Technical Report No. 32-729

Reliability Considerations in the Design, Assembly, and Testing of the Mariner IV Power System

Kirk M. Dawson
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Kirk M. Dawson
G. Curtis Cleven
Charles D. Fredrickson

G. E. Sweetnam, *Manager,*
Spacecraft Secondary Power Section

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

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The probability of success for the 1964 *Mariner IV* (Mars) and the 1962 *Mariner II* (Venus) spacecraft power systems was almost equal, even though the former had considerably more parts and had to operate 2½ times longer. On *Mariner IV*, cruise science instruments were to be used during the transit to examine interplanetary phenomena, and two additional experiments—TV photography and planet occultation—were to be performed at Mars encounter. Power for the 575-lb craft is supplied from a system consisting of solar panel, battery, energy conversion, and load switching equipment. Operation of this equipment is presented and related to overall system reliability. Criteria for sizing solar panel and battery capacities are given, as well as the present power and energy margins for the *Mariner IV* power system. Particular emphasis is placed on those items that were included in the system design to increase overall reliability. Reliability was obtained by providing redundant functional elements and appropriate failure-sensing and switching circuits; by designing a flexible system that can meet unexpected problems using ground commands and on-board logic; by using parts that have proven reliable in a space environment; by screening these parts and applying them with considerable derating; and by thoroughly testing the resultant product before flight. In addition, an adequate amount of telemetry information was made available to diagnose problems and to aid in their solution. The power system telemetry point selection is reviewed and related to overall system reliability.

Author

I. INTRODUCTION

Providing power for interplanetary spacecraft capable of continuous unattended operation in a hostile environment for 6000 or more hours places unusually stringent requirements on the system design. In such a mission, reliability considerations carry through from the overall system mechanization and the design of each functional element to the selection and screening of individual components. The philosophy is to design the best system possible, construct it with the best parts available, and assemble and test it with all possible care. In addition, the system is designed to be as invulnerable as possible to internal failures and to failures in those systems that form interfaces with the power system.

Two spacecraft were launched from Cape Kennedy, Florida, by the Jet Propulsion Laboratory (JPL) for the Mars 1964 mission: *Mariner III,* which was unable to deploy its solar panels, owing to the failure of its aerodynamic shroud to separate from the spacecraft, and was lost after 8 hr when the battery was depleted, and *Mariner IV,* which was launched on November 28, 1964, on a 325-million-mile, 8-month journey to Mars. Six cruise science instruments were to be used on *Mariner IV* to examine interplanetary phenomena during the transit, and two additional experiments – TV photography and planet occultation – were to be performed at encounter with Mars.

The *Mariner IV* power system performs two major spacecraft functions:

1. It generates standard voltages for distribution to spacecraft power users.
2. It controls the turning on and off of various loads.

The system that was designed to carry out these functions was made up of a power source containing four photovoltaic solar panels with a combined active area of 70 sq ft; a 1200-w-hr silver–zinc battery to provide power during periods when the spacecraft was not Sun-oriented; dc regulating devices; 2400-cps inverters; 400-cps inverters; and battery charging, load switching, and frequency control devices.

Figure 1 shows the spacecraft with its solar panels extended for flight. The central structure is an octagon with eight "bays" for electronic equipment. All power system electronics equipment is contained in two of these bays, and the battery is mounted inboard and directly adjacent to the bay containing the power regulators, bay VIII. During launch the panels are folded up and the solar pressure vanes are folded back beneath the panels. At spacecraft–*Agena* separation, squibs are fired to deploy the panels and extend the solar pressure vanes. Within 30 min after separation, the attitude control system provides Sun orientation for the solar panels. Transfer to solar power operation is accomplished automatically when the panel voltage exceeds that of the battery. Once the Sun is acquired, the battery charger begins to recharge the battery. This continues until the battery is fully charged.

For 16 hr after Sun acquisition, the spacecraft is rolled at 3.55 mrad/sec to calibrate the science magnetometer. At the end of this period the Canopus tracker is turned on and the star Canopus is acquired to provide proper roll orientation. This is necessary in order to point the high-gain radio antenna at Earth when the spacecraft–Earth distance exceeds the capability of the omnidirectional (low-gain) antenna.

Between 2 and 10 days after launch, a midcourse correction is normally required to obtain a proper planet miss distance. This operation involves rotating the spacecraft through pitch and roll turns to point the spacecraft's midcourse correction rocket motor in the proper direction. Depending on the magnitude of the pitch turn, battery power may or may not be required. (In the case of *Mariner IV*, battery power was not used.) After "motor burn," the spacecraft automatically reacquires the Sun and Canopus, and the gyroscopes turn off. After the midcourse maneuver, a cruise mode is started that lasts until planet encounter (assuming a second midcourse correction is not required). Shortly before Mars encounter, the encounter electronics are turned on, and optical sensors mounted on a motorized scan platform lock on and track the planet as 22 television pictures are recorded on magnetic tape. The pictures are played back during the postencounter phase.

As the spacecraft passes behind Mars, its radio signal is cut off from the Earth. By measuring the refraction and attenuation of the signal during the short period in which this takes place, it is hoped to learn something about the Martian atmosphere. Figure 2 shows the spacecraft's flight path.

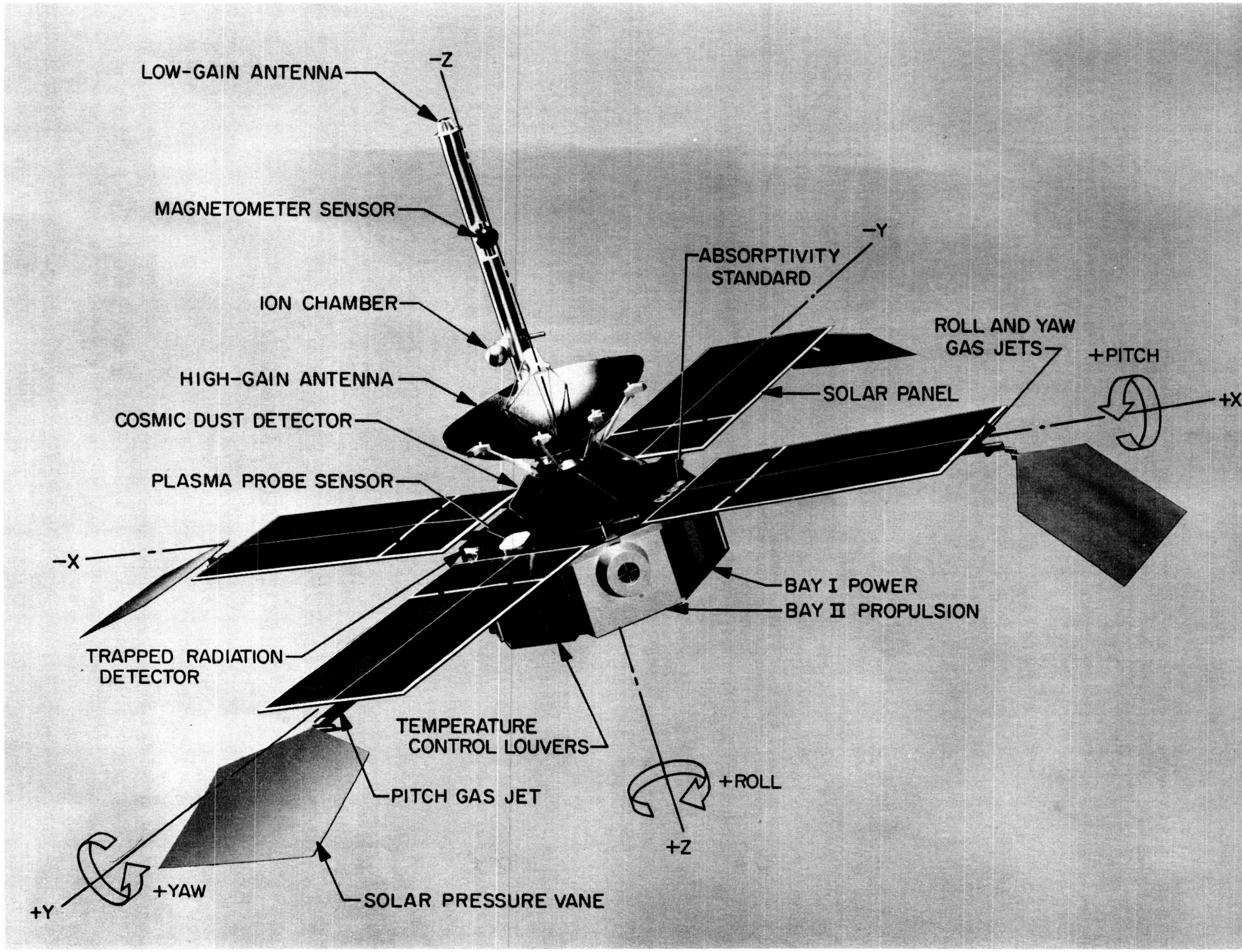

Fig. 1. *Mariner IV* spacecraft, with solar panels extended for flight

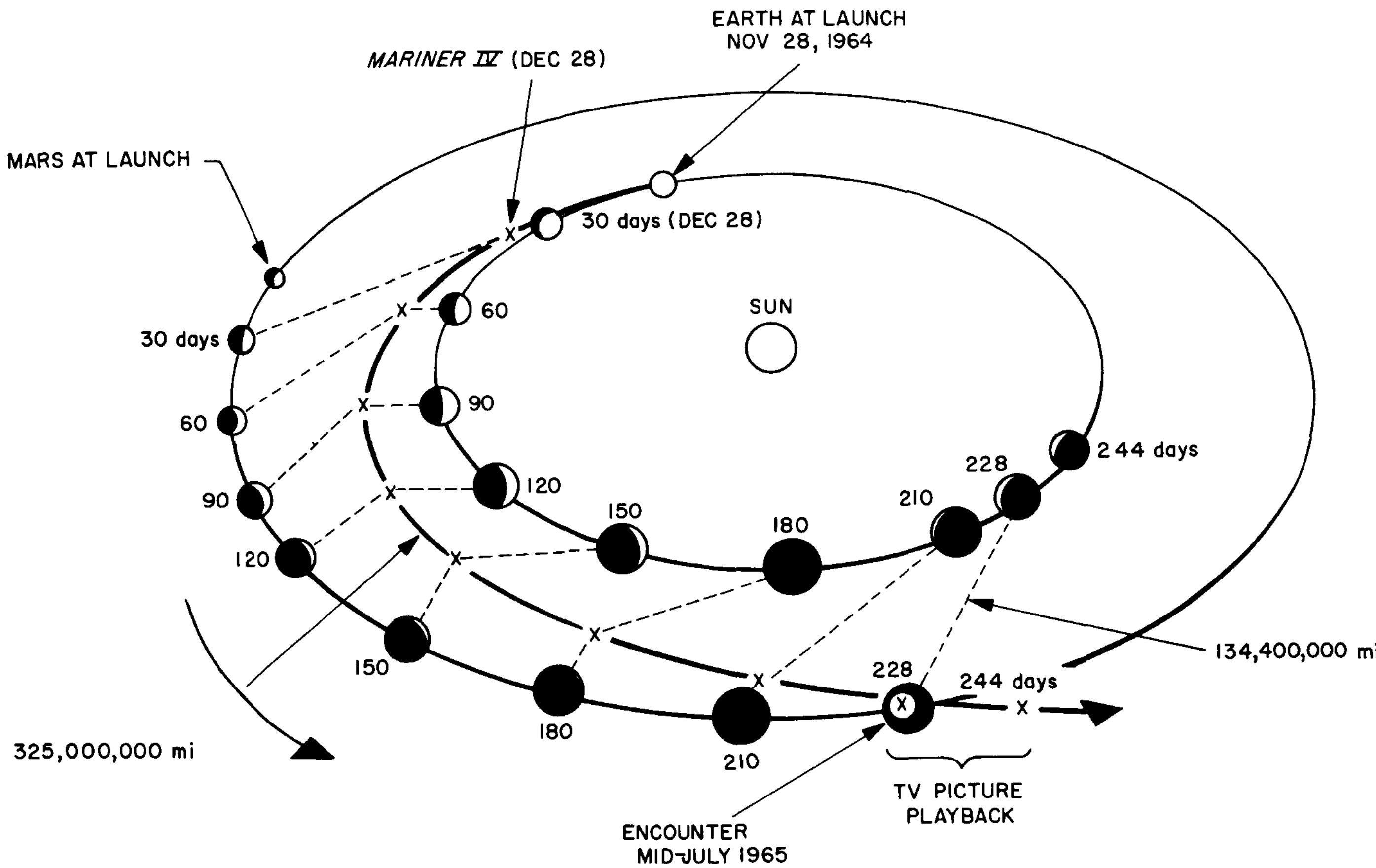

Fig. 2. *Mariner IV* trajectory to Mars

II. SYSTEM MECHANIZATION

A. Energy Source

Figure 3 shows the functional block diagram of the power system. The main source of spacecraft power is from four photovoltaic solar panels with a combined active area of 70 sq ft. Each panel consists of four electrically isolated sections containing 1764 P-on-N silicon solar cells. Electrically, each section is made up of 21 parallel combinations of 84 cells in series. These sections are connected to the unregulated power bus through blocking diodes that prevent reverse current flow if a short should occur in any one section. This ensures that a short circuit in the solar panel cell matrix would reduce the power capability by only 1/16 at most.

In order to remove the battery as a series element in the reliability calculations after the midcourse maneuver, the solar panels were sized to carry all loads required

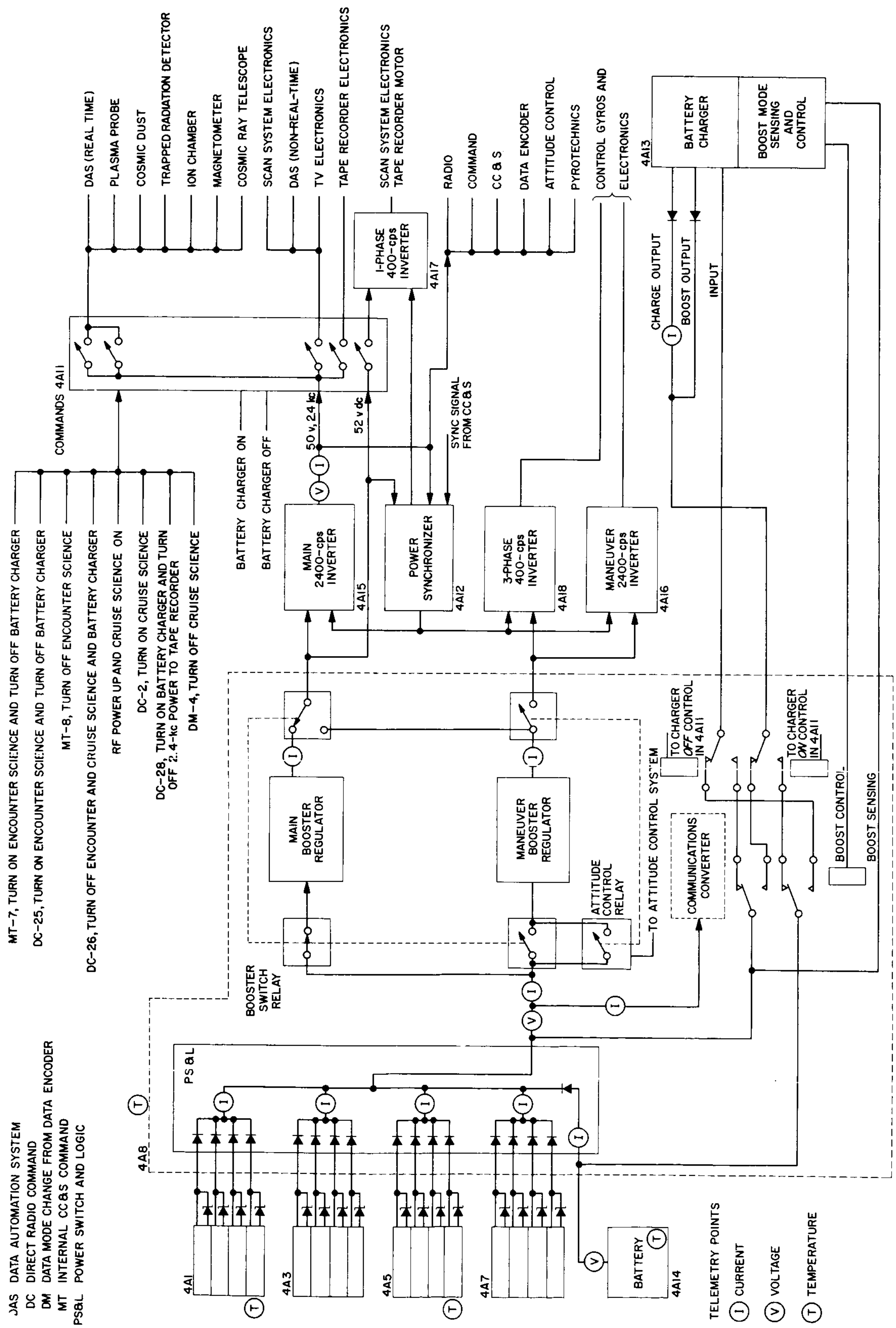

Fig. 3. Functional block diagram of *Mariner IV* power system

after the midcourse maneuver. This significantly increased the theoretical probability of performing the encounter sequence successfully.

Because the launch trajectories allow the spacecraft to spend some time in the Earth's shadow and because there is a large solar panel power capability near Earth, it was necessary to limit the panel output voltage immediately after the spacecraft left the Earth's shadow. Solar panel temperatures as low as $-80°C$ could be expected, and these would lead to voltages as high as 68 v if no limiting were provided. Although regulation equipment could have been designed to handle the high voltage peak, it could have been done only by allowing a substantial decrease in regulator efficiency and a reduction in component derating factors. To remedy the problem, the voltage output of each section was limited to 50 v by a string of six 50-w zener diodes.

Figure 4 shows the solar panel power vs voltage (at steady-state temperatures) for three environments: (1) near Earth, (2) at 100 days after launch, and (3) at Mars encounter. Also shown is the Mars encounter load.

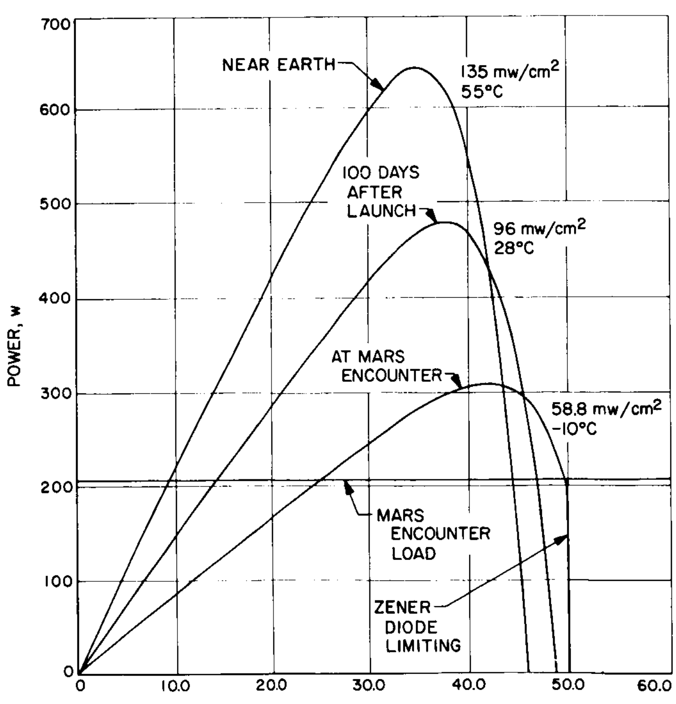

Fig. 4. Nominal solar panel power vs voltage at steady-state temperatures for three space environments

During the early design phase, when the exact solar panel output power, the exact power system loads, and the exact conversion equipment efficiencies were not known, the anticipated nominal panel power was reduced by a factor of 20.5% to obtain an "available-power" number that could be used in the design. This percentage was broken down as follows:

1. Five percent for operation off the solar panel maximum-power point.
2. Seven percent for solar panel design and test tolerances.
3. Eight and one-half percent for efficiency and load uncertainties.

The substantial power margins shown in Fig. 4 are due primarily to increased solar cell efficiency resulting from better control (less cell degradation) during the panel fabrication than was initially predicted.

B. Energy Storage

During periods when the spacecraft is not Sun oriented, spacecraft power is obtained from a silver–zinc 18-cell secondary battery. The battery has a capacity of 1200 w-hr at launch and is capable of withstanding a continuous float charge. The battery forms an interface with the unregulated bus through a blocking diode in such a way that it is disconnected from the bus whenever the solar panel voltage exceeds that of the battery.

Normally, the battery is used to supply power from 7 min before launch until solar panel Sun acquisition. From Sun acquisition until the midcourse maneuver, 2 to 10 days, the battery is partially recharged using the flight charger. After the maneuver, the battery is fully recharged and the charger is turned off. After this time the battery is a redundant energy source and is not normally expected to be used again. For increased reliability, the battery capacity was chosen so that both the launch and maneuver phases could be completed without battery recharging. Figure 5 shows that for the longest possible launch and maneuver periods, a total of 682 w-hr would be removed from the battery. Designing in this manner removed the battery charger as a series element in the reliability calculations.

C. Power Regulation

The voltage-regulating elements of the power subsystem consist of two booster regulators, each capable of operating at power levels up to 150 w. The regulators

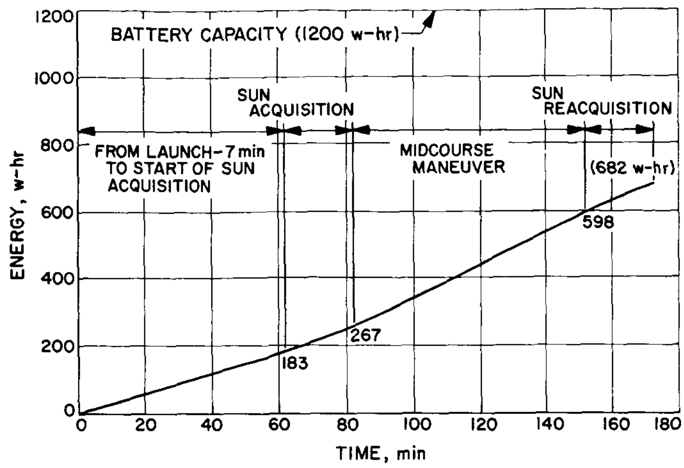

Fig. 5. Worst-case battery drain vs time

accept power from the unregulated bus at voltages ranging from 25 v dc, the lowest battery voltage, to 50 v dc, the highest solar panel voltage, and add sufficient voltage to bring the outputs up to 52 v dc $\pm$ 1%. The main booster regulator is normally on throughout the entire flight, supplying power to all spacecraft loads except the communications converter, which accepts unregulated power directly from the battery or solar panels.

The maneuver booster regulator is used to power a large part of the attitude control system and is on during the launch and midcourse maneuver phases. Turn-on of the maneuver booster regulator is normally controlled by the attitude control system. The main reason for using two regulators is to ensure increased reliability. If a failure should occur in the main regulator that allows its output to leave a 47- to 59-v range for a period of 2 to 3 sec, on-board logic senses a failure, starts the maneuver regulator, and permanently transfers all spacecraft loads to this unit. In order to protect the power users from a failure that would allow the regulator to go to its maximum output voltage—resulting in a 68-v output for 2 to 3 sec—overvoltage protection limits the voltage to 60 v dc. All spacecraft areas can stand this 16% overvoltage for 3 sec.

Choice of the "booster type" regulator, as opposed to the "down regulating" switched regulator, was based primarily on reliability considerations. Even if both regulators should fail, a diode shunt path exists around the regulators that would allow the inverters to run directly from the solar panels. Solar panel output voltage varies from about 43 v near Earth to 50 v at Mars. Thus, a failure of both regulators near Mars would mean a dc output to the inverters of approximately 48 v dc—two diode drops exist between the panels and the dc output—instead of 52 v dc. Tests have shown that even at this reduced output the spacecraft could complete the mission.

D. DC-to-AC Inversion

The main power for spacecraft users is a 100-v (p–p), 2400-cps square wave obtained from a dc-to-ac inverter. Users take this power and, by using transformer-rectifier combinations, obtain the needed dc voltages for their equipment. This method of distributing energy—known as the Edison Company approach—has proved superior to dc distribution on the Mars *Mariner* for two reasons:

1. User voltage requirements vary greatly—especially in the space science area where many instruments require more than 1 kv. Moving these high voltages around the spacecraft in cables is difficult.
2. Greater design flexibility is obtained, since users can change voltage requirements late in a program without affecting the power system or requiring changes in the spacecraft cabling.

Under normal conditions the main inverter receives dc power from the main regulator. An identical inverter receives dc power from the maneuver booster regulator and supplies 100-v (p–p), 2400-cps voltage to the attitude control system. Also running from the maneuver regulator is a 28-v rms, 400-cps, 3-phase inverter that delivers step square-wave power to the gyroscope spin motors. Figure 6 shows the 3-phase waveform. Using this waveform in preference to a sine wave saved approximately 1 lb in inverter weight and had no adverse effect on gyroscope operation. A 400-cps, single-phase, square-wave inverter supplying nominal outputs of 56 and

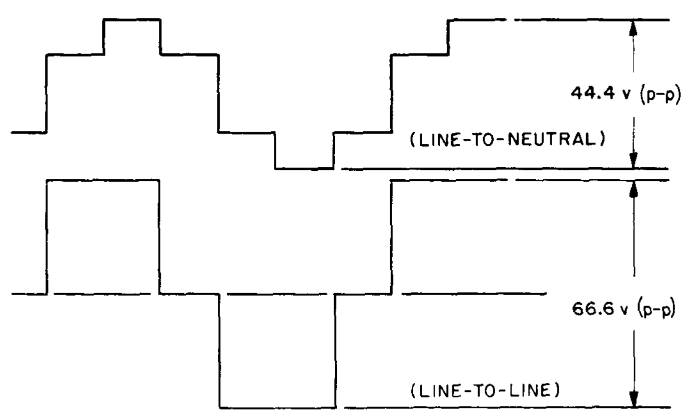

Fig. 6. Voltage waveform of 400-cps, 3-phase inverter with resistive load

65 v (p–p) to the science scan platform and video storage system, respectively, operates from the main regulator. This inverter is off except at Mars encounter.

E. Power Frequency Synchronization

The power synchronizer unit provides a synchronizing signal or a frequency-stable driving voltage for all power subsystem inverters. A 38.4-kc signal received from the Central Computer and Sequencer (CC&S) system is counted down to provide both 2400-cps single-phase and 400-cps single-phase and 3-phase signals. In the 2400-cps and 400-cps single-phase inverters, these signals are used to frequency-synchronize the units and obtain 0.01% stability. The 400-cps, 3-phase inverter is actually a power amplifier driven from the power synchronizer with an accuracy of 0.01%. If the CC&S should fail to produce the 38.4-kc signal, or if a CC&S failure should result in a signal frequency of twice 38.4 kc, a 38.4-kc oscillator internal to the power synchronizer starts, automatically disconnects the CC&S input, and picks up as the frequency source. While the power system is running on its internal oscillator, its frequency is controlled to $\pm 2\%$. If the internal oscillator or the synchronizer countdown chain should fail, the 2400- and 400-cps single-phase inverters self-oscillate within 5% of the desired frequency. All spacecraft systems can operate satisfactorily at frequencies in this worst-case range of $\pm 5\%$.

F. Power Distribution

Some spacecraft systems always receive power whenever the power subsystem is operating; others are turned on and off during various parts of the mission by on-board logic or direct radio command. The actual switching of these loads is done by the power system in the

Table 1. Power system command capability

Control input	Source	Type of signal	Required action by power system	Remarks
1. Encounter start	CC&S MT-7[a]	Isolated circuit closure (permanent)	a. Connect planet science to primary 2.4-kc power source b. Connect cruise science to primary 2.4-kc power source c. Turn on 400-cps, single-phase supply d. Connect tape machine to primary 2.4-kc power source e. Turn off battery charger	Redundant connection
2. Encounter start	C/D DC-25	Isolated circuit closure (pulse $>$ 100 ms)	Same as No. 1. above	Backup for No. 1
3. Encounter terminate	CC&S MT-8	Isolated circuit closure (permanent)	a. Disconnect planet science 2.4-kc power source b. Disconnect redundant cruise science 2.4-kc power source c. Turn off 400-cps, single-phase supply	
4. All science experiments and battery charger off	C/D DC-26	Isolated circuit closure (pulse $>$ 100 ms)	a. Same as No. 3. above b. Same as No. 3. above c. Same as No. 3. above d. Disconnect main cruise science 2.4-kc power source e. Turn off battery charger	Backup for No. 3, plus functions as noted
5. Transmitter power up, cruise science on	Spacecraft separation connector	Series interruptions on one isolated circuit	a. Connect cruise science to primary 2.4-kc power source b. Provide RF power-up signal by opening normally closed relay contact (irreversible in flight)	
6. Cruise science on	C/D DC-2	Isolated circuit closure (pulse $>$ 100 ms)	a. Connect cruise science to primary 2.4-kc power source	
7. Battery charger on	C/D DC-28	Isolated circuit closure (pulse $>$ 100 ms)	a. Turn on battery charger b. Turn off tape electronics	
8. Change to data mode 4	Data encoder	Isolated circuit closure	a. Disconnect cruise science from primary 2.4-kc power source	

[a] MT—Internal CC&S command, C/D—Command decoder, DC—Direct radio command.

power distribution assembly. This unit accepts commands from other spacecraft systems and translates the commands into relay closures. Since loads are switched in units (i.e., cruise science experiments and encounter science experiments), central control of the switching, as opposed to individual user switching, proved highly successful. Table 1 shows the power system command flexibility.

In order to prevent random switching of loads, considerable care was taken in the design of the distribution electronics to make them insensitive to noise and transients. Also, all inputs into the assembly, which come from various spacecraft systems, are isolated to prevent interaction in a failure mode.

G. Battery Charging

After solar panel Sun acquisition, the battery is recharged using the flight battery charger. The charger takes power from the unregulated bus and delivers a current-limited, voltage-regulated charge to the battery. After the battery is fully charged, the capability exists to turn off the charger by direct radio command to prolong battery life. If the battery should be used again, as a result of losing Sun acquisition, a radio command capability exists for reapplying the charger.

In addition to its battery-charging function, the charger may also be used to remove the solar panel–battery combination from an unnecessary battery-sharing mode. Such a mode could be entered if a power transient should instantaneously exceed the maximum power capability of the solar panels. If this should happen during the latter part of the mission when the battery charger is turned off in the charge mode, on-board logic senses the condition, shifts the charger into a current-limited, constant-voltage mode of high capacity, connects the input of the charger to the battery and the output to the unregulated power bus, and boosts the system out of the sharing mode. When sharing ceases, the logic returns the unit to its normally *off* position. Boosting is inhibited during periods when battery sharing should normally occur—launch and midcourse maneuver. If unnecessary battery-sharing should occur near Earth when the charger is charging the battery in a normal manner—very unlikely owing to the large power capability of the solar panels in this region—on-board logic senses the sharing condition, stops the battery-charging operation, and awaits a ground command. A direct radio command can then be sent to initiate the boost mode. This implies, obviously, that the functions of battery charging and boosting are mutually exclusive, since they are performed using the same electronics.

H. Engineering Telemetry

Proper selection of engineering telemetry points is an important part of the power system design. Figure 3 shows the location of 18 of the 22 power system telemetry measurements. Not included are the three solar panel standard-cell measurements and the bay I temperature transducers (all of the electronics except the two dc regulators and the battery). Table 2 lists all 22 telemetry measurements. Measurements are taken at different

Table 2. Power system telemetry measurements

Channel	Measurement	Range	Deck rate
109	PS&L[a] output voltage, v dc	23 to 53	High
203	Dual booster regulator input current, amp	0 to 10	Medium
204	PS&L current to communication, amp	0 to 5	
205	Main booster regulator output current, amp	0 to 5	
206	Battery voltage, v dc	23 to 40	
207	Main 2.4-kc output voltage, v dc	40 to 60	
216	Battery charge current, amp	0 to 1	
221	Maneuver booster regulator output, amp	0 to 5	
222	Solar panel 4A1 current, amp	0 to 5	
223	Solar panel 4A5 current, amp	0 to 5	
224	Solar panel 4A3 current, amp	0 to 5	
225	Solar panel 4A7 current, amp	0 to 5	
226	Battery drain current, amp	0 to 10	
227	Main 2.4-kc output current, amp	0.5 to 2.5	
401	Bay I temperature, °F	25 to 175	Low
407	Bay VIII temperature, °F	25 to 175	
409	Solar panel 4A1 temperature, °F	−40 to 160	
415	Standard cell current, ma	0 to 100	
416	Radiation resistant cell current, ma	0 to 100	
417	Standard cell voltage, mv	0 to 100	
428	Battery temperature, °F	25 to 150	
429	Solar panel 4A5 temperature, °F	−40 to 160	

[a]Power switch and logic.

Table 3. Data transmission rates for power system telemetry

Deck rate	Time between data samples, sec: 33⅓ bps	Time between data samples, sec: 8⅓ bps
	Data mode 1	
High	4.20	16.8
Medium	42.0	168
Low	840	3360
	Data mode 2	
High	12.6	50.4
Medium	126	504
Low	2520	10,080

sample rates depending on the location in the sampling sequence and the spacecraft data mode. Table 3 shows the rates of transmission of the engineering data. With the exception of the solar panel standard-cell measurements, all measurements are self-explanatory.

Selection of the points to be monitored was originally based on the following criteria:

1. The currents flowing into, and out of, all functional elements should be known.

2. The temperatures of all functional elements should be known.

3. Input and output voltages of all elements should be measured.

4. Using power system loading, sufficient information should be available to verify the proper operation of other spacecraft systems.
5. The failure of any one transducer should have a minimum effect on the knowledge about the system.

Unfortunately, a sufficient number of telemetry channels was not available to meet all the criteria. In general, since the system was voltage-regulated, currents were considered more indicative of proper performance than voltages. Voltage measurements were limited to the system input and output and to the battery. In many cases, placement of the current monitors was such that if a transducer should fail, its monitored parameter could be inferred from the other operating monitors.

I. Solar Panel Standard-Cell Transducers

In order to more accurately determine the current–voltage characteristics of the solar panels in flight, the outputs of three "standard" solar cells are telemetered. The open-circuit voltage and short-circuit current of two cells, identical with those used in the solar arrays, form the bases of the predictions. These readings are compared with readings taken on the Earth, and the open-circuit voltage and short-circuit current of the four-panel combination are determined. These readings in turn lead to determination of solar panel maximum power capability and spacecraft power margins. The third cell is insensitive to radiation; that is, it is a standard P-on-N silicon solar cell that has been degraded by radiation to a point where further radiation has little effect on cell output. By comparing its short-circuit output in flight with the short-circuit output of the other standard cell, it is hoped to determine whether radiation is degrading the solar panels.

III. DESIGN, MANUFACTURING, TESTING, AND QUALITY ASSURANCE PROCESSES

The reliability considerations described below were applied to the design, manufacture, and testing of the Mars *Mariner* spacecraft. Spacecraft reliability was achieved through a cooperative effort by the *Mariner* Project Office and the Quality Assurance and Reliability Office of the Jet Propulsion Laboratory and the quality assurance organizations of the vendors who supplied components and subsystems.

The process of designing, manufacturing, testing, and providing quality assurance for the system can be represented by the flow diagram shown in Fig. 7. Obviously, each step in the diagram is important to the overall program of providing flight-qualified hardware.

A. Reliability Analysis

In the initial design phases of the *Mariner IV* mission, it was shown that the power system mechanization used on the *Mariner II*, which encountered Venus in December, 1962, would be inadequate for the longer mission to

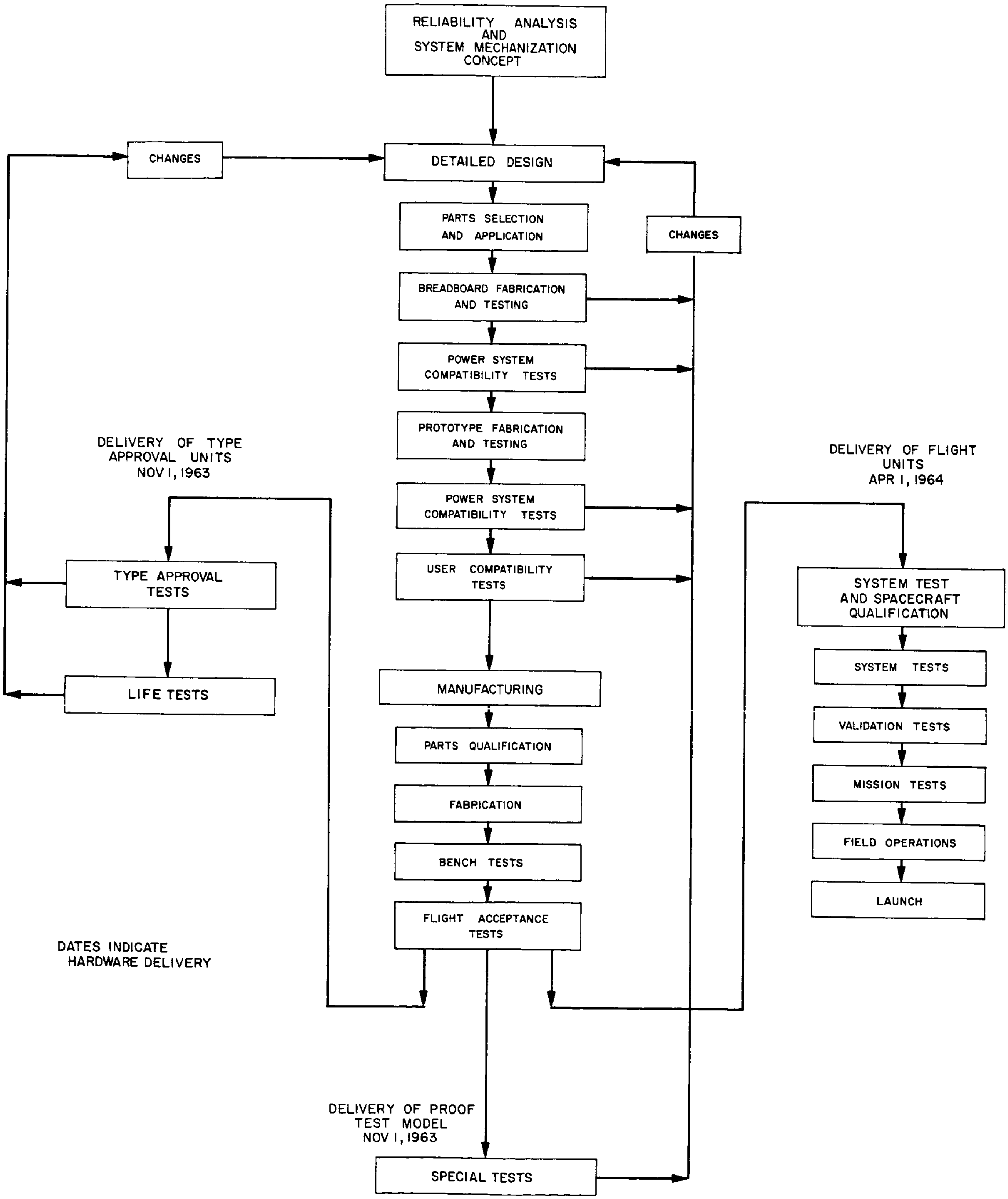

Fig. 7. Flow diagram for design, manufacturing, testing, and quality assurance processes

Mars. The calculated reliability of the *Mariner II* power system was 0.72 (Ref. 1), based on a mission time of 2600 hr. The calculated probability of success of this system for the 6000-hr Mars mission would be 0.46. To get a higher calculated reliability for the *Mariner IV* mission, the various functional blocks of the power system were analyzed for their calculated reliability. The booster regulator, with its 76 parts, had a calculated failure rate of 0.016 failures per thousand hours and was a large contributor to the system's unreliability. Because of the booster's critical nature it was, as mentioned earlier, made redundant.

The largest source of unreliability in the power system was the power synchronizer, with its 219 parts. All but 12 of these parts can be charged against the 400-cycle, 3-phase inverter, as the 2.4-kc inverters have the capability of self-oscillating. Rather than use one phase of the 3-phase inverter to drive the tape recorder at encounter, a condition that would have made the mission dependent upon the power synchronizer for the full 6000 hr, a 400-cycle, single-phase inverter with a self-oscillating capability was used. This made the power system dependent on the synchronizer only through the midcourse maneuver, which, on a normal mission, should occur within 240 hr after launch.

Another design mechanization that was used solely to increase reliability was the fusing of the noncritical telemetry circuits. One fuse in the power regulator assembly removed the 106 parts associated with telemetry as a source of unreliability.

By using reliability approaches such as those discussed above, coupled with the criteria for sizing the solar panels and battery, it was possible to raise the reliability of the *Mariner IV* power system considerably over that possible with the *Mariner II* system. The power system reliability, as computed for several points along the mission profile (Ref. 1), is shown in Table 4.

Table 4. Power system reliability

Event	Time, hr	Estimated reliability
First maneuver	44	0.99
Second maneuver	288	0.96
	750	0.95
	2,600	0.87
	4,500	0.79
Encounter	6,000	0.72
End playback	6,213	0.71

As noted above, the estimated reliability of *Mariner II* was 0.72. Thus, the estimated probability of success for the *Mariner IV* power system, 0.71, is about equal to that for *Mariner II*, although the *Mariner IV* reliability is based on a greater parts count and on a mission lifetime almost 2½ times that of the *Mariner II* flight.

B. Detailed Design

After the functional requirements of each power system element had been determined, detailed design was begun.

Functional element reliability, the process of designing reliable inverters, etc., involves, to a large extent, parts selection and application. The power conversion equipment—excluding solar panels and the battery—contains 913 electrical components, and failures in some areas would obviously lead to termination of the mission. In order to minimize the possibility of such a failure, parts selection and application were closely controlled, and, whenever possible, component selection was made from the Jet Propulsion Laboratory's Preferred Parts List (JPL-PPL). This list contains part types that have been rigorously tested and found suitable for space applications. In many cases JPL High-Reliability parts were used. These are parts that are manufactured to strict JPL specifications from raw material to finished product. All power subsystem parts that were not High-Reliability types were screened by the power system manufacturers to JPL screening specifications. Table 5 shows the percentages of High-Reliability, PPL, and non-PPL parts (parts not listed on the JPL Preferred Parts List) in the power system. Included in the 27.9% total of non-PPL parts were such items as specially made chokes and transformers that were not commercially available and hence would not be in the JPL-PPL.

In applying these High-Reliability or screened parts in circuits, the units were severely derated in power, voltage, etc., by the designer to meet JPL standards. A list was compiled for all power subsystem parts showing the stress rating of the individual units vs the actual stress level in the circuit. The stress ratio of these two numbers was a closely watched item.

As part of the functional element design process, breadboard units were fabricated. After the individual breadboards had been tested, they were mated to other power system breadboards to ensure compatibility. The results of these tests were fed back into the detailed design process to improve overall system performance.

Table 5. Status of component parts in conversion subsystem

Unit	High-Reliability parts	PPL[a] parts	Non-PPL parts	Total[b]
Power regulators assembly	92 (26.4%)	144 (41.4%)	112 (32.2%)	348
Power distribution assembly	90 (52.9%)	31 (18.2%)	49 (28.9%)	170
Power synchronizer	88 (40.2%)	74 (33.8%)	57 (26%)	219
Battery charger	24 (27.0%)	44 (49.4%)	21 (23.6%)	89
2400-cps inverters (2)	8 (20.5%)	24 (61.5%)	7 (18%)	39
400-cps, 1-phase inverter	2 (9.1%)	18 (81.8%)	2 (9.1%)	22
400-cps, 3-phase inverter	1 (3.9%)	18 (69.2%)	7 (26.9%)	26
Total	305 (33.4%)	353 (38.7%)	255 (27.9%)	913

[a]Preferred Parts List (see text, Section IIIB).
[b]Total High-Reliability and PPL parts, 658 (72.1%).

Prototype units were manufactured after breadboard testing was completed. This hardware was identical with the flight gear in mechanical layout and electrical characteristics, and its manufacture served both to "debug" the manufacturing process and to provide a flight equivalent test set. The testing of this hardware revealed changes that were required before production of flight hardware.

C. *Manufacturing and Quality Assurance Coverage*

In order to achieve the highest possible reliability for the completed power subsystem, extremely cautious handling methods and techniques had to be exercised in the fabrication and electrical bench-testing. The fabrication was performed in a dust-free, pressurized "clean room," access to which was restricted to the necessary minimum of personnel. Caps and gowns were required of everyone, and gloves were used by those who actually handled the power system while it was being built. All of the special equipment, tooling, materials, and components to be used in the fabrication were cleaned before being brought into the clean room. It was the function of the quality assurance organizations of both JPL and the manufacturers to maintain the standards of cleanliness, as well as to perform the inspection during fabrication, to provide continuous surveillance during bench testing, and to be responsible for the proper handling techniques and movement of the power system through the various steps in the fabrication and test phases. Special handling frames were designed to hold the subassemblies constituting the power system. The frames provided a means of handling the subassemblies without touching the magnesium chassis and also formed part of a dust-tight plastic box used for transporting or storage of a subassembly.

Fabrication, inspection, and test flow plans were made for each type of assembly. The flow plans outlined in detail each step of the manufacturing and test processes. Included in the plans were a description of the operation involved, the number of applicable inspection or test procedures, and, for reference, the JPL drawing or specification. The flow plans and the manufacturing and test procedures were developed together in advance of actual fabrication in order that sufficient time would be allowed to procure flight-quality materials, to avoid scheduling problems, and to minimize peak work loads. The inspection steps were inserted on the flow plan where it was felt that inspection was necessary to ensure quality of manufacturing. Inspection procedures were written for each inspection step and included visual aids and detailed instructions. These procedures informed the inspector that in addition to a normal inspection for workmanship and cleanliness, he should look for such items as component value and location, wire size, color-coding, and proper termination points for wires and components.

A log book was kept for each subassembly by the JPL Quality Assurance and Reliability Office. This log book contains fabrication records, inspection reports, Material Review Board[1] actions, and test data. Included in the

[1]A JPL Materials Review Board reviews all hardware that does not conform to specification and determines the disposition of the hardware.

fabrication records are the serial numbers of every component used in a subassembly and the circuit symbol indicating where the particular component was used. This serial number is the number assigned to each component during the screening process for component parts. The screening data are filed under the serial number for each type of component screened. These records provide excellent traceability for all components used in the actual manufacturing. If a component failed, the component and its screening data were given to the Component Parts Evaluation Group at JPL for analysis.

The electrical bench-testing consisted of a resistance-continuity test prior to the application of power, followed by two identical performance bench tests. One of the two performance tests was done before and one after the application of a polyurethane conformal coating to the circuit boards and components. The conformal coating acts as an insulator and as a seal to prevent contamination. The subassemblies were not removed from the clean room until after the application of this conformal coating. Figure 8 shows the 2400-cps inverter after conformal coating.

D. Flight Acceptance Testing

Each flight power system is extensively tested before delivery to the spacecraft assembly area. This Flight Acceptance (FA) testing for the conversion equipment electronics consisted of the environmental tests shown in Table 6, plus considerable bench testing. Again, similar tests were performed on the flight batteries and solar panels.

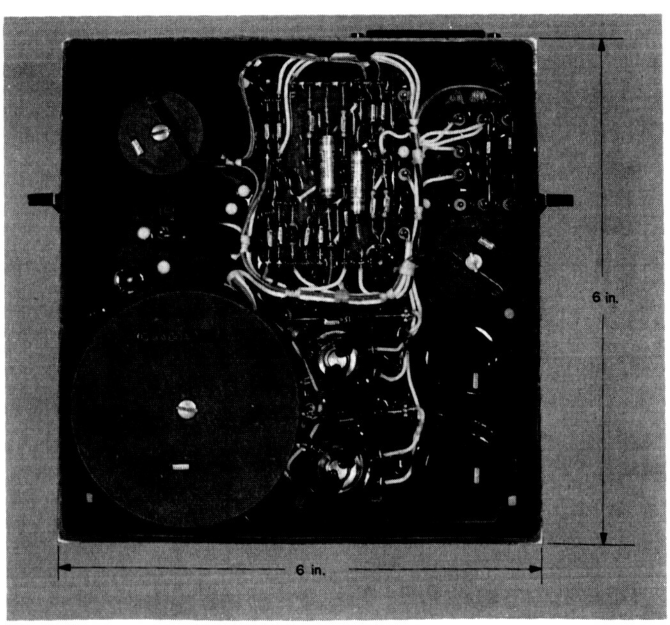

Fig. 8. Main 2.4-kc inverter

Table 6. Environmental flight acceptance tests

Test	Stress level
Vibration (complex wave)	Operation: 6 sec at 9-g rms noise 200 sec at { 3-g rms noise plus 1.5-g rms sine 15–40 cps 6-g rms sine 40–250 cps 3-g rms sine 250–2000 cps }
Vacuum–temperature	Operation: 10^{-4} mm Hg 2 hr at 0°C (+32°F) 40 hr at 55°C (+131°F)

E. Type Approval Testing

One of the first production units of the battery, solar panel, and conversion equipment was designated a Type Approval (TA) set and was used to verify that the designs were both electrically and mechanically stable at stress levels considerably higher than those expected in flight.

Table 7. Environmental type approval tests

Test	Stress level
Handling shock	Free-fall corner drop
Explosive atmosphere	Operation in a fuel–air mixture
Humidity	Operation in 95% relative humidity
Shock	Five 200-g shocks, 0.5- to 1.5-msec pulse, 3 axes
Static acceleration	±14 g, 3 axes, 5 min
Vibration (low frequency)	Operation: ±1.5 in., 1 to 4.4 cps 3 min, 3-g peak from 4.4 to 15 cps
Vibration	Operation: 18 sec at 14-g rms noise 600 sec at { 5-g rms noise plus 2-g rms sine 15–40 cps 9-g rms sine 40–250 cps 4.5-g rms sine 250–2000 cps }
Vacuum–temperature	Operation: 10^{-4} mm Hg 4 hr at −10°C (+14°F) 12 days at +75°C (+167°F)

Table 7 summarizes the TA tests performed on the conversion equipment electronics. Similar tests were performed on the TA batteries and solar panels.

F. Life Testing

The TA batteries and conversion equipment, after having completed their test sequence, were committed to a life test in vacuum at temperatures much higher than expected operating values. For the conversion electronics, the temperatures were cycled between the plus and minus TA temperatures as shown in Table 8. This test was originally designed to locate any latent problems before the launch date, but has been continued past launch to provide further information on the *Mariner* power system design. At the time this Report was written, the equipment had performed flawlessly for over 5500 hr under these severe conditions.

Table 8. Conversion equipment life test

Temperature	Duration, hr[a]
75°C (+167°F)	328
Ambient	160
−10°C (+14°F)	160
75°C (+167°F)	328
etc.	etc.

[a]Eight hours was allowed for transition from one temperature to the next.

G. Testing of the Proof Test Model Spacecraft

One set of flight-qualified hardware was delivered to the Spacecraft Assembly Facility in November 1963, to support the Proof Test Model (PTM) spacecraft, which was used for design verification of the flight spacecraft. This prototype of the final flight spacecraft was subjected to a variety of tests, many at more severe environmental levels than would normally be expected in flight, in order to verify spacecraft designs and performance. In general, when the PTM testing revealed the need for a design change, the change was first incorporated and tested in the PTM before incorporation in the flight spacecraft. As an indication of the extent to which the PTM was used, it had accumulated 1250 hr of test time before launch.

H. Flight Spacecraft Qualification

The flight spacecraft were subjected to a 7-month testing program. This testing, however, was intended not for design verification, as was the PTM testing, but for verification that the equipment operated normally to the design specification. It was thus a flight qualification program.

The tests conducted on the flight spacecraft included the following: subsystem tests to determine the performance of each subsystem while it was operating from spacecraft power; intersystem tests to functionally test all interfaces between subsystems; and systems tests to establish the functional integrity of the complete spacecraft, including redundancy modes. The spacecraft telemetry channels were calibrated and parameter variation tests were conducted. As a climax to the testing at JPL, the spacecraft were subjected to a vibration test and a 250-hr mission test in a vacuum–solar environment. This mission test and a final system test constituted the basis of acceptance. The spacecraft were shipped to the Air Force Eastern Test Range only upon the successful completion of these tests. (A successful system test is defined as one in which no major failures occur.)

The total test time accumulated on *Mariner III* up to launch was 900 hr; that for *Mariner IV* was 860 hr.

IV. CONCLUSION

By applying reliability considerations to system mechanization, detailed design, parts selection, testing, and all other phases of spacecraft development, it was possible to significantly increase the reliability of *Mariner IV* above what would have been possible using the *Mariner II* design.

REFERENCE

1. *Reliability Assessment of the 1964 Mariner Mars Spacecraft,* Document No. PRC R-362, Planning Research Corporation, West Los Angeles, California, July 22, 1963.