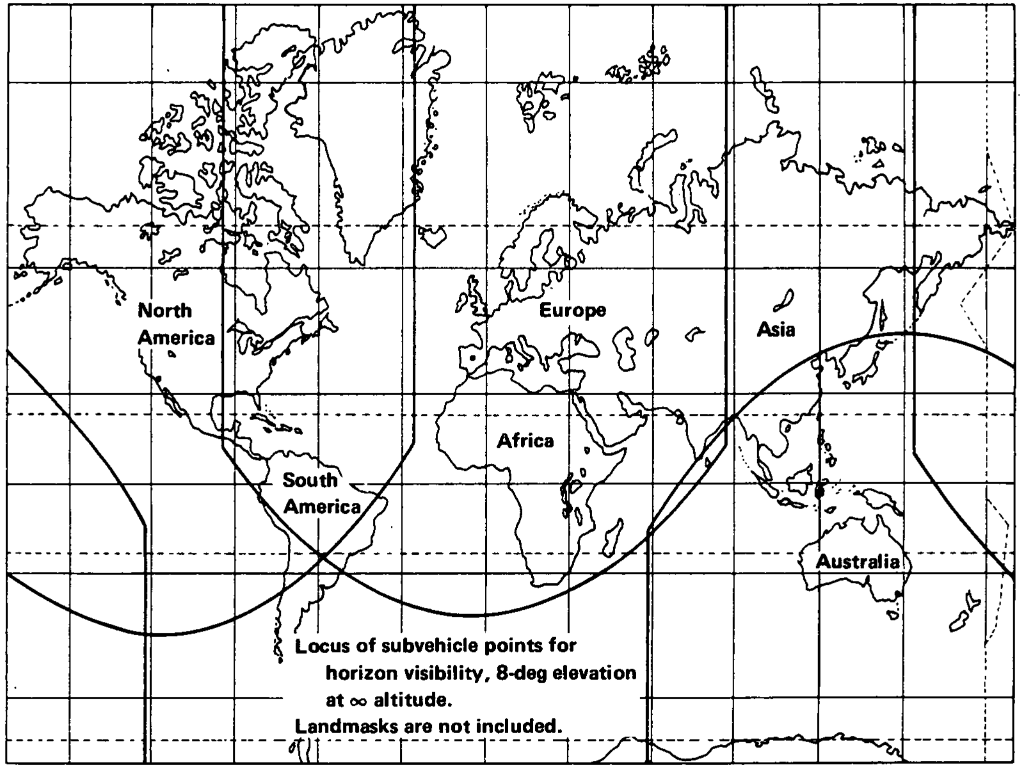


FIGURE 8–4.—Station coverages for the three polar-mounted 85-ft DSIF antennas at Goldstone, Madrid, and Canberra.

two-way, phase-locked mode. A less accurate one-way Doppler mode is sometimes used by stations that are merely listening to the spacecraft transmissions. Drifting of the spacecraft crystal-controlled oscillator limits the precision when the spacecraft receiver is not locked onto the Earth transmitter's frequency. When two separate but intercommunicating DSN stations have the spacecraft in view, three-way modes are possible, with one station in a two-way mode and the other in a one-way mode. The accuracy of DSN ranging is approximately ± 15 m one way (three-sigma value). Range rate accuracy varies with the magnitude of the Doppler shift.

The standard DSN site with its 85-ft dish depends upon 14 subsystems (ref. 1). See figure 8–5. A few important features are:

(1) Antenna mechanical subsystem—Most of the 85-ft dishes are S-band, Cassegrain feed, and monopulse in operation. The antennas point with an accuracy of 0.02° in a 45-mph wind.

(2) Antenna microwave subsystem—Some of the most critical and sophisticated DSIF components are included here: Cassegrain simultaneous lobing feeds, traveling-wave masers, and parametric amplifiers. Beamwidths to the half-power points are $0.32 \pm 0.03°$ and $0.36 \pm 0.03°$ for receive and transmit modes, respectively.

Acquisition-aid subsystem—DSN stations 11, 41, 42, and 51 are equipped with S-band antennas with beamwidths of 16° to help lock the 85-ft antennas, with their much narrower beamwidths, onto the spacecraft.

The Goldstone Mars station (DSS–14) is equipped with a 210-ft dish on an azimuth-elevation mount. This big antenna is more sensitive than the 85-ft dishes and is essential for the tracking of Pioneer spacecraft over 100 million miles away. The nominal beamwidths to the half-power points of the 85-ft dishes are 0.135° and 0.145° for receive and transmit, respectively. During Pioneer operations the beamwidths have appeared to be about 0.20°. Pointing accuracy is 50 arcseconds.

The Ground Communications Facility

NASCOM consists of those circuits, terminals, and switching centers that link the dispersed stations of all three NASA networks together and to their respective control centers. NASCOM is a real-time network; that is, the stations and control centers can exchange data, teletype, and voice messages almost instantaneously.[25] The Pioneer Mission Operations Center at Ames Research Center (fig. 8–6) can, for example, dispatch commands in real time to any one of the Pioneer spacecraft as long as a DSN antenna somewhere in the world is in contact with it.

The GCF utilizes NASCOM for its long-distance traffic (fig. 8–7). Except for the Goldstone-JPL information flow, DSN traffic converges

[25] Limited mainly by the finite speed of light.

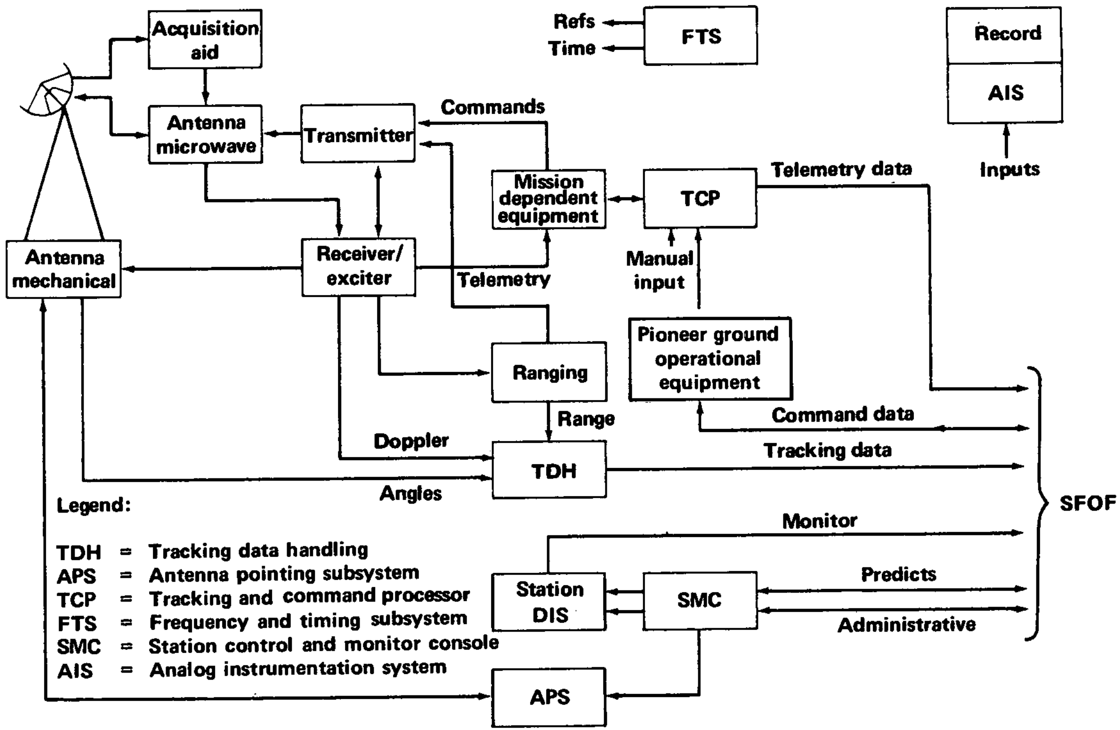


FIGURE 8-5.—Block diagram of the DSIF subsystems.

FIGURE 8-6.—Pioneer Mission Operations Center at Ames Research Center.

first on several NASCOM overseas switching centers, which in turn route it to the central computer-controlled switching center at Goddard Space Flight Center, Greenbelt, Maryland. From Goddard, the traffic is directed to the address indicated on the message. In the case of Pioneer, the SFOF at JPL is the major addressee, although individual DSIF stations can address one another. Goddard Space Flight Center manages NASCOM, but JPL has operational control of the circuits it is using at any given moment.

The five subsystems of the GCF are: (1) inter-station transmission, (2) SFOF communications terminal, (3) DSS communications terminal, (4) DSIF internal communications, and (5) SFOF internal communications (fig. 8–8). In other words, the GCF includes considerable terminal equipment not considered part of NASCOM proper. As figure 8–9 indicates, tracking data flow back to the SFOF primarily by teletype. Most scientific data are recorded on tape and airmailed to the SFOF, where the tapes are verified and then shipped to Ames Research Center for further processing (ch. 9).

The voice circuits are used primarily for coordination and control between the SFOF and the DSN stations. The high-speed data circuits transmit up to 2400 bits/sec with time delays only slightly greater than the

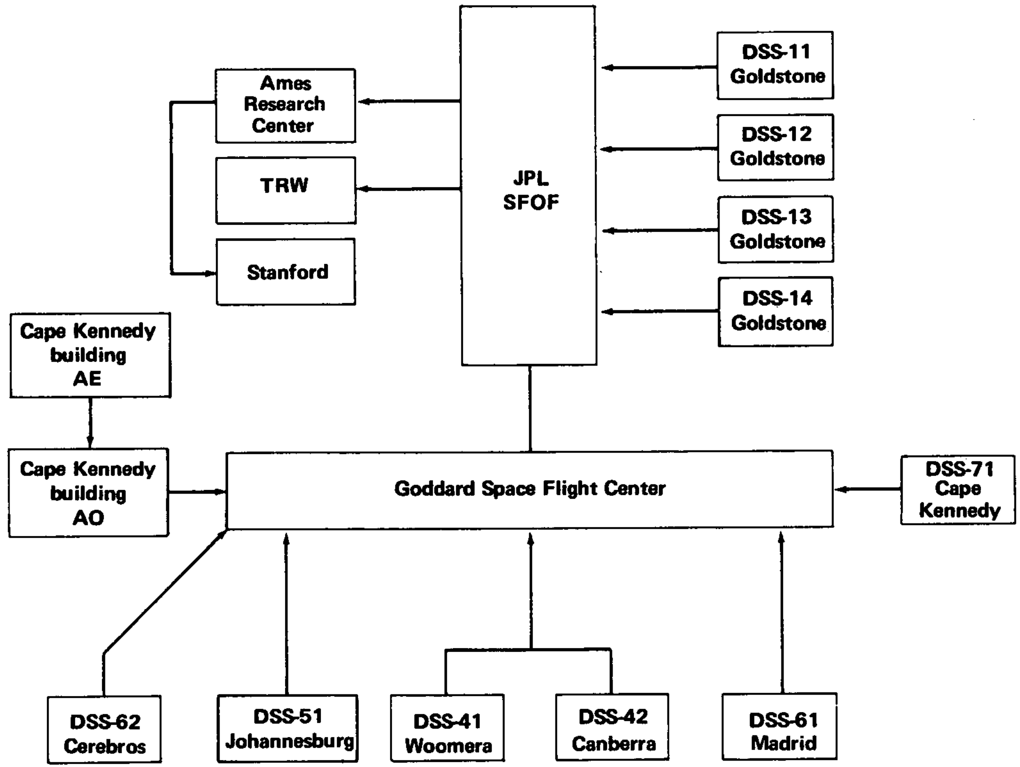


FIGURE 8–7.—Generalized communication traffic routing diagram for GCF. Overseas stations are handled by Goddard's NASCOM lines.

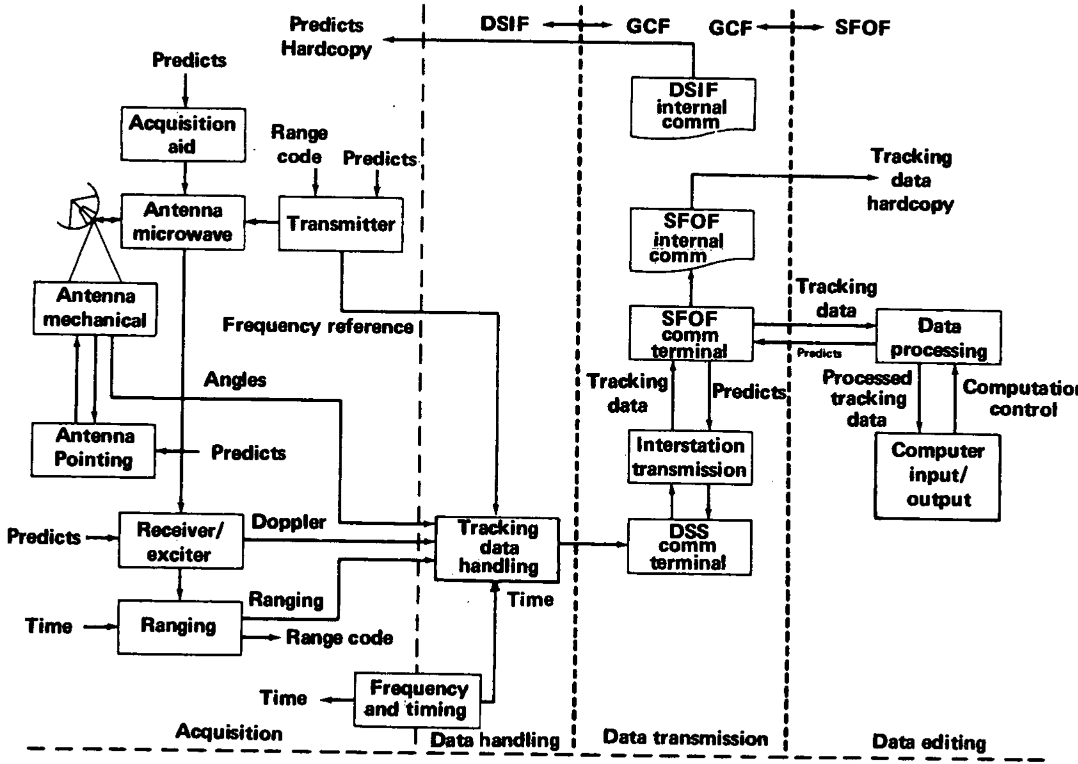


FIGURE 8–8.—Interface diagram for the DSIF, GCF, and SFOF showing the five GCF subsystems.

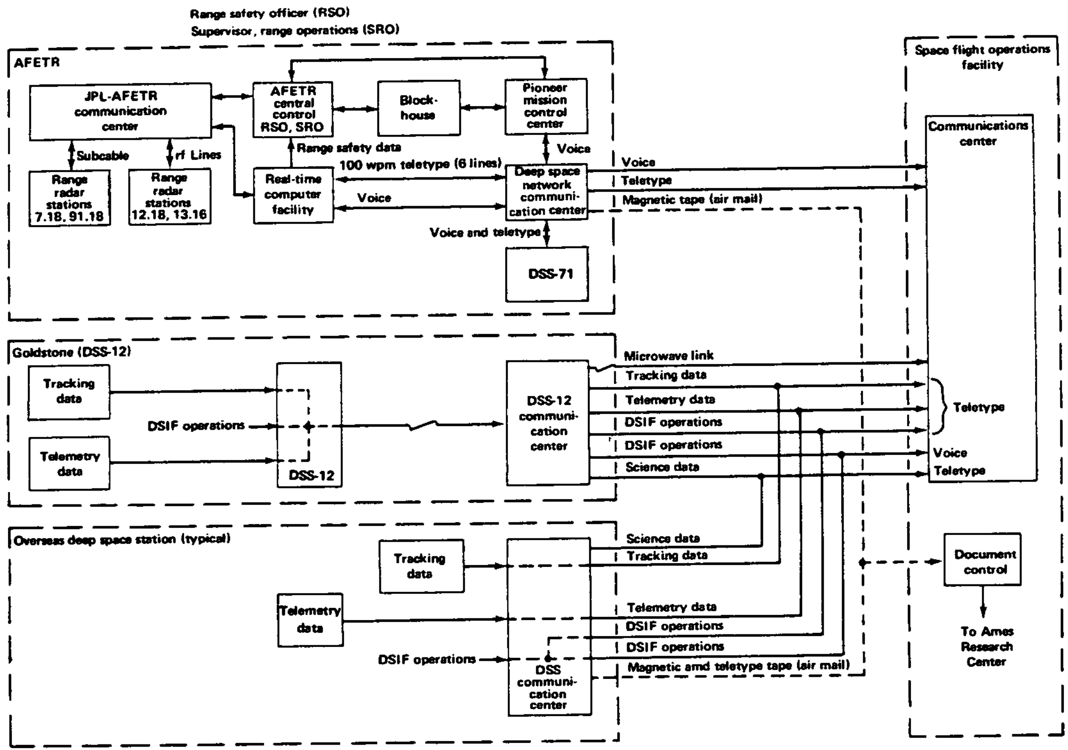


FIGURE 8–9.—Telemetry- and tracking-data flow from the ETR and DSIF to the SFOF and Ames Research Center. Tapes are sent first to the SFOF for verification and then to Ames.

time it would take light to travel the same distance. In effect, "real time" means delays of only tenths of a second at most. The teletype circuits, however, are a little slower, with ¼-sec delays at each control point (up to a maximum of three control points). The site communication's processors, however, introduce delays of 30 to 120 sec. During the early parts of its flight, a Pioneer often transmits at its maximum rate of 512 bits/sec, which is greater than the teletype rate of 60 words/min. During this time it is possible to call up selected blocks of data via the teletype circuits in order to assess the condition of the spacecraft.

The Space Flight Operations Facility

The focal point of DSN activity is a modernistic four-story building at JPL, in Pasadena; this building is the SFOF. The SFOF terminal of the GCF occupies part of the basement. Above are the computers, displays, controls, and facilities for mission control, a major part of which involves DSN control.

Brief descriptions of the eight subsystems that make up the SFOF follow:

(1) Data Processing Subsystem (DPS)—The function of the DPS is the ingestion of DSIF tracking data and its subsequent processing into the formats required for display and control. General-purpose digital computers are the mainstay of the DPS.

(2) Computer Input/Output (I/O) Subsystem—Consoles, printers, and plotters provide one interface between the Data Processing Subsystem and SFOF users.

(3) Data Processing Control and Status (DPCS) Subsystem—Three consoles are used here to monitor and control the Data Processing Subsystem.

(4) Telemetry Processing Subsystem (TPS)—The TPS performs real-time and non-real-time processing of all telemetry except that received on teletype. The TPS carries DPS processing several steps further, depending upon the desires of the mission controllers and experimenters.

(5) Timing Subsystem—This SFOF subsystem generates, distributes, and displays accurate time signals throughout the SFOF.

(6) Display Subsystem—This subsystem drives the automatic displays, the Mission Display Board, and the orbital parameters display in the SFOF. It provides one more interface between the SFOF user and the spacecraft and DSN.

(7) Simulation Data Subsystem—The Simulation Data Subsystem supplies simulated telemetry and tracking data during network tests and training exercises.

(8) Operations Support Subsystem—This subsystem is a catchall for such SFOF activities as document control, transmitting services, support planning, etc.

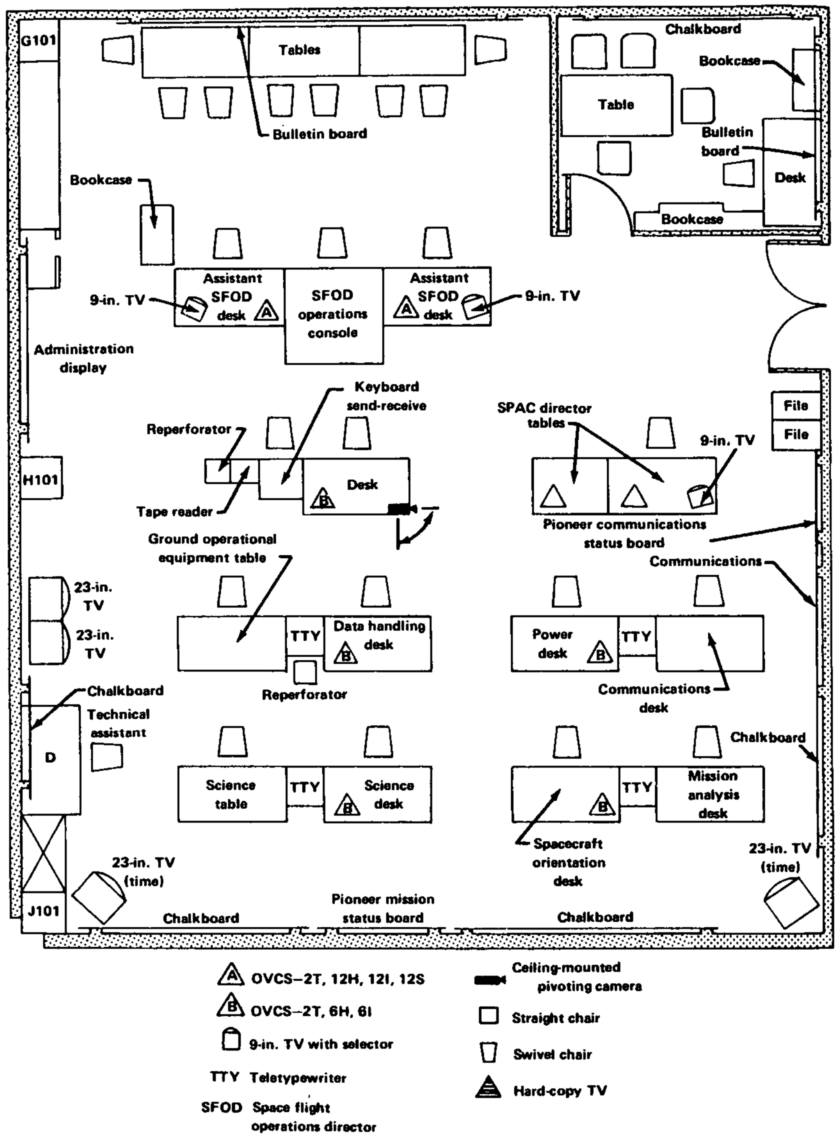


FIGURE 8–10.—Floor plan of the Pioneer mission support area at the SFOF.

Ames personnel controlled the spacecraft from the Pioneer Mission Support Area at the SFOF during critical portions of the flights (fig. 8–10). Spacecraft/orientation maneuvers, however, were controlled from Goldstone and, in the case of the partial orientation maneuver of Pioneer 6, from Johannesburg.

Special Pioneer Requirements Placed on the DSN

The DSN was better equipped during the launches of the Block-II Pioneers than it was when Pioneers 6 and 7 first probed deep space. With the 210-ft Goldstone antenna in operation, the DSN could track and communicate with spacecraft most of the way around the Sun—the Pioneers are some 150–200 million miles away before the Sun's radio noise overwhelms their telemetry signals. As success followed success in the Pioneer Program, scheduling the tracking time of the big DSN dishes and those borrowed from the MSFN became a more difficult task. The swelling number of manned and unmanned lunar spacecraft added to the tracking burden.

The finite resources of the DSN dictates careful planning to avoid supersaturation. The Office of Tracking and Data Acquisition, at NASA Headquarters in Washington, serves as a focal point where requirements, priorities, and resources are weighed for all NASA missions and all three NASA networks. A standard yet rather flexible procedure has developed. The project requiring tracking and data acquisition support issues a "requirements document" called a SIRD (for Support Instrumentation Requirements Document). The SIRD collects priorities, requirements, and other important factors for all Pioneers in space and those being readied for the launch pad. In response to each SIRD, JPL issues an NSP (or NASA Support Plan) relating how it plans to meet the requirements. Goddard Space Flight Center does the same for MSFN support. Each document must be the result of considerable negotiation and balancing of priorities. The SIRDs are updated frequently to reflect changing demands. For example, the launching of a new Pioneer or the loss of signal from an old one might be significant enough to require SIRD updating.

Requirements set forth in the Pioneer SIRDs over the years have been voluminous. In the interest of brevity, only portions of the Pioneer SIRD issued prior to the launch of Pioneer E are summarized in tables 8–2 through 8–6 (ref. 2). These tables do reveal the complexity and magnitude of the tracking and data acquisition tasks for a spacecraft of moderate size, as do figures 8–11 and 8–12. The conditional nature of assigning tracking and data acquisition is revealed in the following list of priority-requirements criteria:

Priority I (Emergency).—This priority applies only to coverage required to investigate or correct a spacecraft or scientific-instrument anomaly if prompt action is necessary to safeguard achievement of primary mission objectives.

Priority II (Critical).—These requirements are mandatory for attaining primary mission objectives.

(1) Launch plus 30 days, continuous coverage from DSN stations with GOE

TABLE 8–2.—*Critical Pioneer Mission Operations Involving the DSN*

Operation	Orbit phase	Operation period	Purpose
1. Separation from third stage	58.5 sec after third-stage burnout		Separate spacecraft from spent third-stage; Mission Control: JPL/SFOF
2. Orient spacecraft spin axis normal to the spacecraft-Sun line (Type-I orientation maneuver)	Automatically initiated at separation from third stage	Nominally 0–10 min	Orient spacecraft for maximum solar energy to provide spacecraft power utilizing the on-board solar array; Mission Control: JPL/SFOF
3. Initial acquisition by DSN	Acquisition of downlink telemetry before launch plus one hour	Approximately 15 min	Acquire spacecraft during transfer to heliocentric orbit and assess spacecraft health; Mission Control: JPL/SFOF
4. Spacecraft data-mode changes and experiment turnon	Commanded from ground after uplink lock and assessment of spacecraft health and orbit	Approximately 6 hr after acquisition	Prepare spacecraft for scientific data collection; Mission Control: JPL/SFOF
5. Orient spacecraft spin axis normal to plane of ecliptic (Type-II orientation maneuver)	Commanded from DSS–12 with other DSS stations as alternates within several days of launch	One complete DSS pass takes approximately 8–10 hr	Establish final reference orientation of spacecraft spin axis; Mission Control: DSS–12 (Goldstone); except for partial Type-II orientation maneuver for Pioneer 6 made from Johannesburg
6. Cruise phase (nominal mission)	Begins upon injection into heliocentric orbit and upon completion of experiment turnon and Type-II orientation maneuver	Continuous tracking coverage for first month after launch; two tracking missions coverage per day through nominal lifetime; dependent on schedule conflicts, and with high-priority DSN,	Collect scientific data; major coverage provided by DSS containing Pioneer GOE for assessment and analysis of real-time science and engineering data; Mission Control: Ames Research Center

		intent should be to provide continuous coverage	
7. Cruise phase (extended mission)	Begins upon completion of cruise phase of nominal mission	One tracking mission coverage per day; tracking missions to be provided by the DSN 210-ft antenna at Goldstone, Cal.; extended mission to continue until spacecraft cannot provide useful science data or when spacecraft is beyond DSN capability	Collect scientific data; Mission Control: Ames Research Center
8. Solar-flare coverage phase	Begins when Mission Operations Manager requests coverage for reported Class-II-Bright or above solar flare during cruise phase	Nominal mission: continuous coverage for 30 to 50 hr; extended mission: within capability of 210-ft antenna system net to provide coverage over a 50-hr period	Collect maximum scientific data in spacecraft vicinity during high solar activity
9. Geomagnetospheric tail analysis (Pioneers 7, 8, and E)	Nominal period of analysis established prior to launch, actual required coverage period provided upon analysis of the resultant trajectory	Continuous coverage from syzygy-minus-5-days to syzygy-plus-15 days	Define boundaries and characteristics of the geomagnetospheric tail
10. Lunar occultation (Pioneers 7 and E)	The probability of a lunar occultation indicated from analysis of the nominal trajectory; definite times established upon detailed analysis of resultant trajectory; simultaneous view periods from Stanford University and Goldstone necessary	Continuous coverage from entrance minus 10 hr to exit plus 10 hr	Provide lunar occultation analysis utilizing the Stanford University on-board instrument in conjunction with the Stanford 150-ft tracking system and DSS–12 and DSS–14 at Goldstone, Cal.; coverage also required from other stations (Australia, Spain, or South Africa) if period occurs during overlap view with Goldstone and Stanford University

TABLE 8–2.—*Critical Pioneer Mission Operations Involving the DSN* (Continued)

Operation	Orbit phase	Operation period	Purpose
11. Superior conjunction or analysis of Sun's corona during solar occultation (Pioneers 6, 7, 8, 9, and E)	Part of the extended mission and to be fixed to a given period upon analysis of resultant heliocentric orbit	Continuous coverage within the capabilities of the 210-ft antenna beginning one month prior to and ending one month following solar occultation	Provide analysis of Sun's corona characteristics during superior conjunction
12. Reorientation maneuvers during cruise phase	As determined by the Mission Operations Manager	During a complete tracking pass at DSS–12 Goldstone.	Possibility of Mission Control's being moved to DSS–12 during this maneuver; to provide spacecraft spin axis orientation as determined by the Mission Operations Manager
13. Spacecraft anomalies	As determined by the Mission Operations Manager	Continuous coverage until anomaly has been corrected or it has been decided that it cannot be corrected, as determined by the Mission Operations Manager.	Possibility of Mission Control's being moved to JPL/SFOF or remaining at NASA/ARC as determined by the Mission Operations Manager

TABLE 8–3.—*General Pioneer Tracking Requirements as of March 1969*[a]

Pioneer	Nominal mission
6	DSS–14 daily coverage 4–8 hr/day; absolute minimum, DSS–14 daily coverage 3 hr/day
7	DSS–14 daily coverage; 4–8 hr/day; absolute minimum, DSS–14 daily coverage 3 hr/day
8	DSS–12, –42, –51, –62 and DSS–11, –42, –61: continuous coverage; absolute minimum, DSS–12, –42, –51, –62 and DSS–11, –41, –61; two tracking missions/day for a total of 16 hr/day
9	DSS–12, –42, –51, –62; continuous coverage; absolute minimum, DSS–12, –42, –51, –62 and DSS–11, –41, –61 and MSFN; two tracking missions/day for a total of 16 hr/day, with 1 hour overlap
E	DSS–12, –42, –51, –62; continuous coverage; absolute minimum, DSS–12, –42, –51, –62 and DSS–11, –41, –61 and MSFN; two tracking missions/day for a total of 16 hr/day

[a] These requirements vary with time, of course. This table is illustrative only.

(2) Thirty-first day to end of mission, two passes per day (coverage period 16 hr or greater)

(3) For duration of mission, at least one horizon-to-horizon two-way-Doppler tracking mission per week not to be on same day of the week

(4) Coverage of specific scientific events that offer single time periods within the flight mission when the data may be retrieved; when in effect, this requirement to take priority over all those noted above

(5) Solar-flare coverage, 30–50 hr from flare initiation for Class-II Bright or greater; upon occurrence, this requirement to take priority over all previously stated requirements above

The above requirements are reduced to 3–4 hr per day to end of mission because of spacecraft-Earth distance, spacecraft- and ground-antenna characteristics, or because only one 210-ft antenna is available for operational support.

Priority III (*Critical*).—These requirements are time-sensitive for other-than-primary mission objectives:

(1) Two final operational readiness tests.

(2) Countdown for launch.

The requirements stated under Priority II above may, during brief periods, be reduced to Priority III to insure optimum use of the DSN resources in the best interests of NASA. This upgrading of priority classification can only be made by the Pioneer Project Manager or the Pioneer Mission Operations Manager.

Priority IV (*Non-critical*).—These requirements are mandatory for attaining primary mission objectives with no risk:

TABLE 8–4.—*Typical Tracking Requirements for a Pioneer Flight*

Time/distance coverage	Data required	Data presentation
Class–I requirement [a] A. Launch-vehicle second-stage engine cutoff (SECO) to SECO-plus-60 sec (fig. 8–11). B. Launch-vehicle third-stage burnout to third-stage spacecraft separation (minimum of 60 sec of data if available). Class–II requirement [a] A. SECO to SECO plus 180 sec. B. Ascension (ETR Station 12) rise to Ascension set.	Time, azimuth, elevation, range Data points per sec: 1/10 minimum, 1/6 desired, 1/3 maximum [b]	The data to be converted for presentation in NRT by teletype to the SFOF as follows: (a) Decimal raw-data format (b) Orbital elements and injection conditions of parking orbit (c) Orbital elements and injection conditions of transfer orbit assuming nominal third-stage burn (d) Orbital elements and injection conditions of transfer orbit based on actual third-stage burn
Class–III requirement [a] SECO to third-stage ignition; third-stage spinup to third-stage burnout; DSS tracking coverage sufficient to define the free-flight orbit (figs. 2–6 through 2–9).	Acceleration	Voice link and/or single-sideband data link in NRT; initially launch plus approximately 2 hr, and as required to meet accuracy requirements [c]

[a] See table 8–4 for priority definitions.

[b] These orbital criteria are stated to fulfill project orbital determination requirements only; they in no way reflect the tracking requirements established by the celestial mechanics experiment of J. Anderson. The celestial mechanics experiment will use as a data source the standard DSN two-way Doppler tracking data with the capability for 60-sec readout.

[c] The accuracy of the orbit based upon tracking data received from Deep Space Stations will be as follows:

Injection: 10 km and 2-Hz two-way Doppler
Injection-plus-10 days: 200 km and 5-Hz two-way Doppler
Injection-plus-180 days: 1000 km and 5-Hz two-way Doppler

(1) Thirty-first day to end of mission, continuous coverage

(2) For duration of mission, one horizon-to-horizon two-way-Doppler tracking mission every 4 days

(3) Coverage of specific solar events of high scientific value unrelated to specific flares

(4) Station time required to investigate a specific spacecraft characteristic or an operational hardware or software anomaly.

TABLE 8–5.—*Pioneer Telemetry Data-Acquisition Requirements*

Channel				
No.	Frequency	No. of segments	Data rates	Recording interval and accuracy
6A 7A	2292.037037 MHz 2292.407407 MHz	Transmitted in a 7 binary-bit data word on a 2048-Hz square wave, which is biphase-modulated with a time-multiplexed PCM bit train, using a non-return-to-zero-mark format. Telemetry data are formatted into 32-word main frames.	512, 256, 64, 16, and 8 bps selectable by ground command; when the convolutional coder is operating, a parity bit will be transmitted with each data bit; this will have the effect of doubling the transmitted bit rate to obtain the same data rates noted above (fig. 8–23).	Telemetry data are received, demodulated, synchronized, recorded and transmitted to an on-site SDS-920 computer. The computer decommutates and formats the telemetry data and outputs a portion of this data on teletype to the Ames Mission Operations Center. The aforementioned takes place at stations having Pioneer GOE during a tracking mission. At stations not having GOE, the output of the ground receiver is recorded during a tracking mission. During noncoverage tracking periods the telemetry data will be stored onboard for later readout. This mode is selectable by ground command.

[a] DSS–12 only.

TABLE 8–6.—*Pioneer Ground Command Requirements*

Coverage

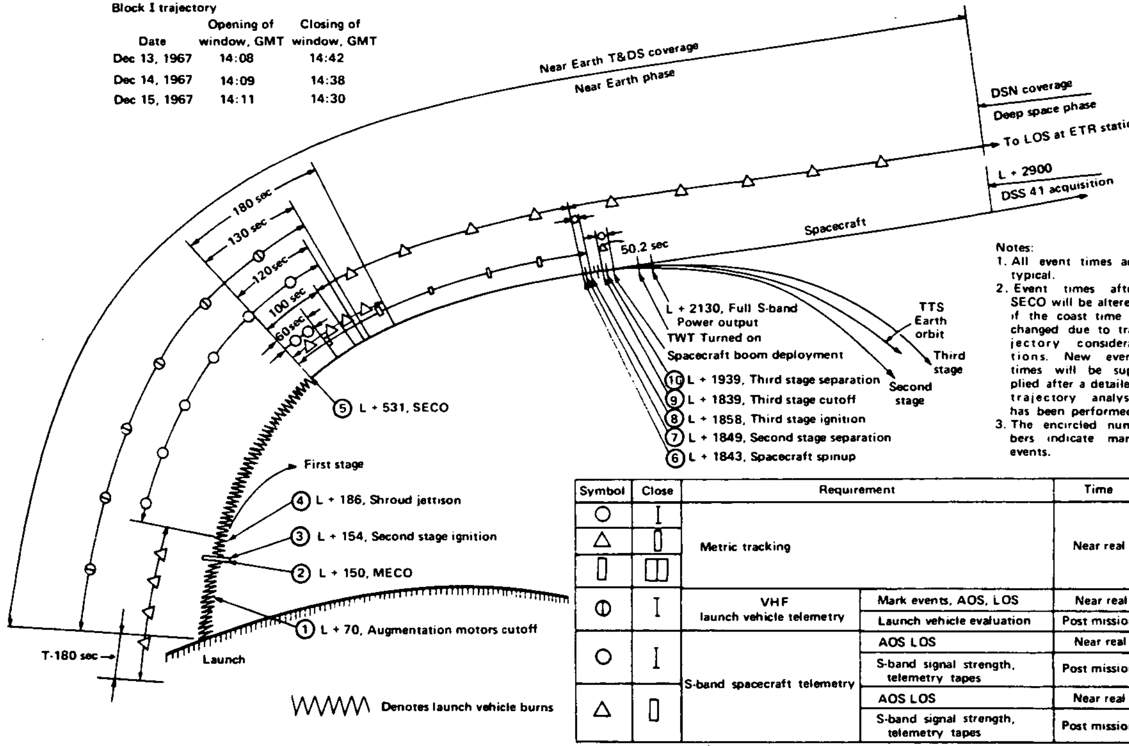
Time or phase of orbit
  Rise to set of DSS equipped with Pioneer mission-dependent GOE.
DSN stations
  DSS–12—Echo, Goldstone, Cal.
  DSS–42—Tidbinbilla, Australia
  DSS–62—Cebreros, Spain
  DSS–51—Johannesburg, South Africa
  DSS–71—Cape Kennedy, Fla. (launch checkout only)

Signal

Frequency
  Receiver 1, Channel 6B 2110.584105 MHz
  Receiver 2, Channel 7B 2110.925154 MHz
Modulation
  FSK/PM
Coding
  150-Hz and 240-Hz tones representing 0 and 1, respectively
Required transmitter power
  Variable; depends on spacecraft receiver signal strength which is dependent on range of 10-kW transmitter at a spacecraft signal strength threshold of −150 dBm
Transmission time required to execute commands
  Transmission time equals the transit light time to the spacecraft plus 23 sec; the transit light time varies with spacecraft range relative to Earth; the command message is 23 bits in length transmitted at 1 bit per sec.
Format
  23-bit word (See ch. 4 for details.)

Special equipment

Pioneer mission-dependent GOE
  Command encoder: produces encoded commands; indicates verification in DSS TCP computer and transmits 23-bit command message
  Computer buffer: provides connection for GOE and special-purpose equipment with the DSS TCP computer; provides serial input of command message from command encoder to DSS TCP computer
  DSS TCP computer operational program tapes: provide computer program to supply real-time processed data to NASA/ARC Mission Operations Center; also provides alarm and verifies information for command activity and Mission Control
  Permissive command tapes: provide computer with allowable commands for transmission.
DSS Mission-independent equipment
  DSS TCP computer: SDS–910/920 computer verifies command prior to transmission and checks bit-by-bit during transmission; provides stop signal on any non-permissive or non-verified command during transmission.



Block I trajectory

Date	Opening of window, GMT	Closing of window, GMT
Dec 13, 1967	14:08	14:42
Dec 14, 1967	14:09	14:38
Dec 15, 1967	14:11	14:30

Symbol	Close	Requirement		Time
○	I	Metric tracking		Near real
△	▯			
▯	◫			
Φ	I	VHF launch vehicle telemetry	Mark events, AOS, LOS	Near real
			Launch vehicle evaluation	Post mission
○	I	S-band spacecraft telemetry	AOS LOS	Near real
			S-band signal strength, telemetry tapes	Post mission
△	▯		AOS LOS	Near real
			S-band signal strength, telemetry tapes	Post mission

FIGURE 8–11.—Near-Earth tracking and telemetry requirements for the Pioneer 8-flight. From: JPL Rept. 607–90, p. 40.

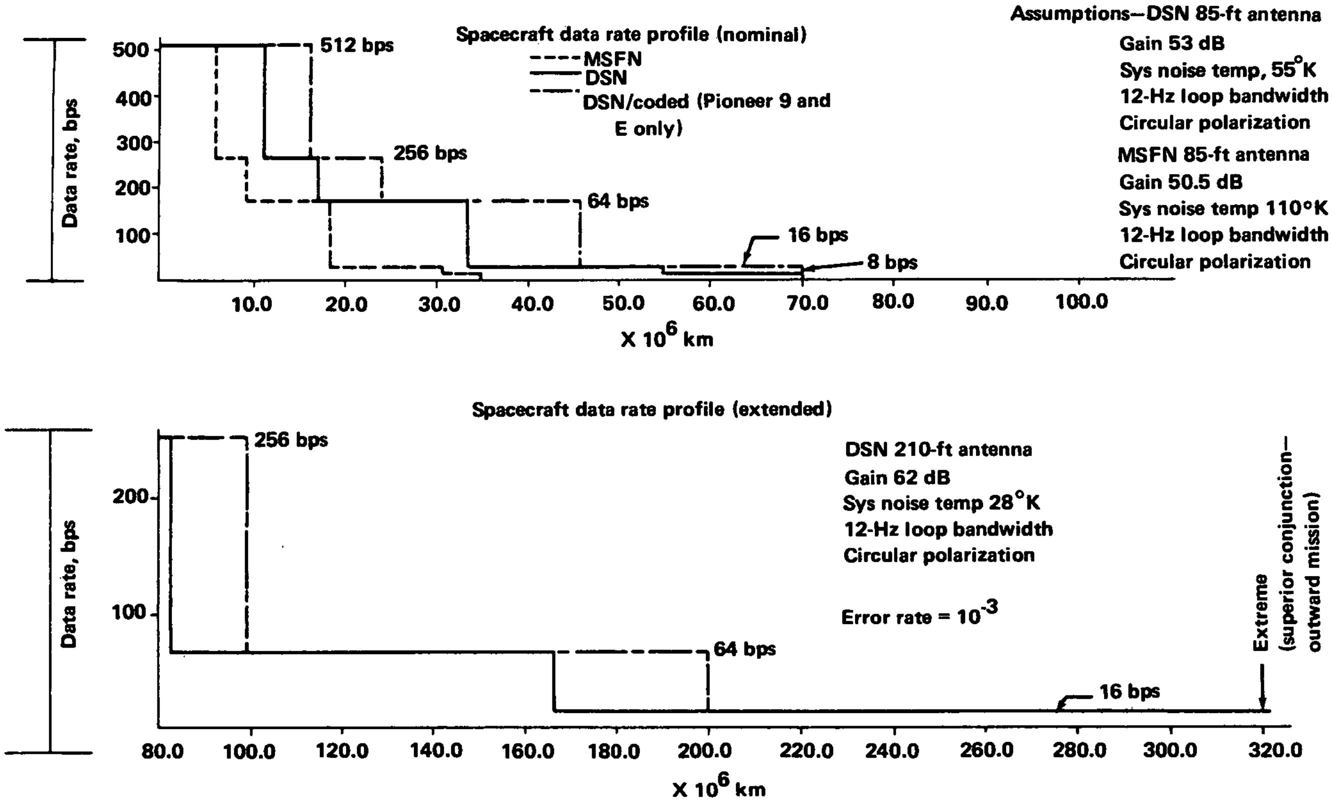


FIGURE 8–12.—Pioneer data rate profiles as functions of distance from Earth. From: SIRD–69, p. 510.

The above requirements are reduced to 6–8 hr per day to end of mission because of spacecraft/Earth distance, spacecraft- and ground-antenna characteristics, or because only one 210-ft antenna is available for operational support.

Priority V (Non-critical).—These requirements are not mandatory for attaining primary mission objectives:

(1) Any tracking coverage in excess of 24-hr coverage

(2) Operational integration testing

(3) Station time required to test a proposed modification to operational hardware or software.

SPECIFIC PIONEER NETWORK CONFIGURATIONS

The terrestrial facilities that NASA pooled to meet the requirements of the Pioneer flights consisted of parts of the following facilities:

1. The DSN, which included the DSIF, GCF, and SFOF
2. The MSFN, which provided 85-ft-dish support on occasion
3. NASCOM, which contributed many circuits to the DSN's GCF
4. The AFETR (Air Force Eastern Test Range), which supplied much of the ground environment from the launch pad downrange 5000 miles to Ascension Island, i.e., the Near-Earth Phase Network

Each Pioneer flight could be divided logically into two main phases: near-Earth phases and deep-space phases. The successful injection of the spacecraft into a heliocentric orbit was the event that effectively separated the two phases (fig. 8–11). At this point, somewhere over the Indian Ocean, the spacecraft would be handed over completely to the DSN and cooperating MSFN stations. Each phase of tracking required a different configuration of tracking, data acquisition, command, and ground communication equipment (ref. 3).

Near-Earth-Phase Network Configurations

The equipment committed to the Pioneer Program varied slightly from flight to flight, as detailed in table 8–7. The stations along the AFETR had the primary responsibility for tracking (or "metric data") during the launch and Earth-orbit portions of the flights. Cape Kennedy has many radars, radio interferometers, and a great variety of optical tracking equipment. AFETR and MSFN downrange stations and Range Instrumentation Ships (RIS) also possess impressive complements of tracking radars and telemetry receiving equipment. Data are fed back to Cape Kennedy via submarine cables and radio links.

The DSN station at the Cape (DSS–71) was an integral part of all DSN configurations supporting Pioneer flights during the near-Earth passes. JPL also maintains a field station at Cape Kennedy that provides an

TABLE 8–7.—*Configuration of Tracking and Data Acquisition Stations During Near-Earth Phases*

Station number		Location	Tracking radars	Telemetry	Use during Pioneer flights			
					6	7	8	9
AFETR	1	Cape Kennedy	FPQ–6 FPS–16, TPQ–18	vhf, S-band	X	X	X	X
	3	Grand Bahama	FPS–16, TPQ–18	vhf	X	X	X	X
	7	Grand Turk	TPQ–18	vhf	X	X	X	X
	91	Antigua	FPQ–6	vhf, S-band	X	X	X	X
	12	Ascension	FPQ–18, FPS–16	vhf, S-band	X	X	X	X
	13	Pretoria	MPS–25	vhf, S-band	X	X	X	X
	—	*Twin Falls* (ship)	FPS–16		X	X	X	X
	—	*Coastal Crusader* (ship)			X	X	X	X
MSFN	1	Bermuda	FPS–16, FPQ–6	vhf	X	X	X	X
	2	Ascension			X	X	X	X
	3	Tananarive, Malagasy	Capri	vhf	X	X	X	X
	4	Carnarvon, Australia	FPQ–6	vhf	X	X	X	X
	5	Goddard Space Flight Center, Greenbelt, Md.			X	X	X	X
	6	Guam				X	X	X
	7	Hawaii				X	X	X
DSN	71	Cape Kennedy			X	X	X	X

TABLE 8-7.—*Configuration of Tracking and Data Acquisition Stations During Near-Earth Phases*—Concluded

Station number	Location	Tracking radars	Telemetry	Use during Pioneer flights			
				6	7	8	9
72	Ascension			X	X	X	X
51	Johannesburg			X	X	X	X
—	SFOF, Pasadena			X	X	X	X
—	Building AO, Cape Kennedy			X	X	X	X

operational interface between the SFOF, in Pasadena, and the Air Force and Goddard Space Flight Center groups. In view of the manifold operations at Cape Kennedy, their complex interactions, and the immense detail required for effective coordination, such interface groups are essential. The JPL field station also contains an Operations Center with abundant displays of different types to help JPL personnel operate range instrumentation under their control. Critical tracking and telemetry data are also routed to the SFOF through the field station.

All launches from Cape Kennedy are under the direct control of the Air Force until the spacecraft leave ETR jurisdiction somewhere beyond Ascension. Because it is responsible for range safety, the Air Force monitors launch vehicle status data and tracking information. Commands to terminate the mission through the destruction of the launch vehicle are also an Air Force prerogative—one which was exercised during the launch of Pioneer E on August 27, 1969,

In summary, the near-Earth phase of a Pioneer flight is scrutinized by dozens of radars, theodolites, and interferometers from Cape Kennedy to South Africa. Telemetry and tracking data flow back to the Cape and the SFOF where they are monitored by the Air Force, NASA, and JPL personnel. Operational control rests with the Air Force during the launch phase and is handed over to NASA when the Pioneer spacecraft has been injected into heliocentric orbit.

Deep-Space-Phase Configuration

After leaving Earth orbit, the Pioneer spacecraft quickly ascended beyond the 500- to 1000-mile ranges of the AFETR and MSFN tracking radars. From then on, they were tracked, communicated with, and commanded by the primary DSS stations listed in table 8–1. MSFN and other DSIF stations worked the Pioneer spacecraft on an as-needed basis. Communication traffic flowed back to the SFOF and NASA/ARC over GCF lines; commands, of course, moved in the opposite direction (fig. 8–13).

Each of the primary DSS stations was outfitted with so-called "mission-dependent" equipment that accommodated general-purpose DSIF machinery to specific Pioneer requirements. In Pioneer vernacular, the DSS gear was called Ground Operational Equipment (GOE). No special equipment was installed at the SFOF, although a general-purpose mission-support area was reconfigured for the Pioneer missions (fig. 8–10). Additional mission-dependent equipment was installed at Ames Research Center (fig. 8–6). Since the presence of Pioneer mission-dependent equipment constituted the major difference between a DSS station in the Pioneer

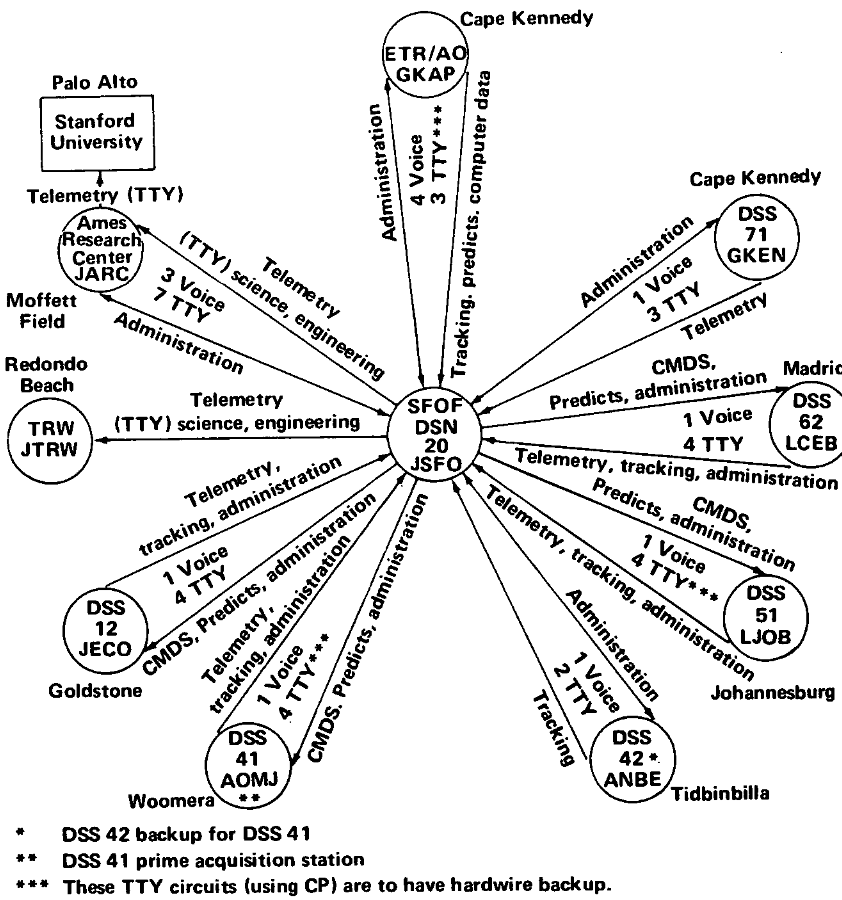


FIGURE 8–13.—GCF channels established for Pioneer 8.

configuration and any other mission-dependent configuration, a few details are in order.

The Pioneer GOE was designed to make maximum use of the general-purpose DSS equipment, particularly the Telemetry and Command Processor (TCP) equipment (the SDS–910 and SDS–920 computers) (fig. 8–14). Type-I GOE, consisting of five racks of electronic hardware, plus a module tester, a test transponder, and an instructor control set, was installed only at the Goldstone site. The primary overseas DSS stations received Type-II GOE, consisting of three racks only. The two extra racks at Goldstone were the recorder and display racks employed during the spacecraft Type-II orientation maneuver. The Woomera site (DSS–41) possesses no GOE equipment. The specific pieces of equipment in both types of GOE are indicated in the labels on figures 8–15 and 8–16. As the block diagram, figure 8–14, indicates, the GOE was actually specialized interface between the antenna and the DSS equipment.

TELEMETRY CAPABILITIES

Telemetry capabilities were provided as follows:

(1) Provided up to 120 frames of continuous spacecraft telemetry data for near-real-time teletype transmission to the SFOF (TCP)

(2) Provided selected spacecraft engineering data for near-real-time teletype transmission to the SFOF (TCP)

(3) Provided local typewriter printout in real time at each DSN station of selected spacecraft engineering data periodically and upon request (This capability accommodates operational requirements during spacecraft initial acquisition as well as routine orbital operations.) (TCP)

(4) Drove local DSIF displays for spacecraft parameters necessary for uplink acquisition and verification of spacecraft receiver lock (TCP)

COMMAND CAPABILITIES

Command capabilities were provided as follows:

(1) Encoded manually inserted commands in a 23-bit format compatible with the Pioneer spacecraft command system (GOE)

(2) Generated an FSK signal suitable for exciting the DSIF S-band transmitter phase modulator at a rate of 1 bit per sec (GOE)

(3) Established a means for preventing the transmission of any command not entered in the "permissive command list" (TCP)

(4) Verified that the transmitted command corresponded to the manually inserted command and terminated transmission of commands in which errors are detected (TCP)

(5) Provided a typewriter printout of spacecraft command status, a notation of the transmitted commands and their time of transmission.

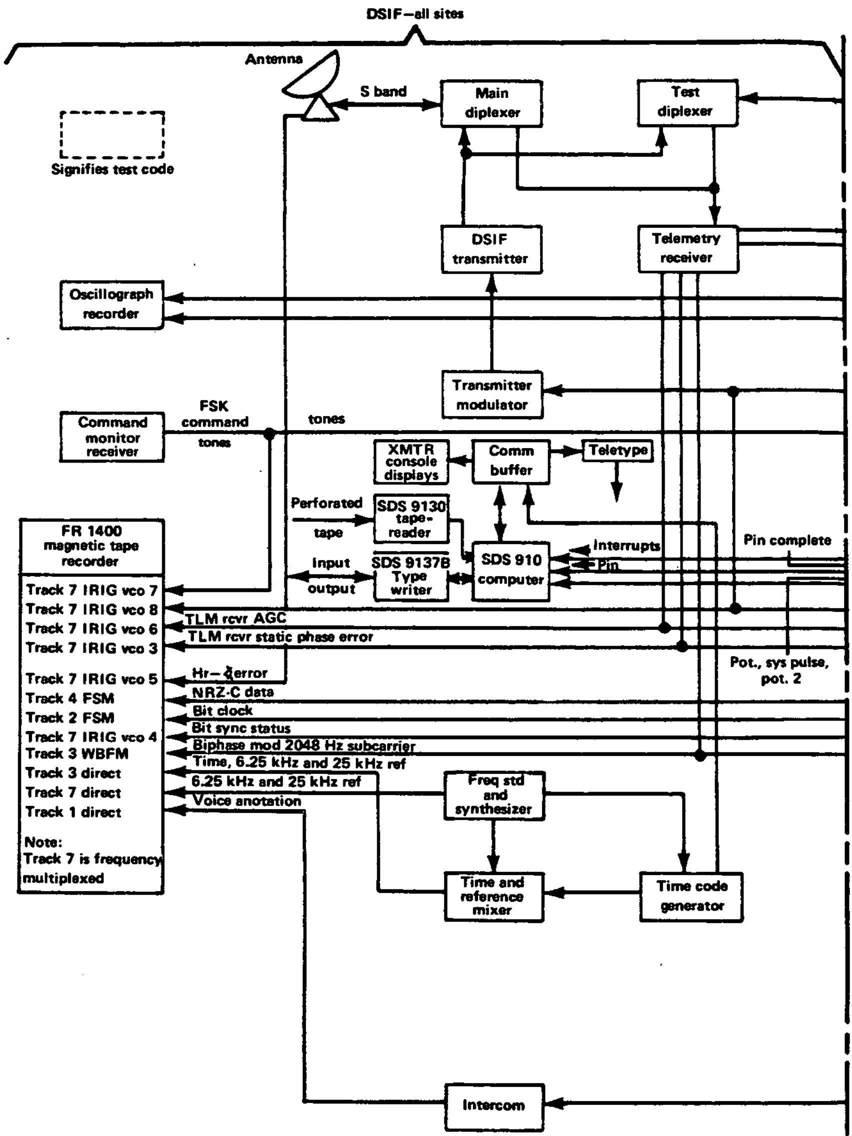


FIGURE 8-14.—Functional block diagram of Pioneer GOE at the DSIF sites.

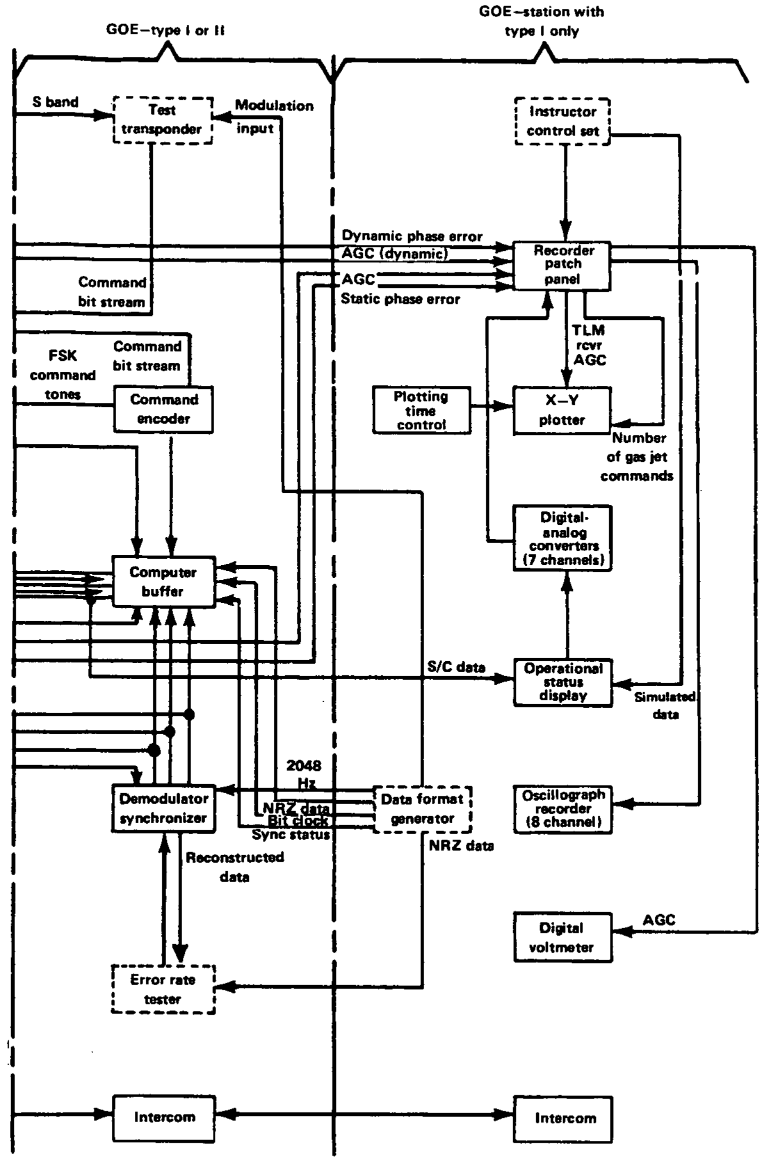


FIGURE 8–14.—Concluded.—Functional block diagram of Pioneer GOE at the DSIF site.

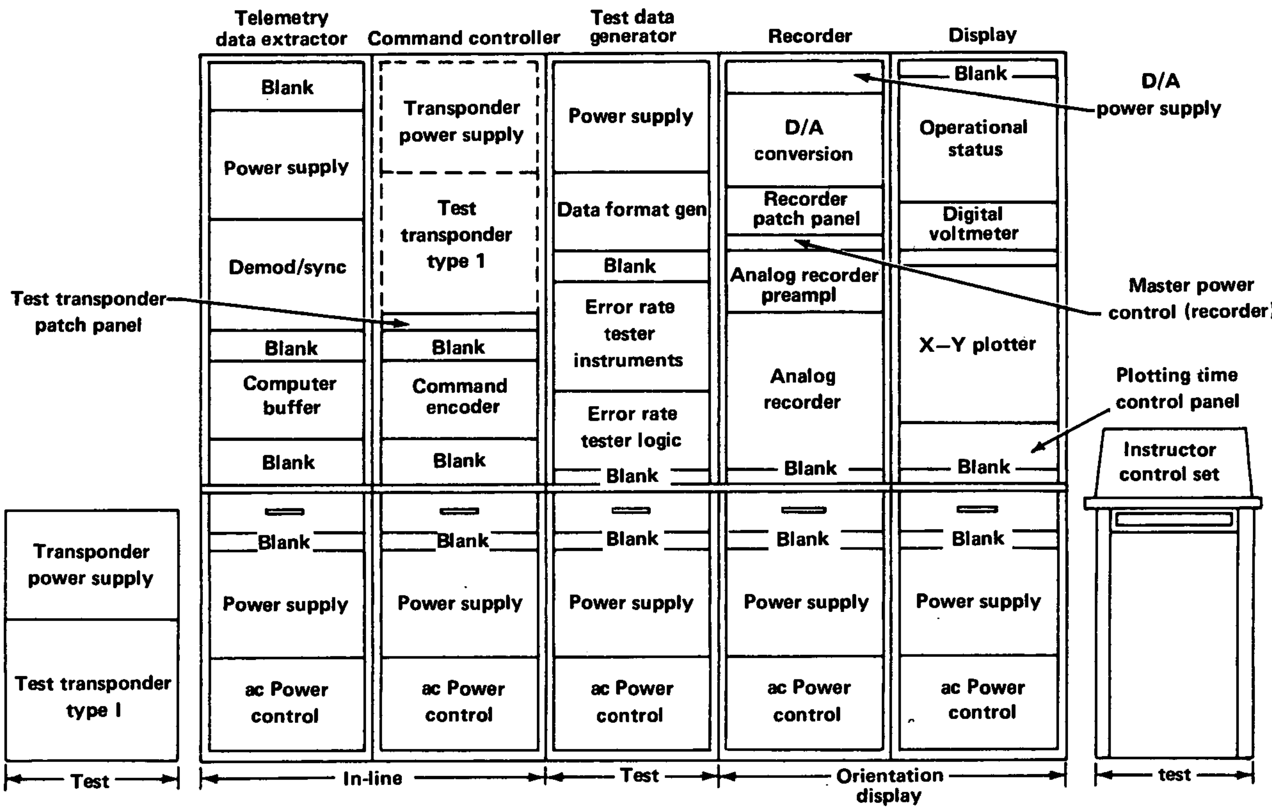


FIGURE 8–15.—The five racks of Type-I GOE at Goldstone.

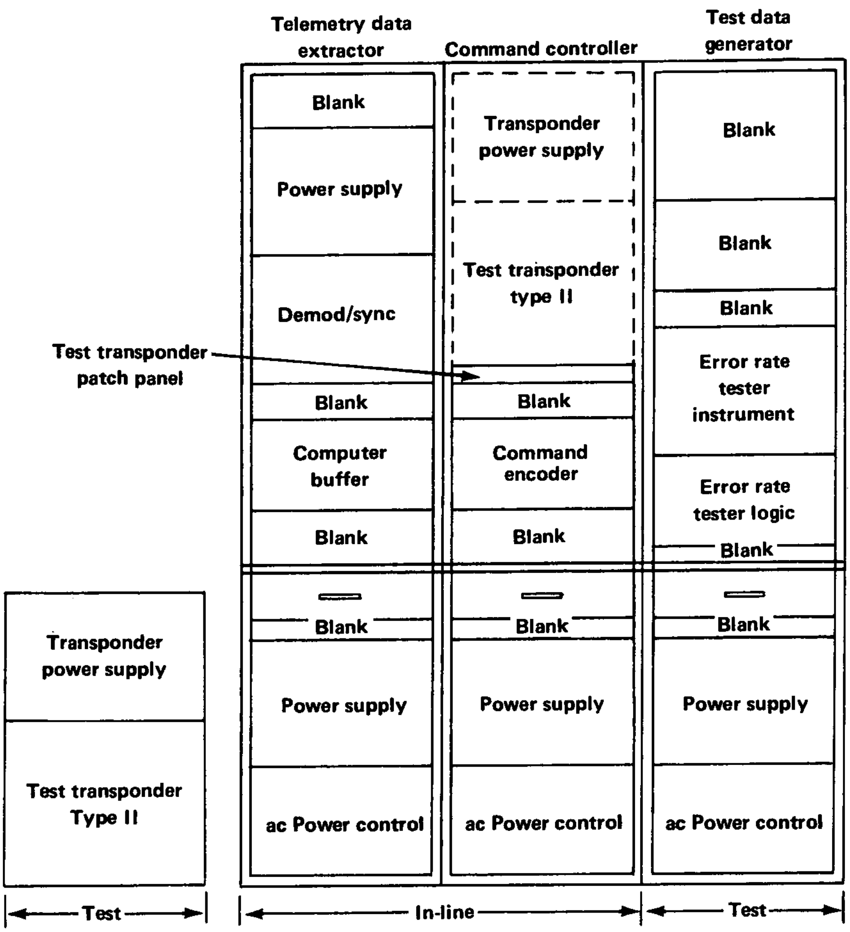


FIGURE 8-16.—The three racks of Type-II GOE at overseas DSIF stations.

(Verification that the command had been executed on the spacecraft was done by the controller at the Pioneer Mission Operations Center at Ames Research Center.) (TCP)

(6) Organized command data and command status into a format suitable for near-real-time teletype transmission to the SFOF (Again, command verification was done at Ames.) (TCP)

The second DSN facility that assumed specific configurations especially tailored for the Pioneer mission was the Ground Communications Facility. The configurations varied from flight to flight and even during the same mission. It is impractical to catalog all these changes; the arrangement for Pioneer 8 was rather typical, and it is reproduced in figure 8-17.

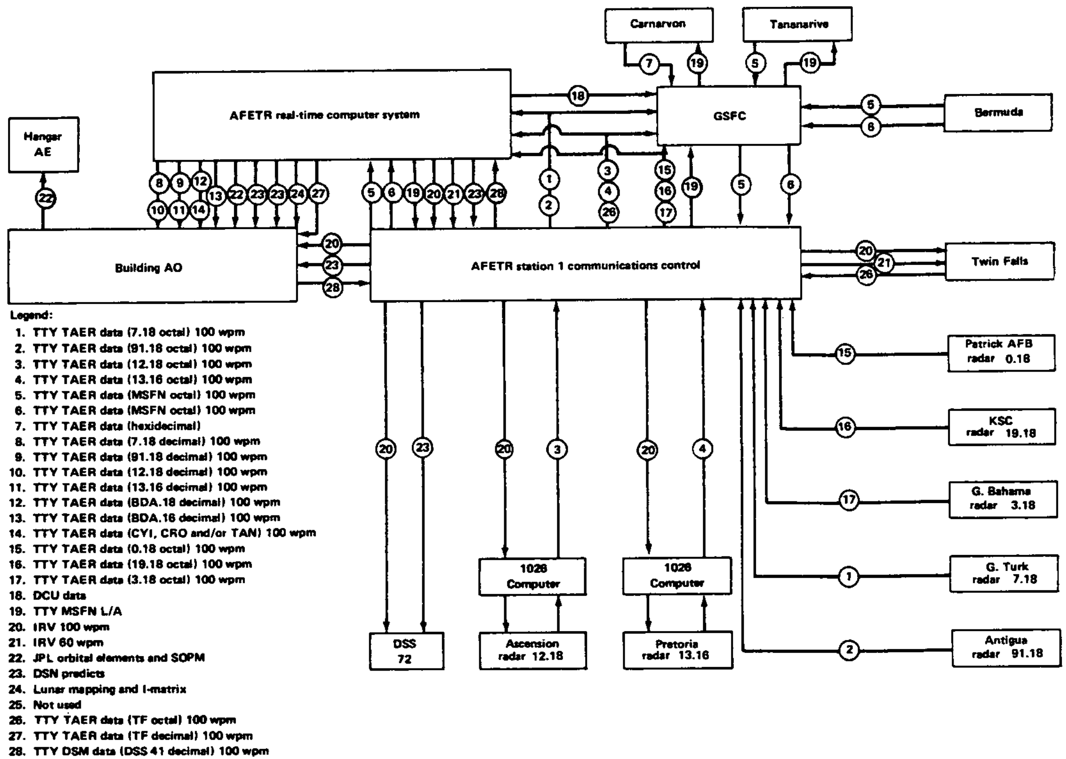


FIGURE 8-17.—Flow diagram for near-Earth-phase tracking data for the Pioneer 8-mission.

A similar situation existed with the SFOF; it is another general-purpose facility that was modified to accommodate Pioneer requirements. A Pioneer Mission Support Area was set up at JPL's SFOF for use as an operational control center during the launch phases of Pioneer flights when activity was high. Spacecraft performance and scientific data analyses were also carried out there. One of the special SFOF Mission Control Rooms and an associated Flight Path Analysis Area were made available to the Pioneer Project during critical phases of each mission. A data flow diagram (fig. 8–18) illustrates the routing of data within and without the SFOF during the Pioneer Program. The dispatch of data packages to Ames Research Center (fig. 8–18) completed the DSN role in data processing and handling. Ames processed data tapes and passed scientific data on to the experimenters, completing the data link from spacecraft to scientist.

Cruise portions of the flights were controlled at Ames where spacecraft and instrument expertise were readily available. SFOF space was used for control only during the launch phase or in the event of extremely critical periods.

REFERENCES

1. ANON.: DSN Capabilities and Plans, Vol. I. Description of Deep Space Network Capabilities as of 1 July 1968. JPL Rept. 801-1, 1968.
2. ANON.: Support Instrumentation Requirements Document, Pioneers 6, 7, 8, 9, and E. NASA, Mar. 1969.
3. RENZETTI, N. A.: Tracking and Data System Final Support for the Pioneer Mission, Pioneer 6 Pre-Launch to End of Nominal Mission. JPL Rept. TM 33–426, 1970. (A similar report exists for each Pioneer flight.)

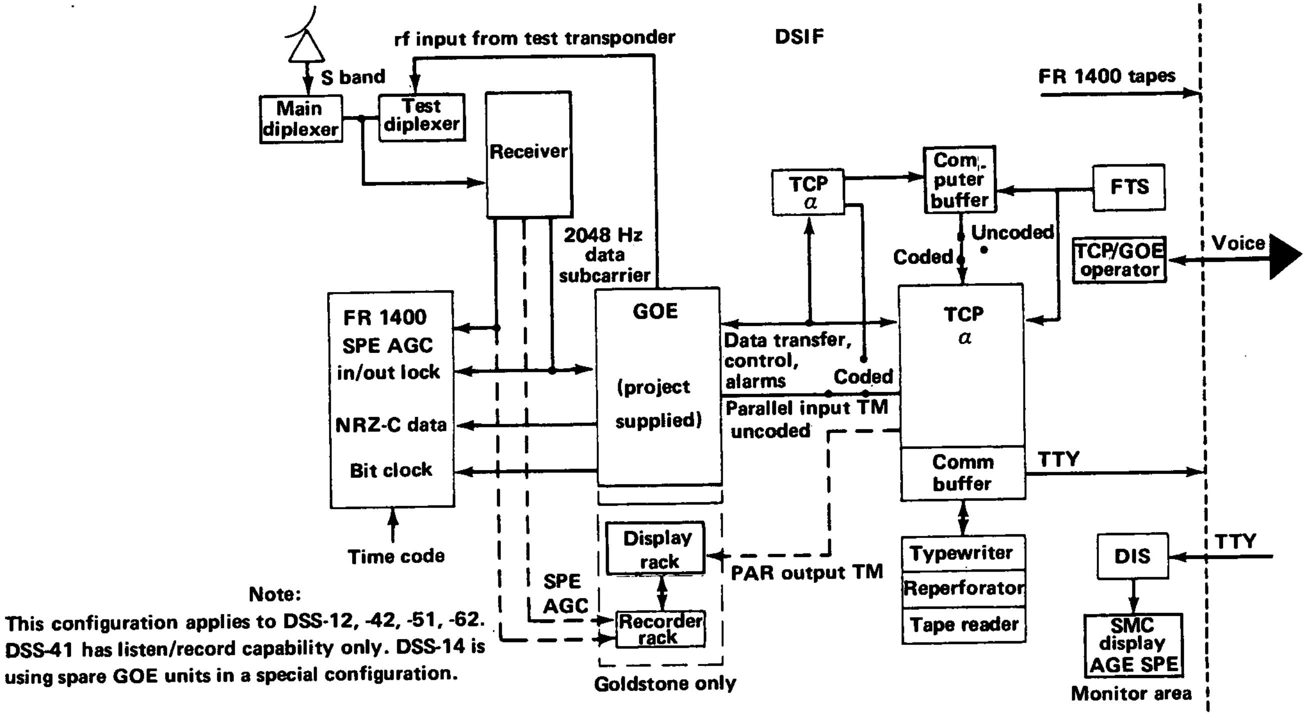


FIGURE 8–18.—DSN telemetry configuration.

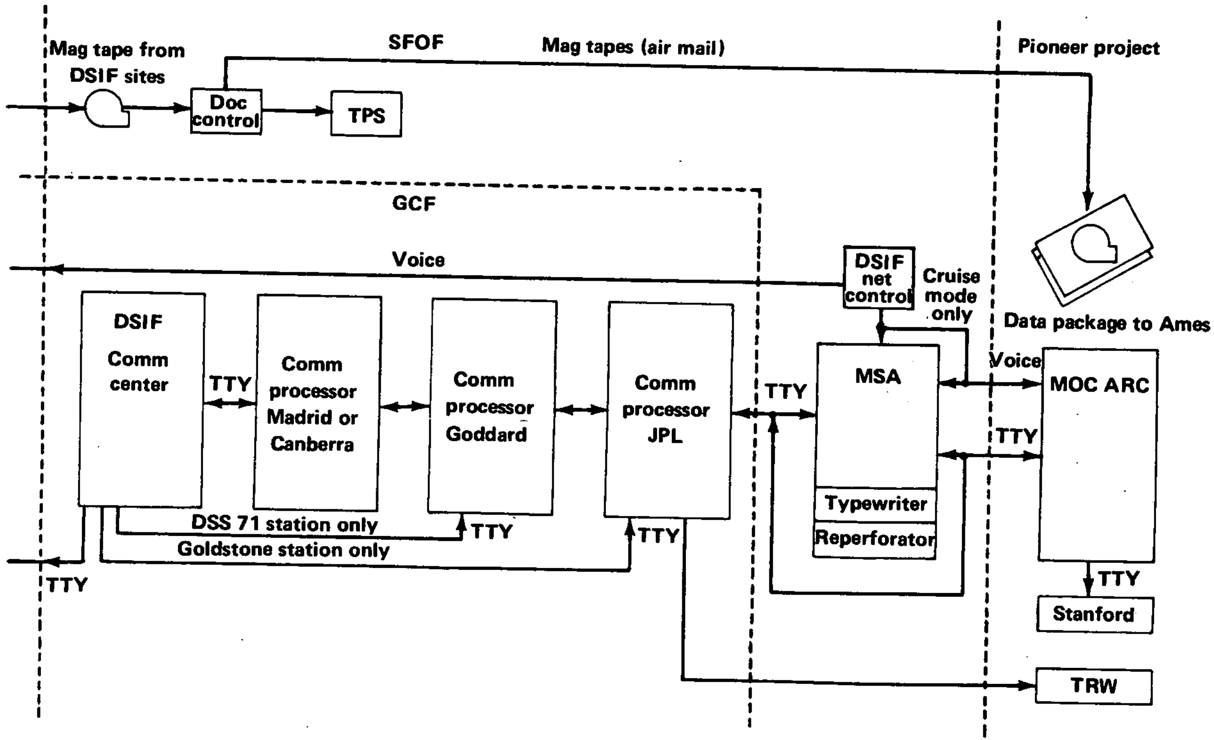


FIGURE 8–18.—DSN telemetry configuration.—Concluded.

CHAPTER 9

Pioneer Data-Processing Equipment

PIONEER SPACECRAFT radioed back to Earth two kinds of data: (1) scientific data for the several Pioneer experimenters, and (2) engineering data to permit the mission controllers to assess the operational condition of the spacecraft. Referring to figure 8–18, one sees that the telemetry data follow two paths between the DSN stations, which receive it directly from the spacecraft, and the experimenters and Pioneer project personnel. Pioneer telemetry data are recorded directly on magnetic tape as they arrive from deep space at the DSN stations and airmailed to JPL for verification and then to Ames Research Center. This is the first route, and all data follow it. At Ames, they are processed on the Pioneer Off-Line Data-Processing System (POLDPS) for subsequent transmission to the experimenters on digital magnetic tapes in formats compatible with their computer facilities. Some of the telemetry data, however, also follow a second route. These are dispatched immediately from the DSN to Ames Research Center via teletype through JPL's SFOF. These are called "quick look" data; they are used for checking the scientific instruments and for retransmission (after some processing) to ESSA to help forecast solar activity. Data from the Stanford radio propagation experiment are handled differently. As described in chapter 5, proper operation of this experiment requires the near-real-time feedback to Stanford of information on the Stanford receiver status. This information is relayed by teletype from Ames Research Center to Stanford a few miles away. In addition, engineering data flow via teletype from the DSN station to the SFOF and thence to both Ames and TRW Systems for analysis. At Ames, these engineering data are used to assess the condition of the spacecraft and help make operational decisions.

The data-processing facilities at Ames and TRW Systems are described below.

PIONEER OFF-LINE DATA-PROCESSING SYSTEM

In 1964 when JPL computers were heavily loaded, the decision was made to construct the processing line at Ames Research Center. Bids from ten companies were received in late 1964 as a result of NASA solicitation. Computer Sciences Corporation (CSC) received the prime POLDPS contract on January 7, 1965. Astrodata, Inc., was the subcontractor responsible for building the POLDPS hardware. By the summer of 1965, POLDPS was ready for operation.

Magnetic tape represented the only practical way to transmit the bulk of the data from the active Pioneer spacecraft—teletype facilities could not handle the volume. At each DSN station, two Ampex FR–1400 tape recorders operating in parallel prepared analog tapes of the transmissions received from the Pioneers. Tape loading times for each machine were staggered to avoid the loss of data. One set of tapes containing all recorded data were selected and shipped first to JPL for verification to ensure the quality of reproduction (fig. 9–1). The tapes were then sent to the Pioneer Off-Line Data Processing System at Ames Research Center.

During 1969, Pioneer tape shipments averaged 400 (9200-ft) tapes per month, each containing 4 hr worth of data with half-hour overlaps. POLDPS processed and sorted out these data, preparing an average of 400 (2400-ft) tapes per month for the experimenters. The preparation of over 15 experimenter tapes per working day indicates that POLDPS was extremely active during 1969, when four Pioneers were transmitting data back to Earth (fig. 9–2).

The input to POLDPS consists of the FR–1400 magnetic tapes received by airmail from DSS sites around the world. The following seven channels are recorded on these tapes at 5.5 in./sec:

(1) Voice (containing station events)

(2) Bit clock data from the DSS demodulator/bit synchronizer

(3) Universal time and 6.25- and 25-kHz reference signals

(4) NRZ-C data (see ch. 4)

(5) The biphase-modulated 2048-Hz subcarrier containing the time-multiplexed PCM-data bit train

(6) Spare channel

(7) Various DSS data, such as static phase error, sync status, antenna error, command tones, etc.

POLDPS processes these tapes in a two-level system. (fig. 9–3) The first level, called the tape processing station (TPS), produces a multifile digital tape that serves as the input to the second level of processing, which consists of the Pioneer off-line direct-coupled system (POLDCS). POLDCS generates separate experimenter tapes that are IBM-compatible and in the formats and densities desired by the individual Pioneer experimenters.

TAPE PROCESSING STATION

The TPS consists of an FR–1400 Tape Recorder-Reproducer, an STL Bit Synchronizer/Demodulator, an SDS–910 computer, Astrodata interface hardware, and the necessary software to control the equipment and process the input data into a multifile digital tape acceptable to POLDCS (ref. 1). Data recorded at any DSS can be processed even if the station does

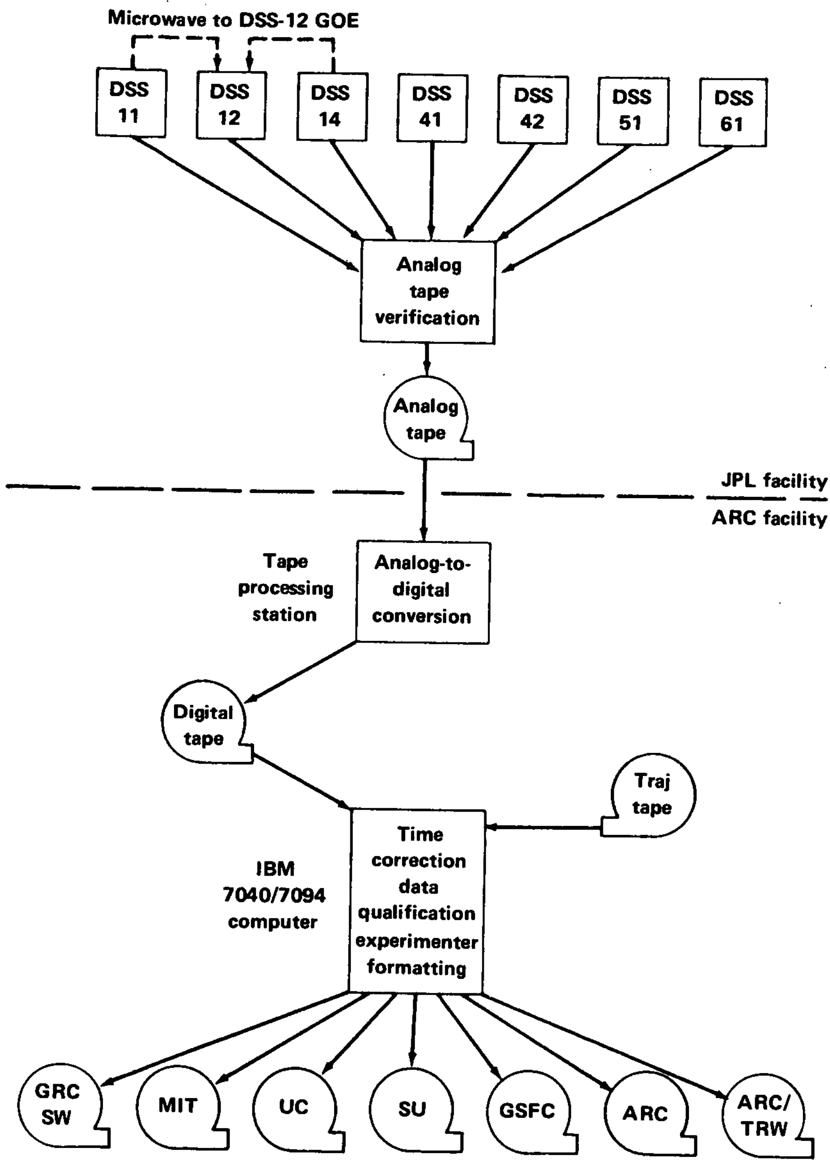


FIGURE 9-1.—The Pioneer off-line data processing station showing data flow paths.

not possess the Pioneer-unique GOE described in chapter 8. The TPS performs the following functions:

(1) Establishes frame and word synchronization of the telemetry data from either the NRZ bit stream or the biphase-modulated subcarrier

(2) Provides a bit clock from the recorded signal or the output of the bit

FIGURE 9-2.—Pioneer Off-Line Data Processing System (POLDPS) at Ames Research Center.

synchronizer

(3) Generates time information (in days, hr, min, sec, and msec) from the bit-clock channel

(4) Demodulates PCM signals and sync information from the raw data

(5) Digitizes analog functions

(6) Converts the FSK command data into digital format

(7) Records the above information on IBM-compatible computer tapes at 556 characters per in.

All TPS operations are performed automatically except for the handling of tape reels, patch-board wiring, installation of plug-in units, and TCP control.

PIONEER OFF-LINE DIRECT-COUPLED SYSTEM

POLDCS digests the TPS output tapes and performs the following functions:

(1) Selects the "best" telemetry data from multiple input sources

(2) Evaluates the quality of the telemetry data

(3) Converts ground station time into spacecraft time

(4) Calibrates the engineering data

(5) Decommutates the evaluated telemetry data

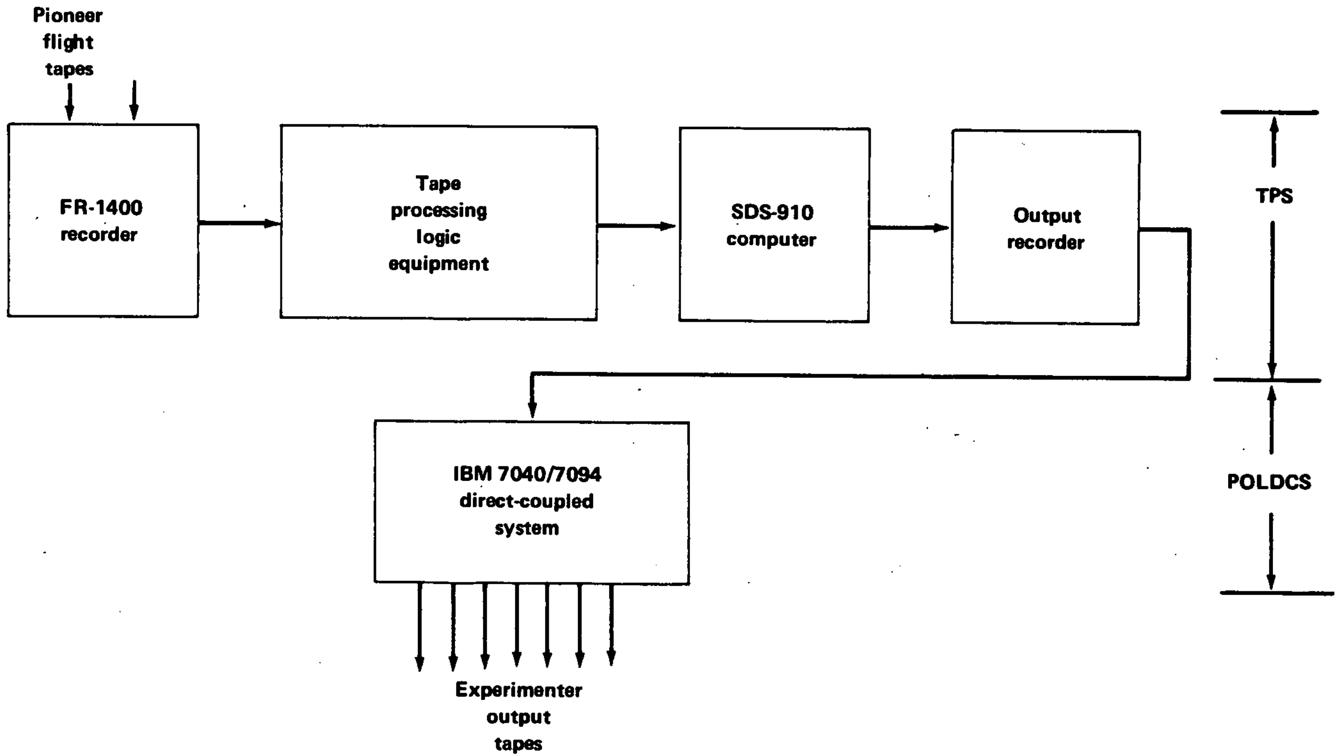


FIGURE 9–3.—Block diagram of Pioneer off-line tape processing.

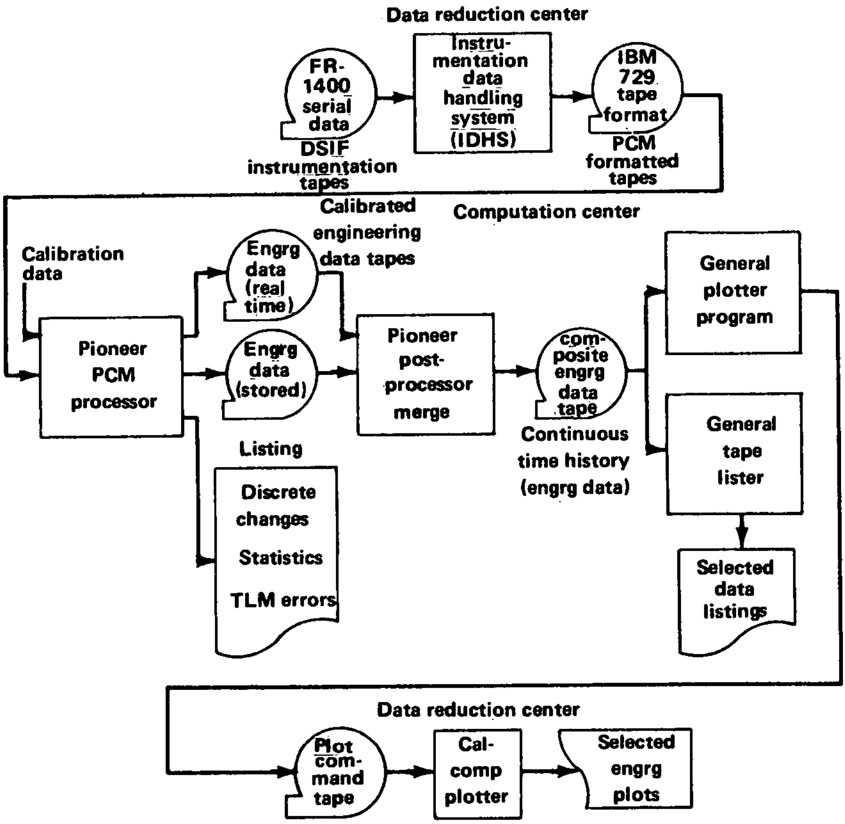


FIGURE 9–4.—Block diagram of data processing line at TRW Systems used on occasion for Pioneer tapes.

(6) Prepares the magnetic tapes for the individual Pioneer experimenters

Because the Pioneer data rates are usually very low (in comparison to those from scientific satellites, for example) NASA felt that some indication of data quality ought to be added to the experimenter tapes. The so-called data condition indicator (DCI) was established, using information from Channel 7 of the analog tapes received at Ames. A numerical code added to the experimenter tapes indicated the following conditions for each spacecraft word:

40	Filled word
36	Replaced value (known words only)
22	FSK command inserted
20	Command from another DSS station

10 DSS receiver out of lock
04 Bit error probability in excess of 10^{-3}
02 DSS bit synchronizer out of lock
01 Parity error
00 Good data

The experimenter tapes are supplemented by trajectory tapes giving the spacecraft position as a function of time. The basic trajectory tapes are prepared at JPL, but Ames reprocesses them to put them in the formats and densities requested by the experimenters.

POLDPS has remained substantially the same throughout the Pioneer program. The only significant change was made for Pioneer 9, which carried the convolutional coder designed to improve the quality of Pioneer data. The effects of the addition of the convolutional coder upon Pioneer telemetry are described in chapter 4.

DATA PROCESSING AT TRW SYSTEMS

TRW Systems receives copies of the FR–1400 analog tapes made at the DSS stations for the first 4 days following a launch. As spacecraft prime contractor, TRW Systems was primarily interested in the engineering data on these tapes. The output from the TRW Systems data-processing line consists mainly of tabulated engineering data and automatically plotted engineering parameters suitable for assessing spacecraft performance as a function of time (fig. 9–4).

The FR–1400 tapes are first formated at the TRW Systems Data Reduction Center by the Instrumentation Data Handling System (IDHS). Formating prepares them for an IBM 7094, which next edits, sorts, and calibrates the data. In addition, the Pioneer General Data Processing Program performs the following operations:

(1) Prepares statistics in the form of minima, maxima, and averages for each engineering parameter

(2) Summarizes those telemetry errors detected

(3) Monitors digital data

The computer prints out tabular data and prepares a magnetic tape for input to a CalComp Plotter, which draws the graphs of desired engineering parameters.

The TRW Systems processing line described above was not used on a regular basis.

REFERENCE

1. ANON.: Pioneer Off-Line Data Processing System—System Manual. Computer Sciences Corp. Rept., 1965.

Bibliography

Included here are all engineering reports and papers of significance dealing with the four Pioneer systems: spacecraft, instruments, launch vehicle, and tracking and data acquisition. Because a great deal of Pioneer descriptive information is available only in the program specifications, a comprehensive list of these follows this bibliography. Volume III includes the bibliography covering Pioneer operations and scientific results.

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Pioneer Specifications

THE TWO A-SERIES SPECIFICATIONS were first used during the solicitation of spacecraft and instrument proposals. They were updated frequently later. Specification A–6669 was updated with almost every modification of the contract with TRW Systems. P-series specifications were issued early in the program and replaced by the PC-series. The PC numbers under 100 apply to Block I; numbers between 100 and 199 apply to Block II. Specifications were frequently updated and revised.

A–6669	Spacecraft and Associated Ground Equipment (12–1–63)
A–7769	Scientific Instrument Specifications (12–31–64)
P–1	Documentation Procedures (1–2–64)
P–2	Amendments to NASA Quality Publications (12–1–63)
P–3	Amendment to MSFC-PROC-158B (6–19–64)
P–4	Project Development Plan
P–5	Launch Vehicle Performance Analysis for the Pioneer Program (12–1–64)
P–6	Classification Guide for Project Pioneer (12–15–64)
P–8	An Analysis of the Effects of the Spacecraft Systems and TAD Launch Vehicle on the Pioneer Trajectories (8–1–64)
P–9	Simulator Description (8–12–64)
P–10	Pioneer Time Resolution of Telemetry (8–28–64)
P–11	Procedures for the Preparation and Processing of Range Documentation (2–15–65)
P–12	Pioneer Canberra GOE ARC/STL Acceptance Test Results, July 1, 1965 (7–1–65)
P–13	Pioneer Johannesburg GOE ARC/STL Acceptance Test Results, July 1, 1965 (7–1–65)
P–14	Pioneer Ascension GOE ARC/STL Acceptance Test Results, July 1, 1965
P–15	Pioneer Goldstone GOE ARC/STL Acceptance Test Results, July 1, 1965 (7–1–65)
P–16	Pioneer Goldstone GOE Approved Engineering Configuration (9–1–65)
P–17	Engineering Configuration Pioneer GOE Serial 002 (Canberra) (9–1–65)
P–18	Engineering Configuration Pioneer GOE Serial 003 (Johannesburg) (9–1–65)
P–19	Engineering Configuration Pioneer GOE Serial 004 (Cape Kennedy)
P–20	Engineering Configuration Pioneer GOE Serial 005 (Ascension) (9–1–65)
P–21	Preliminary Evaluation of Pioneer Compatibility with CKAFS Facilities (8–6–65)
PC–1	Spacecraft/Scientific-Instrument Interface Specification (3–6–64)
PC–2	Spacecraft/DSIG/GOE Interface Specification (8–3–64)
PC–003	Project Development Plan (11–16–64)
PC–010	S/C & Associated Ground Equipment (A–6669) (renumbered)
PC–011	General Instrument Specification (A–7769) (renumbered)
PC–013	Pioneer GOE Specification (8–3–65)
PC–020	Pioneer Solar Array Checkout at Table Mountain (6–5–64)

PC–021	Spacecraft/Scientific-Instrument Interface Specification
PC–023	Spacecraft/Launch Vehicle Interface Specification (7–19–65)
PC–025	Scientific Instrument Integration Activities (8–1–64)
PC–030	GOE Installation, Integration and Compatibility (4–19–65)
PC–046	Pioneer Flight Operations (2–19–65)
PC–047	Flight Operations Test Plan (8–2–65)
PC–050	Procedures for Pioneer-A Flight Operations Test (11–24–65)
PC–051	Pioneer A Flight Operations—Detailed Task Sequence
PC–052	Pioneer A Flight Operations—Detailed Task Sequence for Participating Groups
PC–053	Pioneer A Flight Operations, Stanford Procedures (11–15–65)
PC–054	Pioneer-B Flight Operations
PC–060	Pioneer Off-Line Data Processing System at ARC (8–28–64)
PC–062	Pioneer Permissive Command Tape Program (10–20–67)
PC–064	Pioneer-6 and 7 DSIF Operational Computer Program Requirement Specification (5–23–68)
PC–070	Pioneer Solar Array Checkout at Table Mountain (PC–070) (renumbered)
PC–071	Activities at the Air Force Eastern Test Range (6–19–64)
PC–072	Scientific Instrument Integration Activities
PC–073	Pioneer Spacecraft/DSS-71/GOE Compatibility Test Specifications (8–12–65)
PC–080	Tests of Scientific Instruments at ARC
PC–081	Scientific Instrument Test Requirements (1–29–65)
PC–083	Experiment Tests at Malibu Coil Facility—Master Test Procedures (3–31–65)
PC–084	Procedure for GOE/DSIF Compatibility Tests (5–18–65)
PC–085	Pioneer Spacecraft/DSIF-71/GOE On-Stand Compatibility Test Procedures (5–27–66)
PC–090	Pioneer-A Trajectories
PC–091	Pioneer-B Trajectories
PC–092	Pioneer GOE-DSIF User's List (6–14–65)
PC–093	Maintenance and Configuration Control (11–1–65)
PC–094	Data Format Generator, Type II—Operation and Maintenance Manual Ground Operational Equipment (5–16–66)
PC–111	Pioneer Instrument Specification (12–23–64)
PC–121	Spacecraft/Scientific-Instrument Interface Specification (1–22–65)
PC–122	Spacecraft/Scientific-Instrument Interface Specification (4–5–67)
PC–123	Spacecraft/Convolutional Coder Unit Interface Specification (10–25–67)
PC–130	GOE/Convolutional Coder Installation, Integration and Compatibility Specifications
PC–146	Pioneer Space Flight Operations (7–67)
PC–147	Flight Operations Test Plan (7–13–67)
PC–148	Pioneer D Test Plan (8–20–68)
PC–152	Pioneer-C Flight Operations (Sequence of Events)
PC–153	Pioneer D Space Flight Operations —Procedures
PC–154	Pioneer-E Flight Operations
PC–155	Pioneer-C Flight Operations
PC–160	Pioneer C/E Off-Line Data Processing System at ARC (3–15–68)
PC–161	EGSE Computer Programming Specifications for the Scientific Instruments (10–21–66)
PC–162	Simulation Operation Program (7–67)
PC–163	EGSE Computer Program Required for CCU
PC–164	Pioneer VIII and Pioneer D DSIF Operation Computer Program Requirement Specification (6–20–68)
PC–165	Pioneer Space Weather Program

PC–166 Pioneer VIII and IX Operational Computer Program Requirement Specification (11–29–69)
PC–167 Pioneer 9 Operational Decoder (5–69)
PC–168 TPS/Convolutional Coder Modification Interface Specification (5–20–68)
PC–171 Activities at the Air Force Eastern Test Range (5–24–67)
PC–173 Pioneer Spacecraft/DSS-71/GOE Compatibility Test Specification (8–9–68)
PC–174 ETR/Pioneer Compatibility Test Plan (7–12–67)
PC–180 Tests of Scientific Instruments at ARC
PC–181 Scientific Instrument Test Requirements for Systems Tests of Pioneer C/D/E
PC–182 SPAC and POLDPS Checkout Magnetic Recordings (9–27–67)
PC–183 Convolutional Coder Test Requirements
PC–184 Procedure for GOE/CCU Installation and Checkout
PC–186 ETR/Pioneer Compatibility Test Procedure
PC–187 Spacecraft Dolly Proof Load Procedure
PC–188 Pioneer VI and VII DSIF Operational Computer Program Test Procedure (4–1–67)
PC–190 Pioneer Trajectories
PC–191 Pioneer D Trajectory Characteristics
PC–192 Pioneer E Trajectory
PC–193 GOE/Maintenance and Configuration Control Specification (8–25–67)
PC–194 Pioneer-C RF Equipment and Trajectory Information
PC–195 CCU Description and GOE Modifications
PC–196 Pioneer-D RF Equipment and Trajectory Information
PC–197 Pioneer-E RF Equipment and Trajectory Information (4–18–69)
PT–1 Pioneer Trajectory Group Computer Report I, Coordinate Transformation Programs (8–16–65)

Reliability and Quality Assurance[1]

RELIABILITY CONTROLS used as a function of program phasing are shown in figure A–1 with a time line at the top indicating the approximate number of months on each phase through the first launch and the controls applied during each phase. What is considered to be the more significant of these controls will be described in the following three figures.

Figure A–2 describes the organization designed around the major controls with specific tasks listed. A particular individual within the reliability area was assigned to each of these disciplines. In the case of design reviews this individual was required to schedule, define package content, prepare and distribute minutes, see that action items were accomplished on schedule, and provide the best reviewers possible. In the area of specifications, the responsibility was for preparation of environmental specifications and signoff on all other specifications. Of utmost importance should be that the callouts in the specifications are realistic, not too tight, but adequate to account for drift.

For manufacturing and test surveillance two individuals were assigned. These individuals covered floor problems and attended all problem area meetings held by the functional activity. They were responsible for monitoring changes that affected their area and had an input as to the disposition of material that failed. The test surveillance monitor reviewed the test procedures and reviewed all test setups prior to the start of testing.

In the area of failure reporting, instruction was given as to the use of forms and pickup points where the forms were to be deposited. A failure review board was established which was responsible for failed part analysis and for appropriate and timely fixes. In addition to the involved specialists at TRW, this board had as members the system engineer and reliability engineer from NASA/ARC.

Reliability design tradeoffs were made throughout the initial phase of the program. Emphasis was given in particular to the use of proven designs

[1] Because of the great importance of long spacecraft life in the exploration of deep space, this appendix is devoted to a detailed description of the Reliability and Quality Assurance Program employed by NASA and the spacecraft contractor, TRW Systems. This text is extracted from the paper "Interplanetary Pioneer Success Story," prepared for the IEEE WINCON 70 meeting, by T. M. Lough (TRW Systems), J. Mulkern (NASA Ames), and B. Roseman (TRW Systems). These individuals had important roles in formulating the program and it is appropriate here to use their own words in describing it.

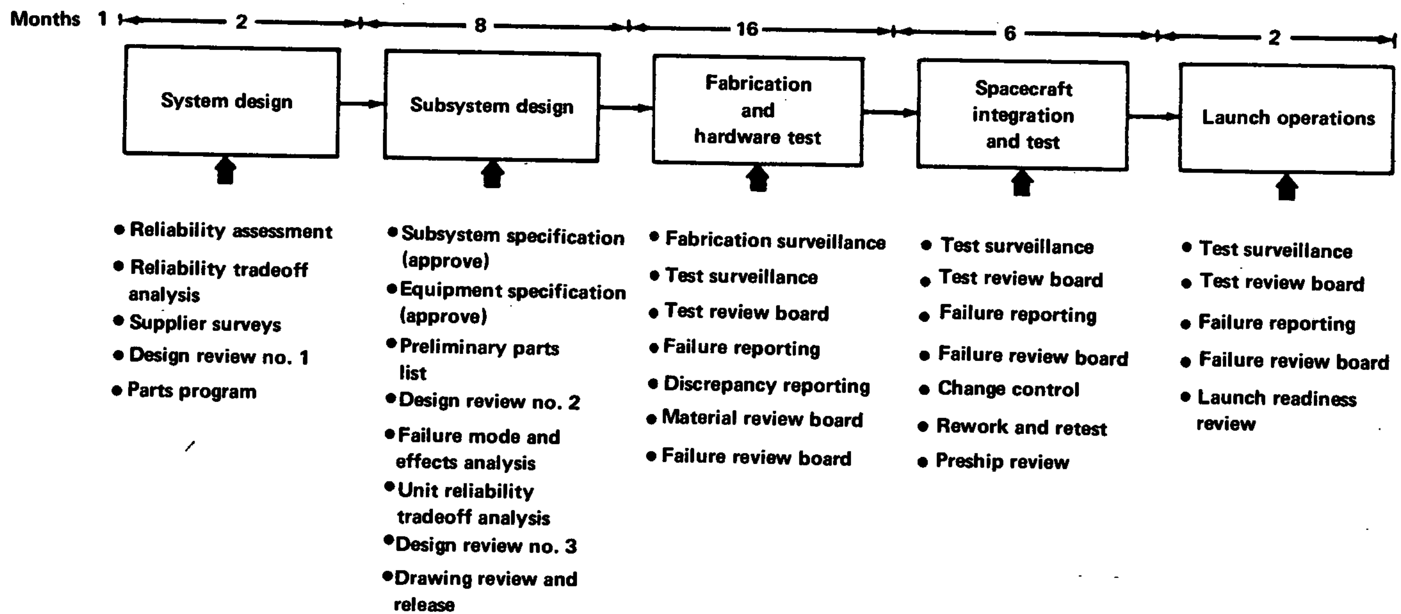

FIGURE A–1.—Pioneer A–E reliability controls.

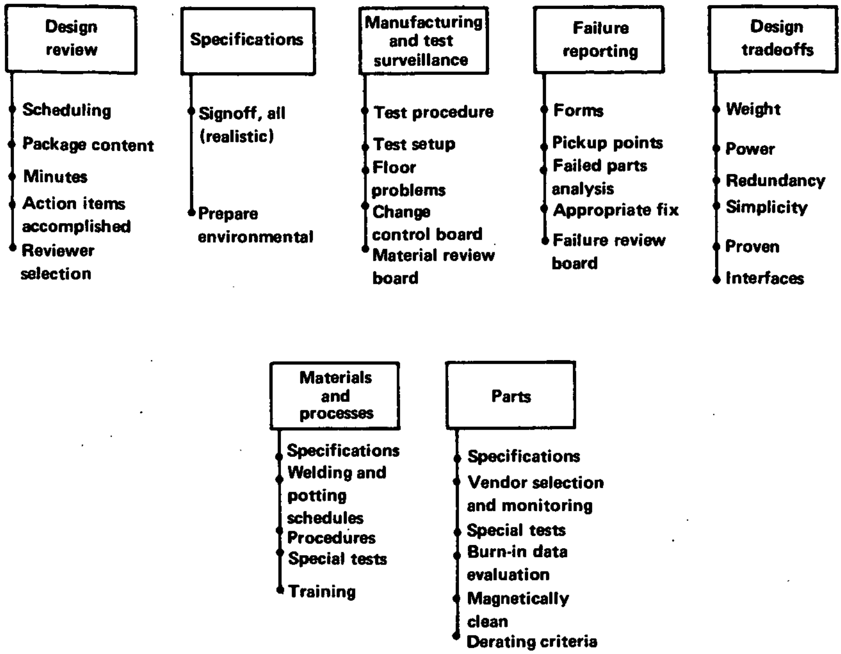

FIGURE A–2.—Reliability organization and functions.

and knowledge of interface parameters. Types of redundancy were evaluated, with consideration continually being given to weight and power.

Materials and processes personnel provided specifications and welding and potting schedules. They were continually involved in special testing and training when problem areas occurred.

Parts discipline has one of the greatest impacts on reliability and therefore this area was given considerable emphasis. The area was managed by a parts engineer assigned to the reliability program. He was responsible for the parts list and deviations to the parts list, high reliability part specifications, special tests and in-house receiving tests, evaluation of burn-in data, and for obtaining magnetically clean parts that were reliable.

Table A–1 describes the significant reliability elements associated with the design phase of the program. Part types used were limited. That is, the designer had to select his parts from a prepared list. This allowed for a better selection of high quality vendors and made vendor control simpler. This program started when the Minuteman program was in full swing and approximately $20 million had been spent with parts vendors on a high reliability parts program. Consequently all our part specifications were written new and were based on Minuteman criteria and wherever possible certified Minuteman vendors were used. Reliability engineers were assigned

TABLE A–1.—*Significant Reliability Elements* (*Design*)

General
Part specifications based on Minuteman criteria
Reliability engineer assigned to each major design area
Redundancy selected on greatest improvement/lb
Design reviews
Kept small (12 to 25 knowledgeable engineers, including NASA)
Data packages, concise and early
Separate reviews for drawings (producibility)
High reliability parts program
100 percent burn-in (100 to 250 hr)
Parameter drift screening
100 percent environmental and life test sampling
No new types, processes or production lines
Part types limited (Parts Deviation Board)

TABLE A–2.—*Significant Reliability Elements* (*Manufacturing and Test*)

General
Reliability surveillance during manufacturing and test
Environmental test evaluation criteria established for each equipment
Failure evaluation system
Failure reporting forms simple, drop stations convenient
Cause of failure defined rapidly (dissection, X-ray, etc.)
Determine and implement corrective action
Concurrency by Failure Review Board (TRW and NASA participants)
Test philosophy
Equipment tests—development, life, qualification, acceptance
System tests—development, qualification, integration, acceptance (space simulation)
Maximum test time possible (minimum 1000 hr/system)

to each major design area to give assurance that tradeoffs were performed with reliability in mind. Redundancy was selected primarily on the basis of the greatest improvement per lb. Eighteen lb of redundancy theoretically brought the reliability from approximately 0.6 to 0.9 for 6 months in space. Specification review and approval and responsibility for the environmental test specifications were assigned to reliability engineers.

We learned early in our design review program to keep the number of persons in attendance small, that is, between 12 and 25 persons; to select knowledgeable and outspoken engineers for design evaluation, to have data packages that were concise and were distributed early enough that reviewers had time to digest the material contained. Finally . . . separate reviews for drawings [were needed] since in the general overall design review there was not time to go over drawings in sufficient detail. This particular discipline contributed to seeing that the producibility of the equipment was at an acceptable level.

The high reliability parts program was considered of major importance and contained 100 percent burn-in, that is, burn-in of all parts from 100 to 250 hr depending on the part type. It also included parameter drift screening; i.e., monitoring the drift of various parameters over the interval the part was in burn-in, and 100 percent environmental and life test sampling. No new types of parts were allowed with one exception which will be discussed later. Table A–2 describes the significant reliability elements associated with the manufacturing and test phase of the program. Manufacturing and test surveillance was provided by two reliability engineers. This consisted primarily of assembly and test setup procedure evaluation, evaluation of test procedures to determine that the environmental test criteria for each equipment was correctly assigned to disclose design weaknesses, participation in the disposition of failed material, and control of changes as well as assignment of test requirements following a failure. Preship and prelaunch evaluation of flight scheduled equipment consisted of reviewing each equipment's history carefully before it was assigned to flight status. In the area of failure evaluation, failure report forms were kept simple and drop stations were located convenient to the manufacturing and test stations. Failure definition was determined expeditiously (within 2 days normally) by means of dissection, x-ray, etc. These results were used to assign fixes and served the Failure Review Board in making their decision.

The Pioneer program test philosophy was very simple. The equipment was made to operate; therefore let it operate as much as practical. A minimum of 1000 operating hours were accrued per system prior to each launch. On occasion at the Cape, when the launch was postponed for launch vehicle reasons, the spacecraft was turned on and left operating in one case for as long as a week. The test program for equipment included development, life, qualification and acceptance testing. System tests included development, qualification, integration and acceptance; the system acceptance tests culminating in a space simulation of 7-day duration.

This describes in brief detail the type of reliability program utilized. Now, what was the result of this program and where could the program have been improved? Figure A–3 shows that design areas were the major problem contributors indicating added emphasis needed to be placed on design reviews. Notice the combination of circuit, packaging and test set design accounts for approximately 50 percent of the total problems. The other major contributors were processes, procedures, and visual aids, which accounted for approximately 35 percent of the major problems.

Figure A–4 shows the areas in which major problems were detected. It is somewhat disconcerting to see the large number of problems, 23 percent, that were not detected until integration test. This would indicate that

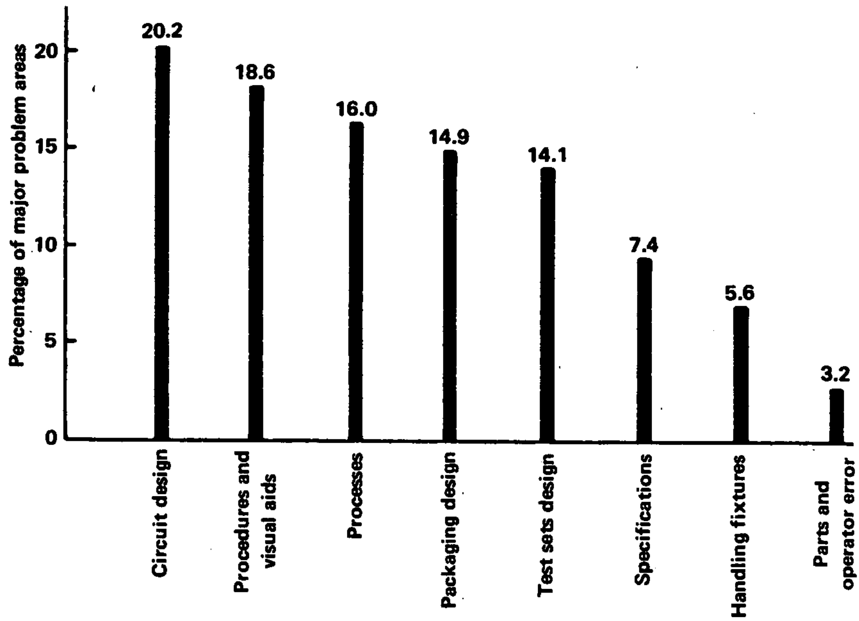

FIGURE A-3.—Causes of major problem areas (prelaunch).

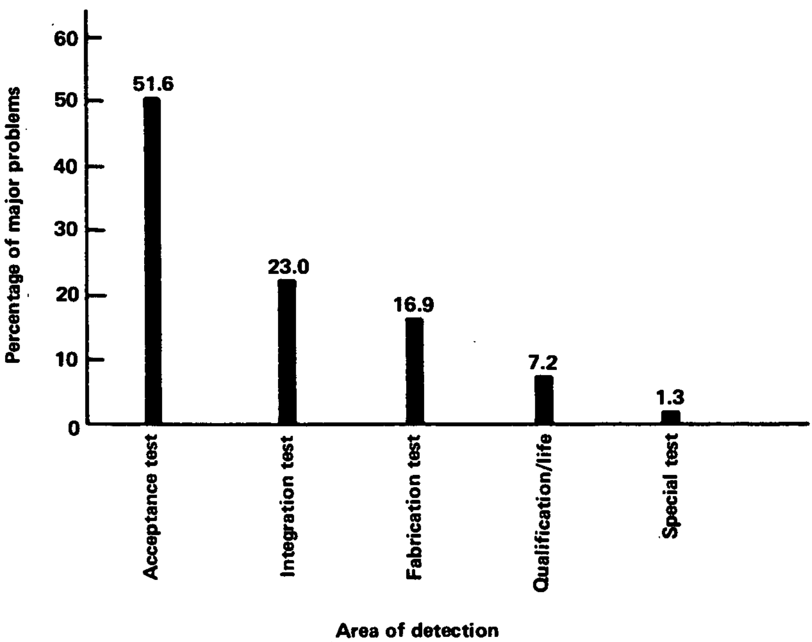

FIGURE A-4.—Percentage of major problems by area detected.

TABLE A–3.—*Post-launch Anomalies*

Six anomalies noted in eleven spacecraft years of operation
One anomaly degraded mission
  Cause—New innovation device (PSCR Sun sensor)
  Threshold sensitive to ultraviolet radiation
  Experiment viewing direction lost on Pioneer 7 (two experiments of little value)
  Fix (ultraviolet filter) included on Pioneer 9
Five anomalies produced no mission degradation because
  Redundancy available by ground command (1)
  Self-heal (2)
  Function not required following orientation (2)

probably the test sets did not satisfactorily simulate interface conditions. Another major fault of the reliability program can therefore be assigned to not including test equipment in the design review program. The final chart, table A–3 in the Pioneer A–E discussion, shows that there were six anomalies noted in 11 spacecraft years of operation. Only one of these failures caused degradation to the mission, and this was because of the use of an unproven device, something that a good reliability engineer should avoid if at all possible. The reason for using this device, a photo silicon control rectifier, as a Sun sensor was to save approximately three-quarters of a lb in weight. The viewing direction during spin has been lost on Pioneer 7 which makes two of the experiments of little value. A study subsequently revealed the threshold of this device to be sensitive to ultraviolet radiation and since a fix could not be implemented until Pioneer 9 it is anticipated that Pioneers 6 and 8 will fail in a like manner. Five other anomalies have been noted, none of which have degraded the mission. It is entirely possible that the self-heal anomalies will reoccur again as failures.

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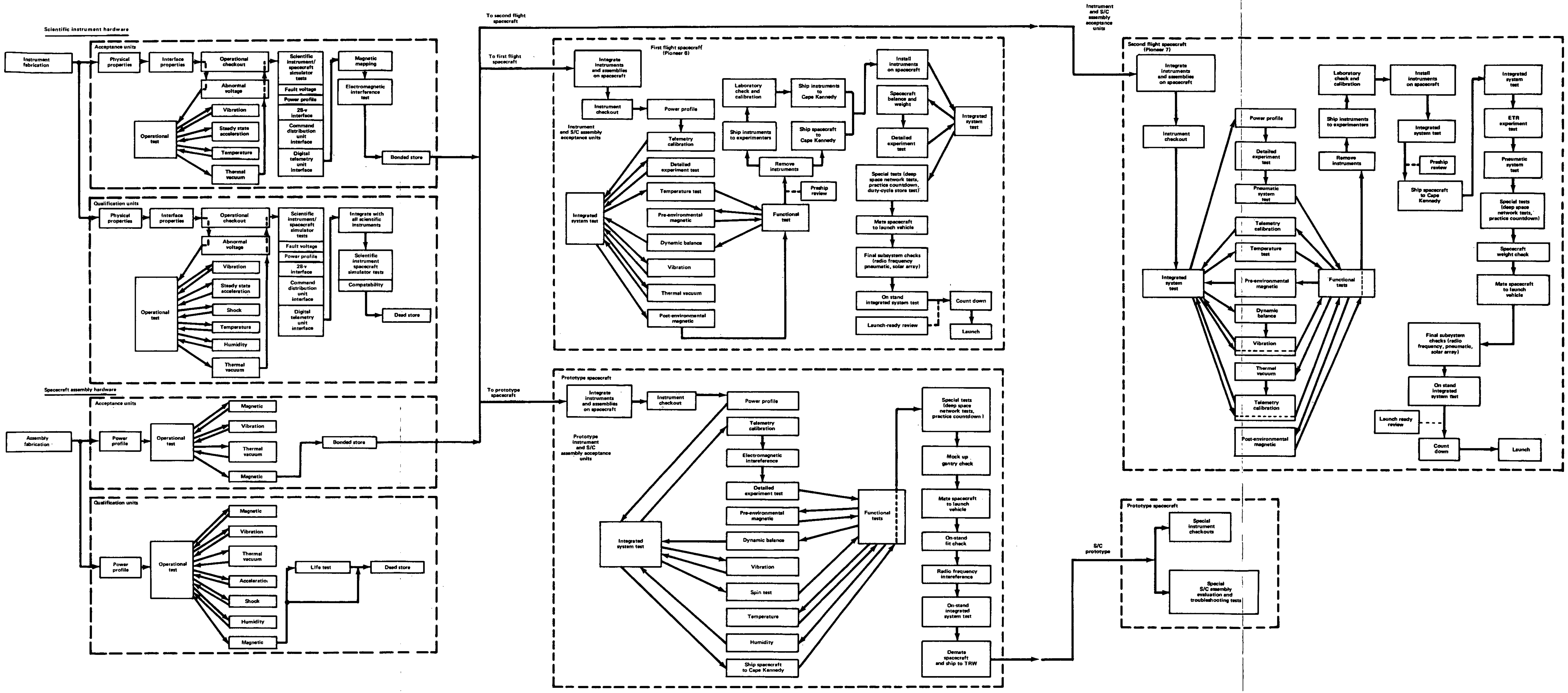

FIGURE 6-1.—Pioneers 6 and 7 test program.

FOLDOUT FRAME 1

FOLDOUT FRAME 2

FOLDOUT FRAME 3