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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Technical Report No. 32-882

Mariner Mars 1964 Project Report: Spacecraft Performance and Analysis

JET PROPULSION LABORATORY

CALIFORNIA INSTITUTE OF TECHNOLOGY

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PREFACE

On November 28, 1964, *Mariner IV* was successfully launched on a trajectory that would take it within 150,000 miles of Mars. On December 5, 1964, the spacecraft performed a successful midcourse maneuver, altering its flight path so that it would pass within 6118 miles of the planet. The flight took approximately 7½ months, during which time a great deal of scientific information was gathered concerning the near-Earth environment and interplanetary space. On July 14, 1965, *Mariner IV* photographed the surface of Mars, and telemetered to Earth the most advanced scientific and technical data regarding the planet yet recorded.

This volume of the *Mariner* Mars 1964 Project Report describes the system- and subsystem-level performance of the *Mariner IV* spacecraft from the November 28, 1964 launch through the end of Mission Phase I on October 1, 1965, and includes discussions of operations planning and problem investigations performed in support of the *Mariner IV* mission.

The *Mariner* Mars 1964 mission required the use of a great many new techniques in designing, building, and operating unmanned spacecraft. The success of *Mariner IV* has made these techniques significant, particularly in the light of future space programs. It is hoped that the operations and results documented in this volume will be a useful reference for those planning future space missions.

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Mariner Mars 1964

ELEMENTS OF THE *MARINER* MARS 1964 PROJECT REPORT

The *Mariner* Mars 1964 Project Report consists of the following volumes:

Mariner Mars 1964 Project Report, Mission and Spacecraft Development,Volume I: From Project Inception Through Midcourse Maneuver (TR 32-740, Vol. I)

Mariner Mars 1964 Project Report, Mission and Spacecraft Development, Volume II: Appendixes (TR 32-740, Vol. II)

Mariner Mars 1964 Project Report, Mission Operations (TR 32-881)

Mariner Mars 1964 Project Report, Spacecraft Performance and Analysis (TR 32-882)

Mariner Mars 1964 Project Report, Scientific Experiments (TR 32-883)

Mariner Mars 1964 Project Report, Television Experiment, Part I: Investigators' Report (TR 32-884, Part I)

Mariner Mars 1964 Project Report, Television Experiment, Part II: Picture Element Matrices (TR 32-884, Part II)

Tracking and Data Acquisition Support for *Mariner* Mars 1964, Volume II: Cruise to Post-Encounter Phase (TM 33-239, Vol. II)

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I. INTRODUCTION

This Report describes *Mariner IV* spacecraft performance during Mission Phase 1, November 28, 1964 through October 1, 1965. The Report was prepared by members of the Spacecraft Performance Analysis and Command Group (SPAC).

SPAC was one of three analysis and command groups in the *Mariner IV* space flight organization. SPAC areas of responsibility included: 1) the determination of spacecraft performance by the analysis of telemetry data; 2) the recommendation of command sequences for the correction of nonstandard performance or the optimization of spacecraft performance; 3) the analysis of flight spacecraft problems through suitable ground tests of nonflight spacecraft; and 4) the training of operations personnel for key events in the spacecraft flight sequence through problem simulation using nonflight spacecraft test models.

Two types of information are presented in this document: spacecraft-related, and SPAC-related. Sections II and III deal with spacecraft performance from the system and subsystem points of view. Sections IV and V describe SPAC organization and planning.

Mariner IV performance is reviewed from the standpoint of overall system behavior in Section II. Particular emphasis is placed on the effects of individual subsystem behavior on overall spacecraft performance, e.g., the relationship of attitude control subsystem pitch- and yaw-axis limit cycles on spacecraft radio-received signal strength.

In Section III, spacecraft performance is analyzed on an individual subsystem basis. Each subsystem is described in sufficient detail that subsystem performance data can be easily interpreted. All inflight performance anomalies and failures are described in detail. Finally, recommendations are presented for improvement in spacecraft equipment and space flight operations techniques.

SPAC organization and function is described in Section IV. SPAC line and flight organizations, operations philosophy, and interfaces with other organizations involved in *Mariner IV* flight support are discussed. An evaluation of SPAC performance during the mission presents specific recommendations for increasing SPAC effectiveness.

Section V presents the planning methods used by SPAC in preparation for key flight sequence events such as launch, trajectory-correction maneuver, and encounter with Mars. The various command sequences and spacecraft operating modes proposed during the flight are discussed in detail and the reasons for selection or rejection are presented.

Appendixes include: 1) a detailed chronology of significant events during Mission Phase 1; 2) graphs of engineering data received via telemetry; 3) problems and failures of spacecraft equipment during the mission; 4) detailed descriptions of the mechanization of each of the commands that could be transmitted to the spacecraft; 5) the test plan used by SPAC in preparation for Mars encounter; 6) a chart for the conversion of mission day, day of year, and day of month; and 7) a list of personnel participating in SPAC activities.

II. SPACECRAFT PERFORMANCE

A. *Mariner IV* Description

Mariner IV was one of two spacecraft launched during the *Mariner* Mars 1964 Project. It was one of the *Mariner* Mars 1964 design series developed specifically to provide a Mars flyby mission capability during the 1964 opportunity.

The primary objective of the *Mariner* 1964 mission was to conduct close up scientific observations of the planet Mars and to transmit the results of these observations back to Earth. Secondary objectives were to perform certain field and particle measurements in interplanetary space and in the vicinity of Mars, and to provide experience and knowledge about the performance of the basic engineering equipment of an attitude-stabilized spacecraft during a long duration interplanetary flight farther from the Sun than Earth orbit.

The *Mariner* Mars 1964 design (Fig. 1) represented an extension of the *Mariner R* (1962 Venus flyby) design modified to permit flyby missions to Mars and to accommodate a complement of scientific instruments appropriate to the mission. The Mars flight plan included:

1. Launch by an *Atlas/Agena*
2. Establishment of a cruise mode
3. Performance of a trajectory-correction maneuver early in flight, if necessary
4. Gathering of interplanetary data
5. Start of planetary experiments and the gathering of data about the planet Mars
6. Playback of data acquired during Mars encounter.

Major differences from the *Mariner R* design reflected the longer flight time to Mars, a trajectory away from the Sun, and the steadily decreasing solar energy input to the spacecraft.

The spacecraft, fully stabilized in attitude, used the Sun and the star Canopus as references. Cold-gas jets pointed the spacecraft in all three axes, and external torques were counteracted in two axes by changing the aspect of movable solar pressure vanes to the Sun. Gyroscopes were available for initial Sun and Canopus acquisition and for inertial control during trajectory corrections.

The *Mariner* power subsystem used photovoltaic cells arranged on panels with a body-fixed orientation for cruise operations and a rechargeable battery for launch, trajectory-correction maneuvers, and backup. Power-conversion equipment delivered regulated, 2.4-kc square wave, 400-cps ac, and unregulated dc electricity for distribution to the spacecraft subsystems. A central computer and sequencer (CC&S) provided synchronizing signals for frequency regulation, and performed the sequencing of onboard switching.

The spacecraft guidance subsystem permitted trajectory-correction maneuvers and the postinjection propulsion subsystem (PIPS) was capable of executing two such corrections.

A two-way S-band radio subsystem carried telemetry to Earth, commands to the spacecraft, and angle-tracking, doppler and ranging information for orbit determination. There were two antennas: a low-gain and a fixed, high-gain; either could be used to transmit or receive. Switching between antennas could be performed by onboard

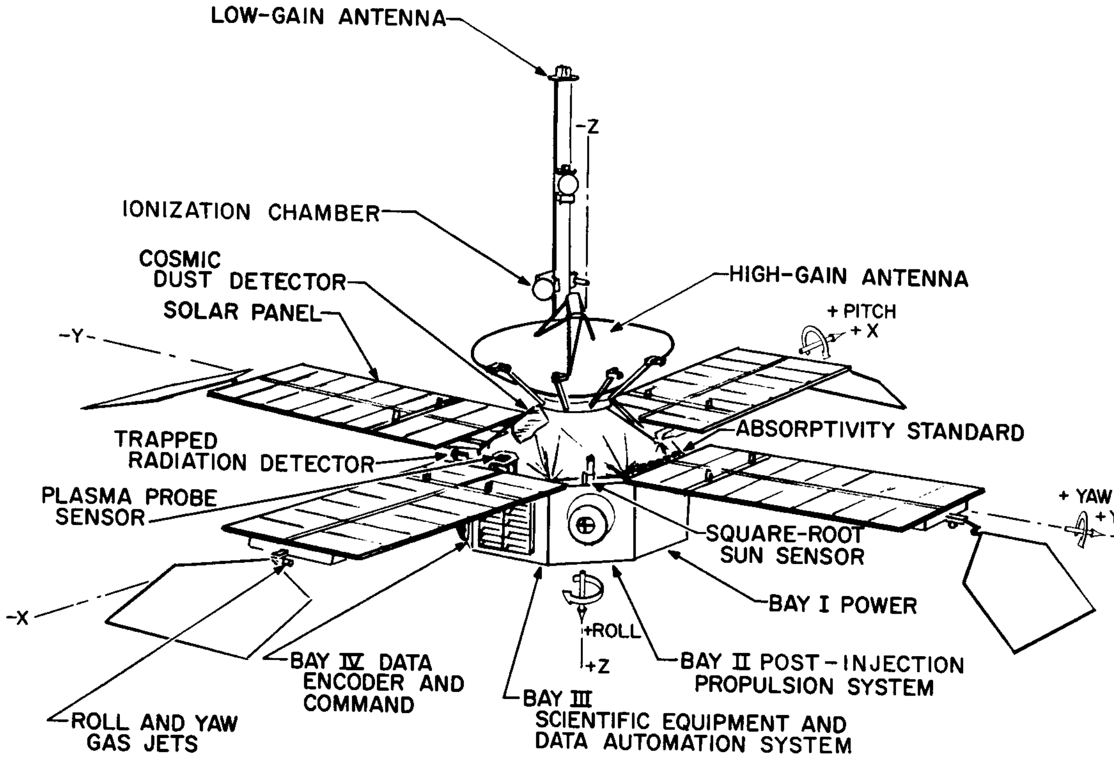


Fig. 1. *Mariner* Mars 1964 spacecraft viewed from above

logic or by ground command. The command subsystem detected and decoded incoming command messages and passed them to the various onboard equipment.

Two types of commands were used: 1) Direct Commands (DCs), which resulted in direct action by the receiving subsystem; and 2) Quantitative Commands (QCs), which were transferred to the CC&S and stored for later use. A data encoder subsystem formatted, sequenced, and when needed, provided analog-to-digital conversion of the telemetry data.

The spacecraft carried a number of scientific instruments to measure fields and particles between Earth and Mars and in the vicinity of Mars; they included:

1. Cosmic ray telescope
2. Cosmic dust detector
3. Trapped radiation detector
4. Ionization chamber
5. Plasma probe
6. Magnetometer

A data automation subsystem (DAS) furnished control and synchronization, performed necessary data conversions and encoding functions, and buffered the science data, transmitting it to the data encoder at the various appropriate rates and times.

For the planetary encounter the spacecraft was equipped with: 1) a television camera, Fig. 2; 2) a scan platform to properly orient the camera; 3) a video storage subsystem; and 4) nonreal time (NRT) data-handling electronics for the television data.

B. *Nominal Flight Sequence*

One of the most significant features of the *Mariner* Mars 1964 spacecraft design was the concept of a spacecraft which, although fully commandable from the ground, could complete its entire mission from launch to end of mission without ground-based intervention or support (except for the trajectory-correction maneuver). This design feature allowed avoidance of complete dependency on the command link but, at the same time, provided the capability to control the mission and alter the nominal flight sequence by ground-command action, if necessary.

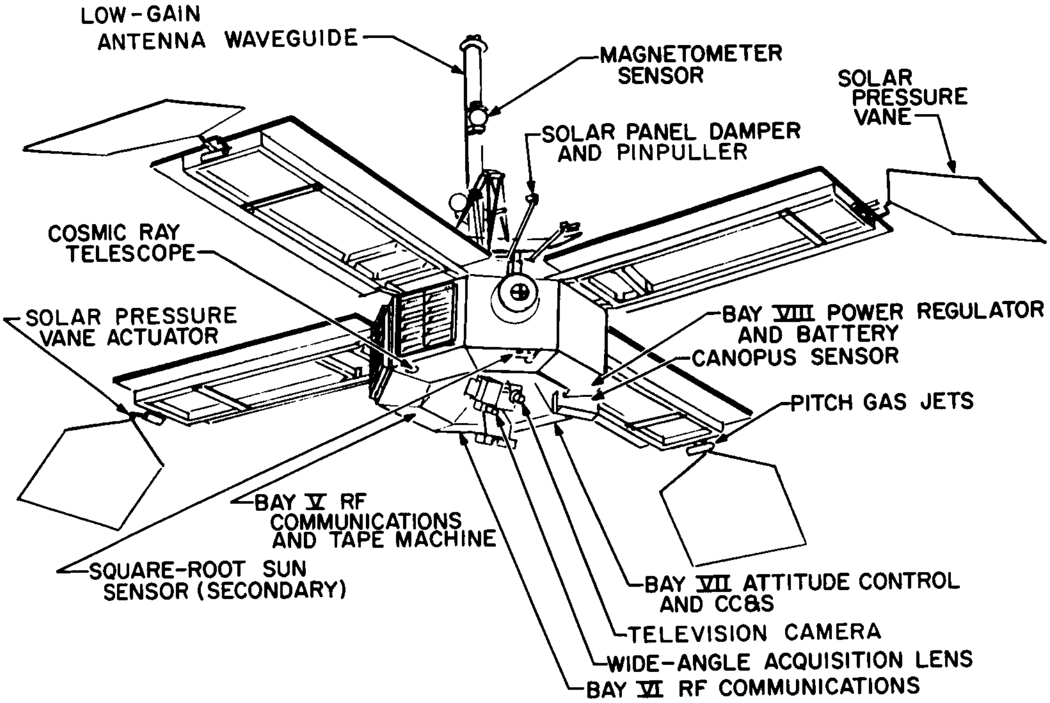


Fig. 2. *Mariner Mars 1964* spacecraft viewed from below

Table 1 provides the nominal flight sequence adopted for *Mariner IV* before its launch. In general, the flight sequence adopted before launch was followed; however, many of the onboard encounter events were preempted by ground command to allow the CC&S- or the sensor-initiated events to serve as backup functions.

C. *Mariner IV* Performance

The *Mariner IV* spacecraft was launched from the Air Force Eastern Test Range (AFETR) at Cape Kennedy, Florida, on November 28, 1964 (the 333d day of the yr); it was the second of two spacecraft scheduled by the National Aeronautics and Space Administration (NASA) for the 1964–65 Mars opportunity. From launch until the nominal end of mission on October 1, 1965, the performance of the spacecraft was excellent. The interplanetary cruise portion of the mission yielded a vast amount of useful scientific and engineering data, and during the planetary encounter phase, all primary mission objectives were met successfully. On October 1, 1965, the spacecraft was still performing well after more than 7300 hr of flight. At this time, the combination of high-gain antenna pointing errors and excessive Earth-probe range exceeded the limits required for obtaining useful telemetry data, so the spacecraft transmitter was commanded to switch from the high-gain antenna to the low-gain antenna. This configuration permitted the Deep Space Network (DSN) to track the spacecraft signals periodically, and should permit the reacquisition of telemetry data if the spacecraft is still operating when the Earth-probe range has again diminished to an acceptable value. This possibility is enhanced by the fact that throughout mission Phase I, all engineering subsystems operated normally, without any discernible degradation in performance.

The nominal flight sequence of events for the *Mariner* spacecraft was designed to be totally automatic, requiring no action from the ground for a normal flight. Such a design relieved the mission from any dependence upon the ground-to-spacecraft communication link (uplink) provided that the CC&S operated correctly. The ground command function would be necessary to the success of the mission only if the CC&S were to fail. This was the cornerstone of the functional redundancy philosophy employed in the Mariner design which provided for independent and physically different means of performing all critical functions. When both means are available, as was the case in the *Mariner IV* mission, a strict reliance upon the onboard functional branch may lead to a flight sequence which is less than optimum. This proved true in a number of cases during the flight, with the result that thc nominal sequence was preempted or modified at those times when it was advantageous to do so. The major departures from the nominal sequence of events are shown in Table 2.

Table 1. Flight sequence adopted before launch

No.	Event	Time	Source	Destination	Comments
1	Verify number of encounter update pulses to be inserted at T − 7 min	T − 15 min	Launch complex equipment (LCE)	CC&S	Number of pulses inserted determines time from launch to CC&S command MT-7, which must vary with trajectory flight time.
2	Switch to internal power	T − 7 min	LCE	Power	
3	Clear counters and insert encounter update pulses	T − 7 min	LCE	CC&S	
4	Release LCE relay hold	T − 5 min	LCE	CC&S	
5	Switch video storage to launch mode	T − 4 min	LCE	Video storage	Monitor data encoder Channel 221 (maneuver booster-regulator output current) to determine if the video storage subsystem is in the launch mode.
6	Release CC&S real-time inhibit	T − 3 min	LCE	CC&S	
7	Clear relay release	Event 6 + 120 sec			
8	Lift-off	T = 0	Event		Provided Event 7 occurs within tolerance relative to Event 6.
9	Spacecraft injection	I (T + 29.2 to 48.0 min)	Event—*Agena D*		
10	Spacecraft separation	S = I + 2.6 min	*Agena D*—timer		
	a. Radio power up and turn on cruise science		Inflight disconnect connector	Power	
	b. Remove relay hold and enable CC&S		Inflight disconnect connector	CC&S	
	c. End video storage subsystem launch mode	S + 0 to 5 min	Inflight disconnect connector	Video storage	This signal is NAND with the first end-of-tape signal after separation and switches the video storage tape recorder motor to 400-cps, single-phase power for the rest of the flight.
	d. Remove plasma 10-kv inhibit		Inflight disconnect connector	Plasma probe	This will allow the plasma probe to supply 10 kv to its sensor.
	e. Arm pyrotechnics		Pyrotechnics arming switch (PAS)	Pyrotechnics	This switch wired in parallel with separation-initiated timer (SIT) for redundancy.
	f. *Agena* isolation amplifier turned off		PAS	Data encoder	
	g. Turn on attitude control subsystem		PAS	Attitude control	Initiate Sun acquisition. Roll spacecraft to calibrate magnetometer.
	h. Separation-initiated timer activated				
11	Arm pyrotechnics	S + 30 ± 20 sec	SIT	Pyrotechnics	See comment 10 e, above.
12	Deploy solar panels and solar pressure vanes. Unlatch scan platform.	Event 11 + 160 ± 80 sec.	SIT		
13	Deploy solar panels and solar pressure vanes. Unlatch scan platform.	T + 53 min	CC&S L-1		Backup to Event 12.
14	Turn on attitude control subsystem	T + 57 min	CC&S L-2	Attitude control	Backup to Event 10 g. Backup: DC-13.

Table 1. Flight sequence adopted before launch (cont'd)

No.	Event	Time	Source	Destination	Comments
15	Sun acquisition complete	Event 12 or 13 + 0 to 20 min			If DC-13 was used to turn on attitude control, Canopus acquisition will begin at Event 15.
16	Turn on solar pressure vanes and Canopus sensor. Initiate roll search about the spacecraft Z axis.	T + 997 min	CC&S L-3	Attitude control	Backup: DC-13. Inhibit magnetometer calibration roll signal.
17	Canopus acquisition complete	Event 16 + 0 to 60 min			(This is at 0.1 deg/sec).
18	Pitch turn duration and polarity	T + 2 to 10 days	QC-1-1	CC&S	
19	Roll turn duration polarity	T + 2 to 10 days	QC-1-2	CC&S	
20	PIPS burn duration	T + 2 to 10 days	QC-1-3	CC&S	
21	Arm first trajectory-correction maneuver	M − 10 min	DC-29	Pyrotechnics	This will ensure that the PIPS transfer relays are in the proper state for first maneuver.
22	Remove propulsion inhibit	M − 5 min	DC-14	Pyrotechnics	This will ensure that the pyrotechnic relays are not inhibited.
23	Initiate maneuver sequence	M	DC-27	CC&S	
24	Maneuver sequence start	M + 0			
	a. Turn on gyro cones for warmup		CC&S M-1	Attitude control	
	b. Switch to Data Mode 1		Attitude control	Data encoder	Backup: DC-1.
25	Begin maneuver	M + 59 to 60 min	CC&S M-2	Attitude control	
	a. Spacecraft to inertial control (all axes). Autopilot on. Canopus sensor off.				
	b. Set turn polarity		CC&S M-3 or $\overline{M\text{-}3}$[a] CC&S M-4		
	c. Start pitch turn		CC&S M-4		
26	a. Stop pitch turn	Event 25 + 1 to 1319 sec (1-sec resolution)	CC&S $\overline{M\text{-}4}$		Magnitude inserted by QC-1-1 prior to maneuver start. (1000 sec equals 180-deg turn).
	b. Reset turn polarity		CC&S $\overline{M\text{-}3}$		
27	a. Set turn polarity	Event 25 + 22 min	CC&S M-3 or $\overline{M\text{-}3}$		
	b. Start roll turn		CC&S M-5		
28	a. Stop roll turn	Event 27 + 1 to 1319 sec (1-sec resolution)	CC&S $\overline{M\text{-}5}$		Magnitude inserted by QC-1-2 prior to maneuver start. (1000 sec equals 180 deg turn).
	b. Reset turn polarity		CC&S $\overline{M\text{-}3}$		
29	Ignite PIPS motor	Event 25 + 44 min	CC&S M-6	Pyrotechnics	Relay pulsed.
30	Stop PIPS motor	Event 28 + 0.080 to 102.360 sec (0.040- to 0.080-sec resolution)	CC&S M-7	Pyrotechnics	Relay pulsed. Magnitude inserted by QC-1-3 prior to maneuver start.

[a]Bar over symbol is convention indicating opposite function.

Table 1. Flight sequence adopted before launch (cont'd)

No.	Event	Time	Source	Destination	Comments
31	a. Commence automatic reacquisition of references	Event 25 + 50 min	CC&S M-1 and CC&S M-2	Attitude control	If CC&S L-3 has failed to operate in Event 15, the Canopus sensor and solar pressure vanes will be turned off by this signal. This condition can be corrected by sending DC-13.
	b. Switch to Data Mode 2		Attitude control	Data encoder	Backup: DC-2.
32	Sun acquisition complete	Event 31 + 0 to 20 min			
33	Canopus acquisition complete	Event 32 + 0 to 60 min			
34	Turn off maneuver counter	Event 25 + 139 min	CC&S	CC&S	Internal function to CC&S which permits subsequent maneuver to be executed.
35	Complete tracking and send trajectory commands for second trajectory-correction maneuver				If a second trajectory-correction maneuver is required.
36	Perform Events 18, 19, and 20				
37	Arm second maneuver	M − 10 min	DC-23	Pyrotechnics	This will ensure that the PIPS transfer relays are in the proper state for second maneuver.
38	Perform Events 22 through 34				
39	Switch bit rate to 8⅓ bits/sec	E − 192 days	CC&S MT-6	Data encoder	Backup: DC-5.
40	Set Canopus sensor cone angle No. 1	E − 137 days	CC&S MT-1	Attitude control	Backup: DC-17.
41	Transmit via high-gain antenna; receive via low-gain antenna.	E − 131 days	CC&S MT-5	Radio	Backup: DC-10.
42	Set Canopus sensor cone angle No. 2	E − 103 days	CC&S MT-2	Attitude control	Backup: DC-17.
43	Set Canopus sensor cone angle No. 3	E − 68 days	CC&S MT-3		Backup: DC-17.
44	Set Canopus sensor cone angle No. 4 Begin encounter sequence	E − 30 days	CC&S MT-4		Backup: DC-17.
45	Turn on encounter science	E − 6⅔ hr (nominal)	CC&S	Power	Backup: DC-25. The Television subsystem (TV), nonreal time (NRT) DAS, scan subsystem, and video storage subsystem receive power and cyclic encounter science sequencing begins. The recording sequence is inhibited. The scan subsystem begins rotating the platform from limit to limit. Actual time of this event will be between E − 6⅔ and E − 10 hr.
	Science instrument cover removed			Pyrotechnics	
	a. Turn off battery charger		Power	Power	
	b. Turn on video storage		Power	Video storage	
46	Wide-angle acquisition (WAA)	E − 4 hr (approx)	Scan	DAS	The scan subsystem acquires the planet and begins tracking.

Table 1. Flight sequence adopted before launch (cont'd)

No.	Event	Time	Source	Destination	Comments
	Switch to Data Mode 3		DAS	Data encoder	Backup: DC-3. After Data Mode 3 is established scan subsystem, TV and DAS engineering data and additional cruise science data are transmitted in real time. No data are recorded at this time.
47	Narrow-angle acquisition (NAA)	E − 80 to E − 35 min	TV and/or narrow angle Mars gate	DAS	Backup: DC-16. Signals from the TV and/or narrow-angle Mars gate (NAMG) indicate when their view is on the planet.
	Inhibit scan platform motion		DAS	Scan	Removes 400-cps power from scan actuator. Emergency Backup: DC-24.
48	Begin recording	Event 47 + 60 to 204 sec			In addition to TV pictures, all real-time data are recorded.
	Start video storage tape recorder		DAS	Video storage	Video storage start delayed until TV is prepared to take picture.
49	End of tape in the video storage	Event 48 + 22.8 min	Video storage	Video storage	Video storage will stop itself and inhibit further start commands from the DAS.
50	Closest approach to Mars	E = 0	Trajectory Video storage	DAS	Indicates when the video storage tape is full.
	a. Switch to Data Mode 2		DAS	Data encoder	Backup: a. DC-2. b. Internal signal at Event 48 + 25.2 min.
	b. Inhibit further start tape commands to video storage		DAS	DAS	The DAS *start tape* commands to video storage are permanently inhibited after the first *end-of-tape signal to DAS* signal which follows the initiation of Picture 18. An internal DAS function inhibits further DAS *start tape* commands after Picture 22 (Event 48 +25.2 min). This is a backup to first part of Event 49 performed in video storage.
51	*Mariner* occulation by Mars	E + 1 to 4 hr	Trajectory		During the last few minutes preceeding the occulation, the radio signals transmitted between the spacecraft and DSIF will refract through the atmosphere of Mars.
52	Radio signal reacquired	Event 51 + 0.5 to 2 hr	Trajectory		Spacecraft emerges from behind Mars.
53	Switch off encounter science	Event 45 + 13½ hr	CC&S MT-8	Power	Backup: DC-26.
54	Begin video storage playback a. Switch to Data Mode 4 b. Switch off cruise science instruments	Event 53 + 6⅔ hr	CC&S MT-9 Data encoder	Data encoder Power	Backup: DC-4.

Table 1. Flight sequence adopted before launch (cont'd)

No.	Event	Time	Source	Destination	Comments
55	Video storage switch from the first track to the second track	Event 54 plus 112 ± 3 hr	End of tape loop in video storage	Video storage	Backup: DC-22.
56	Switch to Data Mode 2		DC-2	Data encoder	Optional after receiving all the recorded data desired.
	Turn on cruise science instruments			Power	

Table 2. Major deviations from the nominal flight sequence

No.	Deviation
1	The spacecraft transmitter was switched via ground command from the triode cavity amplifier used at launch to the traveling-wave tube (TWT) amplifier to avoid transmitter degradation problems inherent in the cavity amplifier.
2	The Canopus sensor brightness gate logic was disabled via ground command to eliminate spurious gate violations thought to result from illuminated dust particles in the sensor field of view.
3	An early encounter sequence was commanded during cruise to deploy the science instrument cover in order to remove dependence upon ground commands at encounter and to position the scan platform to the optimum position as a protection against failure.
4	The planetary encounter sequence was modified by ground command to provide the maximum in backup capability for all critical functions.
5	A second encounter sequence was commanded in order to obtain calibration data for the television electronics.
6	Maneuver-inhibit commands were inserted into the spacecraft after encounter as a means of providing protection against an inadvertent trajectory-correction maneuver.
7	The spacecraft receiver was maintained on the low-gain antenna for the entire mission after it was verified that command capability via the low-gain antenna existed for nearly all of the mission due to an interferometer effect between the two antennas.

Table 3 provides a chronological sequence of events during the *Mariner IV* mission, starting with liftoff and concluding with the transfer of the spacecraft transmitter to the low-gain antenna on October 1, 1965.

Although spacecraft performance was generally excellent, the *Mariner IV* mission was not without incident. Several design deficiencies became apparent, unexpected environmental effects invalidated the automatic reacquisition features in the Canopus sensor logic, and a number of spacecraft performance characteristics were

Table 3. *Mariner IV* sequence of events

Date	Time, GMT	Event
November 28, 1964	14:22:01	Liftoff.
	15:07:09	RF power up, cruise science on, plasma probe high-voltage on, video storage launch mode off, CC&S relay hold off, Agena Channel F telemetry off.
	15:07:10	Pyrotechnics armed, separation-initiated timer (SIT) started, isolation amplifier for Agena Channel F telemetry disabled, attitude control on.
	15:10:10	Solar panels deployed and scan platform unlatched via SIT.
	15:15:00	CC&S L-1 event (deploy solar panels); no action, preempted by SIT.
	15:19:00	CC&S L-2 event (turn on attitude control); no action, preempted by PAS.
	15:30:57	Sun acquisition complete, start of magnetometer calibrate roll.
29	06:59:00	CC&S L-3 event (transfer solar pressure vanes from *erect* to *operate* and turn on Canopus sensor); spacecraft into roll search.
30	11:02:47	Gyroscopes off, spacecraft roll acquired the star Canopus after three roll override commands (DC-21).
December 4	14:35:00	Trajectory-correction maneuver initiated (DC-27).
	14:47:31	DC-13 (inhibit maneuver) transmitted following loss of roll acquisition. Maneuver attempt cancelled. Roll override commands were transmitted later and Canopus reacquired.
5	14:25:00	Trajectory-correction maneuver initiated (DC-27).
	16:09:11	Start of PIPS motor burn.
	16:58:19	Reacquisition of Sun and Canopus complete. One roll override command (DC-21) required.

Table 3. *Mariner IV* sequence of events (cont'd)

Date	Time, GMT	Event
December 6, 1964	00:00	Plasma probe experiment began to return abnormal data.
7	12:29:41	Canopus acquisition lost, spacecraft in roll search until acquisition of star γ Velorum.
13	14:09:00	DC-7 (switch power amplifiers) transmitted. TWT into 90 sec warmup; then standard mode at 40.2 dbm output.
17	16:06:22	Canopus reacquired. One roll override (DC-21) required.
	17:30:00	DC-15 (Canopus gate override) transmitted.
January 3, 1965	16:59:54	CC&S MT-6 (switch bit rates) transferred telemetry rate to 8⅓ bits/sec.
February 11	06:54:43	DC-25 (encounter science on) transmitted. Spacecraft encounter science on, science instrument cover deployed, scan platform started, battery charger off, boost mode enabled. Subsequent commands turned encounter science off and reconfigured the spacecraft to cruise with science instrument cover deployed, battery charger off, and boost mode enabled.
February 27	17:02:19	CC&S MT-1 (Canopus sensor cone-angle update) switched the sensor cone angle from 100.2 deg to 95.7 deg.
March 5	13:12:37	CC&S MT-5 (transfer spacecraft transmitter to high-gain antenna).
April 2	14:25:15	CC&S MT-2 (Canopus sensor cone-angle update) switched the sensor cone angle from 95.7 deg to 91.1 deg.
May 7	17:27:25	CC&S MT-3 (Canopus sensor cone-angle update) switched the sensor cone angle from 91.1 deg to 86.5 deg.
June 14, 1965	15:51:45	CC&S MT-4 (Canopus sensor cone-angle update) switched the sensor cone angle from 86.5 deg to 82 deg.
July 14	14:27:55	DC-25 (encounter science on) transmitted. Encounter science turned on and scan platform started upon command receipt at spacecraft.
↓	15:41:49	CC&S MT-7; no action, preempted by DC-25.
↓	17:10:18	DC-24 (inhibit scan) transmitted. Scan platform stopped with television camera at 178.45-deg clock angle upon receipt at spacecraft.
July 14, 1965	22:10:29	DC-3 (transfer data encoder to Mode 3) transmitted.
	23:42:00	WAA at spacecraft. No action, preempted by DC-24, DC-3.
15	00:17:21	NAA at spacecraft. Television recording sequence started.
↓	00:31:42	DC-26 (all science off) transmitted. All science was switched off upon command receipt at the spacecraft. A subsequent DC-2 command turned cruise science on again.
↓	05:01:49	CC&S MT-8; no action, preempted by DC-26.
↓	11:41:50	CC&S MT-9; cruise science turned off, video storage in playback mode, and data encoder transferred to Mode 4.
August 3	03:20:33	DC-2 (transfer data encoder to Mode 2 and turn on cruise science) transmitted. Previous commands turned off video storage playback power and turned off the battery charger.
26	21:06:52	DC-13 (inhibit maneuver) transmitted. Minimum turn and motor burn duration commands also sent to provide maximum failure protection.
27	19:40:00	DC-17 (Canopus sensor cone-angle update) transmitted. Switched the sensor cone angle from 82 deg to 77.7 deg.
30	20:30:00	DC-25 (encounter science on) transmitted. Encounter science turned on, scan platform started upon receipt of command at spacecraft.
	23:35:26	DC-16 (NAA) transmitted. Started television recording sequence for haze calibration.
31	01:25:00	DC-4 (all science off, transfer data encoder to Mode 4) transmitted. Video storage playback of haze calibration data started.
September 1	06:29:00	DC-2 (cruise science on, transfer data encoder to Mode 2) transmitted. Previous commands turned off video storage playback power and turned off the battery charger.
October 1	21:30:17	DC-12 (switch receiver and transmitter to low-gain antenna) transmitted. End of Mission Phase I.

discovered in flight which had not been observed in prelaunch testing. In addition, failures occurred in two of the scientific instruments carried aboard *Mariner IV*. One of these, traced to the failure of an incorrectly-mounted resistor, caused a partial loss of useful data from the plasma probe. The other, involving the unexplained failure of the Geiger–Müller (GM) tube associated with the ion-chamber experiment, eventually led to the short circuiting of a power supply and the loss of all ion-chamber data. Table 4 presents a summary of the problems and unexpected performance characteristics which were encountered during the mission. A Problem/Failure Report (P/FR) was initiated for each problem that was observed.[1]

Table 4. Spacecraft problems during mission

No.	Problem
1	The solar pressure vanes failed to deploy to the proper positions.
2	An electrical lockout caused one set of solar pressure vanes to fail in the adaptive mode.
3	Periodic violations of the Canopus sensor brightness gates, possibly by illuminated dust particles in the sensor field of view, caused losses of roll acquisition in the standard, automatic roll control mode.
4	A calibration error in prelaunch space-simulator testing resulted in spacecraft temperatures which were generally lower than expected.
5	Failure of a bleeder resistor in the plasma probe experiment caused a partial loss of data for the major part of the mission.
6	Failure of a GM tube induced a power supply failure in the ion chamber experiment, causing the loss of all data from that experiment.
7	A larger than anticipated radio-frequency (RF) interference effect between the high-gain and low-gain antennas was experienced during the later portions of the mission.
8	Transient effects in the operation of the attitude control gas valves resulted in minimum impulses in excess of design values.
9	Large, unexpected variations were present from time to time in the radio TWT helix current.
10	Voltage supply loading changes associated with the antenna switching logic produced spacecraft transmitter frequency changes.
11	Battery voltage exceeded design limits.
12	During the planetary encounter, it was discovered that an unexpected logic state during the recording of television Picture 22 prevented the automatic transfer of the data encoder to Mode 2 upon receipt of the second end-of-tape signal.
13	During postencounter cruise, the cosmic-dust detector instrument began exhibiting anomalous behavior.

These problems and performance characteristics were primarily concerned with individual subsystems, and are described in detail in the subsystem sections of this Report. Many of them during the flight had a pronounced effect upon the systems aspect of the flight operation, and caused some modification to the flight operations philosophy. The following material is presented in an attempt to provide some perspective into the *Mariner IV* mission from an overall systems point of view.

1. Launch-to-Canopus Acquisition

Launch was accomplished on November 28, 1964, at 14:22:01 GMT after a flawless countdown. Both *Atlas* and *Agena* performed well, injecting the spacecraft on a trajectory well within launch tolerances. The Deep Space Instrumentation Facility (DSIF) at Cape Kennedy, Florida (DSIF 71), was able to track the spacecraft until approximately launch plus 7 min. At the nominal time for shroud ejection, the received carrier power from the spacecraft increased by about 15 db, indicating shroud ejection. Shortly afterward, the spacecraft telemetry confirmed that the solar panels were reacting to sunlight. When DSIF 41 (Woomera, Australia) acquired the spacecraft, the data encoder event channels had changed from a reading (octal) of *6 3 1 1* to *0 4 4 3*, indicating that all separation functions had occurred normally; i.e., pyrotechnics was armed, pyrotechnics current flowed in A and B channels, the planetary scan platform was unlatched, all four solar panels were deployed, cruise science was turned on, and the radio power was switched to the high-power mode.

At 15:15:05 GMT, telemetry showed an event counter reading of *1 5 5 3*, indicating that CC&S event L-1 had occurred on time. In addition to the Counter 2 event expected at L-1, events were also observed in Counters 1 and 3. Inasmuch as the number of event-register changes prior to L-1 indicated the proper deployment of the solar panels, it was concluded that the extra events were the result of pyrotechnics current in both channels A and B. This would be possible if the solar panel pinpullers had shorted after they were fired, a distinct possibility confirmed both by ground tests and by *Mariner III* flight data (where one extra pyrotechnics current event was observed). The CC&S L-2 event occurred normally, backing up the already completed *Agena* separation function of turning on the attitude

[1]Also see Appendix C.

control Sun sensors and acquisition logic. Sun acquisition was completed at 15:31:04 GMT, and the spacecraft went into a programmed roll to furnish calibration data for the magnetometer.

The first data samples of the solar pressure vane position telemetry after solar panel deployment indicated that all four vanes had been deployed successfully, but that all four vanes had traveled so far past the nominal angle that they were much nearer than expected to the plane of the solar panels. Despite the overshoot upon deployment, the center of pressure of the spacecraft remained behind the center of gravity. Thus the effect of the solar pressure vanes upon the control of spacecraft attitude was stable, rather than regeneratively unstable. While such an effect was of second order, the tendency was to conserve attitude control gas rather than to increase its expenditure.

At about 1659 GMT the first of a number of data encoder position skips and deck resets in the telemetry commutation cycle was observed. Abnormal indications from the plasma probe and the cosmic-dust detector coincided with the skips. The plasma probe failed to receive some of its normal stepping pulses, and the cosmic-dust detector showed spurious hit indications, both of which are phenomena associated in prelaunch testing with electrical transients on board the spacecraft. The anomalies continued for 23 min, after which the spacecraft returned to normal operation. Subsequent analysis of the spacecraft telemetry data showed no evidence of any performance degradation of the data encoder, the plasma probe, or the cosmic-dust detector which could be attributable to, or be the cause of this transient behavior. Although an acceptable hypothesis had been advanced, the phenomenon remains unexplained because verification was not possible. A second period of commutator position skipping, deck resets, anomalous plasma probe readings and cosmic-dust detector data occurred on December 6, 1964, when the plasma probe began to show a degradation in performance. This period also indicated symptoms of abnormal electrical transients on board the spacecraft, but, beyond this similarity, there was no correlation between the two events. It has since been established that the second period of transient activity was the direct result of a gradual piece-part failure in the plasma probe, and that once the part had completely failed, no further effect was observable in the remainder of the spacecraft.

On November 29, 1964, at 06:59:03 GMT, the CC&S L-3 event was observed in the data. The Canopus sensor telemetry confirmed that it was powered, and the spacecraft went into a normal roll search. Before launch a standard sequence of events during Canopus acquisition had been formulated, because it was anticipated that star identification might pose a serious problem, and because the celestial geometry near launch made it probable that the first star acquired would not be Canopus. The sequence adopted involved allowing the spacecraft to acquire any object that fulfilled the Canopus sensor brightness logic intensity requirements and to become roll stabilized to that star. All data which might provide evidence as to the roll orientation of the spacecraft would then be gathered and evaluated. Based on this evaluation, a recommendation for any command action would be formulated, and then implemented during the next DSIF 11 (Pioneer site, Goldstone, Calif) pass.

An acquirable object entered the Canopus sensor field of view after approximately 60.5 deg of roll search, was acquired, and the gyros were turned off by the acquisition logic. A review of the telemetry immediately indicated that the star acquired was not Canopus. Although tentatively identified as the star Markab, later analysis showed that the sensor actually had made a false acquisition on Earthlight reflected into the sensor optics. Review of the telemetered light intensity data from the sensor showed that the background intensity due to Earthlight was much higher than anticipated, leading to the possibility that a relatively dim star, normally much below the sensor acquisition level might, in fact, ride high enough above the background to become acquirable. As a result, the recommendation for command action was postponed until early in the next DSIF 11 pass when: 1) a more complete analysis of the data would be available, and 2) increasing Earth range reduced the Earthlight background.

Later, on November 29, 1964, at 13:12:57 GMT, data indicated that the gyros had turned on and the spacecraft was in roll search in its automatic reacquisition mode. At approximately 1326 GMT, an acquirable object entered the Canopus sensor field of view and the gyros were automatically turned off at 13:29:15 GMT. Telemetry indicated that the star acquired was approximately one-quarter of the expected Canopus brightness. A review of all data gathered from the L-3 event to this acquisition showed that the star intensity map obtained during the automatic reacquisition did not correlate with that expected for a search beginning at Markab, and that the roll-axis control during the first acquisition appeared much noisier than during the second. These facts led to the hypothesis that the first object acquired was a cluster of stars whose brightness was augmented by the high background light from the Earth, and that the spacecraft had actually been drifting in roll in response to a very noisy input illumination to the sensor.

At this point it was decided to proceed with the Canopus acquisition via ground command from DSIF 11. The first roll override command (DC-21) was transmitted to the spacecraft and verified in the telemetry at 09:14:13 GMT on November 30, 1964. The gyros turned on and the spacecraft went into a normal roll search. After nearly 60 deg of roll, another star was acquired. An examination of the star intensity map for this search showed an almost absolute correlation with the *a priori* map of the sector between the star Regulus and the star Naos. With this first positive indication of the spacecraft roll orientation, the decision was made to continue with Canopus acquisition. A second DC-21 moved the roll reference to the star γ Velorum. The final DC-21 resulted in Canopus acquisition and the initiation of the pretrajectory correction cruise phase of the mission.

2. Pretrajectory-Correction Cruise

For the most part, this portion of the early flight was without incident, except for two roll-control transients, one of which caused a momentary loss of Canopus acquisition. The first transient occurred on November 30, at 1341 GMT, when the telemetry indicated an abrupt change in the roll error signal followed by a rapid excursion from one side of the roll deadband to the other. Some effect in the pitch and yaw axes was also noted. Examination of the telemetry data yielded no correlation with any spacecraft function or event and the transient was attributed to either noise within the attitude control subsystem or to some causal agency external to the spacecraft. The second roll transient, which occurred on December 2, 1964, at 10:09:00 GMT, was the first one which demonstrated all of the characteristics which later became associated with a specific type of repeated roll transient. Canopus brightness had been consistently indicating 100% of the expected Canopus intensity. Suddenly, for one sample, the brightness went to 130% of expected intensity, and, in the same frame of data, a roll error signal significantly less than the error corresponding to the lower edge of the roll deadband was observed. In the next engineering data frame, 12.6 sec later, Canopus brightness had dropped to 50% of the expected intensity, the roll error signal was larger than the high edge of the roll deadband, and the gyros had turned on. Reacquisition occurred almost immediately, and the gyros turned off after a total on time of 207.5 $\pm$ 3.4 sec. Analysis of the data indicated that all of the roll-position errors observed could not be valid, inasmuch as the spacecraft dynamic response would not be fast enough to provide either the initial large negative error or the subsequent recovery. The problem, at that time, remained unexplained, although all indications were that the spacecraft had tracked a very bright object which had passed through the field of view of the sensor.

As the spacecraft temperatures began to stabilize after Canopus acquisition, it began to appear that the stabilization point would be from 6°F to 10°F colder than the nominal temperatures predicted for the near-Earth portion of the flight. However, spacecraft temperatures were within the preflight estimated tolerances. A review of the thermal control activities prior to launch indicated that a possible source of error in the final determination of spacecraft thermal properties was the calibration of the solar simulator during *Mariner* tests in the 25-ft space simulator. Investigation showed that the calibration was incorrect, but that the error was linear in nature, so that while the percentage error would remain constant with decreasing solar input to the spacecraft as it approached Mars, the absolute error would decrease, causing actual temperatures to converge with the prelaunch predictions. As predicted, spacecraft thermal control remained very good throughout the flight.

Midway through November 30, 1964, the Earth detector output became unsaturated somewhat earlier than expected. Because of the additional confidence in the determination of roll orientation afforded by the Earth detector, it was recommended that the trajectory-correction maneuver be performed as early as possible, all other things being equal. A thorough evaluation of the information at hand indicated that the optimum date for the first trajectory correction attempt would be on December 4, 1964 during the DSIF 11 pass, with December 5 and 6 as alternates.

3. Trajectory-Correction Maneuver

On December 4, 1964 an attempt to correct the trajectory of the spacecraft was initiated. The maneuver sequence was terminated before the trajectory correction was accomplished because of an unexpected loss of roll attitude shortly after maneuver sequence initiation. On the following day, December 5, another maneuver was attempted and successfully completed.

About 50 sec after the first maneuver sequence was initiated on the spacecraft, the spacecraft dropped lock on Canopus and went into a roll-search mode, searching for another acquirable object. A DC-13 was transmitted to the spacecraft to abort the maneuver because it had been determined that there was insufficient time to reacquire Canopus with successive DC-21s before the CC&S initiated the pitch-turn maneuver. After maneuver abortion, the spacecraft maneuver clock responded properly to the quantitative commands that had been stored in the onboard logic, demonstrating that the affected subsystems would respond properly during the next maneuver attempt.

The second maneuver attempt was performed on December 5, 1964 after a satisfactory sequence of events and procedures, that would prevent a recurrence of roll search during the maneuver, was developed. The plan that was adopted used a spacecraft design feature that allowed the stopping of spacecraft roll search and the reestablishment of the correct roll attitude without rolling the spacecraft through almost 360 deg of arc. The spacecraft roll gyro would be placed in the inertial (rate-integrating) mode via DC-18 to *freeze* its position in inertial space, and then the spacecraft would be backed up with successive DC-18s until the Canopus sensor plane was properly oriented. Roll attitude would then be referenced to Canopus again using a DC-19.

Table 5 lists the DC-18 maneuver sequence available for use during the second midcourse maneuver attempt. The sequence incorporated Space Flight Operations (SFO)-based constraints needed because of the special sequence execution conditions.[2]

Table 5. Tentative DC-18 trajectory-correction maneuver sequence

Time	Event
M − 5 min	DC-27 execution and time of execution is requested. The DC-27 time of execution is to be based on an estimated zero-roll error crossing at 1435 GMT ± 15 min. The preferred slope of the error-signal-vs-time plot is negative, i.e., the direction of roll should be away from the limit at which spacecraft roll attitude was lost during the first maneuver attempt.
M = 0	DC-27 executed by spacecraft.
M + 1 min	Automatically load DC-18 into ground command operational support equipment (OSE).
M + 36.5 min	Latest time that an indicated roll search can be corrected.
M + 37 min	Latest time that DC-18 No. 1 can be transmitted from DSIF 11. If no DC-18's have been transmitted by this time, remove DC-18 from ground command OSE and insert DC-13.
M + 54 min	Latest time for Canopus acquisition/reacquisition confirmation. Load DC-13 into ground command OSE.

The second midcourse maneuver was initiated at 1425 GMT and successfully completed. Canopus acquisition was maintained until the inertial control mode was established at M + 60 min, making the use of DC-18 or DC-13 unnecessary. The roll position error signal and the roll rate were approximately zero when DC-27 was sent, providing optimum initial conditions for the roll channel at the start of the maneuver.

Telemetry during the maneuver sequence verified that all turns were correct and that the PIPS motor burn appeared normal. Some of the pressure data for the motor differed from those obtained in ground tests, but the differences were attributable to the problems inherent in dynamic testing in a simulated space environment. Tracking data indicated that the doppler change at PIPS motor burn was nominal, and subsequent trajectory computations verified that the new aiming point was well within all tolerances. The maneuver moved the trajectory from a 151,000-mi miss distance on the wrong side of the planet to a 6000-mi miss on the side of the planet that satisfied all science and engineering constraints.

The return of the spacecraft to a cruise configuration after the trajectory correction was uneventful. Sun reacquisition required less than 5 min, roll search was initiated normally, and on December 5, 1964 at 16:47:56 GMT, acquisition of the first acquirable star that passed into the Canopus sensor field of view was complete. One roll override command (DC-21) was sufficient to reorient the spacecraft to the star Canopus.

4. Interplanetary Cruise

Following the reestablishment of cruise mode operations after the trajectory correction, the spacecraft entered a phase of the mission (the interplanetary cruise) which lasted, with one brief exception, until the start of the planetary encounter on July 14, 1965. The one exception was the science cover deployment exercise on February 11, 1965. During this exercise, encounter science was cycled via ground command. Included in the final portion of the cover deployment exercise was a command sequence that modified the cruise configuration somewhat, turning off the battery charger and enabling the battery charger boost mode.

[2]The actual sequence performed is listed in the Chronology, Appendix A.

The interplanetary cruise portion of the mission included the execution of six CC&S master timer (MT) commands, the transmission and execution of two ground commands to modify the cruise configuration of the spacecraft, and two failures in the cruise science instruments. The first of these failures resulted in a degradation in the performance of the plasma probe, and the second led to the catastrophic failure of the ion chamber. The two ground commands were transmitted to place the spacecraft in an alternate roll control mode in order to prevent losses of the roll reference and to switch the spacecraft transmitter from the triode cavity power amplifier to the traveling-wave tube (TWT) power amplifier in order to maximize the probability of mission success.

a. Plasma probe failure. At approximately 0000 GMT on December 6, 1964, the first indications of a failure on board the spacecraft were noted. Science data analysis reported that the plasma probe voltage did not appear to be stepping properly. Two hr later a deck skip was noted in the engineering telemetry coincident with plasma probe and cosmic-dust detector data anomalies. For the next 23 hr, data encoder, plasma probe, and cosmic-dust detector anomalies occurred intermittently. Subsequent analysis of plasma probe data showed a steady degradation in quality until the data reached a point where it was undecipherable by conventional means. Throughout the 23-hr period, the engineering data remained normal, with only the abnormalities in the commutation cycle to indicate any spacecraft malfunction. From the time that the plasma probe data reached a level from which there was no further degradation through the end of Phase I of the mission, there were no deck skips or resets observed in the data which were not correlated with onboard switching functions. All indications were that an isolated piece-part failure in the plasma probe resulted in sporadic generation of radio-frequency (RF) noise for the duration of the propagation of the failure, and that when the part had failed completely, the removal of the noise source precluded any further interaction with the rest of the spacecraft. Further analysis confirmed the failure mechanism, and led also to the development of a data-processing technique which allowed the recovery of a significant portion of the plasma data by the experimenter.

b. Ion chamber failure. After a Class 2 solar flare on February 5, 1965, the ion chamber GM 10311 tube began to count at an excessive rate. None of the other radiation instruments on the spacecraft showed any evidence of abnormal radiation levels; it was felt, therefore, that a failure may have occurred in the tube itself since the ion chamber instrument with which the GM 10311 tube was associated showed no degradation in performance. This hypothesis was substantiated by a duplication of the flight data by the life-test unit after being exposed to radiation levels corresponding to those experienced by *Mariner IV*. A careful review of the spacecraft telemetry data yielded no correlation between the GM 10311 tube performance and any function or condition occurring elsewhere in the spacecraft.

On March 17, 1965, during the telemetry blackout over DSIF 51 (Johannesburg, South Africa), which was supporting the *Ranger* Project, a second, related failure occurred, and when the spacecraft was acquired by DSIF 11 the telemetry data showed that the GM tube counting rate had dropped to zero and the ion chamber was returning no data at all. Analysis indicated that the second failure could be attributed to a failure of the power supply common to both the ion chamber and the GM tube indicated by a short circuit in the tube itself.

c. Power amplifier switching by command. Early in the *Mariner IV* flight, a formal recommendation was made to SPAC by telecommunications personnel to switch the radio to the TWT amplifier via ground command action. This action represented the implementation of a radio flight plan formulated prior to the launch. The recommendation was approved by SPAC, the Space Flight Operations Director (SFOD), and the Project Manager. Command action (DC-7) was initiated on December 13, 1964, and the transfer from the cavity amplifier to the TWT amplifier was successfully completed.

This decision, based upon an evaluation of the characteristics of the two different amplifiers, revolved around the following arguments. The TWT had a predicted lifetime far in excess of the minimum *Mariner* mission requirements and its known failure modes led to immediate, catastrophic failure. The cavity amplifier, on the other hand, was known to show definite aging effects which limited its useful lifetime to approximately the length of the *Mariner* mission. It was necessary to launch using the cavity amplifier because the TWT did not have the low-power mode required for the launch phase, but if the cavity were employed throughout the flight it was probable that aging would degrade its performance to the point where it would not be capable of returning useful data to Earth at the planetary encounter. As a result, operation of both the cavity and the TWT would be required for a successful mission. However, if the TWT were employed from some early point in the mission, it would be capable of serving for the entire mission, barring catastrophic failure. In the event of TWT failure at some later time during the mission, the cavity amplifier would be available, undegraded by excessive use, for the balance of the flight. Thus, the mission could be successfully completed if either the TWT, or a combination of the TWT and cavity were capable of operation. The additional backup capability afforded by the latter mode of operation led to its adoption for the flight.

Another factor that contributed to the decision to transfer to the TWT by ground command was that the spacecraft was designed with the capability to automatically transfer from one power amplifier to the other if the amplifier output were to drop below a fixed point. Since the most likely failure mode of the TWT was catastrophic failure, automatic switching to the cavity amplifier would take place if the TWT failed. However, the most likely failure mode of the cavity was gradual degradation, therefore if command capability were lost, the spacecraft could reach a state in which the cavity amplifier output was so degraded that useful data could not be returned, and yet not degraded enough to cause automatic switching to the TWT.

Following the power amplifier switch via DC-7, the TWT operated as expected with one exception. Beginning on December 22, 1964 and continuing through December 31, 1964 telemetered data indicated that the TWT helix current was varying more than anticipated, and was generally tending to increase. It finally stabilized about 0.4 ma higher than the initial value. Discussions with the manufacturer subsequently established that helix-current variation was a known characteristic of TWTs used in space applications and should be considered normal.

d. Canopus sensor gate inhibit. Before the first trajectory correction attempt, two roll transients occurred, one of which caused the loss of Canopus, although it was subsequently reacquired without incident. When the first attempt failed due to a loss of roll acquisition, there was enough available evidence to question the capability of the spacecraft to remain attitude stabilized in normal roll control. All remaining doubt was removed when on December 7, 1964 at 12:29:41 GMT the spacecraft again lost Canopus acquisition and went into roll search, reacquiring finally on the star γ Velorum. Since roll orientation was critical during the cruise phase only when the spacecraft was transmitting or receiving via the high-gain antenna, it was decided to allow the Canopus sensor to remain acquired to γ Velorum until a plan of action could be formulated.

The plan submitted by SPAC and approved by the Project Manager involved reacquiring Canopus, and then sending a ground command to disable the Canopus sensor brightness gate logic (DC-15). The necessity for command action was emphasized by four roll transients, each of which caused momentary loss of acquisition during the next 10 days. Several models were proposed to explain the transients, but the one which appeared to best fit the data in all cases attributed the transients to dust particles illuminated by the Sun. This theory accounted for the large error signals observed, which could not be explained in terms of normal spacecraft motion, and it delineated the mechanism by which acquisition was lost, i.e., the high brightness of the nearby particles would produce violations of the high-intensity gate logic, thus placing the spacecraft into an automatic override mode. Examination of the telemetry data indicated numerous brightness transients during the early flight, in addition to those severe enough to cause gate violations. The conclusion was that the spacecraft would continue to suffer periodic losses of roll acquisition throughout the mission if it were allowed to continue in the normal, roll control mode.

At that time, since the spacecraft was both transmitting and receiving via the low-gain antenna, the only consequences of the normal roll control mode were the transients injected into the spacecraft when the gyros and the maneuver booster-regulator were turned on and off. Although this was undesirable, in tests on the proof test model (PTM) spacecraft the gyros had been cycled many hundreds of times without any apparent degradation. For a spacecraft transmitting on the high-gain antenna, however, the loss of Canopus acquisition would be accompanied by a loss of downlink communication from the spacecraft. During the period of the mission when a large number of stars was acquirable, this implied a requirement for blind transmission of commands in order to recover both Canopus acquisition and telemetry data. If the command capability were lost altogether, estimates by guidance and control personnel were that the high-gain antenna would be available for less than 10% of the time during cruise, and that the probability of being Canopus-acquired during encounter would be less than 0.10.

Transmission of a DC-15 command to the spacecraft would have removed the Canopus sensor brightness gates which were used for acquisition and prevented the initiation of roll search due to the observed high brightness gate violations. Since the spacecraft response would be slow compared with the speed of the violations, the sensor would remain oriented properly toward Canopus regardless of the transient brightness sensed. The only remaining mechanism for this type of loss of acquisition with the gates removed was that the sensor could follow another object instead of Canopus if the object both were brighter than the star and moved slowly enough through the sensor field of view to allow spacecraft roll response.

The major drawback to the use of DC-15 was that the gyro control unit was controlled through the gate logic, so that with the gates disabled there could be no gyro turn-on except by ground command in the event that Canopus acquisition was lost. The *Mariner* design not only used the gyro turn-on under these conditions for the automatic reacquisition, but also was able to ensure that the receiver was accepting signals from the low-gain antenna so that command capability would exist for all roll attitudes. With DC-15 in effect and the spacecraft receiving via the high-gain antenna, a loss of Canopus acquisition could not be corrected for from 66⅔ hr to 133⅓ hr, the time required for the onboard logic to automatically switch the receiver back to the low-gain antenna, thus restoring command capability. The problem became most severe if the spacecraft were perturbed in such a manner as to cause the loss of both Sun and Canopus acquisition. Without the gyros there was a possibility that Sun reacquisition within the required time would not be possible. The probability that the spacecraft would experience a disturbance large enough to knock it significantly off of the Sun line was estimated to be quite low (about 10^{-4} to 10^{-6}); hence, the risk of this failure mode was accepted.

The DC-15 command action was initiated on December 17, 1964. A single roll override command (DC-21) was sufficient to return the spacecraft roll orientation to the star Canopus. DC-15 was then transmitted, and all indications were normal. Throughout the balance of the mission, although a significant number of roll transients were observed in the telemetry, roll acquisition to the star Canopus was maintained without any problem.

One of the hypotheses offered to explain the roll transients observed early in the flight was that micrometeoroids impinging on the spacecraft either knocked loose dust held to the spacecraft electrostatically or produced spalling, possibly of thermal control paint, and that these secondary particles passing through the view of the Canopus sensor were responsible for the sensor brightness gate violations. The latter portion of the interplanetary cruise, however, indicated no correlation between micrometeoroid activity as reported by the cosmic-dust detector experimenter and the occurrence of roll transients.

Prior to the time for the CC&S to command the switch from transmit via low-gain antenna to transmit via high-gain antenna, the DC-15 decision was reviewed, since the loss of Canopus acquisition in this condition would automatically produce a loss of telemetry. In view of the continuing occurrence of roll transients, it was decided to remain in the DC-15 condition. A more severe problem was presented by the apparent requirement to switch the receiver to high-gain antenna several weeks later. Analysis of the spacecraft RF characteristics in the transmit via high-gain antenna and receive via low-gain antenna mode showed, however, that an RF interference effect between the signals from two antennas into the receiver produced an unexpected ability to maintain command capability from the three prime tracking stations assigned to *Mariner* for all but a relatively brief period throughout the remainder of the mission. This removed the last major disadvantage associated with DC-15, that of not being able to take immediate command action to turn on the gyros in case of a loss of attitude stabilization if the spacecraft were receiving via the high-gain antenna, and led to the decision to maintain a DC-15 condition throughout the entire mission.

e. Interferometer effect. Within a few days of the CC&S command MT-5 antenna switch, sufficient information became available to show that both the spacecraft-received signal strength and the ground-received signal strength were varying in a manner which indicated that some amount of RF interference was being experienced. That this was a potential problem had been recognized long before launch; however, for the nominal characteristics of the communications subsystem it was felt to present a problem only when the Earth was nearly aligned with the boresight of the high-gain, directional antenna and the spacecraft was either transmitting or receiving via the low-gain omniantenna. At the boresight of the high-gain antenna, the increase in antenna gain above that of the low-gain antenna was very nearly equal to the isolation between the two antennas. As a result, RF leakage via the high-gain antenna was approximately the same order of magnitude as the normal signal via the low-gain antenna. The path length from the ground receiving antenna to each of the spacecraft antennas then determined the phase relationship of the two interfering signals, causing either constructive (additive) or destructive (subtractive) interference.

The fact that interference between the spacecraft antennas was present as early in flight as February 22, 1965 on the spacecraft-to-Earth link (downlink) and February 18 on the Earth-to-spacecraft link (uplink) was unexpected and indicated that the exact RF configuration of the spacecraft was not well understood. The signal strengths at all of the DSIF stations began to drop rapidly on February 22 and 23, causing SPAC to review the possibility of recommending an early switch of the transmitter to the high-gain antenna. Analysis by telecommunications personnel of the signal characteristics of *Mariner IV* and of the data gathered in prelaunch antenna tests resulted in a set of predicted values of signal level as a function of spacecraft aspect angle (which was a function of mission time), that showed that the first null in the interference pattern would still yield signal strengths above the absolute telemetry threshold. Thus, although a relatively large number of bit errors might be expected, the telemetry data received would still be of sufficient quality to allow normal spacecraft data analysis by SPAC. It was predicted that the null would last for 3 to 5 days, and then the received signal strength from the spacecraft would begin to increase as the changing relative path length from the two antennas to the Earth reached a point where the interfering signals began to come into phase with each other. The CC&S-controlled switch of the spacecraft to the high-gain antenna was scheduled for just after the time of this predicted peak signal strength. On the basis of this information SPAC recommended to the *Mariner* Project that no command action be taken to effect an early antenna switch.

The interferometer pattern calculated from the signal strengths reported by the DSIF in the succeeding few days matched very closely with the predictions, and, although a bit-error rate as high as 3% was sometimes observed, it proved quite feasible to provide continuous analysis of the spacecraft data until the antennas were switched by the CC&S.

The CC&S-controlled transfer of the spacecraft transmitter to the high-gain antenna (MT-5 event) occurred as expected on March 5 at 13:02:37 GMT. The increase in signal strength at DSIF 42 (Tidbinbilla, Australia) matched the predicted change very closely and all telemetry indications verified a normal transfer. It was noted that coincident with the switch of the transmitter to the high-gain antenna there was a 14-db increase in the spacecraft-received signal strength. Subsequent analysis indicated that the change of state of the circulator switch which controlled the antenna switching also changed the phase relationship of the interfering signals on the Earth-to-spacecraft link, effectively changing position of the interferometer pattern, so that the uplink signal was near a peak rather than a null. An effort was initiated to gather sufficient data after MT-5 to allow remapping of the uplink interferometer pattern.

Originally it had been predicted that command capability via the low-gain antenna could be maintained through April 8, 1965. The predictions based on spacecraft automatic gain control (AGC) readings after MT-5, however, included a very deep null in the interference pattern which would have precluded command action using the DSIF 10-kw transmitters for the period from March 24 through April 1, and from the DSIF 13 (Venus site, Goldstone, California) 100-kw transmitter from March 26 through March 30. The alternative action of allowing or commanding the spacecraft receiver to transfer to the high-gain antenna necessitated a review of the attitude control mode of operation of the spacecraft as well as the telecommunications configuration. With DC-15 in effect, the automatic star reacquisition feature of the *Mariner* Mars 1964 design is not available in the event that Canopus acquisition is lost, so that proper reacquisition of Canopus requires ground command action. Except for the command blackout period during the predicted null, command action would be possible from the Venus site 100-kw transmitter through the postencounter period. The maximum loss of both uplink and downlink communications, then, would be the period between Goldstone tracking passes. If the spacecraft were allowed to transfer to receive via the high-gain antenna, however, a loss of roll orientation could result in a command blackout of up to 133⅓ hr, the maximum time until the spacecraft automatically switched its receiver back to the low-gain in the event of a loss of uplink communications.

After extended discussion, it was decided to configure the spacecraft in such a manner that the transfer of the spacecraft receiver to the high-gain antenna would take place automatically on March 26 at 09:03:54 GMT. In order to accomplish this it was required that the DSIF refrain from establishing uplink RF lock with the spacecraft until two consecutive CC&S cyclic pulses (CY-1) occurred. At the second CY-1, the receiver would transfer from one antenna to the other, and at each subsequent CY-1 would transfer again as long as the uplink RF lock was withheld. CC&S CY-1 No. 42 was allowed to occur on March 23 at 14:24:24 GMT while DSIF 11 was in one-way RF lock (ground transmitter off), and no further two-way RF lock passes (ground transmitter operating and spacecraft phase-locked loop in-lock) were scheduled. This method of effecting the antenna transfer had the additional advantage over ground command that the transfer could be aborted by resuming the normal two-way lock schedule. This advantage was employed when, just before CY-1 No. 43 (which would have caused the transfer) it became apparent that the decrease in signal strength received by the spacecraft was less than predicted. As additional telemetry became available the interference pattern predictions were revised, finally indicating that the null would probably still leave a substantial command margin for even the 10-kw transmitters. As soon as sufficient confidence was generated in this new value, the DSIF schedule was amended to include a two-way pass prior to CY-1 No. 43, thus inhibiting the transfer of the spacecraft receiver to the high-gain antenna.

Analysis of the revised pattern indicated that the error in prediction of the spacecraft received signal strength was due to a difference in the isolation characteristics of the circulator switch that controlled the antenna from the nominal values which had been assumed. Predictions of the command margin throughout the remainder of the mission showed a positive margin for all but a brief period during May and June, when command capability could still be maintained from the DSIF 13 100-kw transmitter. In view of this it was decided to remain in the receive via low-gain antenna configuration throughout the mission, pending a final review as the 10-kw command blackout approached in May.

As expected, the spacecraft-received carrier-power for transmission from 10-kw transmitters at the prime DSIF stations dropped below −139 dbm, the worst-case command threshold on May 3. It rose above −139 dbm again on June 22. During most of this period command capability from the 10-kw transmitters was doubtful and, for some 27 days in the middle of the period, impossible. As a result of an SPAC review of the command situation, it was recommended that the spacecraft receiver remain switched to the low-gain antenna because: 1) few failures could be hypothesized that were correctable by ground command action, but that could not wait until the 100-kw transmitter at DSIF 13 could be used; and 2) a significant risk of losing roll attitude and therefore all command capability, existed in the receive via the high-gain antenna configuration. Throughout the period of doubtful or impossible 10-kw command capability, the 100-kw transmitter at the Goldstone Venus site was kept ready for command transmission on short notice.

Since both the normal encounter sequence plan and any possible emergency procedures involved the command subsystem, it was recommended by SPAC that a series of periodic command-loop lockups be undertaken to:

1. Provide data on the behavior of the command subsystem
2. Verify optimum command-loop lockup procedures
3. Assure the maintenance of a high degree of readiness of DSIF equipment and operators

Command-loop lockups were accomplished from DSIF 13 with the 100-kw transmitter on April 9 and again on April 13, from DSIF 11 and 51 on April 28, and from DSIF 42 on April 30. The second DSIF lockup (April 13) was performed after the 10-kw command blackout had begun. All exercises of the command subsystem were accomplished normally. Throughout the 10-kw command blackout, DSIF 13 continued to perform periodic command-loop lockups with the 100-kw transmitter. On June 28, command-loop lockups from the 10-kw sites were resumed and continued on a regular basis through encounter.

f. CC&S MT events. Six CC&S MT events occurred during the interplanetary cruise phase of the *Mariner IV* mission. Although all of the commands issued by the CC&S were correctly transferred to the appropriate subsystems on the spacecraft, several of them were accompanied by anomalous indications in the telemetry or in the ground-received signal characteristics. However, a thorough analysis of the telemetry data produced no indication of anything other than normal performance by the spacecraft.

The first MT command issued was the bit-rate switch event (MT-6) which occurred on January 3, 1965 at 16:59:54 GMT, as expected. The data encoder responded normally, switching from a rate of 33⅓ bits/sec (bps) to a rate of 8⅓ bps. Coincident with the MT-6 event, DSIF 41 (Woomera, Australia) reported a decrease of about 1 db in the received carrier power as indicated by the station AGC monitor. Throughout the balance of the mission the ground-received carrier power agreed with predicted values well within the calibration tolerances of the AGC monitors, indicating normal performance by the spacecraft. To this time the cause of the apparent decrease at MT-6 remains unexplained.

The first Canopus sensor cone-angle update (MT-1) was issued by the CC&S on February 27 at 17:02:03 GMT. The event appeared normal in every respect so far as the data were concerned, and it was subsequently verified by telemetry that the cone angle had changed from the preset position of 100.2 deg to the MT-1 position of 95.7 deg. Analysis of the data, however, showed that a skip in the data encoder commutation cycle had occurred at the time of the event. The commutator, at low deck positions 308, 419, and 439 was apparently subjected to a transient which resulted in a reset of the synchronizer before receipt of its normal stepping pulse. As a result, the next readout of the low decks was for measurements 301, 401, 421. While it was known that the *Mariner* synchronizers were susceptible to transients, all of the prelaunch test experience indicated that power transients or high-level hardline-carried broad-spectrum noise (e.g., electrical arcing), were necessary to cause deck skips or resets. Either this conclusion was erroneous, or there was some type of power transient or noise burst on the spacecraft coincident with the MT-1 event. If the latter were the case, there was no doubt that there was a malfunction of undetermined nature coincident with MT-1. In this light several possible failure modes were proposed and tested on the proof test model spacecraft, but all test results were negative. At the end of the mission the deck skip remained unexplained.

The second Canopus sensor cone-angle update (MT-2) was observed in the telemetry data on April 2 at 14:25:15 GMT, as expected. Although the update appeared normal and the new cone angle of 91.1 deg was verified in the data, analysis of the tracking reports from the DSIF subsequent to the MT-2 event indicated an apparent decrease in the average ground-received carrier power of about 1.5 db. The decrease was confirmed by all three tracking stations over the next two weeks. A review of the detailed records from DSIF 51 showed no abrupt change in receiver AGC at or near the time of MT-2. Instead it appeared that the ground-received carrier strength decreased gradually from March 31 to April 3 by about 1 db more than would be expected from nominal antenna gain and range loss considerations. From April 3 to 9 the ground-received carrier strength decreased more slowly than predicted so that at the end of this time, some of the earlier loss had been recovered. This anomalous behavior is still unexplained, although there is now considerable doubt that it was in any way connected with the MT-2 event. Instead it is felt that the AGC calibration procedures at the DSIF stations were responsible.

The third Canopus sensor cone-angle update (MT-3) was received in the data on May 7 at 14:28:15 GMT. In view of the anomalous data encoder deck skip at MT-1 and the received signal strength decrease possibly associated with MT-2, a thorough analysis of the data was performed at MT-3. All data indicated a cone-angle update to 86.5 deg that was normal in every respect.

The fourth Canopus sensor cone-angle update from 86.5 deg to 82 deg by CC&S command MT-4 occurred at 15:51:45 GMT on June 14. The event was normal in every respect.

g. Frequency shift at CY-1. A momentary loss of signal strength was noted by personnel at DSIF 42 coincident with the first CC&S cyclic pulse after MT-1 (CY-1 No. 34). The decrease in signal strength was great enough to cause both receivers at the station to give an out-of-lock indication, although the duration was so short that the telemetry demodulator remained in lock. The subsequent recovery was very rapid and all indications were that the spacecraft was functioning normally. Very careful observation was maintained during the next CC&S cyclic pulse, CY-1 No. 35, which occurred on March 4 at 03:42:23 GMT while DSIF 11 was tracking. There was an apparent 2-db decrease in signal strength for about 5 sec coincident with the CY-1. One receiver at DSIF 11 gave an out-of-lock indication momentarily, the other remained in lock throughout cyclic period.

The only characteristic of the two cyclics, which appeared significantly different from the previous cyclics for which no signal dropouts had been noted, was the fact that both occurred while the station was in one-way lock with the spacecraft. Most of the previous cyclics had occurred while the station was tracking in two-way lock. An investigation of the DSIF records showed momentary dropouts of the spacecraft signal coincident with all cyclics that had occurred in one-way lock, and no dropouts for any cyclics that had occurred during two-way tracking. This piece of information led to the investigation of the doppler data taken during the time of the dropouts, which showed that during the dropouts the spacecraft transmitted frequency increased by an average of 27.3 cps. This frequency shift explained the signal strength decreases at the DSIF stations, because the ground receiver was unable to track the carrier frequency with infinite speed. Ground tests were then performed on the proof-test model spacecraft, and it was determined that the frequency shift was a normal characteristic of the *Mariner* Mars 1964 radio subsystem during cyclic pulses in one-way lock, although the relatively poor resolution of the instrumentation used in ground testing precluded its discovery in the course of normal prelaunch tests. The shift was actually produced by a slight loading of the auxiliary oscillator supply voltage by logic circuitry whose state was controlled by the CC&S CY-1.

The logic circuitry in question was that which controlled the switching of the receiver between the two antennas if two-way RF lock was not obtained, at least periodically. If a CC&S CY-1 pulse were issued when the spacecraft was not in lock with a DSIF transmitted signal, then a logic gate was set which would allow the antenna transfer to take place when the next CY-1 pulse was issued. Lockup before the next CY-1 pulse would reset the logic gate again. It was the setting of the logic gate that loaded the auxiliary oscillator power supply, thus changing slightly the transmitted frequency. Resetting the logic had no effect upon the spacecraft transmitted frequency, since the reset occurred only when the spacecraft was locked to the ground signal so that the spacecraft transmitted frequency was derived from the ground frequency.

After the discovery that the frequency shift was a design characteristic of the spacecraft, it became apparent that the only disadvantage associated with allowing it to occur was the possible loss of some small portion of the telemetry as the result of ground receiver losses of lock. Scheduling the DSIF to track two-way during all cyclic pulses, however, also had associated with it a disadvantage: should it have become necessary to allow the receiver to transfer to high-gain antenna in order to command the spacecraft during the command blackout period, the time of automatic transfer would have been dependent upon the time of the last two-way period. If a CY-1 pulse occurred while the DSIF was tracking two-way, the earliest automatic transfer to high-gain antenna would be two cyclics later (133⅓ hr). If on the other hand two way tracking had been terminated just before the CY-1, then the earliest automatic transfer would be one cyclic later (66⅔ hr) if there were no subsequent lockups. As a result of these factors SPAC recommended that all pulse times be covered two-way during the portions of the flight where command capability existed via the low-gain antenna from 10-kw transmitters in order to avoid any data loss, and that the spacecraft be tracked two-way only just prior to CY-1 times during any periods where the command capability from 10-kw transmitters was marginal, to prevent any avoidable extended loss of command capability.

h. Best-lock frequency. As the flight progressed during the cruise phase, the DSIF experienced increased difficulty in obtaining two-way lock with the spacecraft during normal operations. Problems were encountered even when the received carrier power at the spacecraft was relatively strong. Investigation showed that when two-way lock was finally achieved, the transmitted frequency used by the station was significantly different from the predicted frequency. The prelaunch calibrations for the *Mariner IV* radio subsystem were consulted and it was suggested by the cognizant personnel that, for the ambient radio temperatures, the minimum-lockup-time frequency had shifted by over 2 kc. When this new frequency was tried the DSIF was unable to lock up at all, indicating a probable shift in the characteristics of the transponder. Since almost immediate lockup is mandatory in the event of any spacecraft emergency, an intensive investigation was initiated to find some way of providing an inflight calibration of the radio subsystem frequency characteristics. Several operational procedures were suggested for DSIF use that would provide the maximum amount of useful information for this recalibration in the course of normal DSIF operation. The procedures were approved and put into use on a regularly scheduled basis. The information resulting allowed radio subsystem cognizant personnel to recalibrate the transponder and to provide a continuing monitor on any further shifts. With the revised minimum-lockup-time frequency that this effort provided, DSIF lockup times were reduced to several seconds on the average.

i. Battery voltage. A possibly serious anomaly, first noticed just after the science cover deployment exercise and continuing throughout the mission, was the increasing battery voltage. Before the cover-deployment sequence, the battery charger was on and trickle-charged the battery at less than 10 ma. Battery voltage had risen to 34.4 v, indicating a full charge on the battery. At the end of the science cover deployment sequence, the spacecraft was reconfigured to a battery-charger-off, boost-mode-enabled state. For a brief time, the battery voltage dropped and then began a gradual rise that brought the voltage to 35.0 v. Initially it had been felt that a voltage of 35.0 v would indicate a possible battery failure. As a result of the *Mariner IV* battery behavior, a review was made of the principal parameters affecting battery voltage and it was discovered that the original estimate was somewhat low.

Two hypotheses concerning the state of the *Mariner IV* battery were put forward:

1. The battery voltage increase was a normal consequence of the 0-*g* gravitational field, temperature effects, and the small charge-current produced by the battery voltage transducer. In this case, no battery failure would be expected because the pressure

required to crack the cell case is approximately 20 times that generated in a heavily charged battery (about 5 psi).

2. The battery case was already cracked and the electrolyte (45% KOH) was slowly leaking out of one cell. The resulting decrease in conductivity of that cell would be interpreted by the battery voltage transducer as a higher battery voltage. If this were the case, battery failure would have to be expected if any power demands were made upon it.

The personnel responsible for the battery operation indicated that there was no positive support for the latter hypothesis, and that it should not be assumed to be the case unless the battery voltage reached 38.0 v. It remained, however, a possibility and as a result, SPAC recommended that no action be taken that could increase spacecraft power loads to the point that any battery drain would be required, unless it was judged necessary to the success of the mission.

Although the battery voltage continued to rise throughout the flight, the rate of increase leveled off as aphelion was approached. At the end of the mission on October 1, 1965 the battery voltage appeared constant at 37.2 v.

j. Solar pressure vane performance. The first activity of the solar pressure vanes was observed on December 2 and 3, 1964 when the +X and −X vanes (located on solar panels 4A1 and 4A5) moved in the proper direction to reduce the external torques about the spacecraft yaw axis. Performance analysis showed that the X-axis vanes had considerable effect in compensating for torque imbalances during the mission; however, their effectiveness was hampered by the fact that the variation of torque imbalance as a function of time was much higher than anticipated, although it was within design tolerances.

During portions of the mission the X-axis vanes were able to reduce the torque imbalances in the yaw axis to values on the order of 1–4 dyne-cm. During the remainder of the mission, the torque imbalances were somewhat larger due to the slow response time of the vanes in adapting to any step change in imbalance, but on the whole the X-axis vanes verified the feasibility of using solar pressure vanes in an adaptive mode to maintain the mean attitude at the center of the spacecraft system dead band.

The +Y and −Y solar pressure vanes (located on solar panels 4A3 and 4A7) did not move at all during the mission, except for some mechanical slippage apparently coincident with postinjection propulsion motor burn. It was probable that they failed as a result of electrical lockup. A characteristic of the solar pressure vane stepping motor was that if an attitude-control gas valve were fired within 30 msec of stepping motor activation, the solar pressure vane actuators would be permanently locked up. Data indicated that, within the time resolution of the telemetry, a gas valve had fired coincident with solar pressure vane electronics turn-on at CC&S command L-3. The possibility of lockup was expected, since the mode of spacecraft roll control changes at L-3 when the solar pressure vanes are turned on.

The failure of the Y-axis solar pressure vanes to operate in the adaptive mode was considered a design failure. The possibility of solar pressure vane lockup was recognized during the system test program before launch. Some modifications of the hardware were made, but there was insufficient time to make all the required design changes and to perform flight-acceptance testing to completely correct the situation before launch.

Throughout the mission, there was no evidence that the vanes were able to produce any effect in the damping mode by means of position changes in response to thermal input changes to the bimetal mounting arms. Torque variations were of such magnitude compared with damping mode restoring torques, that if such an effect did exist, it was masked completely.

k. PIPS bladder incompatibility. On February 3, 1965, an increase of about 4 psi in PIPS fuel tank pressure was observed in the data. This measurement had previously shown no change after the midcourse maneuver. Tests conducted by propulsion subsystem cognizant personnel indicated that the increase was probably caused by an incompatibility between the fuel tank bladder material and the hydrazine fuel, a problem that had been noted on most previous propulsion subsystems. Although estimates were made which indicated, based upon telemetry, that the worst-case pressure rise by the time of Mars encounter might be 40 psi (still well within tolerance) no further increase was observed during the balance of the mission.

l. Science cover deployment exercise. During the period from launch until planetary encounter the only noncruise activity of the spacecraft was the ground-command-initiated science cover deployment exercise, Fig. 3, performed on February 11. The objective of this exercise was threefold: 1) to deploy the science cover, 2) to pre-position the scan platform to the optimum encounter position, and 3) to turn off the battery

COMMAND VERIFICATION PHASE

Cruise Phase	DC-3	DC-2	DC-26	DC-2	DC-28	Decision point	DC-25	DC-24	DC-28	DC-3	DC-2	DC-26	DC-2
Cruise science on			Cruise science off	Cruise science on								Cruise science off	Cruise science on
NRT DAS off							NRT DAS on					NRT DAS off	
No signals from NRT DAS to VSS							Stop record signals to VSS from NRT DAS, start record signals inhibited					No signals from NRT DAS to VSS	
TV off							TV shutter cycling					TV off	
Mode 2 engineering and science telemetry	Mode 3 all science telemetry	Mode 2 engineering and science telemetry	Mode 2 engineering telemetry only	Mode 2 engineering and science telemetry						Mode 3 all science telemetry	Mode 2 engineering and science telemetry	Mode 2 engineering telemetry only	Mode 2 engineering and science telemetry
VSS 2.4-kc power off							VSS 2.4-kc power on		VSS 2.4-kc power off				
No power available to record motor							VSS record motor off					No power available to record motor	
Scan 2.4-kc power off							Scan 2.4-kc power on					Scan 2.4-kc power off	
Scan platform stopped							Scan platform searching	Scan Platform Stopped					
400-cps 1-phase inverter off							400-cps 1-phase inverter on					400-cps 1-phase inverter off	
Battery charger on, boost mode off			Battery charger off, boost mode on		Battery charger on, boost mode off		Battery charger off, boost mode on		Battery charger on, boost mode off			Battery charger off, boost mode on	
Science cover up							Science cover deployed						
Standard cruise phase configuration established after midcourse termination	DC-3 switches D/E to Mode 3 so that DC-2 will have a measurable effect.	DC-2 switches D/E to Mode 2 which checks command decoder associated with DC-2. This is to give confidence cruise science can be turned back on.	DC-26 turns off cruise science which checks power distribution logic to assure that encounter science can be turned off.	DC-2 turns cruise science on reducing the power transient at DC-25.	DC-28 turns on battery charger which checks power distribution logic to assure that VSS (tape) 2.4 kc can be turned off.	**At this time, the decision is made whether to proceed with the deployment sequence or return to the cruise configuration.**	DC-25 turns on encounter instruments, starts scan search mode and deploys science cover.	DC-24 is timed to position scan platform with TV pointed for optimum Mars pictures.	DC-28 turns off VSS 2.4-kc power and turns on the battery charger.	DC-3 allows knowledge of TV shutters for calculation of DC-26 transmit time so shutter will be closed to Mars.	DC-2 switches D/E to Mode 2 so there will be telemetry after DC-26.	DC-26 is timed to arrive so that TV shutter is closed when encounter power and 400-cps 1-phase inverter are turned off.	DC-2 turns on cruise science and the RT DAS and reestablishes the cruise mode.

Fig. 3. Science cover deployment sequence

23-1

23-2

Cruise phase	DC-25	MT-7	DC-24
Cruise science on			
NRT[d] DAS[e] off	NRT DAS on		
No signals from NRT DAS to VSS[f]	Stop record signals to VSS from NRT DAS; start signals inhibited		
TV off	TV shutter cycling		
Data Mode 2 engineering and science telemetry			
VSS 2.4-kc electricity off	VSS 2.4-kc electricity on		
No power available to record motor	VSS record motor off		
Scan 2.4-kc electricity off	Scan 2.4-kc electricity on		
Scan platform stopped	Scan platform searching		Scan platform stopped
400-cps, 1-phase inverter off	400-cps, 1-phase inverter on		
Battery charger off, boost mode on			
Standard cruise phase configuration established during cover drop exercise.	Preempts MT-7; verifies DC-25 capability if needed in recycle; time chosen to allow scan search to planet as backup	Backup to DC-25; if DC-25 has failed, transmit DC-24 to arrive 8-sec after MT-7 occurs at spacecraft.	Scan platform positioned to establish scan/planet geometry. Position chosen to permit optimum TV pictures.

RECYCLE SEQUENCE, IF NEEDED

Cruise phase	DC-3	DC-2	DC-26	DC-2	DC-28	DC-25
Cruise science on			Cruise science off	Cruise science on		
NRT[d] DAS[e] off			NRT DAS off			NRT DAS on
No signals from NRT DAS to VSS[f]			No signals from NRT DAS to VSS			Stop record signals to VSS from NRT DAS, start signals inhibited
TV off			TV off			TV shutter cycling
Data Mode 2 engineering and science telemetry	Data Mode 3 all science telemetry	Data Mode 2 engineering and science telemetry	Data Mode 2 engineering telemetry only	Data Mode 2 engineering and science telemetry		
VSS 2.4-kc electricity off					VSS 2.4-kc electricity off	VSS 2.4-kc electricity on
No power available to record motor						
Scan 2.4-kc electricity off			Scan 2.4-kc electricity off			Scan 2.4-kc electricity on
Scan platform stopped						Scan platform searching
400-cps, 1-phase inverter off			400-cps, 1-phase Inverter off			400-cps, 1-phase inverter on
Battery charger off, boost mode on					Battery charger on; boost mode off	Battery charger off; boost mode on
Standard cruise phase configuration established during cover drop exercise.	Allows knowledge of TV shutter for calculation of DC-26. Transmit time; permits early look at TV and NRT DAS status.	Allows spacecraft engineering telemetry when DC-26 is transmitted.	Transmit time chosen so that TV shutter is closed at turn-off. Permits recycling of NRT-DAS, scan and TV.	Turns on cruise instruments and RT DAS; power surge of encounter science on reduced; if DC-2 fails, backup is DC-25.	Permits recycling VSS; eliminates possible adverse effects of DC-26, DC-2, and DC-25 with VSS 2.4-kc electronics on.	VSS, NRT-DAS, scan and TV recycled. Time chosen to allow scan search to planet as backup.

Cruise phase	DC-3	WAA[a]	NAS[b]	NAS + 56.4 to 200.4 sec	DC-16	2nd EOT[c]	EOT After Picture 19 Starts	End of Picture 22	DC-26	DC-2	MT-8	MT-9
Cruise science on									Cruise science off	Cruise science on		Cruise science off
NRT[d] DAS[e] off									NRT DAS off			
No signals from NRT DAS to VSS[f]				Start and stop signals to VSS from NRT DAS; record sequence initiated		Start signals from DAS inhibited in VSS	Start signals to VSS inhibited; stop signals to VSS from NRT DAS		No signals from NRT DAS to VSS			
TV off									TV off			
Data Mode 2 engineering and science telemetry	Data Mode 3 all science telemetry	**	**				Data Mode 2 engineering and science telemetry		Data Mode 2 engineering telemetry only	Data Mode 2 engineering and science telemetry		Data Mode 4/1 recorded data RT engineering data
VSS 2.4-kc electricity off												
No power available to record motor				VSS record motor alternately on/off with each picture		VSS record motor stopped by VSS internal command			No power available to record motor			Playback of TV pictures initiated
Scan 2.4-kc electricity off									Scan 2.4-kc electricity off			
Scan platform stopped	Scan platform stopped if no recycle	*	***									
400-cps, 1-phase inverter off									400-cps, 1-phase inverter off			
Battery charger off, boost mode on												
Standard cruise phase configuration established during cover drop exercise.	Permits evaluation of TV and NRT-DAS status; allows accurate time for DC-16 and DC-26 to be chosen.	Backup to DC-3 in prepositioned scan sequence; *In recycled sequence wide-angle sensor tracks planet. **DAS switches data encoder to Mode 3 (backup).	Issued when narrow-angle sensor "sees" planet. ~~**DAS switches data encoder~~ to Mode 3 (backup). ***Stops scan platform if in search or track mode.	First start-record command to VSS issued by NRT-DAS at proper time in frame.	Backup to NAS timed to arrive after two pictures could be recorded in latest normal sequence.	Recording stops at 2nd EOT signal; prevents recording data over first picture.	Data mode switch to occur at first EOT signal sent to NRT-DAS after start of Picture 19.	Backup logic in NRT-DAS to inhibit start signals to VSS and cause data mode switch.	Backup to VSS stop record relay circuit; timed to arrive 5-sec after Picture 22.	Reenergizes cruise instruments and RT DAS; backup switch to data Mode 2.	Backup to DC-26 turns off engineering science and 400-cps 1-phase power.	Energizes playback motor in VSS. Initial data will be Mode 1; approx. 1-hr later recorded data starts.

a = wide-angle acquisition
b = narrow-angle signal
c = end-of-tape signal
d = nonreal time
e = data automation subsystem
f = video storage subsystem

Fig. 4. Encounter sequence

24-2

charger and enable the boost mode. The exercise was carried out over the DSIF 11 tracking station. A total of 12 ground commands were transmitted to the spacecraft. All were received and executed normally. All objectives of the exercise were fully achieved, and the spacecraft was returned to the cruise state without difficulty.

Throughout the entire exercise, only one anomaly was observed. After an apparently normal command lockup, a loss of command lock occurred. About 6 min later, command lock was regained; there was no explanation for the loss. The balance of the sequence went smoothly; telemetry confirmed that the science cover was deployed, the scan platform was pre-positioned to within 0.7 deg of the optimum position, and all systems functioned without any apparent degradation throughout the entire sequence. Only about 44 min of cruise science data were lost during the periods that cruise science was turned off.

A nonreal time analysis of the cover deployment operation on February 11 revealed a severe roll transient had occurred that was completely unnoticed during the actual operation because of noise in the data. As one of the objectives of the early deployment of the science cover was to preclude the possibility of particles dislodged by a science cover deployment at encounter from causing a loss of Canopus acquisition, the exercise must be considered highly successful.

5. Planetary Encounter

The basic design of the *Mariner* Mars 1964 spacecraft provided for a completely automatic planetary encounter sequence without any ground action required. Because the ground command option was available, however, there existed a number of possible backups to the onboard functions, and in addition, there existed a number of alternate modes which might have been superior in any given situation to the normal mode of operation. In order to provide the best possible sequence of events for the encounter, the Encounter Planning Working Group (EPWG) was formed in January 1965 to operate in parallel with SPAC for the purpose of investigating all spacecraft encounter-related operations and modes, and recommending to the *Mariner* Project Manager a detailed encounter sequence of events. The EPWG was composed of appropriate SPAC, division, and subsystem representatives, and was chaired by the *Mariner* Spacecraft Project Engineer. Also formed was the Occultation Working Group for the purpose of recommending the configuration of the Space Flight Operations Facility (SFOF), the DSIF stations, and the spacecraft during occultation of the spacecraft by Mars to provide the most meaningful occultation data.

It was at the recommendation of the EPWG that the science cover deployment exercise was performed on February 11, 1965 to configure the spacecraft for encounter. At that time, the science cover was deployed and the scan platform was positioned to an angle that would permit useful television data in the event that a scan platform failure was experienced before the actual encounter.

The fourfold objective, to which the EPWG worked, was to:

1. Develop an encounter sequence that provided the maximum assurance of obtaining useful television and occultation data.
2. Determine the operational mode providing the greatest assurance of maintaining attitude stabilization throughout encounter, thereby assuring real-time planetary field and particles data and occultation data.
3. Provide for full utilization of backup methods for all critical functions.
4. Select appropriate alternate modes for any functions both functionally-critical and time-critical.

A recommended encounter sequence of events, Fig. 4, was formulated by the SPAC Director based upon the findings of the EPWG and was approved by the *Mariner* Project Manager.

While minor departures from the nominal sequence were made, none affected the logic of the sequence or its operation on the encounter day.

The major problem in encounter operations was the transmission delay between Earth and the spacecraft. Approximately 13 min were required from data transmission at the spacecraft until data presentation in SPAC; over 12½ min were required from command initiation at a DSIF station until command execution at the spacecraft. Thus, immediate response to anomalous indications in the data could only reduce the effective reaction time in an emergency to something over 25 min. Conversely, normal command action as a part of the sequence of events could be taken only if it was assumed that nothing abnormal had occurred in the 13 min prior to command initiation, and that nothing abnormal would occur during the 12½ min before command execution. Since most of the commands to be transmitted during the planetary encounter were time-critical or functionally-critical, it was most important to prevent accidental losses of command lock and, in the event that any occurred, to provide for minimum-time reacquisition of the command loop. Only the DSIF station personnel could act on the former, but the latter was achieved by maintaining the command subcarrier at the slightly offset, minimum-lockup-time frequency normally used for initial lockup, but not for command transmission. This offset was not large enough to significantly increase the probability of dropping lock at encounter signal levels, yet it offered a reasonably small time between pseudonoise code coincidences for a command loop out-of-lock. While this minimum lockup time provision was never required during the subsequent planetary encounter sequence, the flawless operation of the command detector and decoder in accepting and executing all commands verified the practicability of this approach for critical operations in the blind or with very long delays.

a. Planetary acquisition and record sequence. The encounter sequence was initiated with the transmission of DC-25 from DSIF 51 on July 14 at 14:27:55 GMT. This time was selected on the basis of the latest flyby trajectory information to initiate scan search at such a time that, given nominal performance, the scan platform would be pointing at the planet at the nominal narrow-angle acquisition time if the scan platform failed to inhibit either via ground command or internal logic. The command was verified in the data at 14:52:32.3 GMT. A low-rate deck skip was observed coincident with the command execution. Deck 400 was reset after reading out channel 409, but prior to being stepped to the next position, so that the next reading of that deck was position 401. While data encoder deck skips are not totally unexpected with a DC-25, they are not usual if the battery charger is off, so that its state is not changed by the command. Spacecraft response to the command was normal; NRT power was turned on, video storage subsystem 2.4-kc power was turned on, the scan platform went into a normal search, and power levels and temperatures increased as predicted.

The CC&S MT-7 event was observed in the telemetry data at 15:53:48.6 GMT, but inasmuch as it had been preempted by DC-25, it had no effect upon spacecraft performance.

Prior to the planetary encounter a determination was made by the television experiment personnel that a scan position of 180 deg would yield the best television data. This then was selected as the nominal position at which to inhibit the scan platform. An inhibit command (DC-24) time was predicted by the Space Science Analysis and Command Group (SSAC) scan analyst based upon scan platform performance data during scan search. The DC-24 was initiated at DSIF 51 at 17:10:18 GMT for a predicted scan platform position of 179.9 deg. Command execution was verified in the data at 17:34:55 GMT with the scan platform positioned to 178.45 deg, well within the tolerances. Based upon this position, wide-angle acquisition (0.7 probability) was predicted for 2355 GMT, with less than 0.01 probability of it occurring prior to 2324 GMT.

A normal two-way RF transfer was made from DSIF 51 to DSIF 11 at 2010 GMT. The ground transmitter frequency was set to a value which would permit transmission to the spacecraft through entrance of the occultation region without a frequency adjustment, and command modulation was reapplied at the nominal offset frequency at 21:00:30 GMT. Command-loop lockup was observed in the data at 21:54:19 GMT, indicating a lockup time just under 30 min, a normal time for the low uplink carrier power level at the spacecraft.

Beginning just prior to 2000 GMT and continuing through the encounter, anomalous indications were noted in the magnetometer X-axis data which appeared to be cyclic with a period of approximately 16 min. Although unexplained to date, the period corresponded to coincidences between the real time (RT) and NRT portions of the DAS, which count at different rates. The anomaly had no effect upon the quality of planetary data gathered.

Had the scan platform been allowed to search for and track the planet automatically, acquisition of the planet by the scan sensor would have initiated a switch to Mode 3 data (all science data) several hours before the recording sequence. With the scan platform prepositioned, however, the expected time of wide-angle acquisition (WAA) was so near the beginning of the television record sequence that insufficient time was available for television performance analysis prior to the latest time to transmit commands. As a result, a commanded switch to all science data (DC-3) was included in the encounter sequence. The DC-3 was transmitted from DSIF 11 at 22:10:29 GMT, was received by the spacecraft at 22:23:07 GMT, and was observed in the data at 22:35:08 GMT. Subsequent analysis by the television and NRT DAS cognizant personnel indicated that television sequencing was normal.

WAA occurred at the spacecraft at 23:42:00 GMT on July 14. No effect was noted, other than the indication in the science telemetry data, because the scan platform had been inhibited by DC-24 and the switch to Mode 3 data had been accomplished by DC-3. The backup command (DC-16) for the initiation of the recording sequence was transmitted on July 15, 1965, at 00:11:57 GMT. It was anticipated that this command would arrive at the spacecraft after two pictures had been recorded by the video storage subsystem if normal narrow-angle acquisition by the onboard sensors had initiated the sequence.

On the basis of the latest trajectory information and the DC-16 transmission time, SSAC recommended that the latest time for transmission of the DC-26 command to back up the termination of the recording sequence should be 00:36:20 GMT. The actual time for DC-26 transmission was to be determined from the narrow-angle acquisition (NAA) indication in the data, if NAA acquisition was observed earlier than 00:34:40 GMT. In order to provide the necessary real time data analysis, a ground telemetry analyst (GTA) was stationed at DSIF 11 to report the data by voice line to SSAC. The Goldstone GTA reported NAA in the data at 00:29:59 GMT and confirmed it on the next data frame. The DC-26 was initiated, upon SSAC's request for immediate transmission, at 00:31:42 GMT. A DC-2 command to turn cruise science back on followed at 00:32:40 GMT, and it was followed by more DC-2 transmissions beginning at 00:37:00 GMT and continuing every 5 min until a total of six DC-2 commands had been transmitted to the spacecraft.

Although the recording sequence appeared to start normally, anomalous indications of status were observed in the science data which could not be interpreted until after the video data playback was completed on July 24. The first anomaly was observed at 00:34:57 GMT and appeared to be an end-of-tape (EOT) event, although none was expected until approximately 00:42:30 GMT. One of the postulated failure modes for the record sequence involved the failure of the DAS to send, or the video storage subsystem to receive, tape stop commands, with the result that the video storage subsystem would have recorded continuously. Given this failure mode, the first EOT event would have been expected at about 00:35:30 GMT. If it were assumed that the tape was improperly positioned past the EOT foil, then the observed event would correlate with expected telemetry indications for this failure mode. If so, a second event would have been expected at approximately 00:40:08 GMT. A second end-of-tape event was observed in the data at 00:40:50 GMT. Since the uncertainty in time of occurrence of an event based upon the time of first observation in the data is +0.0 and −50.4 sec, this time also correlated with the postulated failure mode. With the possibility that the failure mode existed clearly established, it was expected either that no further EOT events would occur, or that they would occur each 309 sec. This was not the case, because a third EOT event was observed in the data at 00:43:20 GMT, the proper time for track change in a nominal sequence. In the next data frame the false shutter event (distinguishable from EOT events only by its timing) was observed. This also would be expected in a nominal sequence. No further events were observed in the science data.

The contradictory indications in the data made it impossible to determine what had taken place until it was determined during video playback what data, if any, were stored upon the tape. It was expected, however, that if the sequence were in fact normal, the second true EOT would occur during the recorder runup prior to Picture 22 and would immediately switch the data encoder to Mode 2 data. The mode switch would be observed in the data at about 00:55:24 GMT. The actual switch to Mode 2 data was not observed until 00:55:46.6 GMT, coincident with the end of Picture 22, indicating that the data mode switch was initiated by the DAS rather than the video storage subsystem. This tended to lend support to the hypothesis that the video storage subsystem was not running during the latter portion of the recording sequence. The engineering data upon return to Mode 2, however, indicated the correct number of event-register changes for a nominal recording sequence.

After the planetary encounter, investigation showed that the *Mariner IV* spacecraft had demonstrated a tendency toward spurious EOT events in the Mode 3 science data although the previous occurrences had been associated with some ground action such as *boost mode inhibit off* or *narrow-angle Mars gate stimulus on/off.* Thus, the two anomalous EOT events could have been triggered by noise in the system, or were the result of some normal transient in the television sequencing.

That the recording sequence was, in fact, normal throughout was verified during the playback of the stored data. All of the postulated potential failures would have resulted in less than 21 pictures, while only the normal sequence would have yielded the 21 pictures plus 21 lines of Picture 22 which were found to be present on the tape. Some additional light was thrown on the events at encounter when laboratory testing showed that the switch to Mode 2 data at the end of Picture 22 was normal. The mode switch could have occurred immediately upon an EOT signal during Pictures 19 through 21, but was inhibited after the start of Picture 22. If the end-of-tape occurred during Picture 22 (as it did at encounter) the video storage subsystem would stop recording and reject any further *start* commands, but the DAS would not issue the *mode switch* command to the data encoder until the Picture 22 *stop* command was issued to the video storage subsystem.

This fact had not been uncovered before launch because none of the *Mariner* tape machines had a tape of the precise length required to place the second end-of-tape in the period of actual data transfer for Picture 22. During prelaunch testing, the *Mariner IV* second end-of-tape occurred during the runup for Picture 22, but before the start of video data. The *Mariner III* end-of-tape events occurred during the rundown from Picture 21, and MC-1 (the PTM) had an extra-long tape, so that the DAS always initiated the mode switch after 22 pictures.

The DC-26 command (all science off) was observed in the telemetry data on July 15 at 00:56:23.0 GMT and SPAC confirmed *all-zeros* science data. The first DC-2 command (cruise science on) followed at 00:57:20 GMT. After confirming that cruise was on and appeared normal, the DSIF was requested to cease transmission of DC-2 commands. Event register changes confirming the reception of the remaining five DC-2 commands by the spacecraft were subsequently received at the expected times. Further analysis of the telemetry indicated that all engineering and science subsystems were performing the same as before encounter, and that the spacecraft had returned to a normal cruise state. The remaining encounter event, MT-8, occurred several hours later, appearing in the data at 05:14:39 GMT. This event, a CC&S command to turn off encounter science, had been preempted by ground command and did not affect the state of the spacecraft.

b. Occultation. The spacecraft RF signal was lost at the DSIF on July 15 at 02:31:12 GMT as *Mariner IV* entered the Earth-occultation region of Mars. Four stations were tracking: DSIF 11 and DSIF 42 as prime stations and DSIF 12 (Echo site, Goldstone, California) and DSIF 41 as backup. All stations were able to obtain useable data. The occultation occurred about 8 min later than the time estimated prior to closest approach. While this is presently unexplained, an error analysis is being performed on the trajectory program and its inputs by the cognizant Flight Path Analysis and Command Group (FPAC) personnel to determine the source of the error in the estimate of closest approach distance and the subsequent flight path.

After spacecraft entrance into the occultation region, the 100-kw transmitter at DSIF 13 was turned on in an attempt to obtain two-way tracking upon spacecraft exit from the occultation region. The spacecraft RF signal was reacquired by DSIF 42 at 03:25:06 GMT and by DSIF 41 shortly thereafter. Useful data were obtained by both sites and by the wideband, open-loop receivers at DSIF 11 and DSIF 12. Analysis of the telemetry by SPAC subsequent to the exit indicated that no apparent changes in the state of the spacecraft had taken place during the occultation and that performance was normal.

6. Playback

Video data playback was initiated by the CC&S MT-9 event on July 15 at 11:41:49.8 GMT. The event was observed on Earth at 11:53:53.3 GMT, and the expected switch to Mode 4/1 data was confirmed. An anomaly was noted at MT-9 when two Register 2 events were received rather than the expected one. CC&S CY-1 No. 83 was coincident with MT-9. Since the relays involved are driven from two separate points in the sequencing chain, it was believed that the two events were sufficiently separated in time that each caused the event register to increment, where normally the register input would not have recovered sufficiently from the first input to accept the second.

After approximately 68 min of engineering data, the first Mode 4 (picture) data were observed in telemetry at 13:01:58 GMT. From this point the playback proceeded in an entirely normal manner through the termination of the record playback on August 3. During this period, the data from each of the pictures were transmitted to Earth twice, with the exception of the last few lines of Picture 22. The data recovery rate was quite high, and the performance of the spacecraft was normal throughout.

In view of the uncertainty associated with the recording sequence as to the EOT events, it was decided that command alerts should be called for each of the suspect periods during playback. After the end of Picture 4, DSIF 42 locked up with the spacecraft command detector and maintained lock until after

Picture 5 had been identified in the data and it was verified that video data had been recorded on the day of encounter. Similarly, DSIF 11 locked up with the command detector after the end of Picture 8, and maintained lock until Picture 9 could be verified. DSIF 51 locked up with the command detector and maintained lock until after the time required to reach the point on the tape where the tape rundown after Picture 9 had stopped. Thus, during all of the periods which could have coincided with the spurious events received on July 15 a minimum-reaction-time command capability was provided to ensure that corrective action could be taken, if necessary, to obtain the most useful data from the playback. During all three command alerts, all spacecraft data indicated that performance, both in playback and at encounter, was normal.

An additional alert was called at the time of the expected video storage subsystem track change, although the command detector was not locked up. The alert was called because it was possible, due to the low speed with which the EOT foil passed the EOT sensor during playback, to obtain multiple track changes. The track change occurred without incident, but an interesting sidelight took place which provided the answer to the question of how many pictures had been recorded on the tape. Because of the physical geometry of the EOT sensor and the playback and record heads in the video storage subsystem, the playback head will play, upon track change, a number of seconds of data recorded prior to the EOT. It was expected that this portion of the tape would be for the start-up for Picture 22 based upon prelaunch testing, and thus in the absence of video data, engineering data would be transmitted to Earth. Instead, the data received were video data, and upon decoding it, it was determined that it was Mars data from the 21st and 22nd lines of Picture 22, thus verifying that 21 complete pictures and a portion of the 22nd had been successfully recorded.

The first complete transmission of the stored data ended on July 24 at 19:26:33 GMT and a second transmission began automatically. This second playback was necessary to ensure that any data missing from the first transmission of the pictures might be recovered and to provide duplication of the remainder of the data for comparison purposes. Spot checks of the data from both transmissions confirmed that the data were repeatable, so it was planned to return the spacecraft to the cruise mode at the end of the second playback.

One anomaly occurred during the second playback of the video data. At the first EOT signal, two events were recorded in data encoder Register 3. The track change was normal, however, and the extra EOT event was attributed to dirt or foreign material on the foil. Based upon prelaunch test data, this extra event was not unexpected.

Shortly before the playback phase of the mission was terminated, SSAC and the television experimenters recommended to the *Mariner* Project Office that a second recording sequence be considered. Such a sequence could provide information on the behavior of the television electronics for uniformly black pictures at each of the various gain settings. With this possibility in mind, the video storage subsystem representative and SPAC recommended that playback be terminated in such a manner that some portion of the Mars data would not be erased in the event that another recording sequence were performed. By stopping the tape at some position other than the EOT foil, from 1 to 10½ pictures could be protected, since in a recording sequence the tape would run only until the EOT foil had been crossed twice.

This recommendation was accepted, and the playback was terminated in Line 18 of Picture 22, thus protecting all of the complete pictures on the second track. The playback could have been terminated and cruise mode restored by transmission of a DC-2, which would inhibit the playback motor, switch to Mode 2 data, and turn on cruise science, but it would have allowed the 2.4-kc power to the video storage subsystem to remain on. Since this was an unfused power supply, a line-to-line short would have resulted in a catastrophic failure of the spacecraft power subsystem. For this reason, it was decided to include a DC-28 command to turn off the 2.4-kc power to the video storage subsystem. This command also disabled the boost-mode and turned on the battery charger, so it had to be followed by a DC-26 command to turn off the battery charger again and reenable the boost mode. Finally, a DC-2 could be used to switch to Mode 2 data and turn on the cruise science.

The DC-28 was transmitted on August 3 at 03:08:33 GMT, the DC-26 at 03:14:33 GMT, and the DC-2 at 03:20:33 GMT. Telemetry verified that video storage 2.4-kc power turned off, battery charger turned on, and Mode 1 data (indicating video data not present) received at 03:22:37 GMT. The DC-26 command event and battery charger off were observed in the data at 03:28:37 GMT, and coincident with it, the expected data encoder medium-rate reset which generally accompanies large power transients. DC-2 was verified in the data at 03:48:02 GMT as the data encoder switched to Mode 2 data and cruise science came on. All cruise science instruments were verified to be operating as before encounter, and all engineering subsystem performance was verified as normal.

7. Postencounter Cruise

The spacecraft continued to perform well throughout the final 8 wk of Phase I of the *Mariner IV* mission to Mars. One spacecraft anomaly was noted when the cosmic-dust detector instrument sporadically began to return apparently abnormal data. Problems occurred which caused the postponement of command sequences twice, but investigation verified that in each case the problems were groundbased at the DSIF rather than associated with the spacecraft itself.

Prior to the planetary encounter, proposals had been made to the Project Manager as to the operational procedures to be implemented after the completion of the video data playback. Each of the interested groups was encouraged to propose tests or operations which would lead to the optimum recovery of science and engineering data. After reviewing the proposals the Project Manager laid down four basic guidelines to govern postencounter operations and testing:

1. No test or operation should be undertaken that would interfere significantly with the gathering of cruise science data unless it was necessary to mission success.
2. All tests and operations should be nondestructive and carry minimum risk to the continuous operation of the spacecraft.
3. Postencounter activities should enhance the possibilities of reacquiring an active spacecraft at next closest approach to Earth in 1967.
4. Subsystem performance verification tests would not be considered to be of extremely high scientific or engineering value.

These guidelines led to the approval of a series of tests and operations to be conducted during the remainder of the portion of the postencounter cruise in which it would be possible to maintain continuous telemetry coverage. The majority of the tests were passive in nature, involving DSIF tracking techniques and equipment. Active tests included inflight calibrations of the *Mariner IV* radio and command subsystems and a *dark* encounter sequence to provide television calibration data. The radio and command subsystem tests were considered necessary to any attempt to reacquire the spacecraft in 1967, and the television calibration was requested by the television experimenters as necessary to provide an insight into the possible fogging of the Mars pictures resulting from some unknown phenomenon in the television electronics. Three flight operations were also approved: 1) updating of the Canopus sensor cone-angle, which was necessary to maintain telemetry and therefore cruise science data as long as possible; 2) the insertion of the maneuver-inhibit and minimum turn and burn duration commands as protection against a failure-induced, inadvertent trajectory-correction maneuver before the 1967 reacquisition period; and 3) the transfer of the spacecraft transmitter to the low-gain antenna, which is the reacquisition configuration for 1967.

a. Television haze calibration sequence. The initial attempt to perform the television haze calibration sequence, Fig. 5, on August 21 was cancelled after DSIF problems occurred which posed a threat to the ability to react via ground command to any nonstandard occurrence onboard the spacecraft. A second attempt was successful on August 30.

The first attempt was scheduled on August 21 because the star Altair, with an apparent +0.8 magnitude, would be at the proper cone angle to be photographed. While such a small light source would not be resolved by the *Mariner* television subsystem, the effects of the light source should be apparent in the pictures, thus providing calibration data for the television optics as well as the electronics. The planned command sequence followed closely the planetary encounter sequence of events, without, of course, the onboard events.

The sequence was to be performed from DSIF 13 using the 100-kw transmitter. The spacecraft radio subsystem was locked up normally, and then command modulation was applied. Shortly afterward, one of the safety interlocks at the station tripped and the transmitter automatically shut off. After an investigation by station personnel indicated that the transmitter had been operating normally, the transmitter was turned on again and command modulation was reapplied. Again an interlock tripped, shutting off the transmitter. A brief analysis by the station personnel indicated a probable failure of the interlock circuitry.

Cruise phase	DC-25	DC-3	DC-24	DC-16	NAS[a]	NAA[b] + 60 to 72 sec	1st EOT[c]	2nd EOT	End of Picture 22	DC-2	DC-26	DC-22	DC-4	Pictures 1 through 5	DC-28	DC-26	DC-2
Cruise science on											Cruise science off						Cruise science on
NRT DAS off	NRT DAS on										NRT DAS off						
No signals from NRT[d] DAS[e] to VSS[f]	Stop record signals to VSS from NRT DAS, start record signals inhibited					Start and stop signals to VSS from NRT DAS, record sequence initiated		Start signals from DAS inhibited in VSS	Start signals to VSS inhibited, stop signals to VSS from NRT DAS		No signals from NRT DAS to VSS						
TV off	TV shutter cycling										TV off						
Mode 2 engineering and science telemetry		Mode 3 all science telemetry			*				Mode 2 engineering and science telemetry		Mode 2 engineering telemetry only		Mode 4/1 recorded data RT[g] engrg data		Mode 4/1 engineering telemetry only		Mode 2 engineering and science telemetry
VSS 2.4-kc power off	VSS 2.4-kc power on												TV picture playback initiated		VSS 2.4-kc power off		
No power available to record motor	VSS record motor off					VSS record motor alternately on and off with each picture		VSS record motor stopped by VSS internal command			No power available to record motor						
Scan 2.4-kc power off	Scan 2.4-kc power on										Scan 2.4-kc power off						
Scan platform stopped	Scan platform searching		Scan platform stopped		**												
400-cps 1-phase inverter off	400-cps 1-phase inverter on										400-cps 1-phase inverter off						
Battery charger off, boost mode on															Battery charger on, boost mode off	Battery charger off, boost mode on	
Standard cruise phase configuration established after playback termination	DC-25 turns on encounter instruments and starts scan search mode.	DC-3 allows knowledge of TV filters and shutters for calculation of DC-16 transmit time to allow all gain levels to be recorded.	DC-24 is timed to position scan platform so that the star Altair is in the TV field-of-view.	DC-16 is timed to arrive after a green filter during the 12-sec NAA period. This will allow all gain levels to be recorded.	Caused by DC-16. *DAS switches data encoder to Mode 3. **Stops scan platform if in search mode.	First start record command to VSS issued by NRT DAS at proper time in frame (NAA + 60 to 72 sec).	First EOT switches recording from Track 1 to Track 2 during runup of Picture 1, saving one track of Mars data for playback in 1967 if desired.	Second EOT stops recording sequence, prevents recording data over Mars data on other track. (Occurs in Picture 11.)	Data encoder transferred to Mode 2 at end of Picture 22.	DC-2 is used as a backup to Mode 2 transfer by DAS at end of Picture 22.	DC-26 is timed to arrive so that TV shutter is closed when encounter power and 400-cps 1-phase inverter are turned off.	DC-22 switches VSS tracks so that Picture 1 of space will be played back first.	DC-4 energizes VSS playback motor. Initial data will be Mode 1; about 20 min later picture data starts.	Five pictures are played back in order to have one at each gain setting.	DC-28 turns off VSS 2.4-kc power, switching the data encoder to Mode 1 of Mode 4/1, and turns on the battery charger.	DC-26 turns off the battery charger and enables the boost mode.	DC-2 turns on cruise science and the RT DAS and transfers the data encoder to Mode 2.

a = narrow-angle signal
b = narrow-angle acquisition
c = end-of-tape signal
d = nonreal time
e = data automation subsystem
f = video storage subsystem
g = real time

Fig. 5. Television haze calibration sequence

Since such a failure would not permit the maintenance of continuous command capability, it was decided that the operation could not be conducted from DSIF 13. The received carrier power at the spacecraft had risen to the marginal command capability level for 10-kw transmitters during the preceding few days, so the SPAC telecommunications representative recommended having DSIF 12 transmit to the spacecraft. Two-way RF lock was established normally with the spacecraft, then command modulation was applied and the command detector locked up. All telemetry indications were normal, so the first command in the sequence, *encounter science on* (DC-25), was transmitted at 22:22:00 GMT. Shortly after transmission, the telemetry indicated that the carrier power at the spacecraft was fluctuating severely. Several samples of the command-detector monitor telemetry channel indicated momentary drops of lock by the command detector as the carrier level at the spacecraft dropped below threshold. Since these momentary out-of-lock indications appeared to signal either a DSIF failure or a spacecraft failure, a decision was made to terminate the attempted haze calibrate sequence. If the DC-25 was not received by the spacecraft, no further action would be required. If, on the other hand, the DC-25 command to turn on encounter science was executed, an emergency command sequence would be employed to return the spacecraft to a cruise condition.

At 22:52:04 GMT the data indicated the receipt and execution of the DC-25 command. After verification that the spacecraft was responding normally to the command, the emergency sequence, Table 6, was initiated.

The redundant commands in the sequence listed in Table 6 were designed to avoid the possibility that one of the momentary command dropouts would fall just before or during the receipt of any of the commands, thus inhibiting execution of the command. During the actual emergency return-to-cruise-state sequence, however, none of the redundancy was actually required, each command being executed normally on the first transmission. A total of seven DC-2 commands were transmitted before verification of the first one in the data was possible. An intensive investigation of the telemetry data showed that the spacecraft was in a normal cruise condition with no anomalies apparent in the data. Although the following day was also acceptable for obtaining television data using the star Altair as a light source, no exercise could be scheduled for August 22 without the complete assurance that: 1) there had been no failure or problem in the spacecraft radio or command equipment, and 2) all problems at DSIF 12 and DSIF 13 had been discovered and corrected. Since no such assurance could be given in the less than 24 hr available, the Project Manager decided to forego photographing Altair and that rescheduling of the haze-calibration sequence would be delayed until a full analysis could be made of spacecraft and DSIF performance.

Table 6. Emergency command sequence for return to cruise configuration

Time	Command	Remarks
T = 0	DC-28	Turn off video storage 2.4-kc power and turn on battery charger.
T + 2 min	DC-28	Redundant command.
T + 8 min	DC-26	Turn off battery charger and all science. T = 0 is chosen so that this command will arrive at the spacecraft when the scan platform is at the desired clock angle. The 8 min delay between the first DC-28 and DC-26 is designed to allow verification via telemetry of the DC-28 execution.
T + 10 min	DC-26	Redundant command.
T + 12 min	DC-2	Turn on cruise science.
T + 14 min	DC-2	Redundant command.
T + 19 min and each 5 min thereafter	DC-2	Additional redundant commands. DC-2 commands are sent each 5 min until telemetry has verified a normal return-to-cruise state.

Early in the week following the unsuccessful attempt to perform the haze-calibration sequence it was verified that the problems encountered were station problems. At DSIF 13 the interlock circuitry had failed, and investigations at DSIF 12 uncovered a degraded frequency synthesizer. After both sites had been reported operationally ready the haze calibration sequence was rescheduled for August 26 after the maneuver inhibit command sequence on August 25. The maneuver inhibit exercise on August 25 was postponed, however, after it was determined that DSIF 13 had further interlock problems that would require 24 hr to correct. DSIF 12 was considered for use in sending commands, but no firm commitment from the DSN office as to its operational status could be obtained on such a short time scale. The Project Manager directed that the maneuver inhibit sequence be rescheduled for August 25 contingent upon the correction of the DSIF 13 interlock problems, and that the haze calibration sequence be rescheduled for August 30 at the earliest.

The second attempt to perform the television haze calibration exercise took place on August 30 and was successful. The exercise differed from the first attempt only in that no star of sufficient magnitude was available as a light source. As a result the data gathered in the sequence applied only to the television electronics, rather than to both electronics and optics. The sequence was performed using DSIF 12 and a 10-kw transmitter because it was felt that the operational status of DSIF 12 more than compensated for the additional transmitter power available at DSIF 13, a research and development site not designed specifically for flight operations. This use of the smaller transmitter was possible only during the few days that the RF interferometer effect aided, rather than degraded, communication ability.

The sequence was initiated with a DC-25 command at 20:30:00 GMT on August 30. The sequence proceeded without incident as each command was received and executed in a normal manner. The command to initiate the television recording sequence (DC-16) arrived at the spacecraft at 23:51:22 GMT. It had been anticipated that the apparently erroneous EOT indications in the science data during the planetary encounter on July 15 might be repeatable, and that some indication of their source might be obtained during this recording sequence. No erroneous EOT indications were observed, however. The DC-16 was observed in the data on August 31 at 00:06:52 GMT and the EOT events were observed in the data at 00:08:19 GMT and 00:20:15 GMT, as expected. The nonstandard time to first EOT was due to the tape position at the start of the haze-calibrate sequence; the playback of the encounter data had been terminated just prior to the foil on the tape which signals the EOT. This allowed a calibrate sequence that would *record-over* only half of the planetary data, the video storage subsystem automatically inhibiting after the foil had been passed twice. Under these conditions the transfer back to Mode 2 data was initiated by the DAS at the end of 11 NRT frames (22 television pictures).

The balance of the sequence proceeded as planned, with playback data appearing in the telemetry at 01:56:16 GMT (Mode 1 data) and the first picture beginning at 02:00:46 GMT. The stored data were played back until the first five pictures, which included all of the television gain settings, had been completed (September 2 at 04:39:23 GMT). One hour later the DSIF 41 10-kw transmitter locked up with the spacecraft command detector and the sequence to return the spacecraft to cruise configuration was begun. The first command (DC-28) was transmitted at 06:17:00 GMT, and the first Mode 2 data, indicating the completion of the haze calibration sequence, were observed on Earth at 07:00:32 GMT. Analysis of the engineering and science telemetry indicated that all subsystems were performing as they had prior to the initiation of the sequence.

b. Maneuver inhibit commands. On August 26, four ground commands were transmitted to the spacecraft in order to protect against failures which might cause an inadvertent trajectory-correction maneuver to start. Discussions as to the desirability of inserting a DC-13 (maneuver inhibit) into the spacecraft had been held prior to the planetary encounter, but the recommendation of the EPWG and of SPAC had been against the maneuver inhibit because:

1. The risk was relatively small that such a failure would occur prior to encounter.
2. Multiple failures would be required to actually cause a motor start.
3. Without the motor burning, a trajectory-correction sequence would not be catastrophic.
4. With nearly continuous command capability and continuous telemetry monitoring, an inadvertent maneuver could be aborted via ground command.

After the planetary encounter, however, the fact that a major portion of the mission objectives had been completed and the addition of a new objective, the enhancement of the probability of a 1967 reacquisition, dictated that a review of the maneuver inhibit proposal be made.

The SPAC representatives recommended without question that the maneuver inhibit command be transmitted and a DC-13 transmission was scheduled for August 12 during the regular DSIF 13 tracking period. Additional analysis was performed to determine the desirability of also inserting minimum turn and burn duration QC, with the result that just prior to the scheduled DC-13 transmission an informal recommendation to transmit all four commands was presented to the Project Manager. In view of the proposed change, the Project Manager postponed any command action until a more formal review and recommendation could be made.

A recommendation to transmit both DC-13 and minimum value QC was presented to the Project Manager and approved by him on August 23, 1965. The maneuver-inhibit command exercise, including all four commands, was scheduled for August 25. Shortly before the transmission of the first command, problems were encountered in the operation of the DSIF 13 transmitter which forced the postponement of the exercise for 24 hr. The following day the DSN office reported that the 10-kw transmitter at DSIF 12 had been returned to a state of operational readiness. The SPAC telecommunications representative recommended to the SPAC Director that since DSIF 12 was prepared to transmit commands, it should be used rather than DSIF 13, which was subject to interlock problems because of its safety problems and its developmental, rather than operational, nature. This recommendation was accepted, and on August 26 at 21:06:52 GMT, DC-13, the first command of the sequence, was transmitted to the spacecraft. The three QCs followed approximately on 8-min centers. Telemetry from the spacecraft indicated that all commands were successfully received.

c. Canopus sensor cone-angle update. On August 27, 1965, a ground command was transmitted to the spacecraft to update the Canopus sensor cone angle to the first optional position. All previous cone-angle updates had been commanded automatically by the onboard CC&S, but no master-timer event was provided in the *Mariner* design for more than the four updates required through the playback phase of the mission. Thus any updates beyond the MT-4 event had to be commanded from the ground. Analysis indicated that while the nominal day for loss of Canopus due to exceeding the limits of the field of view of the sensor was September 7, adverse tolerances could combine to cause loss of acquisition as early as August 27.

The update command, DC-17, was transmitted from DSIF 12 at 19:40:00 GMT and was verified in the telemetry at 20:11:01 GMT. The telemetry data verified the new cone angle setting of 77.8 deg.

d. Communications capability. When playback ended on August 3, 1965, the spacecraft-received carrier power with command modulation applied was below −139 dbm, the worst-case command threshold for the 10-kw transmitters at the prime DSIF stations. The capability to command the spacecraft from prime stations was regained, as predicted, on August 21 when the received-carrier level rose above −139 dbm. The received-carrier level again dropped below the worst-case threshold on September 3 and remained at low levels for the balance of the period.

For a significant portion of the period after September 7, the received-carrier power without command modulation applied was sufficiently close to absolute RF threshold that the resulting low signal-to-noise ratio produced a large frequency jitter which was translated into the telemetry carrier. Each time the spacecraft was locked up, two-way ground-received carrier power dropped below the absolute telemetry threshold, causing a loss of all data. To prevent this, two-way tracking was forgone, and the spacecraft was locked-up only periodically for short durations to prevent the spacecraft from transfering to the receive-via-the-high-gain-antenna mode.

The average ground-received carrier power reached the worst case telemetry threshold of −159.6 dbm, about September 3. From that point, until October 1, 1965, the ground-received carrier power dropped steadily as the spacecraft high-gain antenna pointing error increased. On October 1, as predicted, the carrier level was fast approaching absolute telemetry threshold. At 21:30:17 GMT, a DC-12 command (transmit and receive via low-gain antenna) was transmitted from the 100-kw transmitter at DSIF 13. Telemetry was lost at 22:05:07 GMT, marking the end of the nominal mission. The transfer to low-gain antenna precluded the cycling of the receiver between the antennas by the radio subsystem logic circuitry, thus permitting access from the 100-kw transmitter as desired. In addition, the transfer of the transmitter to the low-gain antenna placed the spacecraft in the proper configuration for the projected attempt to reacquire the spacecraft in 1967.

During the 308 days of flight through the termination of telemetry, the *Mariner IV* radio subsystem returned well over 50 million measurements to the Earth. A total of 85 commands were transmitted to the spacecraft and all were received and executed normally. The last command, DC-12, was accepted by the spacecraft at a distance of more than 191 million mi from Earth.

III. SUBSYSTEM PERFORMANCE

A. *Structure and Mechanisms*

1. Description

The spacecraft subsystem is structurally integrated into a rigidly supported functional spacecraft. The major requirements on the structure occur during all prelaunch phases of the operation and during the boost phase and trajectory-correction maneuvers. The cruise portion of the flight requires that the structure maintain the mechanical alignments and that the spacecraft not fall apart. The *Mariner IV* structure was composed of the octagon structure, superstructure, and science platform structure.

The spacecraft mechanisms consist of those items that were required to actively function during the mission:

1. The boost dampers reduced the vibration inputs to the solar panels and the low-gain antenna during boost. They also statically positioned these items.
2. The SIT and PAS initiated onboard commands and applied power to the pyrotechnics subsystem at spacecraft/booster separation.
3. The solar panel deployment spring deployed the solar panels to the cruise position. Switches near the panel hinges gave telemetry indications that each panel opened.
4. The cruise dampers positioned the panels in the deployed position and damped solar panel excursions during propulsion maneuvers. Latches on the dampers engaged the panel to prevent subsequent separation.
5. The scan actuator rotated the science platform and instruments to search for and point to the Mars surface. The total scan amplitude was 180 deg. The actuator was controlled by the scan subsystem electronics.
6. The scan inhibit switch inhibited power to the scan actuator until the pin-puller, which torsionally restrained the platform, was fired. It also gave a telemetry indication that the pin-puller had released the platform.
7. The science instrument cover protected the planet-oriented instruments and sensors from sunlight and cosmic dust. It could be unlatched by either a solenoid or by a lanyard which was released at the start of platform scanning motion. The cover was deployed by two clock springs. A change in the spacecraft identification telemetry channel data number (DN) indicated the cover was deployed.

2. Performance

Because the function of all the structural and many of the mechanical items was passive after launch, little telemetry was transmitted about the performance of these items. As a result, the inflight performance of these components can only be deduced from other flight information. This is also true in the case of the electronic packaging design and the electronic cable harnessing.

All information gained during the flight indicated that the structure, electronic cabling, mechanical devices, and the electronic packaging design performed as designed. Telemetry was recovered in flight on the performance of the following mechanical devices:

a. Separation-initiated timer and pyrotechnics arming switch. Telemetry event indications verified that either the PAS or the SIT energized the pyrotechnics control assembly (PCA). The PAS closure arming the PCA was designed to occur at spacecraft/*Agena* separation; the SIT switch closure was a backup designed to occur about 40 sec after separation. It was not determined whether the PAS or the SIT energized the PCA, because telemetry data covering spacecraft separation were not available.

The first available data encoder event counter indications were received about 3 min after separation. The telemetry indicated a normal SIT-actuated solar panel deployment and unlatch of the planetary scan platform.

b. Solar panel deployment. Telemetry indications from the solar-panel deploy-switches, through the data encoder event counters, indicated that each of the four panels had been released by the pyrotechnic pin-pullers and had been deployed by the panel deployment springs to within 20 deg of the fully open position. Temperature and power measurements later confirmed panel deployment.

c. Cruise dampers and latches. Based on the successful trajectory-correction maneuver, it was deduced that the panels did deploy and latch in the fully open position and that any panel excursions during the PIPS motor firing were adequately damped. Had the dampers and latches failed to perform as designed, the center-of-gravity shifts caused by misaligned or excessively vibrating panels would have degraded the accuracy of the maneuver.

d. Scan actuator. The scan actuator was operated during the science cover deployment exercise for approximately 11 scan cycles. During the scanning period of 127 min in which the actuator functioned normally, the average scan cycle was 11 min 52.9 sec.

At encounter the actuator again operated within the design limits. During the 108 min of operation, the average scan cycle time was 11 min 53.2 sec. An analysis of the scan reversal points verified that the actuator reversed at the first set of limit switches. During the aborted television haze calibration sequence the actuator operated successfully for 66 min with an average scan cycle period of 11 min 53.8 sec. During the 139-min operating period of the completed sequence, the scan cycle was 11 min 53.5 sec.

The scan actuator was pressurized to 30 psia when it was assembled. At launch the pressure was 29.5 psia. During the science cover deployment, during encounter and during the television haze calibration sequence, the pressure was 28.8, 26.6, and 25.5 psi respectively. This decrease in pressure was attributed to seal leakage and porosity in the magnesium actuator cover.

e. Scan inhibit switch. The data encoder event register indicated that the scan inhibit switch functioned as designed when the pin-puller, which latched the science platform, was fired at spacecraft injection. The subsequent successful operation of the scan actuator verified this conclusion.

f. Science cover. On February 11, 1965, the single-shot science cover solenoid, initiated from the PCA, successfully unlatched the science cover. The proper deployment of the cover was verified by a change in the spacecraft identity Telemetry Channel 414 data number, and by a decrease in the platform temperatures. Temperature change on the scan platform caused two data number changes in the Channel 414 resistance measurement at other times during flight.

3. Recommendations

Based on the flight performance of the structures and mechanical devices, it is recommended that although the scan actuator components were designed to function normally in hard vacuum, additional effort be directed toward finding improved methods for sealing actuators.

B. Telecommunications

1. General

The *Mariner IV* telecommunications system was designed to perform three functions:

1. Track the angular position, radial velocity and range of the spacecraft to enable the FPAC and tracking data analysis areas of the SFOF to compute the parameters of the spacecraft trajectory.
2. Provide engineering telemetry data from the spacecraft to evaluate spacecraft performance. Provide scientific telemetry data from the spacecraft to permit scientific evaluation of the mission and to transmit stored television pictures of the planet Mars.
3. Provide ground command capability to the spacecraft as required.

These functions must be performed from launch to at least encounter plus 20 days, including a trajectory-correction maneuver, providing the trajectory and maneuver meet the constraints set forth in the functional specification.

2. Telecommunications System Performance

A review of flight data indicates that all functional requirements have been met, such as:

a. Tracking. Tracking data were received by the DSIF 24 hr a day after *Mariner IV* launch in either the one-way or two-way mode, except for short periods when *Ranger* tracking requirements took precedence over *Mariner* requirements. Using doppler and angle-tracking data from DSIF stations, FPAC personnel were able to determine the course of the spacecraft and continually monitor the spacecraft trajectory to verify and improve their trajectory predictions. Initial tracking data after injection were used to determine the distance and time by which the spacecraft would miss the nominal aiming point, and to define a trajectory correction for their error. Following the trajectory-correction maneuver, tracking data were used to verify that the maneuver had occurred as planned and that the spacecraft was indeed on course.

An additional task was performed in support of the tracking function during the occultation experiment, which was designed to gather information on the Martian atmosphere by monitoring changes in radio signals from the spacecraft as it passed behind the planet.

b. Telemetry. During all periods when the spacecraft was being tracked, telemetry data were received from the spacecraft. Analysis of received signal level data from DSIF indicated that the telemetry threshold requirement of less than 1 bit error in 200 bits was met. Indeed, during the picture playback the theoretical bit error rate was estimated to be less than 10^{-5}. Although the telemetry link bit-error rate during the picture playback could not be experimentally evaluated, the extremely clean picture data were evidence of superior performance. During the mission, over 350×10^6 bits of data were sent to Earth by *Mariner IV* and these data were relayed to the SFOF for evaluation, logging, and further processing.

c. Command. From two days after launch, when the first command was sent to the spacecraft, until the end of the TV picture playback, a total of 56 commands were sent to *Mariner IV*. In addition, 19 more commands were sent during the postencounter phase, bringing the total to 85 by the end of Phase I of the mission. All of these commands were successfully received and executed by the subsystems to which they were directed. Analysis of spacecraft received signal level telemetry at the time of command transmission indicates that the command threshold bit error rate of 10^{-5} was met for all commands. Since all commands were correctly received and executed, it can be inferred that no bit errors occurred.

During operational planning for the encounter phase, it became obvious that extremely accurate command timing would greatly enhance the value of encounter sequence commands. By utilizing the *timed start* mode of the ground command subsystem it was possible to accurately time the initiation of commands and hence their arrival and execution times on the spacecraft. The accurate positioning of the scan platform for the TV picture-taking sequence was evidence of the success of this technique.

d. Communications capability. Theoretical performance predictions were made for the *Mariner IV* mission using a summation of all known gains and losses between transmitters and receivers to generate curves of predicted received-power levels at the spacecraft and at the ground stations. By comparing predicted signal levels at any time during the flight with system thresholds, it was possible to define a performance margin as the amount by which the predicted nominal exceeded threshold. In order to reasonably assure that the telecommunications system performance would not decrease below the threshold during the mission, the design criterion was such that the performance margin must exceed the linear sum (in db) of the known system adverse tolerances. This criterion was met for all required phases of the mission for the command function, and for all phases except a short transition period between low- and high-gain antenna coverage for telemetry.

It should be noted that even during the period when the telemetry system adverse tolerances were not covered, the performance margin never went below nominal threshold. Thus in this region the telemetry

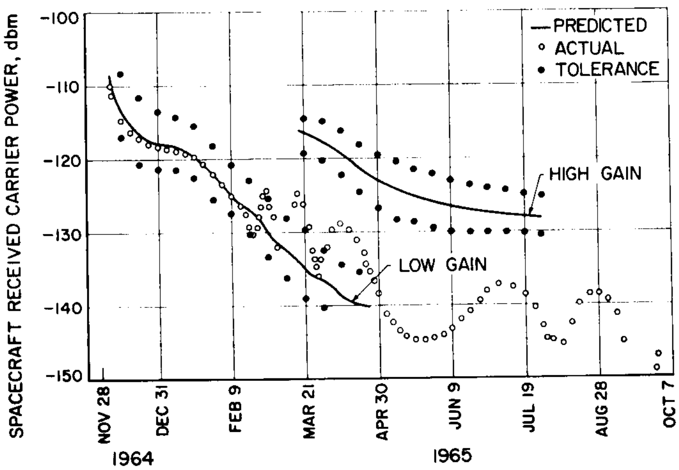


Fig. 6. Uplink performance via low-gain antenna

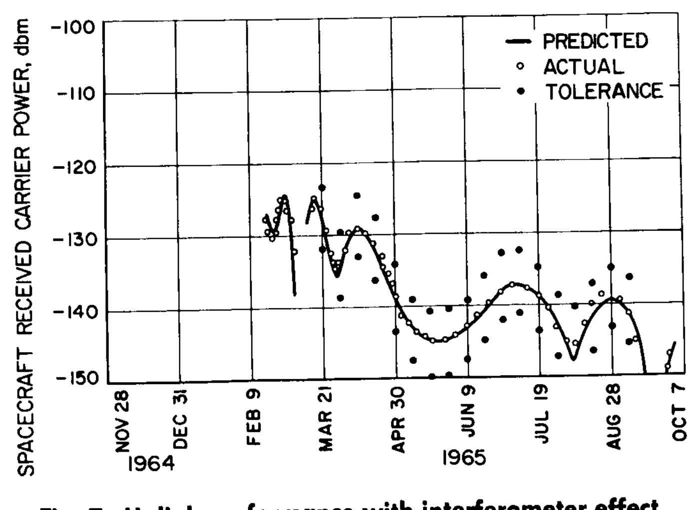


Fig. 7. Uplink performance with interferometer effect

system performance would be marginal, depending on the way in which the tolerances break. This was a result of an antenna position compromise that had to be made between midflight performance and encounter performance, sacrificing a short period in midflight to more adequately cover the encounter.

Uplink performance. During the *Mariner IV* mission, all signals transmitted from Earth to the spacecraft were received via the spacecraft low-gain antenna. Theoretical received signal level predictions are shown in Fig. 6 and 7 for the Earth to spacecraft link. Figure 6 shows predicted signal levels and the associated tolerances

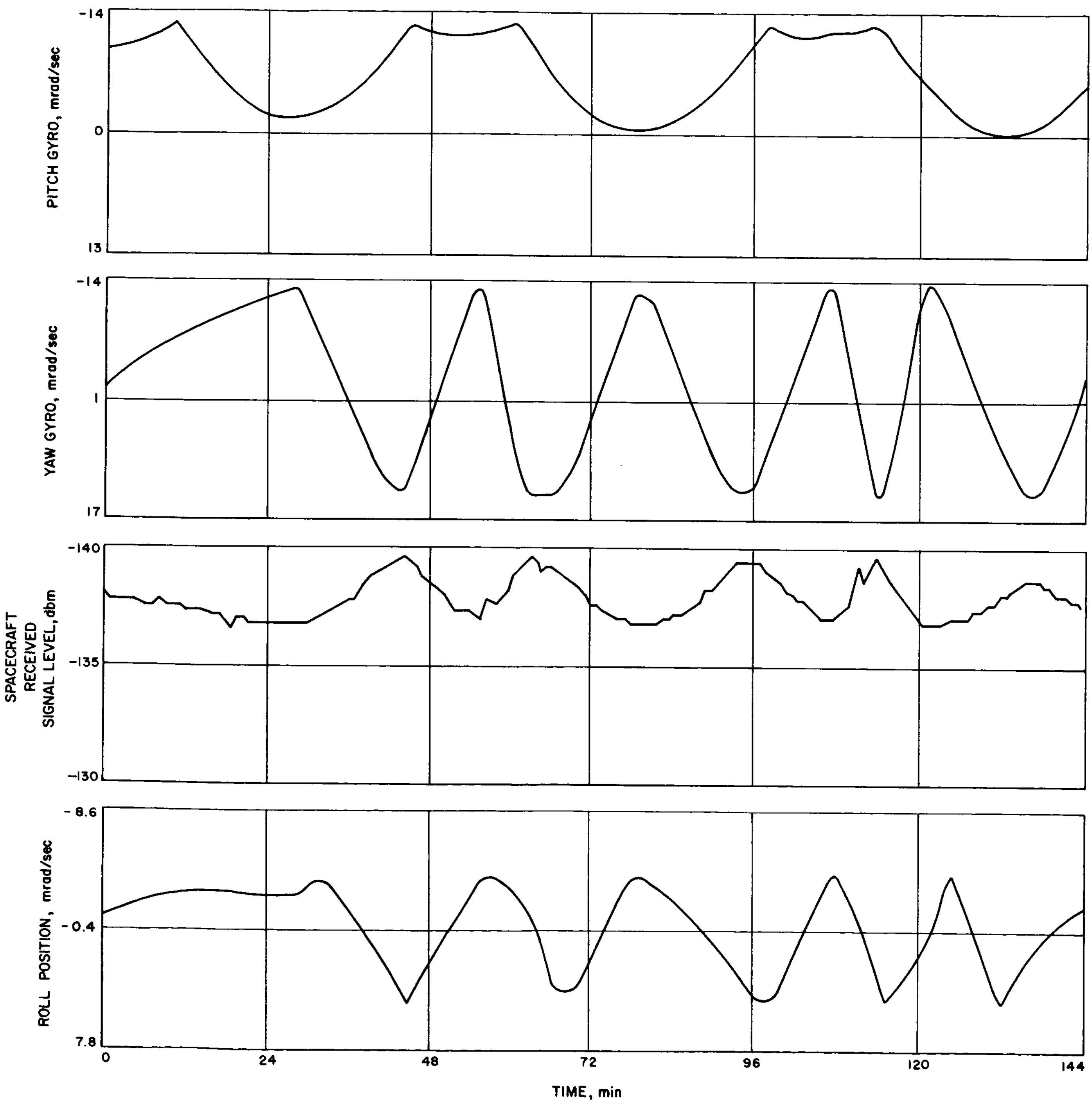


Fig. 8. Spacecraft attitude effects on uplink performance

during the portion of the flight when the spacecraft-to-Earth vector fell on the low-gain antenna alone. Figure 7 shows the same predictions during the period when the spacecraft-to-Earth vector intersected both the low- and high-gain antennas, and shows the resulting radio-frequency interference pattern. For comparison with predictions, received carrier power data have been taken using a calibrated AGC voltage on the spacecraft. Figures 6 and 7 show the recorded signal level data on curves of predicted data. The agreement is quite good for all phases of the flight.

Data points as shown on Fig. 6 and 7 are based on spacecraft AGC averages over a long period of time, since the instantaneous value at any given time may be subject to several possible perturbations. Most significant among possible perturbations is the variation in the exact point on the antenna pattern seen by the Earth due to attitude control limit cycles. Since the slope of the antenna-gain-vs-look-angle curve is very steep for many places on the antenna pattern, small variations in look angle can be translated into significant variations in spacecraft received-signal level, Fig. 8. Other items contributing to instantaneous variations in AGC telemetry were noise on the AGC telemetry, variations in DSIF transmitter power output, and variations in DSIF transmitter frequency.

Downlink performance. During the first 96 days of the *Mariner IV* mission, spacecraft-to-Earth communications were via the spacecraft low-gain antenna. A switch was made to the high-gain antenna on March 5, 1965, and the spacecraft continued to transmit via the high-gain antenna until October 1, 1965, when it was commanded back to the low-gain antenna as a condition for the cruise around the Sun. The spacecraft transmitted via a cavity amplifier for the launch and for the first 15 days of the mission. On December 13, 1964, a commanded switch to the TWT amplifier was made.

Figure 9 shows predicted received signal levels for the spacecraft-to-Earth link for the *Mariner IV* mission. Also shown in Fig. 9 is DSIF-received carrier-power data taken during the mission. These data points represent averages, over several station passes, of AGC data taken at DSIF 11. It should be noted that the data points shown in Fig. 9 may not, in many cases, correspond to signal levels reported by the station during the flight. The station-reported signal strengths have been corrected for known calibration errors in order to present a true picture of downlink performance.

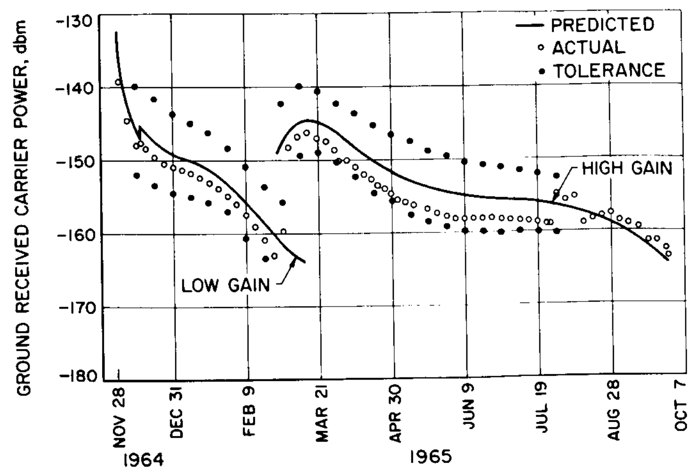


Fig. 9. Downlink performance

e. Summary. Performance of both the uplink and downlink averaged near nominal predicted values and within the assigned tolerance limits. Telecommunication system design criteria for the *Mariner* mission were thus met, with the conclusion that all parameters significantly affecting the *Mariner IV* telecommunications system had been considered, and the functional requirements as defined for the *Mariner IV* telecommunications system had been fulfilled.

3. Radio Subsystem

a. Description. The radio subsystem for the *Mariner IV* spacecraft was required to receive either a modulated or an unmodulated RF carrier transmitted from ground stations of the DSN, demodulate command and ranging signals if present, coherently translate the frequency and phase of the RF carrier, modulate the translated carrier with telemetry and ranging signals, and retransmit it back to Earth. Figure 10 is a block diagram of the spacecraft radio subsystem.

The received signal was a 2116 Mc carrier that could be modulated by a composite command signal and/or a coded ranging signal. This signal was demodulated by the automatic phase control (APC) receiver, which tracked the carrier signal in a phase-locked loop. If command or ranging modulation were present, it was sent to the command detector or the ranging channel, respectively. When the receiver was phase locked to the receiver carrier, it generated a filtered phase reference for the transmitter exciter which was coherent with the received carrier. The phase of the transmitted signal was thus related to the phase of the received signal by the constant ratio of 240/221. The signal generated by the

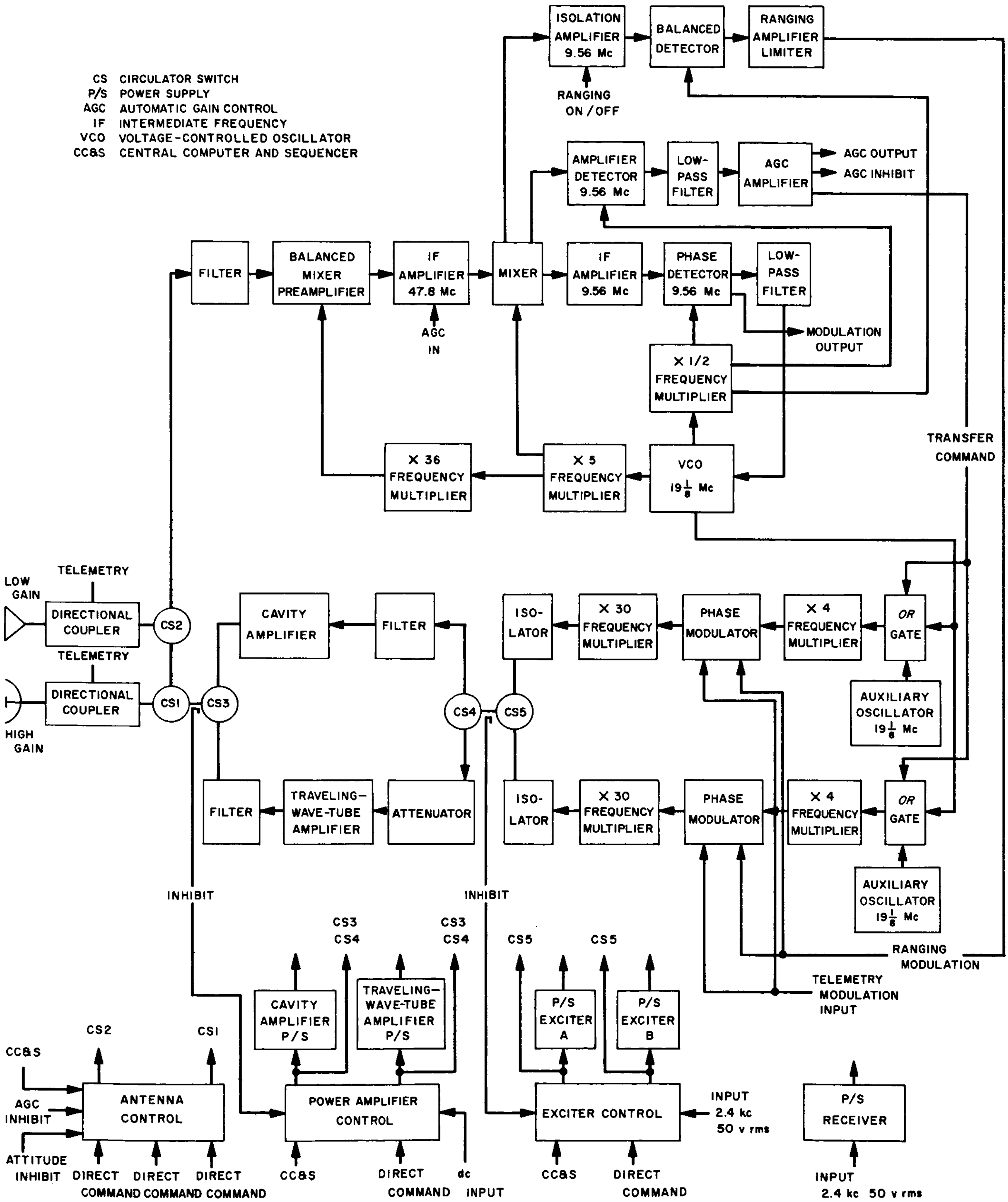


Fig. 10. Radio subsystem block diagram

transmitter exciter was phase modulated continuously by a composite telemetry signal and by a ranging signal if ranging was turned on.

When the ground stations were not transmitting to the spacecraft, the spacecraft transmitter exciter was controlled by an auxiliary crystal oscillator. This provided a capability for noncoherent one-way tracking of the spacecraft and ensured that spacecraft telemetry would be available even when the ground stations were not in uplink lock with the spacecraft.

To provide high reliability, redundant exciters, power amplifiers, and power supplies were incorporated into the transmitter. Either exciter could be used with either power amplifier through a circulator switching network. Similarly, either power amplifier could be used for transmitting through either antenna within the restrictions of the spacecraft logic. The control of switching between power amplifiers and exciters was provided by either ground command or on-board failure detection logic. For ground command control, the receipt of the appropriate direct command caused the control unit to transfer power from the active to the inactive element and to reverse all necessary input-output functions. In the case of switching by onboard failure detection, power monitors sampled the level of both the exciter and power amplifier RF power outputs. If the output power dropped below a preset level, the next cyclic pulse

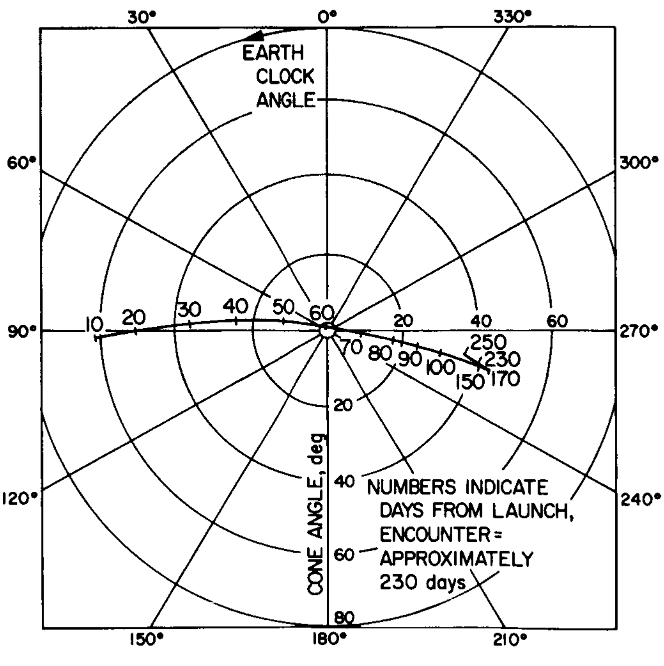


Fig. 11. Spacecraft-to-Earth vector diagram

(CY-1) from the spacecraft CC&S caused the control unit to switch the offending element.

The *Mariner IV* RF power amplifiers formed a hybrid configuration consisting of one cavity amplifier and one TWT amplifier. This configuration was chosen as the best system to fulfill mission objectives with available hardware and technology. The cavity amplifier was used during the launch phase in a low-voltage mode to prevent arcing when the spacecraft pressure passed through the critical region as the spacecraft evacuated in the vacuum of space. As the spacecraft separated from the *Agena* stage of the launch vehicle, the cavity amplifier power supply was switched to the high-voltage mode, and the cavity amplifier began transmitting at full power. It was not possible to use the TWT amplifier during this period since it had no low-voltage mode and was not packaged to operate in a partial vacuum. Several days after launch, a switch to the TWT amplifier was directed by ground command and this configuration was maintained throughout the balance of the mission.

With two antennas and the associated circulator switches, three transmitting and receiving modes are available:

1. Transmit low-gain, receive low-gain
2. Transmit high-gain, receive high-gain.
3. Transmit high-gain, receive low-gain

These modes provided the required coverage during all phases of the mission. Selection of the proper mode for a given time in flight was by CC&S command with ground command backup. Failure mode switching was also available to switch the spacecraft receiver to the low gain antenna should an automatic roll reacquisition be necessary and between the two antennas in the event of uplink communications failure.

By designing antenna pattern coverage to take advantage of the unique characteristics of the Mars 1964 minimum energy trajectories, considerable savings in spacecraft weight and complexity were realized. Figure 11 shows a plot of the spacecraft-to-Earth vector for a typical trajectory across the spacecraft coordinate system. During the latter portion of the flight, when it was necessary to use a high-gain antenna to communicate over the long ranges involved, the angular variation of the Earth track was small. This permitted the use of a fixed, high-gain antenna and enabled spacecraft designers to eliminate the heavy and complex antenna pointing equipment required to steer a moveable antenna.

b. Performance. The radio subsystem aboard the *Mariner IV* spacecraft operated continuously with no evidence of any malfunction or degradation of performance from time of launch to completion of the mission. The anomalies that were observed were well understood and are reported in the order that they were recorded.

Traveling-wave tube helix current variation. Following the switch to the TWT amplifier by ground command on December 13, 1964, the TWT operated as expected with one exception. Beginning December 31, telemetry indicated that the TWT helix current was varying more than anticipated, and was generally tending to increase. It finally stabilized about 0.4 ma higher than the initial value. Discussions with the manufacturer indicated that helix current variation was a known characteristic of TWTs and should be considered normal. Subsequent tests conducted at JPL with two TWTs from the same lot that *Mariner IV* used confirmed this.

Continued observation of helix current after its stabilization value (about 7.7 ma) was reached has shown no further variations, except those attributable to decreasing spacecraft temperature.

Interferometer effect. The *Mariner IV* radio subsystem was so designed that the transmitter or receiver could be switched to operate with the high-gain antenna or the low-gain antenna on the spacecraft. The switching operation was done with circulator switches. Because the *effective* isolation provided by the circulator switches was finite, more than one path was possible for transmission of RF energy through the subsystem. The net result of the finite isolation was that transmission or reception of the same signal over both antennas on the spacecraft was possible. At a given nonlinear element, constructive or destructive interference occurred depending upon the relative amplitudes and phases of the interfering signals. Because the two antennas were many wavelengths apart, the relative phase between each antenna signal, and therefore the relative amplitude of the composite signal, varied with look angle to the spacecraft. The resulting amplitude variation with spacecraft cone and clock angle produced the interferometer pattern shown earlier in Fig. 7.

The leakage mechanism in the RF circuits was a combination of actual circulator isolation and circulator termination mismatch return loss. That is, an unwanted RF signal may wind up at a circulator port via wrong-way circulation and via reflection from a mismatched termination with right-way circulation. The actual unwanted signal magnitude at the port in question was then dependent upon the relative magnitudes and phase difference between the signals via the two paths. Hence, the leakage was a function of circulator isolation, termination mismatch, and interconnecting line lengths.

Early in the *Mariner* radio subsystem design, the problem of circulator switch leakage was recognized, but the decision was made to continue the design. For the planned portion of the flight where the interferometer pattern occurred — over the solid angle of the high-gain main beam — the spacecraft could have the transmitter and receiver connected to the high-gain antenna. The large gain, coupled with normal isolation to the low-gain antenna, would make the interferometer effect for this mode negligible.

Mariner IV left Earth in the transmit low-gain, receive low-gain mode. Prelaunch predictions based on an assumed isolation of 25 db indicated that +7 db to −11 db maximum variations from the nominal low-gain patterns could exist over that portion of the low-gain pattern encompassing the high-gain antenna main beam (around 38-deg cone angle). Until the time that the transfer from low-gain transmitting to high-gain transmitting occurred (MT-5, 28-deg cone angle), the maximum variations encountered were approximately ±4 db from nominal for the uplink signal and approximately ±3 db for the downlink signal.

After the mode switching occurred, the interferometer effect ceased to exist on the downlink because, for the new antenna configuration, the interfering signal was negligible compared with the primary, high-gain antenna signal. In uplink communications, however, the interferometer effect remained quite pronounced, as expected, since the spacecraft receiver antenna configuration had not changed, although the phase and amplitude relationships of the interfering signals did change. Subsequent mapping of the antenna gains allowed the received carrier power data and the spacecraft signal (2116 Mc) to be fitted to analytical interferometer patterns, yielding uplink communications predictions for the balance of the mission and indicating that the effective isolation from the high-gain antenna to the receiver was on the order of 12.5 db for the transmit high-gain, receive low-gain mode.

The interference effect upon the uplink could have been avoided entirely by transferring the receiver to the high-gain antenna at this point. It was desirable, however, to remain on the low-gain antenna as long as possible to permit command capability in the event that roll orientation was lost. The logic which would normally command a transfer of the receiver to the low-gain antenna if roll acquisition were lost was disabled when DC-15 was in effect.

Fortunately, the interferometer pattern was so disposed about the spacecraft that a pattern maximum occurred during encounter. At look angles other than over the high-gain antenna main beam, the normal low-gain antenna pattern still existed. Thus, the interferometer pattern provided higher gain than normal for the critical portion of the mission and yet had wide coverage for command reception capability in the event the spacecraft lost attitude control.

Frequency shifts at CY-1 events. Both receivers at DSIF 42 dropped lock momentarily coincident with the CC&S cyclic command (CY-1) on February 27, 1965. Analysis of the data showed that these transients were present every time a CY-1 occurred while the radio subsystem was in one-way lock. A series of tests on the *Mariner* proof-test model spacecraft provided the explanation.

The first CY-1 after a two-way tracking period changed the state of a flip-flop in the radio control circuitry. The flip-flop, which derives its power in common with the auxiliary oscillator, draws more current in one state than in the other. The dc power-supply voltage, in turn, is slightly current dependent, so that a change of state of the flip-flop results in a slight change of voltage to the auxiliary oscillator, causing a shift in spacecraft transmitted frequency. Ground testing indicated that a change of approximately 10 mv in the power-supply voltage was to be expected, so that the frequency shift at S-band would be about 30 cps.

When the spacecraft is in two-way lock, the radio operates from the voltage controlled oscillator (VCO) which does not have a dc power supply in common with the current unbalanced flip-flop. Thus, the slight shift in frequency is not observed when in two-way lock. It was not observed during prelaunch system testing because sufficiently sensitive instrumentation to measure such small frequency change is not part of a normal test configuration.

Best lock frequency investigation. The best lock frequency is the frequency onto which the spacecraft receiver APC loop will lock in minimum time. It also corresponds to the average frequency at which the receiver VCO operates when out of lock. If this frequency is known closely enough, the average time to acquire two-way lock can be minimized.

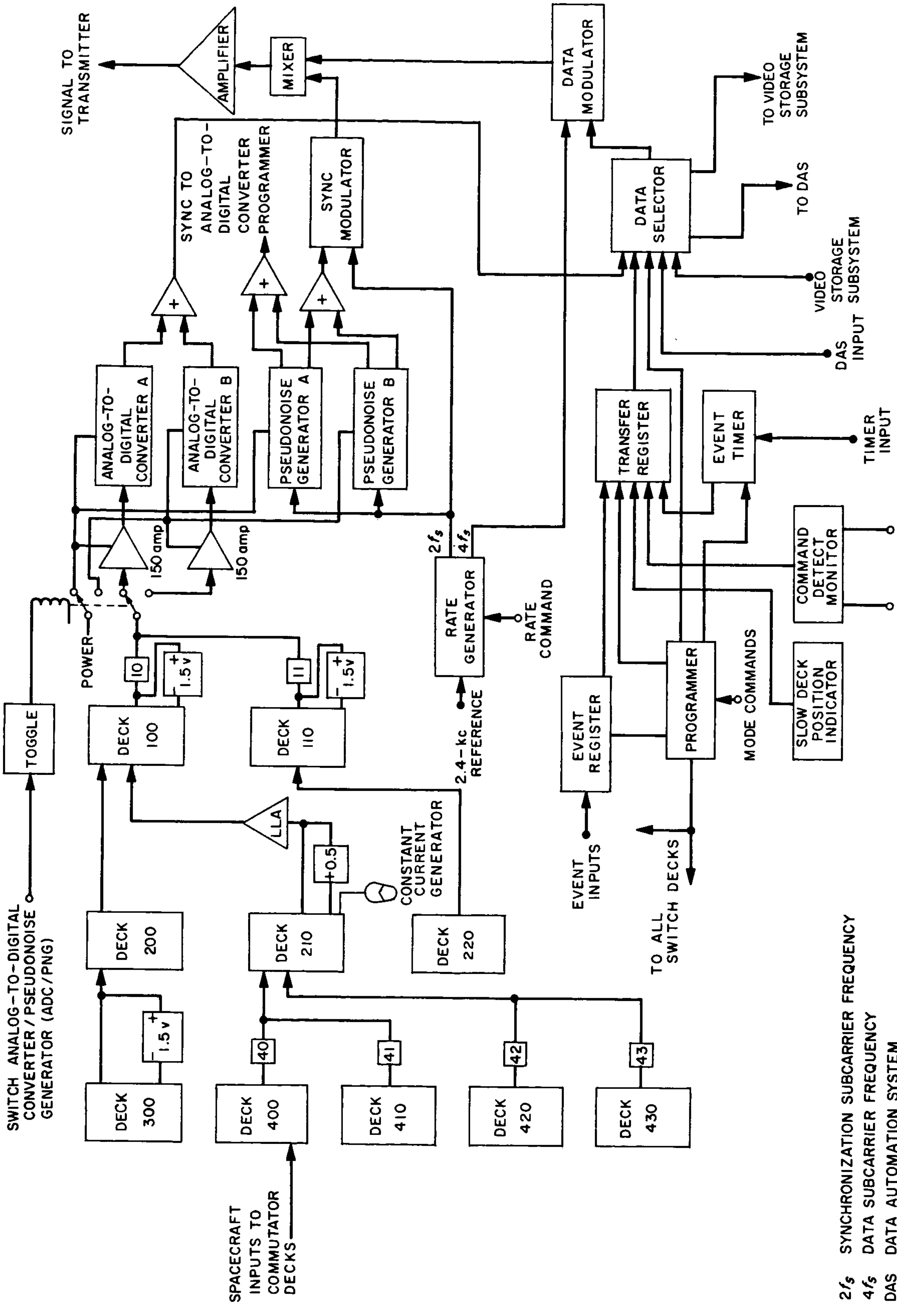
Due to aging and temperature differences, the best lock frequency may change from the prelaunch value. For this reason, two methods were proposed and were used to continuously update the estimate of the best lock frequency.

The first method involved the direct measurement by DSIF of the frequency giving the shortest lock-up time, and the only requirement was that the DSIF stations modify slightly the procedures for acquiring two-way lock. The second method consisted of recording spacecraft static phase error (SPE) telemetry values when out of lock and then averaging the spacecraft SPE readings over a sufficiently long period of receiver out-of-lock conditions. The average telemetry value was then converted to spacecraft VCO frequency from the appropriate calibration table. Using the updated best lock frequencies, ground stations repeatedly obtained two-way lock with the spacecraft within a few seconds after the signal arrived at the spacecraft. Prior to obtaining these data it occasionally took more than an hour to obtain two-way lock.

4. Data Encoder Subsystem

a. Description. The *Mariner IV* data encoder subsystem accepted engineering data from 90 analog channels, time multiplexed it into a predetermined sequence, and converted it into 7-bit binary words. It also accepted digital data from science, video storage, and command subsystems and time multiplexed (commutated) this with the engineering data. The data encoder subsystem generated a cumulative count of specific spacecraft events, and a cyclic, binary, pseudorandom code from which bit and word sync were obtained. A composite telemetry signal consisting of telemetry data bi-phase modulated on a square wave subcarrier and linearly added a synchronization subcarrier was generated and sent to the RF phase modulator. Two data rates were provided in order to take advantage of the higher signal strengths available during the early part of the flight to support a higher bit rate. The lower rate was used after the Earth to spacecraft range was so large that ground received signal level decreased to the point where excessive bit errors were introduced at the higher bit rate.

Figure 12, a block diagram of the *Mariner IV* data encoder, shows the main functional elements. Analog



2fs SYNCHRONIZATION SUBCARRIER FREQUENCY
4fs DATA SUBCARRIER FREQUENCY
DAS DATA AUTOMATION SYSTEM
LLA LOW-LEVEL AMPLIFIER

Fig. 12. Data encoder subsystem block diagram

data from telemetry transducers in the various spacecraft subsystems arrives at the solid state commutator where it was time multiplexed into a predetermined sequence and converted to a pulse-amplitude-modulated (PAM) signal for presentation to the analog-to-digital converter (ADC). The commutator also provided unique words of frame synchronization information for the ground decommutators to use to identify the various data words.

In the ADC the sequenced PAM data from the commutator was converted into 7-bit binary words which in turn went to the data selector for insertion into the data stream.

To obtain bit and word sync for telemetry, a cyclic 63-bit pseudorandom code was used. This code was generated by a pseudorandom code generator which was driven by the sync subcarrier frequency of $2f_s$, and was synchronized with data and sync subcarriers. Bit and word sync were derived through a series of gates which generated pulses that synchronized the commutator, the ADC, and all digital data inputs.

To provide bit and word sync information to the ground equipment, the pseudorandom code was phase-shift key (PSK) modulated onto a sync subcarrier. The ground demodulator generates a local pseudorandom code identical to the code received from the spacecraft and steps the two codes past each other until they are in phase. Word gates like those on the spacecraft then derive accurate word and bit sync pulses.

To obtain the $2f_s$ and $4f_s$ frequencies used for sync and data subcarriers and to provide basic subsystem timing, the spacecraft 2400-cps power is divided down via a digital counter chain in the frequency divider. Data bit rate is also determined in the frequency divider by switching in additional counters to obtain the lower bit rate. This switching function was controlled by a CC&S command with ground command backup. At launch the data rate was 33⅓ bps, with the CC&S commanding a switch to 8⅓ bps on January 3, 1965.

In case of catastrophic failure of an ADC or pseudorandom code generator, a redundant ADC/pseudorandom code generator chain was available. The redundant combination could have been activated by a ground command which removes power from the failed unit and applies power to the standby unit.

In addition to the processing of analog data, several digital data inputs were accepted by the data encoder. The spacecraft command system provided an input from the command detector indicating whether or not the command detector was in lock and the frequency of the command detector VCO. Event registers that maintained a cumulative count of spacecraft events provided a 7-bit digital output which was commutated together with the engineering data words. The science subsystem provided a digital input consisting of data from the scientific instruments on board the spacecraft, and television picture data from the video storage subsystem were input during the post encounter period when pictures were being played back.

Four data modes were available, either by CC&S command or ground command. They were: 1) engineering data only, 2) engineering and science data, 3) science data only, and 4) stored video and science data with short blocks of engineering data every 9 hr. Mode 1 was used during checkout of the spacecraft and critical maneuvers where engineering data are of primary interest. Mode 2 was used during the cruise phase where both science and engineering data were required. Mode 3 was used during planetary encounter to obtain maximum real time scientific data and Mode 4 was used during the post encounter playback of stored data.

b. Performance. The data encoder subsystem operated normally throughout the flight. As predicted before launch several data encoder deck skips occurred coincidentally with changes in the spacecraft power profile. These skips were due to the susceptibility of the commutator to electrical transients, and had no effect on the mission other than a change of reference times for science frame count and deck sync.

Shortly after launch, while the spacecraft was passing through the Van Allen radiation belt, there were several deck skips that are as yet unexplained. However, after that time, all deck skips were associated with changes of spacecraft power or other electrical transients.

Deck skips and resets. Deck skipping is described as a skipping of some channels by the commutator. Specifically, the effect was observed at frequent intervals between L + 157 min and L + 180 min on *Mariner IV* and at L + 207 min on *Mariner III*. Commutator skipping at these times was not fully explained, although several theories were advanced, e.g., radiation effects, outgassing effects, and plasma probe high-voltage arcing. Since these specific times, occasional skips have been noted on *Mariner IV* which have been correlated with arcing of the plasma probe on the spacecraft or with changes in the spacecraft power profile. Table 7

Table 7. Data encoder deck skips and resets

Date	Time, GMT	Event
November 1, 1964	16:59:00	Rate 3/4 skipped 409, 410, then reset and advanced one count.
	17:00:28	Rate 1 skipped at word 119 which made science PN code start one word only.
	17:00:40	Rate 2 reset from 208, skipping word 209.
	17:00:53	Rate 3/4 skipped from word 411 to 400.
	17:04:00	Rate 2 skipped position 206 and 207.
	17:04:13	Rate 2 reset to 200 from 207, skipping 208 and 209.
	17:04:25	Rate 3/4 reset to 400 from 201, skipping 402 through 419.
	17:04:37	Rate 1 sync occurred one word early. 100 deck reset at one of the two 8 positions in the science deck. Rate 2 reset to 200 from 201, skipping 202 through 209.
	17:04:50	Rate 3/4 reset to 410 from 400, skipping 401 through 409.
	17:05:15	Rate 1 sync occurred one word early. Again a 100 deck reset at position 8.
		Rate 2 reset to 200.
	17:05:27	Rate 3/4 reset to 400.
	17:06:05	Rate 1 sync occurred one word early. 100 deck reset at position 8.
		Rate 2 reset to 200 from 203, skipping 204 through 209.
	17:06:17	Rate 3/4 reset to 410 position.
	17:06:42	Rate 1 sync occurred one word early. 100 deck reset at position 8. Rate 2 reset to 200 from 202, skipping 203 through 209.
	17:07:18	Rate 3/4 reset to 410 position.
	17:07:43	Rate 1 sync occurred one word early. Probably 100 deck reset at position 8.
	17:07:55	Rate 3/4 reset to 400 position.
	17:09:28	Rate 2 skipped 208, going from 207 to 209. Rate 3/4 reset to 410 position.
	17:11:21	Rate 2 skipped 208, going from 207 to 209. Rate 3/4 reset to 400 position.
	17:15:33	Rate 2 reset to 200 from 208, skipping 209.
	17:16:36	Rate 2 skipped 205, going from 204 to 206. Rate 3/4 reset to 410 position.
November 1, 1964	17:18:29	Rate 2 reset to 200 during first 9 words of engineering frame since 205 okay but 220 occurred during 225 time indicating a reset from position 5 to 0 on rate 2. Rate 3/4 reset to 400 position.
	17:19:20	Rate 2 skipped position 204 but not 214 and 224. It read out 205, 214, 224. Indicates three rate 2 decks out of step.
	17:19:33	Rate 2 read out 206, 215, 224. Indicates three rate 2 decks out of step.
	17:19:45	Rate 2 read out 207, 216, 226. Indicates three rate 2 decks out of step.
	17:19:58	Rate 2 read out 208, 217, 227. Indicates three rate 2 decks out of step.
	17:20:11	Rate 2 reset to 200, skipping 209.
	17:20:23	Rate 2 reset to 200; i.e., 200 deck sync again.
	17:20:35	Rate 3/4 reset to 410 position.
	17:21:26	Rate 2 skipped position 205 and succeeding channels were incorrect until the deck was reset at Channel 209. Channels 219 and 229 were skipped as a result of the reset.
	17:22:28	Rate 3/4 reset, 410 skipped.
December 6	02:03:04	Rate 2 reset, skipped 209; 400 deck not affected.
	06:21:34	Rate 3/4 reset, skipping from 416 to 401. Implied reset before 06:21:22 GMT.
	09:02:14	Rate 2 skipped 207, read out 208, 217, 227; 400 deck reset. Next reading 201 = 90. Skipped 418, 419, and 400.
	13:29:46	Rate 3/4 reset, skipped 408, 409, and 410.
	13:42:10	Rate 3/4 and rate 2 reset, skipped 209.
7	03:41:08	Rate 1 reset, skipped 119 and 206.
	06:48:36	Rate 3/4 reset.
February 11, 1965	03:53:15	Rate 2 reset to 200 from 206 and Rate 3/4 reset to 400 from 416 (DC-26 battery charger off transient).
	06:54:43	Rate 1 reset, Rate 2 reset to 200 from 203, and Rate 3/4 reset to 410 and 407 (DC-25 encounter science on transient).
	10:27:08	Rates 1, 2, and 3/4 reset (DC-26 battery charger off transient).

Table 7. Data encoder deck skips and resets (cont'd)

Date	Time, GMT	Event
February 11, 1965	17:02:05	Rates 3 and 4 reset coincident with CC&S event MT-1.
June 10	14:40:32.8	Data encoder rate 3/4 skipped from 409 to 401.
June 30	03:28:37	Data encoder rate 2 skipped from 209 to 200, then reset to 201. Rate 3/4 skipped from 408 to 410 and 308 to 300.
September 2	06:54:32	Rate 1, 2, and 4 decks skipped coincident with battery charger turn off.

Table 8. Deck skips and resets during SAF testing

Date, 1964		Spacecraft
February		MC-1
March	17	
April	22	
April	23	
May	2	
	18	
	25	MC-2
	26	MC-1

lists the deck skips and resets observed during the mission.

The commutator is composed of semiconductor switches arranged in groups of 10, called decks. The decks are grouped to form frames, and each frame is sampled at a different sampling rate. As an example, the fast frame is composed of two decks which are sequenced in parallel. The outputs are switched alternately, so that a length-20 frame is formed. Each deck is sequenced by a sequencer, which turns each switch on and off in order. These sequencers are essentially free-running, but must run in synchronism. To ensure that the sequencers stay synchronized, a synchronizer is used. The output of the last position of each sequencer is connected to the synchronizer. The synchronizer produces an output pulse whenever it receives an input pulse from either of the two sequencers. Thus, if one sequencer gets ahead of the other, it will reach the last position, thereby regaining synchronism.

In most cases, the deck skip had been caused by premature triggering of the synchronizer, which in turn had reset the sequencers to the first position. The result was the skipping of the remaining measurements on the deck, depending on where the reset had occurred. A few of the deck skips had been caused by the actual skipping of the sequencer from the first to the second or third position in a sampling sequence. This had been due to transients on the sequencer clock line.

This anomaly had been observed occasionally since February, 1964, when the first deck skip was observed in the Spacecraft Assembly Facility (SAF) on MC-1 PTM. Table 8 lists the deck skips and resets observed in SAF system testing.

Analysis of these anomalies showed that in every case a deck skip was correlated with a fairly severe spacecraft power transient, e.g., switching gyros on and off, boost-share power mode, science on and off, and high-voltage arcing in the science instruments. Further lab testing showed that the data encoder synchronizer gave an output pulse with either a 0.7-v, 2-sec pulse on the input, or a 6-v pulse on the +6-v power or ground lines. It became obvious, then, that a deck skip was caused by the spacecraft inducing noise pulses into the data encoder of such magnitude as to produce transients on the synchronizer lines. This may be attributed in part, to the fact that as a telemetry processor, the data encoder is the center of the spacecraft signal ground tree and that it has approximately 150 signal connections routed via the upper ring harness.

Extra events. An extra event was observed in data encoder, Register 2, instead of the single event normally associated with CC&S command MT-9. The anomalous indication is believed due to the nonsimultaneous closure of the two relays associated with MT-9.

No further anomalous indications have been observed from the spacecraft data encoder during the flight. All encoder functions such as bit rate changes and mode switches have occurred as planned including the Mode 1/Mode 4 switching during picture playback.

5. Command Subsystem

a. Description. The spacecraft command subsystem is required to detect and decode any of 29 DC or three QC that may be sent to the spacecraft by the DSN ground stations. If the command received is a DC, a logical output must be provided to the appropriate subsystem for execution of the command. If the command received is a quantitative command, the magnitude and polarity information it contains are stored by the CC&S for use during the execution of a maneuver. When a command word is received, the command detector must be in lock or the command will be inhibited. The command words are designed in such a way that at least two bit errors must have occurred in order for a wrong command to be executed. Telemetry information is provided to the data encoder including: 1) command detector in lock or out of lock, 2) command detector VCO frequency, and 3) event pulse indication of command reception.

The command subsystem consists of two major functional units: the command detector and the command decoder. Also included is a transformer rectifier unit which converts the spacecraft 2.4-kc voltage into the ac and dc voltages required by the subsystem.

A simplified block diagram of the detector is given in Fig. 13. The detector receives the composite command signal subcarrier from the radio subsystem. The subcarrier contains the command word information (a sequence of binary digits) in the form of a biphase sinusoidal signal added to a synchronization (sync) signal. The sync signal is composed of a pseudonoise (PN) sequence Modulo 2 added to a square-wave signal at a clock frequency defined as the $2f_s$ frequency. The sinusoidal signal is also at $2f_s$ frequency.

The detector consists of two major channels where synchronization and command word detector functions are accomplished. The detector sync channel effectively correlates the incoming sync signal with a detector-generated PN sequence Modulo 2 added to a square-wave signal at f_s frequency. The correlation of the two PN signals results in the error signal in the detector phase

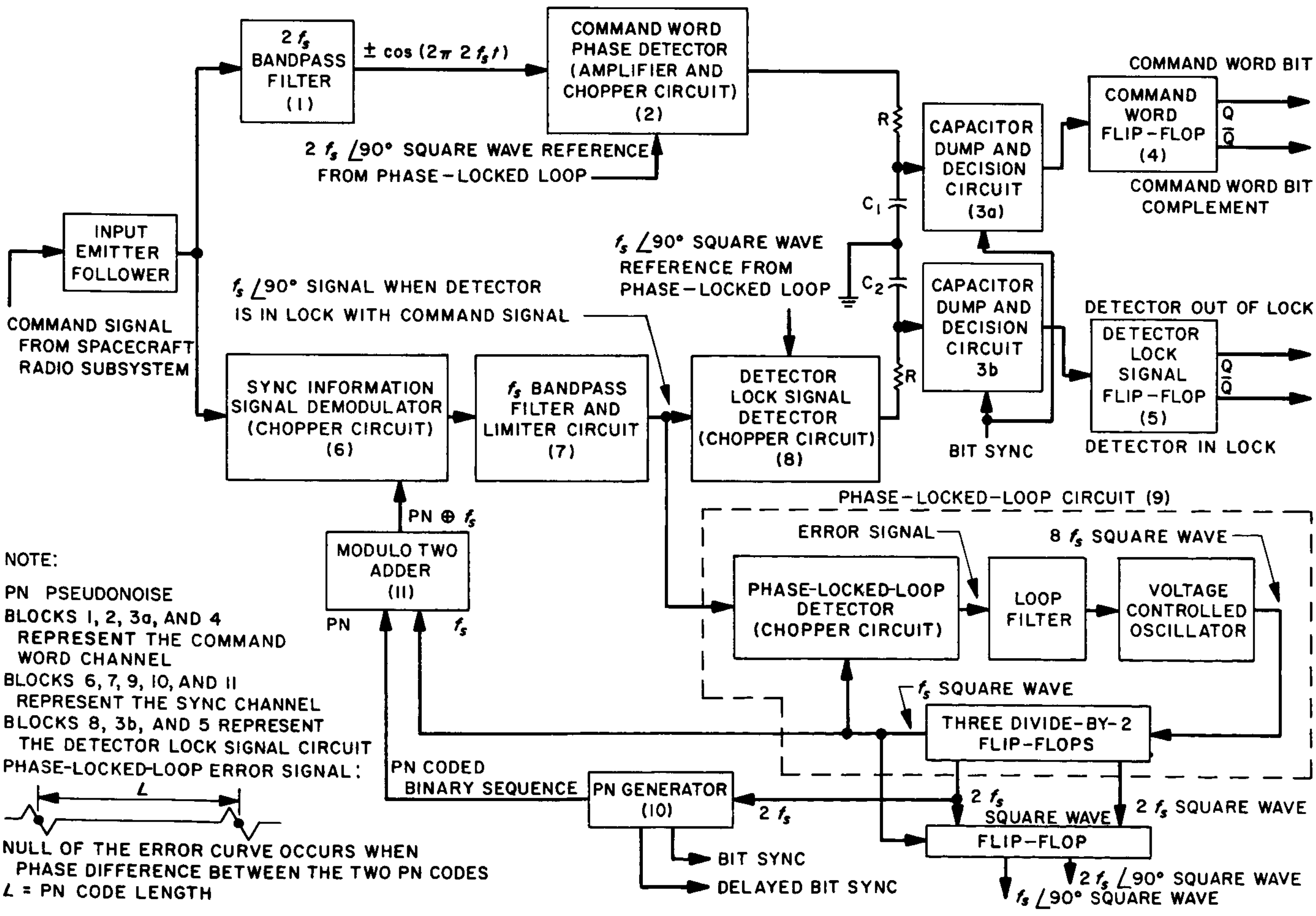


Fig. 13. Command subsystem command detector block diagram

lock loop (PLL) which drives the indicated VCO. To accomplish phase lock, a frequency offset is set between the transmitted and the detector $2f_s$ frequencies, which causes a continually changing phase shift between the two PN signals until correlation occurs. When the transmitted and the detector-generated PN sequences are in correlation, the null point of the error curve is achieved and the output of the VCO is in phase and frequency coherence with the transmitted signals. Any reasonable phase transient will now be tracked by the PLL which is in lock with the command signal.

The detector command channel filters the command word information signal from the composite commanded signal and recovers the command word bits by multiplying the signal with a phase reference from the detector sync channel. The multiplied signal is then applied to a matched filter (an effective integrator over the bit time) which optimally detects the transmitted bit level. The command signal is corrupted by noise assumed to be gaussian with a flat spectral density over the noise bandwidth of the matched filter. Threshold for the detector is defined at a bit error rate of 1×10^{-5}, and occurs at a composite command signal-to-noise ratio of 19.5 ± 1 db in a 1-cycle noise bandwidth. The reconstructed bits and bit sync pulses, emanating from the detector PN generator (PNG) each cycle of the PN sequence, are directed to the command decoder.

A lock signal is generated by the detector which effectively measures at each bit sync time the correlation between the transmitted and detector-generated PN sequences, compares the correlation output to a threshold bias signal, and makes a decision as to whether or not the subsystem was in lock over the preceding bit time interval. The detector lock signal is directed to the decoder and inhibits its function when an out-of-lock indication occurs.

A simplified block diagram of the command decoder is given in Fig. 14. Upon recognition of the decoder start bits, the decoder sequences a binary counter with the detector bit sync pulses. At the correct bit sync pulse, the decoder matrix is interrogated. Depending on the address bits stored in the decoder address storage flip-flops, one of the NAND gates is gated to close an output isolation switch (or switches) in the case of a DC. For a QC, the interrogation permits the QC bits to be

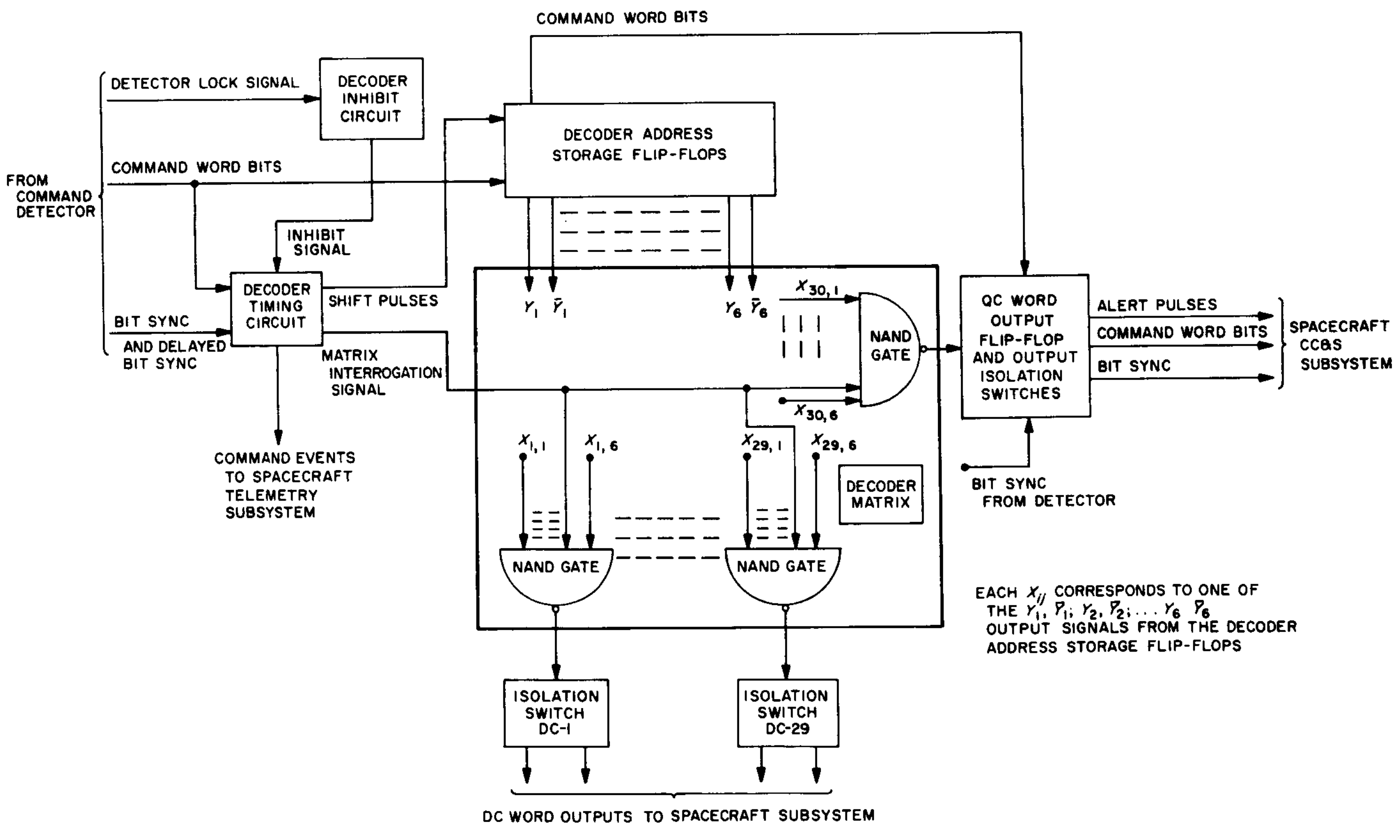


Fig. 14. Command subsystem command decoder block diagram

directed to the spacecraft CC&S subsystem along with QC bit sync pulses. When no QC bits are transmitted to CC&S, alert pulses are directed, one per second, to the subsystem to clear the QC storage registers.

If an out-of-lock indication occurs during decoder function, the decoder generates an inhibit signal which immediately stops the decoder command word processing and allows no further decoder functioning until 26 consecutive command zero bits are directed to the decoder after the lock signal indicates *in lock* once again.

The command word is comprised of 26 bits, as shown in Fig. 15. The first three bits of a command word (always 110) begin the command decoder function. Bits 4 through 9 are the command address bits. They completely identify which command word has been sent. The remaining bits constitute quantitative information bits. The *Mariner* command subsystem has the capability of processing and executing 32 command words. Twenty-nine of the command words are DC, which cause a momentary switch closure in the recipient subsystem or subsystem circuitry. DC makes use of only the decoder start and address bits of the command word. The remaining command words are QC which direct to the CC&S subsystem bits 9 through 26 representing either roll, pitch, or velocity information necessary for the two possible midcourse maneuvers.

Three command signals are directed to the spacecraft telemetry subsystem to be telemetered to Earth. The signals time-share the telemetry transmission channel and are sampled at a rate determined by the telemetry

DIRECT COMMAND FORMAT

COMMAND BIT NO.	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26
COMMAND BIT IDENTIFICATION	COMMAND DECODER START			COMMAND ADDRESS						ADDRESS PARITY See Note 1		BITS 12–26 HAVE NO SIGNIFICANCE IN DC'S. IN QC'S THEY FORM PART OF THE CC&S COMMAND. REFER TO QUANTITATIVE COMMAND FORMAT														
COMMAND BIT VALUE	1	1	0	VARIABLE								*ZERO* FOR DC'S; VARIABLE FOR QC'S														

QUANTITATIVE COMMAND FORMAT

		1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26
COMMAND BIT NO.		1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26
CC&S COMMAND BIT NO.										1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18
CC&S COMMAND BIT IDENTIFICATION										CC&S ADDRESS: See Note 2			CC&S ADDRESS: Register Selector		See Note 3 ODD PARITY	TIME VALUE See Note 5											See Note 4 POLARITY
QC COMMAND BIT VALUES	PITCH TURN	1	1	0	0	1	1	0	0	0	1	1	1	0		VARIABLE											
	ROLL TURN	1	1	0	0	1	1	0	0	0	0	0	0	1		VARIABLE											
	MOTOR BURN	1	1	0	0	1	1	0	0	0	1	0	1	1		VARIABLE											

NOTES:

1. COMMAND BITS 10 AND 11 ARE ADJUSTED TO ENSURE AGAINST SINGLE BIT ERRORS CAUSING AN INCORRECT COMMAND WORD OUTPUT.
2. COMMAND BITS 9 THROUGH 11 (CC&S COMMAND BITS 1 THROUGH 3) ARE NOT USED QUANTITATIVELY BY CC&S BUT ARE USED TO REMAIN COMPATIBLE WITH PREVIOUSLY DESIGNED HARDWARE (*MARINER* VENUS 1962).
3. COMMAND BIT 14 (CC&S COMMAND BIT 6) IS ADJUSTED IN QC'S TO GIVE AN ODD NUMBER OF *ONE* BITS IN COMMAND BITS 9 THROUGH 26 (CC&S BITS 1 THROUGH 18).
4. COMMAND BIT 26 (CC&S COMMAND BIT 18) MUST BE A *ONE* TO PRODUCE A CLOCKWISE (POSITIVE) SPACECRAFT ROTATION ABOUT THE SPECIFIED SPACECRAFT AXIS. A *ZERO* IN THIS BIT POSITION WILL RESULT IN A COUNTERCLOCKWISE (NEGATIVE) SPACECRAFT ROTATION ABOUT THE SPECIFIED SPACECRAFT AXIS. POLARITY BIT FOR MOTOR BURN COMMAND IS ALWAYS *ONE*.
5. COMMAND BITS 15 THROUGH 25 (CC&S COMMAND BITS 7 THROUGH 17) ARE A PSEUDOBINARY CODE REPRESENTATION OF THE TURN OR MOTOR BURN DURATION.

Fig. 15. Command word format

mode. A listing of the signals and a description of their information content follow:

1. Detector bit sync pulses are sent to the telemetry subsystem to be conditioned into information concerning the detector VCO frequency. Between telemetry data sample times, reference pulses at the spacecraft 2.4-kc frequency are counted in the interval between two consecutive bit sync pulses. The number counted is indirectly proportional to the VCO frequency. Thus, an average of the VCO frequency, over an approximate 1-sec interval, is telemetered each telemetry-data sample time. From this telemetry data, the proper ground command signal frequency may be ascertained so that minimum command lock acquisition time may be accomplished.

2. The detector lock signal is directed to the telemetry subsystem to be converted into one word of information regarding the detector lock condition. If the lock signal indicates *out of lock* for at least 1 sec in the interval between telemetry sampling times, the detector lock telemetry data will be at the binary state indicating out of lock at the next sampling time. If no out-of-lock indications occur, the binary state indicating *in lock* will result. The detector lock data indicates when the command subsystem is capable of detecting and processing the command signal.

3. Command event pulses are telemetered approximately 10 sec after the command decoder has recognized the decoder start bits of a command word. The event pulse indicates normal decoding timing function upon reception of the start bits. It is also normally coincident with a command address recognition and a DC execution by the decoder. In the case of a QC command, a second event pulse will be telemetered, immediately after the last QC bit is directed to the CC&S subsystem.

b. Performance. During the *Mariner IV* mission, the performance of the spacecraft command subsystem was excellent. A total of 85 commands was sent to the spacecraft through the completion of the first phase of the mission. Table 9 is a listing of commands and the times at which they were sent. Analysis of spacecraft telemetry indicated that all commands were received and correctly acted upon.

Table 9. Ground commands received by *Mariner IV*

Date	Time, GMT	Command
November 30, 1964	09:13:00	DC-21 (roll override).
	10:45:00	DC-21.
	10:57:09	DC-21.
December 4	13:05:00	QC-1-1 (pitch turn duration and polarity).
	13:10:00	QC-1-2 (roll turn duration and polarity).
	13:15:00	QC-1-3 (motor burn duration).
	13:45:00	DC-29 (first maneuver arm).
	14:05:00	DC-14 (release maneuver inhibit).
	14:35:00	DC-27 (start midcourse maneuver).
	14:47:31	DC-13 (inhibit maneuver).
	15:22:00	DC-21.
	15:32:00	DC-21.
	16:02:00	DC-21.
	22:40:00	DC-21.
	23:04:00	DC-21.
December 4, 1964	23:05:00	DC-21.
	23:06:00	DC-21.
	23:40:00	DC-21.
	23:57:00	DC-21.
	23:58:00	DC-21.
5	13:05:00	QC-1-1.
	13:10:00	QC-1-2.
	13:15:00	QC-1-3.
	13:45:00	DC-29.
	14:05:00	DC-14.
	14:25:00	DC-27.
	16:52:00	DC-21.
13	14:09:00	DC-7 (switch power amplifiers).
17	16:00:00	DC-21.
	17:30:00	DC-15 (Canopus gate override) transmitted.
February 11, 1965	03:29:29	DC-3 (switch to Mode 3) transmitted. Spacecraft telemetry to Mode 3.

Table 9. Ground commands received by *Mariner IV* (cont'd)

Date	Time, GMT	Command
February 11, 1965	03:36:13	DC-2 (switch to Mode 2) transmitted. Spacecraft telemetry to Mode 2.
	03:53:15	DC-26 (encounter science off) transmitted. Spacecraft cruise science off, battery charger off, boost mode enabled.
	04:15:51	DC-2 transmitted. Spacecraft cruise science on.
	04:32:39	DC-28 (battery charger on) transmitted. Spacecraft battery charger on, boost mode disabled.
	06:54:43	DC-25 (encounter science on) transmitted. Spacecraft encounter science on, scan cover deployed, platform scan started, battery charger off, boost mode enabled.
	08:59:23	DC-24 (inhibit scan) transmitted. Spacecraft scan platform stopped at 177.9-deg clock angle.
	09:13:51	DC-28 transmitted. Spacecraft video storage 2.4-kc power off, battery charger on.
	09:30:56	DC-3 transmitted. Spacecraft telemetry to Mode 3. Television camera shutter normal.
	10:21:20	DC-2 transmitted. Spacecraft telemetry to Mode 2.
	10:27:08	DC-26 transmitted. Spacecraft encounter science off, television shutter positioned, battery charger off, boost mode enabled, cruise science off.
	10:49:35	DC-2 transmitted. Spacecraft cruise science on.
July 14	14:27:55	DC-25 (encounter science on) transmitted.
	17:10:18	DC-24 (inhibit scan) transmitted.
	22:10:29	DC-3 (transfer data encoder to Mode 3) transmitted.
15	00:11:57	DC-16 (narrow-angles acquisition) transmitted.
	00:31:42	DC-26 (all science off) transmitted.
	00:32:40	DC-2 No. 1 (transfer data encoder to Mode 2 and turn on cruise science) transmitted.
	00:37:00	DC-2 No. 2 transmitted.
	00:42:00	DC-2 No. 3 transmitted.
	00:47:00	DC-2 No. 4 transmitted.
July 15, 1965	00:52:00	DC-2 No. 5 transmitted.
	00:57:00	DC-2 No. 6 transmitted.
August 3	03:08:33	DC-28 (turn off video storage 2.4-kc power and turn on battery charger) transmitted.
	03:14:33	DC-26 (turn off battery charger) transmitted.
	03:20:33	DC-2 (transfer data encoder to Mode 2 and turn on cruise science) transmitted.
21	22:37:21	DC-25 received at spacecraft. Encounter science turned on and scan platform in search mode.
	23:35:21	DC-28 No. 1 received at spacecraft. Video storage 2.4-kc power turned off and battery charger turned on.
	23:37:21	DC-28 No. 2 received at spacecraft. No effect.
	23:43:34	DC-26 No. 1 received at spacecraft. Battery charger and all science turned off.
	23:45:34	DC-26 No. 2 received at spacecraft. No effect.
	23:47:34	DC-2 No. 1 received at spacecraft. Cruise science turned on.
	23:49:34	DC-2 No. 2 received at spacecraft. No effect.
	23:54:21	DC-2 No. 3 received at spacecraft. No effect.
	23:59:21	DC-2 No. 4 received at spacecraft. No effect.
22	00:04:21	DC-2 No. 5 received at spacecraft. No effect.
	00:09:21	DC-2 No. 6 received at spacecraft. No effect.
	00:14:21	DC-2 No. 7 received at spacecraft. No effect.
26	21:22:32	DC-13 received at spacecraft. Inhibit midcourse maneuver.
	21:30:56	QC-1-1 received at spacecraft. Load clockwise 0.18 deg pitch turn.
	21:39:20	QC-1-2 received at spacecraft. Load clockwise 0.18 deg roll turn.
	21:47:44	QC-1-3 received at spacecraft. Load 0.08 sec motor burn duration.

Table 9. Ground commands received by *Mariner IV* (cont'd)

Date	Time, GMT	Command
August 27, 1965	19:55:50	DC-17 received at spacecraft. Canopus cone angle updated to 77.8 deg.
30	20:45:56	DC-25 received at spacecraft. Encounter science on and scan platform in search mode.
	21:26:20	DC-3 received at spacecraft. Data encoder transferred to Mode 3.
	23:04:29	DC-24 received at spacecraft. Scan platform motion inhibited.
	23:51:22	DC-16 received at spacecraft. Narrow angle acquisition logic actuated.
31	00:21:14	DC-2 received at spacecraft. No effect, data encoder previously returned to Mode 2 data by DAS logic.
	00:59:56	DC-26 received at spacecraft. All science off.
	00:04:56	DC-22 received at spacecraft. Video storage charge tracks.
	01:40:57	DC-4 received at spacecraft. Data encoder transferred to Mode 4/1 data.
September 2	06:33:05	DC-28 received at spacecraft. Video storage 2.4-kc power off, battery charger on.
	06:39:05	DC-26 received at spacecraft. Battery charger off.
	06:45:05	DC-2 received at spacecraft. Cruise science power on, data encoder to Mode 2.
October 1	21:48:01	DC-12 received at spacecraft. Transmitter switched to low-gain antenna.

The following are items of interest and problem areas noted during the *Mariner IV* mission.

Command procedures. Normal procedures for sending commands called for the ground command OSE $8f_s$ frequency to be set to the nominal spacecraft command subsystem $8f_s$ design frequency after the command detector was in lock and prior to sending commands. This procedure was followed during all periods of the mission when spacecraft telemetry was available to indicate to SPAC that the command detector had achieved and was maintaining lock. However, when the spacecraft data encoder was in Mode 3 or 4, no engineering telemetry was available to indicate the lock status of the command detector.

A special procedure was adopted to insure maximum likelihood of the command detector being in lock when commands were sent with no telemetry indication of detector in lock. Instead of returning the ground command OSE $8f_s$ frequency to nominal prior to sending commands, the $8f_s$ offset used to achieve detector lock was left in. Thus if the command detector should become transient and momentarily drop lock without telemetry to indicate this to SPAC, it would attempt to relock automatically.

VCO frequency monitoring. To determine the frequency at which to set the ground command OSE in order to optimize command detector lockup, it is necessary to know the average out-of-lock frequency of the spacecraft command detector. Spacecraft Telemetry Channel 104 contains VCO frequency information as well as an indication of command detector lock. An IBM 1620 computer program was written to average the readings of VCO frequency telemetry and convert them to engineering units. *Mariner* operations technical assistants ran this program for 500 telemetry samples per week when the command detector was out of lock and forwarded this information to the command subsystem cognizant engineer. Figure 16 shows the VCO frequency history.

Command detector lockup exercises. To ensure that SPAC and DSIF personnel maintained their proficiency in carrying out command procedures and to periodically verify command operations, a series of command detector lockup tests was initiated February 3, 1965 and carried on throughout the mission. These tests consisted of cognizant personnel following standard command procedures and locking up the command detector without sending commands. These tests were carried out at all prime and backup stations including the 100-kw station at DSIF 13. As a result, a high degree of confidence was maintained in the ability to command the spacecraft as necessary.

Command polarity verification. During compatibility testing with DSIF and the spacecraft, it was discovered that it was possible for the command subcarrier transmitted by the DSIF station to be of the wrong polarity with respect to the spacecraft command detector. To resolve this problem and ensure that all ground stations in the Net were transmitting a signal of the correct polarity, a modulation checker was obtained. This modulation checker was then carried to each station and a qualified operator verified correctness of the

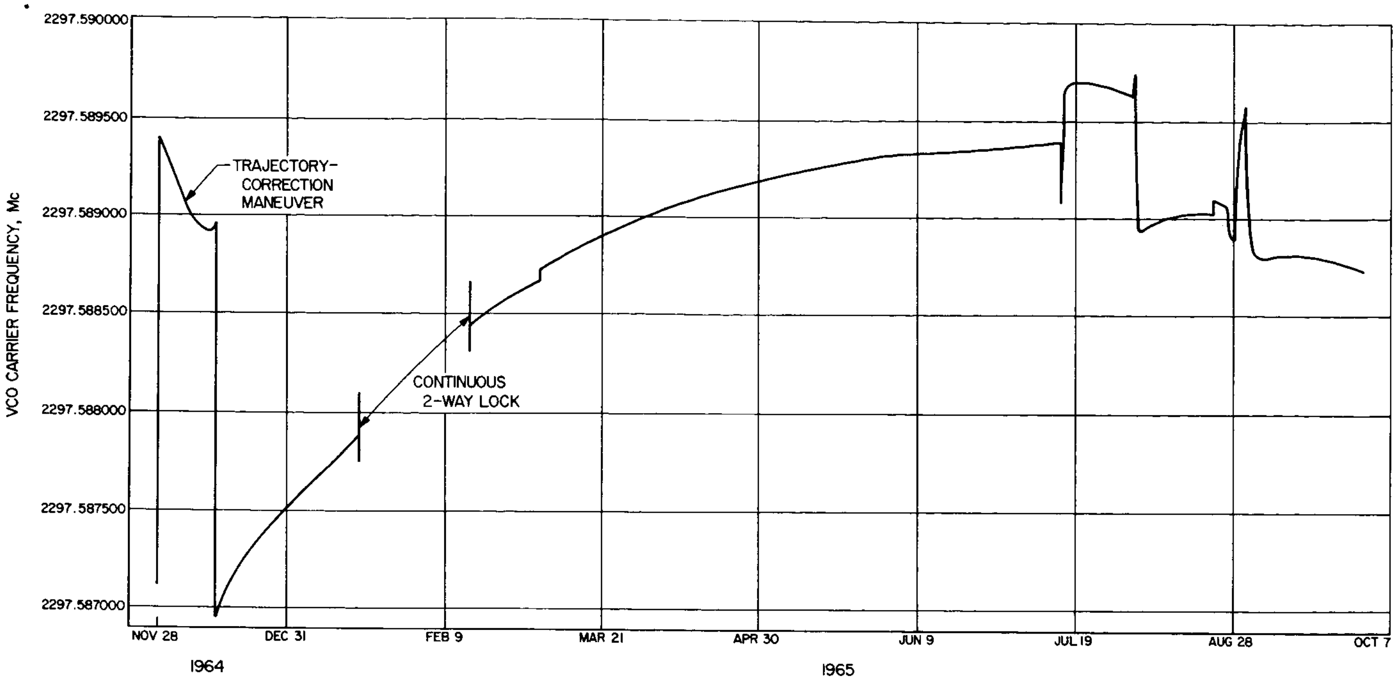


Fig. 16. VCO frequency history

command subcarrier polarity. Subsequent to this verification, the ground command subsystem at each station was frozen to ensure against inadvertent polarity reversals during the mission.

Ground command OSE tape punch anomaly. On December 17, 1964, during preparation of the ground command OSE for transmission of DC-15, the spacecraft command detector went out of lock when the ground command OSE tape punch at DSIF 41 was switched to the standby mode. Subsequent investigation revealed that a power transient occurred in the ground command OSE when the tape perforator standby switch was operated. This caused momentary inhibition of the command signal which in turn caused the spacecraft command subsystem to drop lock. After it was verified that the problem was only concerned with the tape perforator and standby switch, it was decided that the perforator should not be placed in standby condition while the ground command OSE was locked up with the spacecraft. No further problems were encountered by use of this procedure.

Drop of lock during $8f_s$ normalization. During command lockup prior to the early science cover deployment operation on February 11, it was observed that the command detector telemetry indicated out of lock while the ground command OSE $8f_s$ was being normalized. Subsequent investigation revealed no evidence of malfunction of either the ground command OSE or the spacecraft command subsystem. The command detector was then relocked and all indications were normal. All 12 commands of the cover deployment sequence were properly received and acted upon, and no further anomalous indications were found.

c. Summary. No other problem areas arose concerning the command subsystem and performance through the end of picture playback continued to be normal. Since all commands sent to the spacecraft were properly acted upon, it can be concluded that all commands were received with no bit errors, though it is not possible to perform a statistical bit error test on a spacecraft in flight.

6. Video Storage Subsystem

a. Description. The *Mariner IV* video storage subsystem, which consists of a hermetically-sealed tape transport unit and associated electronic equipment, is required to accept and record digital television picture data from the DAS at a fixed rate of 10,700 bits/sec, and to store a minimum of 20 pictures of video data. These data are then reproduced at a rate of 8⅓ bps during the picture playback period. The total bit capacity of the tape recorder must be at least 5.24×10^6 bits, contained on two tracks of 0.25-in. tape, 30-ft long. Design bit error rate is less than 1 bit error in 10^5 bits, not including errors accumulated while establishing lock or due to track changes.

The *Mariner IV* tape recorder has three operating modes, described as follows:

Launch mode. During the launch phase of the flight, the tape transport is operated at record speed, although no actual recording takes place; this is to preclude any possibility of tape spillage due to launch phase shock and vibration. The record motor is started shortly before launch and continues running until after the spacecraft separates from the *Agena* rocket. When the spacecraft separates from the *Agena* rocket, the EOT pulse is combined in an AND gate with the *Agena* separation signal. The AND result is applied through an OR gate and driver amplifier to a latching relay. This pulse resets the latching relay which in turn deenergizes a bias relay, thereby removing power from the record motor and locking out the launch mode circuitry. After the record motor and tape coast to a stop, the EOT foil is positioned at a point approximately 2.5 ft beyond the record/reproduce head. This is the normal tape position from which to start the encounter record sequence.

Record mode. Application of 2400-cps power and single-phase 400-cps power prior to encounter energizes all sections of the recorder system except those associated with the launch mode. Assuming that the tape is positioned properly, the recorder system is ready to record at least 20 picture sequences while making two complete pases of the tape, changing tracks after the first pass and returning to the initial track after the second pass. Response to record commands is inhibited after the EOT foil passes the sensor contacts the second time.

The absence of a playback command signal from the data encoder conditions the system for recording by applying power to the record head drivers, as shown in the record mode functional diagram, Fig. 17. Thus, whenever the recorder system is not operating in the reproduce mode, it is in the record mode and will record or erase the tape.

The start-stop relay applies power to the record motor when in the start position, so that record operation is controlled by the DAS via its start and stop pulses.

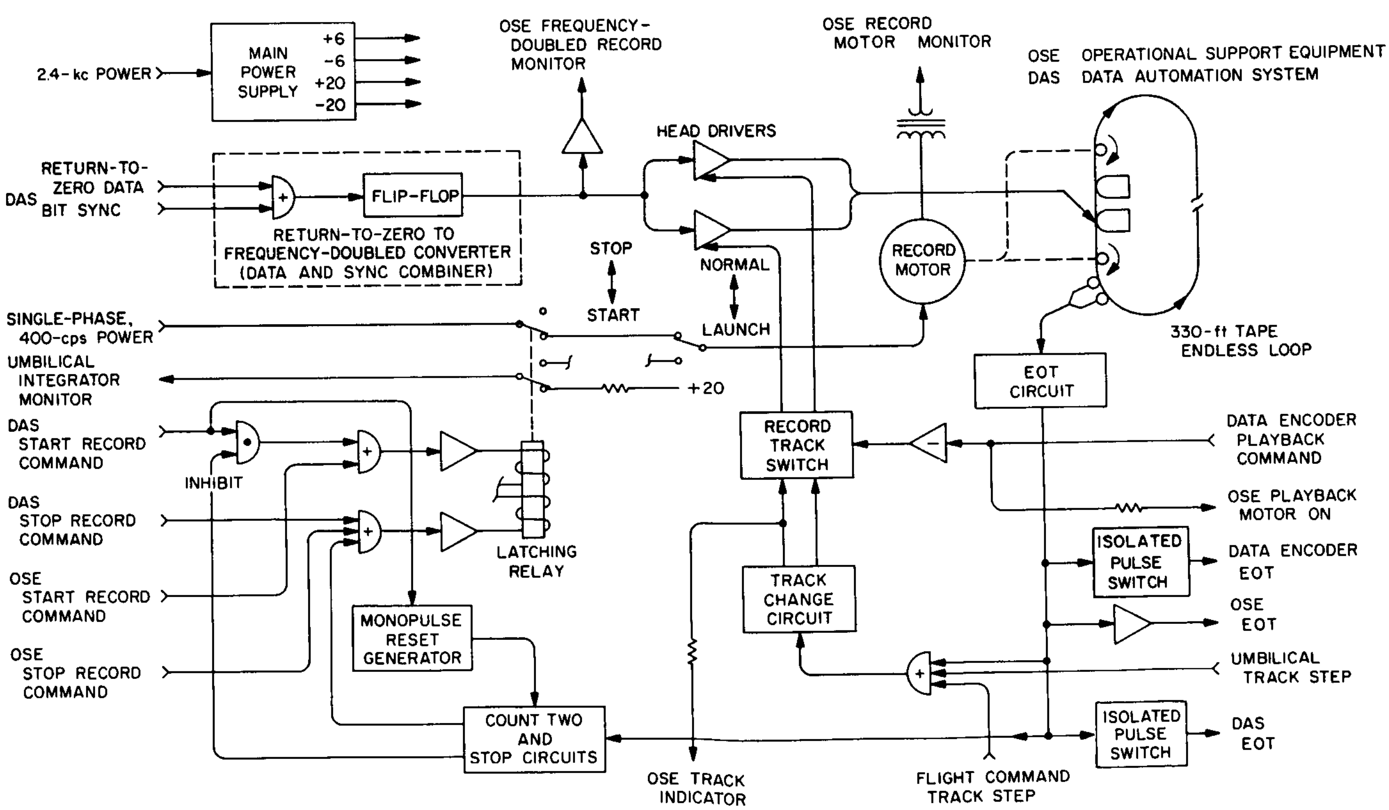


Fig. 17. Video storage subsystem record mode block diagram

During recording, a pulse is generated for each complete pass of the tape by the EOT circuit as the end-of-tape foil passes over the sensor contacts. This EOT pulse is applied internally to the track change and count-two sections of the recorder system and externally to the DAS and data encoder. In a normal planetary encounter record sequence, the DAS ignores the first EOT pulse and responds to the second if it occurs between the beginning of the nineteenth and the start of the twenty-second picture sequence. The DAS is mechanized to terminate encounter record sequences upon receipt of the second EOT or at the end of the twenty-second picture if an EOT pulse has not been received.

Track changes take place whenever the EOT circuit generates a pulse. The track change circuit incorporates a magnetic latching relay arranged in a bistable flip-flop configuration. EOT pulses also trigger the count-two-and-stop circuits of the recorder system to inhibit DAS start record commands upon receipt of the second EOT signal. The counter is reset to zero by the first DAS start record command acting through a monopulse reset generator. Further reset of the counter is impossible unless 2400-cps power is removed from the system.

Reproduce mode. After planetary encounter and the data have been recorded, the recorder system is switched into the reproduce mode, Fig. 18, by a command from the data encoder. Playback signals are amplified and operated on in such a manner as to reproduce the frequency-doubled code waveform that was recorded. Bit sync is separated from the data and compared in phase with the master bit sync from the data encoder. These comparisons determine indirectly the frequency of the VCO which dictates playback tape speed. When the correct speed and phase are achieved, the tape-signal-derived bit sync and the master bit sync are in the proper phase relationship for accurate readout of the data. If a stable phase relationship has been established, the recorder system is said to be phase-locked. The reproduce mode may continue indefinitely after encounter, while the system automatically changes tracks once for each pass of the

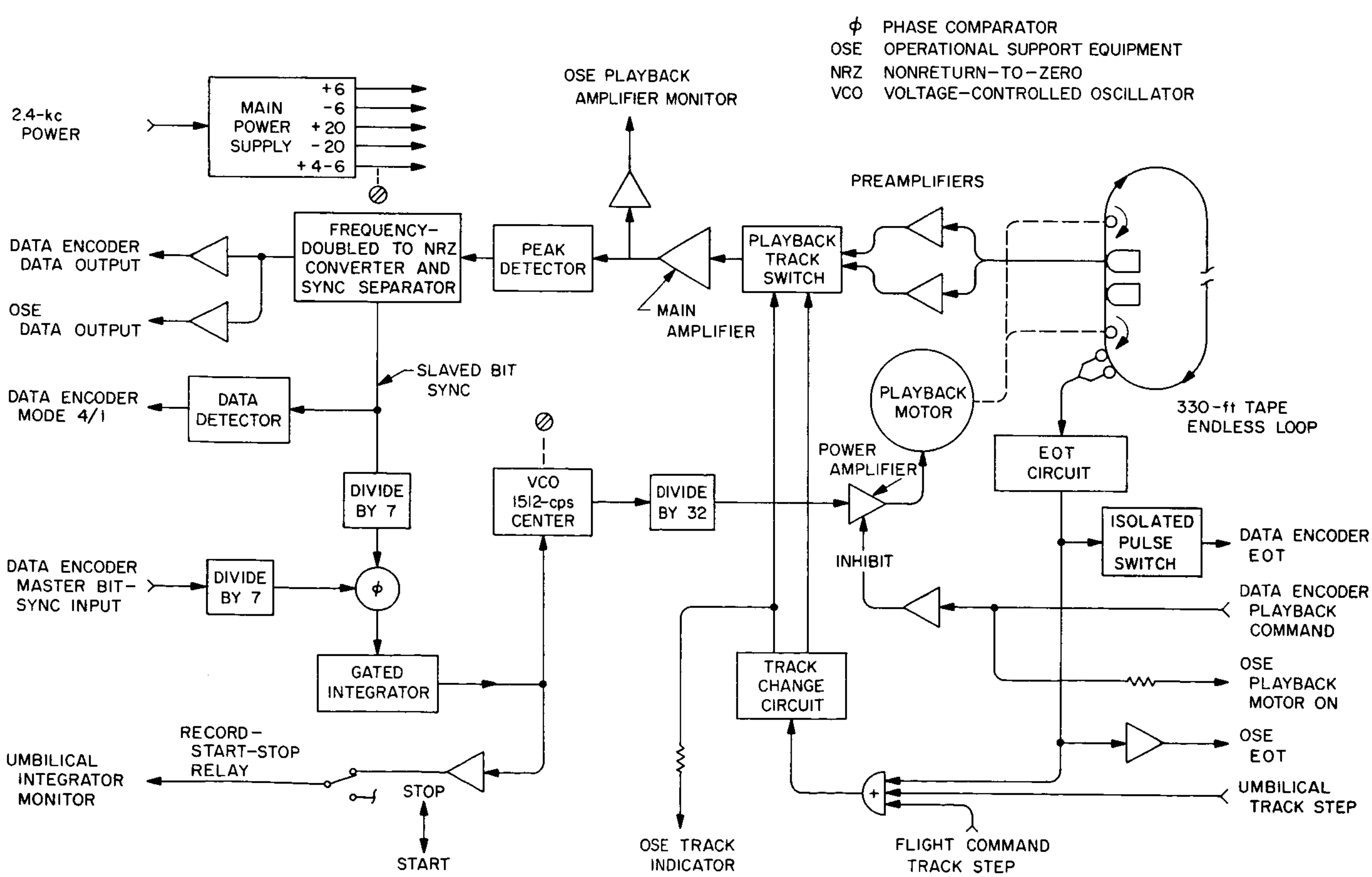


Fig. 18. Video storage subsystem reproduce mode block diagram

tape. Complete reproduction of twenty picture sequences will require from 8 to 10 days. Because sections of the tape were erased between record sequences, data will not be present at the output of the system for approximately 2 hr between picture data blocks. Provision has been made in the recorder system for detecting this condition and automatically switching the data encoder from stored data playback to engineering data as long as the data are absent.

Passage of the tape over the reproduce head gap produces a signal which is roughly the derivative of the tape flux. This signal, in the order of 50 μv peak-to-peak, is amplified by low-noise preamplifiers located in the tape transport subassembly and routed to the playback track switch and main amplifier. Signals from both tracks are applied simultaneously to the playback track switch where one signal is selected at a time.

Each complete pass of the tape loop generates an EOT pulse as the EOT foil passes over sensor contacts. As in the record mode, the EOT pulse is applied to the track-change circuits and is also fed out to an event counter in the data encoder.

b. Performance. The *Mariner* Mars 1964 video storage subsystem mission functional specifications stipulated the two prime requirements of storing twenty television pictures of Mars and playing them back at an error rate less than 1 in 10^4 bits. As indicated by Table 10, the video storage subsystem exceeded the first requirement

Table 10. Picture start and stop times of the *Mariner IV* playback

Picture No.	From		To	
	Date, 1965	Time, GMT	Date, 1965	Time, GMT
1	July 15	13:01:58	July 15	21:38:07
2	15	23:32:27	16	08:08:03
3	16	10:04:28	16	18:39:54
4	16	20:35:12	17	05:10:12
5	17	07:07:45	17	15:43:18
6	17	17:40:32	18	02:16:10
7	18	04:13:25	↓	12:49:03
8	18	14:46:13	↓	23:21:51
9	19	01:19:32	19	09:55:35
10	↓	11:52:52	19	20:28:29
11	↓	22:25:33	20	07:01:26
12	20	08:57:42	20	17:33:15
13	20	19:28:51	21	04:04:32
14	21	06:01:22	21	14:37:00
15	21	16:34:43	22	01:10:21
16	22	03:07:13	↓	11:42:53
17	22	13:40:59	↓	22:16:37
18	23	00:15:04	23	08:50:41
19	↓	10:48:08	23	19:23:48
20	↓	21:21:10	24	05:56:50
21	24	07:55:01	↓	16:30:40
22	24	18:28:33	↓	19:26:33

Picture No.	From		To	
	Date, 1965	Time, GMT	Date, 1965	Time, GMT
1	July 24	21:21:53	July 25	05:57:54
2	25	07:53:36	25	16:29:15
3	25	18:24:31	26	03:00:09
4	26	04:56:24	↓	13:32:03
5	26	15:27:47	↓	00:03:39
6	27	02:00:57	27	10:36:39
7	↓	21:08:41	↓	12:33:01
8	↓	07:40:41	↓	23:04:57
9	28	18:12	28	09:37
10	29	04:45:32	28	20:09:57
11	29	15:19:13	29	06:43:10
12	30	01:51	29	17:16:43
13	↓	12:21	30	03:46:24
14	↓	22:55:28	30	14:20
15	31	09:27	31	00:52
16	31	20:02:15	↓	11:24
17	August 1	21:59:57	↓	06:31
18	↓	08:28	August 1	17:09:00
19	↓	19:07:10	2	03:42:33
20	2	05:40:00	2	14:15:27
21	2	16:14:48	3	00:39
22	3	02:36	3	03:36:02

by actually recording twenty-one full pictures and approximately one-tenth of the 22nd picture.

Although an absolute playback error rate for the playback could not be determined, due to the large amount of redundant information that is contained in a television picture, it was possible to perform a preliminary error rate analysis of the playback data. Also, since the recorder was played back twice, the first playback can be compared with the second playback. The redundant picture elements in the first picture playback were examined in great depth and the error rate was found to meet the specifications. Random samples of the first and second playbacks, which have been compared in all cases, confirmed this.

Significant video storage events through the end of playback are shown in Table 11.

Recorder pressure seal. Because the recorder case temperature change during the mission was very small, it was assumed that all of the case pressure drop was due to gas leaking through the seals. The pressure drop of 0.86 lb was considered quite good in light of the design tolerance of 19.0 lb.

Encounter performance. During the encounter recording sequence, two abnormal false shutter indications were observed and the data encoder switch to Data Mode 2 was thought to be 22 sec late.

The two false shutter indications could have come from either the DAS or from the video storage subsystem. A series of tests have been run on the video storage subsystem type-approval unit in an attempt to simulate the events observed on the *Mariner IV* spacecraft. The flight events could not be reproduced. Tests on the PTM spacecraft have indicated that the DAS can produce these events when 400-cps power is switched, although no verification of the source is possible.

With respect to the data encoder switch to Mode 2 data event, the DAS logic design is such that the DAS will not issue a data encoder Mode 2 data command until the end of Picture 22 if the video storage subsystem second end-of-tape signal is received after the start of Picture 22. During the *Mariner IV* spacecraft encounter, the second EOT occurred in the twenty-first line of Picture 22; consequently, the DAS did not issue a switch to Mode 2 data command until the

Table 11. Video storage subsystem flight events

Date	Time, GMT	Event
November 28, 1964	14:18:01	Launch mode on. Verified by Telemetry Channel 221.
	14:22:01	Atlas liftoff. Recorder pressure 22.76 psi, recorder temperature 60.31°F.
	15:07:00	(Approximate) Spacecraft/Agena separation.
	15:09:10	Calculated time for VSS launch mode off. Spacecraft telemetry data was available. Confirmation from Channel 221 delayed until 1514.
July 14, 1965	14:52:31	Encounter power on via DC-25 command. VSS power on confirmed by Telemetry Channel 227.
	00:29:59	Narrow angle acquisition.
	00:30:36	Start of first picture television data.
	00:34:59	Abnormal false shutter indication in science data.
	00:40:52	Abnormal false shutter indication in science data.
	00:42:46	Expected first end-of-tape event. Verified at 00:43:22.
	00:44:24	Expected second end-of-tape and switch to Mode 2 data.
	00:44:46	Mode 2 data 22 sec later than expected. Engineering telemetry verified EOT events normal.
15	11:53:53	CC&S MT-9 command to start playback.
20	03:15:32	Track change on Line 111 of Picture 11.
24	19:26:33	Track change on Line 21 of Picture 22.
29	11:33:27	Track change on Line 111 of Picture 11. Extra event observed.
August 3	03:36:02	Playback terminated on Line 18 of Picture 22.

end of Picture 22. It is concluded that the switch to Mode 2 data time observed at 00:44:46 GMT, July 14, reflects proper spacecraft operation, although this was not understood at the time.

At the start of picture playback and again at the start of the second playback, there were 68 sec of old data prior to the first recorded Mars data. It was not erased during the encounter sequence because the tape traveled 0.68 in. farther during run-down at the completion of the launch sequence than it did after the second EOT during encounter. This is a clear indication of friction buildup.

The Mode 1 data spaces between pictures were approximately 117 min long. Before launch, they averaged 123 min. The difference in run-down distance accounts for 1 min of this 6-min discrepancy. To account for the remaining 5 min, the acceleration time in the record mode would have to have been at least 0.25 sec larger than in the prelaunch condition. Both friction buildup and decreased 400-cycle encounter power voltage could account for this change. While no direct measurement of the 400-cycle voltage is made, it is felt that any voltage change would be in the opposite direction, again supporting the friction buildup hypothesis.

Calculations of utilization of the available tape indicate that the tape must have been moving 7% slower during Mode 1 data spaces than during Mode 4 data spaces. This is normal and is the same as it was before launch.

The observance of two telemetry events at the track change during the second playback indicates that the conductive foil used to stimulate track change events had become contaminated by the second playback. This is not unlikely since *Mariner IV* already had a large number of passes on the tape machine at the time of launch.

The major events in the flight history of the video storage subsystem from launch through the end of the second playback after encounter are shown in Table 11.

Television haze calibration sequence. During the black-space television recording sequence performed on August 30, 1965, a total of 10½ pictures were recorded. The first five pictures were played back for analysis purposes, Table 12. Video storage subsystem characteristics were the same as during the playback of the planetary pictures.

Table 12. Television haze calibration picture playback times

Picture No.	From		To	
	Date, 1965	Time, GMT	Date, 1965	Time, GMT
1	August 31	02:00:46	August 31	10:36:38
2	↓	12:30:40	31	21:06:30
3		23:01:03	September 1	07:37:10
4	September 1	09:31:40	1	18:07:30
5	1	20:03:20	2	04:39:23
6	2	06:34:46	2	06:48:32

7. Mission Operations

a. Support. Telecommunications support for the *Mariner* SPAC consisted of telecommunications analysts representing the entire communications system and subsystem analysts representing radio, data encoder, command, video storage, and ground telemetry. In addition, in-depth backup analysis support was provided by Telecommunications Analysis Team (TCAT), which consisted of Telecommunications Division personnel experienced in all phases of telecommunications system operation, plus cognizant engineers from each of the telecommunications subsystems.

During critical phases of the mission such as launch, trajectory-correction maneuver, and encounter, support was provided on a 24 hr-a-day basis. During the cruise phase, daily checks of performance were made by each subsystem and alarm limits were constantly monitored by the SFOF computer system. Analysts from all subsystems were on call during the cruise phase, and if any telemetry data violated predetermined limits, the cognizant person was called. Despite some initial misgivings as to the reliability of computerized alarm monitoring, the system proved to work very well and provided real time monitoring of data even when cognizant analysts were not present.

b. Data sources and display. During the *Mariner IV* mission, spacecraft telemetry data were made available in several forms, some of which proved to be very useful and others which have not been of significant value for data analysis. A list of the various available data sources is presented along with a discussion of their usefulness to telecommunications system analysis.

Unprocessed telemetry data. While in a somewhat inconvenient format for data analysis, the unprocessed telemetry (raw) data as displayed on the 60 wpm printers proved to be the most reliable source of real-time data during the mission. With the use of overlays, decoding matrices and a little practice, the raw teletype (TTY) data could be converted to a useable form in a matter of seconds. These data were usually several seconds ahead of the IBM 7040 formatted data output, and for this reason the raw data were sometimes used even when formatted data were available. Whenever a question arose as to the validity of a datum point on output from any source, the raw data were used as the deciding factor. Even though the many computer outputs displayed the data in a more convenient form, it is felt that the raw data display capability was more than justified by its superior reliability alone.

Formatted telemetry data. Use of the IBM 7040 computer permitted the telecommunications system telemetry channels to be displayed on a separate format and to be converted to data numbers. This was by far the most convenient data display available to telecommunications during the *Mariner* mission. During the prelaunch phase, it was decided to display the 7040 data as data numbers instead of in engineering units. In retrospect, it appears that this was a good decision. It became easier to evaluate data quality in real time using data numbers instead of engineering units to spot bit errors and *out of locks.* However, for postflight analysis and evaluation of system performance, a display in engineering units would be more convenient to use.

JPEDIT. JPEDIT was an NRT output of the total data stream in a convenient format for performance analysis, data records, etc. Its information content was supposed to be the same or better than the 7040 formatted data. After several attempts to utilize JPEDIT during the mission, it was found that it consistently contained more time regressions and other erroneous indications than the 7040 output. If a questionable datum point were found in JPEDIT, it was always checked against the 7040 data and the raw data. This low confidence in JPEDIT made it of questionable value to telecommunications system analysis, and it did not play a significant role in mission operations.

EDPLOTM. EDPLOTM was a program that would take the data used for JPEDIT and plot it out in data numbers versus time for performance analysis and data records. The EDPLOTM function appeared to be a valuable tool and a series of display formats were developed and used. However since its data source was essentially the same as JPEDIT, it contained the same time regressions and other erroneous indications. For this reason, the EDPLOTM program, although it could have been of great value, was not effectively used for telecommunications analysis during the *Mariner* mission.

SSDM. SSDM was a program whose output was a condensed version of JPEDIT. Essentially it would give an output only when a data number changed in a specified channel. This program was also of practically no usefulness to telecommunications analysis because of the erroneous indications it consistently contained.

MDL. The Master Data Library (MDL) will contain a listing of the best data available from all sources for the *Mariner* mission. This will also include selected station functions as well as spacecraft data. Telecommunications plans were to use this data, when available, to perform a statistical analysis of the uplink and downlink signal level data from the flight. It is also desirable to be able to get EDPLOTM runs of the MDL for a handy, reliable source of mission data. At present, it is not known if this will be made available, but it seems incongruous to have gone to considerable effort to generate a best data source and not make full use of it.

c. Computer programs. To facilitate the rapid, accurate analysis of telecommunications system data from a long mission such as the *Mariner,* it is necessary to make extensive use of automatic data processing whenever possible. Computer programs were requested over a year before launch to provide predictions of system performance, AGC calibrations and evaluation of system performance during the mission. The following paragraphs describe the programs requested, their disposition and their use.

AGCM. This program provides the capability to take station calibration data from before and after the tracking pass and generate curves and lookup tables of received signal level vs AGC voltage. It also generates tables of received signal level vs a subcarrier oscillator frequency which is recorded on magnetic tape as a permanent record. Along with each curve, coefficients of third through fifth order least-square fits are generated for use during postflight analysis of signal-level data.

This program was ready and checked out well before launch and was used during critical phases of the mission for near-real-time evaluation of signal-level data. An addition was made to the program to accommodate calibrations from S-band stations as well as L-to-S conversion stations. Further use of this program will be made during postflight statistical analysis of DSIF and spacecraft data.

CPPM. CPPM was the *Mariner* communications prediction program that generated curves and tables of predicted system performance using inputs from trajectory SAVE tapes, DAP antenna pattern tapes, and cards of telecommunications system parameters. This program was completed in its first stage, which did not have the capability of utilizing DAP tapes, over a year before launch. This version was used with antenna patterns that were tediously generated by hand to perform preliminary performance predictions.

The final version, which included DAP handling capability, was completed in a frenzy of activity shortly before launch and used to generate sets of final prelaunch predicts. Due to the shortness of time when the DAP processing capability was being programmed, several compromises had to be made and the final version of CPPM was somewhat bulky and time consuming to use, and was quite mission dependent. The version of CPPM finally used for *Mariner* was very satisfactory within its limitations as ultimately proved by the excellent agreement between predicted and actual values for the entire flight.

Program X. This program was requested over a year before launch to provide an up-to-date analysis of uplink and downlink performance during the flight. When it became obvious that this program would not be available, its scope was modified to that of a more detailed postflight analysis program for which an RFP is being generated. If this program in its original form had been programmed as requested for the *Mariner* flight, a more rapid and accurate evaluation of problems such as variations in downlink AGC calibrations would have been possible. Instead, hand plotting had to be used to achieve a far more time-consuming and less accurate result.

d. General problem areas. During the course of mission operations, several general problems were noted that are worthy of mention. This discussion will be limited to operational problems rather than technical problems which will be discussed with their respective subsystem reports.

SFO/DSN Failure/Problem Reports (F/PRs). The SFO/DSN F/PR system did not provide correct, complete answers to the satisfaction of the initiator. Many of the F/PRs written by the telecommunications area were answered in a token manner and closed out. It is strongly recommended that for future missions a SFO/DSN F/PR system be devised that will be conscientiously followed through, rather than merely being accepted as was the present one. Final closeout should require the signature of a cognizant section manager and a system manager. Unanswered reports or reports with unsatisfactory answers should be subject to the same level of concern that the present flight equipment P/FRs are. Only when everyone has confidence in the system and those responsible for the system require a complete, correct and timely action will an F/PR system serve a useful purpose.

Communication between SPAC and DSIF stations. On numerous occasions during the mission, anomalous indications affecting the telecommunications system would be observed in spacecraft telemetry data or by the DSIF stations. Most of these anomalies, after investigation, proved to be attributable to a ground station problem such as antenna hitching, cockpit error, transmitter power variations, etc. However, on many occasions the information necessary to rapidly and accurately isolate the problem was not readily available from the stations. As a consequence, many hours of the spacecraft analysts' time were spent in finding answers to P/FRs that were not concerned with the spacecraft.

A factor contributing to the difficulty in ascertaining origin of problems that occur at the station could be that the station crew was unaware they had a problem until the effect of the problem was observed. One reason for this may be the lack of instrumentation to adequately monitor station functions during a tracking pass. A typical example of instrumentation difficulties is in monitoring the S-band transmitter at the stations. Spacecraft telemetry would indicate power or frequency variations that were not verified by ground station data until additional checks had been performed.

Another problem is the actual transfer of information from the DSIF station to SFOF such that SPAC is always aware of the station condition during critical periods. During the excitement and turmoil of high activity phases of the mission, station personnel are sometimes unable to keep completely up to date on reporting of significant events, or potential problems sometimes go unnoticed. The only really satisfactory solution to this problem is to have selected station functions displayed in the SFOF in real time. Not only would this provide analysts with up-to-date information on station events concerning the telecommunications system, but it would also free station personnel for more important operational tasks.

The *Mariner* mission also pointed out that station functions can play an important part in flight operations. A good example is the use of AGC to provide roll attitude information during Mode 3 data and during roll search. The AGC display system used was somewhat hurriedly devised and did not have the desired resolution, but with more time and effort a reasonable system could be implemented.

Some of the facilities to implement a real-time display of station function "telemetry," such as digital input system (DIS) and signal-to-noise ratio estimator (SNORE), are presently being installed or checked out with DSIF stations. Since the telecommunications system includes the ground stations as well as the spacecraft, ground station information is as necessary to the telecommunications analyst as spacecraft data. It would be of definite advantage to future missions to include station functions in their real time data display requirements.

C. Power Subsystem

1. Description

The *Mariner IV* power subsystem performed two major spacecraft functions:

1. It generated standard voltages for distribution to spacecraft power users.
2. It controlled 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-whr 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.

a. Energy source. Figure 19 shows the functional block diagram of the power subsystem. The main source of spacecraft power is 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 no more than about 6%.

In order to increase system reliability, the solar panels were sized to carry all loads required after the midcourse maneuver. This significantly increased the theoretical probability of performing the encounter sequence successfully by not requiring battery operation except for a nonstandard situation.

Because the launch trajectories required the spacecraft to remain in the Earth's shadow for a period of up to 20 min, and because there was 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 series string of six 50-w zener diodes across each section.

Figure 20 shows the predicted solar panel maximum power capability vs time curves for the *Mariner IV* spacecraft. Also shown is the actual Mars encounter load.

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 conversion efficiency and load uncertainties

The substantial power margins shown in Figure 20 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 whr at launch. 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.

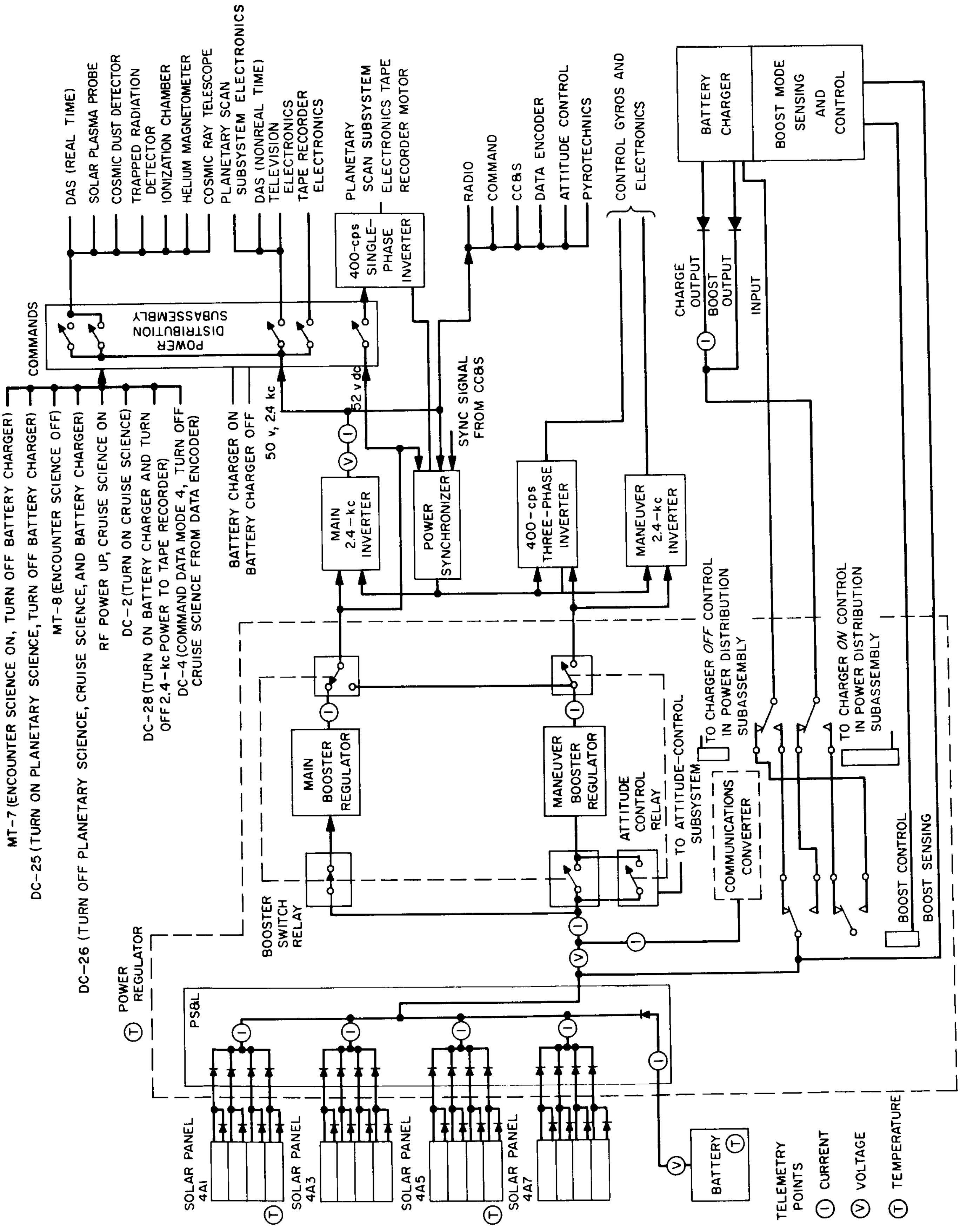


Fig. 19. Power subsystem block diagram

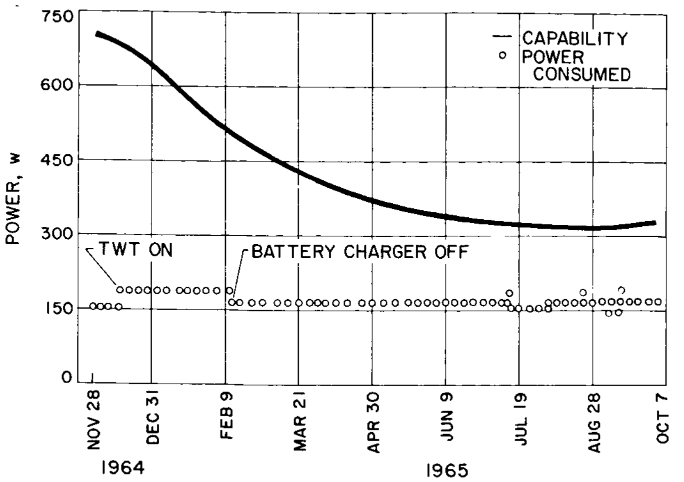


Fig. 20. *Mariner IV* solar panel power capability

Normally, the battery is used to supply power for from 7 min before launch until solar panel Sun acquisition. From Sun acquisition until the midcourse maneuver,[3] 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. Designing in this manner removed the battery charger as a series element in the power system.

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 accept power from the unregulated bus at voltages ranging from 25 vdc, the lowest battery voltage, to 50 vdc, the highest solar panel voltage, and add sufficient voltage to bring the outputs up to 52 vdc $\pm$ 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.

[3]Two to 10 days

If a failure should occur in the main regulator that allows its output to deviate from 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 vdc. All spacecraft equipment can stand this 16 percent 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 vdc instead of 52 vdc, as there are two diodes in series between the panels and the dc output. Tests have shown that even at this reduced output the spacecraft could complete the mission, though performance would be somewhat degraded.

d. Dc-to-ac inversion. The main power for spacecraft users is a 100-v (p-p), 2400-cps squarewave 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 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. Transmission of these high voltages through the spacecraft cabling is difficult.

2. Greater design flexibility is obtained, since users can change voltage requirements late in a program without affecting the power subsystem 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 subsystem. Also powered from the maneuver regulator is a 27-v rms, 400-cps, 3-phase inverter that delivers step squarewave power to the gyroscope spin motors. A 400-cps, single-phase, squarewave inverter supplying nominal outputs of 56 and 65 v peak-to-peak (p-p), to the science scan platform and video storage subsystem, 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 CC&S subsystem is counted down to provide both 2400-cps, single-phase and 400-cps, single-phase and 3-phase signals. In the 2400- 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 single frequency of twice 38.4 kc, a 38.4-kc oscillator internal to the power synchronizer starts, automatically disconnects the CC&S input, and takes over as the frequency source. While the power system is running on its internal oscillator, its frequency is controlled to ±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. (The 3-phase 400-cps inverter does not.) All spacecraft subsystems are designed to operate at frequencies in this worst-case range of ±5%.

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

Table 13. Power subsystem command capability

Control input	Source[a]	Type of signal	Required action by power subsystem	Remarks
1. Encounter start	CC&S MT-7	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)	a. 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.

in the power distribution subassembly. This unit accepts commands from other spacecraft subsystems and translates the commands into relay closures. Since loads are switched as groups[4] central control of the switching, as opposed to individual user switching, proved highly successful. Table 13 shows the power subsystem command flexibility.

f. Battery charging. After solar panel Sun acquisition, the battery is recharged by 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 was turned off in the charge mode, on-board logic would 1) sense the condition, 2) shift the charger into a current-limited, constant-voltage mode of 45-w capacity, 3) connect the input of the charger to the battery and the output to the unregulated power bus, and 4) boost the system out of the sharing mode. When sharing ceased, the logic would return the unit to its normally off position. Boosting would be 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[5] on-board logic would sense the sharing condition, stop the battery-charging operation, and await a ground command. A direct radio command could then be sent to initiate the boost mode.

g. Engineering telemetry. Proper selection of engineering telemetry points is an important part of the power subsystem design. Figure 19 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 14 lists all 22 telemetry measurements.

h. Solar panel standard-cell transducers. To more accurately predict 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-n* silicon solar cell

Table 14. Power subsystem telemetry measurements

Channel	Measurement	Range	Deck rate
109	PS&L[a] output voltage, vdc	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, vdc	23 to 40	
207	Main 2.4-kc output voltage, vdc	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.

[4]E.g., cruise science experiments and encounter science experiments.

[5]Very unlikely owing to the large power capability of the solar panels in this region.

that has been degraded by radiation to a point where further radiation has little effect on cell output. Comparison of its short-circuit output in flight with the short-circuit output of the other standard cell allowed a determination of the effect of radiation upon the solar panels.

2. Performance

With the one exception of a higher-than-expected battery voltage, operation of the power subsystem was completely normal and predictable throughout the flight of *Mariner IV*.

a. Solar panels. Operation of the solar panels began in a normal manner 61 min after launch when Sun acquisition occurred. During the 10-min period during which the spacecraft was in the Earth's shadow, the solar panel temperature slipped to an estimated 0°F. This is much warmer than the −112°F which would result from a maximum shadow time of about 20 min. As a result of the relatively warm solar array temperature, no zener diode operation was required.

Figure 21 shows the solar panel currents immediately after Sun acquisition. The wide differences in current result from variations in solar panel temperatures. After 1 hr the temperatures equalized and the currents merged.

During the remainder of Phase 1 of the mission, flight data obtained in the performance of the solar array indicated that the individual panels performed as predicted and no abnormalities or degradations were encountered during the 307 days of deep space exposure. Presented in Fig. 22 is a plot of the solar panel output currents and the PS&L voltages as a function of time in flight. It was observed that prior to January 27, 1965, the maximum spread of the output currents of the panels was 0.15 amp. On January 27, assuming the predicted 20°C gradient between the surface temperature of the

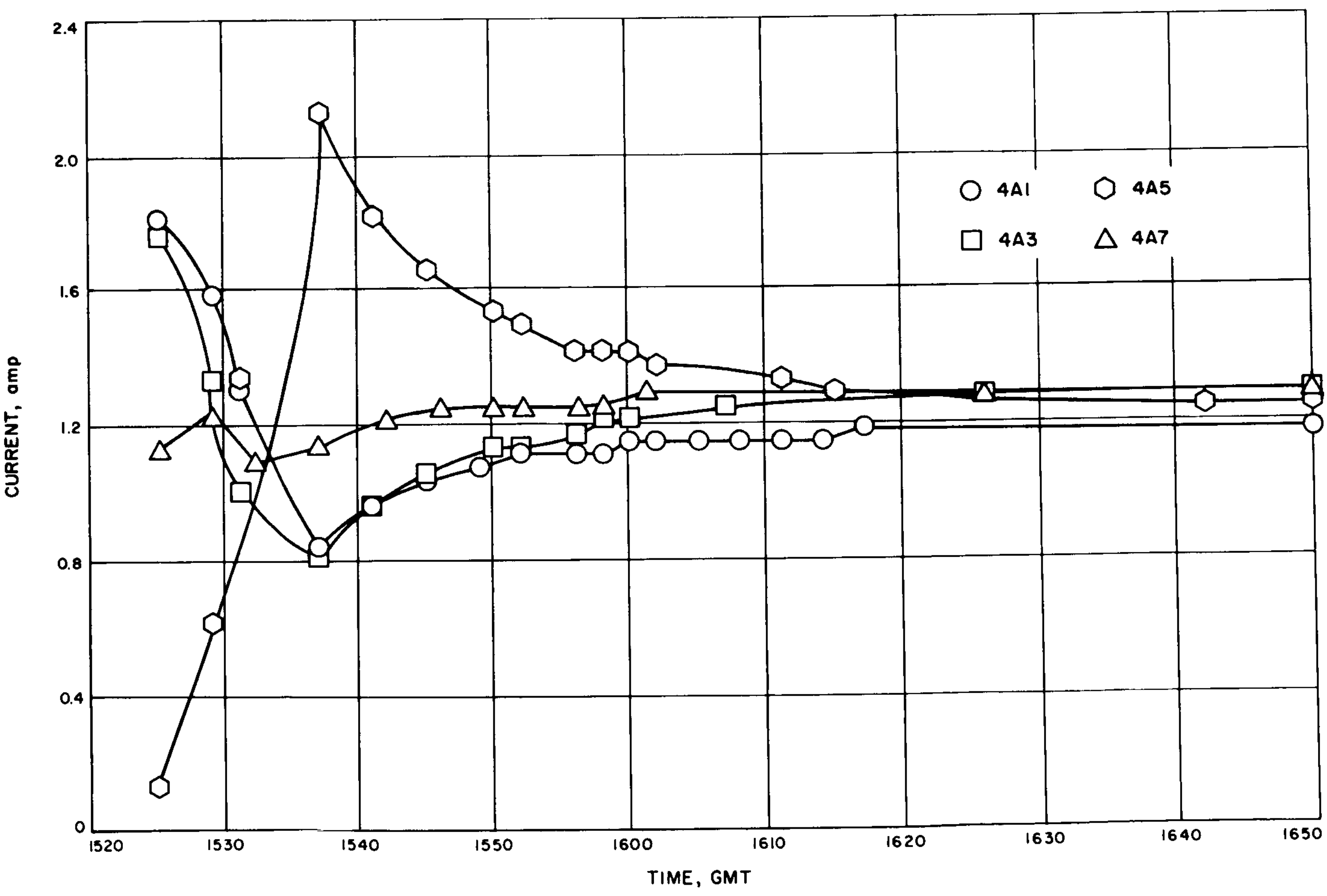


Fig. 21. Solar panel currents at Sun acquisition

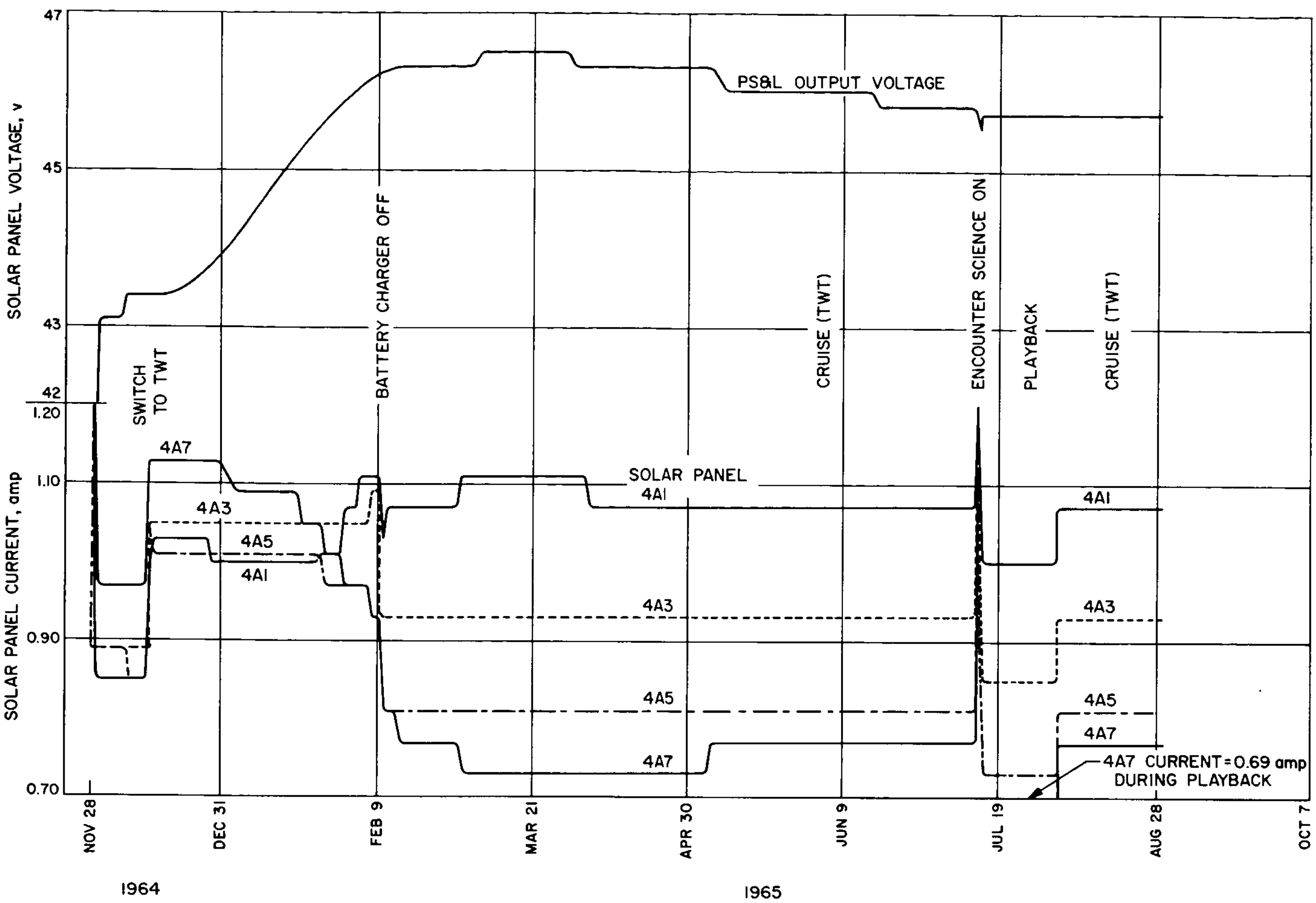


Fig. 22. PS&L output voltage vs solar panel current

panel and the zener diode shunt regulator,[6] the zeners would have been at the proper temperature to begin limiting the output voltage of the panel. That this did occur is manifested in the relative adjusting of the load sharing of the four panels during the subsequent days. Between February 1 and March 31, the solar-array sharing gradually changed, resulting in a maximum difference of 0.4 amp among the four panel currents. The differences in solar panel currents gradually changed throughout the mission as the zener diodes and the solar panel sections gradually changed their thermal–electrical equilibrium point.

As explained in the Description Section, a short-circuit-current, open-circuit-voltage transducer was mounted on one of the panels to aid in the evaluation of the solar array's flight performance. This transducer, which was monitored throughout the flight, consisted of three solar cell assemblies all similar to the cell assemblies of the array. However, one of the cells was instrumented to record its open-circuit voltage, a second was instrumented to record its short-circuit current and a third cell, which had been irradiated with 1 mev electrons to a 10^{15} electrons/cm^2 flux level before assembly, was instrumented to record its short-circuit current. The third cell served as a radiation-resistant standard. Before flight, the output of this transducer was calibrated with the short-circuit current and open-circuit voltage of the array.

Plots of the flight performance of the $I_{SC} - V_{OC}$ transducers are shown in Figs. 23–25. The initial flight data from these cells read approximately 2 telemetry data numbers (about 3%) lower than the predicted values. This deviation is considered to be within acceptable limits of predictability. It is observed that the outputs of the cells generally follow their predicted time performance curves and do not indicate the occurrence

[6]Located on the panel spars.

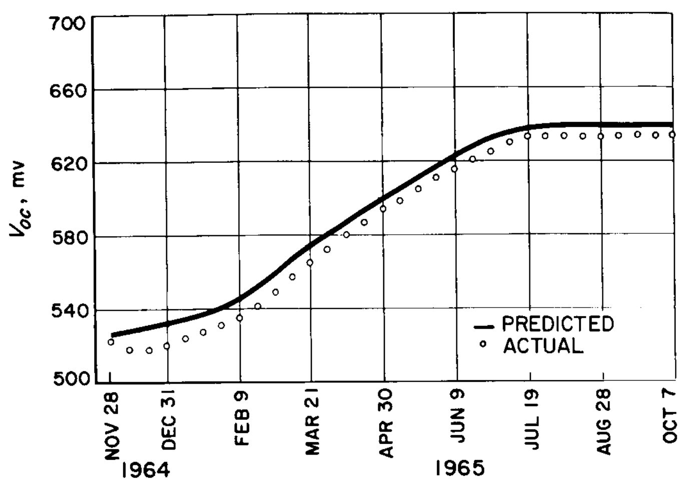


Fig. 23. Standard cell voltage

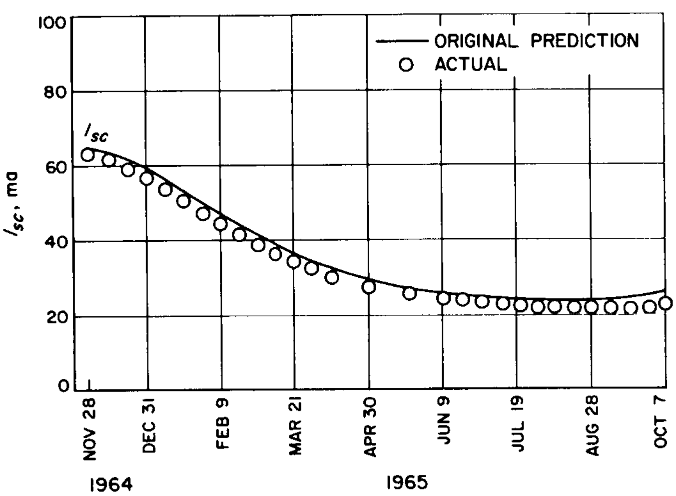


Fig. 24. Standard cell current

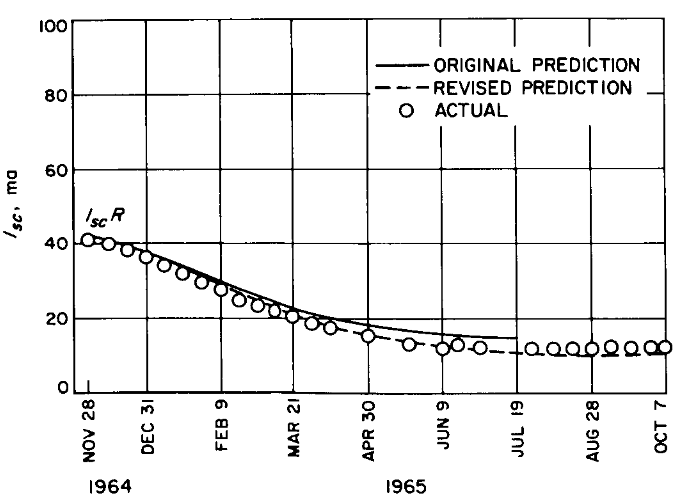


Fig. 25. Radiation resistant cell current

of degradation. This absence of degradation to the transducer offers additional confirmation of the undegraded condition of the array during flight.

b. Battery. During the prelaunch and launch activities the battery was used for 68 min. The total energy drain during this time was approximately 8 amp-hr. Immediately after solar panel Sun acquisition, the battery charger began charging the battery. Figure 26 shows the battery-voltage battery-charge current relationship after Sun acquisition. Five days after launch the battery voltage had risen to 34.8 vdc and the charge current had decreased to 0.010 amp. Because of the relatively small pitch turn of 39 deg during the midcourse maneuver, the solar panels supplied all spacecraft power and the battery was not required.

On February 11, 1965, the battery charger was turned off to prolong battery life and on February 19 the battery voltage reached a new high of 35.0 vdc. After that time the battery voltage increased steadily, except for brief periods during which changing spacecraft loads resulted in higher battery temperature and lower battery voltage, Fig. 27. The cause of this increasing voltage was the source of much concern throughout the flight. Two theories were proposed to explain the increase:

1. The increase was a normal consequence of the 0-g gravitational field, temperature effects, and the small (1.6 ma) charge current produced by the battery-voltage telemetry transducer.

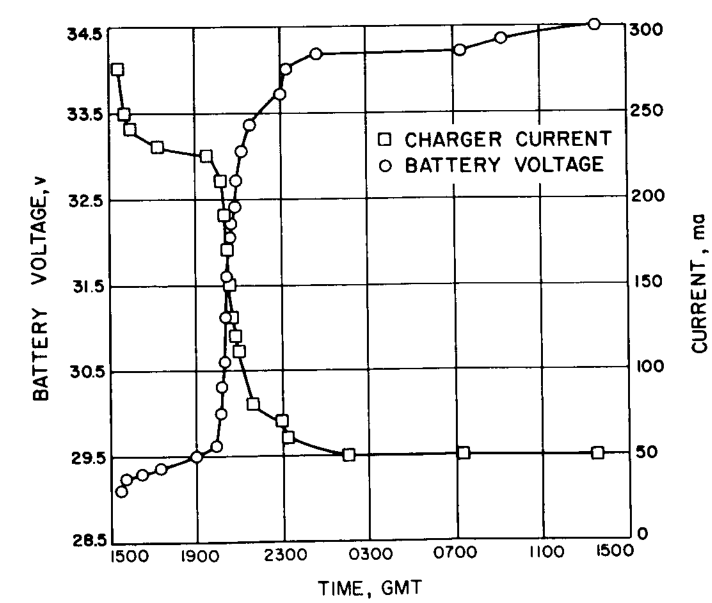


Fig. 26. Battery voltage and charge current after Sun acquisition

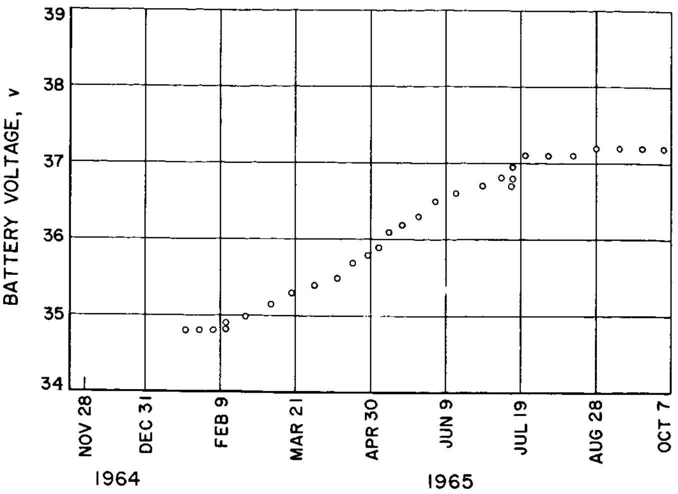


Fig. 27. Battery voltage increase during flight

2. The battery case had cracked and the electrolyte was leaking out slowly and evaporating, producing an open cell, which the battery-voltage telemetry transducer interpreted as increased voltage.

Because the voltage did not exceed 38.0 vdc, a limit at which theory 2 would become more likely, there is no way of determining which theory was correct. Numerous attempts were made to resolve the problem[7] but no solutions were reached. A definite answer to the battery-voltage anomaly, using *Mariner IV* data, appears unlikely. Only continued study of silver-zinc batteries may provide the answer.

c. Conversion equipment. Operation of the power conversion equipment was normal in all respects. During the flight the main inverter output voltage varied over the range 50.4 to 51.0 vac, the higher voltage being maintained during the low-power television picture-playback phase and the lower value being reached during the early magnetometer-calibration part of the flight. Before DC-15 was sent to inhibit the turn-on of the maneuver booster-regulator by the gyro-on signal as the result of an attitude control disturbance, it was turned on and off a total of 22 times by the attitude control subsystem. No problems were noted.

As a result of the science cover deployment exercise on February 11, 1965, and the actual encounter sequence on July 14, 1965, all elements of the power distribution assembly were checked and found to operate properly. All commands were processed in a normal manner.

Operating performance characteristics of the conversion system as a whole agreed well with both prelaunch system tests and manufacturing bench tests.

d. Power subsystem loading. In all cases power subsystem loading, as reflected on the battery or solar panel energy source, was identical[8] to the equivalent system test loading observed before launch. No anomalies were noted. Figure 28 is a plot of the actual spacecraft power demand-vs-mission phase.

3. Recommendations

Although operation of the *Mariner IV* power subsystem was satisfactory throughout the flight, several changes could be made to the design that would improve subsystem reliability and performance.

a. Three-phase inverter frequency control. The present 400-cps, three-phase inverter will not operate if the power subsystem synchronizer frequency-dividing network should fail. This could be corrected by driving the three-phase generation circuit directly from the 2400-cps main inverter-output instead of from the synchronizer frequency-dividing network.

b. Battery-voltage telemetry monitor. The battery-voltage telemetry monitor should be changed so that no current is injected into the battery. This would remove the undesirable trickle charge current into the battery that resulted in problems described earlier in the battery flight analysis section.

c. Power regulator changes. Several circuit changes could be made in the power regulators that would further improve transient response and lead to circuit simplification. Also, the booster-regulator's failure-sensing circuit could be simplified and made more reliable by a redesign.

D. Central Computer and Sequencer

1. Description

The *Mariner* CC&S subsystem is the central timing and synchronizing source on board the spacecraft. Spacecraft timing and synchronization are accomplished

[7]Including life tests with identical batteries, tests on individual cells, conferences with representatives of the battery manufacturers, etc.

[8]Within the tolerance of the telemetry measurements.

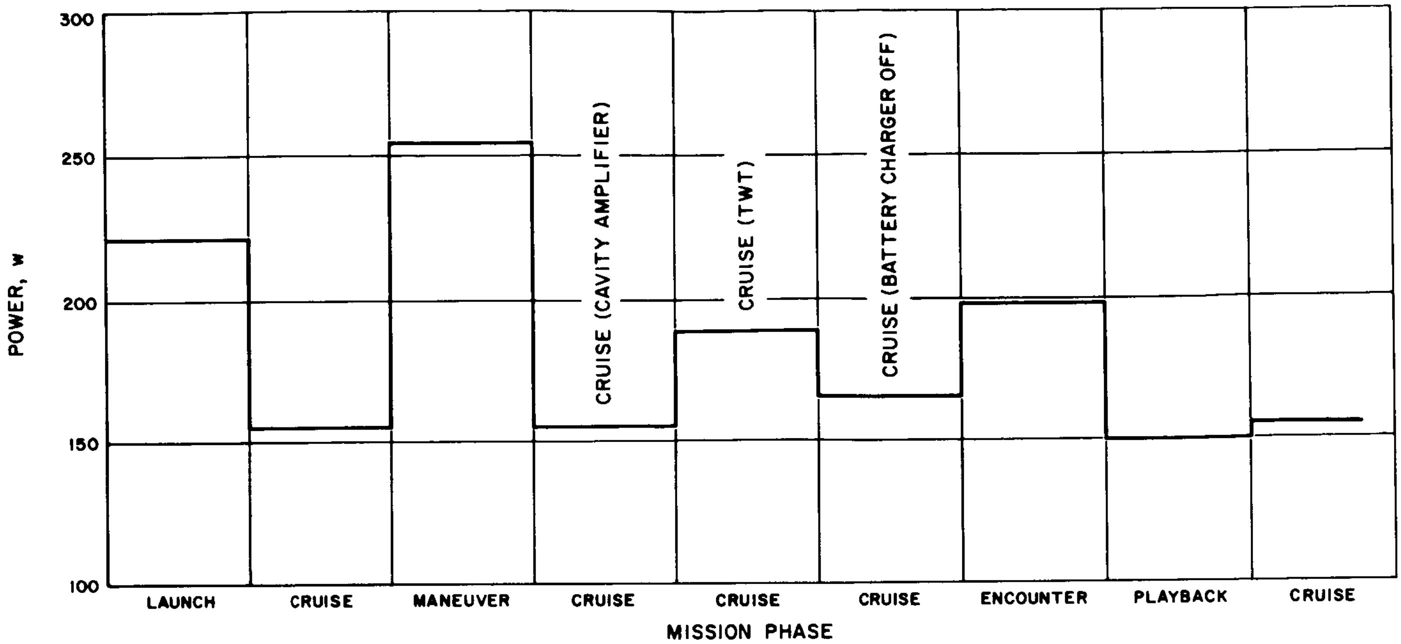


Fig. 28. *Mariner IV* power profile

through a CC&S-supplied 38.4-kc ($\pm 0.01\%$) signal to the power subsystem. The 38.4-kc signal is used as the frequency source for the power subsystem inverters, resulting in a common frequency reference for each subsystem.

In addition, the CC&S provides three commands shortly after launch to get the spacecraft into a cruise configuration. It is capable of receiving, storing, and executing commands for the trajectory-correction maneuver, and provides nine commands during the cruise portion of flight to automatically initiate mission-dependent functions at the proper time. A cyclic command (every 66⅔ hr) is also generated for the duration of the flight to provide an automatic switching capability to the radio subsystem under certain failure conditions.

2. Performance

All functional requirements specified for the CC&S were met during the flight, and operation continued to be normal. Table 15 lists the flight sequence of CC&S commands along with the observed times of occurrence.

Power was initially applied to the CC&S at approximately launch minus 4½ hr on November 28, 1964. Operation was uninterrupted and normal from that time to the termination of telemetry coverage. The CC&S 28v supply (Channel 309) remained at 27.8v throughout the flight.

The three launch-phase commands L-1, L-2, and L-3 occurred normally, as did the nine master-timer events (MT-1 through MT-9).

Two trajectory-correction maneuvers were initiated with the *Mariner IV* spacecraft, the first on December 4, 1964 and the second on December 5, 1964. The first maneuver was terminated after the loss of Canopus acquisition shortly after initiation of the sequence. The abort command (DC-13) did not inhibit the CC&S from issuing all the programmed maneuver commands successfully although it did inhibit the spacecraft from reacting to any of them. The second attempted midcourse maneuver was successful and was completed without incident. Tables 16 and 17 include event times for the two midcourse maneuvers.

a. Accuracy. CC&S clock accuracy is dependent on the temperature of the crystal oscillator. Below 72°F the oscillator frequency increases with increasing temperature, and above 72°F it decreases with increasing temperature. The particular crystal used in *Mariner IV* was slightly lower than its rated frequency of 38.4 kc at all temperatures, resulting in a negative frequency error.[9]

[9]I.e., the clock was running slow.

Table 15. CC&S commands

Event	Date	Expected time, GMT	Observed time, GMT
Inhibit Release	November 28, 1964		14:19:01
Launch			14:22:01
L-1 (deploy solar panels)		15:15:00	15:14:52 to 15:15:06
L-2 (Sun acquisition)		15:19:00	15:18:52 to 15:19:05
L-3 (Canopus acquisition)	29	06:59:00	06:58:49 to 06:59:00
Cyclic No. 1 (CY-1 No. 1)	29	16:59:00	16:59:00 to 16:59:14
CY-1 No. 2	December 2	11:39:05	11:38:56 to 11:39:10
1st (aborted) Midcourse Maneuver	4	(see Table 16)	13:05:00 to 17:54:08
CY-1 No. 3	5	06:19:09	No data
2nd (actual) Midcourse Maneuver	5	(see Table 17)	13:05:00 to 17:44:11
CY-1 No. 4	8	00:59:13	00:59:01 to 00:59:14
CY-1 No. 5	10	19:39:17	19:39:12 to 19:39:25
CY-1 No. 6	13	14:19:21	14:19:21 to 14:19:33
CY-1 No. 7	16	08:59:26	08:59:17 to 08:59:30
CY-1 No. 8	19	03:39:30	03:39:27 to 03:39:39
CY-1 No. 9	21	22:19:34	22:19:23 to 22:19:35
CY-1 No. 10	24	16:59:38	16:59:32 to 16:59:45
CY-1 No. 11	27	11:39:42	11:39:41 to 11:39:53
CY-1 No. 12	30	06:19:46	06:19:37 to 06:19:50
CY-1 No. 13	January 2, 1965	00:59:51	00:59:47 to 00:59:59
MT-6 (change bit rate)	3	16:59:54	16:59:54
CY-1 No. 14	4	19:39:56	19:39:12 to 19:40:03
CY-1 No. 15	7	14:20:01	14:19:26 to 14:20:17
CY-1 No. 16	10	09:00:06	08:59:36 to 09:00:26
CY-1 No. 17	13	03:40:11	03:39:45 to 03:40:36
CY-1 No. 18	15	22:20:16	22:19:56 to 22:20:46
CY-1 No. 19	18	17:00:21	17:00:05 to 17:00:55
CY-1 No. 20	21	11:40:27	11:40:16 to 11:41:07
CY-1 No. 21	24	06:20:32	06:20:26 to 06:21:16
CY-1 No. 22	January 27, 1965	01:00:38	01:00:37 to 01:01:28
CY-1 No. 23	29	19:40:44	19:39:58 to 19:40:48
CY-1 No. 24	February 1	14:20:51	14:20:10 to 14:21:00
CY-1 No. 25	4	09:00:57	09:00:21 to 09:01:11
CY-1 No. 26	7	03:41:04	03:40:32 to 03:41:23
CY-1 No. 27	9	22:21:12	No data
CY-1 No. 28	12	17:01:18	17:00:57 to 17:01:47
CY-1 No. 29	15	11:41:27	11:41:09 to 11:41:59
CY-1 No. 30	18	06:21:35	06:21:22 to 06:22:13
CY-1 No. 31	21	01:01:44	01:01:35 to 01:02:25
CY-1 No. 32	23	19:41:53	No data
CY-1 No. 33	26	14:21:02	14:22:03 to 14:22:54
MT-1 (Canopus update No. 1)	27	17:02:05	17:01:29 to 17:02:19
CY-1 No. 34	March 1	09:02:13	09:01:27 to 09:02:18
CY-1 No. 35	4	03:42:23	03:41:41 to 03:42:32
MT-5 (transfer to high-gain antenna)	5	13:02:25	13:01:47 to 13:02:37
CY-1 No. 36	6	22:22:34	22:21:56 to 22:22:47
CY-1 No. 37	9	17:02:44	17:02:12 to 17:03:03
CY-1 No. 38	12	11:42:55	11:42:28 to 11:43:18
CY-1 No. 39	15	06:23:06	06:22:44 to 06:23:34
CY-1 No. 40	18	01:03:18	01:03:00 to 01:03:50
CY-1 No. 41	20	19:43:29	No data
CY-1 No. 42	23	14:23:41	14:23:33 to 14:24:24
CY-1 No. 43	26	09:03:54	09:03:50 to 09:04:40
CY-1 No. 44	29	03:44:06	03:43:17 to 03:44:07
CY-1 No. 45	31	22:24:19	22:23:35 to 22:24:25
MT-2 (Canopus update No. 2)	April 2	14:24:25	14:24:25 to 14:25:15
CY-1 No. 46	3	17:04:32	17:03:52 to 17:04:42
CY-1 No. 47	6	11:44:45	11:44:11 to 11:45:01
CY-1 No. 48	9	06:24:58	06:24:29 to 06:25:19

Table 15. CC&S commands (cont'd)

Event	Date	Expected time, GMT	Observed time, GMT
CY-1 No. 49	April 12, 1965	01:05:12	01:04:47 to 01:05:38
CY-1 No. 50	14	19:45:26	19:45:05 to 19:45:55
CY-1 No. 51	17	14:25:40	14:25:24 to 14:26:14
CY-1 No. 52	20	09:05:54	09:05:43 to 09:06:33
CY-1 No. 53	23	03:46:09	03:46:03 to 03:46:53
CY-1 No. 54	25	22:26:23	22:26:21 to 22:27:11
CY-1 No. 55	28	17:06:38	17:05:51 to 17:06:41
CY-1 No. 56	May 1	11:46:54	11:46:11 to 11:47:02
CY-1 No. 57	4	06:27:09	06:26:30 to 06:27:21
CY-1 No. 58	7	01:07:24	01:06:50 to 01:07:41
MT-3 (Canopus update No. 3)	7	14:27:25	14:27:25 to 14:28:15
CY-1 No. 59	9	19:47:39	19:47:10 to 19:48:01
CY-1 No. 60	12	14:27:55	14:27:30 to 14:28:20
CY-1 No. 61	15	09:08:09	09:07:50 to 09:08:41
CY-1 No. 62	18	03:48:24	03:48:11 to 03:49:01
CY-1 No. 63	20	22:28:40	22:28:31 to 22:29:22
CY-1 No. 64	23	17:08:55	17:08:52 to 17:09:42
CY-1 No. 65	26	11:49:11	11:48:21 to 11:49:12
CY-1 No. 66	29	06:29:27	06:28:42 to 06:29:33
CY-1 No. 67	June 1	01:09:43	01:09:03 to 01:09:53
CY-1 No. 68	3	19:49:59	19:49:23 to 19:50:14
CY-1 No. 69	6	14:30:15	14:29:44 to 14:30:35
CY-1 No. 70	9	09:10:31	09:10:05 to 09:10:56
CY-1 No. 71	12	03:50:47	03:50:25 to 03:51:16
MT-4 (Canopus update No. 4)	14	15:50:57	15:50:54 to 15:51:45
CY-1 No. 72	14	22:31:02	22:30:46 to 22:31:36
CY-1 No. 73	17	17:11:18	17:11:06 to 17:11:57
CY-1 No. 74	20	11:51:34	11:51:27 to 11:52:17
CY-1 No. 75	23	06:31:49	06:30:58 to 06:31:49
CY-1 No. 76	26	01:12:05	01:11:19 to 01:12:09
CY-1 No. 77	28	19:52:21	19:51:39 to 19:52:29
CY-1 No. 78	July 1	14:32:36	14:31:59 to 14:32:49
CY-1 No. 79	July 4, 1965	09:12:52	09:12:19 to 09:13:10
CY-1 No. 80	7	03:53:08	03:52:40 to 03:53:30
CY-1 No. 81	9	22:33:23	No data
CY-1 No. 82	12	17:13:39	17:13:22 to 17:14:07
MT-7 (turn on encounter science)	14	15:53:49	15:53:15 to 15:54:05
MT-8 (turn off encounter science)	15	05:13:52	05:13:49 to 05:14:39
CY-1 No. 83 and MT-9 (start picture playback)	15	11:53:53	11:53:53.12 to 11:53:53.24
CY-1 No. 84	18	06:34:08	Arrived during Mode 4 data
CY-1 No. 85	21	01:14:23	Arrived during Mode 4 data
CY-1 No. 86	23	19:54:38	19:54:23 to 19:54:40
CY-1 No. 87	26	14:34:52	14:30:47 to 14:34:59
CY-1 No. 88	29	09:15:06	Arrived during Mode 4 data
CY-1 No. 89	August 1	03:55:21	Arrived during Mode 4 data
CY-1 No. 90	3	22:35:36	22:35:07 to 22:35:57
CY-1 No. 91	6	17:15:50	17:15:25 to 17:16:15
CY-1 No. 92	9	11:56:04	No data
CY-1 No. 93	12	06:36:18	06:36:03 to 06:36:53
CY-1 No. 94	15	01:16:32	01:16:21 to 01:17:12
CY-1 No. 95	17	19:56:46	19:56:40 to 19:57:30
CY-1 No. 96	20	14:36:59	14:36:08 to 14:36:59
CY-1 No. 97	23	09:17:12	09:16:25 to 09:17:15
CY-1 No. 98	26	03:57:26	03:56:44 to 03:57:34
QC-1		21:45:56	21:45:16 to 21:46:06
QC-2		21:54:20	21:53:42 to 21:54:32
QC-3		22:02:44	22:02:04 to 22:02:54
CY-1 No. 99	28	22:37:39	22:37:01 to 22:37:52
CY-1 No. 100	31	17:17:52	Arrived during Mode 4 data

Table 15. CC&S commands (cont'd)

Event	Date	Expected time, GMT	Observed time, GMT
CY-1 No. 101	September 3, 1965	11:58:03	11:57:35 to 11:58:25
CY-1 No. 102	6	06:38:16	06:37:52 to 06:38:42
CY-1 No. 103	9	01:18:28	01:18:09 to 01:18:59
CY-1 No. 104	11	19:58:40	19:58:25 to 19:59:16
CY-1 No. 105	14	14:38:51	14:38:42 to 14:39:32
CY-1 No. 106	17	09:19:03	09:18:58 to 09:19:49
CY-1 No. 107	20	03:59:15	03:58:24 to 03:59:15
CY-1 No. 108	22	22:39:26	22:38:40 to 22:39:31
CY-1 No. 109	25	17:19:37	17:18:56 to 17:19:47
CY-1 No. 110	28	11:59:47	No data
CY-1 No. 111	October 1	06:39:58	06:39:26 to 06:40:17

At launch the CC&S clock was running about 0.00070% slow. As the Bay VII temperature decreased during the flight (CC&S is mounted in Bay VII), the clock error increased as predicted, and on August 5, 1965 was running about 0.0011% slow. CC&S clock frequency error in percent as a function of time is plotted in Fig. 29.

On August 5, 1965 the CC&S clock had lost a total of 188.5 sec in 250 days, resulting in an accumulated clock frequency error of −0.00087%. CC&S clock error in seconds as a function of time is plotted in Fig. 30.

b. Bay VII temperature variations. At launch the Bay VII temperature was 73°F, reached a maximum of 75°F about 14 hr after launch, and stabilized at 64.4°F by 25 hr after launch. During the trajectory-correction maneuver the temperature went up from 64.4°F to a maximum of 71.2°F following the motor burn, and then back down to a stable 64.4°F about 5 hr after Sun reacquisition. During cruise, Bay VII temperature decreased from 64.4°F following the midcourse maneuver to 57.6°F just prior to encounter. During encounter the temperature rose to 59.6°F a few hours after encounter science was turned on and stabilized at 56.6°F following the turn-off of all science (MT-9). The temperature remained stable at 56.6°F until the end of picture playback and return to cruise mode configuration, whereupon it restabilized at 58.6°F. Bay VII temperature is plotted as a function of time on the reverse scale of Fig. 29.

Table 16. First trajectory-correction maneuver attempt (aborted)

Event	Time observed, GMT December 4, 1964	Best estimate of actual time, GMT
QC1-1 sent		13:05:00
QC1-1 verified by CC&S	13:06:06 to 13:06:18	13:06:07
QC1-2 sent		13:10:00
QC1-2 verified by CC&S	13:10:55 to 13:11:08	13:11:07
QC1-3 sent		13:15:00
QC1-3 verified by CC&S	13:15:57 to 13:16:11	13:16:07
DC-27 sent		14:35:00
DC-27 received by CC&S	14:35:47 to 14:35:53	14:35:53
Pitch start	15:34:59 to 15:35:11	15:35:08 + 1.42, − 0.42 sec
Pitch stop	15:39:11 to 15:39:24	15:39:19 + 1.42, − 0.42 sec
Duration		251 ± 1.84 sec
Duration from data encoder timer		249.9 ± 0.42 sec
Roll start	15:57:02 to 15:57:15	15:57:08 + 1.42, − 0.42 sec
Roll stop	16:11:19 to 16:11:32	16:11:20 + 1.42, − 0.42 sec
Duration		852 ± 1.84 sec
Duration from data encoder timer		851 ± 0.42 sec
Motor start	16:19:05 to 16:19:18	16:19:08 + 1.42, − 0.42 sec
Motor stop	16:19:18 to 16:19:31	16:19:29 + 1.42, − 0.42 sec
Duration		21 ± 1.84 sec
Duration from data encoder timer		20.6 ± 0.42 sec
Sun reacquisition start	16:24:58 to 16:25:11	16:25:08
CC&S counter overflow	17:54:01 to 17:54:15	17:54:08

Note: The estimate of the actual time of the start and stop commands of the pitch, roll, and motor burn was determined by extrapolation of the data encoder timer readings (Channel 220). The resolution given was determined as follows:

+1, −0 sec is the resolution of the time tags

±0.42 sec is the resolution of the data encoder timer

The total resolution is therefore:

+1.42, −0.42 sec

Table 17. Second trajectory-correction maneuver, December 5, 1964 attempt

Event	Time observed, GMT December 5, 1964	Best estimate of actual time, GMT
QC1-1 sent		13:05:00
QC1-1 verified by CC&S	13:06:05 to 13:06:18	13:06:08
QC1-2 sent		13:10:00
QC1-2 verified by CC&S	13:11:08 to 13:11:20	13:11:08
QC1-3 sent		13:15:00
QC1-3 verified by CC&S	13:15:58 to 13:16:11	13:16:08
DC-27 sent		14:25:00
DC-27 received by CC&S	14:25:47 to 14:25:52	14:25:51
Pitch start	15:25:10	15:25:10.6 ± 0.3 sec
Pitch stop	15:28:52 to 15:28:54	15:28:53 ± 1 sec
Duration		222.4 ± 1.3 sec
Duration from data encoder timer		223.4 ± 0.42 sec
Roll start	15:47:09 to 15:47:12	15:47:10.6 ± 0.3 sec
Roll stop	16:01:18 to 16:01:21	16:01:20 ± 1.3 sec
Duration		849.4 ± 1.6 sec
Duration from data encoder timer		850 ± 0.42 sec
Motor start	16:09:10 to 16:09:12	16:09:10.6 ± 0.3 sec
Motor stop	16:09:29 to 16:09:34	16:09:31 ± 1.1 sec
Duration		20.4 ± 1.4 sec
Duration from data encoder timer		20.58 ± 0.42 sec
Sun reacquisition start	16:15:10 to 16:15:12	16:15:10.6 ± 0.3 sec
CC&S counter overflow	17:44:07 to 17:44:21	17:44:10.6 ± 0.3 sec

Note: The estimate of the actual time of the start and stop commands of pitch, roll, and motor burn was determined by use of the analog recording from the high-speed data line. This allowed time tagging to within ±0.1 sec and improved the overall resolution somewhat from the TTY data (compare with Table 16). The pitch start command was tagged to within ±0.3 sec which was used to estimate the start times of roll and motor burn.

c. Operational anomalies. Four anomalies have occurred during the flight that may have been associated with the CC&S; none of the four affected spacecraft operation, however.

The first was observed, initially, coincident with Cyclic No. 34 and involved an apparent drop in received RF signal strength. Further checking indicated that this phenomenon had occurred previously and, in fact, occurred each time a cyclic was received while the spacecraft was being tracked in one-way lock. The problem was traced to an auxiliary oscillator in the radio subsystem that operated when in one-way lock, and would shift frequency slightly when a cyclic was received, due to additional loading on the power supply. The frequency shift appeared at the DSIF receiver as a drop in received signal strength. After it had been established that this was a design characteristic of the radio subsystem, no further action was necessary.

The second anomaly was a data encoder deck skip coincident with CC&S command MT-1 (first Canopus update). The data encoder has had a history at skipping decks, but the skips were usually caused by spacecraft power transients. No appreciable power transient should have occurred due to command MT-1 and the deck skip was not anticipated. Tests were run on the PTM in an attempt to duplicate the phenomenon but to no avail. Component failures were simulated in the CC&S to purposely introduce noise into the system, but no deck skips occurred. All other MT events occurred on the spacecraft without incident. The MT-1 command differs from the other MT commands in that a spare relay is set simultaneously with MT-1. No other MT event has a spare relay set coincident with it, but there is no obvious reason why two relays set at the same time should be any more apt to cause the data encoder to skip than just one. At the present time there is no precise explanation as to the cause of the data encoder deck skip.

The third anomaly occurred coincident with Cyclic No. 75, and involved erroneous readings in Event Register Channels 115 and 116. The cyclic was received by Station 41 while tracking in one-way lock, and the resultant drop in RF received signal strength caused an out-of-lock indication for several seconds. The out-of-lock indication resulted in bit errors being introduced into the data for several seconds and caused the channels being interrogated to indicate erroneous readings. Since the cyclic occurred during the interrogation of Channel 115, the erroneous readings in the event registers of Channels 115 and 116 were assumed to have been the result of the out-of-lock condition. Similar errors were noted in other telemetry channels coincident with cyclic occurrences. Subsequent event register readings were normal.

The fourth anomaly concerned the receipt of two events in the CC&S event register (Channel 115 Register 2) following MT-9, instead of the normal and expected indication of one event. Since there was a

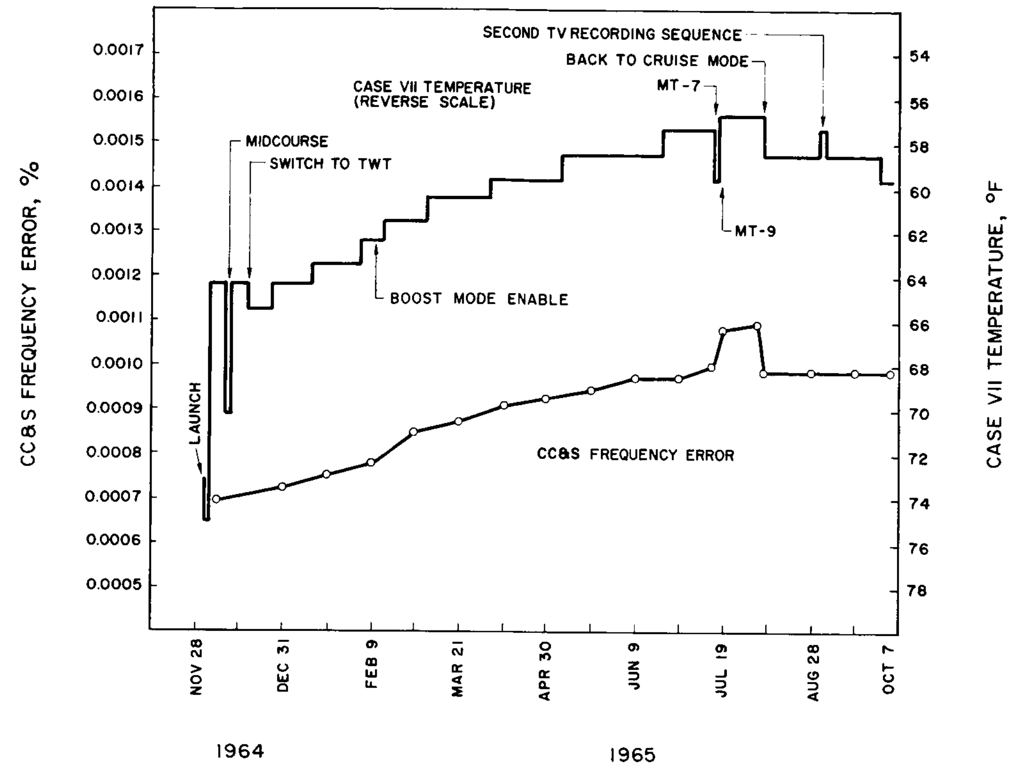


Fig. 29. CC&S clock frequency error

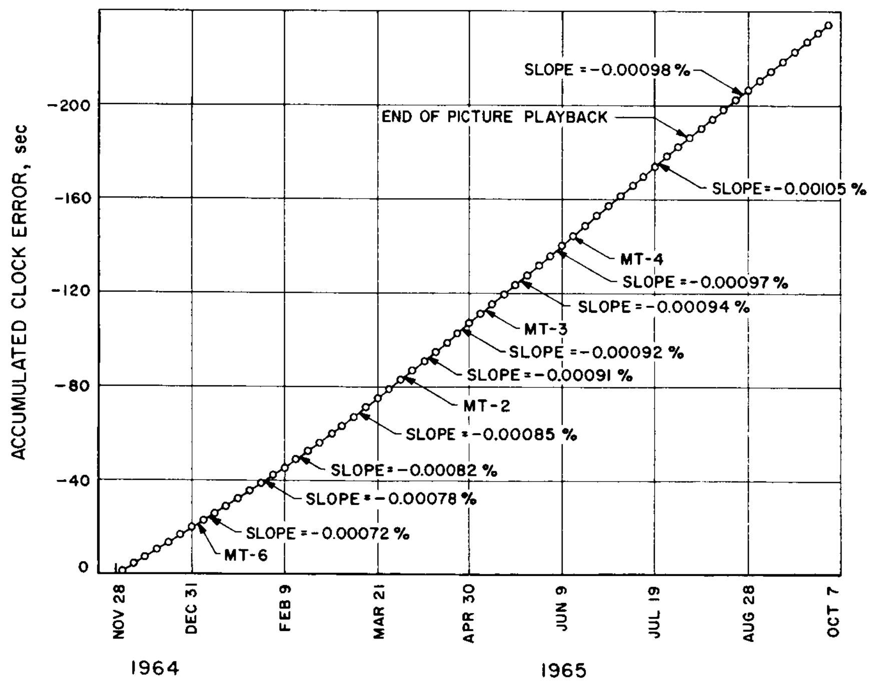


Fig. 30. Cumulative CC&S clock error

cyclic command (Cyclic No. 83) coincident with MT-9 and two separate relays were involved (one spring loaded and one latching), the assumption is that the two relay closures may have occurred at slightly different times and caused the generation of two events. For the data encoder to react to two event pulses, the pulses must be at least 2 msec apart, (assuming proper data encoder operation), which means that the two relay closures would have had to be at least 2 msec apart; however, the inputs to the two relay drivers are within a few μsec of each other.

Previous testing in the CC&S lab has indicated that relays kept in storage for several months without being exercised, such as was the case with the MT-9 relay, tend to switch more slowly the first time than do relays exercised periodically.[10] It has not yet been determined whether this phenomenon could create enough difference in switching time to cause the two events. Another source of difference in relay switching time is associated with the different relay types involved (one spring loaded and one latching), but this difference is small compared to 2 msec. Further analysis is being done in an attempt to better understand the problem.

d. Future operational capability. All command functions of the CC&S have been completed with the exception of the 66⅔-hr cyclics. The cyclic commands will continue, barring an associated circuit malfunction, and the CC&S clock will continue to provide timing and synchronizing information to the spacecraft unless interrupted in a similar manner. Although the capability for only one more motor burn exists, any number of midcourse maneuver sequences can be initiated and executed by the CC&S.

3. Recommendations

The functional design of the *Mariner* CC&S was adequate, although some packaging and circuit design changes could be made to make the unit more reliable. The accomplishment of the improvements listed below would require a major redesign of the CC&S, however. Recommendations for improvements of future equipment are listed below in order of importance.

1. Redesign of packaging and potting techniques of magnetic shift registers and counters
2. Elimination of back-side printed circuit wiring
3. Reduction of component density to eliminate stacking and provision for specified 3/16-in. body-to-terminal lead lengths
4. Provision of proper spacing for subassembly harnessing
5. Replacement of shift registers with units having wider margins and not requiring sensistor temperature compensation
6. Redesign of input decoder subassembly to reduce complexity
7. Redesign of magnetic counters to increase margins so that initial test time is reduced and the need for optimization of adjustments is eliminated

[10]The cyclic relay was exercised every 66⅔ hr.

E. Attitude Control Subsystem

The attitude control subsystem establishes and maintains three-axis stabilization of the spacecraft using the Sun and the star Canopus as references. Two-axis Sun stabilization aligns the spacecraft Z-axis with the spacecraft–Sun line, keeping the solar panel surfaces presented fully to the Sun. Roll stabilization about the Z-axis with the star Canopus as the reference insures that the high-gain antenna beam will include Earth during the latter portions of the mission.

The attitude control subsystem orients the spacecraft to the commanded attitude during the midcourse trajectory correction and maintains it during the motor firing interval. After the correction, Sun and Canopus acquisition are re-established.

In addition to the normal cruise control, a number of alternate modes of operation including inertial roll control are designed into the attitude control subsystem.

1. Description

The variety of requirements placed upon the attitude control subsystem during the mission dictate that the subsystem consist of a number of separate functional units each operating in a manner appropriate to the attitude control mode required for any particular portion of the mission. The largest part of the *Mariner* mission is flown in the interplanetary cruise mode. In cruise the attitude control subsystem must orient and maintain the X- and Y-axes normal to the Sun line and at the desired angular position about the Z-axis. The primary means of control is a system of cold gas thrusters.

a. Gas jets. The spacecraft is maintained at certain angular positions, or is rotated at prescribed angular rates through the application of appropriate torques obtained by the expulsion of gas through pairs of jet nozzles, located so as to produce couples about each of the principle axes. The gas jets are opened and closed by solenoid valves operated by switching amplifiers. The switching amplifiers respond to signals from sensors which measure spacecraft angular position and rate.

The gas supply is contained in two completely independent nitrogen-gas systems. Assuming flight equipment is well-enough controlled during manufacture, assembly, and testing to preclude any severe weaknesses or manufacturing flaws, the most probable failure mode would be loss of gas pressure due to random sticking of the gas valves. The redundancy of gas valves between the two half-systems permits normal operation if a valve fails closed; and if it fails open there are two opposing valves to counteract it, one in its own half-system and one in the redundant half-system. After all of the gas in the failed half-system has been expended, enough remains in the redundant half-system to allow the completion of the mission.

b. Inertial sensors. Gyros provide error signals proportional to angular rate. The rate error signal from a gyro may be fed to a switching amplifier to maintain an angular rate within the rate deadband orientation. Angular rates other than zero are obtained either by torquing the gyros or by providing a bias signal to the switching amplifiers. Torquing the gyro at a constant rate will produce a rate error signal, operating the gas jets, until the spacecraft is rotating at the same rate at which the gyro is being torqued. The rate error signal will then go to zero. A fixed-rate bias signal to the switching amplifier will operate the jets until a rate is established in which the gyro rate error signal just cancels the bias signal.

The rate error signal from the gyro may be integrated to give position error. When operating the switching amplifier from position error signals, the gyro rate error signals may also be used as damping signals to damp out spacecraft oscillations about the desired position.

c. Celestial sensors. Sun sensors produce error signals proportional to angular position displacement about the pitch (X) and yaw (Y) axes. Similarly, the Canopus star sensor will, when a star is in its field of view, produce error signals proportional to the angular displacement of the star from the sensor boresight line (only the angular position error component about the Z-axis is sensed).

If these three-position error signals are fed to the switching amplifiers, they cause gas jet firings which accelerate the spacecraft so as to reduce the errors. Hence three-axis stabilization is achieved. In the absence of a damping force, however, the spacecraft will continue to oscillate about the null positions indefinitely. The rate signals from the gyros might be mixed with the position error signals to damp out the oscillations. However, since the gyros are not operating throughout most of the flight, another source of damping must be found.

When the gas jet pairs are opened in response to error signals, the regulated flow of gas produces a constant torque, hence a constant angular acceleration.

The attitude control gyro power is turned on prior to launch. The attitude control subsystem power is turned on when the spacecraft separates from the *Agena* stage. The turn-on is one function of the pyrotechnic arming switch. The CC&S also issues a backup command to turn on the attitude control subsystem power at lift-off plus 57 min.

Alignment of the Z-axis with the spacecraft–Sun line begins at this turn on of power. Acquisition Sun sensors looking away from the normal Sun direction supplement the cruise Sun sensors so that the Sun is in the field of view of at least one set of Sun sensors regardless of the spacecraft angular orientation. The Sun sensors produce angle-position error signals which cause the switching amplifiers to open the appropriate gas jets. The thrusts from the jet pairs produce torques which rotate the spacecraft and thus reduce the angular position errors. Oscillations are damped out by mixing a rate damping signal from the rate gyro with the position error signal. Sun stabilization occurs within approximately 20 min depending upon spacecraft initial angular position and rate.

When the Z-axis is aligned with the spacecraft–Sun line, the Sun gate is illuminated and switches the acquisition sensors off; stabilization is maintained by the cruise Sun sensor signals. The Sun gate signal also enables the roll-spin or roll-search mode.

After pitch- and yaw-axis Sun stabilization has been accomplished, a controlled turn rate about the Z-axis at either one of two rates is executed. Normally, Sun stabilization will occur well in advance of CC&S L-3 (T + 997 min). A spin signal is then fed to the switching amplifier firing the roll jets and accelerating the spacecraft to a constant 3.5 milliradians/second (mrad/sec) rate. The roll spin provides calibration data for the magnetometer.

At T + 997 min, the Canopus sensor is turned on by CC&S L-3. L-3 also switches the input of the amplifier from the spin signal of 3.5 mrad/sec to the search signal of 2 mrad/sec. The spacecraft decelerates appropriately, then remains in the search mode until a star of sufficient brightness appears in the field of view of the Canopus sensor, which then initiates star acquisition and switch-to-cruise mode.

An inoperative gyro failing to provide a rate signal to buck out the spin (or search) signal would cause the spacecraft to continue to accelerate indefinitely. To prevent such a failure from being catastrophic, the jet firing times are electrically integrated. After a jet firing time slightly in excess of normal, the input signal (spin or search) to the switching amplifier is interrupted, allowing back-up of roll control to be used.

A star of sufficient brightness in the field of view of the Canopus sensor will operate the acquisition gate. The signal from this gate places the spacecraft in the cruise mode, turning off all gyros and switching in derived-rate networks. The acquisition gate signal also switches the Canopus sensor error signal into the switching amplifier in order to provide roll position control.

The long cruise period of almost 8 mo created a desire to enhance reliability by using a passive stabilization technique in place of the gyros. A system of proved performance had been developed for the *Mariner II* Venus spacecraft, called *derived rate switching amplifier compensation,* which was used on *Mariner IV*. Fundamentally, it is a lag feedback around the switching amplifier with a long time constant of about 100 sec. Whenever there is a switching amplifier output to activate the gas jets, the long time constant produces a feedback signal to the input. While the gas jets are on, the spacecraft is accelerating and the feedback appears the same as a rate feedback obtained by gyro integration of acceleration (thus the term *derived rate*), provided that: 1) the gas jets are on for a much shorter period than the derived rate time constant, and 2) the initial spacecraft rate is nearly zero, because the derived rate signal is not correlated with the actual spacecraft rate. For conditions other than these, the control system damping is increasingly less efficient. The derived rate system, however, is capable of controlling any cruise mode disturbance. For single-axis operation, the system has enough acquisition capability to replace a gyro that is providing no output, though it is quite inefficient in that mode.

A cruise mode limit cycle is established by controlling the maximum angular excursion by means of the switching amplifier deadband, and by controlling the minimum rate increment by means of a minimum-on-time circuit in the switching amplifier. That is to say, whenever the angular error exceeds the deadband value, the gas jets are activated to reverse the direction of motion. In addition, the gas jets are caused to remain on for a minimum of 20 msec even though the input signal is of less duration. The 20 msec minimum-on-time circuit establishes the minimum torque impulse capability of the control system. The derived rate damping will reduce the angular rates to the minimum impulse level, which is about ± 2 deg/hr on either side of null. The angular position deadband on either side of null is 0.5 deg for pitch and yaw and 0.25 deg for roll.

Direct command DC-15 (acquisition gate override) provides a means of bypassing the acquisition gate, thereby stabilizing roll position using any star bright enough to produce position error signals. After normal star acquisition, direct command DC-21 (roll override) provides a means of interrupting the acquisition gate signal, thus commanding further roll search for another star.

Additional options are afforded by DC-18 (inertial roll control) which places the roll axis only in inertial control while pitch and yaw are still controlled in position by the Sun sensors, and by DC-20 (roll drift) which removes all roll control. In the inertial roll control mode the roll position may be commanded from the ground in 2¼ deg increments. After the first DC-18 initiates the inertial roll mode, each subsequent DC-18 will trigger a one shot which applies a positive command current into the roll-gyro torquer for a time equivalent to a positive 2¼ deg roll turn. A DC-21 command will trigger another one shot, giving a negative 2¼ deg turn.

Each of the alternate modes – inertial roll control, roll drift, or acquisition gate override – may be reset and normal control established through the use of an additional command, DC-19 (normal roll control).

d. Trajectory-correction maneuver. During the trajectory-correction sequence the spacecraft is placed in inertial control on all axes.

This control mode maintains spacecraft angular positions in inertial space, under control of the gyros. The mode is commanded by the CC&S and is maintained throughout the maneuver sequence.

A large integrating capacitor is inserted in the gyro feedback loop to the torquer, integrating the gyro rate signal output and thereby providing position information. Whenever some disturbance produces a spacecraft angular rate, the gyro pick-off amplifier provides a corrective signal to fire the jets and also an electrical current to the gyro torquer. The flow of torquer current produces a charge accumulation on the capacitor proportional to the position error introduced. When the jets have reduced the rate to zero, the capacitor will deliver torquing current of opposite polarity. This current produces a rate opposite that of the initial disturbance. Torquing current will cease to flow when the capacitor charge (and hence the position error) is reduced to zero.

During the period of PIPS rocket motor burn, the rate, plus position-error signal is used by the autopilot to maintain stabilization by appropriate positioning of four jet vanes in the rocket exhaust.

The rocket motor thrust vector is aligned for the trajectory correction by the performance of commanded pitch and roll turns. The desired thrust vector orientation is determined on the ground from tracking data and the necessary turn information is sent by radio command and stored in the CC&S. After maneuver initiation, the CC&S commands generate a torquing current to produce a precision current that will torque the gyro at 0.18 deg/sec. The polarity of the current, hence the direction of the turn, is determined by the presence or absence of an additional CC&S command. The precision torquing current torques the gyro, producing a gyro output which opens the appropriate jets, accelerating the spacecraft. When the spacecraft angular velocity reaches 0.18 deg/sec, the error signal goes to zero, the gas jets are closed, and the spacecraft continues to turn at a constant rate. The duration of the turn (and hence the magnitude of the turn angle) is determined by the timing functon of the CC&S counter, which, at overflow, commands an interruption of the flow of torquing current. The spacecraft angular rate then produces a gyro output which opens the appropriate jets until the rate is reduced to zero. This new orientation is then maintained by the spacecraft in the inertial control mode.

The midcourse autopilot is used to control the attitude of the spacecraft during the motor burn interval. Control is accomplished by continuous adjustment of the angular positions of four jet vanes mounted in the midcourse motor nozzle. Since the motor is not mounted along any of the spacecraft body axes, the motion of each jet vane is controlled by a mixture of the signals of the three gyros.

Each jet vane has its own control system, consisting of an autopilot amplifier, a jet vane actuator and a feedback loop. Power to the autopilot is switched on only during the midcourse correction sequence. The three gyro signals and the feedback signal are summed in different ratios at the input to each amplifier. The jet vanes are then adjusted so that the motor thrust vector passes through the spacecraft center of gravity, nulling the gyro error signals.

e. Solar pressure vanes. An auxiliary two-axis Sun-stabilization system employs solar pressure to provide a restoring torque. Solar pressure vanes are automatically positioned to balance the solar pressure about the spacecraft center of gravity at the correct spacecraft Z-axis alignment.

Every time the pitch or yaw jets are opened, the solar vane electronics operate solar vane actuators which make step changes in the angular position of the appropriate solar vanes.

As opposite solar vanes are stepped toward and away from the plane normal to the Sun line, the solar center of pressure is shifted, creating a solar torque of the same sense as the opened jet as shown in Fig. 31.

The gas jets fire only when the position error has exceeded the deadband limit, for a spacecraft in normal cruise mode. In normal limit cycle operation, the limit on one side of the null position is reached as many times as the limit on the other side. If a torque bias exists, however, the number of times that a correction of one polarity is required exceeds that for the other. The differential stepping of the solar pressure vanes shifts the center of pressure in such a manner as to oppose the bias torque, driving the mean position error to zero.

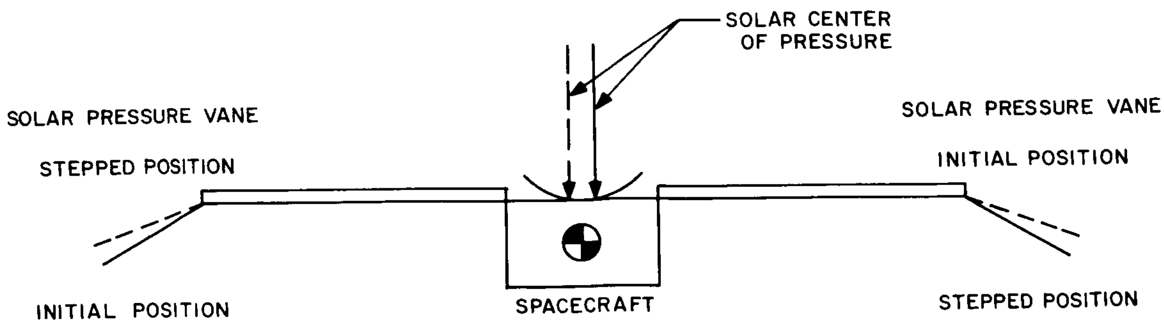


Fig. 31. Solar pressure vanes

2. Performance

The attitude control subsystem performed in a satisfactory manner throughout the mission. Sun acquisition was maintained without incident throughout the mission except during the midcourse trajectory correction sequence. Some problems were encountered in maintaining Canopus acquisition until an alternate control mode was initiated with the transmission of a ground command disabling the automatic reacquisition logic in the star tracker. There were no other anomalies observed during the flight which might have placed any of the mission objectives in jeopardy, although problems were observed in the solar pressure vane system operation.

a. Initial Sun acquisition. Separation of the spacecraft from the *Agena* occurred in the Earth shadow 15:07:10 GMT on November 28, 1964. There were no Sun sensor error signals in the shadow, but the three single-degree-of-freedom gyros about each spacecraft axis sensed the tumbling of the spacecraft and turned on the attitude control jets until the rates were reduced to within the deadband of the switching amplifier. The rate deadband was ± 1.27 mrad/sec in pitch and yaw and ± 0.33 mrad/sec in roll. When post-separation telemetry first became available, the rates were already within the deadband, indicating that the separation rates were very low.

The actual time of exit from the Earth shadow took place at 15:17:35 but Sun sensor signals were first noted at 15:16:37 GMT due to the refraction of sunlight through the Earth atmosphere.

When the spacecraft was completely in the sunlight, the pitch error was seen to be equal to -1.0 deg, and for all practical purposes the pitch axis had acquired. The yaw Sun sensors were saturated and had established a rate determined by this saturation equal to -2.2 mrad/sec (-0.126 deg/sec). The yaw sensors came out of saturation at 15:29:56 GMT at an angle of $+11.5$ deg. This indicated that at the start of yaw acquisition, the yaw angle was about $+106$ deg.

A Sun gate event occurred at 15:30:59 GMT. This indicated that the Sun gate cone-angle field of view was 2.2 deg, which is within the design tolerance.

The Sun gate initiated the magnetometer calibration roll as required. A search signal was switched into the roll axis control system and the spacecraft accelerated to a rate of -3.55 mrad/sec (-0.203 deg/sec), which was the design nominal value.

During the magnetometer roll mode, the pitch and yaw errors were observed to limit cycle against one side of the deadband. The pitch axis stayed at $+8.7$ mrad edge and yaw at the -8.5 mrad edge. This was as expected and was due to crosscoupling from the cross-products of inertia.

b. Star acquisition. Before launch it was felt that star identification would be a major problem. The only information from the Canopus sensor, other than an error signal, was a brightness measurement and a knowledge of the cone angle of an object within ± 5 deg. The absolute calibration of the Canopus sensor for all the stars in the sky was not known and the problem was further complicated because the brightness signal included the integrated background of stars in the field. However, during the roll-search mode, the tracker made good relative measurements of the star, plus background brightness. This information was basic to the star identification procedure.

A map-matching technique was developed to identify objects seen by the Canopus sensor during the roll search mode. As an aid to establishing the map-matching technique, other corroborating information was used for initial acquisition. This included a fixed, wide-angle (26 deg by 83 deg) field-of-view Earth detector to provide an output if Canopus was in the star sensor field of view; magnetometer information obtained during the calibration mode for rate and position information; and low-gain antenna pattern variations to also provide rate and position information. The magnetometer and antenna information was relatively crude and could only be used with a low weighting function on their corroboration.

Fundamental to the map-matching was an *a priori* telemetry map of sensor brightness output versus clock angle (angle about the Sun line measured from Canopus). A reasonably sophisticated mathematical model of the Canopus tracker and the sky, including the Milky Way, was developed so that with trajectory information, a 7094 program printed a map of the expected telemetry output of the brightness channel seen during roll search. This then was matched with an actual telemetry map to identify observed objects.

Initially, an extensive computer program was devised to process the telemetry data to produce the actual telemetry map. This was statistically correlated with the *a priori* map, Earth detector output, magnetometer data, and low-gain antenna data. Whenever an object was acquired subsequent to a roll search, a calculation was made of the probability that each acquirable object had been acquired. If the object was not Canopus, a roll override command would institute another roll search and another computer run was made until Canopus was identified as acquired.

The computerized version of the map-matching was necessary due to the lack of knowledge of the tracker response to the innumerable objects and integrated background that would be seen by the tracker. Until the tracker was actually calibrated in flight, the uncertainties required that the best possible analysis techniques be prepared beforehand.

In addition to the computerized star identification program, a second map-matching technique was developed and implemented during the flight which was much simpler, but required actual flight experience to prove reliability. A continuous strip-chart recorder was employed to plot in real time the star tracker brightness telemetry. An *a priori* map derived from the computer program was transcribed to a transparent overlay in the same scale as the real time telemetry plot so that they could be instantaneously compared during the roll search. The process turned out to be surprisingly good and became the primary technique for star identification. It was quite accurate and the fastest possible way of identifying objects. One of the mission accomplishments can be said to be the development of the procedure consisting of excellent *a priori* maps matched in real time with brightness telemetry data, as shown in Fig. 32.

Initial roll acquisition was commanded by the CC&S to take place at 06:59:00 GMT on November 29. Just before this command (CC&S L-3), the spacecraft was in the magnetometer calibration mode, rolling at a rate of -3.55 mrad/sec (-0.203 deg/sec). At the roll acquisition command, the search command signal was switched in and the roll rate was reduced to -2.35 mrad/sec (-0.135 deg/sec), the upper edge of the deadband about the nominal search rate. An acquisition occurred at 07:07:47 GMT. At this point, Earth light reflections on the Canopus sensor optics caused a bright enough background in conjunction with a spurious error signal to cause the acquisition logic to discontinue the roll search. Even though there was no acquirable object, a "logic" acquisition had been effected. The clock angle at this point was about 129 deg.

Even though the logic circuitry was satisfied, there was no real error signal and the roll axis was essentially in a drift mode with the attitude control gas jets firing at random due to background noise. The cumulative action caused the spacecraft to drift primarily in a direction of increasing clock angle. At some unknown time during the drift, the star Alderamin was acquired at a clock angle of 171 deg, after drifting about 42 deg. Alderamin was about 0.05 times as bright as Canopus, with a bright background, and the error signals were too noisy to determine the time of acquisition.

On November 29, at 13:12:34 GMT, the spacecraft went into an automatic roll search. At 13:26:15 GMT after 13 min and 41 sec of roll search, the star Regulus was acquired at a clock angle of 278 deg. Lock was maintained on Regulus until the next day, since Canopus acquisition was not necessary to the mission at the time.

At 09:13:47 on November 30, DC-21 (roll override) was sent and the star Naos was acquired at 09:20:56 GMT, at a clock angle of 338 deg. Another DC-21 at 10:45:09 GMT caused γ Velorum to be acquired at a clock angle of 343 deg at 10:46:00 GMT. Canopus was acquired by a DC-21 initiated at 10:57:57 GMT and lock was obtained at 10:59:38 GMT.

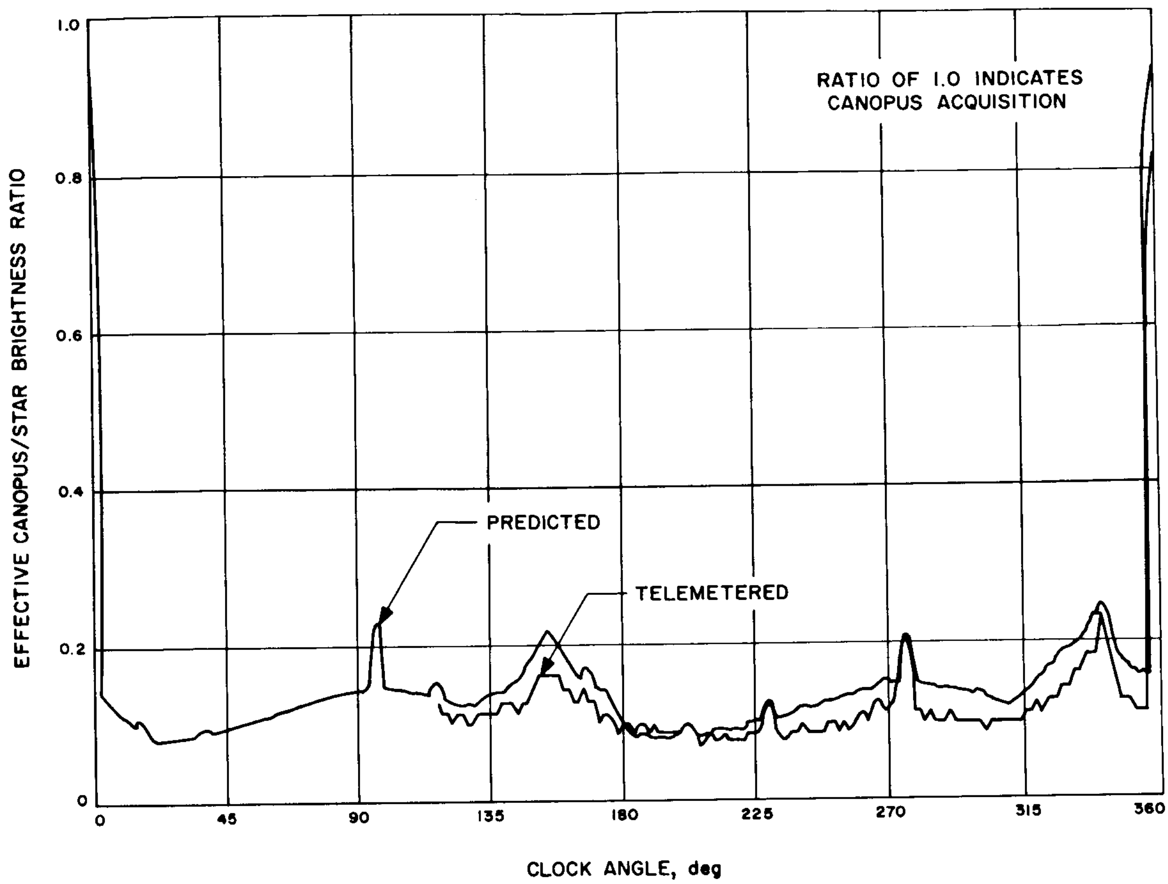


Fig. 32. Star map of December 4, 1964 reacquisition

c. Solar pressure vane operation. The first solar pressure vane measurements were coincident with the turn-on of the adaptive mode actuators at the time of CC&S L-3 (initial star acquisition). It was then observed that the vanes had been deployed beyond the nominal position angle of 35 deg below the plane of the solar panels. They were deployed in the following positions: 1) +X at 21 deg, 2) −X at 15 deg, 3) +Y at 17 deg, and 4) −Y at 7 deg.

The deployment failure was due to an unsuspected source of friction that was present during ground testing, but not present in the space environment. In spite of the overtravel, the center of pressure of the spacecraft was behind the center of gravity such that there was a net restoring torque of about 1.1 dyne-cm/deg about the pitch axis and 1.7 dyne-cm/deg about yaw.

In the adaptive mode of operation, stepping motor actuators rotated the solar pressure vanes about their deployed positions to cancel out unbalanced solar torques. An unbalanced torque would cause the valves for one direction to fire more frequently than the ones for the other direction. The vanes were stepped 0.01 deg by the actuator for each valve firing, in phase with that firing. The differential firing of the valves caused the vanes to be rotated in a direction to cancel the torque causing the difference.

The +X and −X vanes, which control the yaw axis, worked properly in the adaptive mode by canceling out an initial unbalanced torque of about 25 dyne-cm, probably due to the skewed angle of the high-gain antenna. The +Y and −Y vanes were definitely in a failed mode. The most likely explanation is that the actuators were locked up electrically. The design was such that within the first 30 msec of actuator turn-on, if a gas jet fired, the actuator would lock up. The telemetry data indicated that it was very likely that a jet fired at that time.

The thermal actuators were designed to provide damping such that after the stepping motor actuators cancel out the unbalanced torque, the limit cycle damps out within the deadband of the gas subsystem. The thermal actuators are exposed differentially to sunlight by a Sun shade so that the varying angle with respect to the Sun line causes the vanes to be rotated in a direction to reduce slightly the restoring torque. This should have caused the limit cycle oscillation to be damped out eventually.

Because the +X and −X vanes were operating in the adaptive mode to reduce the unbalanced torques, they were expected to eventually operate in the thermal mode to damp out the limit cycle. After the unbalanced torque was reduced to about 5 dyne-cm, the disturbance torques proved to be quite variable. The torques changed by as much as 5 dyne-cm between valve firings. Over longer periods, such as several weeks, the disturbance torques varied as much as 25 dyne-cm. The maximum restoring torque of the spacecraft over the ±½-deg limit cycle was 0.85 dyne-cm and the vane damping was 0.30 dyne-cm[11]. It was not possible for the thermal actuators to perform their function under these circumstances.

[11]This represents degraded performance due to overtravel at deployment.

These varying disturbance torques were obviously not solar torques. It is probable that they were due to the normal leakage of the gas valves. The normal valve leakage of 3.0 cc/hr at standard conditions actually provides 16 dyne-cm/valve of torque if it is assumed that the nominal gas laws apply. There were 4 valves controlling each axis, and the leakage has been observed in testing to vary with valve actuation. Therefore, even though the average leakage was known, it was obvious that the leakage, and hence the torques, could vary considerably over a period of time. If the seating characteristics of the valves changed over a period of time, long-term variations in the disturbance torques would be expected, and as has been discussed, if the seating for each valve firing was slightly different, then short-term variations would be observed.

Since the long-term disturbance torques did vary, it should be noted that the +X and −X vanes were observed moving and canceling out these disturbances in the adaptive mode of operation, Fig. 33. Figure 34 shows a representative limit axle plot for all three axes during cruise.

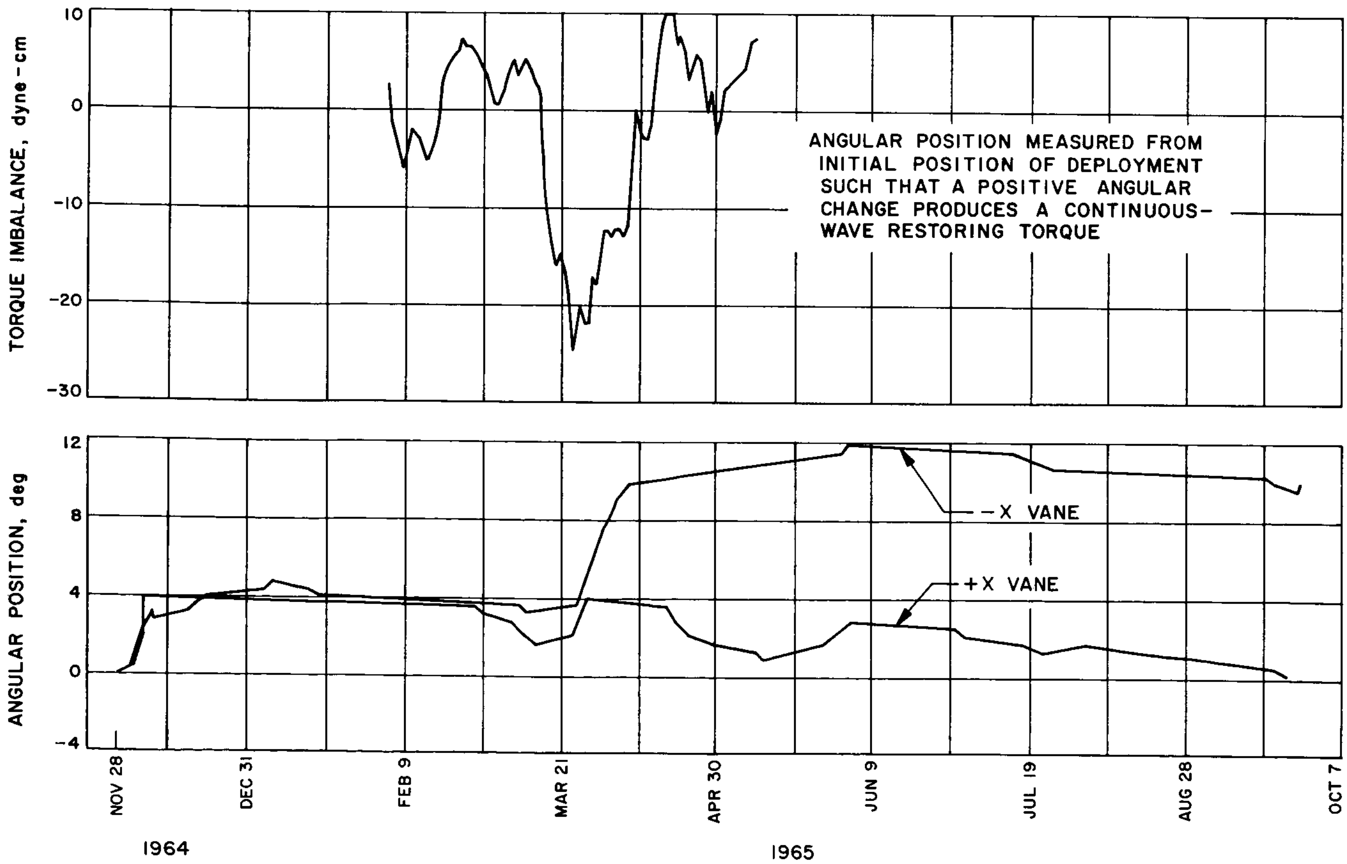


Fig. 33. +X and −X solar pressure vane movement during mission

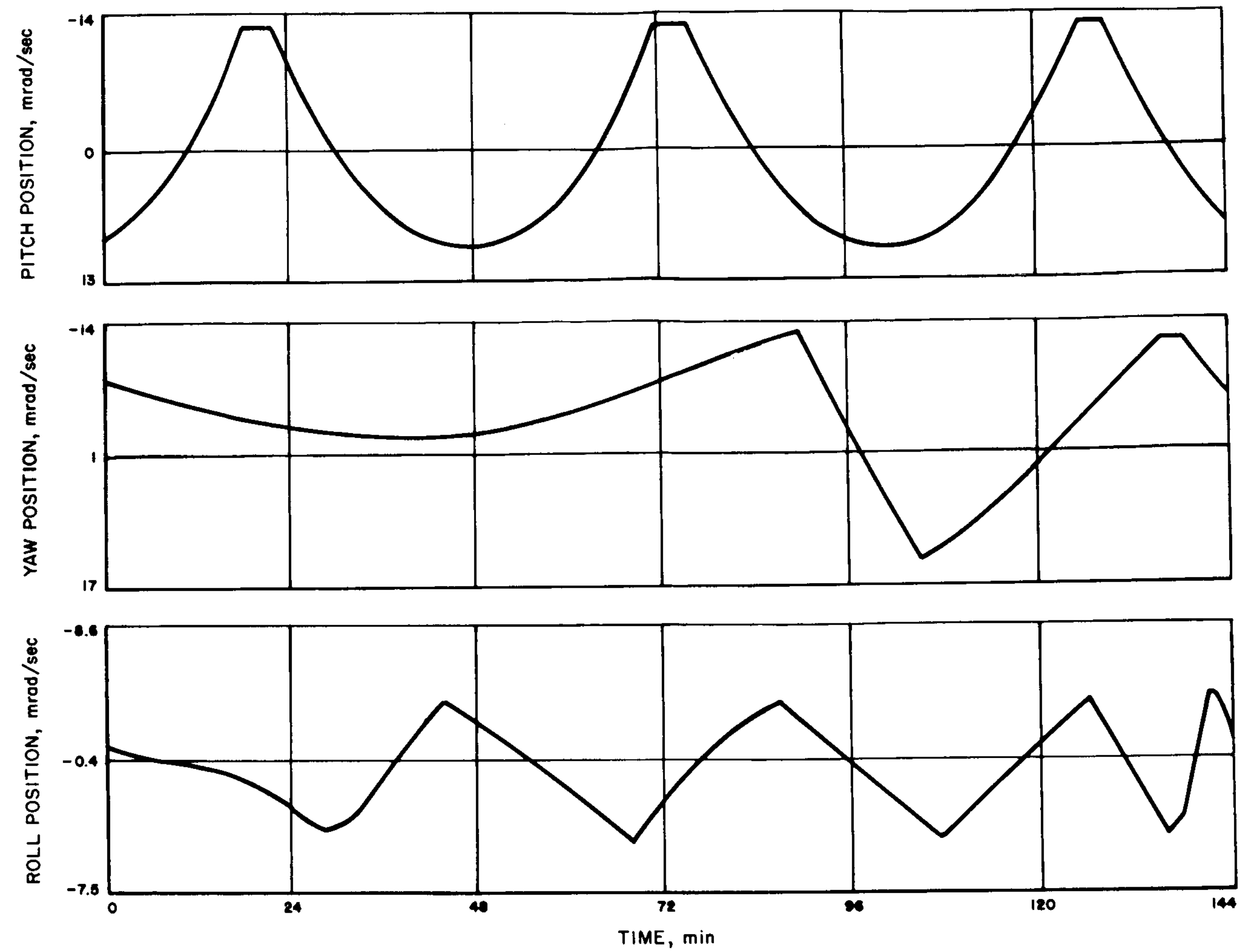


Fig. 34. Representative limit cycles during cruise

d. First trajectory-correction maneuver attempt (aborted). The start of the first trajectory-correction attempt (initiation of DC-27) took place at 14:35:00 GMT on December 4. This command turns on the gyros for warmup 1 hr before the start of the commanded turns. At 14:35:47 GMT the gyros came on and started to spin up. The spin-up of the gyros caused a disturbance in the attitude control system and the pitch, yaw, and roll error signals were driven over to one side of the deadband. The roll channel, which was of particular interest, started limit cycling against the negative side of the deadband with an angular excursion of 1 DN which was a resolution of 0.41 mrad. After 49.3 sec of operation (11 frames of data) a large transient was observed in the roll-error channel equivalent to −16.0 mrad. The next data frame (4.2 sec later) showed the spacecraft going into automatic roll search, which continued until Earth light reflections caused an acquisition to occur at 14:52:32 GMT at a clock angle of about 90 deg.

The fact that the roll-error signal was forced to one side of the deadband by the gyro spin-up was a completely expected, normal event. The huge transient in the roll channel and subsequent loss of acquisition necessitated the termination of the maneuver and caused it to be postponed for one day. The one sample of roll error of −16.0 mrad was impossible to explain by actual spacecraft motion, because the impulse required is catastrophic, and would have been observed in the pitch and yaw channels and on the roll gyro, which it was not. It was first thought that the signal was an electrical transient of some kind.

In view of the experience and analysis in connection with the loss of star acquisition in general, which generated the minute-particle theory as the cause of losses of acquisition, it now seems that the loss of Canopus at the initiation of midcourse falls into the same category. It is postulated that the time correlation of gyro warmup to subsequent loss of acquisition is due to the gyro starting transient shaking loose particles from the solar panels that were then driven in front of the Canopus sensor by solar pressure or spacecraft static charge. An instantaneous bright flash would cause a high gate violation and a false error signal. Since the resolution of events in this mode was 4.2 sec, it was possible to miss obtaining a sample of the flash in the brightness channel.

Ten DC-21s were transmitted before the spacecraft reacquired Canopus. The high degree of correlation between the *a priori* star maps and the telemetered data can best be seen for this reacquisition. See Fig. 32.

The performance of the spacecraft at initiation of gyro warmup prior to the start of the second and successful midcourse sequence was identical to the first except for the loss of acquisition.

e. Second trajectory-correction maneuver attempt (successful). The start of the second trajectory-correction maneuver attempt took place at the initiation of DC-27 at 12:25:13 GMT on December 5. The gyros went on and the gyro spin-up caused the roll limit cycle to stay on the negative side of the deadband, as in the first maneuver try. It should be pointed out that this is normal. After the gyros were up to speed, the roll limit cycle was normal.

The maneuver consisted of a counterclockwise (ccw) pitch turn of 39.16 deg and a cw roll turn of 156.07 deg. The pitch turn starting transient was seen at 15:25:12 GMT and the stopping transient at 15:28:54 GMT. The start roll transient was at 15:47:10 GMT and stopping transient at 16:01:19 GMT. The turns were correct in every respect. The amount of overshoot in the rate, plus position measurement scan during the starting and stopping transients, was within the tolerances expected for the attitude control subsystem acceleration of 0.45 mrad/sec. This was the only good confirmation available of the steady-state operation of the gas jets.

The start of motor burn was at 16:09:10 GMT and stop at 16:09:30 GMT.

Sun reacquisition was commanded by the CC&S at 16:15:10 GMT. Since the pitch turn was of such a small magnitude, the Sun acquisition was short, taking place at 16:21:07 GMT. Again the Sun gate field of view was seen to be 2.2 deg.

The Sun gate event started roll search for Canopus and the first star acquired was γ Velorum at 16:44:36 GMT at a clock angle of 342.8 deg. A roll override command was initiated at 16:52:50, Canopus was acquired at 16:55:00 GMT, and regular roll control was established.

f. Autopilot performance. The successful trajectory-correction maneuver was initiated at 14:25:13 GMT on December 5 with gyro warmup. At 15:25:12 GMT the attitude control subsystem was switched to inertial mode and the commanded turns were started. The ccw pitch turn was 39.16 deg and the cw roll turn was 156.07 deg. These turn angles include a vernier adjustment in pitch and roll to compensate for a known center-of-gravity (cg) misalignment error. The action of the autopilot under the known cg misalignment rotates the thrust vector to the proper direction in inertial space.

A vernier adjustment was also made to the motor burn time to compensate for jet vane drag. The compensation was based on the predicted initial conditions to the autopilot from the attitude control subsystem. On the basis of the commanded turn angles, it was expected that solar torques would keep both pitch and yaw angular positions held against the negative side of the attitude control deadband. This initial angular offset caused the autopilot to drive the jet vanes hard over, producing 6-lbf drag. The integral of the drag force over the transient period following motor ignition provided the magnitude of the motor burn time correction. The actual position of the spacecraft in the pitch and yaw attitude control deadband was exactly as predicted.

The midcourse motor was ignited at 16:09:09.6 GMT and burned for 20.4 sec. Four data points on the gyro signals were received during the motor burn period. Although this did not provide enough data for a thorough analysis, three events of significance appeared to fit the observed points. These were: 1) the first data point after motor ignition indicated a transient overshoot in pitch and yaw[12]; 2) for the remaining three points during the burn period, pitch and yaw maintained an equal and opposite offset, indicating a cg misalignment angle of approximately 2.2 mrad or 0.064 in.; and 3) a very slow motion in the roll channel may have been a low-amplitude limit cycle due to friction in the jet vane actuators. This limit cycle was theoretically predictable and was observed in analog computer simulations, but was not observed in previous spacecraft

[12]This was expected from the known, initial conditions on the autopilot.

because of insufficient data during the midcourse maneuvers.

The entire maneuver and the performance of all subsystems was very satisfactory. Subsequent tracking data verified that the *Mariner IV* midcourse was very accurate.

g. Cruise performance

Pitch and yaw control. The pitch and yaw control systems have performed well throughout the entire flight. One outstanding anomaly is the fact that the minimum rate increment was twice what it should be, but this is a system problem that is also observed in the roll channel, and is discussed at length in the gas system section. The position deadband was ± 8.6 mrad and the limit cycle rate increment is ± 18.4 μrad/sec. The design value rate increment was ± 9.0 μrad/sec.

Roll control. The roll control system, like pitch and yaw, exhibits an excessive rate increment (50% too large). The deadband is ± 4.5 mrad and rate increment is ± 14.0 μrad/sec; the design value being ± 9.0 μrad/sec.

A significant problem in roll control observed throughout the mission is the roll transient problem. The evidence gathered indicates that roll transients are caused by bright flashes, external to the spacecraft, detected by the star sensor. Extensive PTM tests tended to deny the possibility that the cause could be an internal electrical problem. In addition, analysis seemed to rule out the possibility that energetic particles or radiation were interacting with the star sensor. The only reasonable theory left was the external flash hypothesis. In the early part of the flight, when the telemetry operated at the high data transmission rate (12.6 sec between samples) there were several instances of excessive brightness samples coincident with roll transients. If the transients were caused by the bright flashes, the magnitude of the transients indicated that their duration must be on the order of less than a half second. This would explain why a flash is seldom detected at the time of a transient.

The evidence led to the use of DC-15 to deactivate the brightness gate logic to prevent loss of acquisition. The validity of that decision is proved by the fact that from that time there was never a loss of acquisition in spite of about 40 observed roll transients. Subsequent to MT-6 the low data rate of 50.4 sec between samples precluded the observance of most brightness transients. On June 9, however, a 25-times-Canopus brightness telemetry sample was obtained coincident with a roll transient.

The source of the bright flashes is still not known positively. Some indication of the probable source is found in the fact that of the four times that the spacecraft was mechanically disturbed from the cruise mode, two were accompanied by vigorous roll transients. The first was at gyro turn-on during the initial midcourse attempt and the second was at the science cover deployment. The mechanical disturbances themselves could not explain the transients, but particles shaken from the spacecraft such that they floated by the star sensor could cause flashes. Before the flight, the cognizant engineer reported that the extreme sensitivity of the star sensor made it possible to detect particles 0.005 in. in diameter at 2500 ft from the sensor. It was not suspected, however, that the spacecraft would present a serious problem as a source of particles.

An explanation was sought for the source of the continuing roll transients taking place during the long cruise period. The hypothesis was advanced, with some analytical justification, that undetectable micrometeorites striking the solar panels could detach surface particles, which float by the sensor. Occasionally, very small rate impulses in the pitch or yaw channels were observed to be correlated with roll transients. It was suggested that a micrometeorite struck a solar panel, imparting a pitch or yaw impulse, and dislodging a particle which caused a roll transient. However, these pitch and yaw impulses can as easily be explained as crosscoupling effects from the roll axis, due to gas jet and moment arm misalignment. The magnitude of the impulses is about 15 to 20% of the roll impulse, about the same observed in normal limit cycle operation. It is interesting that the polarity of the pitch and yaw impulses are always the same, independent of the roll polarity. This would be the case if the positive and negative roll jets have the same misalignment components about the pitch (or yaw) axes.

It must be noted that there has never been a correlation of a roll transient with a cosmic dust detector micrometeorite event. However, this, along with the foregoing discussion, does not rule out micrometeorites as a cause for spacecraft particle generation. This hypothesis postulates micrometeorites below the detectable level. It does mean, though, that the micrometeorites hypothesis is not verifiable, even though it stands as the most reasonable explanation for continuing roll transients.

Table 18 lists all of the roll transients reported throughout the mission.

Control gas actuator subsystem. The gas-usage rate throughout the mission was between 3×10^{-3} and

Table 18. Roll transients observed

Date	Time, GMT	Event
November 29, 1964	13:12:57	Roll acquisition lost, gyros on. Low-gate violation.
November 30	13:41:00	A transient was noted in the roll position telemetry. Not severe enough to cause the loss of Canopus acquisition.
December 2	10:09:00	Canopus acquisition lost, gyros on.
4	14:36:31	Approximately 52 sec after the initiation of the midcourse sequence, roll acquisition was lost and the spacecraft went into roll search.
	23:29	Roll transient.
5	16:56	Roll transient.
7	12:29:41	Canopus acquisition lost, gyros on. Spacecraft in roll search.
8	10:43	Roll transient.
9	05:35:36	Gyros on, spacecraft in roll search during one sample of the roll position channel.
13	20:48:27	Acquisition of the star γ Velorum lost, gyros on. Reacquisition was immediate.
14	22:51	Roll transient.
17	05:36:15	Brightness transient noted on the Canopus intensity channel. No loss of acquisition occurred.
	07:16:43	Acquisition of the star γ Velorum lost, gyros on. Reacquisition of γ Velorum was normal.
26	20:04:04	Roll transient.
31	00:53:46	Roll transient.
January 4, 1965	18:49:14	Roll transient.
6	11:16:27	Roll transient.
8	05:01:15	Roll transient.
13	14:13:51	Large roll transient.
20	23:46:00	Roll transient.
February 4	13:55	Roll transient.
5	04:16	Roll transient.
7	05:40:00	Large roll transient. Disturbances were also seen in pitch and yaw.
February 11, 1965	06:58	A large roll transient was observed coincident with science cover deployment and scan search initiation.
12	02:53:22	Roll transient.
16	13:47:24	Roll transient.
27	13:56	Roll transient.
March 11	04:47	Roll transient.
14	02:17	Roll transient.
18	01:03:50	Roll transient.
April 7	11:10	Small roll transient.
11	06:18	Small roll transient.
15	09:55	Extremely small roll transient.
24	23:47	Roll transient.
June 9	11:41:05	A vigorous roll transient was observed coincident with a large change in Canopus brightness and a visible impulse on the spacecraft yaw axis.
10	19:54	Small roll transient.
11	15:40	A roll transient was observed coincident with slight impulses in the pitch and yaw axes.
12	08:49	Small roll transient.
14	15:53	Roll transient.
16	11:38	Small roll transient.
19	04:50	Small roll transient.
27	07:14	Roll transient.
	18:47	Roll transient.
July 3	09:56	Small roll transient.
5	09:27	Small roll transient.
6	03:10	Small roll transient.
12	05:40:56	Small roll transient.
	06:23:46	Small roll transient.
August 4	03:32	Small roll transient.
	03:42	Small roll transient.
	16:17	Small roll transient.

4×10^{-3} lb per day, indicating a probable total gas system lifetime in excess of 4 yr from launch. The gas usage rate is illustrated in Fig. 35.

During the flight it was noted that the minimum rate increments that establish limit cycle performance were about 200% in pitch and yaw, and 150% in roll, of the design values. The attitude control gas system operates in two modes: (1) steady state, where the valves are activating longer than one second, and (2) pulse mode, where the valves are activated for 20 msec. Analysis of flight data indicates that in the steady state mode the design acceleration was achieved from the constant gas flow.

The pulse mode establishes the limit cycle rates by generating a minimum impulse bit proportional to the amount of gas expelled from the valve during one 20-msec pulse. The minimum rate increment was designed to be that which is generated by a minimum impulse bit obtained from 20 msec of steady state flow. Actually the dynamic flow characteristic of the valve-plenum chamber-nozzle system has been found to create a gas flow about 30 times the steady state flow during the first 3 msec of valve actuation. Analysis indicates that this fully explains the observed rate increments for all three axes.

h. Encounter operation. During the encounter period, the only discernible attitude control activity was the scan platform motion effect on the attitude control limit cycle activity. This was expected and was used to verify proper scan operation.

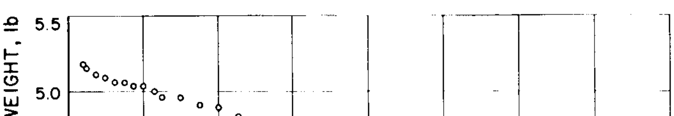


Fig. 35. Attitude control gas usage

No planetary effects upon the attitude control subsystem operation were observed. No gravity gradient torques, atmospheric torques, stray light reflections, or radiation effects could be detected in the telemetry data.

3. Recommendations

a. Roll transient problem. The in-flight solution to the roll transient problem was to disable the Canopus sensor brightness gates. This was not without disadvantage, because the automatic reacquisition logic for the attitude control subsystem is also disabled. Since the high-brightness gate is required for initial acquisition only[13], there is no reason that logic circuitry could not be included to disable it automatically at gyro turn-off. Whenever the gyros were off for cruise operation, the high gate would be removed, therefore a bright flash would not start a roll search, yet all of the other automatic features of the subsystem would still be enabled.

b. Solar pressure vanes. From the beginning of the flight, it has been noted that the damping mode, as designed for the *Mariner* spacecraft, is not effective for attitude control. It appears that a different design concept such as active control would be required before solar pressure damping could provide a practical means of controlling spacecraft attitude.

The adaptive mode, however, has proved to be quite adequate for its purpose. The basic problem in the adaptive mode operation revolved around the extremely slow response of the vanes, especially compared with the rate of change of torques acting on the spacecraft. If the *Mariner* damping mode is discarded, the step size for the adaptive mode could be increased considerably[14] and the offset torques would be rebalanced quickly, providing more efficient operation.

c. Gas valve transient flow. No reliable method had been developed at JPL before launch to measure the minimum impulses required from the gas jets, hence, the transient effect went unobserved until the flight. Because this is a primary spacecraft system, some considerable effort should be made to develop this capability as a regular flight test operation.

One simple way to solve the transient effect in the valve design would be to restrict the poppet valve exhaust orifice such that it approaches the gas jet

[13]The only object brighter than Canopus in the sensor field of view, during the flight, is the Earth at initial acquisition.

[14]To perhaps 1 deg.

nozzle in area. This would diminish the transient flow to the steady state level. Decreasing the minimum-on-time is not practical, since the transient flow takes place in the first few msec.

F. Pyrotechnics

1. Description

The *Mariner* pyrotechnics control assembly (PCA) supplies power and performs switching for seven functions:

1. Pin retraction of eight solar panel latches
2. Pin retraction on one science platform latch
3. First start, PIPS motor
4. First stop, PIPS motor
5. Second start, PIPS motor
6. Second stop, PIPS motor
7. Solenoid release, science cover

The PCA receives 2.4-kc power from the bus, rectifies this and charges capacitor banks. The energy in these banks is switched upon command by solid-state switches (silicon-controlled rectifiers or SCRs) to actuate pyrotechnic devices and the science cover solenoid release.

Telemetry measurements for in-flight monitoring of the subsystem performance was limited to non-quantitative event register indications. An indication of nominal current being delivered to each squib bank during pyrotechnic firing and an indication of nominal voltage appearing at the output to the solenoid during actuation constituted the PCA flight engineering information.

2. Performance.

The subsystem performance appeared normal. PCA events were received as expected.

Auxiliary events indicating squib firings were received upon backup command for solar panel deployment. This occurred during both *Mariner III* and *Mariner IV* launches and was accepted as a normal condition resulting from solar panel pin puller squib shorts. A nominal-voltage-to-solenoid event was received as a part of the planetary encounter sequence. This indicated that one channel of the redundant PCA was still nominally charged and functioning at encounter.

Critical parameters such as capacitor bank voltage and leakage currents of the SCRs and capacitor banks were not available.

3. Recommendations

No specific recommendations are presented for this subsystem as a result of the flight information. Various recommendations relating to the results of ground testing and considered design improvements have been documented elsewhere.

G. Postinjection Propulsion Subsystem (PIPS)

1. Description

The system is a monopropellant-hydrazine, regulated-gas-pressure-fed, constant-thrust rocket.

The principal system components are:

1. Nitrogen tank (3000 psia high-pressure gas reservoir)
2. Pressure regulator for reducing fuel tank inlet pressure to a constant 310 psia
3. Propellant tank and propellant bladder which contains the hydrazine
4. Rocket engine
5. Explosive valves

Thrust initiation and termination are controlled by explosive valves which fire simultaneously to initiate nitrogen, propellant and oxidizer flow; and terminate nitrogen and propellant flow. Ganged valves in parallel are used for the two-start requirement. A 15-cc slug of N_2O_4 is injected for each of the two starts resulting in roughly, 1 sec of hypergolic bipropellant operation which heats the catalyst bed so that monopropellant decomposition can be sustained.

System design and operational philosophy is aimed at:

1. Maximizing reliability
2. Maximizing start and shutoff reproducibility
3. Minimizing preflight handling and spacecraft interactions
4. Minimizing inflight electrical signals and sequencing
5. Minimizing number of system components

2. Performance

a. Trajectory-correction maneuver. Inflight telemetry coverage of the *Mariner IV* PIPS has been excellent to date. The reduced data for the nitrogen tank pressure, propellant tank pressure, and oxidizer start cartridge pressure are plotted in Fig. 36–38 respectively.

As depicted in Fig. 36, nitrogen tank pressure remained constant up to the time of the postinjection maneuver indicating a leak-tight system through boost and during the 7-day pre-correction maneuver coast period. This nominal nitrogen tank pressure would have supported a maximum correction maneuver as limited by the amount of fuel available, and resulted in a maximum predicted velocity increment *capability* of 86.97 m/sec.

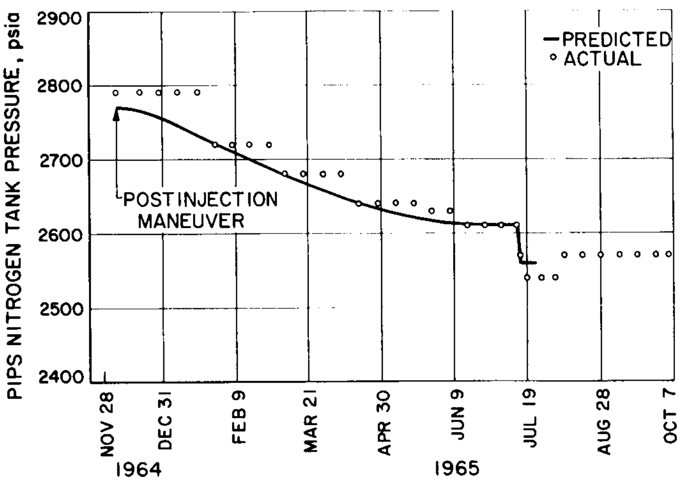


Fig. 36. Nitrogen tank pressure

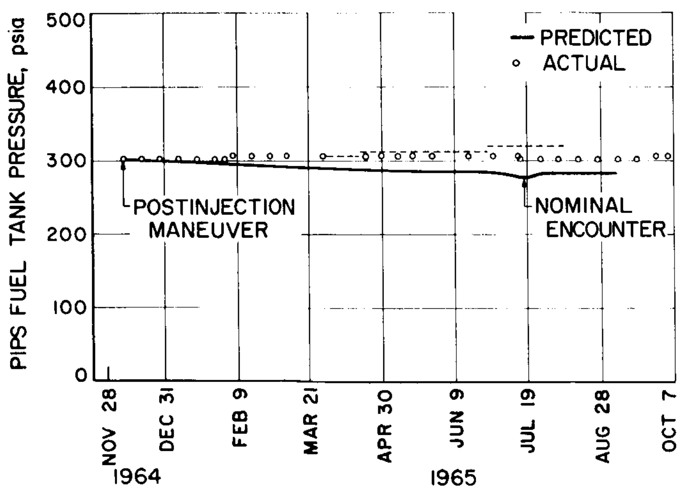


Fig. 37. Propellant tank pressure

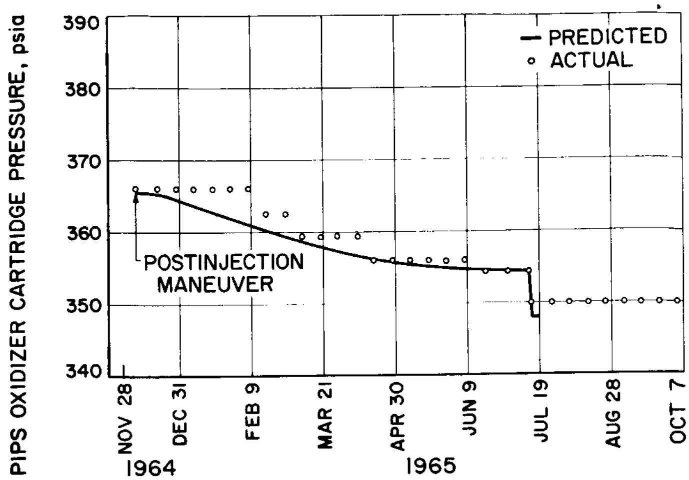


Fig. 38. Oxidizer pressure

The propellant tank pressure, Fig. 37, and oxidizer pressure, Fig. 38, maintained a constant pressure prior to launch and up to the time of the correction maneuver.

Based on the pressure regulator setting for the *Mariner IV* PIPS, and nominal engine performance at the expected jet vane deflection, an engine vacuum thrust of 49.40 lbf was predicted. This thrust level along with the velocity increment requirement of 16.70 m/sec and a spacecraft weight of 574.74 lbm resulted in a predicted motor burn time of 20.06 sec. From the doppler shift data acquired during the correction maneuver and from the thrust chamber pressure transducer, motor ignition and thrust termination were verified, the burning time being as predicted. Further verification of normal PIPS operation during the trajectory-correction maneuver is provided by the post-maneuver propellant tank and nitrogen tank pressures and temperatures. Using these data and premaneuver nitrogen and propellant tank pressures and temperatures, the delivered velocity increment was calculated to be within 5% of the commanded maneuver. It should be noted that this discrepancy between the aforementioned calculated velocity increment and the command increment (16.70 m/sec) is well within the accuracy of the computation and does not necessarily indicate an error in the execution of the maneuver.

It is significant to note that because of the inclusion of a timer shutoff mechanism in the *Mariner* spacecraft a stringent total impulse predictability and reproducibility requirement was placed on the PIPS. From the data received, the propulsion subsystem performed very well and within its design limits. In addition, the propulsion subsystem remained leak tight throughout the entire mission, presenting no anomalous torques to the spacecraft, and provided a reliable second trajectory correction capability.

b. Diagnosis of abnormal conditions.

Trajectory-correction period. The rocket-motor chamber-pressure transducer indicated a higher than expected value and the propellant-tank pressure transducer indicated a lower than expected value. However, by comparing the propellant tank pressure transducer calibration curve against the TA test history data, nonlinearity in the region of operating pressure was apparent. In addition, the motor chamber pressure transducer was not pressure calibrated prior to flight.

Cruise period. On February 3, 1965 a pressure rise of approximately 4 psi was noted in the fuel tank when all other propulsion oriented pressures and temperatures were dropping. This event was not totally unexpected. A small amount of bladder/hydrazine incompatibility was expected from past test history; however, in attempting to explain the pressure rise the following three cases were considered:

Case I — Pressure leaking through the nitrogen explosive valve (shutoff)

Case II — Pressure that is locked up between the nitrogen explosive valve and the regulator

Case III — Pressure rise due to hydrazine/bladder incompatibility

Cases I and II were examined by calculating the leak rates that would result in the indicated pressure rise. These data were not conclusive because the very low magnitude of the pressures involved is on the threshold of the telemetry resolution.

In support of Case III several bladder tests were initiated and the preliminary data indicated that bladder/hydrazine incompatibility fits into the pattern of the actual data very well. Since the life-test system bladder test results were similar to the actual flight conditions, an attempt was made to duplicate the degraded condition of the life-test bladder. Under controlled conditions, a sample of bladder material was subjected to the temperature/pressure profile of the life test. At ambient temperature in excess of +100°F, decomposition of hydrazine, caused by the sample bladder, produced an appreciable rate of pressure increase in the sample vessel. Since reaction rates such as those obtained with this bladder/hydrazine combination are known to be extremely temperature dependent, the effect on the system due to pressure rise is proportional to the increase in the absolute value of the temperature. For temperatures below +100°F the rate of decomposition falls off sharply. However, it should be noted that even at room temperature, +70°F, a finite decomposition occurs.

3. Recommendations

a. PIPS. It would be most desirable to develop a completely compatible bladder for a system of this type in future applications. Although the present bladder design proved capable and in fact accomplished the mission objectives, a totally compatible bladder would be a necessity in more sophisticated propulsion systems.

A significant weight and reliability gain would be realized by using a spontaneous catalyst ignition technique instead of the bipropellant ignition system presently used. Use of a spontaneous catalyst would also provide for a multistart concept enhancing the present two-start capability.

b. SPAC evaluation

1. The actual flight SPAC operations were, in general, operated in a satisfactory manner; however, the data simulation for test purposes could be improved so that it would be more meaningful to propulsion personnel.
2. Since the necessity for a certain amount of documentation is unavoidable, the repetition of inputs should be kept to a minimum. Daily reports for instance, during inflight cruise phase could be reduced to a weekly report with daily reports made on a verbal basis except during emergencies.
3. A direct line of verbal communication should be established between the SPAC representatives and the technicians that distribute the incoming data. This would help resolve problems of data distribution in a faster manner.

H. Temperature Control

1. Description

a. Temperature control subsystem. The temperature control subsystem is comprised of all devices and design features employed for the purpose of maintaining temperatures within specified bounds for subassemblies, assemblies, and components of the spacecraft. Included hardware are louvers, radiation shields, surface coatings, and thermal insulation. Design details influenced by temperature control requirements include component placement, methods of attachment, structural material and configuration where conduction is important, and surface finish of structure components.

The *Mariner* Mars 1964 bus temperature-control design provides isolation from solar heating, minimum internal resistance to radiative and conductive heat transfer, and active control for all bays except II and IV. Protection from the variable Sun input is provided by a multiple layer aluminized mylar upper thermal shield. Relatively uniform internal temperatures are obtained by using good thermally conducting joints and structural materials and by treating interior surfaces to produce high emittances. Bimetallic-actuated variable-emittance louvers reduce the temperature variations caused by changes in solar heating and internal power dissipation.

External science and appendages are separately and passively controlled by surface property selection, regulation of conduction paths, and selective Sun shielding. Conductive and/or radiative thermal coupling with the actively controlled bus is provided where possible to minimize both transient and long-term temperature variations.

b. Absorptivity standard. The absorptivity standard is an engineering experiment flown to enable the spacecraft temperature control designers to:

1. Measure the solar absorptance α_s of four typical spacecraft surfaces in real sunlight.
2. Measure the change or degradation of the surface properties with time in space, and
3. Determine the effects of basing flight predictions upon the temperatures measured during space simulator testing.

The instrument works upon the principle of measuring the temperature of an insulated flat plate normal to the solar irradiation. The temperature measurement system is designed to circumvent the possible 3°F inaccuracy of the typical spacecraft thermister data. The time at which a known temperature is reached is signalled rather than telemetering a continuously varying temperature history. The sensor system uses the mercury thread in special thermometers to short out a series of resistances, providing large step change in the telemetry signal.

The four sample surfaces are: 1) ARF-2, a zinc oxide-potassium silicate white paint; 2) Cat-a-lac black paint; 3) aluminum silicone paint; and 4) polished aluminum. The ARF-2 sample contains a black stripe to prevent freezing of the thermometer mercury and the polished aluminum sample contains an ARF-2 stripe to cool the sample to a level compatible with the structural plastic.

2. Performance

Monitored temperatures remained within allowable limits throughout flight. Flight temperatures were generally lower than pre-launch predictions because simulator tests were conducted at higher solar intensities than those seen in flight[15]. The transient behavior of the spacecraft in flight was similar to that exhibited in prelaunch testing.

The flight results indicated that the thermal design succeeded and that all thermal control hardware functioned as designed. Discrepancies between predictions and flight data were attributable to an imperfect understanding of test data and spacecraft characteristics, not to hardware degradation or malfunctions. Specifically, louvers, thermal shields, and surface coatings used on *Mariner IV* reliably fulfilled mission requirements. The relatively small Earth-to-Mars temperature drop within the bus provided confirmation of proper louver behavior. The absence of unexplainable temperature anomalies verified the proper operation of the temperature control subsystem as a whole. Louver position indicators showed only fair correlation with corresponding bay temperatures, however.

a. Cruise temperatures. As shown in Table 19, initial cruise temperatures were considerably lower than predicted on the basis of space simulator testing. Those items most sensitive to the solar input had the largest discrepancies. The ion chamber and associated electronics were 30°F cooler than predicted; the magnetometer sensor was 20°F cooler, and the solar panels were 15°F low.

Postlaunch checks in the JPL 25-ft space simulator indicated that the spacecraft was probably tested at a solar intensity 11% high at Earth cruise and 17% high at Mars cruise. The absorptivity standard black and gray samples agree with these findings, as do the solar panel temperatures. The problem appears to have been caused by improper calibration or interpretation of the reference thermopile, but no significant source of thermopile error has yet been discovered.

[15]Temperature histories are shown in detail in Appendix B.

Table 19. Flight temperature evaluation

Channel	Temperature measurement	November 30, 1964 Earth cruise, °F	Predicted Earth cruise, °F	July 13, 1965 Mars cruise, °F	Predicted Mars cruise, °F	December 5, 1964 Midcourse maximum, °F	August 2, 1965 Playback[a] minimum, °F	Operating temperature limits, °F
401	Bay I	76	89	60	62	88	54	14 to 167
421	Bay II	71	85	41	46	94	33	35 to 125
402	Bay III	70	74	51	55	73	34	14 to 122
423	Bay IV	72	80	54	57	75	47	14 to 167
404	Bay V	66	69	60	60	73	59	14 to 149
424	Crystal oscillator	72	77	69	69	74	68	14 to 167
405	Bay VI	71	75	70	73	73	69	14 to 167
426	Bay VII	63	68	58	57	70	57	30 to 131
407	Power regulator	95	106	83	87	114	77	14 to 167
408	PIPS nitrogen tank	70	83	47	51	136	38	35 to 125
428	Battery	75	85	60	64	81	57	40 to 140
409	Solar panel 4A1	136	150	9	20	135	8	10 to 175
429	Solar panel 4A5	136	150	11	20	134	9	10 to 175
410	Canopus sensor	60	62	52	51	68	49	0 to 100
430	Lower ring	63	69	55	56	72	53	−300 to 300
411	Scan actuator	71	82	50	53	77	44	−30 to 200
431	Upper ring	87	104	46	52	109	37	−300 to 300
434	Upper thermal shield	223	268	82	110	221	80	−200 to 300
435	Lower thermal shield	−126	−95	−131	−105	−126	−131	−200 to 300
436	Tape recorder	68	71	58	60	70	58	14 to 149
437	SPITS	37	46	14	26	39	9	−20 to 122
418	Television	38	46	13	26	39	9	−4 to 104
438	Trapped-radiation detector	81	97	49	55	84	39	14 to 122
419	Ion chamber	70	100	−19	10	95	−28[b]	−22 to 158
439	Magnetometer	34	54	−35	−19	102	−68	−40 to 131
217	PIPS fuel tank	72	80	49	52	84	39	35 to 125
218	Attitude control +X/−Y nitrogen tank	70	78	54	54	77	50	40 to 140
219	Attitude control −X/+Y nitrogen tank	71	76	51	53	76	45	40 to 140

[a]Cruise science off. [b]Estimated.

The above discrepancy does not provide the whole answer to incorrect flight temperature prediction, however, since different components indicated different amounts of intensity mismatch. For example, the solar panels, bus, and ion chamber results require 10, 20, and 30% reductions, respectively, to explain the flight results. After correcting for the solar intensity, however, flight temperatures are within the expected prediction accuracy. The remaining errors are associated with the failure to correct for inherent space simulator limitations, such as decollimation of the solar beam and extraneous inputs of thermal radiation. The *Mariner* flight data have resulted in an improved understanding of these error sources.

b. Deviations from post-launch predictions. Correlation of temperature-time predictions, based on initial flight data and simulator test results, is generally good, subject to the following considerations.

A flight anomaly, apparent on inspection of predicted vs actual flight data, was that induced by incorrect solar panel temperatures during simulator tests. The dummy panels were forced to 150° and 0°F at Earth and Mars, respectively. Corresponding flight temperatures are 135° and 10°F. On the basis of these tests, the Earth-to-Mars temperature drop was overestimated for bays which have radiative inputs from the panels; these are primarily odd-numbered bays. The resultant error is small and conservative, but the requirement for careful test interpretation is clear.

The magnetometer temperature gradually departed from the predictions, as shown in Fig. 39. By encounter,

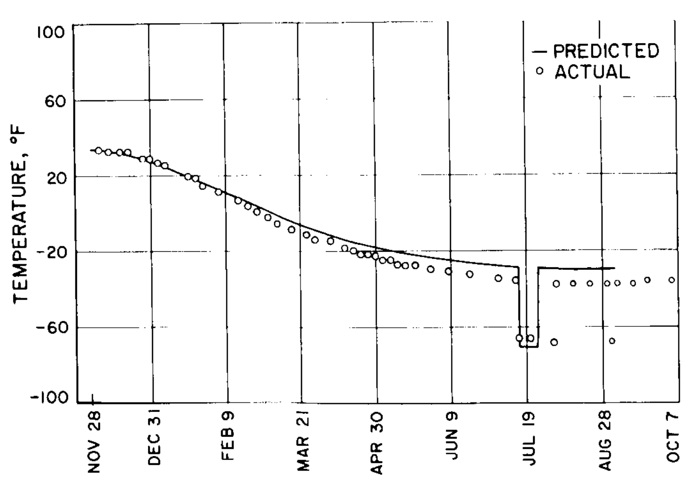


Fig. 39. Magnetometer temperature

the magnetometer was 6°F below the anticipated temperature. This instrument had significant heat inputs from both the Sun and internal power dissipation, and a deficiency in either input could cause such a drop. The playback mode turn-off of the magnetometer produced a smaller temperature drop than experienced during simulator tests, however; and this result gives a firm indication of below normal power dissipation in the unit.

The lower thermal shield temperature was somewhat lower than that observed during simulator tests, and the temperature drop during flight was slightly less than expected. These data indicate that a small heat input, about 2% of a solar constant, to the bottom of the bus existed during the simulator tests.

c. Launch and maneuver transients. Launch temperatures were sufficiently low to prevent overheating during ascent, parking orbit, and Sun acquisition. The launch azimuth resulted in a moderate thermal environment. The spacecraft separation occurred in the shadow of the Earth, and the Sun acquisition time was relatively short. At Sun acquisition, the temperature of the Canopus tracker (100°F upper limit) was 70°F. A shroud-off verification was obtained during the parking orbit by comparing the temperatures of two solar panels.

No constraint was placed on the trajectory-correction maneuver for thermal reasons. The pitch turn selected did not cause extreme solar inputs. A mild heating transient was caused by the increased solar heating, increased power dissipation, and motor burn. All temperatures remained within acceptable limits. The magnetometer sensor heated more rapidly than expected; the discrepancy stemmed from an oversimplified transient thermal analysis. Many of the maneuver events were verified by observation of changes in spacecraft temperature distribution. A qualitative confirmation by temperature measurements was obtained for changes in power distribution, extent and direction of pitch and roll turns, and motor firing.

d. Thermal effects of internal power changes. Gyro turn-ons during the early days of flight caused some variation in bus temperatures, particularly in Bay VII. Long term gyro operation caused a 10°F rise in temperature in this bay.

Fifteen days after launch, the power amplifiers were switched in the radio transmitter in Bay VI. The resulting increase in power dissipation resulted in a 9°F increase in the temperature of Bay VI and a general increase of 1°F in the bus temperature. These changes were very nearly the same as experienced in pre-flight space simulator tests.

The cover drop and battery charger turn-off on the 75th day of the flight combined to lower bus and scan platform temperatures. Bay I dropped 5°F and cooled adjacent bays slightly. The scan platform temperature dropped 8°F due to the increase in unblocked radiation area and the decreased heat input from the bus. Since the Mars cruise temperature predictions were based implicitly on a charger-off condition, the bus temperatures after the turn-off were nearer the nominal values.

The slight rise in television temperature at the beginning of the scan sequence coupled with the temperature drop after television turn-off provided confirmation of normal television power dissipation and of a normal science cover drop. These data are in good agreement with corresponding space simulator results.

e. Encounter and playback results. Encounter warming transients, Fig. 40, were similar to those experienced during the cover drop exercise. The battery charger turn-off earlier in flight resulted in more pronounced temperature rises at encounter than occurred during corresponding simulator tests.[16]

Bus temperatures, particularly those in Bay III, showed a sizable temperature drop during the playback sequence due to the turn-off of cruise science. This drop averaged about 1°F more than experienced in pre-flight tests, probably because the louvers were more nearly closed (and therefore less effective) in flight. The magnetometer temperature-drop was 12°F less than expected, which indicates that the power dissipation in this sensor was less than had been assumed. A gradual decrease in power consumption during the flight seems likely. The ion chamber temperature transducer dropped

[16]The net power increase was greater in flight than in tests.

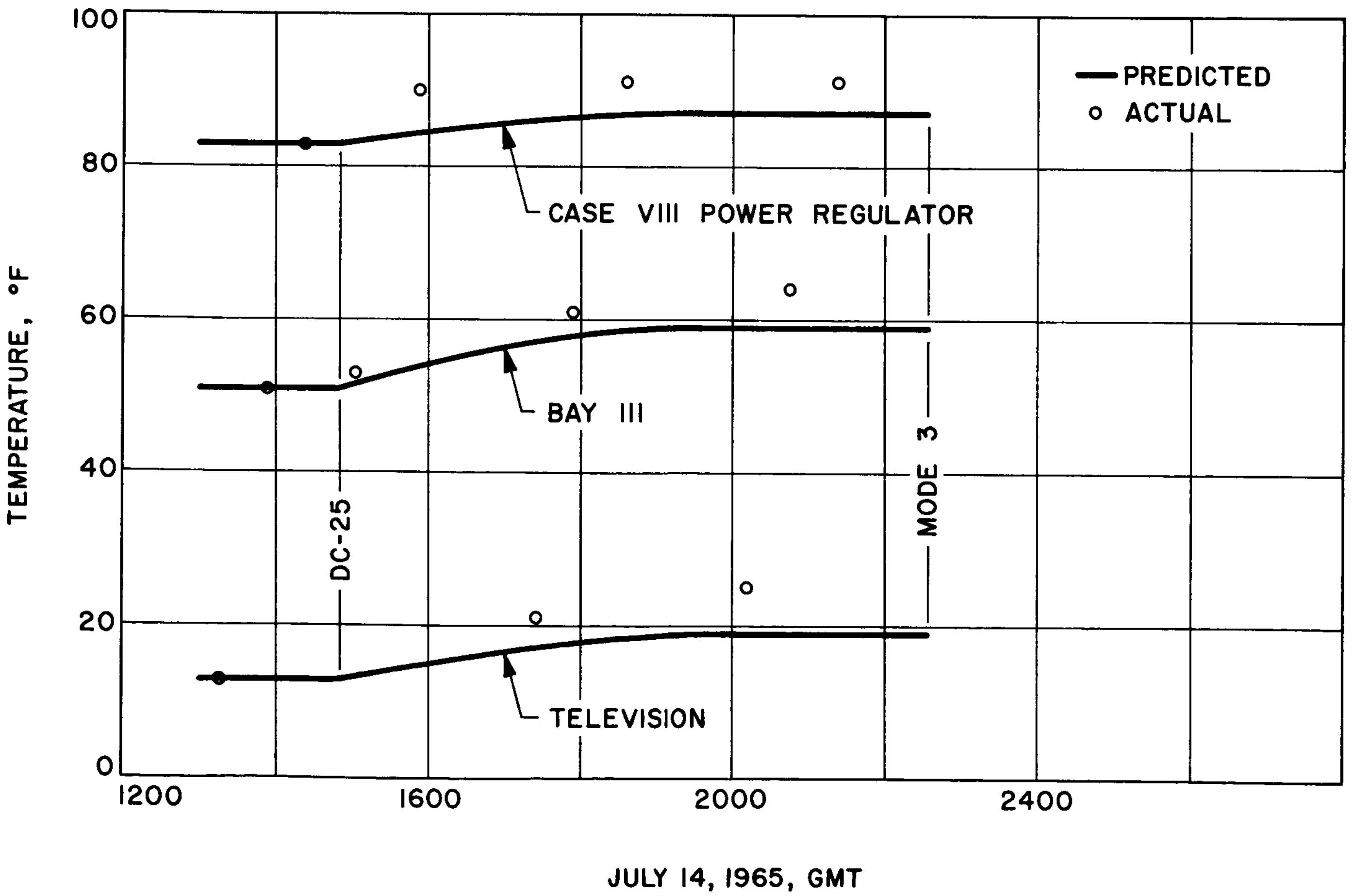


Fig. 40. Encounter temperature transients

off scale during playback which prevented an estimation of this unit's power consumption. Such an estimate would have been very helpful in failure mode analysis.

Bus temperatures returned to their pre-playback levels at the end of playback. This repeatability rules out louver hysteresis caused by bearing friction.

f. Absorptivity standard. The absorptivity standard has provided good information regarding the degradation of surfaces in space and the problems arising from the testing in the JPL space simulator. The determination of α_s in space has not been successful because the radiation and conduction losses are too large. The ratio of α_s at any given step to the α_s at the initial step is plotted in Fig. 41.

One of the first results noticed was that the ARF-2 paint was yellowing or degrading at a rate much greater than expected due to ultraviolet (UV) exposure. This has continued at a pace about ten times as rapid as measured during the development of this coating. Subsequent ultraviolet exposure tests on ARF-2 samples prepared at the same time as the flight surfaces show rapid degradation, raising the question of the validity of predicting a paint's performance without additional tests of samples prepared and applied by the user.

It was found that the wiring was reversed between the Cat-a-lac black and aluminum silicone sample sensors. The sensor idiosyncrasies made this deduction concrete by mid-January 1965. This embarrassing problem probably is due to the cable harness being fabricated to an obsolete drawing. Once this error was discovered, the data for the Cat-a-lac black paint was very consistent and predictable, showing negligible degradation. On February 13, 1965 another interesting event occurred when the black sample switched a month prematurely. This was attributed to a design oversight. The thermometers are insensitive to 100-g accelerations when the mercury meniscus is within the capillary; but, when the meniscus extends into the gas chamber, the

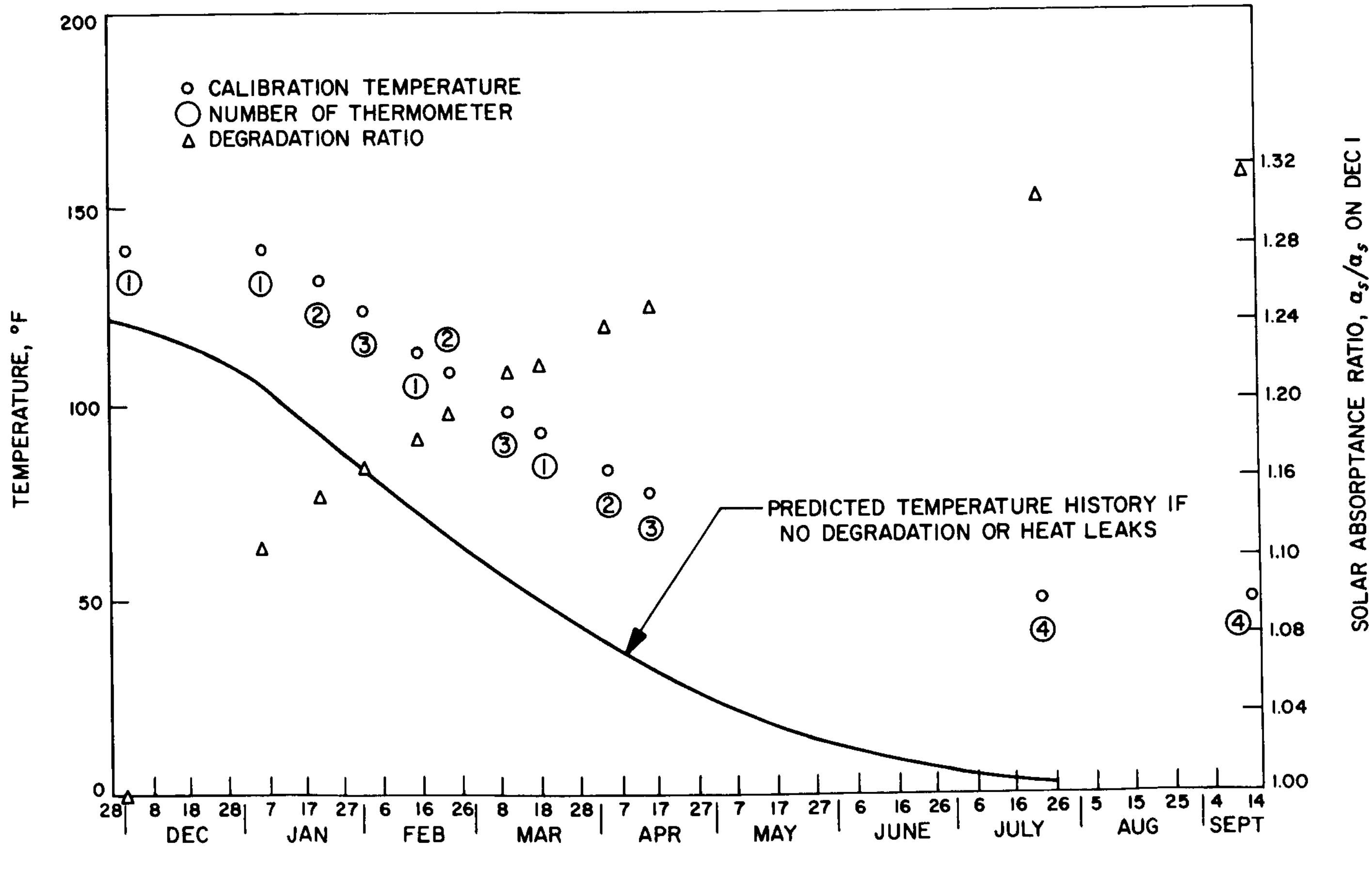


Fig. 41. Absorptivity standard performance

thermometer is vulnerable to shocks. During the midcourse maneuver, the sample temperature was such that the mercury of the No. 4 thermometer was probably advanced into the gas chamber and the accelerations of the squibs and midcourse motor shook some mercury loose, changing the calibration. Each sensor contains four mercury thermometers in this sensor; the first three thermometers overlap in the temperature range of 275 to 208°F, while the fourth thermometer spans 180 to 130°F.

The aluminum silicone paint had degraded more than expected but the degree is uncertain due to sensor calibration shifts.

The polished aluminum sample showed an early degradation greater than that due to the ARF-2 stripe alone. After the first two steps, the data paralleled the degradation due to the stripe alone. This would indicate that perhaps the metallic surface was vacuum cleaned early in the flight and then did not further degrade. Two P/FRs were written against the sample. The third (final) step of the No. 3 thermometer never occurred and the first step of the No. 4 thermometer occurred early. The missing step is attributed to a broken thermometer and the early step is due to the same design oversight mentioned for the black sample.

The absorptivity standard showed that the space simulator has solar simulation intensity problems as well as spectral mismatch problems. Comparison of the simulator test and flight data of the black and aluminum silicone samples reveal that the simulator intensity was 10% higher than the simulator instrumentation said. The α of the ARF-2 and polished aluminum for the JPL solar simulation are respectively 37 and 9% greater than for sunlight.

3. Recommendations

a. Temperature control subsystem. On the basis of *Mariner* IV flight results, the following recommendations are made:

1. Future flights should include temperature transducers specifically located to aid in thermal data analysis. On *Mariner IV*, for example, solar simulator decollimation effects could have been evaluated with a transducer on the low-gain antenna; transducers on the high-gain feed and plasma probe would have provided valuable information on the UV degradation of white paints.
2. A more conservative approach to the temperature control of isolated components is recommended. Small items such as the magnetometer are inherently sensitive to small changes in their thermal balance. Contingency heater power or lightweight active control devices should be considered for such components in future missions.
3. Comprehensive and painstaking checkout of space simulator facilities should be performed before committing to a flight design based on test. Particular emphasis should be placed on the evaluation of solar simulator performance and on the detection of extraneous heat inputs from chamber walls.
4. The use of temperature control surfaces subject to degradation in flight should be avoided. Sunlit white paints should be eliminated when possible by the use of Sun shields and/or configuration control.
5. Improved louver position indicators should be developed.
6. Parametric tests should be performed to evaluate the effects of changes from design thermal conditions. Parameters of interest are internal power dissipation, solar heat inputs, and radiation and conduction coupling.

b. Absorptivity standard. The information gained from the absorptivity standard has proved of much value even though all objectives were not met. The following recommendations are made:

1. An instrument, although perhaps not the present absorptivity standard, to verify α_s predictions and to measure the degradation of surface properties of new surface treatments would be of value on future flights.
2. If the present instrument concept is to be used in the future, the design should be improved to reduce the heat leaks.
3. Type approval level tests on modified hardware have shown that the unit is structurally overdesigned. A lighter, modified structure and a redesign of certain components is recommended.
4. The temperature sensor system had problems but was basically sound. The problems encountered in flight can be easily remedied if this design concept is used again.

5. The use of mercury step thermometers to calibrate resistance thermometers may be useful in other design applications.

I. Data Automation Subsystem

A digital computer is required to provide a unified electrical interface between the varied science instruments and the spacecraft. On *Mariner IV*, this subsystem was called the data automation subsystem (DAS). The DAS automatically controls and synchronizes the data gathering sequence of all science instruments and processes and formats all the diversified science data into a single, continuous bit stream of 1s and 0s for the telecommunications subsystem, with a separate bit stream for the video storage subsystem.

The DAS sends and receives commands as required by the spacecraft or mission profile, and processes ground commands to the science instruments. All science data is formatted and identified so that every bit is unique and can be referenced to its source in time and space.

1. Description

Each of the science instruments is sampled at a different rate and in a different format. The data are converted from the form presented by the instrument such as serial, parallel, analog-to-digital conversion, pulses which must be counted, etc., into a single, serial bit stream which is transmitted to the spacecraft telemetry system for transmission to Earth. Some data are submitted to the tape machine for recording and later transmission to Earth. The only functions which bypass the DAS (going from and to the science instruments directly from the spacecraft), are the 2.4-kc power from the power subsystem and the temperature measurements which are performed by the data encoder. The DAS receives commands from the spacecraft and issues commands to the spacecraft, as required to perform the science mission. The DAS provides identification information to the data such that each block of data bits is uniquely identified so that those science measurements represented by any block of data may be placed in time and thus in space through the knowledge of the spacecraft trajectory. The DAS processes not only scientific measurements, but also performs science subsystem performance measurements to help in the engineering evaluation of the science payload. Also, performance data assist in the calibration of the science instruments to provide confidence in the science data.

The DAS weighs approximately 12 lb and requires a maximum of 10.7 w. The system contains 11,021 electronic components in a volume of approximately 380 in.^{3} ($6 \times 7\frac{1}{2} \times 8\frac{1}{2}$ in.). The acceptable operating temperature range extends from $-10°$ to $+75°$C. With a few minor exceptions, all signals between the DAS and the spacecraft or between the DAS and the science instruments, are ac-coupled through transformers.

a. Real time DAS. The real time (RT) DAS is that portion of the DAS which operates continuously. It is turned on after launch and remains on until after the Mars encounter. The RT DAS sequences the cruise instruments and collects their data for transmission to the data encoder and thence to Earth. The cruise instruments consist of the following: helium magnetometer; plasma probe; ion chamber; cosmic dust detector; trapped radiation detectors; cosmic ray telescope.

These instruments gather data in the near-Earth environment, in deep space (in the interplanetary environment), and in the near-Mars environment.

For the majority of the mission, the spacecraft telemetry system is in the Mode 2 or cruise mode. In this condition, each spacecraft data frame contains 420 bits of data and is composed of two portions: one portion contains 140 bits of spacecraft engineering data from the data encoder; the other portion contains 280 bits from the data automation system. This is the science portion of each frame. The DAS divides the 280 bits into 10-bit segments so that there are 28 words in the science portion of the format. The science instruments do not necessarily require 10-bit words; some require more, some less. Therefore, the 10-bit words are, in some cases, combined or divided to provide the necessary length data word for the appropriate measurements.

During the Mars encounter period when the NRT DAS is operating, the real time (RT) DAS also sequences the data gathering from the scan subsystem and the television subsystem (except for video data). These data are combined with the other cruise science data and, when the telemetry switches to Mode 3, then all 420 bits of each frame of data are used by science and contain science data. The 280 bits originally transmitted remain essentially the same, while the extra 140 bits contain the scan and television information, along with additional measurements of some of the cruise information.

Conversion of analog data to a digital format is one of the prime functions of the DAS. For the helium magnetometer, which has three axes (X, Y, and Z), the

RT DAS performs three simultaneous A/D conversions. Six unipolar analog-to-pulse width (A/PW) converters are installed physically in the magnetometer. These comprise three bipolar ADCs when operated in conjunction with three 10-bit registers in the DAS. To perform the conversion, the DAS sends a start pulse to the A/PW converters in the magnetometer. This begins the analog comparison. Simultaneously with the start pulse, the DAS begins to count 55.5 kc into the three DAS registers (the X, Y, and Z registers). When an analog reference becomes equal to the input of magnetometer data, a stop-pulse is sent from the appropriate converter to the corresponding DAS register. This pulse stops the 55.5 kc count in the DAS counter-shift register. It also determines the polarity of the conversion. This polarity is indicated by the tenth bit in the DAS register; 0 indicating a positive conversion, 1 indicating a negative conversion. The remaining nine bits contain the binary number proportional to the analog signal which was measured, 511 being full scale (binary all ones) and equal to 6v.

After the three stop pulses, all three registers then have data indicating the magnetic amplitude for three axes. At the appropriate moment, the data are shifted from the X register into the telemetry bit stream. Then the Y register is shifted, and finally the Z register. In this manner, the three analog-to-digital A/D conversions are performed simultaneously, but the data flow into the bit stream in a bit-by-bit fashion. Similar A/D conversions are performed in the plasma probe instrument, in the scan instrument, and in the television instrument. In each of these cases, however, each conversion is unipolar, leaving the tenth bit always zero.

The plasma probe has 72 steps in its sequence. The DAS provides a stepping pulse to advance the next level prior to each A/D conversion. The plasma probe only requires one ADC. Similarly, the scan has only one ADC; however, it has six different inputs to the converter. These are switched with reed switches in the scan upon command from the DAS prior to each A/D conversion. The television operates essentially the same as the scan, with one ADC having two inputs and switched by reed switches as commanded by the DAS. The UV instrument requires two ADCs with three inputs. One input is connected to one ADC and the other two inputs are alternately switched to the second converter upon command of the DAS. This completes the analog-to-digital conversion system of the RT DAS.

The particle instruments require a different technique for data conversion. For the trapped radiation detector the DAS provides a 20-bit counter. Each frame of 420 bits of science and engineering data requires 50.4 sec to be transmitted at the cruise bit rate of 8⅓ bits per sec (bps). Thus, the frame period is 50.4 sec. There are five trapped radiation detectors and one geiger tube (which is associated with the ion chamber experiment). Every eight DAS frames, one of the six detectors is observed by the DAS through a gate which opens for 45 sec and allows the pulses for this interval to accumulate in a 20-bit counter. If this counter should count high enough to place a one in the 20th stage, then a divide-by-four pre-scaler is inserted into the count so that the counter now becomes a 22-bit counter. After the 45-sec gate has closed, the most significant ten bits of the counter are parallel-dumped into the X shift register of the DAS and are shifted out to the telemetry bit stream. Then the second ten bits of the counter are parallel-dumped into the X register and are shifted out to the telemetry bit stream. Thus the telemetry bit stream contains twenty bits in series indicating the number of pulses from one of the detectors in a 45-sec period. If the 20th bit is a 1, the data must be multiplied by four and added to 2^{19}, the original number of counts which caused the 20th stage to shift to 1. Of the six detectors, four are observed once in every eight DAS frames. The other two detectors are each observed twice during the eight-frame period.

The ion chamber is handled in a somewhat different manner by the RT DAS. This is because there is usually a larger period between pulses for the ion chamber than for the other particle detectors. For instance, during normal cruise the ion chamber may only pulse once every 7 or 8 min, compared to several pulses per sec for the trapped radiation detectors. Therefore, the DAS accommodates the dual possibilities for the ion chamber by measuring the time between pulses and/or the total number of pulses occurring within each frame. This is accomplished by one 10-bit counter in the DAS which is divided into two portions. The first portion is a 3-bit counter. This counts the number of pulses in a frame, up to the maximum of seven. In the average frame, the number of pulses in this counter will be either 0 or 1. The second portion of the counter consists of six stages which count time from the beginning of the frame until the first pulse occurs. A time pulse occurs every ten bits of telemetry data, or every 1.2 sec at 8⅓ bps. Therefore, each frame has a maximum possibility of 42 time pulses (equalling 50.4 sec) in this six-stage counter. This would occur if no ion chamber pulses were counted. If the ion chamber did pulse during this frame, a count would be noted in the first three bits of the counter. The time of this pulse would be noted in the second six bits of the counter because, when the pulse arrives, the time count stops. Now, if an additional pulse were to arrive, it would also be counted in the first three bits while the time count would remain unchanged, representing only the time of the first pulse. The same is true for three pulses, four, five, and six. On the seventh pulse, however, the time counter is reset and connected as an overflow counter to the first three bits so that time is lost, but now the measurement is performed by a single nine-stage ion pulse counter. This is useful during a flare period where the ion chamber pulses a large number of times in a frame. Under these conditions, the precise period between pulses is relatively unimportant compared to the total number in a given frame. In this manner, a total of 511 pulses can be counted. The tenth bit in this counter is simply the indicator for whether the six bits are time or overflow pulses from the basic three-stage counter. A 0 indicates that the 6 bits are time and a 1 indicates all 9 bits represent the total number of counts from the ion chamber for this particular frame of 50.4 sec.

The cosmic ray telescope has two 10-bit registers divided into several counters. However, the DAS treats these as two 10-bit counters. To transfer the data from the cosmic ray telescope to the DAS, the DAS selects one of the two counters and transmits 111-kc pulses to it while simultaneously counting the 111-kc pulses in one of the DAS registers. The cosmic ray telescope counts the 111 kc in its counter. When this counter reaches overflow, a stop pulse is sent to the DAS. The DAS then stops sending 111-kc pulses and stops counting. The DAS now has a number in its register which is indicative of the number originally in the cosmic ray telescope counter. This number, which is the data complement, is now shifted bit-by-bit into the telemetry data stream.

The remaining instrument in the cruise science complement is the cosmic dust detector. This instrument has two 8-bit output registers. The DAS makes a parallel conversion from one of the cosmic dust registers into the DAS X-register, i.e., eight wires carry eight bits simultaneously. This is initiated by the DAS dump command and causes the cosmic dust detector to alternately select one of its two registers. Thus it is seen that the RT DAS is required to perform many types of data conversions. This is required because of the varied design of the instrument interfaces. In addition to all of the DAS data conditioning functions, the DAS also maintains a clock which counts every bit, every word, and every frame up to a maximum of 2,048 or 2^{11}. This counter overflows approximately every 28⅔ hr at 8⅓ bps. At this time the count starts over again and calibration signals are sent to each of the instruments which require them: the magnetometer, the cosmic dust detector, and the cosmic ray telescope all use these calibrate signals to prove the validity of the scientific data generated by these instruments. The DAS clock frame count is shifted to the telemetry bit stream to uniquely identify each frame of data. The RT DAS also provides status information of various spacecraft events pertaining to the science mission and a pseudonoise sequence at the beginning of each frame to use as a reference starting point for the data.

b. NRT DAS. NRT DAS is dormant for the majority of the mission. It is normally turned on a few hours prior to Mars closest approach. At this time, the scan, television, NRT DAS, and video storage subsystem are all turned on and begin functioning. The NRT DAS is completely independent of the RT, and vice versa. Each has a separate power supply and sequencer. The NRT primary function is to provide the sequencing for the television recording portion of the encounter.

After turn-on, the NRT DAS begins issuing shutter commands to the television every 48 sec. It issues stop-tape commands to the video storage subsystem two times in each 144-sec period. It also issues line start and stop commands to the television, and A/D conversion commands similar to those issued by the RT DAS. At 8⅓ bps, the telemetry bit rate, each bit period requires 120 msec. This is used as the sync for line start. That is, each line of the television picture requires 120 msec to record.

The NRT timing is derived from a crystal oscillator which runs at 111 kc. This frequency is counted down through a 15-stage counter to make a basic period of 115.2 msec. All the microsequencing in the NRT DAS is performed in this period. The basic sync is the telemetry bit sync. When a bit sync occurs, this 15-stage counter begins to count. When it reaches 115.2 msec the count is inhibited until the next bit sync pulse (which occurs 4.8 msec later). Furthermore, each bit sync pulse is counted in a nine-stage counter which counts up to 200. This makes up the 200 television lines. The output of this counter is fed into a subframe counter which counts each 24-sec period (200 television lines). The subframe counter counts to six before recycling. Thus, an NRT frame is made up of 144 sec which are six 24-sec periods, each 24-sec period being made up of two hundred 120-msec periods, the 120-msec period being the basic line rate. In the 144-sec NRT frame, a television picture is recorded during the first 24-sec period; during the third 24-sec period, a second picture is recorded, to make an overlapping pair.

Recording occurs at 10.7 kc, which is the rate of transmission of data from the DAS to the video storage subsystem. The television, however, must scan at a high rate, with the result that the data from the television to the DAS occurs at 83.3 kc. This mismatch of data rates requires that the NRT DAS provide buffering between television and video storage subsystem. This is accomplished through the use of two 1320-bit buffers. While one buffer is being loaded with information, the second buffer is being read out to the video storage subsystem. When the second buffer is empty, the first buffer is read out to the video storage subsystem and the second buffer is filled with new information. In this manner the buffers transmit alternate television lines to the video storage subsystem.

Each television line of information contains 1200 bits of television data, derived from 200 elements with six bits per element, for a total of 1200. In addition, the DAS provides a 31-bit pseudonoise sequence to identify the start of each line of data. Also provided are: four bits for frame count to indicate of which picture pair this line is a member, three bits for subframe count to indicate which of the two pictures of the pair this line is a member, nine bits for the line number count, three bits for commutated digital television performance data, one bit of RT information, ten bits of commutated analog performance data, and an average of 23 spare bits at the end of each line.

The various blocks of bits are dropped into the buffer at various times within the 120-millisecond period. They are generated at various rates; for instance, the 31-bit pseudonoise sequence is generated at 13.9 kc, but the 1200 television bits are generated at 83.3 kc and only take up 14.4 msec of the total 120-msec line period. In addition to the 200 lines of television information sent to the video storage subsystem, an extra buffer load of information is sent at the beginning of each television picture. This is required because 80 zeros must be sent to the video storage subsystem to synchronize its timing before proper recording can commence.

All sequencing of the television and the buffers and the transmission of the data to the video storage subsystem begins at power turn-on and continues throughout the operation. The only function inhibited is the issuance of start-tape commands. These commands are not issued until the narrow-angle planet-in-view signals occur. Prior to that time, in the nominal encounter sequence, the scan system would have locked onto the planet and begun to track it. When planet tracking commences, the scan subsystem sends a wide-angle planet-in-view to the NRT DAS. This causes the DAS to issue a switch-to-Mode 3 command to the data encoder, which in turn causes all of the 420 RT telemetry bits to be allotted to the science RT DAS.

Then, when the planet comes into the field of view of the narrow-angle Mars gate and the television, these instruments send a narrow-angle planet-in-view to the NRT DAS. This initiates the NAS condition which in turn causes the NAA condition. At NAS, a signal is sent to the scan instrument, inhibiting any further motion. That is, 400-cycle power is removed from the scan motor.

The condition of NAA allows the recording sequence to start. Depending on the condition of the DAS clock, the recording sequence will commence when the clock reaches the appropriate point in timing after NAA has occurred. When this time is reached, a start-tape command is issued to the video storage subsystem. This starts the record motor on the video storage subsystem, which takes several seconds to reach record synchronization speed. A shutter command is issued to the television 3.6 sec after the tape-start command. The television then takes a picture which is retained in the vidicon. The DAS then immediately sends a television line-start command which causes the television to scan one line of the television picture. This information is loaded into the buffer and then to the tape machine, as previously described. After 24 sec and 200 lines, the first picture has been recorded and a tape-stop command occurs to stop the tape machine. Another start occurs 20.4 sec later. Then, 3.6 sec later, the next shutter occurs, followed by the next picture. Finally, 24 sec later, the stop-tape command occurs, followed by a 68.4-sec period before the next start-tape command. Thus, in a 144-sec period, the television has taken three pictures, two of which have been recorded by the tape machine. This sequence is repeated until one of the conditions required for termination of television recording occurs.

After the beginning of Picture 19, if the second end-of-tape occurs, recording ceases; that is, the DAS will not send any more start-tape commands. If the second end-of-tape does not occur by the end of Picture 22, further recording is inhibited regardless of end-of-tape. In addition to this mechanization of the DAS, the video storage subsystem automatically stops itself and inhibits further starts when it reaches the second end-of-tape. If the second end-of-tape occurs during Pictures 19, 20, or 21, a switch-to-Mode 2 data command is immediately sent to the data encoder. If the second end-of-tape occurs during Picture 22, the switch-to-Mode 2 data command is issued at the end of Picture 22. In either case, this mode switch command causes the telemetry system to be switched back to the condition of 280 bits for the RT DAS and 140 bits for the engineering data encoder. No further science events occur until the NRT DAS and encounter instruments are turned off several hours after Mars closest approach. Later, the RT DAS and cruise instruments are turned off and the tape machine commences the playback. This is essentially the end of the mission for the science system and the DAS, although the option exists after playback is complete, to again turn on science and the RT DAS in the cruise mode.

2. Performance

The DAS was designed to sequence the science instruments and to process science data into a format suitable for RT telemetry and NRT recording. Every DAS function has performed exactly as intended for the entire mission and continues to operate in an exemplary manner.

a. RT DAS. At the turn-on of science power after launch, the DAS began operation, sequencing the RT science instruments in their data gathering and then processing this data from all of the various forms from different instruments into a continuous, serial stream of data to the telecommunications subsystem. Since that turn-on, the RT DAS has functioned continuously, with two short commanded interruptions, precisely correct, without a single anomaly. Every bit of data produced by every instrument was sent to telemetry at exactly the correct instant in exactly the correct form.

In the more than 26,000,000 sec of operation during Phase I of the *Mariner IV* mission, over 200,000,000 bits of data have been processed by the RT DAS. One flip-flop has changed state well over two trillion (2×10^{12}) times without missing a beat. The system operated correctly from almost 32 to 80°F at various points in the mission.

b. NRT DAS. The NRT DAS was operated three times during the mission. It was first turned on for a short period at the science cover deployment exercise. The correct operation at this time gave confidence that encounter would be successful.

At Mars, the NRT DAS was turned on for the second time, after five months of non-operation in space. Its primary function was to sequence the TV system and, through the use of buffers, to change the 83-kc TV data to 10.7-kc data for the tape machine and to synchronize the tape machine recordings in proper sequence. In addition, the NRT DAS had to generate identification information and interleave TV engineering data and RT science data with the TV picture data transmitted to the tape machine. Furthermore, the initiation and cessation of the recording sequence and the switching of telemetry modes was controlled by the NRT DAS. Table 20 lists the encounter event times as determined from DAS data to the best possible accuracy. These times are all referenced to TTY print times (except command initiate times), so that, to be absolutely correct, 0.3 sec of fixed TTY delay should be subtracted from each time.

Every NRT function was performed exactly as designed. Every goal was fulfilled, thus providing the first pictures of Mars.

The only DAS events which could, even remotely, be considered non-standard for the entire mission, were two extra EOT events monitored through the RT data. These events were, in all probability, noise spikes on the EOT line. The DAS monitor flip-flop is quite sensitive and noise pulses would trigger it just as full-sized EOT pulses from the video storage subsystem would. The DAS was performing exactly as designed in every aspect, even when recording noise spikes. It is especially interesting to note that, functionally, the DAS ignored the spikes, even while monitoring them. It was purposely designed to not act on any EOT pulse prior to the beginning of Picture 19.

c. TV haze calibration. The NRT DAS was turned on for 4 hr, 14 min and 1.6 sec during the television calibration recording sequence on August 30, 1965. The first shutter pulse occurred 20.3 sec after turn-on. Television and DAS personnel computed time for the initiation of DC-16 to cause the second recorded picture to be taken through the red filter No. 2 in gain 01 (the minimum possible gain under these conditions). This calculation was based on knowledge of the filter position at turn-off at Mars encounter (green No. 2) and on the period of NRT operation for the initial attempt on August 21, 1965 (1 hr, 6 min, 14 sec), plus observation of Mode 3 data to derive the RT/NRT relationship.

Table 20. DAS data times at encounter

Event	Ground initiate	Spacecraft	Ground verify	Uncertainty[a]	RT frame	Bit
DC-25	14:27:55	14:40:32.8	14:52:32.3	−0.1 +1.0	411	273
MT-7	NA[b]	15:41:48.9	15:53:48.6	±0.3 sec	484	247
DC-24	17:10:18	17:22:55	17:34:55	−0.1 +1.0	604	401
DC-3	22:10:29	22:23:07	22:35:08	−0.1 +1.0	962	139
WAA (SPIV)	NA	23:42:00.3	23:54:01.6	±10.8 sec	1056/0	100
NAS	NA	00:16:50.1	00:28:51.5	±12.6 sec	41	295
NAA	NA	00:17:21.1	00:29:22.5	±0.1 sec	42	133
First start tape	NA	00:18:29.6	00:30:31	±0.1 sec	43	284
No. 1 shutter	NA	00:18:33.1	00:30:34.5	±0.1 sec	43	313
No. 2 shutter	NA	00:19:21.1	00:31:22.5	±0.1 sec	44	293
No. 3 shutter	NA	00:20:57.1	00:32:58.5	±0.1 sec	46	253
No. 4 shutter	NA	00:21:45.1	00:33:46.5	±0.1 sec	47	233
First EOT event	NA	00:22:30.3	00:34:31.7	±25.2 sec	48	190
No. 5 shutter	NA	00:23:21.1	00:35:22.5	±0.1 sec	49	193
No. 6 shutter	NA	00:24:09.1	00:36:10.5	±0.1 sec	50	173
DC-16	00:11:57	00:24:36	00:36:37	−0.1 +1.0	50	395
No. 7 shutter	NA	00:25:45.1	00:37:46.5	±0.1 sec	52	133
No. 8 shutter	NA	00:26:33.1	00:38:34.5	±0.1 sec	53	113
No. 9 shutter	NA	00:28:09.1	00:40:10.5	±0.1 sec	55	73
Second EOT event	NA	00:28:23.1	00:40:24.5	±25.2 sec	55	190
No. 10 shutter	NA	00:28:57.1	00:40:58.5	±0.1 sec	56	53
No. 11 shutter	NA	00:30:33.1	00:42:34.5	±0.1 sec	58	13
Third EOT event	NA	00:30:54.3	00:42:55.7	±25.2 sec	58	190
No. 12 shutter	NA	00:31:21.1	00:43:22.5	±0.1 sec	58	413
No. 13 shutter	NA	00:32:57.1	00:44:58.6	±0.1 sec	60	373
No. 14 shutter	NA	00:33:45.1	00:45:46.6	±0.1 sec	61	353
No. 15 shutter	NA	00:35:21.1	00:47:22.6	±0.1 sec	63	313
No. 16 shutter	NA	00:36:09.1	00:48:10.6	±0.1 sec	64	293
No. 17 shutter	NA	00:37:45.1	00:49:46.6	±0.1 sec	66	253
No. 18 shutter	NA	00:38:33.1	00:50:34.6	±0.1 sec	67	233
No. 19 shutter	NA	00:40:09.1	00:52:10.6	±0.1 sec	69	193
No. 20 shutter	NA	00:40:57.1	00:52:58.6	±0.1 sec	70	173
End NAMGA	NA	00:42:14.7	00:54:15.2	±25.2 sec	71	400

[a] Applies to *Ground initiate* when applicable; when *Ground initiate* is NA, *Uncertainty* applies to *Spacecraft* column.
[b] NA, abbreviation for *Not applicable*

Table 20. DAS data times at encounter (cont'd)

Event	Ground initiate	Spacecraft	Ground verify	Uncertainty[a]	RT frame	Bit
No. 21 shutter	NA	00:42:33.1	00:54:34.6	±0.1 sec	72	133
No. 22 shutter	NA	00:43:21.1	00:55:22.6	±0.1 sec	73	113
Last stop tape	NA	00:43:45.1	00:55:46.6	±0.1 sec	73	313
Switch to Mode 2	NA	00:43:45.1	00:55:46.6	±0.2 sec	73	313
DC-26	00:31:42	00:44:21.5	00:56:23.0	−0.1 +0.2	74	197
DC-2 (No. 1)	00:32:40	00:45:19.6	00:57:21.1	±0.3 sec	75	261
Telemetry lost	NA	02:18:44	02:30:45.8	±1.0 sec	110	343
Enter occultation	NA	02:19:11	02:31:12	NA	111	148
Exit occultation	NA	03:13:04	03:25:06	NA	175	209
MT-8	NA	05:01:49	05:13:52	±0.3 sec	304	404
MT-9	NA	11:41:49.8	11:53:53.3	±0.2 sec	781	68

[a]Applies to *Ground initiate* when applicable; when *Ground initiate* is NA, *Uncertainty* applies to *Spacecraft* column.
[b]NA, abbreviation for *Not applicable*

All events were precisely as predicted for this exercise. Table 21 lists the event times as determined from command initiation times and DAS data to the best possible accuracy.

3. Recommendations

a. DAS. Every goal was achieved, considering the constraints which were in effect during the design, fabrication and testing of the DAS. Had different constraints been in effect the program probably could have achieved the same success with less effort and less cost.

b. SPAC. SPAC activities were satisfactory, particularly after operations had been shifted so that all performance analysis personnel were physically located together. However, since operations in the future will doubtless involve more persons, a need is felt for closer operational ties between SPAC and SSAC. For example, where SPAC and SSAC are separated, the science SPAC representative should be supplemented by cognizant individuals from each of the science subsystems. This team would concentrate on the rapid analysis of science instrument performance for dissemination to others in SPAC and to SSAC. On the other hand, the SSAC operation might concentrate most heavily on analyzing the data for its scientific implications. Implicit in this change is the need for proper and sufficient additional output devices to support this SPAC team. The scheme in use

Table 21. TV haze calibration sequence events

Event	Initiate time	Spacecraft time	Earth verify time	Tolerance
DC-25	20:30:00	20:45:55.3	21:01:13.3	+2 sec −0
First shutter pulse		20:46:15.6	21:01:33.6	±0.1 sec
DC-3	21:10:24	21:26:19.4	21:41:37.5	+2 sec −0
DC-24	22:48:33	23:04:28.6	23:19:47.0	+2 sec −0
PN of Frame 0		23:51:20.5	00:06:39.0	−0.3 sec
DC-16	23:35:26	23:51:21.7	00:06:40.2	+0.4 sec −0
First start tape		23:52:36.0	00:07:54.5	±0.1 sec
First recorded shutter (green No. 2—00 gain)		23:52:39.6	00:07:58.1	±0.1 sec
End of tape No. 1		23:53:23.8	00:08:42.3	±25.2 sec
End of tape No. 2		00:05:09.4	00:20:27.9	±25.2 sec
Switch to Mode 2		00:17:39.4	00:32:58.0	±16.8 sec
DC-2	00:05:00	00:20:55.8	00:36:14.3	+2 sec −0
DC-26	00:44:00	00:59:56.0	01:15:14.9	+1.3 sec −0

at present by SPAC seems appropriate. That is, a raw printer giving data reducible to the science format and a high speed printer displaying selected parameters for each two subsystems. An effort of this magnitude is necessary to guarantee the timely, comprehensive reports required during critical mission phases.

J. Cosmic Ray Telescope

1. Description

a. Purpose

1. To detect and measure trapped corpuscular radiation in the vicinity of Mars.

2. To measure the flux and energy of alpha particles and protons as a function of position and time in interplanetary space.

Figure 42 shows a block diagram of the cosmic ray telescope and associated circuits leading to the development of the 10-bit words which are read into the data automation subsystem.

b. Function

Detectors. The complete detector system consists of three solid-state surface-barrier detectors, D_1, D_2, and D_3, arranged as a telescope. The depletion depth of each detector is 200 μ. The area of D_1 and D_2 is 2.38 cm^2, and of D_3 is 4.6 cm^2. The detectors are arranged so that the half-angle of the cone of acceptance of the telescope is 20 deg, and the geometrical factor is approximately 0.3 cm^2-ster. Absorbers are interposed between D_1 and D_2, and between D_2 and D_3. When an ionizing particle passes through such a surface-barrier detector, the detector puts out a pulse of charge whose size is proportional to the energy lost by the particle in the depletion layer of the detector. One ion-electron pair (equivalent to 1.6×10^{-19} coulombs charge) is generated for every 3.5 electron volts (ev) of energy lost.

Calibration. For operation in flight and during spacecraft testing, a calibrate mode of instrument operation is incorporated in the cosmic ray telescope. This utilizes the permanent installation of low-level Am^{241} alpha particle sources in front of each of the three detectors. These sources are encapsulated between two thin layers of Mylar and have only enough energy to penetrate to the detector on which they are mounted. In the data

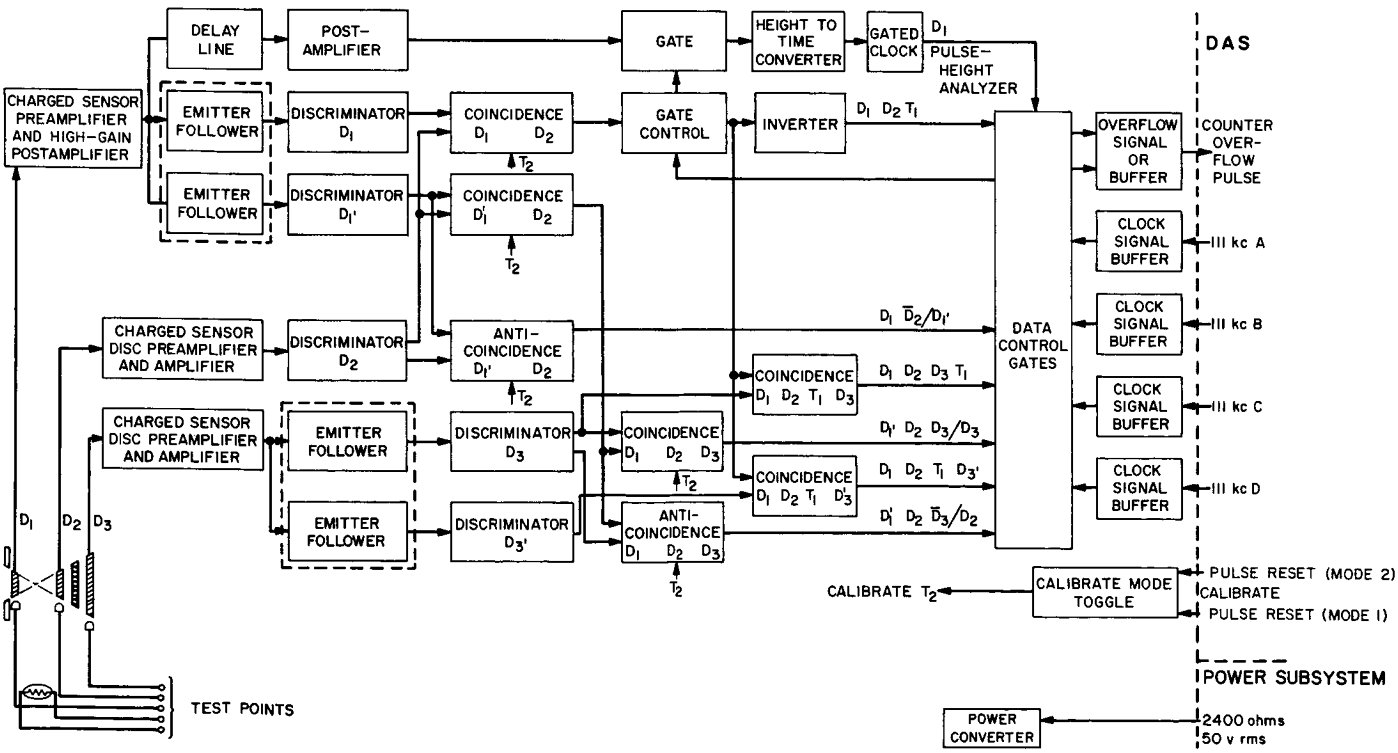


Fig. 42. Cosmic ray telescope block diagram

mode of operation, therefore, they produce no coincidences or pulse-height analyses but only contribute to the $D_1'\bar{D}_2$ rate. In the CRT-calibrate mode of operation, the coincidence circuits are disabled, allowing the individual count rates on D_2 and D_3 to be recorded and pulse height analysis of the $D_1\bar{D}_2$ events to proceed.

Amplifiers, discriminators, and pulse-height analyzers (PHAs). A charge-sensitive preamplifier and an amplifier is connected to each detector. A voltage pulse, proportional to the charge collected from the detector, is generated at the output of an amplifier. The amplifiers connect to pulse height discriminators D_1, D_1', D_2, D_3, and D_3', respectively. Discriminators D_1, D_1', D_2, and D_3 pass pulses only if the input to them exceeds a level corresponding to an energy loss of 210 kev. The threshold of D_3' is 400 kev. In addition, the amplifier associated with detector D_1 connects to a height-to-time converter which sorts pulses corresponding to energy losses between 210 kev and 5.2 Mev into one of 128 channels.

Logic circuits. The coincidence-anticoincidence logic provides the following outputs: D_1', $\bar{D}_2/D_1$, $D_1'D_2D_3/D_3$, and $D_1'D_2\bar{D}_3/D_2$ in the normal/calibrate modes. These three coincidence rates are recorded in different portions of the 10-bit register II which is read out as Words B and D. The $D_1'\bar{D}_2$ rate is prescaled by 4 before being recorded. One bit in register I (Words A and C) indicates if one or more coincidences $D_1'D_2\bar{D}_3$ have been received since the last readout. In the calibrate mode, the registers record the singles rated D_1', D_2 and D_3 as shown. In addition, the first double or triple coincidence after register I is read out by the DAS opens a gate which allows the height to time converter to analyze the pulse from D_1 associated with the coincidence. Pulses are provided to two indicator bits in register I which show whether the coincidence analyzed was double $D_1D_2\bar{D}_3$, triple $D_1D_2D_3$ or triple $D_1D_2D_3'$. In calibrate mode, the PHA simply records the first pulse from D_1 without coincidence.

Registers. Two 10-bit registers record the current rate and pulse height analysis data as shown in Fig. 42. The registers are read out alternately by the DAS. To accomplish this, the DAS adds pulses to the register at 111 kc/sec until the register overflows. The DAS counts the number of pulses required to produce an overflow. The registers are cleared by readout, and the PHA logic is reset.

Power converter. The power converter changes the 2400-cps spacecraft power into various dc voltages required by the detectors and electronics.

2. Performance

The instrument performed well throughout Mission Phase I of the *Mariner IV* interplanetary flight. Data were obtained within the design objective of the instrument for the counting rates, the PHA, and the calibrate mode. The PHA peak shifted down three channels during the first five or six weeks of flight, but after this period of time no more shift was observed. The shift was caused by a shift in a zener diode. This same shift in the PHA has been observed on other cosmic ray telescope instruments flown on other spacecraft by the University of Chicago. However, this small shift in the PHA does not affect the data.

3. Recommendations

Two design changes in the cosmic ray telescope would improve the data output considerably. The first change would add an anticoincidence shield to the instrument. This shield is used to reject secondary particles produced in the spacecraft.

The D_1 rate Word 6 assigned to the cosmic ray telescope overflowed approximately four times and Word 22 overflowed one time per readout. The second change in the telescope would add sufficient binary circuits to prevent the rate words from overflowing.

K. Cosmic Dust Detector

1. Description

a. Physical. The cosmic dust detector is a single assembly containing a sensor plate mounted on the top cover of an electronics chassis with a microphone bonded to one side and penetration capacitors on both sides, which provide measurements of dust particle impacts by direct penetration, and microphonic techniques. The instrument configuration aboard the spacecraft is such that the sensor is exposed to space, and the electronics chassis is beneath the thermal blanket.

b. Functional. The sensor is an aluminum impact plate, 0.030-in. thick, with a crystal acoustical transducer bonded to one side. The transducer will detect acoustical waves set up in the plate by impinging particles. Both sides of the plate are coated with 15,000 Å of nonconducting material over which is evaporated a thin film of aluminum. The impact plate-dielectric-aluminum film combination on each side constitutes a penetration detector in the form of a capacitor. With a static potential applied across the capacitors and a limiting resistor, an impacting particle which penetrates the dielectric will produce a voltage pulse across the resistor, which in turn will trigger the penetration circuitry. The films are used on both sides of the impact plate to provide an indication of the impacting particle's incoming direction. Data from the capacitors are stored in a three-bit binary accumulator. The microphone is the portion of the sensor from which momentum analysis is derived. The microphone output is amplified, rectified, and pulse height to pulse width converted, with the resultant being a direct indication of the impacting particle momentum. The microphone is inhibited for 30 msec after a particle impact, to allow for analysis of the particle. Thus, the penetration capacitor and microphone data combine to produce the following:

1. Particle momentum data (pulse-height analysis)
2. Particle incoming direction relative to the spacecraft
3. An accumulation of hits that may have occurred during the readout period
4. Binary accumulation in the microphone nonresetting hit accumulator

2. Performance

Shortly after launch, on November 28, 1964, bidirectional impacts were noted in the cosmic dust data, which continued until approximately 1730 GMT. System test data had shown this to be the result of electromagnetic noise. Since the instrument at this time would also be detecting acoustical creaking in the spacecraft due to rapid temperature changes, and separation of real hits from spacecraft generated noise would be extremely difficult, it was concluded that instrument operation under these conditions was normal.

On December 6, these fingerprints were again noted in the data. They continued from approximately 0200 GMT on December 6 to 0400 GMT on December 7. Investigation revealed that during this exact period, an intermittent arc had developed in the plasma probe high voltage supply. The events in the cosmic dust data were in agreement with the plasma probe anomalies.

The thin film capacitor sensors failed to register a dust particle impact during the first 146 days of flight (through April 23, 1965). The assumptions were that the films had failed or were damaged by impacts before power turn-on. Hypervelocity firings were conducted at Goddard Space Flight Center on the original backup sensor using iron spheres. This sensor worked in a satisfactory manner. It is the opinion of the experimenter that the particles impacting the *Mariner IV* sensor were most probably silicates, or cometary icy fragments; and it is quite conceivable that the threshold mass sensitivity of the films to these particles is one to two orders of magnitude greater than for the iron spheres. The microphone is sensitive to total mass only, not particle composition or density.

On April 24, 1965, a film hit was recorded in coincidence with a microphone event, indicating normal film operation. The microphone PHA was saturated. Another such coincidence was recorded on April 27. There are two films on the sensor, one on each side, and each one recorded a hit: the direct side was hit April 24; the retro side on April 27. These two events were considered to be real in view of the fact that they occurred during the period of maximum flux experienced to that time; and the microphone PHA was saturated, indicating particles of sufficient mass to activate the films.

3. Recommendations

Testing at Goddard Space Flight Center revealed that impacts on the sensor with no power applied could permanently short the penetration capacitors. On *Mariner IV*, during the period from shroud ejection to science power on, there was no power applied to the sensor. It is recommended on future flights, that a keep-alive voltage be applied to the sensor when the instrument power is off, to prevent the possibility of permanently shorting the sensor capacitors.

L. Trapped Radiation Detector

1. Description

a. Physical. The trapped radiation detector consists of three GM tubes, a solid-state proton detector, and a power supply within a single magnesium chassis. The electronics for the solid-state proton detector are contained in a separate chassis within the main chassis.

The GM tubes are referred to as Detectors A, B, and C, and are sensitive to electrons and protons within given energy ranges, Table 22. The solid state proton detector is referred to as Detector D or $D_1 - D_2$ (two discrimination levels), and is sensitive to protons within given energy ranges.

b. Functional

GM tubes. Detectors A, B, and C are EON type 6213 GM end window counters, which measure the total

Table 22. Detector characteristics

Detector	Unidirectional geometric factor	Omnidirectional geometric factor	Particles to which sensitive	Dynamic range
	cm^2 ster	cm^2		
A	0.044 ± 0.005	~0.15	Electrons: $E_e > 45$ kev Protons: $E_p > 670 \pm 30$ kev	From galactic cosmic ray rate of 0.6 counts/sec to 10^7 counts/sec
B	0.055 ± 0.005	~0.15	Electrons: $E_e > 45$ kev Protons: $E_p > 550 \pm 20$ kev	From galactic cosmic ray rate of 0.6 counts/sec to 10^7 counts/sec
C	0.050 ± 0.005	~0.15	Electrons: $E_e > 150$ kev Protons: $E_p > 3.1$ Mev	From galactic cosmic ray rate of 0.6 counts/sec to 10^7 counts/sec
D_1	0.065 ± 0.003	—	Electrons: none Protons: $0.50 \leq E_p \leq 11$ Mev	From in-flight source rate to 10^6 counts/sec
D_2	0.065 ± 0.003	—	Electrons: none Protons: $0.88 \leq E_p \leq 4.0$ Mev	From in-flight source rate to 10^6 counts/sec

number of charged particles passing through their sensitive volumes. The sensitive volume of each tube is shielded such that low energy particles can enter only by passing through the window at the end of the tube. By allowing for omnidirectional flux of higher energy particles, a directional measurement can be made of the low energy particles.

The windows of Detectors A and B are approximately 1.4 mgm/cm^2 of mica, making the tubes sensitive to electrons ≥ 40 kev and protons ≥ 500 kev. A shield defines the angle through which the particles may enter the window. This look angle is 60 deg and lies along an axis that is 135 deg to the Sun–probe line for Detector A, and 70 deg to the Sun–probe line for Detector B.

A visor is incorporated on the end of each tube housing to protect the detectors from exposure to direct sunlight during cruise.

An aluminum foil of approximately 20 mg/cm^2 is placed behind the aperture on Detector C to increase the energy threshold to 150 kev for electrons and 3 Mev for protons. The look angle of 60 deg for Detector C is along an axis 70 deg to the Sun–probe line. The shield configuration is the same as that of Detectors A and B.

The GM counter outputs are shaped by means of saturating current amplifiers before being sent to the DAS.

Solid state Detector $D_1 - D_2$. Detector D is a silicon surface-barrier diode, having a thickness of 35 μ, and covered with a nickel foil, 0.242 mg/cm^2 of air equivalent, to make it light tight. The detector, used as a two-channel proton spectrometer, is capable of detecting protons in a high electron field. Sensitivity to electrons is minimized by: 1) making the detector too thin to absorb energy from electrons, 2) setting the discrimination level much higher than the expected electron energy loss and, 3) using delay line pulse clipping. The energy discrimination levels are: 1) Channel D_1, 500 kev and 11 Mev and, 2) Channel D_2, 880 kev and 4 Mev. The look angle of Detector D is 60 deg and lies along an axis 70 deg to the Sun–probe line.

The Detector D output is fed into a linear-charge sensitive amplifier (preamplifier), which is followed by a series of highly-stable negative-feedback voltage amplifiers (post amplifiers). The output of the second post amplifier is fed in parallel to two more post amplifiers having different voltage gains. The outputs of these amplifiers are in turn transferred to two identical amplitude discriminators with a common reference supply. Gains of the post amplifiers are set to produce pulses at the discriminators that correspond to the particles detected. The discriminator reference supply is temperature compensated to keep the discrimination levels constant.

2. Performance

Except for one period between February 5, 1965, and March 5, 1965, the instrument performance was normal.

During the above-mentioned period there was an increase in bad data points in the user program, which was attributed to the reduction of low-gain antenna signal strength. When the switch to the high-gain antenna was completed, the problem disappeared.

The performance of the instrument was such that the scientific aims were accomplished. These aims were:

1. The detection and preliminary measurements of the intensity, composition and distribution of any magnetically trapped particles (belts) in the vicinity of Mars.
2. To study the occurrence of energetic particles in interplanetary space, with reference to their identity, energy spectra, angular distribution with respect to the Sun–probe line, and their time histories.

M. Ionization Chamber

1. Description

The ionization chamber experiment on the *Mariner* Mars 1964 spacecraft employs two detectors. The first detector is an integrating ion chamber of the quartz-fiber electrometer variety. The quartz electrometer is housed in a 5-in. diameter stainless steel sphere filled with four atmospheres of argon gas. The second detector is a conventional, halogen quenched GM tube, Type 10311, which was procured from Radiation Counter Laboratories. The GM tube is mounted in a thin-walled shield of stainless steel 5 in. long and approximately ¾ in. in diameter.

The two detectors are mounted to a gold-plated chassis which houses the high-voltage power supply and the pulse amplifying and shaping electronics. To minimize the effects of backscattering and shielding due to the presence of the spacecraft, the detector package is mounted part way up the cylindrical wave guide.

The quartz electrometer, housed within the ion chamber sphere, consists of a quartz collector rod and a thin quartz fiber that is separated from the rod by approximately 0.02 in. The collector is coated with Aquadag and the fiber is coated with a thin film of metal. Although the collector is electrically isolated from the fiber and sphere, it is normally positively charged. This charge is produced by a +360 vdc potential on the fiber, which induces a negative image charge on the Aquadag surface of the collector near it. As a result of this image charge, the fiber is attracted to the collector and momentarily contacts it. During contact electrons pass from the collector to the fiber. When the electron exchange is completed, the fiber springs back to its normal position leaving the rod positively charged to +360 v.

When ionizing particles penetrate the sphere, the gas within the chamber becomes ionized. The resulting positive ions are attracted to the grounded sphere while the electrons are attracted to the collector and tend to neutralize its positive charge. When the collector charge is sufficiently neutralized, the image charge, as previously described, attains sufficient magnitude and the fiber is pulled to the collector and finally contacts it. A surge of electrons passes from the collector to the fiber and through a load resistor. This current pulse is amplified, shaped, and presented to the DAS. Each output represents a fixed amount of charge collected from the gas. Thus, the interval between pulses is inversely proportional to the ionization rate.

The GM sensor comprises a glass tube encased in a stainless steel cylinder. The anode consists of a tungsten wire stretched along the axis of the tube while the cathode consists of a thin, transparent, conductive coating on the inner glass wall. A potential of approximately 900 v is maintained between the electrodes. Charged particles which penetrate both the steel shield and the glass wall and enter into the sensitive volume of the tube ionize the gas molecules. The high electric field between the anode and the electrode accelerates the dissociated ions and electrons. The electrons accelerate rapidly due to their low mass and ionize other gas molecules. The ultimate result of the avalanche effect thus produced between electrodes is a momentary current flow through an appropriate load resistor. The resulting pulse is amplified, shaped, and delivered to the DAS. One output pulse is generated for each charged particle that penetrates the shield and enters the sensitive region of the tube. Thus, the pulse rate is directly proportional to the radiation flux.

Because of a judicious choice of GM tube shield thickness, both the ionization chamber and the GM tube detect particles of the same energy; i.e., electrons of energy greater than 0.5 Mev, protons of energy greater than 10 Mev, and alpha particles of energy greater than 40 Mev are capable of penetrating the walls of either detector. Both sensors have omnidirectional sensitivity. The ionization chamber measures the average rate of ionization and the GM tube measures the omnidirectional flux.

The time between successive output pulses is the important ionization chamber parameter to be measured, while in the case of the GM tube the number of pulses per unit time is the parameter required. In both cases the respective pulses are transmitted on separate lines to the DAS for handling. The DAS measures the time interval between the beginning of a frame and the arrival of an ion chamber output pulse in increments of 1.2 sec or 0.3 sec, depending on the prevailing bit rate. In addition, the number of ion chamber pulses per frame is recorded. If the rate from the ion chamber is one pulse per frame or less (the usual case in quiescent space), the time between ion chamber output pulses can be calculated. For more than one pulse per frame, the exact time between pulses cannot be obtained, but the number of pulses per frame is more significant at these rates and is directly recorded. At rates of 7 or more pulses per frame, all interval-timing bits are reset and the entire ion chamber data word is devoted strictly to counting pulses. An indicator bit is set by the eighth pulse in any one frame and is useful in reducing the data. In this mode the data register is capable of tabulating as many as $(2^9 - 1)$ pulses per frame. Ion chamber data are read out once every frame.

The GM tube registers are capable of recording as many as 5×2^{19} pulses per frame. When 19 bits of the 20-bit register are filled $(2^{19} - 1$ pulses per frame), the next pulse triggers a change of tabulating mode. Incoming GM tube pulses are then prescaled by a factor of four before being recorded in the data register. The change to the prescale mode is signalled by an indicator bit to avoid ambiguity. Data from the GM tube are commutated with data from other sensors and consequently are read out only every fourth frame. Pulses are accumulated for 45 sec or 11.25 sec at 8⅓ and 33⅓ bits/sec respectively.

2. Performance

The *Mariner IV* ionization chamber performed nominally through the first 70 days of the mission. During this period, a profile of radiation intensity was faithfully recorded by the instrument as the spacecraft passed through each of the Van Allen belts, through the boundary of the magnetosphere, and into the relatively stable interplanetary region. Sixty-nine days after the mission began, i.e., January 5, 1965, a solar flare occurred. Both the ion chamber and the GM tube recorded peak activity of the flare. As the flare subsided and the ion chamber returned to the normal background level, the GM tube indicated a slightly higher rate than expected over the downward trend, leveling off at a value approximately 30% above the expected preflare level. On the 84th day of the mission, February 20, after about 10 days of comparatively stable operation at this slightly anomalous level, the GM tube rate rose sharply from approximately 40 counts/sec to 18,000 counts/sec, indicating a spontaneous, self-sustained discharge.

On the 109th day of the mission, March 17, the GM tube counting rate dropped to zero and the ion chamber interval became infinite. No response from either sensor has been observed since that time.

It is calculated that the GM tube had first failed (by spontaneous discharge) after having counted approximately 8×10^8 counts, i.e., after expending only 8% of its rated 10^{10} count life. In an attempt to understand this phenomenon, ten tubes were taken from the batch from which the flight tubes were selected. Seven tubes were operated at various fixed count rates until such time as each tube failed by continuous discharge. Three tubes were exposed to the higher flux levels, but without power applied. No significant shift in the voltage plateaus of these latter tubes was observed.

The tentative conclusion from the results obtained was that the total life-count of a tube depends upon the rate at which the tube counts; the higher the count rate, the more counts a tube will assimilate before the onset of spontaneous discharge. Since each pulse is smaller at higher rates, it is likely that the tubes fail by spontaneous discharge after a certain amount of charge has passed through them. It also appears that the tube selected for *Mariner IV* may have been an especially short-lived tube in spite of the excellent stability displayed in selection tests and also in prelaunch systems tests. It appears that any other tube in the batch would have had a life-count capacity capable of completing the mission.

In an effort to understand the mechanism of the failure mode, it was speculated that perhaps the quench gas, a mixture of chlorine and silicon tetrafluoride, was depleted. This would explain the sustained count rate in the absence of sources. With this in mind, good tubes and failed tubes were submitted to Chemistry Section 326 for analysis and comparison. Analysis of the filling gas as performed by a mass spectrometer indicated no significant differences between gas from good tubes and gas from expended tubes. However, this test is deemed inconclusive due to the poor sensitivity of the available spectrometer. Gas samples have been submitted for further analysis and comparison on a spectrometer with improved sensitivity.

A darkening of the transparent cathode of the expired tubes was noted as they were allowed to operate in the spontaneous counting mode. Chemical analysis of cathode washings showed chlorine, one of the quenching gases, to be present on the cathodes of both good and expired tubes.

The most significant and entirely unexpected finding occurred when the tin oxide cathode of a blackened tube was carefully scraped and the scrapings submitted to spectrographic analysis. Tests showed a remarkable 72% (by weight) tungsten content. The mechanism for tungsten transport from anode to cathode is not yet fully understood.

The subsequent failure of the ion chamber portion of the instrument on the spacecraft, accompanied by the apparent simultaneous drop to zero in the GM tube rate gave rise to speculation that perhaps the power supply, which is common to both detectors, had failed or had been shorted. This speculation was made more credible in view of the tungsten transport phenomenon. It was hypothesized that the anode was steadily weakened by the emission of tungsten until electrostatic attraction by the cathode was sufficient to break the anode. Further, it was theorized that the slender anode might be attracted to the cathode to the extent that physical contact might be made, the ensuing arc providing the energy necessary to weld the anode to the cathode, resulting essentially in a short circuit of the power supply. In order to provide a method of quickly testing the hypothesis, it was necessary to find a means of severing the anode of the tube while the tube was counting normally under applied high voltage. To accomplish this feat, a good tube was mounted to the breadboard circuit. By means of optical lenses, the beam from a ruby laser was focused through the glass envelope of the tube onto the anode. The laser was pulsed, with the tube in normal counting operation and powered by the breadboard ion chamber circuit. The anode burned in two and the energized segment was attracted to the cathode as postulated. However, welding of the anode to the cathode did not occur immediately, but required a period of approximately 40 min, during which time the arcing between the electrodes was visible through the glass tube. A possible reason why the welding was not instantaneous in this case, but may have been instantaneous with the unit in space, was that burning the anode in half resulted in the formation of a small ball of tungsten on the melted ends of the anode. The additional mass of tungsten, slightly larger than the diameter of the anode wire, might be expected to require considerably more energy for welding than an anode wire eroded to the breaking point by tungsten emission. Also, an eroded wire might have significantly less restoring force to counter the electrostatic attraction.

Additional evidence supporting the shorted power supply hypothesis was later obtained. Three failed tubes functioning in their spontaneous discharge modes were successively operated on the breadboard circuit until output from the tubes ceased. In all three cases when each tube finally quit producing the spontaneous output pulses, it was noted that the filaments had broken. In two of the tubes, the anode was observed to have been instantaneously welded to the cathode, resulting in a short circuit of the power supply. The third tube proved to have an open circuit within the tube. This occurred because the break in the anode was located so close to the glass feed-through sheath as to provide insufficient length for the energized segment of the broken anode to reach the cathode.

Further evidence for the postulated failure mode occurred when the life-test unit failed in a manner much the same as the *Mariner IV* instrument. Output of the GM tube dropped to zero after a count-rate profile and a time interval comparable to that of the spacecraft instrument. Simultaneously, the test voltages at all monitor points dropped in a manner that suggested a shorted supply. However, the ion chamber sensor continued to function with only slight increase in the expected period as opposed to the immediate failure of the chamber in space. This apparent discrepancy in the failure modes seems to have been first resolved by subsequent tests on the breadboard circuit. Breadboard tests based on the assumption that the power supply is shorted through the GM tube indicated that, if the life test unit (operating at ambient room temperature) were subjected to cold temperatures comparable to those experienced by the unit in space, the high voltage would decrease to a value below the ion chamber pulsing threshold. This assumption was verified by subjecting the life test unit to low temperatures, in vacuum, after life tests were completed. The ion chamber ceased operation at $-14°F$.

Early in the failure investigation there was one bit of evidence that seemed to contradict the proposed failure mode. Based on tests performed on the breadboard it was calculated that, if a short occurred in the GM tube as postulated, the step function increase in power dissipated by the instrument would probably be detectable in the instrument's thermal performance. A review of thermal data from the *Mariner IV* instrument during the time in question did not indicate any obvious change in dissipated power. However, the apparent discrepancy was resolved during tests performed on the life test unit, in thermal vacuum, after life tests were completed. These tests revealed an increase in power dissipation of only 100 mw when the GM tube was shorted. This resulted in a temperature increase of less than 3°F, a change small enough to challenge the resolving power of the thermal measuring system. Based on this test, personnel from the thermal control area feel that the flight temperature data received during failure are not inconsistent with the postulated failure mode.

Finally, after all pertinent tests were completed on the life-test unit, the failed GM tube was removed from the instrument for observation. Inspection of the tube revealed a broken anode, the end of which made physical contact with the blackened cathode.

3. Recommendations

The Type 10311 GM tube was originally chosen because it was the only tube commercially available which had the proper geometric factor and wall thickness, and was constructed of nonmagnetic materials. Other GM tubes recently made available commercially should be evaluated for potential use in future missions.

The problem of determining the total life-count capability of a given tube early in its life should be investigated. This problem might prove to be as simple as ensuring proper quality control during fabrication and filling of the tube.

In the area of circuit design, when two or more sensors operate from the same power supply, the design should preclude the possibility of a short circuit in either detector affecting the operation of the other, no matter how remote the possibility of a short may appear.

Finally, the necessity for completing life testing of flight hardware early in the program, and certainly before launch, has been demonstrated by the experience derived from *Mariner IV*.

N. Planetary Scan Subsystem

This section describes the planetary scan subsystem performance in flight. To relate the performance to the various operations, a brief description of the subsystem operation is included. The subsystem was energized three times throughout the flight. It was first energized on February 11, 1965 during the science cover deployment sequence, then during the 9-hr planet encounter period on July 14, 1965, and finally during the television haze calibration August 2, 1965. Subsystem performance analysis is made on the various devices having unique engineering characteristics. In spite of their superficially unreliable appearances, these devices have proven to be reliable in operation. These devices with their unique physical and performance characteristics should be applicable to the design of future systems.

1. Description

The planetary scan subsystem was designed and developed to support the planet encounter activities during the 9-hr period prior to the television picture-taking sequence. Its primary function was to orient the television camera correctly for picture taking.

The original operational sequence and the various functions designed into the subsystem were as follows:

a. Searching operation. When power to the encounter instruments is turned on by either the on-board CC&S function MT-7 or the Earth-based command DC-25, the scan subsystem enters into a planet searching operation. Since the television is mounted on a platform which rotates about the roll axis of the spacecraft, the subsystem drives the platform back and forth through 180 deg of arc searching for the planet at a rate of 0.5 deg/sec. This scanning provides planet searching motion in one direction while the motion of the spacecraft provides planet searching motion approximately perpendicular to the scanning plane. Limit switches actuate scan reversals when the platform completes each of the 180-deg rotations. To provide redundancy, two switches connected in parallel and operated in series are incorporated at each of the two scan reversal limits.

b. Tracking operation. Mounted on the platform and bore-sighted with the television is a wide-angle planet sensor. This sensor, which has a 50-deg circular field of view, detects the presence of the planet and senses the position of the planet with respect to the spacecraft. When the planet comes into the field of view of the sensor, the subsystem generates a wide-angle planet-in-view (PIV) signal and switches the subsystem to planet tracking such that the television is oriented toward the desired portion of the planet. When the spacecraft reaches the proper position for picture taking, a scan-inhibit signal is initiated by either the narrow-angle Mars gate, which was also mounted on the platform, or by the television system itself. This signal indicates that the planet is in the television field of view, stops the tracking motion, and initiates the picture recording sequence.

At encounter on July 14, 1965, however, this automatic planet tracking sequence was not exercised and the platform was stopped at a predetermined position through the use of the scan inhibit command DC-24.

c. Telemetry measurements. The wide-angle sensor was constructed such that two differential voltage outputs, those from the X-axis and Y-axis, were formed from the outputs of the sensor's quadrants through a resistor matrix. The scan platform motion affects the X-axis output and this output was used as the error signal for planet tracking. The spacecraft orbital motion affected the Y-axis output which provided a telemetry indication of planet brightness and the relative position of the planet with respect to the spacecraft. The Y-axis output, together with other data required for subsystem performance evaluation, was telemetered back to Earth in near real time. The various data channels and their functions are shown in Table 23.

Table 23. Data channels and functions

Channel	DAS channel number	Function
Scan position No. 1	21	Indicates the pointing direction of the platform.
Y-axis output	29	Indicates the brightness and relative position of planet.
Calibration signal	36	For calibration of scan position data and for indicating proper operation of electronic power supply.
Detector temperature and scan actuator pressure	37	Indicates temperature and pressure in alternate data frames.
Transformer-rectifier voltage	40	For calibration of scan position data and for indicating proper operation of electronic power supply.
Scan position No. 2	42	Indicates the pointing direction of the platform.

2. Performance

a. Science cover deployment exercise. The first block of subsystem flight performance data was obtained during the science cover deployment sequence on February 11, 1965, seventy-five days after launch. The scan subsystem responded properly to DC-25, the encounter science power-on command, by immediately entering the search mode. The correct initial search direction was obtained and the subsystem made a total of ten search cycles. Analysis of the received data indicated the subsystem was operating under the following conditions:

1. All scan reversals were actuated by the primary switches since each of the reversals occurred at the proper limiting positions. The redundant or back-up switches were not used.
2. All searching logic, including the microelectronic integrated circuits operated properly and there were no premature scan reversals due to improper logic state changes.
3. The average searching time had decreased by 4 sec per search cycle from the pre-launch scan period of 717 sec per cycle.
4. The pressure readout indicated that the actuator was operating under full pressure with an insignificant amount of change over the prelaunch value of 30 psia.
5. The subsystem sensor and optics operated at a temperature of minus 3°C as compared to the operating temperature of 0°C for the neighboring instrument, the television optics assembly.

The Y-axis output indicated that no objects having radiation energy of $1.0 \mu w$ or greater in the integrated spectral range of 0.5 to 1.1 μ were observed by the wide angle sensor. The numerical value of this output remained constant at a value corresponding to no significant input.

At that time, it was decided to time the scan inhibit command, DC-24, to stop the scan motion. This operation was performed for two reasons:

1. To determine whether the platform could be stopped accurately at a predetermined position, and
2. To stop the platform at a position which would ensure the attainment of at least some television pictures of Mars in the event of any later spacecraft failure which would prevent platform motion at encounter.

Accordingly, the time for the initiation of DC-24 was calculated based on the desired television pointing direction, in terms of spacecraft clock angle, and the scan speed calculated from the data received for the first six scan cycles. The command time was projected four scan cycles ahead to allow for calculation time, the various transmission delays and time for command processing. Upon receiving DC-24, the platform stopped 0.72 deg away from the desired position.

The success of this platform positioning operation via ground command led to the consideration of using this method as another means of positioning the television camera. Subsequent tests with the proof test model spacecraft indicated that it was possible to pre-position the scan platform over a simulated communication range of 135 million miles with a positioning accuracy of 1 deg. The performance of the scan subsystem during the early science cover deployment sequence is summarized in Table 24.

Table 24. Scan subsystem performance during science cover deployment sequence

Function	Prelaunch or predicted value		Actual data	
	(DN)	Engineering units	(DN)	Engineering units
Average scan search time per cycle, sec		717.0		712.9
Desired platform inhibit position, deg		—		177.19
Actual platform inhibit position, deg		—		177.94
Calibrate signal, v	(85)	0.998	(85)	0.998
Transformer-rectifier voltage, v	(417)	4.896	(417)	4.896
Sensor temperature, °C			(318)	−3.0
Actuator pressure, psia	(143)	30	(146)	30
Y-axis output with no input, v	(261)	3.064	(261)	3.064

b. Encounter. On July 15, 1965, the scan subsystem responded to the DC-25 by initiating a planet searching operation at 14:40:33 GMT at the spacecraft. The subsystem began its search in the correct direction. To evaluate the planet searching performance, a graph of the platform position as a function of search time was generated based on the received real-time flight data. Analysis of the graph indicated that the subsystem was operating properly under the following conditions:

1. The scan reversals occurred at the predicted position as indicated by the magnitudes of the position output voltage. Each of the reversals was actuated by a primary switch.
2. The linearity of scan platform position as a function of search time and the position output voltages at the reversal points were within the specified limits. Therefore, no corrections were required on the established conversion table of scan position voltage to scan position in terms of spacecraft clock angle.
3. All searching logic, including the microelectronic integrated circuits, operated properly and there were no premature scan reversals due to improper logic state changes.
4. The average searching time per cycle was within ½ sec of the rate calculated during the early science cover deployment sequence.
5. The pressure readout indicated that the actuator was operating at a pressure of about 26 psia, or a decrease of 4 psi after some 8 mo of space travel.
6. The subsystem sensor and optics operated at a temperature of minus 8.5°C.

The original encounter operating plan was revised such that, operational and scan subsystem conditions permitting, the scan platform inhibit command DC-24 would be used to pre-position the platform. The command initiation time was to be calculated to stop the platform at a position for optimum pictures. The automatic planet tracking sequence was to be used as the backup operation in the event the pre-positioning could not be exercised or failed to produce the intended results. Because of the occultation experiment, the final prelaunch trajectory change was such that the relative value of television pictures with automatic planet tracking was not optimum.

Accordingly, a command initiation time was calculated based on the desired television pointing direction, the calculated scan search speed averaged over the first seven search cycles, the communication transmission delay and command processing time. The initiation time of DC-24, projected seven scan cycles ahead, was calculated to be 17:10:18 GMT for a desired inhibit position of 179.19 deg in clock angle. The calculation was based on the following selected parameters:

1. Average scan time per cycle = 713.4 sec
2. Communication transmission delay time for the last frame of scan data used in the command time calculation = 11 min 59.8 sec
3. Communication transmission delay time at time of command initiation = 12 min 00.0 sec
4. Command processing time = 37.3 sec
5. Motor coastdown time = 0.2 sec

Upon receipt of DC-24, the scan platform motion was inhibited at a clock angle of 178.45 deg, or 0.72 deg away from the desired position. This positioning was considered acceptable since the television pictures at the actual inhibited position would have essentially the same relative value as pictures taken at the desired pointing position.

At 23:42:00 GMT at the spacecraft, the wide-angle sensor detected the edge of the planet and the subsystem initiated a planet-in-view signal. Twenty-three min later, the planet came into the television camera field of view. The Y-axis output, as the planet entered and swept across the sensor field of view, varied in accordance with the relative planet/spacecraft position and the amount of radiant energy received. Figure 43 shows the actual Y-axis output as a function of GMT (Curve A) along with the calculated Y-axis output based on the pre-encounter trajectory data (Curve B). Because of the Y-axis amplifier feedback adjustment, the Y-axis output was saturated part of the time and the Y-axis output curve has been expanded through extrapolations and calculations. Figure 43 illustrates that Curve B is located to the left of Curve A, indicating that the planet entered the wide-angle sensor field of view about 5 min earlier than expected. This deviation is partially due to an error in the prediction of the spacecraft arrival time and partially due to error in the Y-axis output prediction calculation. The performances of the scan subsystem during the planet encounter period are summarized in Table 25. For comparison, the data received during the early science cover deployment sequence is also shown.

*c. **Television haze calibration exercise.*** The scan subsystem was energized during the television haze calibration sequence on August 8, 1965. The subsystem operated properly throughout the sequence. Upon the receipt of encounter science power resulting from DC-25, the subsystem responded by initiating a planet searching operation in the correct search direction. A total of eleven search cycles was performed with each reversal actuated by the primary limit switches. During this time, the subsystem sensor and optics assembly was operating at a temperature of $-11.0°C$ as compared to

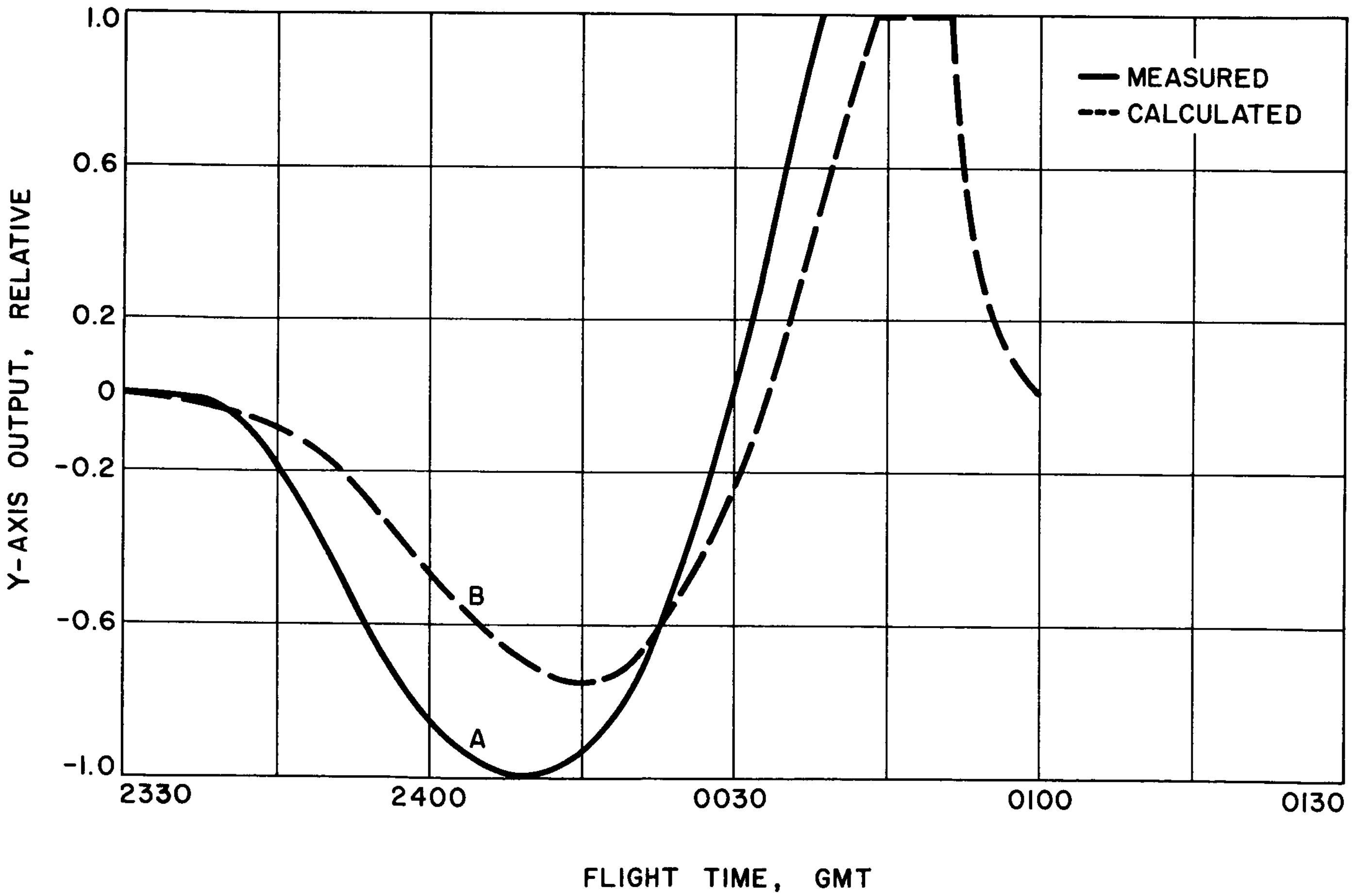


Fig. 43. Scan subsystem Y-axis output at encounter

Table 25. Scan subsystem performance data

Function	Prelaunch or predicted value	Data readout during science cover deployment	Data readout during planet encounter
Average scan time per cycle, sec	717.0	712.9	713.4
Desired platform inhibit position, deg	—	177.19	179.19
Actual platform inhibit position, deg	—	177.94	178.45
Calibration signal, v	0.998(85)[a]	0.998 (85)	0.998 (85)
Transformer-rectifier voltage, v	4.896 (417)	4.896 (417)	4.896 (417)
Sensor temperature, °C	−10°C for encounter operation	−3.0 (318)	−8.5 (351)
Actuator pressure, psia	30 (143)	30 (146)	26 (187)
Y-axis output with no input, v	3.064 (261)	3.064 (261)	3.076 (262)
No. of search cycle performed	—	10	14

[a]Number in parenthesis indicates the Data Number.

−8.5°C at planet encounter. The Y-axis output remained constant at a numerical value corresponding to no significant input, i.e., the Y-axis output indicated that no objects having radiation energy of 1.0 μw or greater in the integrated spectral range of 0.5 to 1.1 μ were observed by the wide-angle sensor. All other signals were well within the specified and predicted limits.

The initiation time for the scan inhibit command (DC-24) was based on the flight data received for the first four scan cycles and a one-way communication delay time of 15 min 18.3 sec. The average searching time per cycle was 713.5 sec, or 0.1 sec longer than that calculated during the planet encounter sequence on July 15, 1965. The calculated initiation time for DC-24, projected seven scan cycles ahead, was 22:48:33 GMT for a desired inhibit clock angle of 148.80 deg. Upon receipt of DC-24, the scan platform motion was inhibited at a clock angle of 148.43 deg, 0.37 deg from the desired position.

d. Overall subsystem performance analysis. As mentioned, a graph of the platform position as a function of planet search time was made to determine the subsystem planet search performance. Data of the first search cycle on the graph indicated that the motor initiated the search operation at a proper speed. This observation revealed that the motor was driving a nominal starting torque and no excessive bonding between the surfaces of the platform shaft and spacecraft structure was experienced. Although the motor assembly pressure had been reduced from 30 to 26 psi, the motor was considered operating in a suitable environment since the assembly was designed to survive the 8-mo journey and to operate at encounter even in a hard vacuum. The pressurization was simply an added safety factor.

The subsystem sensor and optics assembly was operating at a temperature of −8.5°C as compared to the predicted values of −10°C.

To simplify the design requirements of the electronic amplification circuitry, the dc outputs of the sensor were converted to ac signals by a pair of photomodulators, each consisting of a pair of small neon lamps and a matched pair of cadmium selenide photocells. Extreme care had been taken in the mounting of the frail lamps. For proper operation, the lamps and photocells were housed in light-tight compartments. However, it was known that neon lamps fail to fire properly when pulsed in darkness, or after long periods in the non-conducting state in darkness, because the time required for gas ionization increases when the lamps are operating in the dark. This ionization time was reduced by the addition of a radioactive substance to the gas of the lamps. The apparent proper operation of the photomodulator as indicated by Y-axis output provides evidence that this method, along with its resultant advantages, can be utilized reliably for low-level signal modulation.

To conserve weight, volume and power consumption, microelectronic integrated networks were utilized in the searching, tracking and planet-in-view logic circuits. These networks were packaged in welded modules in order to take advantage of their small, flat, packaging dimensions. Since the various operations controlled by these logic circuits occurred as anticipated, they have amply demonstrated their suitability for utilization in circuits which must operate reliably in a space environment.

In addition to indicating the planet position, the Y-axis output from the sensor was also used to determine the amount of reflected solar energy received from the planet. The Y-axis output as the planet swept across the field of view of the sensor is shown in Fig. 43. The data indicated that the sensor received 20 to 30% more energy than expected. Therefore, the calculated energy level and the data used in the energy calculations must be modified for use in designing future systems.

Although all the functions designed into the subsystem were not exercised during planet encounter, the various operations performed in flight coupled with the data received during various ground and life-testing operations indicated that performance was at all times well within the design tolerances.

O. *Plasma Probe*

1. Description

The plasma probe was designed to measure and record the magnitudes of the positive-ion currents intercepted by each of the three sectors of its current-collecting electrode in each of 32 slightly-overlapping energy windows that cover the range from 45 to 9370 ev per unit charge.

A block diagram of the instrument is given in Fig. 44. The principal subsystems include:

1. The high-voltage subsystem modulator, transformer, multiplier, and voltage divider which supplies selected magnitudes of square wave ac to a grid so as to select the various energy windows.
2. The current measurement chain, including the three current-collecting sectors with their individual preamplifiers, of which any one or all three outputs can be gated into the summing amplifier and subsequent circuitry.
3. The calibration signal generator which produces two standard dc currents that are gated periodically into the three sector preamps.

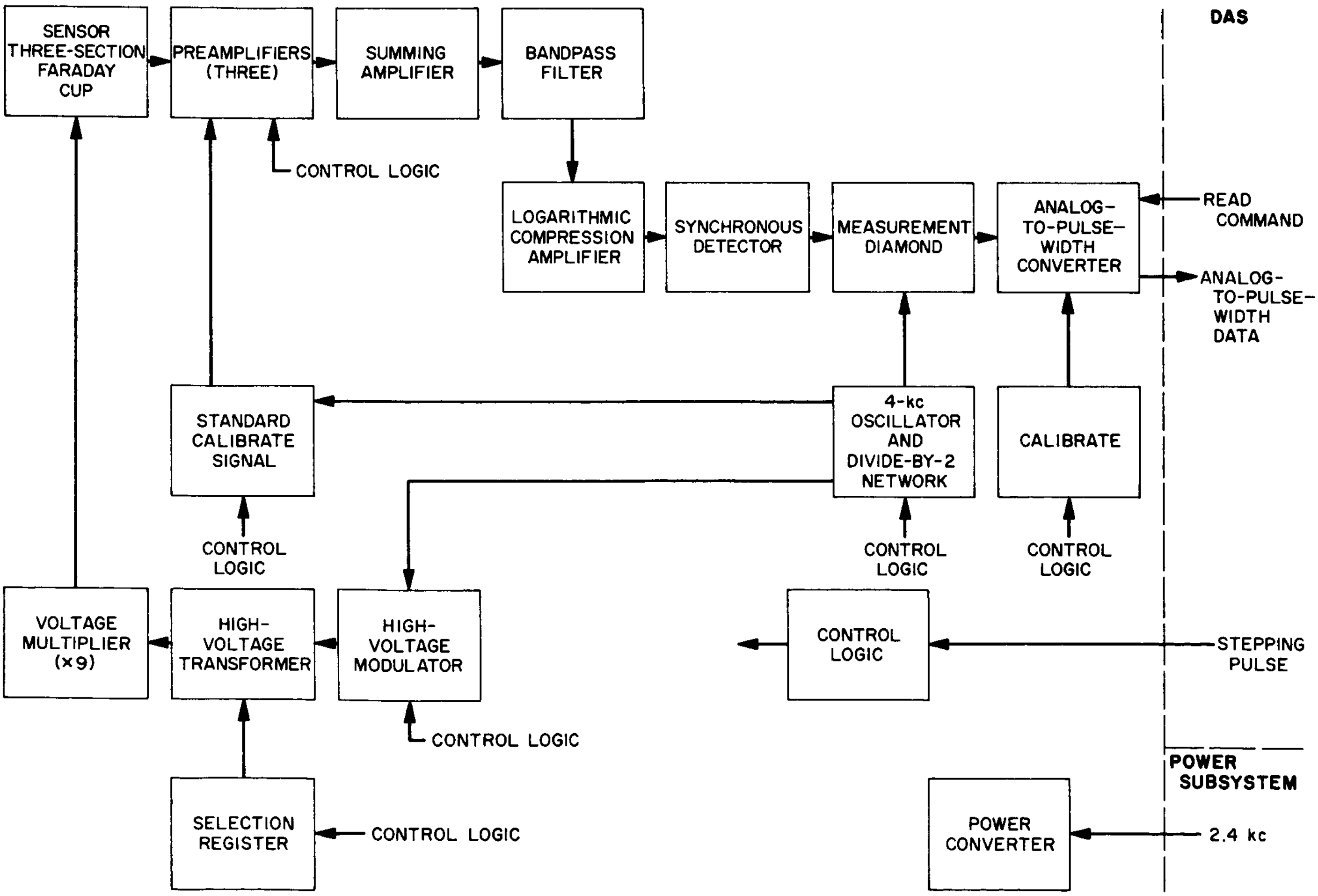


Fig. 44. Plasma probe block diagram

4. The digital control circuitry which steps the instrument through the proper sequence of measurements. The complete measurement sequence comprises 144 measurements, including:
   a. Each of 32 energy windows on each of three collector segments
   b. Each of 32 energy windows with all segments gated in
   c. Two current calibration measurements on each collector segment
   d. The modulation voltage on the 27th energy window
   e. The modulation voltage on the 28th energy window
   f. Two temperature measurements
   g. Six extra marker measurements.

2. Performance

The digital control section of the instrument apparently operated faultlessly throughout the flight. It executed its 144-step cycle 18,783 times before telemetry was terminated. There were no instances of stepping errors that did not have demonstrable causes such as spacecraft commands or (during one 24-hr period) high-voltage arcs.

The calibration circuitry and the current measurement chain appear to have operated in an entirely normal manner and in agreement with prelaunch expectations. This is attested by the great stability of the high-current (3×10^{-9} amp) calibration readings, which were repeatable within one or two numbers for long periods of time, varying with changes in temperature in approximately the predicted manner. The low-current (3×10^{-11} amp) calibration readings exhibited essentially the same behavior, although the readings deviated more from the mean values because of proximity to the noise threshold of the measurement chain.

The high-voltage modulation circuitry (exclusive of the voltage divider) did not exhibit any detectable anomalies, and it is probable that it operated essentially as intended. There is insufficient evidence for a positive evaluation of its performance, however, for two reasons:

1. During the first 8 days of the mission, the two voltage-calibration measurements were stable and varied predictably with temperature, but as only two of the 32 voltage levels were sampled, one can only infer that the values of the other 30 were approximately normal from the fact that the solar-wind data obtained appears normal.
2. Failure of the high-voltage resistor in the voltage divider eliminated the voltage calibrations during the rest of the mission and so garbled the data that even a fairly major drift of the high-voltage output might be difficult to detect from the data. No evidence for such a drift has been seen, and it is clear that (excepting the resistor) no catastrophic failure occurred in any portion of the instrument.

The voltage divider consists of two resistors in series: R339, a 300-meg high-voltage resistor, and R342, a 499-kilohm ¼-w resistor. Its two functions in the circuit are:

1. To supply to the calibrate circuitry a low voltage that is proportional to the output of the voltage multiplier, and
2. To discharge the capacitance of the multiplier between current readings.

The very long time constant for discharge of the voltage multiplier after the large increase in resistance pushed most of the readings up to an energy above that of the solar wind and introduced considerable uncertainty into the energy of all the readings.

From the data gathered between launch and resistor failure November 28, 1964, and December 5, 1964, significant information has been extracted. Velocity, density, and flux have been calculated for most of the 1400 plasma spectra obtained, and calculations of temperature and flow direction are in progress.

For the rest of the high-rate data, December 6, 1964, to January 3, 1965, the situation was not encouraging. The discharge time constant was such a large fraction of the duration of the measurement cycle that very few measurements showed any current at all except for one day in mid-December when the solar-wind velocity was exceptionally high.

During January, February, and March the data indicated that, after the switch to low data rate, the discharge time constant was short enough relative to the interval between readings that most of the data from the third sector sampled were taken at or near the proper energy. It was therefore easily possible to reconstruct enough of the plasma spectrum to derive fairly reliable estimates of the solar-wind velocity.

Because the back resistance of the high-voltage diodes in the multiplier was extremely temperature-sensitive, the discharge time constant increased steadily with time as the spacecraft cooled; sometime in March or April it had become so long that no simple scheme of data analysis was valid any more, and virtually no information has been gleaned from this later data.

Although more careful investigation of the properties of high-voltage resistors following the failure revealed that they were not a good choice for the application in which they were used, there is obviously a considerable element of luck involved. This fact was demonstrated by the two post-launch life tests. The MC-4 plasma electronics package was operated in vacuum at JPL for approximately 5050 hr from December 27 to July 22. The MC-0 electronics package was operated in vacuum at MIT for 4690 hr ending on September 6. No change in the properties of the resistor was apparent in either test.

3. Recommendations

Apart from the resistor failure, for which the remedy is obvious, the limitations of the plasma probe were already clear before launch, but the schedule did not permit further improvements. The limited sensitivity of the probe is a major problem on which considerable effort has been expended both at Massachusetts Institute of Technology (MIT) and at JPL for future missions. The noise level in the *Mariner IV* instrument makes current readings below about 10^{-11} amp unreliable. Improved circuit design and more effective shielding between the modulating grid and the collector can reduce the noise level by a factor of five and hopefully even ten. The uncertainty of the magnitude of the modulating voltage on the various steps is the second major limitation; it can be overcome in a straight-forward manner by incorporating a more complex sequence of voltage calibrations, as is being done in later versions of the probe.

P. Magnetometer

1. Description

The helium-vapor magnetometer uses the principle that absorption of light by a cell containing helium gas is a function of the angle between the ambient magnetic field and the optic axis. The instrument was mechanized with calibration circuitry which, by command from the DAS, superimposed magnetic steps of -80, -40, $+40$, and $+80$ gamma at the magnetometer sensor. This sequence of steps allowed calibration of the combined instrument output through the DAS for periodic checks of both linearity and scale factor. The cycle was initiated at the beginning of each DAS sequence. The in-flight calibration data indicated that both the instrument linearity and scale factor remained stable throughout the mission, and were identical to prelaunch calibration of the instrument.

a. Sensor. The resultant light, after passage through the helium absorption cell, is monitored for absorption as a function of time by an infrared (IR) detector. The detector output represents a measure of the field before refinement. Surrounding the sensor is a tri-axial array of Helmholtz coils which creates a sweeping magnetic vector for comparison with the IR detector output.

b. Oscillator and phase shifter. The oscillator and phase shifter produces two sinusoidal currents with a phase angle of 90 deg between them. One current is sent to the Z-axis Helmholtz coil and the other is commutated between the X and Y axes of the coil system, thereby creating a rotating field in either the X-Z or Y-Z sweep planes.

c. Commutator. The commutator switches the X and Y components of the sweep field on alternate cycles and synchronously switches the X and Y servo amplifiers.

d. RF power supply. The RF power supply is used to maintain a flow discharge in the helium lamp to supply resonant light and in the absorption cell, Fig. 45, to maintain helium atoms in the metastable state.

e. Servo amplifiers. Each servo amplifier: 1) converts the first harmonic component of the IR detector output that is in phase with the corresponding component of the sweep vector to a dc current and 2) nulls one component of the external magnetic field by applying the dc current to a Helmholtz coil.

f. Magnetometer outputs. The analog output of the magnetometer is proportional to the nulling current for each axis and is taken directly from the servo loop. The analog signal is fed to analog-to-pulse-width converters, and these converters are the interface with the DAS.

2. Performance

The performance of the magnetometer during all phases of the mission was nominal; no anomalies were observed in its performance. During the period before Canopus acquisition, the magnetometer was successfully used to assist in the determination of the spacecraft

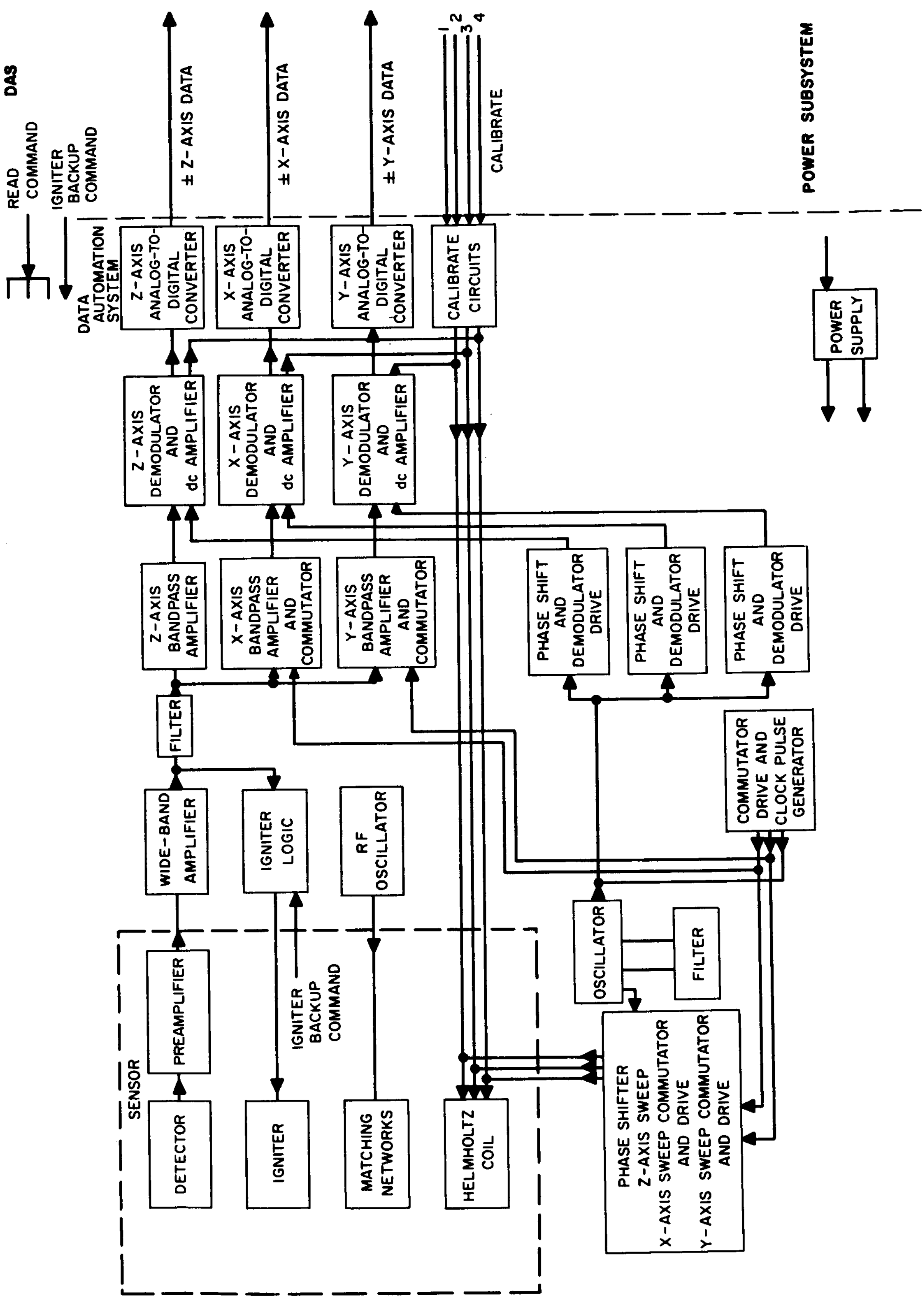


Fig. 45. Magnetometer block diagram

orientation, and during this roll period the X and Y components of the spacecraft magnetic field were determined.

The magnetometer was useful in establishing the time that the gyros came on or went off, because it indicated a shift of several DN whenever the gyros came on.

During periods of low interplanetary magnetic field activity, there were periods in excess of half an hour when there were no changes in the magnetometer output, indicating that the noise level of the magnetometer was well below the 0.7 γ quantization limit of the DAS. This low-noise behavior has been the best indicator of good magnetometer performance and minimal degradation during the mission.

The instrument calibration remained steady during the entire mission, indicating that there was no shift in output scale factor or linearity from prelaunch calibration. The inflight calibration steps were of sufficient amplitude to exercise all six analog-to-pulse-width converters.

The performance of the instrument during and after encounter was nominal; therefore, the results of the planetary measurements can be accepted with confidence.

3. Recommendations

At the present time there is an advanced-development program being conducted on the helium magnetometer at JPL. By and large, the results of this program indicate that the *Mariner IV* magnetometer sensor design was nearly optimum. The areas for improvement are primarily the optimization of the electronic design to reduce weight and power and to increase noise performance. Some improvements in mechanization and qualification testing have also been suggested by life-test results.

Q. Television Subsystem

1. Description

The television subsystem consists of a single camera employing a narrow-angle telescope. The camera is a shuttered system utilizing a slow-scan vidicon capable of storing an image with negligible degradation for the 24-sec frame time. The subsystem will take and encode two-color television pictures upon receipt of frame and line start signals from the DAS.

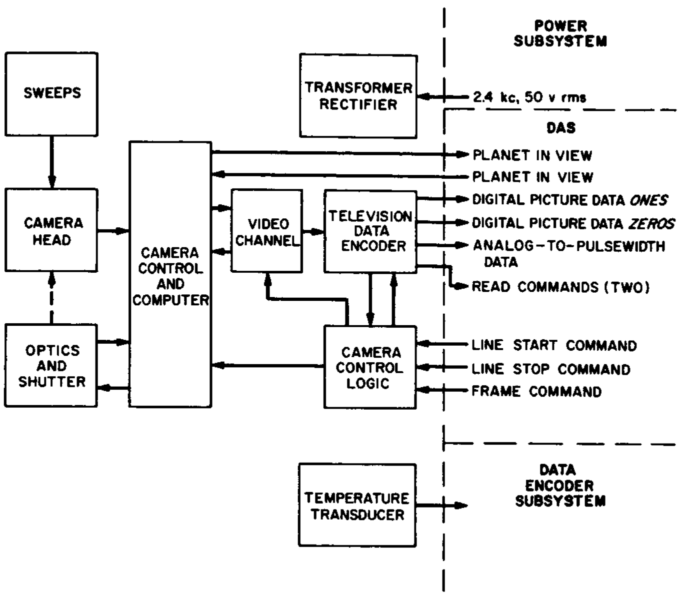


Fig. 46. Television subsystem functional block diagram

A functional block diagram is shown in Fig. 46. A more detailed block diagram showing circuitry blocks is shown in Fig. 47. The subsystem is divided into eight functional parts; a description of each follows.

a. Optics and Shutter. This part is the mechanical assembly of the telescope and the combination shutter-filter. The telescope is an $f/8$ Cassegrain with a 12-in. effective focal length and a 1.05 by 1.05 deg field of view. The shutter contains both shutter and filter functions. This is accomplished by the use of a disk containing four cutouts; two contain red filters and the other two, green. These filters are introduced into the light path by a rotating solenoid and remain until a second current pulse energizes the solenoid. Eight such pulses cause one complete rotation of the shutter filter wheel.

b. Camera Head. The camera head contains the vidicon tube, preamplifier circuitry, 110-kc oscillator, and filtering and distribution circuitry. The vidicon tube is a photosensor and provides the electrical video signal. The 110-kc oscillator provides vidicon beam modulation which serves as a carrier for amplitude-modulated video information. The video signal is amplified by the tuned preamplifier. The filtering and distribution circuitry provides filtering and distribution of the voltages required for operation of the vidicon tube.

c. Sweeps. This part consists of the horizontal and vertical sweep circuitry and the 4-kc oscillator. The

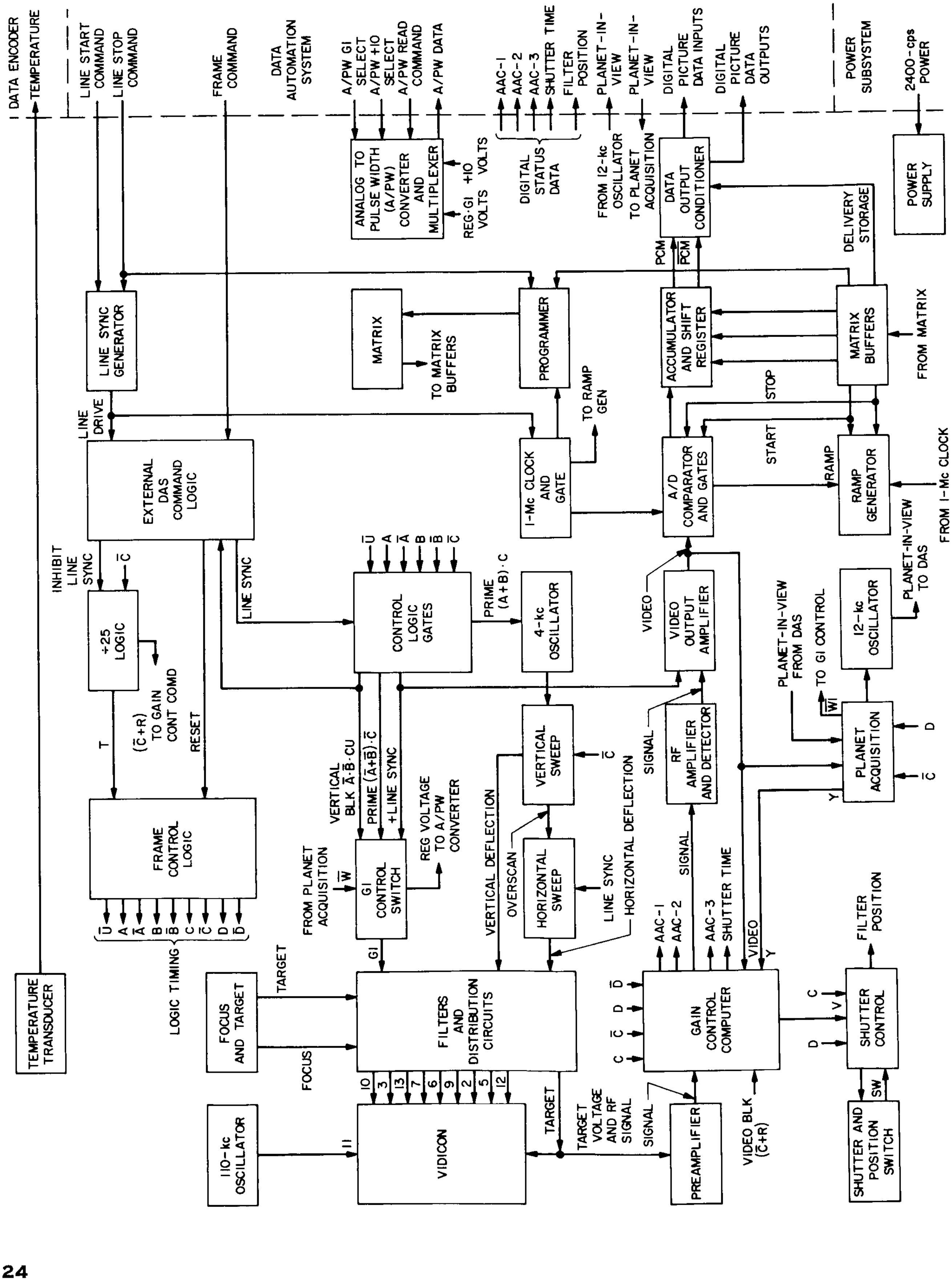


Fig. 47. Television subsystem circuit block diagram

sweep circuitry provides the voltages necessary for horizontal and vertical deflection of the vidicon beam both during picture readout and when the vidicon target is being erased and primed for the next picture. The sweep voltages are adjusted to scan a 0.22 by 0.22-in. sq area of the vidicon target. The 4-kc oscillator is gated on during the interval between pictures and causes the vertical sweep voltage to vary at that rate. This provides erasure of the image on the vidicon target.

d. Camera control and computer. The gain control computer circuitry, shutter control circuitry, G1 control circuitry and planet acquisition circuitry are contained in the camera control and computer. This circuitry performs simple computer operations and provides control of certain functions that affect the subsystem operation and dynamic range. The video signal level is sampled and video gain adjusted accordingly. The shutter exposure time is decreased if the video signal exceeds a preset level. The G1 control circuit regulates the amount of vidicon beam current for readout, prime, and erase of the vidicon target. The planet acquisition circuit provides a planet-in-view signal to the DAS when the video signal exceeds a preset level.

e. Camera control logic. The camera control logic consists of the external DAS command logic, divide-by-25 counter logic, frame control logic, and control logic gages circuitry. These circuits provide logic control of all camera timing functions such as shutter, picture readout, target erase and prime, and various switching functions necessary to the above. The control logic operates synchronously from external frame and line timing information received from the DAS.

f. Video channel. The video channel provides amplification, demodulation and dc clamping for the video signal. This part consists of the RF amplifier and detector and the video output amplifier. A feedback channel is provided so that video gain control can be exercised. The output from the video output amplifier is directed to the television data encoder.

g. ADC. The television ADC provides A/D conversion of the television picture data to 6 bits per picture-element pulse-code-modulation (PCM) conversion. These data are transferred to the DAS at an 83,333 bps rate. The data output conditioner, accumulator and shift register, analog-to-digital comparator and gates, 1 Mc clock and gate, ramp generator, matrix, matrix buffers, programmer and OSE buffers circuitry perform this encoding function. In addition, A/PW conversions are performed on certain critical television performance data and are coupled to the DAS by the A/PW converter circuit. The line sync generator circuit reconstructs line synchronization information and couples it to the camera control logic.

h. Power supply. The power supply transforms the 2400-cps square wave power into the voltages, with appropriate regulation, necessary to operate the television subsystem.

The television subsystem provides several outputs:

1. PCM 83.3-kc data to the DAS; this is the digitized video data.
2. Planet-in-view signal to the DAS; this signal indicates the subsystem has received sufficient illumination from a source or has received a planet-in-view command from the DAS.
3. Multiplexed A/PW digital pulse output; this is an analog-to-digital pulse width conversion for two critical television parameters, signal voltage level and G1 readout voltage.
4. Digital status data; five digital bits are parallel dc coupled to indicate the status of the video gain control circuitry, color filter position, and shutter exposure time.

2. Performance

a. Science cover deployment exercise. The television instrument was turned on and operated three times during the mission: during the science cover drop, February 11, 1965; during the encounter, July 14; and during the television haze calibration exercise performed on August 30, 1965.

The command to deploy the science cover (DC-25) also turned on the television subsystem. When the switch to Mode 3 data was made, real time telemetry data on the television subsystem was received. Reduction of the data from 60 frames in Mode 3 provided the results shown in Table 26.

Word 41 data indicated proper alternation of the filter position and also indicated the shutter was operating properly. The data indicated the shutter exposure time was 200 msec and also that no planet-in-view signal had been received nor generated. The gain control computer remained inhibited in the minimum gain state and the video level was less than 2 v. All these indications were normal and expected for the cover drop operation.

Table 26. Television subsystem performance data

Parameter	Prelaunch	Science cover deployment February 10, 1965	Encounter July 14, 1965
TV G1 voltage (DAS word 31), photometer mode, vdc	−43.4 ± 0.2	−43.4 ± 0.2	−43.4 ± 0.2
TV G1 voltage (DAS word 31), normal mode, vdc	−46.4 ± 0.2		−46.4 ± 0.2
TV +10 voltage (DAS word 39), vdc	9.57 ± 0.01	9.50 ± 0.01	9.51 ± 0.01

No anomalies were detected during the operation. The turn-off of the TV subsystem was timed to ensure the shutter was in a closed position.

b. Encounter. The TV subsystem was turned on by DC-25 on July 14 for the encounter operation. The subsystem operated for approximately 7¾ hr prior to the switch to data Mode 3. Reduction of the data received prior to NAA agreed with the February 10 data (Table 26). It is known also from tests performed prior to launch that the voltage measuring device varies an order of magnitude more than the voltage. The measuring device is also more temperature sensitive. Again, the Word 41 data were as expected. The shutter filter indication was proper as were the exposure time of 200 msec, the lack of a planet-in-view signal, less than 2 v of video indication, and the minimum gain position of the computer.

On Frame 41 (the DAS frame counter is reset concurrent with WAA) the Word 41 data indicated a television planet-in-view. All other television data remained the same. In testing and calibration prior to launch, there was always an over-2-v-of-video indication before the light intensity was high enough to cause a television planet-in-view. It is therefore considered that the television subsystem received the planet-in-view signal from the DAS with the Mars gate as the originator.

In the next Frame No. 42, the data indicated an over-2-v-of-video signal. In Frame No. 43, the voltage indication from Word 31 changed abruptly, which indicated that the television had switched out of the photometer mode to the normal mode (Table 26) and the inhibit was released from the gain control computer. The computer remained in the minimum gain state since the video signal was large enough. The video signal, however, was not large enough to cause the shutter exposure time to switch to the backup 80 msec time. The recording of the first picture was Frame No. 43.

The data remained the same with the filter color alternating, until Frame No. 59. In this frame, the video signal level indicated less than 2 v. In the next frame, the indicator was again over 2 v. Frame No. 59 indicated a green filter and the subsystem was designed such that the gain would change after a green picture with less than 2 v of video. The gain, however, did not change. A close check of the time of Word 41 sampling indicated it occurred during the time when the first six television lines of a frame are blanked out. This condition had occurred several times during systems tests. This was not an anomaly or a problem.

The gain remained at minimum until Frame No. 68, where it stepped up one level. In the previous frame, the video signal was less than 2 v. The gain switched again in Frame No. 70 and switched to maximum in Frame No. 72. The video signal indication remained less than 2 v until Frame No. 72 where it again indicated over 2 v. The spacecraft switched to Data Mode 2 before data could be received on Frame No. 73. All frame numbers refer to the DAS frame counter.

The pictures received demonstrated, along with the telemetry data, that the television subsystem operation was flawless. The optical and electronic focus, resolution, gray-scale rendition and sensitivity were the same as prior to launch.

c. Television haze calibration exercise. A dark encounter exercise was undertaken primarily in an attempt to learn more about the haze found in the pictures of Mars. Secondarily, it was hoped that some explanation of the periodic interference seen in the pictures taken through the red No. 2 filter might be found.

Although an initial attempt, terminated because of ground problems, moved the shutter disc, analysis of the timing indicated an 0.8 probability that the first red picture recorded would be filter No. 1 if the proper sequence of turn-on commands were used.

The predetermined procedure was followed and the encounter sequence initiated at 2030 GMT on August 30, 1965. The Mode 3 telemetry was carefully analyzed to determine whether the AAC-3 function (≤ 2 vpk video) tripped. Additionally, the frame start/end of tape (bit 408) and the television filter (bit 406) functions were monitored for a repeat of the anomalous events seen during the Mars encounter.

The results of the encounter were essentially negative. No video was detected, and the played-back pictures showed no excessive noise indicative of increased dark current in the vidicon. At the maximum gain state, the average video level DN was 61.5_{10} c.f. 59.0_{10} as the quietest "capped optics" picture taken during calibration. Space would appear to be quieter and darker than a test lab as no interference was found. All telemetry data were nominal, including bias voltages and temperature.

3. Recommendations

The television subsystem met all of the design goals and objectives placed on it. For future missions, if the same type of photographic subsystem were used, improvements that could be made are:

1. Obtain a vidicon with improved performance. The vidicon used was at the state-of-the-art. For a future mission, perhaps a vidicon with greater sensitivity, better erase characteristics, and less shading could be developed.
2. Use a system with higher gain to distinguish features in the darker portions of the planet. A better determination of the light characteristics of the planet can now be made from the results of *Mariner IV*.
3. For any future spacecraft system, more data storage capacity and a higher rate is desired. With these capabilities, a higher resolution photographic system could be used.

IV. SPAC ORGANIZATION AND FUNCTION

A. Operations Philosophy

1. SPAC Responsibility and Function

The responsibility of the *Mariner* SPAC group was to monitor and analyze spacecraft performance and make recommendations for courses of action which would enhance the probability of achievement of the mission objectives. This group was one of three analysis groups within the SFO organization established to support the *Mariner* Mars 1964 Project. The other analysis groups were the SSAC group and the FPAC group. These groups were responsible for the analysis of science data and the analysis of the flight path (trajectory), respectively, and for making recommendations for ground commands to enhance mission success from their points of view. The directors of the analysis groups, in addition to the directors of the mission support groups, such as DSIF, the Data Processing System (DPS) and the Operational Communications System (OCS), reported to the SFOD.

In analyzing the spacecraft data the SPAC continually attempted to evaluate the actual performance against the anticipated performance based on the design. Extensive effort was applied to explain every unexpected condition observed. Frequently, the cause was an incomplete understanding of the spacecraft in a space environment.

Under guidelines established by the SSAC and the Project Office the SPAC also attempted to maximize the scientific data recovery. A primary goal throughout the mission was the ultimate attainment of television pictures of Mars and data from the occultation of the spacecraft by Mars. The responsibility of the SPAC group relative to the attainment of scientific data was to assure the acquisition of the planet-oriented data and then, wherever possible within engineering constraints, to assure the acquisition of the maximum amount of interplanetary data. In the *Mariner IV* mission the latter was easily accommodated.

The development of a mission operations plan was undertaken by the SPAC group to execute the responsibilities with which it was charged. SPAC planning efforts were instrumental in the development of the overall SFO sequence of events for the various mission phases.

2. Organization

The *Mariner* SPAC organization had as its nucleus the SPAC Division Representatives, the SPAC Director and the Assistant SPAC Director. SPAC Division representatives were members of the following Technical Divisions at JPL: Division 32, Space Science; Division 33, Telecommunications; Division 34, Guidance and Control; Division 35, Engineering Mechanics; and Division 38, Propulsion. The SPAC Director and Assistant Director represented Division 29, Project Engineering. In this manner, there was available in the SPAC organization knowledge of the spacecraft from each of the broad disciplines which contributed to the design. This nucleus formulated the SPAC organization and was responsible for the implementation of the group's goals. The goals can be summarized as follows: define facility, computer, and staffing requirements; adequately staff the various positions; effect training of the SPAC personnel through procedure generation and test participation; analyze flight data, and recommend actions to be taken during the mission.

Detailed knowledge of the major subsystems aboard the spacecraft was brought into the SPAC organization through subsystem representatives, usually the cognizant engineer or his alternate. Responsibilities of each representative included knowledge relative to the design, fabrication, test history, and capabilities of his subsystem in flight. In addition, the subsystem representative was responsible for the achievement of the detailed analysis of his hardware in flight, analyzing real time telemetry, advising his division representative or the SPAC Director of his subsystem's performance, and providing information relative to the status of this subsystem during any discussion of action to be taken in the event of a non-standard spacecraft condition. With the subsystem representatives supporting the SPAC nucleus, the non-staff portion of the SPAC organization was essentially complete. This organization provided the SPAC Director with only a small group which he had to manage directly, and also provided the desirable feature of having the subsystem representatives be responsible to a member of their own division. The SPAC line organization is shown in Fig. 48.

During the actual flight operations, especially the long-duration critical phases, some modification was necessary in the organization of the *Mariner* SPAC. In the operations the SPAC personnel were organized in a functional manner rather than by individual as described above. During planning phases the Division Representative, for example, is an individual charged with the responsibilities of that position. No alternate may act in his behalf unless duly authorized by the Project Office, SPAC, and the Division in question. This is necessary because his duties involve commitments and recommendations which are binding both on the Project and on his Division. During an operation, however, two facts serve to void this necessity for individual responsibility:

1. It is not possible to require individual responsibility and extended full time support simultaneously from the same person.
2. All commitments and recommendations which require participation by this individual will have been previously made; the responsibility for those occurring during the operation rests with the SPAC Director, the SFOD, and the Project Office.

As a result the operational organization of the *Mariner* SPAC permits the position of Division Representative to become a functional responsibility, so that any qualified person may serve in that capacity.

Early in the mission it was attempted to treat the position of SPAC Director in the same manner. The SPAC Director, however, is still required to be individually accountable during a flight operation, since he is the only spacecraft representative active in flight operations who is recognized directly by the SFOD and the Project Office. Thus, while an alternate could be appointed, neither full responsibility nor authority could be transferred. In order to reduce the burden upon the SPAC Director, his duties were catalogued as to individual or functional responsibility. Those which were individual in nature were reserved to the SPAC Director. The functional responsibilities were assigned to the Assistant SPAC Director position which could be manned by any of the qualified members of the SPAC Director's staff. Figure 49 represents the flight operations organization for SPAC.

Another modification to the organizational structure of the SPAC during flight operations was the elimination of the line structure requirement that all interfacing between the SPAC Direction Team and the individual subsystem representatives be carried out via the appropriate division representative. Prior to launch or during the routine transaction of SPAC business the direct insertion of the division representative is necessary in order to guarantee proper coordination of the division's SPAC effort. During an operation, however, it places on the operation the unreasonable burden of maintaining an additional level in the organization. To

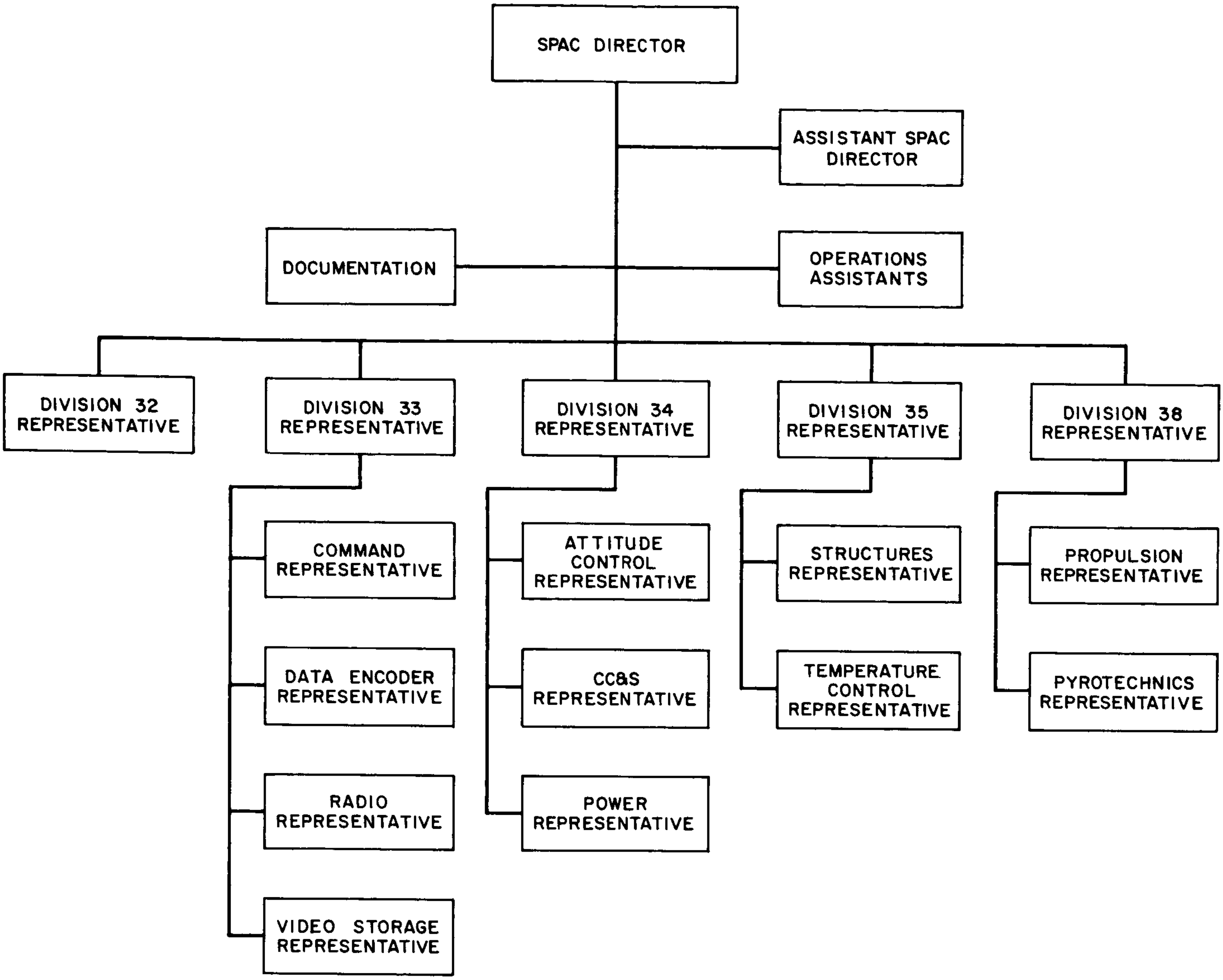


Fig. 48. SPAC line organization

avoid this the subsystem representatives were encouraged to report directly to the Assistant SPAC Director, with all reports or recommendations subject to the division representative's concurrence.

One staff function was performed for the SPAC Director by individuals on loan from the organization of the Space Flight Operation Director. These individuals, called operations assistants, aided in the integration of the mission-oriented SPAC group into the basically mission-independent Spacecraft Performance and Analysis Area. They also aided in verification and checkout of computer programs to be used by SPAC. Subsequently they provided the SPAC interface with the facility and the computers during the critical (high activity) phases of the mission. During the cruise phase the operations assistants reverted back to the Space Flight Operation Directors organization. Their function then was to serve as mission operations controllers during the 24 hr/day tracking and data acquisition activities of the cruise phase. During any of the critical phases of the mission, including encounter, the operations assistants were utilized to augment the activities required of the Space Flight Operation organization.

Additional line support was available to the SPAC Division Representatives through the specialized support from individuals within their divisions who were uniquely aware of various assemblies of the spacecraft or had a special analytical ability. During the critical phases of

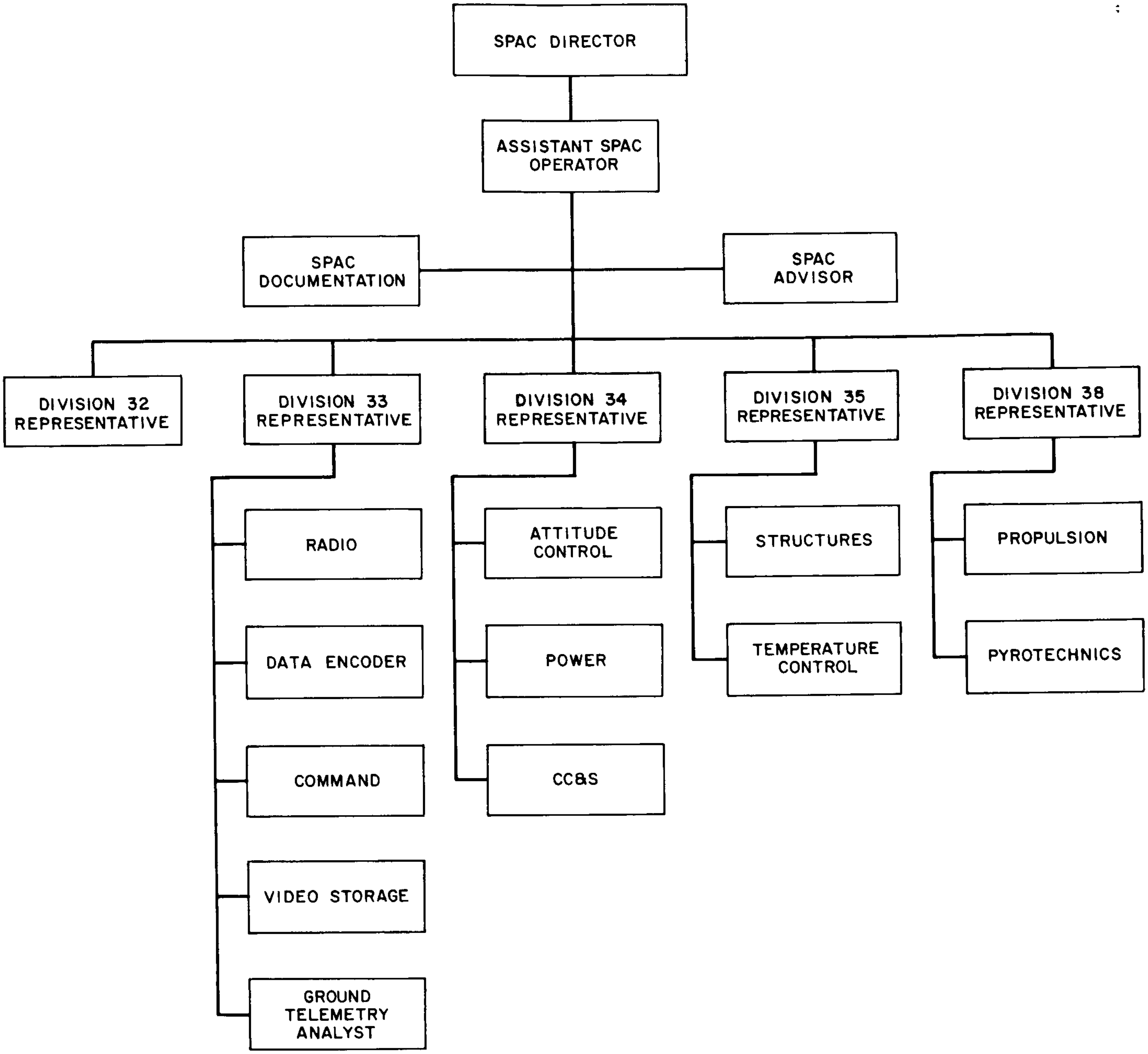


Fig. 49. SPAC flight organization

the mission these individuals were not housed in the Spacecraft Performance Analysis Area, (SPAA) or the *Mariner* Mission Support Area (MMSA), but did have access to the data and could contact their division representative, as required. They could be summoned into the operations area for consultation as the need arose.

The importance of a person on the *Mariner* SPAC group solely responsible for documentation was not recognized until after the launch. The Spacecraft Systems Manager requested of the SPAC, subsequent to the launch, periodic on-lab formal reports of spacecraft performance. In order to achieve this objective, it was necessary for the *Mariner* SPAC to acquire a documentation engineer. With his assistance the SPAC Director was able to publish formal reports covering spacecraft performance for each 6-wk period throughout the mission. He was able also to publish informal weekly reports of the spacecraft performance. Later in the mission this individual was also responsible for helping generate SPAC Operational Procedures. The documentation to support the Encounter Preparation Working Group and the evolution of the final encounter sequence plan was generated with his help.

3. Operations Philosophy

The operations philosophy of the *Mariner* SPAC group was to maintain 24 hr/day coverage by the majority of the SPAC representatives during the critical phases of the mission, such as launch, Canopus acquisition, midcourse, encounter and any additional situations requiring ground command action. The basis for this philosophy was the recognition that: 1) any imminent change of state of the spacecraft should be preceded by an interval of extensive data examination prior to commitment to such an action, and 2) any recent change of the spacecraft should be monitored to verify that all expected results are observed and to ensure against some unexpected effect going unobserved.

In the early phases of the mission (prior to the trajectory-correction maneuver) the intervals between critical phases were manned with only division representatives, or equivalent, support in addition to a member of the SPAC Director's staff. Subsequent to the midcourse maneuver, the coverage by the SPAC was reduced to daily reviews of the data, followed by daily meetings of the SPAC Director and the Division Representatives.

Continuous monitoring of the data was accomplished by an alarm program in the IBM 7044 Computer under the supervision of the SFO Operations Controllers (previously SPAC Operations Assistants). Any violations of alarm limits previously read into the computer would cause the Operations Controller to be alerted. He would, in turn, notify the SPAC Director and the appropriate Division or subsystem representatives who would take whatever action was warranted as a result of the alarm limits being violated.

In situations where an unexpected condition existed, there frequently was some action that could be taken on the ground to provide deeper insight into the condition of the spacecraft, or there existed an obvious need for the transmission of a ground command. More frequently, however, the unexpected condition was the result only of bad data, or perhaps of some unknown condition. One aspect of the SPAC function was to ascertain just what the cause of any unexpected condition was before any overt action was taken. The philosophy which pertained was that no intentional perturbation to the spacecraft would be initiated on the ground unless such action was clearly warranted and justified.

In previous missions this had been interpreted as precluding the transmission of any command unless corrective action for a particular problem was required. The nature of the *Mariner IV* mission, however, was such that an extension of the basic philosophy was evolved during the mission. Ground commands were utilized to preempt automatic features aboard the spacecraft and to exercise a back-up to an automatic function before the need for such a command was demonstrated. The necessity for this change in philosophy resulted from the intricacies of the spacecraft logic and from recognition of the 12-min one-way transmission time of signals between the spacecraft and the Earth at encounter. Before a ground command to correct an unexpected condition aboard the spacecraft, at encounter could reach the spacecraft, that condition which required the corrective command was at least 24 min old. If that amount of time was consumed before a back-up was effective, the utility of the back-up might be so marginal as to be useless.

4. SPAC Interfaces

To successfully complete its task the SPAC group had to have numerous interactions with other elements of the SFO organization. This section describes those interfaces in order to provide an understanding of the broad scope of efforts involved in a spacecraft performance analysis activity.

a. Facility. One of the major interfaces was the interface with the SFOF. The SFOF housed all the SFO activity at JPL during the *Mariner* mission. One segment of the SFOF was the SPAA which housed the SPAC for the first 2 mo of the mission. To properly utilize the SPAA it was necessary for SPAC to define its equipment needs, including telephones, desks, data display devices and communication net requirements. Subsequent to defining the equipment needs it was necessary to derive a layout for the various pieces of equipment. The requirement in this regard was to assure that people needing access to various pieces of equipment could have such conveniently. Since the SFOF and the SPAA were basically mission independent, it was necessary for the *Mariner* SPAC to consider the requirements of other projects also using the SPAA in deriving its final room arrangement. Significant negotiation took place prior to the definition of a room layout which was easily adaptable to both the *Ranger* and the *Mariner* missions. The final room arrangement is shown in Fig. 50.

Because of the desire of the *Mariner* Project Office to provide a working place for *Mariner* personnel during the cruise phase of the mission, it was decided to implement the MMSA. Such an area would provide an interference-free location where *Mariner* personnel could work, meet, and eventually test and execute the encounter sequence when the spacecraft passed the planet in July 1965. The MMSA was completed and occupied by the *Mariner* Project in mid-January 1965.

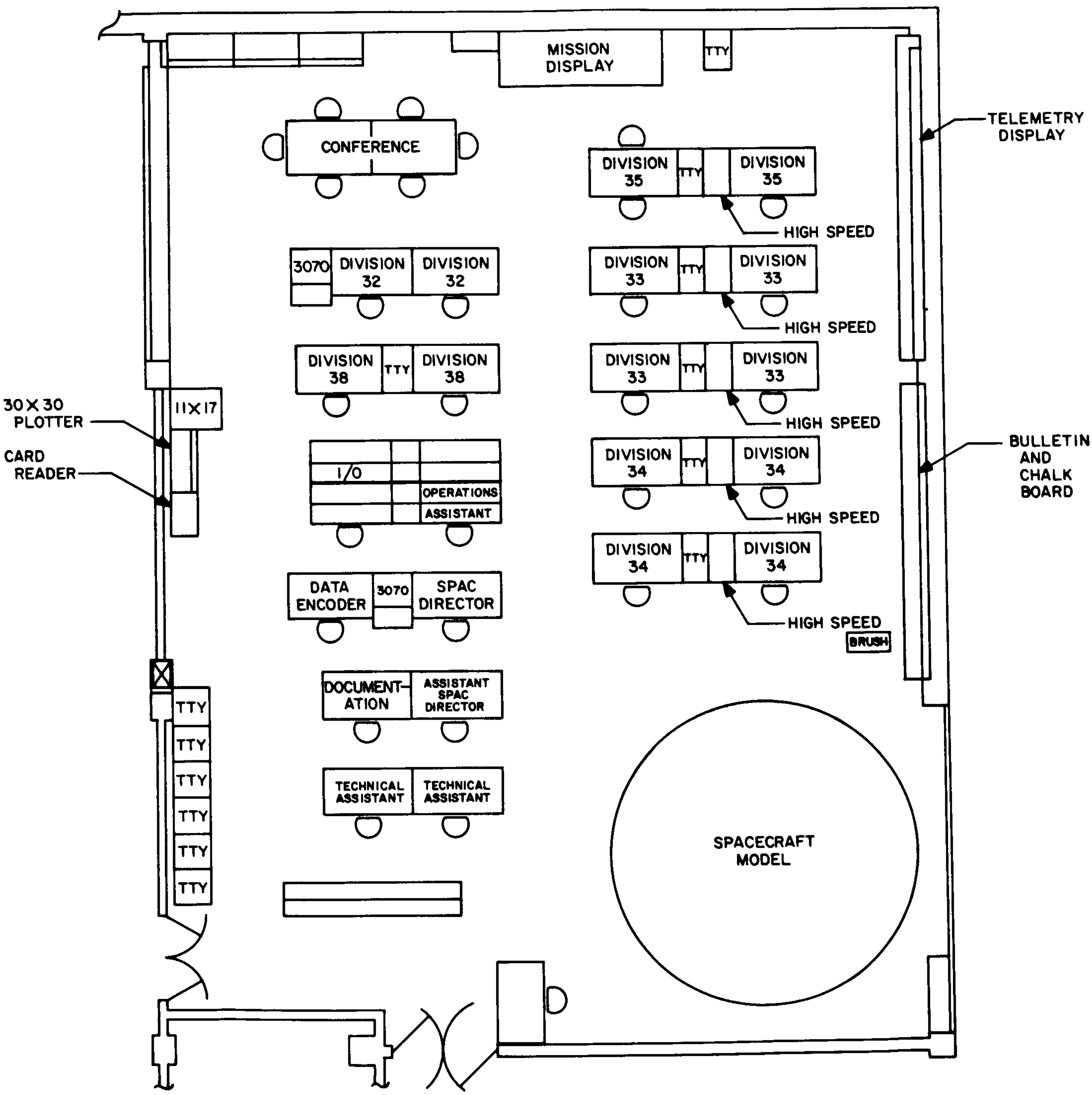


Fig. 50. SPAA configuration

The room arrangement of the MMSA is shown in Fig. 51. This room housed activities which were previously distributed among various areas in the SFOF: SSAC, SPAC, FPAC, DSIF and the SFOD.

b. Data processing system. To accomplish the SPAC responsibility of data analysis, it was necessary to interact with the SFOF Data Processing System (DPS). The SPAC group received output from the DPS as a result of real time and nonreal time data processing. Real time data processing involved decoding, identifying and time tagging data received from the DSIF tracking stations very shortly following its reception at the station. Real time data processing led to an immediate display of the data in desired formats in the user (SPAC) area. Nonreal time data processing involved special computer operations on data stored in the SFOF. In general this data was in excess of 30 min old. During the majority of the mission nonreal time processing was done on a daily basis using the data received during the past 24 hr.

The real time computer processing programs were utilized continuously throughout the mission. All data received in real time from the DSIF stations was processed by the real time editing program and operated on by the alarm program. In situations where the real time programs could not be run due to computer conflicts or computer breakdown, the data from the DSIF stations was interpretable from the output of a teletype machine attached directly to the incoming teletype line. The value of the real time programs was proved to the satisfaction of those reviewing the teletype machine printout.

The SFOF and DPS interfaces were preliminary to the major SPAC task of mission operations, which involved a general interface with the SFOD and the Project Office. In order to succeed in SPAC's responsibility for reviewing the data and suggesting any corrective action that was necessary, extensive effort had to be applied in the areas of test planning, development of operational procedures, and determination of a sequence of events. The goal of the mission operations phase is to have a set of agreed-upon operational procedures and also a sequence of events which would indicate to those concerned what activities were to be followed in the execution of that particular flight sequence. To achieve the twin goals of an operational procedure and a sequence of events, a test plan had to be developed. The test plan utilized by the SPAC was directed toward gradually educating the SPAC personnel to the capabilities of the facility and computer system, then conducting a series of tests in which their knowledge of the spacecraft and its operational modes was tested.

Initially, the tests that were run were conducted using simulated data derived from a computer program based upon inputs from the various technical areas as to what their subsystem telemetry should look like as a function of time within a flight sequence. This type of testing was primarily effective during the initial portions of the testing program when the primary activity consisted of checking out the facility and the computers. The complexity of the *Mariner* Mars 1964 spacecraft, however, made it very difficult to adequately simulate the data to show reactions to spacecraft commands and changes of state due to other sources. Beginning in October, 1964, in order to provide SPAC with more realistic data, the proof test model (PTM) spacecraft which had been previously utilized in the spacecraft test and assembly program was used as the data source. It was now possible to show immediate responses to spacecraft commands and to more adequately test the SPAC group through the use of the capability of simulating failure modes. The operation of the PTM was controlled by the PTM test team in the Spacecraft Assembly Facility. Data from this spacecraft was brought into the SFOF via hard-line.

c. Deep Space Instrumentation Facility. The necessary interaction between the DSIF and the SPAC was handled by the SPAC Director and the Telecommunications Division representative, or their alternates. During critical operations the DSIF Project Engineer was accessible to SPAC only through the SFOD. Since all telemetry was received by the DSIF stations and all commands are transmitted by the DSIF stations, it is natural that much information about the DSIF station at any particular point in the mission be of interest to the SPAC group. The primary interface activity involved the determination of a two-way tracking schedule in conjunction with the FPAC group, discussion regarding the use of the 100-kw transmitter at the Goldstone station, and determination of the procedures involved with sending commands and receiving spacecraft telemetry signals. Periodically it was necessary to request of the DSIF information such as station time, transmitter frequency, ground-received signal level or a readout of some telemetry channel directly from the telemetry processing equipment at the station. When problems occurred in the communication lines bringing telemetry data into the SFOF, it was necessary to request the DSIF to monitor various telemetry channels and report by voice when and if these channels changed in value. This

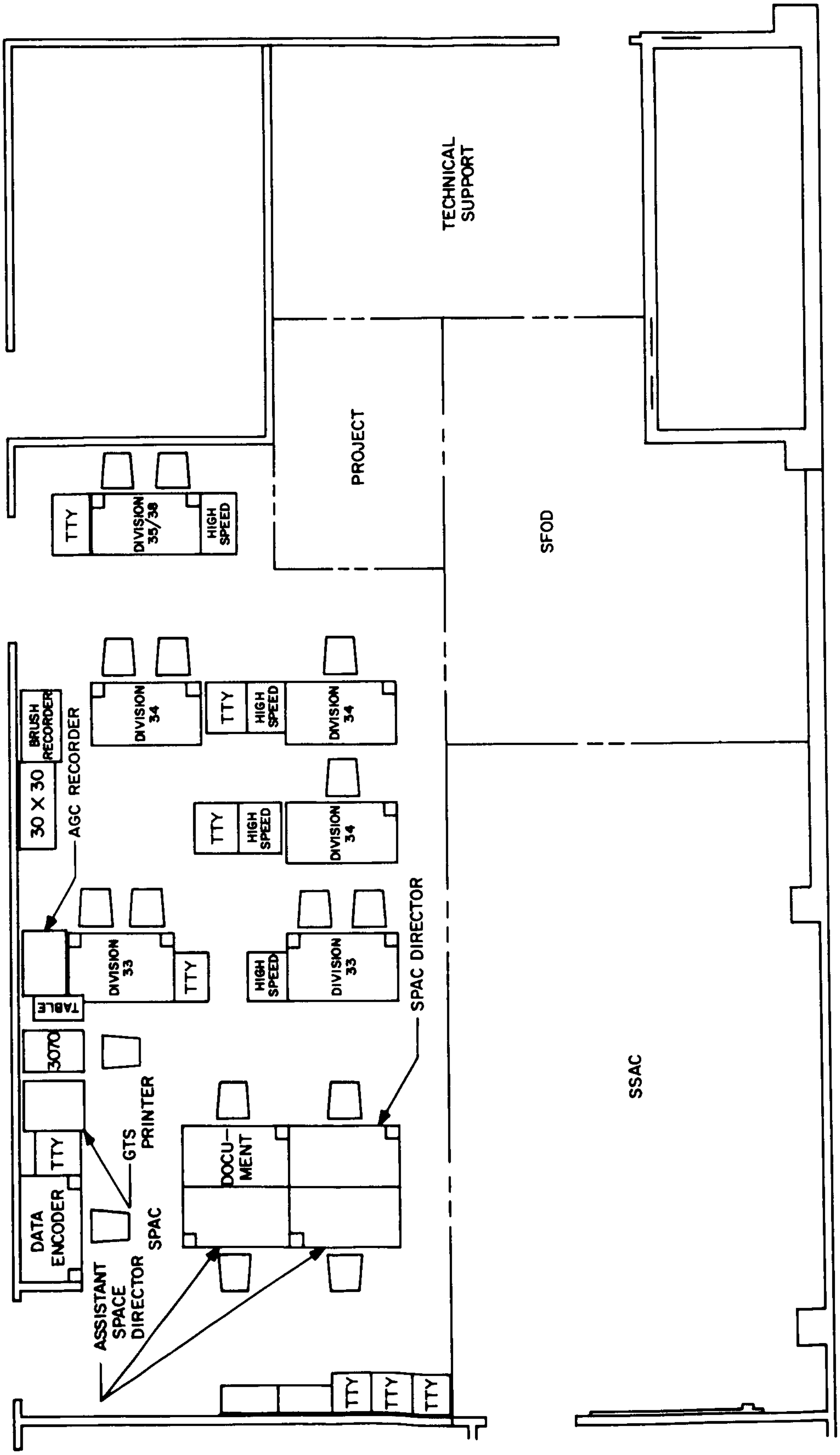


Fig. 51. MMSA configuration

permitted rapid action on the part of personnel in Pasadena, even if they were unable to view the telemetry that was being received at the station at that particular moment.

An example of the special functions which the DSIF was requested to perform was a monitoring of the analog recorders at the stations to fix the time when frequency transients were noted on the downlink signal. One type of frequency transient occurred when the spacecraft cyclic command occurred while the spacecraft receiver was in a one-way lock condition. The SPAC requirement for precise timing of the transients was based on a need for information about the spacecraft CC&S clock frequency.

The DSIF received-signal level was displayed during the late portions of the mission in the *Mariner* Mission Support Area on an analog recorder. This requirement was established after detailed investigations into the implementation of the spacecraft roll inertial control mode indicated that the high gain antenna pointing error was the only measurable roll attitude indicator. This capability to monitor received-signal level permitted the telecommunication analyst in the SPAC to gain some knowledge as to the signal level changes as a function of time.

d. Analysis areas. The interfaces with the other two analysis areas, SSAC and FPAC, were very important to the SPAC in the successful accomplishment of its responsibilities. The SSAC group was able to assist the SPAC in the interpretation of the scientific mission objectives so that the proper determination of the mission plan could be effected. SPAC felt a responsibility to implement the desired goals of the scientific community whenever such could be accomplished without detrimental effect upon the final accomplishment of a successful encounter sequence.

The evaluation of the science data by the science cognizant engineers and scientists provided the SPAC group with an indication of the environment which the spacecraft was observing, in addition to providing some useful inputs to the SPAC relative to the occurrence of various events. Some of the anomalous conditions observed on the *Mariner IV* were evidenced in both the science and engineering telemetry. Correlation of the times of these occurrences in both telemetry types permitted more concrete determination of the cause of such anomalous conditions. The interface between the SPAC and SSAC was effected by the SPAC Director and the SPAC Science representative, each of which had contact with the SSAC Director. Additionally, the SPAC Science representative had direct communication with the science subsystem cognizant engineers who were located in the SSAC area. After the MMSA became operational, direct communications of an informal nature between individual SPAC and SSAC personnel were possible.

The FPAC group provided the SPAC with trajectory information including the geometry of the spacecraft relative to the Sun and Earth and other celestial objects at such times as midcourse and encounter and also the cruise phase. This information permitted detailed and knowledgeable interpretation of the performance of the spacecraft as concerned both the telemetry and the radio signals observed on the Earth. An important function which was satisfied by the FPAC group was the prediction of the frequency of the radio-received signal at the Earth and the proper frequency for transmission to the spacecraft. This was necessary due to the continually changing doppler effect on the radio signal. The availability of such information to SPAC permitted better analysis of the radio signals as well as the rapid lockup of both the uplink and downlink signals.

e. Encounter preparation working group. The Encounter Preparation Working Group (EPWG) greatly influenced the operations within the SPAC group for the latter portions of the mission. The EPWG was one of several groups established by the *Mariner* Project Office in January 1965 to support Space Flight Operations in its preparations for the remainder of the *Mariner* Mars 1964 mission. Basic responsibilities of this group were to augment the SFO organization by studying the potential encounter phase in July at great length and making recommendations to this organization and the *Mariner* Project Office as to what strategy should be employed to maximize the probability of mission success. The major outputs of the EPWG were the recommendations for a science cover deployment exercise during the month of February, and the generation of the encounter sequence plan that was utilized in the July encounter of Mars.

The chairman of the EPWG was the Spacecraft Project Engineer. His committee members were basically the Project Representatives from the various technical divisions and areas of responsibility.

In the preparation for encounter, hidden facets of the spacecraft design and operation were uncovered or rediscovered to a much greater extent than had been anticipated. The acquisition of detailed knowledge relative to the spacecraft design and predicted performance based on testing was very useful in the evolution of the encounter sequence plan. This encounter sequence plan was thoroughly tested prior to encounter with a series of tests conducted with the PTM spacecraft. An additional result of the EPWG effort was the generation of techniques and computer programs required to support the encounter phase; e.g., the program to predict the scan platform position as a function of time based upon real time telemetry input. Another computer program was generated which defined the true spacecraft time based upon the timing associated with real time telemetry at the DSIF stations. Precise timing of certain functions was mandatory for a successful encounter sequence and was established and verified through test and re-evaluation of the systems involved.

5. SPAC Data Output

Enormous quantities of data were generated for SPAC use during the course of the *Mariner IV* mission. The real time data processing was done entirely with the IBM 7044 Computer, and the nonreal time processing was done with the IBM 7094 Computer. The gross design of the data processing system in the SFOF calls for the output of the 7094 to be available in near real time. The *Mariner* SPAC did not utilize this capability, although at times, prior to the mission, it had been anticipated that such would be done. The lack of priority of this capability for *Mariner* precluded additional work to correct computer deficiencies in this area.

There were four basic sources for *Mariner* data throughout the mission. One source was the IBM 7044 Computer which provided three outputs utilized by the SPAC group. One output was from the Stromberg-Carlson 3070 Electrostatic Printer. Another output of the 7044 was an output of selected telemetry measurements on teletype machines modified to operate at a rate of 100 words/min. These devices have been called teleprinters. The third output of the 7044 was a plot of the telemetry on a Milgo 30- × 30-in. plotter. This device had the capability of printing and displaying only 12 selected engineering telemetry channels. The nature of the *Mariner* data was such that this number of channels appeared to be satisfactory.

A second source of data was the 7094 computer which again generated three outputs utilized by the SPAC group: 1) an editor program output called JPEDIT which listed all engineering measurements in a format which provided easy access to the measurements in question, 2) a data suppression program (SSDM) used to reduce printout of non-changing telemetry channels, and 3) a plotting program called EDPLOT. The SSDM output contained the conversion in its output format; this was due to the fact that JPEDIT was an editor output and the engineering unit conversion was made downstream of this particular portion of the 7094 computer processing program. With EDPLOT, selected telemetry channels could be plotted to desired time scales to provide for the SPAC members a graphical display of the telemetry measurements for a segment of time.

A third source of data to the SPAC group was the display of non-decommutated data on teletype machines in the area. This output was taken directly from the teletype lines. The printout of the teletype signals was made useable to SPAC by providing a special printing head on slightly modified printers. Data in this format were used when the computed outputs were not available, or when timing was critical and the delay within the computer to process the data and display it would be too long to permit operational decisions.

The fourth basic data source available to the SPAC was a Franklin printer located in the area. This output device is identical with that utilized on the System Test Complex during the spacecraft system testing. Special conversion equipment was provided to accept the teletype signal and convert it into a signal compatible with the input requirements of the ground telemetry processing equipment. The output of the Franklin printer became available only late in the mission and was primarily useful in the period of time when the recorded television picture data were being transmitted. The real time computer processing program had no convenient means of displaying the output of the television data. It was decided not to expend additional effort attempting to modify the 7044 computer programs to handle this data when the availability of the Franklin printer output was recognized.

Some other special data sources were employed within SPAC. The data conversion equipment used in conjunction with the ground telemetry equipment and the Franklin printers was developed and used by the Guidance and Control analysts to provide a rapid analog display of some important telemetry channels. This technique of displaying digital telemetry to satisfy unique requirements proved very worthwhile throughout the mission, but particularly in the early phases of the mission at the time of Canopus acquisition and the midcourse maneuver.

Each of the outputs available to the SPAC group was intended to fill a specific purpose. The 3070 printer output was utilized continuously throughout the mission by the SPAC Direction Team, and the mission operations controllers to assess the status of the spacecraft, and by the telemetry subsystem analysts to verify the state of their subsystem. These printers had the desirable capability of printing all telemetry channels. All SPAC personnel did not have ready access to a 3070 printer, however, because there were few available in each area. Subsystem data analysts used the output of the high-speed teleprinters, driven by the 7044 computer, as the prime data source throughout the mission. The outputs of these devices were specially formatted with only telemetry measurements of interest displayed to each analyst. The Milgo plotter output was utilized by the attitude control engineers in analyzing the performance of the spacecraft control system in its limit cycle operation.

The JPEDIT output was intended to provide a complete nonreal time record of the spacecraft engineering data in a reasonably compact form, affording each SPAC analyst a reference library of telemetry data. Problems with the 7094 editor program rendered JPEDIT less useful than hoped, especially during the early stages of the mission. SSDM was intended to supplement the JPEDIT, but in a compressed form, much like a nonreal time alarm record. The EDPLOT output was to provide a graphical equivalent to JPEDIT for selected measurements. The utility of both SSDM and EDPLOT were also reduced by the 7094 editor program deficiencies.

All data generated in the *Mariner IV* mission were stored by an organization of the SFOF called Operational Document Control. Possession of all this data on file provided the SPAC representatives with a means of obtaining and reviewing any segment of the mission desired.

An MDL of engineering telemetry has been generated which is a compilation of all the best data recorded at the DSIF stations. The MDL provides an accurate and reliable record of all mission data. The output of the MDL, as far as engineering telemetry is concerned, is in two forms; the first is an engineering lister which is essentially a decommutated listing of all engineering channels in their decimal value format, i.e., no engineering conversion. The EDPLOT program provides the second means of displaying the MDL information. These plots contain nearly all spacecraft generated data, providing a graphical means of reviewing long term data for single measurements.

B. Performance Evaluation

1. General

It should be noted at the outset that there is no way to measure directly the true worth of an SPAC group. The *Mariner* SPAC was chartered to monitor the spacecraft, to determine from the telemetry the state of the spacecraft, and to recommend to the SFOD and the Project Office any action necessary to effect the achievement of the mission objectives. Since the SPAC did perform these functions and since the mission objectives were achieved, then the SPAC activity must be considered successful. The charter, however, allows such a broad spectrum of action that mission success is a very gross indicator of SPAC performance. The spacecraft is designed to perform its mission with no support from SPAC. If it does, then that SPAC group is more successful than one which heavily supported a spacecraft that failed. Obviously if the failure can be traced to an act of either commission or omission by the SPAC, then the SPAC group was at fault; in all other cases no absolute criterion exists by which to measure the effectiveness of the SPAC.

The role of the *Mariner* SPAC is doubly difficult to assess, in that the *Mariner* SPAC recognized the difficulty involved in responding to spacecraft anomalies with effective corrective action, especially at planetary encounter distances with the associated 25-min communications turn-around time. As a result the primary effort was directed toward the implementation of an operations plan which would prevent anomalous behavior rather than correct it. In a number of cases this led to the pre-emption of an automatic spacecraft function, with the result that operation of those automatic functions was never fully verified. Thus no after-the-fact information is available to verify that the action of SPAC was required, and any evaluation of the SPAC performance must rest solely on an assessment of the quality of the analyses which led to the adoption of pre-emption as a flight procedure.

To the SPAC personnel themselves, the best indication of the performance of SPAC was found in the high degree of preparedness for the planetary encounter. A very large number of possible failure modes had been investigated and provided for, either through preventative action or the adoption of a plan for corrective action.

Encounter testing showed that the capability to react was high: telemetry predictions allowed almost immediate recognition of failure modes, exchange of information was quick and complete, and non-standard procedures were well understood. During the actual encounter the SPAC representatives performed their duties with the efficiency and dispatch of a highly professional team.

As with any undertaking of this magnitude, a number of points stand out as primary contributors to success, while others demonstrated considerable room for improvement. These are discussed in the following.

2. Facilities

One of the primary problems faced by the *Mariner* SPAC revolved around the fact that it was the first SPAC group to attempt to operate solely in the new SFOF. The SFOF is a mission-independent facility equipped to support a number of current and projected space flight projects, both singly and simultaneously. As such it has been built with the growth capability required to handle extremely complex missions involving a wide variety of spacecraft design concepts. As might be expected, a facility as flexible as this is extremely difficult to bring to an operational status, and even then will not meet the particular requirements of a specific spacecraft design as well as a simpler facility specifically oriented to that spacecraft.

The combination of a facility which was both new and mission independent and a spacecraft model which had not been previously flown created a number of problems which affected *Mariner* SPAC operations throughout the *Mariner IV* mission. First, each of the projects requiring the use of the SFOF had unique requirements which were not always compatible. The development of an SPAC capability requires a considerable amount of time in advance of the actual flight operations, so that significant conflict existed among the *Mariner*, *Ranger*, and *Surveyor* Projects as to both physical configuration and facility utilization. The *Mariner* SPAC received partial relief when the MMSA became available shortly after *Mariner IV* launch, but computer utilization conflicts remained subject to negotiation throughout the mission. The DPS presented another problem beside utilization conflicts. The computer system, designed to meet the requirements of advanced spacecraft beyond *Mariner*, used certain hardware and software designed to be mission independent. This placed so many constraints on its ability to handle *Mariner* data that only a very small portion of the capability of the computer system was available to the SPAC. In addition, the development and checkout of the hardware was so delayed that the primary goal of the prelaunch testing in which SPAC took part was the checkout of hardware and software. So far as the SPAC activity was concerned, the real time computer programs were more important than the nonreal time programs, so that the real time programs were emphasized to the exclusion of work on the nonreal time programs. Reasons for this included the realization that real time data processing was mandatory to support the mission during the critical phases of launch, Canopus acquisition, and the midcourse trajectory correction, and to provide a real time alarm program to monitor spacecraft engineering telemetry automatically during the cruise phase.

Successful operation of the real time data processing programs was demonstrated sufficiently far in advance of launch to provide the SPAC group with confidence that they would be able to adequately support the mission. The real time data output, however, did continue to experience a variety of problems throughout the portion of the mission that the spacecraft telemetry rate was high (33⅓ bits/sec), so that SPAC was forced to rely heavily on the unprocessed teletype data as a backup. Nonreal time processing was eventually brought to operational status for a few programs, but their usefulness was limited by SPAC's reluctance to rely on data outputs whose dependability had not been demonstrated prior to launch; only two programs required for Canopus acquisition met this requirement.

In retrospect, many problems rising between the SPAC and the DPS resulted from insufficient understanding on the part of all parties concerned. Poor liaison caused a part of this: the requesting SPAC engineers failed to define the problem completely, and the computer programmers did not adequately apprise the requesting engineers of the system constraints and programming progress. Another contributing factor, one which should be expected every time a new spacecraft model is flown, was that without previous flight operations experience on the *Mariner* Mars design, many of the SPAC representatives did not fully understand their own data processing requirements. Not until actual flight experience was gained was it possible to evaluate adequately the programs available for nonreal time processing. This evaluation indicated the need for a significant amount of revision, a requirement for several new programs, and the fact that several of the existing programs were without significant value. The computer system proved, however, to be so complex as to be ponderous; program revisions and new program development are so costly and lengthy a process that they could not be undertaken, especially when the SPAC had already demonstrated the capability to operate in spite of degraded data processing.

The conflicts over the physical facility were relieved by the creation of a separate *Mariner* area which could be used in the manner best suited to the *Mariner* operations without interference from other projects or activities. The recommendation of SPAC is that a similar solution be considered for the data processing problems in future projects. Whether or not this would mean the use of a mission dependent computer system would depend largely upon the type of capability required and the projected utilization of the existing DPS facilities by other projects. If a *Mariner*-type vehicle were to be flown again under conditions similar to *Mariner IV*, the recommendation by SPAC would be that a mission-dependent system be used for SPAC and SSAC telemetry processing. The basic processing requirement would be that the system be capable of displaying real time telemetry data to each user and also be equipped to handle all mandatory operational programs. Nonreal time processing, post processing, and special real time processing would be the responsibility of the user; logging of the incoming new data and a real time telemetry output would provide the input for these programs. Running them off-line would provide the additional benefits of isolating the nonreal time processing completely from the balance of the system and making it possible to modify, discard, or generate programs at any time it becomes advantageous.

Late in the *Mariner IV* mission, after it became clear that the existing data processing and presentation scheme could not satisfy all of the encounter requirements, the SPAC began casting about to determine what steps could be taken to provide the additional capability necessary. As it turned out, the above recommendation for the consideration of mission dependent equipment proved to be the most effective approach. No provision existed in the DPS for processing Mode 4 Data in real time, a capability necessary if the Video Storage analysts and the SSAC operations personnel were to make engineering assessments of their respective subsystems. Although a number of possibilities were examined, the most easily implemented involved the use of a portion of the ground telemetry system (GTS) equipment from the system test complex (STC). Provided that suitable conversion equipment could be found, the GTS could be driven from the raw teletype data. Suitable conversion equipment did exist, in the form of a digital-to-analog converter, designed to operate on the teletype input, which had been used by the Guidance and Control representatives for special data displays during the Sun- and Canopus-acquisition phases of the mission. The resulting combination, while certainly no panacea for data processing ills, more than proved the worth of processing and display equipment designed to a specific task. It should be noted that the SSAC group, which relied more heavily on the DPS output throughout the mission than did the SPAC group, also found that a piece of STC equipment, the real time data translator (RTDT) enhanced their operational capability significantly.

It is not necessarily recommended that total standardization of flight operations and system test data processing and display equipment be attempted. There is no reason to believe without further investigation that the two are compatible. The significant point is that the STC equipment was designed specifically for *Mariner*, and as such provides a number of unique advantages over more-or-less universal equipment. When joint use is possible, then it should certainly be considered, since among other things it lends continuity across the transition from system test data analysis to flight performance analysis. Even if the joint use of SPAC and STC equipment is not possible, the use of mission peculiar equipment in SPAC is recommended for many purposes to which it will bring the capability of extracting a maximum of usable information from the data most efficiently.

Another source of information added to the SPAC subsequent to launch was the analog display of the prime DSIF stations' received signal level (AGC voltage). This capability was implemented to provide an indication of spacecraft roll attitude should it have become necessary to place the spacecraft on roll inertial control. The late addition of the requirement for this capability, coupled with the sudden interest by SPAC in the ground received frequency when it was observed that frequency shifts were occurring simultaneously with the CC&S cyclic pulses, implied that there exist additional information sources in the DSN which could provide data beneficial to the SPAC in determining the state of the spacecraft but are not presently available to the SFOF largely because they have not been requested.

All in all, the facilities available to the SPAC group were for the most part adequate. The physical plant was acceptable at launch, and, after the transfer to the MMSA, were exceptionally good. The consensus of the SPAC members is that conducting operations from the MMSA contributed significantly to the success of the operation. The data processing facilities were generally adequate for real time telemetry, though certainly not optimum. Computer outages provided a continuing problem throughout the cruise, though performance during the critical phases was quite good. Nonreal time data processing never did live up to expectations. Computer program development suffered significantly from: 1) the lack of a firm schedule, 2) the lack of a clear definition of responsibility, and 3) the lack of regular reporting from both sides of the SPAC – DPS interface. Special requirements could be satisfied outside of the SFO system with the use of *Mariner* peculiar equipment when necessary.

3. Organization

a. SPAC internal organization. Review of the SPAC organization reveals that basically the organization was adequate to accomplish its required tasks. It was fashioned along the lines of the *Mariner* project, with Division Representatives reporting to the SPAC Director, and subsystem representative reporting to the Division Representatives. By bringing the influence of the SPAC Division Representatives to bear within their own divisions, the SPAC Director had an effective means of implementing SPAC policies among all members of the organization. This technique proved effective in achieving various tasks desired of the subsystem representatives.

There were three noticeable deficiencies which should be corrected in future missions. First, there was no full-time SPAC data-handling engineer. The requirement for such an individual was recognized much before launch, but this requirement was never adequately implemented. As a result, people who were not thoroughly familiar with many facets of the 7044-7094 computer system in the SFOF were forced to participate in resolution of many computer interface problems. With insufficient time to adequately learn the computer system, the SPAC personnel were not too helpful in resolving computer problems. In retrospect, having the proper personnel on the user side of this interface would have eased the problem experienced in checking out the computer programs. Another of the functions of the data-handling engineer would be to resolve any difficulties in distribution of data or in the establishment of computer program operational requirements in conjunction with the data processing and SPAC personnel. In the future a plan must be worked out to provide such an individual from somewhere in the organization. The experience gained on this project indicates that the dollar-savings possible through the utilization of a data handling engineer would most certainly justify his existence.

The second area that can be noted as a deficiency in the SPAC organization is the relationship between the SPAC Director, the Division 32 Science representatives, and the engineers who report through him to the SPAC. It is concluded that future projects must provide adequate space to permit the science instrument engineers to be face to face with the Division 32 Science representative who must be located in the same room as the SPAC Director. This arrangement will also enhance direct communications as required between the instrument cognizant engineers and other engineers on the *Mariner* SPAC. As was learned in the *Mariner* operations, there are frequent opportunities for exchange of information between the science engineers and the engineers from other spacecraft subsystems. This fact was not sufficiently acknowledged prior to the mission.

In actual practice, it was also learned that the science analysis area can effectively be divided between operational personnel, that is, instrument cognizant engineers, and scientific data analysis individuals. The latter are generally the cognizant scientists and associated investigators. Since the *Mariner* encounter operations were conducted from the *Mariner* Mission Support Area, SPAC can, indeed, see the validity of such a suggestion and offer this recommendation for future missions. In the encounter operations only those individuals from the SSAC who would have a voice in the operational decisions were permitted entrance. A special interface was established with the scientific investigators located in another area. This interface was handled by the SSAC Director and his alternate.

The experience of the *Mariner* SPAC was that the necessity for a documentation engineer cannot be underestimated. Initially, this need was not sufficiently recognized. This was unfortunate, particularly on a mission of such duration, because ability to document progress on a continuing basis is most important. Keeping of accurate records and logs is a prerequisite to good documentation; our efforts in this respect were woefully lacking until the documentation responsibility was given to a single individual.

From an operational point of view, the necessity for adequate procedures cannot be minimized. The documentation engineer is able to provide the assistance required in getting the procedures published. Another benefit accrued by the documentation engineer's participation in the SPAC effort is dissemination of spacecraft performance information on a weekly basis. This effort served to keep the laboratory personnel, including those individuals who were reassigned to other projects after the *Mariner* launch and initial flight operations, informed throughout the mission. These weekly reports seemed to solve one of the big potential problems on a mission of many months' duration: a sense of participation in the flight operations for those personnel who participated heavily on the pre-launch phase.

The documentation engineer provided services which were indispensable in SPAC proposals to project management. As a result of the *Mariner* SPAC experiences, it is recommended that in the future key positions in the project organization be staffed to include one individual whose prime purpose is to give documentation support.

One of the prime contributions to the effectiveness of the SPAC organization was the fact that the majority of the SPAC personnel were spacecraft-design oriented. Many of them had followed the *Mariner* spacecraft in one capacity or another since the inception of the project. The resulting store of knowledge about every facet of the spacecraft available to the SPAC made it possible to assess the state of the spacecraft throughout the mission with a high degree of confidence. While the limited amount of data available through the telemetry link means that certain ambiguities will always exist in determining the state of the spacecraft, it is significant that at the termination of the first phase of the mission on October 1, 1965 the SPAC members had been able either to solve or advance workable, though unverifiable, hypotheses as to the cause of all significant flight anomalies. The necessity for personnel in flight operations who are intimately acquainted with the design of the spacecraft at all levels of detail cannot be emphasized too highly.

b. SPAC external organization. While the internal organization served well with the exceptions noted, the organization external to SPAC, that is, the manner in which SPAC interfaced with the balance of the SFO organization, presented a number of problems. One of the problem areas is discussed in the previous paragraphs, that of the data processing interface. In this case it was the internal organization of SPAC which was deficient; the addition of a full-time data-handling engineer would have provided the necessary liaison to prevent most if not all of the interface problems.

The entire problem of SPAC external organization can be summarized as being one of incomplete definition of responsibility. The intent of the *Mariner* SPAC was to develop an active, spacecraft design oriented capability within the flight operations structure, as is in keeping with the nature of the design of the *Mariner* Mars spacecraft, rather than emphasizing an after the fact data analysis capability. The extended length of the *Mariner* mission, however, led to a degree of active participation in the flight operations which had not been anticipated, let alone experienced in previous projects. As a result, most of the interfaces which the SPAC wished to use had not previously been established, nor was there any full understanding of the interface requirements. No overlap of responsibility existed; in general the problem was one of creating working relations with portions of the flight operations organization which had not previously fallen within the SPAC sphere of activity.

One such interface was the SPAC – DSIF interface. During the actual flight operations all contact with the DSIF was through the SFOD, as it should be to ensure control of the operations during critical phases. Prior to launch, and during the cruise phase when full time SPAC coverage was not required, however, there was no provision for any official communication with the DSIF Project Engineer. This deficiency was rectified somewhat by the evolution of the SFOD's daily status meetings to accommodate a planning function as well as reporting. With both the SPAC Director and the DSIF Project Engineer in attendance, procedures could be established and discussed, then offered to the SFOD and the Project Office Representative for concurrence. In general, however, the major interaction between the DSIF and the SPAC group took place on the unofficial level throughout the entire mission.

Ultimately, this informal contact with the balance of the SFO activities became the rule rather than the exception until this gradual expansion of the SPAC's sphere of influence to permit direct interfacing became the accepted, if never formalized, working arrangement. In this manner working relationships were established with the Tracking Data Advisors (TDA), the Ground Telemetry Advisors (GTA), and SSAC operations personnel, and the test team responsible for the operation of the *Mariner* PTM spacecraft in support of the flight operations. This direct contact approach permitted the full exchange of information and resultant understanding which contributed largely to the smoothness of the encounter operations.

The necessity for full interfacing by SPAC is derived simply from the fact that the SPAC is the only group within the flight operations organization with a body of detailed spacecraft design knowledge at its command, and thus is the unique source of information which greatly affects operational procedures.

The creation in January 1965 of the EPWG to investigate and recommend to the Project Office the steps necessary to enhance the probability of a successful planetary encounter gave the SPAC yet another group with which a direct interface was mandatory. The chairman of the EPWG was the Spacecraft Project Engineer. His committee members were basically the Project Representatives from the various technical divisions and areas of responsibility. Initially these individuals were not the prime members of the *Mariner* SPAC group. The composition of the EPWG changed in the succeeding months, however, with the result that the EPWG organization was primarily composed of individuals from SPAC. That their participation in the EPWG increased their ability to support the encounter operation was amply demonstrated.

4. Preparation and Planning

The majority of SPAC Division representatives were spacecraft design oriented and were required to phase out of these activities to allow more time for flight operations oriented activities. To precipitate this phase-out and to promote thought and discussion by all concerned, weekly meetings of the SPAC nucleus were initiated in July, 1964. SPAC testing commenced at approximately this same time. These two activities were complementary in regard to the establishment of the configuration of the SPAA, the checkout of and familiarization with the computer programs, and the generation of operating procedures for the SPAC. Prior to these meetings other meetings of the SPAC nucleus had been called as necessary to respond to various action items, such as establishment of the SPAC organization and definition of computer programs. These weekly meetings provided a sounding board for all aspects of the SPAC effort; the results of these meetings were instrumental in the success of the *Mariner* prelaunch preparation.

Subsequent to launch and the re-orientation of the mission operations into the more routine cruise phase, a series of daily, except Saturday and Sunday, SPAC meetings was initiated. This series of meetings had the prime purpose of reviewing the spacecraft performance and data processing performance of the preceding day to provide feedback into the Space Flight Operations Direction and the Project Office as to the progress of the mission. From the viewpoint of the SPAC Direction Team the daily meeting technique was extremely effective in providing total SPAC performance coverage, and as a result provided much insight into the operation of the spacecraft by essentially dividing the discussion of spacecraft performance into small, more easily assimilated, segments. Had the meetings been held less frequently, certain aspects of spacecraft performance might have been ignored inadvertently in the press of routine business. Ample opportunity was available to the participants to mention and discuss some of the finer points of spacecraft performance and operation, because on many days there was only minimal reporting to be done. The overall education and training of the SPAC group members was enhanced through the interchange of information of those members in specialized areas of operation. In conclusion, it was the constant interaction between the SPAC Director and SPAC Division representatives which resulted in the ability of SPAC to take the lead in the generation of the mission operations plans that were developed and executed during the flight.

Operational procedures to be used during the critical phases of the mission were generated by the SPAC group prior to the respective phases. Preparation of these procedures required considerable reflection upon what would be done under a great variety of situations; this very act of putting on paper the various plans which could be followed increased to a great extent the SPAC group's preparedness for unexpected conditions. Without having considered in advance as many conditions as possible, it would be questionable that all the necessary considerations to any particular action might be forthcoming when the actual condition occurred in flight. The procedures indeed were complementary to the mission operations, but it is recognized that a greater benefit accrued from generating the operational procedures than from having the procedures in hand at the beginning of the mission phases.

The philosophy of the SPAC followed the Project guidelines of sending as few spacecraft commands from the ground as possible. Each command that was transmitted was thoroughly justified, its effect on the spacecraft was analyzed, the effect of failure modes that could be induced were analyzed, and subsequent to all this analysis the recommendation was made and permission granted by the Project Office for the commands to be transmitted. This rigorous review of anticipated action demanded individual mental and written justification of the intended operations, in order to better understand and explain some of the ramifications of the suggested actions. In effect, if commands could not be justified they were not sent. This philosophy is endorsed and recommended very highly for future mission operations.

One of the most significant assets to the *Mariner* SPAC in preparation for critical phases of the mission was the test program. During the initial testing, the test objectives were the checkout of the DPS, the facility and the analysis area data outputs. As a result the data source was required to simulate nominal telemetry in a nominal flight sequence. While this will adequately demonstrate the ability of the SFO organization in general to support a nominal mission, it does not test SPAC. Since the spacecraft design calls for the successful completion of the mission without SPAC support, then it follows that the SPAC group's primary function must be to anticipate, diagnose and correct anomalies. Nominal spacecraft telemetry simulations cannot exercise this function.

Not until October 1964, just 1 mo prior to the first *Mariner* Mars launch, was an adequate data source provided to the *Mariner* SPAC members. This was the *Mariner* Proof Test Model (PTM) operated in JPL's Spacecraft Assembly Facility (SAF) in Pasadena. With a flight-type spacecraft as the data source, it was possible not only to simulate the launch sequence in a more realistic manner, but also to induce failure, request commands, and show an immediate response to any spacecraft event in the telemetry. The highest praise that can be given to the prelaunch PTM test effort is found in the fact the Canopus acquisition simulation was criticized in the SPAC test critique as unrealistic by the cognizant SPAC personnel, yet almost exactly duplicated the sequence experienced later by the *Mariner IV* in flight. Needless to say, none of the computer simulations came close. Unfortunately insufficient time remained prior to the *Mariner III* launch to fully utilize this new found capability to the extent desirable.

The failure of the aerodynamic fairing covering the *Mariner III* spacecraft and the problems encountered by the SPAC in attempting to salvage the mission reinforced strongly the view that failure mode testing was necessary to provide the SPAC group with the capability required to support adequately the *Mariner* mission. As a result, the balance of the tests supported by SPAC employed the PTM as the data source, and, for the most part, permitted induced failures which required SPAC analysis and command action.

This type of testing was implemented almost exclusively during the post launch testing. The SPAC encounter test operations were intended to exercise the engineering and science performance analysis personnel in the performance of their duties with the objective of developing and maintaining the proficiency required to participate in the critical operations involved in the planetary encounter. Because the personnel involved had been actively engaged in the interpretation of spacecraft telemetry data in order to determine the state of the *Mariner IV* spacecraft from the time of launch, it was not intended to test toward the objective of orientation and familiarization with spacecraft operations, but rather to test toward the objective of exercising all of the interfaces within the DSN which directly affected the SPAC group, of providing a thorough understanding of the spacecraft logic which had been previously unexercised during the flight and of developing in detail the procedures that would be used during the actual flight operations.

Throughout the SPAC encounter test program, the PTM Test Director was requested both to provide the most realistic simulation possible and to induce correctable failures or responses to the space environment at his discretion. Failure mode testing had the additional benefit of creating an air of competition between the SPAC representatives and the SAF test team and operators to the end that the degree of participation in each of these tests by all personnel was the highest observed during the program. Considerable effort was expended by the SAF component of the test effort to induce or simulate failures which were extremely hard to diagnose or which required significant changes in the normal operations procedure. It is the general consensus that the SPAC group would not have been able to attain a state of operational readiness for the encounter phase of the mission had PTM failure mode testing not been employed.

5. Operations

The SPAC group participation in flight operations went very smoothly. The primary reason for this was the realization that during a critical operation spacecraft performance analysis is a full-time task, with no allowance for the presentation, discussion, decision and recommendation of corrective actions. This meant that as many decisions as possible had to be anticipated, so that SPAC response during the operation would be automatic; the recognition of a certain set of spacecraft events or parameters would lead directly to a recommendation of a specific course of action. In this manner the major decisions by SPAC were made in advance, after thorough investigation, lengthy discussion, and careful review, rather than during the operation when the press of time and the urgency for correct action would severely limit the capability to give careful consideration to all pertinent aspects of the problem.

There were two problems encountered by SPAC in flight operations: effort was required to adapt the SPAC organization to be compatible with 24-hr operations support, and some difficulty was involved in coordinating the SPAC spacecraft performance analysis activities with these SPAC recommendation and decision activities which could not be handled in advance.

Initially no cognizance was taken of the fact that certain of the responsibilities of the members of the SPAC group were assignable, while others lay with specific people rather than positions in the SPAC structure. As a result the prelaunch concept called for qualified alternates to replace the primary SPAC members in the less critical portions of any given mission phase. It was not until after the *Mariner IV* launch that it was realized there were certain non-assignable responsibilities which the division representatives and the SPAC Director were required to bear on a 24-hr basis regardless of any arrangements to the contrary. The science cover deployment marked the first time that the SPAC structure as modified for operations was used. Under this scheme the non-assignable duties were specifically separated from the assignable ones. It was then recognized that for a division representative to commit his division to support a specific operations phase, he must necessarily have completed his non-assignable tasks. He may then be replaced in the operation by any alternate qualified to perform the same level of analysis without jeopardizing the coordination effort of that division's SPAC activities.

The SPAC Director's position presents a different problem. Many of his non-assignable responsibilities only begin as the effort shifts from preparation to operation; at the same time the flight operation also brings with it a wealth of routine tasks involving the internal coordination of the SPAC group and the coordination of SPAC's activities with the SFO groups external to it. During the launch and midcourse phases, when SPAC was located in the SPAA and thus isolated physically as well as organizationally from the balance of the flight operations organization, this problem was solved by the addition of an SPAC Operations Assistant. If the SPAC Director were to do his job adequately, that is, pursuing the analysis of spacecraft data and resolving potential problems, he had to be as free as possible from the routine operational problems and constraints imposed by the remainder of the Space Flight Operations Facility and some of the mission operations. The routine features of the Space Flight Operations organization were turned over to the operations assistants for each of the areas.

Observations of previous JPL project operations indicated that there was considerable emphasis on the SPAC Director's participation in routine discussions with other members of the Space Flight Operations organization and his providing liaison between the SPAC and other elements of the SFOF. The *Mariner* SPAC maintained the view that an operations assistant, with sufficient training, could satisfy these requirements and free the SPAC Director to concentrate on the analysis of the spacecraft. This did, however, create the problem of the Director interfacing with the operations assistants. This particular interface required a good deal of effort, but by the time of launch it was felt that the operational assistant was an integral part of the organization and his presence aided the total SPAC operations capability to a great extent.

After the SPAC group was relocated in the MMSA, the problem of physical separation no longer existed. Members of the various flight operations groups could meet face to face, and the majority of the business was transacted over a single communications net, so that the interfacing problem did not exist to the same extent. The first operation performed in the MMSA was the science cover drop exercise in February 1965. Since this operation was via ground command only, it may be considered a non-standard sequence; that is, not provided for in the nominal sequence of events. As such it required that the SPAC Director continuously be available to the SFOD and the Project Manager for consultation and discussion, since the SPAC Director is the official spacecraft representative during the flight operations phase of the project. In order to meet this requirement it was decided that the assignable tasks involved with the routine direction of the SPAC group should pass to the position of Assistant SPAC Director, which could be occupied by any of the SPAC Director's staff. This concept of an SPAC direction team was employed throughout the balance of the mission.

While the SPAC direction team approach solved one problem, it emphasized another, that of providing the necessary coordination between the performance-analysis component of SPAC, i.e., the SPAC representatives and the Assistant SPAC Director, and the recommendation and decision component, i.e., the SPAC Director. This was a part of the general problem of communications of the Project Manager, the SFOD, and the analysis area directors with the members of the analysis and operations groups. The members of the various groups were required to implement actions to be taken in the flight operations but were often the last ones to be informed of specific requirements. Frequently these people would have been able to have more adequate preparation for the ensuing action had they been kept abreast of the proceedings in the closed door meetings held by the SFOD, the Project Manager, advisors, and the area directors. Due to the nature of the closed-door meetings the area directors lost all contact with the members of their groups. Partial relief for this was found when the assistant area directors were permitted to attend the meetings with the express purpose of establishing liaison between the meetings and the group members. This had drawbacks also in that the two prime people from each analysis area were out of the operations area.

The solution finally evolved was the establishment of a private communications net for the Project Manager, SFOD, and the analysis area directors in the *Mariner* MSA. This permitted the area directors to be located near their groups and permitted face-to-face conversations to be carried on with the members of their respective groups without loss of effectiveness relative to the Project Manager and the SFOD. Quicker feedback could be obtained with this system so that a more complete understanding of any contemplated action was achieved at all levels.

A similar system may offer some solution to the problems encountered by future projects in the flight operations area. However, the *Mariner* experience indicates that all projects have totally different communications requirements, based on the nature of the spacecraft design, the mission flown, and the individuals involved in the operation. Rigorous efforts obviously must be applied in future projects in order to understand fully the requirements for communications at the upper levels of the SFO organization.

V. SPAC PLANNING

A. Planning Philosophy

A significant portion of the activity of the *Mariner* SPAC involved preparation for the various phases of the mission. The objectives of this SPAC preparation were to establish an acceptable plan for each phase of the mission, to develop a nominal sequence of events and a set of criteria by which the performance of the spacecraft could be evaluated, to establish the courses of action to be followed in the event of nonstandard performance of the spacecraft, and to develop in the SPAC personnel and procedures the level of proficiency required to carry out the flight operations responsibilities of the SPAC. All of this activity was specifically directed toward the development of an SPAC capability to react properly in the actual operation to any sequence of events, standard or nonstandard.

Throughout the life of the program it became increasingly apparent that the nature of the mission, and the complexity of both the spacecraft and the ground equipment precluded the successful execution of any course of action which had not been detailed and agreed to in advance by all parties concerned. This was especially true in the SPAC area, where the people who are responsible for proposing and evaluating alternative courses of action at a detailed level are the same people responsible for evaluating spacecraft performance. During a critical phase of the flight operation these two activities are mutually exclusive; if the SPAC members concern themselves with evaluating the state of the spacecraft they cannot simultaneously generate a recommended course of action effectively. If their attention is diverted to the formulation and evaluation of a plan of action, then little or no information on the state of the spacecraft is obtained. Obviously, for a satisfactory SPAC operation, both complete knowledge about the current state of the spacecraft and an effective, well understood plan of action are required.

As a corollary to this, even if the state of the spacecraft could be monitored effectively without the full time support of the SPAC members, it is practically impossible to generate spontaneously a good emergency procedure in response to an unanticipated anomaly on the spacecraft. Unanticipated is the key word here. It implies that the source or sources of the problem are unknown, the impact of the failure, if one exists, upon the mission is not understood, and exact response of the spacecraft to any given set of ground actions is uncertain at best. Coupled with this are the facts that time is usually critical and all of the possible ground actions will carry with them disadvantages as well as advantages, each of which must be weighed and evaluated. In thirteen launches of *Ranger* and *Mariner* type spacecraft, six unanticipated spacecraft anomalies (as opposed to trajectory anomalies) have occurred and the only one which was was successfully circumvented was not extremely time critical (*Mariner II* partial power failure). The remaining five spacecraft failed to achieve their mission objectives. It is probable that none of these missions could have been salvaged even if the ground performance had been flawless, but the important fact is that the operations personnel performing the SPAC function were unable to attempt in a timely manner the proper corrective action.

This leads to the first principle underlying the *Mariner* SPAC planning philosophy: the performance of the spacecraft at each step in any sequence of events must be understood and verifiable, and each possible failure mode, its symptoms and its consequences, must be anticipated. The second principle is as important as the first: for each possible failure mode a plan for corrective action must be generated before the fact and agreed upon by the SPAC personnel. For probable or extremely time-critical failure modes the plan for corrective action should be agreed upon in principle by all parties concerned in the flight operations, including the Project Manager, so that reaction time is minimized. Given the proper SPAC preparation for any phase of the flight operation, any anomalous indication in the telemetry would be immediately recognizable, and the possible failure modes which would yield that particular anomaly would be catalogued, along with the recommended corrective action. The SPAC Director, the SFOD and the Project Manager should be thoroughly cognizant of the implications of both the anomaly and action required to cure or circumvent it, so that the decision required to take corrective action is a simple approval or disapproval rather than a complex procedure of proposing, explaining and evaluating.

The third axiom from which the planning philosophy is derived is as obvious and elementary as the first two: take no positive action without a positive reason. The *Mariner* design is sufficiently complex that improper ground command action may place the spacecraft in a state from which recovery is not possible. Any time a command is inserted into the spacecraft, there is the possibility that either the initial state of the spacecraft or the effect of the command are not completely understood, with the result that irrepairable damage may be done to the mission. This risk must be outweighed by the value attached to ground action before any perturbation to the spacecraft state is warranted.

The first activity in the SPAC planning for any particular portion of the mission is the recommendation of a mission plan. For the standard phases of the mission, such as launch, midcourse or encounter, a mission plan already exists in the form of the CC&S programmed sequence of events. This is in line, of course, with the *Mariner* design philosophy which dictates that the spacecraft must be able to perform a nominal mission, except for the midcourse correction, without any ground-based intervention or support. It is also true, however, that the spacecraft is able to complete all of the mission objectives without the CC&S, relying solely on ground commands. In fact, since the CC&S sequence of events is necessarily designed into the spacecraft a considerable time before launch, the response of the spacecraft to its environment during flight may render the prelaunch nominal sequence less than optimum. In this case the probability of achieving mission success may be enhanced considerably by augmenting or pre-empting the CC&S events with ground commands.

It was the responsibility of the SPAC to generate a plan for each phase of the mission or, if a CC&S nominal plan existed, to review all of the possible alternatives and recommend the optimum plan. For certain phases of the mission this task encompasses so much work that a separate *ad hoc* committee may be formed to perform the mission planning and evaluation. This was the case in the *Mariner IV* encounter phase where the Project Manager chartered a special Encounter Planning Working Group (EPWG). Ultimately personnel changes resulted in an EPWG composed for the most part by SPAC members and SFO personnel who interfaced directly with the SPAC, so that the EPWG may be considered largely as an SPAC activity.

In the establishment of a mission plan, the many alternative courses of action were examined in the light of the mission objectives as stated by the *Mariner* Project Office. Any proposed command action from the ground was evaluated by the SPAC group to determine what undesirable side effects or possible failure modes might be induced by it. Where possible, quantitative analyses were made so that the relative desirability of alternate plans might be assessed.

During the later stages of the mission, one particular problem whose significance had not been fully appreciated before became the overriding consideration in developing an encounter plan. This was the transmission delay which, on encounter day, was slightly longer than 12 min. As an example of the impact which this had on planning, consider the event which initiates the planetary television recording sequence. Normally controlled by an onboard narrow-angle planet sensor, it can be backed up by a ground command. For an Earth satellite or lunar mission the standard flight operations practice would be to confirm the onboard event in the telemetry. If it did not occur, however, then the backup command could be transmitted to the spacecraft with a minimal effect to the mission objectives. Application of this type of operation to the *Mariner* mission is a different matter. Confirmation that recording had not started would arrive on Earth after 12 min and recognition, decision and command initiation would require an additional 1 or 2 min, so that the backup command would arrive at the spacecraft 25 min after nominal time for recording initiation or just after the planet had passed out of the field of view of the camera. Obviously for the *Mariner* mission this application of backup commands is impractical. If the backup command is transmitted early, however, so that it arrives at the spacecraft at the same time or slightly after the onboard event is to take place, it is effective in providing a redundant means of performing the required function. This means that the latest possible time to commit the use of this backup is some 20 to 25 min *before* there is any indication of whether or not it is required.

The extension of this same type of reasoning to the balance of the events required during the highly time-critical planetary encounter phase of the mission led to the adoption of a mission plan for encounter which bore little resemblance to the nominal sequence programmed into the spacecraft prior to launch. Perhaps the most significant thing to be noted in this modification of the encounter plan is that the pre-emption of the nominal programmed plan does not indicate that it was deficient in any way. It was developed to provide nominal mission capability to the spacecraft denied any support from the ground and, had such been the case, it would have done the job admirably. The actual plan was designed to make the most efficient use of the full capabilities of the entire DSN-spacecraft complex, which, of course, is the whole point of flight operations planning.

In the development of the mission plan, then, attention is not limited to the spacecraft alone, but must be given to all of the areas and activities within this DSN-spacecraft complex. Many of the possible ground command sequences which are most attractive from a spacecraft point of view when performed correctly carry such large penalties that they are discarded simply because ground-based operational difficulties preclude the guarantee of a sufficiently high probability of correct performance.

After a recommended plan for each phase of the mission had been formulated, it was presented, with some discussion of its advantages and disadvantages, to the Space Flight Operations Director and the *Mariner* Project Office. Final responsibility for all *Mariner* mission plans rested with the *Mariner* Project Manager, who issued the approved plan and coordinated any of the activities required for its implementation.

The approval by the Project Office of a particular plan of attack initiated another activity of the *Mariner* SPAC group, the establishment of a nominal sequence of events for that plan. The sequence of events is a detailed chronology of all of the spacecraft changes of state which will occur during the execution of a particular mission plan. The SPAC members are especially telemetry-oriented since telemetry furnishes nearly all of the information available concerning the state of the spacecraft, so the effort to establish the sequence of events led directly to the tabulation of nominal telemetry values.

The sequence of events with its associated telemetry table provided the SPAC with all of the tools required both to predict and verify all of the state changes of the spacecraft. Not only did this allow immediate recognition of a nonstandard occurrence onboard the spacecraft, but it also placed in the hands of the SPAC analysts the primary diagnostic instrument for determining the cause of any deviation from normal performance.

Once the nominal sequence of events had been established, an effort was initiated for the purpose of anticipating possible failure modes and appropriate corrective action. One aspect of this activity during the *Mariner IV* mission was the generation of a series of sequence flow diagrams which depicted the large number of alternate paths which were possible, either by option or due to failure, at any point in the sequence once it had been initiated. This particular activity forced much thought on the part of the SPAC personnel in the area of nonstandard sequence plans. An important part of nonstandard sequence planning is the increased understanding gained by SPAC of what corrective capability does exist within the spacecraft design.

Nonstandard sequence plans were submitted to the Project Office for approval prior to the actual flight operation in order to provide the capability to take action in a timely fashion whenever a nonstandard condition arose. It was, however, still necessary for each command transmission to the spacecraft to be approved by the *Mariner* Project Office. The major significance of the approval of nonstandard sequences of events was that the SPAC group was now provided with an agreed-upon reaction to various possible circumstances, so that much discussion and debate could be eliminated at the time of any inflight anomaly.

The final test of any plan, of course, is how well it works when implemented. In order to verify that the mission plan, the sequence of events and the alternate plans were adequate, the SPAC conducted a series of tests, often in conjunction with general SFO tests. A large number of the tests conducted after *Mariner IV* launch employed the PTM spacecraft as the data source, rather than using simulated data or available test tapes. So that the SPAC group could be put to the most severe test of its proficiency and so that its procedures could be best validated, the PTM Test Director was allowed to induce or simulate spacecraft malfunctions at his discretion. The SPAC and SSAC operations personnel were then responsible for determining the corrective action required to allow the successful completion of the test objectives.

The contribution to the mission success of this test activity was emphasized by a large number of significant changes to the sequence of events and the operational procedures which followed each of the encounter preparation tests.

In retrospect it appears that the level of preparation applied to the *Mariner IV* SPAC operations, especially prior to encounter, represents little more than the minimum amount required to meet acceptable SPAC performance standards. By planetary encounter the SPAC group and associated SSAC operations personnel were able to conduct the encounter operations confidently and efficiently. This was a requirement, however, instead of an achievement; there still remained a number of potential problem areas which might have compromised the mission had certain failures occurred for which adequate preparation had not been made.

The planning philosophy proved valid. The diligent application of the philosophy to the preparation for encounter provided the SPAC with sufficiently firm guidelines to allow the successful execution of the mission sequence even in the face of adverse circumstances. This planning allowed the establishment of a set of criteria under which certain actions would be taken. These criteria were approved and understood among the *Mariner* SPAC Director, the SFOD, and the Project Manager, thereby guaranteeing a maximum of information flow and the maximum capability to react.

B. Launch Planning

1. General

The launch planning activity of the SPAC group was handled in a somewhat different manner than the planning for the balance of the mission, largely because the primary effort prior to launch was directed toward developing the necessary operations capability within the new and relatively untried SFOF. The SPAC group was deeply involved with establishing data processing requirements for flight operations, determining the hardware utilization which would best fulfill these requirements, developing its own internal organization, and testing in order to learn how to use the facilities at its command. Most of the tests performed in the facility were not designed specifically to bring the SPAC to a state of proficiency, as would have been desirable from an SPAC point of view, but rather were designed to qualify facility hardware and software. Obviously it is fruitless to conduct operations tests with the objective of providing a capability to react to nonstandard events when no such capability exists for standard events. As a result the principles furnishing the basis for the *Mariner* SPAC planning philosophy were not adhered to prior to launch as rigorously as they were during the later stages of the mission.

It should be noted that one of the reasons for this is that the urgent need for in-depth planning had not yet been realized. This type of planning is primarily of value in an active, real time operation. Prior to the *Mariner* Mars project, flight operations at JPL had included a performance analysis function which was largely passive in nature. The basic design of the spacecraft flown had not called for a large scale participation by the performance analysts; specifically, there was no provision for the transmission of any commands, except those required for the midcourse correction, for anything less than a potentially catastrophic emergency. The *Mariner C* design, on the other hand, was provided with a large number of command options designed to be used as necessary, not just limited to dire emergencies. The first real indication that ground command transmission might become a standard part of flight operations came with the realization that an extremely high probability existed that acquisition of Canopus would require weeks or months unless ground commands were employed. With this development the SPAC began to evolve to include an active command function.

2. Launch Plan

The launch of the spacecraft marks the transition from control via the spacecraft test team to control via the flight operations team. In effect this transition occurs during the real time telemetry blackout which occurs prior to post-injection acquisition of the spacecraft by the DSIF. For the *Mariner* flight sequence, this transition period coincides with the booster separation events and the initiation of Sun acquisition. As a result, the first continuous postlaunch data from the spacecraft is obtained after many of the significant events leading to attitude stabilization have occurred. The SPAC group's task, then, is primarily to determine precisely the new state of the spacecraft from the telemetry without being able to monitor it continuously through all of the state changes. Because many of the measurements available can be somewhat ambiguous, the SPAC launch planning consists mainly of determining a unique set of telemetry criteria for each possible spacecraft condition, and, for each nonstandard condition which might adversely affect the mission, of determining the possible corrective actions.

The prime objectives of the launch phase of the mission were the successful injection of the spacecraft on a Mars trajectory, deployment of the solar panels, and acquisition of the Sun. Without these the mission is an immediate failure. The shroud failure on *Mariner III*, for example, precluded all three of the launch objectives and the spacecraft failed upon the depletion of the battery. In addition to the primary objectives, a list of secondary objectives was recognized. They were:

1. Turn on cruise science.
2. Increase radio transmitted power (RF power up).
3. Remove CC&S relay holding current.
4. Turn off video storage launch mode.
5. Attain magnetometer calibration roll.
6. Acquire Canopus.
7. Assess the ability of the spacecraft to perform the midcourse maneuver.

Each of these objectives, primary and secondary, with the exception of injection over which SPAC had no control, had to be examined, the procedure by which its verification could be made determined, and appropriate plans for any corrective action developed.

a. Solar panel deployment. Spacecraft telemetry would permit the SPAC to determine the following conditions relative to solar panel deployment:

1. Whether the pyrotechnic subsystem had been armed by the pyrotechnic arming switch or the separation-initiated timer.
2. Which, or both, of the pyrotechnic subsystems issued a solar panel deployment command.
3. Which of the solar panels were deployed to within 20 deg of the full open condition.

The latter condition infers that a solar panel was completely deployed because of the nature of the solar panel deployment mechanism. In addition, it is possible to observe whether or not the CC&S launch counter did generate the first of its commands to back up the initiation of solar panel deployment. This particular function could be inhibited if the CC&S relay holding current had not been properly released at separation. The mechanization of this feature is such that the holding current would certainly be removed whenever the gyro power was turned off. Should the solar panels be observed to be in their launch position and the CC&S command not having been issued, the gyros would be turned off by ground command. Subsequent to gyro turn-off, the CC&S would be permitted to issue the solar deployment command at some future time predictable from knowledge of the CC&S counting sequence. It would be preferred to transmit the gyro-off command, DC-15, prior to launch plus 93 min. One constraint on the time for the turn-off of the gyros is the recognition that the video storage launch mode is also powered by one phase of the gyro three-phase power supply. In the case of a failure in the video storage launch mode logic,[17] it would be desirable to turn off the gyros such that the video storage end-of-tape foil would have just passed over the

[17]See subheading g., which follows in this Section V.

end-of-tape sensor when the command is executed. Subsequent to the issuance of the CC&S L-1 command for solar panel deployment, it would be permissible to re-energize the gyros to effect the Sun acquisition by the transmission of DC-19.

It is also recognized that whenever the gyros would be on in the future it would be impossible for the CC&S to generate any of its internal commands from the launch counter or from the master timer (end counter). A midcourse maneuver is permissible under the conditions of a relay hold failure.

b. Sun acquisition. Sun acquisition would commence as soon as the attitude control switching amplifiers were turned on. Of course, the Sun acquisition process would not commence until the spacecraft was out of the Earth's shadow: such conditions exist on approximately ¾ of all near-Earth trajectories. Even in the Earth's shadow the attitude control logic would cause the rates about the control axes to be reduced to the rate dead-band limits set by the gyros. The occurrence of the pyro arming event in the telemetry as mentioned in the previous paragraph infers that the pyrotechnic arming switch has been activated and the attitude control electronics should have been turned on; however, if the telemetry indicates that neither the rates about the control axes are decreasing nor that the Sun sensors are generating error signals, this particular function probably had not been accomplished. A backup function is also available as a command from the CC&S launch counter. This command is subject to the same failure mode indicated in the above paragraph for the solar panel deployment. In this condition, the gyros will be maintained in the off mode by ground command for a sufficient length of time to permit the CC&S to generate this backup attitude control turn-on command if such is necessary. Subsequent to the CC&S command, the gyros would be turned on for the enhancement of the Sun acquisition process. The acquisition of the Sun by the spacecraft is completely automatic. No ground commands are possible which aid the Sun acquisition once the electronics have been turned on and the procedure has been initiated.

c. Cruise science turn-on. Verification of cruise science turn-on can be had from observation of the spacecraft telemetry format and noting the change in format of each data frame from ⅔ zeros and ⅓ engineering to ⅔ science and ⅓ engineering. It is only at the turn-on of the Data Automation Subsystem that it begins to transmit science data to the Data Encoder for subsequent transmission to the ground. Analysis of science telemetry permits verification that all science instruments are on. An alternate means of verifying science instrument turn-on is to analyze the power subsystem telemetry. Should the cruise science instruments and the DAS not be turned on automatically at separation there is a ground command, DC-2, which will turn on the cruise science through an alternate path. Should this corrective action be insufficient, cruise science data could only be obtained in the future by also turning on the encounter science instruments. A feature of the design was to turn on cruise science at the time of the initiation of the encounter sequence. This latter approach would not normally be followed prior to encounter because of the degradation to portions of the encounter instruments that would result from excessive turn-on and operation in the cruise phase.

d. Increase of transmitted RF power. RF power transmitted from the spacecraft is maintained at approximately 1 w prior to separation by a relay holding-circuit passing through the spacecraft separation plane. This circuit is to be opened at spacecraft separation. It is powered by the power subsystem maneuver booster which is required to activate the gyros. Should automatic RF power-up not occur, it will be possible to attain increased transmitted power from the spacecraft whenever the gyros are off. Conceivably the entire mission could be attained even with this degraded mode of operation; telemetry during a midcourse maneuver may be degraded, however. The occurrence of RF power-up can easily be verified by reviewing the received signal level at the DSIF stations in addition to reviewing the spacecraft telemetry which will indicate the transmitted power.

e. Canopus acquisition. The orientation of the Canopus sensor toward the star Canopus (Canopus acquisition) was very important to the *Mariner* flight sequence. This technique had not previously been implemented for a flight project, but *Mariner* was dependent upon it to maintain a proper inertial orientation so that the high-gain antenna could be oriented toward the Earth for the latter portions of the mission. Additionally, the attitude of the spacecraft had to be accurately known so the midcourse maneuver calculations could be made. The plan for achieving Canopus acquisition was to roll the spacecraft about the roll axis, i.e., the Sun-directed axis, and having the Canopus acquisition logic stop this roll search when an object of the proper intensity was observed. The initiation of this sequence occurred at 997 min after launch. The acquisition of Canopus could be verified from spacecraft telemetry indicating the brightness of the star being observed. Failure of the spacecraft to internally initiate the Canopus acquisition sequence can be counteracted by the ground command, DC-13, which will override the spacecraft logic, turn on the Canopus sensor, and initiate the roll search mode.

Before launch, a standard sequence of events during Canopus acquisition was formulated because it was anticipated that star identification might pose a serious problem, and because the celestial geometry near launch made it probable that the first star acquired would not be Canopus. The sequence permitted the spacecraft to acquire any object which fulfilled the Canopus sensor acquisition logic requirements (¼ to 8 times anticipated Canopus brightness and a negative error signal) and to become roll stabilized to that star. All data which might provide evidence as to the roll orientation of the spacecraft would then be gathered and evaluated. Based on this evaluation a recommendation for any command action would be formulated and then implemented during the next DSIF-11 (Goldstone) pass. This latter statement is based upon the recognition that should the Canopus sensor acquire a star determined not to be Canopus, there is a capability to roll-override this condition and force the spacecraft to go into a search mode for another star.

A further discussion follows of the procedures and plans developed for identifying the star, to which the spacecraft Canopus sensor was oriented. The only information from the Canopus sensor other than an error signal was a brightness measurement and an indication of the field-of-view of the sensor. The absolute calibration of the Canopus sensor for all the stars in the sky was not known, and the problem was further complicated because the brightness signal included the integrated background of stars in the field.

A map-matching technique was developed to identify objects seen by the Canopus sensor during the roll search mode. As an aid to establishing the validity of this map-matching technique, other corroborating information was used for the initial acquisition. This included information from a fixed wide-angle field-of-view Earth sensor. Magnetometer information observed during the magnetometer calibration roll sequence and low-gain antenna pattern variations were used to provide rate and position information. The magnetometer and antenna information was anticipated to be relatively crude and would be utilized with only a low weighting factor.

Fundamental to the map-matching was an *a priori* telemetry map of sensor brightness output vs clock angle (angle about the Sun line measured from Canopus). A reasonably sophisticated mathematical model of the Canopus sensor and the sky, including the Milky Way, was developed so that with trajectory information a computer program printed a map of the expected telemetry output of the brightness channel seen during roll search. This then was matched with an actual telemetry map to identify observed objects. Initially an extensive computer program was devised to process the telemetry data to produce the actual telemetry map. This was statistically correlated with the *a priori* map, the Earth-detector output, magnetometer data, and low-gain antenna data. Whenever an object was acquired subsequent to a roll search, a calculation was made of the probability that each acquirable object had been acquired. If the object was not Canopus, a roll-override command would have to be instituted to initiate roll search and another computer run would be made until Canopus was identified as the acquired object.

This computerized version of the map-matching technique was necessary due to the lack of knowledge of the sensor response and the innumerable objects and integrated background that would be seen by the sensor. Until the sensor was actually calibrated in flight, the uncertainties required that the best possible analysis techniques be prepared beforehand. Another map-matching technique was developed prior to launch to aid in the identification of the stars. It was recognized that the dependence upon this technique would have to wait for verification of the technique in flight. It consisted of a continuous strip-chart recorder employed to plot in real time the star brightness telemetry. An *a priori* map derived from the computer program was transcribed into a transparent overlay in the same scale as the real time telemetry plot, so that they could be instantaneously compared during the roll search.

f. Magnetometer roll calibration. To increase the value of the magnetometer experiment, the spacecraft was designed to roll at a fixed rate for many hours prior to Canopus acquisition to attain an in-flight calibration of the spacecraft's magnetic field as observed by the magnetometer sensor. This calibration was possible because the observed magnetometer signals in the near-Earth environment would have rotating components due to the external environment (the geomagnetic field) and fixed components due to the spacecraft environment. Knowledge of the geomagnetic field of the Earth permits determination of the spacecraft-induced magnetic field at the magnetometer sensor.

Every effort was made to achieve this objective; however, it was recognized that under certain conditions this could not be achieved. Not only was this calibration desirable from a scientific point of view, but from the operations point of view the inertial orientation of the spacecraft axes should be determinable. This fact would provide information to be utilized in the determination of the spacecraft attitude at the time that Canopus acquisition was initiated. It was recognized that the magnetometer calibration sequence might not be possible for every conceivable situation. The magnetometer calibration roll rate could only be provided under the conditions that Sun acquisition had been initiated properly by the spacecraft and that Sun acquisition had occurred. Should the Sun acquisition not be initiated automatically by the spacecraft, the utilization of a ground command would have to be made to initiate Sun acquisition. The utilization of this command at this time in the sequence would automatically cause the initiation of the Canopus acquisition sequence; this essentially would eliminate the interval of time during which the magnetometer calibration roll could be achieved. There was no capability aboard the spacecraft to command the magnetometer calibration roll. It was solely a function of the automatic features aboard the spacecraft.

g. Turn-off of video storage launch mode. The video storage subsystem record motor was energized during the launch phase to maintain tension on the mylar tape to preclude its spilling off the reel during the vibration environment of the launch phase. The mechanization for turning off this launch mode was to sense the first end-of-tape signal generated by the video storage subsystem subsequent to separation, and upon this signal open the relay connecting the record motor with the gyro three-phase power supply. Failure of this particular logic could be overridden by automatic or commanded turn-off of the gyros.

When it is considered that twelve passes over the video storage tape occur every hour that the machine is running, concern has been expressed about extensive periods of time when the video storage subsystem must be operative. Should the launch mode not stop the video storage subsystem shortly after separation, a ground command, DC-15, would be sent to turn off the gyro power supply and consequently the video storage subsystem. Upon subsequent energization of the gyros by DC-19, the video storage logic would have been locked out and the video storage subsystem would no longer operate until initiated at encounter. A constraint upon the use of the DC-15 command in this condition is that whenever the gyros are off the spacecraft roll rate cannot be controlled and a valid magnetometer calibration roll sequence is not possible. Consequently, it was desirable to postpone the turn-off of the video storage subsystem by ground command in the above-mentioned fashion until at least three or four hours of magnetometer data had been acquired. At the end of this interval the spacecraft would have passed through the geomagnetic field and its calibration data acquired would be minimal. Verification of video storage turn-off was possible by reviewing the power subsystem telemetry readings.

h. Removal of CC&S relay holding current. The function of the CC&S relay holding current is to prevent inadvertent actuation of relays within the CC&S during the launch phase and consequently initiate parts of the mission sequence at an inappropriate time. The relay holding current is normally removed at separation but will also be removed at those times when the gyro power is turned off. Failure to release the relay holding current permits no commands other than the midcourse commands to be successfully issued by the CC&S. Review of the spacecraft power subsystem telemetry readings allows verification that the holding current has indeed been released. Just when the gyros would be turned off to provide a backup to the automatic spacecraft function of releasing the holding current would be a function of what the spacecraft requirements were at that particular time. If the Sun acquisition had been initiated and the solar panels had been deployed, the relay holding current could be maintained on until some later time after the magnetometer calibration roll had been completed. However, if the backup capability provided by the CC&S launch commands were necessary, there would be no choice but to turn off the gyros as soon as reasonable to permit the spacecraft backup commands to be issued. Nominally this would occur prior to launch plus 93 min.

C. Trajectory Correction Maneuver Planning

The trajectory correction maneuver is perhaps the most demanding phase of the *Mariner* mission so far as SPAC is concerned. Normally the accuracy of injection of the spacecraft is such that a trajectory correction is required for a successful mission, but no automatic provision can be made for it in the design, since the correction parameters must necessarily be furnished from the ground some time after launch. Thus, the burden of performing the correction falls upon the SFO organization and FPAC and SPAC in particular. All inputs must be inserted into the spacecraft via ground command and the performance during all portions of the sequence must be monitored carefully, since at best a nonstandard midcourse means an unacceptable planetary miss distance, while at worst a nonstandard midcourse may result in total spacecraft failure.

The design of the *Mariner* Mars spacecraft includes a capability for termination of the midcourse maneuver before rocket motor ignition by the use of a ground command (DC-13), if an anomaly were to jeopardize the successful execution of a maneuver. When an abort is required, the DC-13 causes the attitude control and pyrotechnics subsystems to disregard the maneuver commands issued by the CC&S. After a DC-13, the CC&S continues counting and issuing its event register and timing telemetry signals to the data encoder until its maneuver clock is completely counted out and resets itself.

Barring gross failure a terminated sequence has no real effect, since the maneuver may be rescheduled and performed successfully regardless of the number of previous aborts. Thus the operations philosophy for the midcourse correction phase is determined by the abort capability inherent in the spacecraft design. Based upon knowledge of the design, ground test experience, and previous mission flight experience, it was fully expected that the maneuver would be executed successfully without deviation from the predetermined plan. An anomaly noted during the maneuver sequence has a potentially harmful effect and, therefore, requires that the maneuver be terminated and that the anomaly be evaluated to determine what corrective action, if any, should be taken.

Consideration was also given to ensuring that the spacecraft would enter the midcourse sequence in the proper state. Since subsystem and system testing prior to launch had demonstrated the possibility of logic state changes on the spacecraft due to electrical transient effects or to shock and vibration effects upon relays, it was determined that a reliance on non-verifiable, preset logic states was foolhardy. Thus two additional commands were added to the midcourse maneuver to ensure that the spacecraft was in a proper state to perform the midcourse maneuver. The first, DC-29, set the relays in each independent half of the pyrotechnics subsystem, arming the first set of motor burn start and stop squibs. The desirability of this is obvious if the results are considered, for example, of performing a maneuver on the first burn stop squibs but the second burn start squibs, a distinct possibility if relay contacts are subject to movement as the result of shock during boost. It was also deemed advisable to ensure the arming of the midcourse via the transmission of DC-14, the command to revoke DC-13, maneuver termination. Both commands were ultimately adopted as a part of the normal midcourse command sequence.

Just after Canopus acquisition the first Project Office midcourse meeting was held to establish the guidelines for midcourse operations. Subsequent to this first midcourse meeting the detailed preparation for the midcourse correction was initiated which led to a set of SPAC criteria for termination of a maneuver by DC-13, Table 27. This list was compiled by the SPAC Director with inputs from the divisions shown, and was approved by the Space Flight Operations Director and the Project Manager. The loss of roll attitude ultimately experienced during the first midcourse attempt was never considered for inclusion on the list because of the implicit confidence prevailing that the roll attitude would be maintained unless commanded to change or there was a failure in the roll channel circuitry. The abnormality itself was, however, covered implicitly by the Division 34 criteria. Each division listed in Table 27 was responsible to the SPAC Director for determining that its criteria were satisfied.

Table 27. SPAC criteria for use of DC-13

Division	Criteria
33	Abnormal RF performance of downlink which jeopardizes ability to analyze performance. Large-scale deck skips and resets. (Medium deck skip at autopilot turn-on would not be surprising. Gyro turn-on has caused deck skips occasionally during system test, but not during the flight.) Assuming a properly operating CC&S, Channel 220 (CC&S timing of events) indicating greater than 1 DN from expected value. Inability to decommutate high rate and position 0 of medium-rate data attributable to spacecraft malfunction.
34	Noncharacteristic gyro performance. Wrong turn polarity. Gyros do not turn on. Turn starts and/or stops at wrong time. Main or maneuver booster-regulator failure. Failure of 2.4-kc inverter.
35	None.
38	Oxidizer start cartridge pressure less than 300 psia.

Maneuver execution is affected if the expected impulse cannot be applied to the spacecraft. Although a velocity of 17 m/sec cannot be achieved if nitrogen pressure is less than 650 psi, and the rocket motor starting transient is affected by any drop in fuel pressure, it was decided that DC-13 would not be transmitted for either of these reasons because aborting the maneuver might negate any possibility of correcting the trajectory later, since either anomaly would probably indicate a steady degradation in capability which would be expected to continue.

On December 4, 1964, the first attempt to correct the trajectory of the spacecraft was initiated. The attempted correction was terminated because of an unexpected loss of roll attitude shortly after maneuver sequence initiation. Since this problem had not been specifically planned for, with, of course, the exception of the DC-13 maneuver termination provision, detailed preparation for such a problem was required before a second attempt could be made.

Analysis of the telemetry indicated that about 50 sec after the first maneuver sequence was initiated on the spacecraft, the spacecraft dropped lock on Canopus and went into a roll search mode, searching for another acquirable object. A DC-13 was transmitted to the spacecraft to abort the maneuver because it was determined that there was insufficient time to reacquire Canopus with successive DC-21s before the CC&S initiated the pitch-turn maneuver. After maneuver abortion, the spacecraft maneuver clock responded properly to the quantitative commands that had been stored in the on-board logic, demonstrating that the affected subsystems would respond properly during the next maneuver attempt.

It was decided that the second maneuver attempt could be scheduled for the next day (December 5, 1964) provided that a satisfactory sequence of events and procedures could be developed that would prevent a recurrence of roll search during the maneuver. The plan that was finally adopted used a spacecraft design feature that allowed the stopping of spacecraft roll search and the re-establishment of the correct roll attitude without rolling the spacecraft through almost 360 deg of arc. In the event of loss of roll acquisition the spacecraft roll gyro placed in the inertial (rate-integrating) mode via DC-18 to *freeze* its position in inertial space, then the spacecraft would be backed up with successive DC-18s until the Canopus sensor plane was properly oriented. It was felt that inadvertent roll search was associated with gyro turn-on so that before pitch-turn start the spacecraft roll attitude would be referenced to Canopus again using a DC-19.

Although DC-18 had not been used previously in flight, it would be used without prior test experience if the spacecraft went into roll search. It was decided not to test the DC-18 method in order to reduce the number of times the Canopus sensor was turned on and off during the mission, since power transients are often the cause of electronic equipment failures. Nor was there any direct evidence that showed such a test was warranted. If DC-18 did not prove effective, DC-13 could be used to abort the maneuver again.

Table 5 lists the DC-18 maneuver sequence available for use during the second midcourse maneuver attempt. The sequence incorporates SFO-based constraints needed because of the special sequence execution conditions.

The maneuver plan recognized that timing of the DC-18 sequence was critical, since up to eight DC-18s might be needed to compensate for the angular roll displacement gained before the spacecraft received the first DC-18 that would stop spacecraft motion. Table 28 shows the tentative time sequence evolved for the use of DC-18 during second maneuver attempt, if such action proved necessary.

Table 28. DC-18 time sequence

Time, sec	Event
R = 0	Spacecraft starts roll search
R + 10	SPAC receiver telemetry indication of roll search
R + 20	SPAC requests transmission of DC-18
R + 35	DSIF 11 notified to transmit DC-18
R + 80	DC-18 received by spacecraft
R + 275	DSIF 11 ready to transmit series of DC-18s
R + 755	DSIF 11 transmits DC-18 No. 8
R + 800	DC-18 No. 8 received by spacecraft
R + 860	SPAC requests DC-19

D. Science Cover Deployment Planning

1. General

On February 11, 1965, a ground command initiated a science cover deployment exercise with the threefold objectives to:

1. Deploy the science cover

2. Preposition the scan platform to the optimum encounter position, and
3. Turn off the battery charger and enable the boost mode.

The exercise was carried out over the DSIF 11 tracking station. A total of twelve ground commands were transmitted to the spacecraft. All were received and executed normally. All objectives of the exercise were fully achieved, and the spacecraft was returned to the cruise state without difficulty.

The planning which preceded the science cover deployment exercise was provided primarily by the *Mariner IV* EPWG as a part of their chartered activities of formulating plans and making those preparations required to fully exploit the planetary encounter.

a. Preliminary analysis. During the design phase of the *Mariner* Mars 1964 project it was decided that the mechanization of the planetary experiments and associated data conditioning equipment should revolve about the simplest dormant design possible. Thus the planetary instruments were designed to remain in the stowed position with power off from launch until several hours prior to closest approach to Mars. The only exception to this was the scan platform latch, released at solar panel deployment, which protected the platform and actuator during the launch. While this approach had a number of advantages, it did require that the encounter equipment operate normally after an 8-mo storage in space, it did not allow for ground monitoring of instrument status, and it did not allow a great deal of corrective action capability by the ground in the event that it was discovered at encounter power turn-on that a failure had occurred.

As a result, the question of ground-commanded encounter exercises was investigated by the EPWG to determine if there existed any significant advantage to performing all, or any part of, an encounter sequence during the cruise portion of the mission. These activities were undertaken during the months of December 1964 and January 1965, since, were sufficient reason to perform some command exercise uncovered, it might be desirable to do so while the spacecraft still had low-gain antenna capability, both uplink and downlink.

The investigation by the EPWG showed that there were a number of advantages to performing a premature science cover deployment. The advantages are listed in Table 29.

Table 29. Advantages of premature science cover deployment

Item	Advantage
1	Reduced risk of Canopus loss during encounter due to cover deployment: a. No data loss with roll loss prior to MT-5 b. Confirmed command subsystem availability c. Not a time-critical operation
2	Minimized risk of science cover deployment failure during storage
3	Permitted assessment of encounter equipment
4	Turned battery charger off and enabled boost mode
5	Exercised procedures, personnel, and facilities required for encounter
6	Positioned scan platform for encounter
7	Precluded possibility of premature NAA from reflected light off cover
8	Prevented possibility of encounter data loss due to science calibration
9	Provided baseline data for assessing encounter performance
10	In the event of some malfunction, corrective action could be attempted

Whenever commands are sent to a spacecraft to change its electrical or mechanical configuration a number of risks are involved. Table 30 lists the risks involved in the science cover deployment.

b. Potential failure modes. Particular interest was focused on those failure modes to which the spacecraft was exposed by deploying the cover early, and to which it would not otherwise have been subject.

Table 30. Summary of risks

Item	Risk
1	Scan subsystem malfunction
2	Loss-of-command lock
3	Command subsystem malfunction
4	Power subsystem malfunction
5	Micrometeorite damage to TV
6	Television shutter left open
7	Video storage malfunction

Scan subsystem malfunction. The use of any sequence exercising DC-24 exposed the scan subsystem to an unnecessary failure mode. Before the exercise, the relay which interrupts power to the scan subsystem upon receipt of DC-24, DC-16, NAA, or television planet-in-view (PIV) was in the reset position; i.e., the circuit was completed. Sending DC-24 or DC-16 would set this relay, which must then be reset again by logic circuitry within the scan subsystem at encounter. Should the logic circuit fail to work, assuming DC-24 or DC-16 had been sent during the early deployment, the subsystem would be exposed to a failure mode to which it would not otherwise be susceptible. On the other hand, if the relay had failed to reset, but the scan platform had been stopped at the proper clock angle, this would turn out to be quite favorable. At that time the uncertainty in θ was approximately ± 3 deg. Providing that uncertainty and the nominal value of θ did not change appreciably, this could turn out to be a risk that is actually desirable to take.

Loss-of-command lock. There was no way to determine the command subsystem lock status during Data Mode 3. Concern was expressed over using DC-3 especially since at least 2 hr of Mode 3 Data would be required to accurately determine the TV frame time[18]. Blind command lockup tests were run at the DSIF stations which demonstrated that if command lock was lost during Data Mode 3, it could be reestablished.

Command subsystem malfunctions. Failures of the command subsystem come in two varieties:

1. A command sent to the spacecraft could be wrongly interpreted by the command subsystem. A review of all possible commands, however, showed that none are irrevocable or catastrophic with the possible exception of DC-9 which would have exposed us to the possibility of a receiver lockup associated with the ranging equipment. The probability of the command subsystem wrongly interpreting a command is 0.97×10^{-8} at threshold. The probability of interpreting any command sent as DC-9 other than DC-9 itself is 0.33×10^{-9} at threshold. The command link performance margin at the contemplated time of the early cover deployment was predicted to be 10 db above the sum of the negative tolerances. The probability of executing a wrongly interpreted command was zero for all practical considerations.

2. A command which was required to complete the cover deployment sequence and which was unavailable because of a component failure would have been troublesome. There were three such commands: DC-2, DC-26, and DC-28. DC-26 and DC-28, which were required to recondition the encounter equipment after the cover deployment, would be demonstrated to have complete functional integrity prior to committing to the cover deployment operation. DC-2, which was required to recondition the cruise science, would be functionally checked except for the integrity of approximately eleven components when it was transmitted as the second command in sequence. The probability that these particular components would have failed since they were last checked during the countdown was 1.2×10^{-4}. The effect of a failure in one of these components would have resulted in being unable to have cruise science on without also having encounter science on. The consequence of leaving the encounter science on would be a wearout failure of the TV shutter which was estimated to have a half-life of 60 hr. A choice could be made between cruise science or TV. Such a failure would not affect the alternate cruise science turn-on which occurs at MT-7 or DC-25.

[18]This was required to determine the best transmission time for DC-26 from a TV shuttering point of view.

Power subsystem malfunction. Failures in the Power Distribution Subassembly 4A11 have effects similar to those described above. There were thirteen components whose failure could not be determined by the verification sequence and which would have the same effect as a failure of DC-2. There were six such components whose failure would result in the inability to turn off encounter science. There were also six such components whose failure would leave the 2.4-kc power on to the video storage subsystem. The probability that any of these components would have failed since they were last known to be working properly was 1.6×10^{-3}.

Micrometeorite damage to TV. With the cover open, there existed the possibility of micrometeorite erosion or possibly a cosmic ray deterioration of the encounter optics. This phenomenon, which also affects the Canopus sensor, is not thought to be a problem.

Television shutter left open. An unusual combination of circumstances could have led to a possible problem as follows:

1. After the cover was deployed and encounter power was turned off, a problem could occur if the TV shutter were left open.

2. The spacecraft, during the mission, lost lock on the Sun-line (but managed to recover).
3. While either losing lock, or in the process of reacquiring lock, the Sun passed through the field-of-view of the camera[19].
4. Enough energy could be absorbed by the vidicon surface to destroy it.

The risk of leaving the shutter open, when DC-26 turns the Encounter Science power off, was eliminated by including DC-3 in the sequence to obtain the TV shuttering times. This allowed DC-26 to be sent so that it would arrive at the spacecraft 6 sec after a TV shutter. The shutter is a logically controlled function occurring once every 48 sec. The probability of causing a shutter pulse and therefore leaving the shutter open is extremely low if power is turned off within the first 10 sec following shutter, and reaches approximately one chance in a hundred (10^{-2}) if power is turned off within 2 sec preceding the next shutter. For 0.2 sec, out of 48, the probability is one out of one, for practical purposes, of leaving the shutter open simply because during this period of time it is normally open.

Video storage subsystem malfunction. The Video Storage Subsystem end-of-tape circuitry had the undesirable characteristic that it produced a short (approximately 1 ms) pulse on power turn-off. This pulse has sufficient energy to forward bias transistor Q2, the video storage subsystem channel change relay driver transistor. However, this pulse does not hold Q2 on long enough to ensure reliable relay operation.

With this low energy drive pulse, relay K1 may behave in one of the five following fashions with the noted results:

1. The relay may properly function in that both sets of armature contacts change from one set of contacts to the other. This mode of operation, which occurs approximately 90% of the time, causes the video storage subsystem to effectively change tracks.
2. The relay armature may fail to operate in that the armature contacts remain in their initial condition. This mode of operation, occurring approximately 2% of the time, produces no change in the video storage subsystem.
3. The relay armature making contacts with terminals four and five may function properly while the armature making contact with terminals one and six may end up in a neutral condition wherein the armature does not make contact with either terminal. In this hung-up state, which occurs approximately 4% of the time, both record heads are energized and both playback amplifiers are gated off. Consequently, if any data is sent to the video storage subsystem, this data will be recorded on both tracks; however, since the playback amplifiers are gated off, data cannot be played back. If the channel change relay K1 is in this state when an encounter sequence is begun, the operation of the video storage subsystem will not be affected. During the first tape pass, the first 10½ pictures will be recorded on both tracks. Since the steering contacts four and five are still operable, relay K1 will operate normally at the first end-of-tape, thereby correcting the relay hang-up. During the second tape pass, the next 10½ pictures will be recorded over the first 10½ pictures on one of the tracks, thereby resulting in the storage of 21 complete pictures. Picture playback would be normal. In summary, this type of hang-up does not affect the operation of the system for a normal encounter sequence.
4. The relay armature making contact with terminals one and six may function properly while the armature making contact with terminals four and five may hang up in a neutral state. In this state, which occurs approximately 4% of the time, the video storage subsystem effectively becomes a one-track machine since the channel change relay cannot be thrown by transistor Q2 when one of the steering contacts is not making contact. If 21 pictures are recorded, the second 10½ pictures would be recorded over the first 10½ pictures on the same track. Playback, which would consist of the last 10½ pictures, would also be from this track.
5. Both relay armatures become hung-up. In this state, which was not observed in testing, any recorded data is recorded on both tracks. However, this data cannot be played back since the playback amplifiers are gated off.

During video storage subsystem laboratory and spacecraft testing, relay hang-ups were avoided by sending manual OSE track change commands 1 sec before power was turned off. These commands depleted the energy in capacitors C1 and C2, the

[19]The field of view is approximately 1.05 deg square and is at an angle 120 deg with the −Z axis.

relay energy storage capacitors. With capacitors C1 and C2 discharged, a pulse turning Q2 on had no effect on relay K1 since there was no energy available to drive the K1 coils.

Video storage subsystem electronics failure risk. The risk of incurring a failure in the video storage subsystem electronics by leaving the 2.4-kc power on for the remainder of the mission is as follows:

1. Any small spacecraft power transients during the rest of the mission might have caused the video storage subsystem count-two circuitry to lock-up in the following manner. A transient turns on the monopulse reset generator clearing the count-two-and-stop circuitry to zero. Later transients would generate two counts into the count-two-and-stop circuitry, thereby inhibiting the record sequence.
2. Leaving the video storage subsystem power on would add unnecessary operating lifetime to the video storage subsystem electronics. Unnecessary operating lifetime is considered highly undesirable.

Therefore, these risks were removed by including a DC-28 in the sequence generated to turn off the video storage subsystem 2.4-kc electronics.

2. Considerations

a. Optimum calendar date. The best time to perform the exercise appeared to be prior to the March 5 CC&S MT-5, which switched the transmitter to the high-gain directional antenna. The reasons which made this appear to be the best time were:

1. If the spacecraft had lost roll control as a result of reflected light from dust particles knocked free by the impact of the science cover deployment, it would be very desirable to be able to receive data during the resulting roll. Without telemetry it would have been impossible to evaluate the spacecraft performance until Canopus was again acquired. This would have required a DC-19 followed by DC-21s periodically until telemetry reappeared and Canopus was reacquired. Performing the exercise before CC&S MT-5 would allow the desired exercise to be completed even with the loss of roll control, and later would allow Canopus to be reacquired, since data would not be lost during the roll while transmitting over the low-gain antenna in the near-Earth vicinity.
2. If the science cover would not deploy, it would be desirable to determine this early so that some corrective measures could be employed. For example, if the cover had not deployed, it might be desired to turn the spacecraft so that solar heat is applied directly on the latching mechanism. The earlier this could be performed, the greater the amount of heating which would be possible on a Mars transfer trajectory. The spacecraft battery, command subsystems and CC&S subsystems would all be required for such an exercise, and all appeared to function normally at that time.
3. All subsystems appeared to be functioning normally in early February. Any of them, of course, could have failed before encounter. It would be very desirable, therefore, to put the spacecraft in the most desirable position possible for the encounter itself by using the backup commands at that time, because command was working, instead of waiting until encounter.

Some consideration was given to performing the exercise in late March or early April to allow more time to study the problems and trade-offs more closely. It was felt that very little more could be learned and it was not worth the risk of losing spacecraft telemetry in case of a loss of roll control.

b. Television value. Before the trajectory-correction maneuver, the trajectory-dependent value curves for the television experiment were based on the following considerations:

1. Resolution—degrades with increasing picture-taking distance
2. Blur degradation—occurs with decreasing distance
3. Frame size—decreases with decreasing distance
4. Terminator and near-terminator coverage—a regular function of θ
5. Surface featurě and bright-dark transition coverage—an irregular function of θ and of arrival time.

After the trajectory-correction and before the science cover deployment exercise, the *Mariner IV* trajectory was rather well defined. This made it possible to more closely examine the relative TV value in terms of some parameters which were still variable. It was determined that the relative TV value changed as a function of scan clock angle at which the scan head was positioned for the pass over the planet.

Figure 52 shows the scan geometry in relation to the trajectory near the planet. The spacecraft is located at the center of the coordinate system with the direction to the Sun pointing vertically upward. The scan subsystem is located 120 deg from the Sun direction. The planet denoted orbital plane represents the relative motion of the planet past the spacecraft.

The scan subsystem has associated with it a wide-angle sensor and a narrow-angle sensor, with 25-deg half-angle and 1¼- by 2½-deg fields-of-view, respectively. The scan subsystem is energized at Mars and the scan rotates about the Sun-line between 296-deg and 116-deg clock angle. Clock angle is measured in the plane perpendicular to the direction to the Sun with 0-deg clock at the projection of Canopus onto the clock angle plane. When the planet enters the 25-deg half-angle field-of-view of the wide-angle sensor far enough to trigger the WAA logic, the scan begins tracking the planet. The tracking occurs by setting the scan clock angle equal to the planet clock angle plus a small bias angle which is a function of distance from spacecraft to planet. When the planet enters the field-of-view of the narrow-angle sensor (narrow-angle Mars gate, NAMG), active tracking ceases and the picture-taking sequence is triggered. Following the taking of 21 TV pictures, the sequence is ended.

In Fig. 52 it will be noted that the orbit plane is inclined by about 50 to 55 deg to the Sun-line. This results in rather non-optimum coverage for the TV subsystem unless biasing is included. This can be seen as follows: relative to the two sensors, the active tracking of the planet effectively cancels any motion of the planet through the sensor in the clock angle direction. This means that during tracking the planet changes in cone angle only in the sensor, or climbs vertically up the center line of the sensor. However, when active tracking stops, the scan subsystem can no longer remove the clock angle motion of the planet through the sensor. The planet, therefore, appears at NAA to change direction and pass through the sensor at a skewed angle (equal to the angle the trajectory is inclined to the Sun), with the result that the optic axis does not pass through the center of the planet. This is illustrated in Fig. 53, the view of the planet as seen by the sensors; in any particular case, the scan trace will be parallel with the one shown. The traces will begin near the bright limb, and will be approximately the same length as the one shown. Hence, to achieve near-optimum TV coverage, a bias was built into the wide-angle sensor which effectively displaced the planet in clock angle. However, the bias that could be built into the scan sensor, while adequate to provide good pictures, was insufficient to bias the scan far enough to carry it across the center of the planet.

When positioning the scan by ground command, the scan axis could have been located at the optimal clock angle for TV pictures plus or minus the accuracy in positioning process. In order to determine the allowable tolerances in positioning the scan and in the trajectory

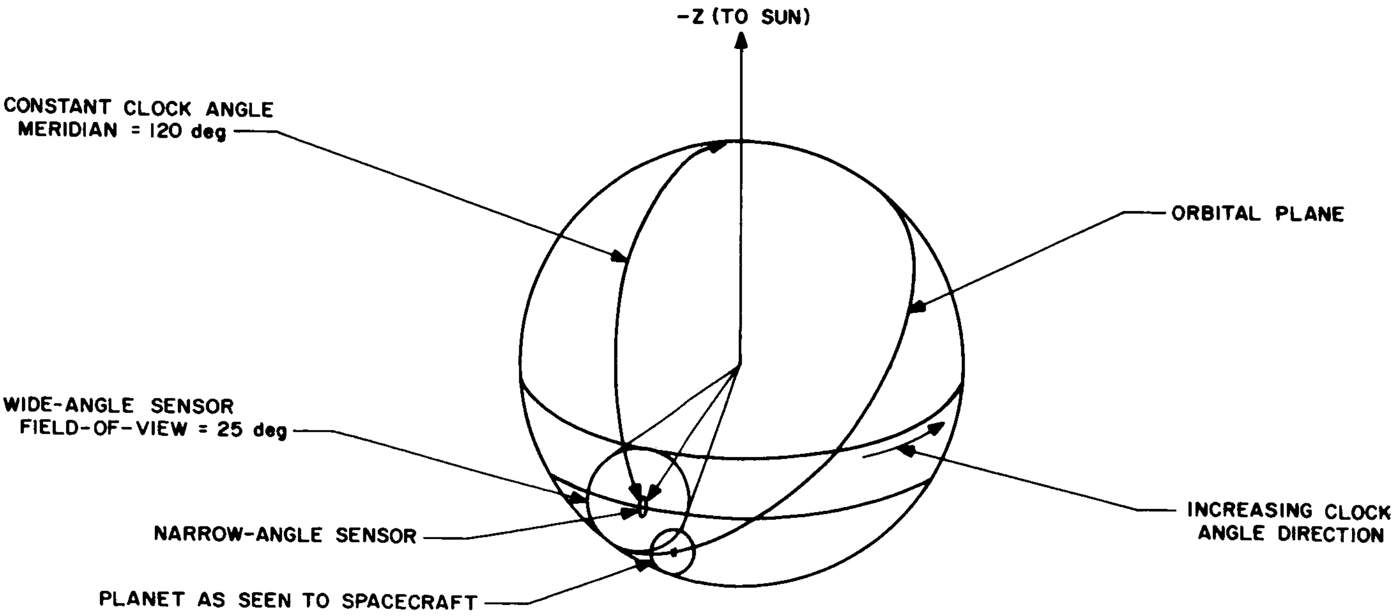


Fig. 52. Scan subsystem trajectory geometry

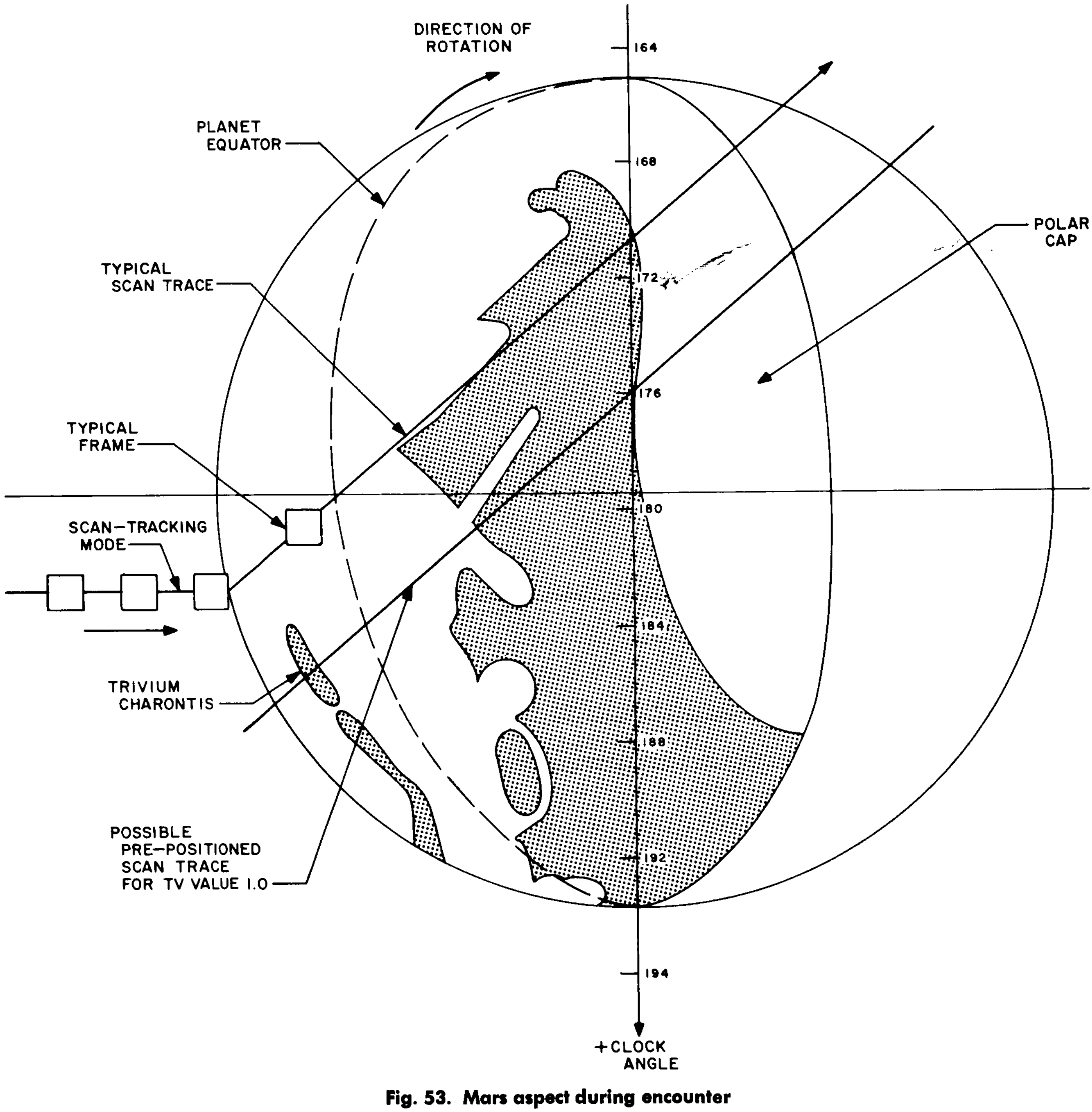


Fig. 53. Mars aspect during encounter

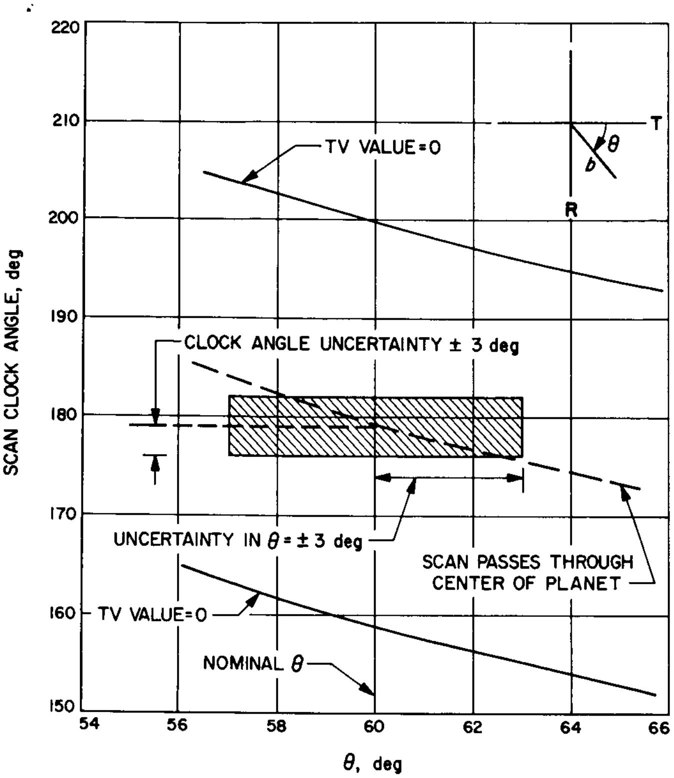


Fig. 54. Scan subsystem clock angle vs θ

uncertainty, Fig. 54 was generated illustrating the clock angle range for a given θ[20] that resulted in the scan being able to see the planet. The box shown cross-hatched in the center of the figure pointed out the region in which it was most probable that the scan subsystem would fall, assuming a $\pm$3-deg uncertainty in positioning the scan and a $\pm$3-deg uncertainty in the orbit θ.

In addition, it was important to know how the uncertainty in trajectory would decrease with time. Prelaunch studies indicated that the uncertainty in the orbit dropped very slowly through the cruise phase. When nearing the planet, a large decrease in the uncertainty in the orbit was expected starting 2–4 hr before closest approach and continuing through encounter. The conclusion was that 1 or 2 days prior to encounter the uncertainty in θ was unlikely to be much smaller than it was on February 11, 1965.

θ is the angle between the **T** vector and the **B** vector in the **RT** plane, where the **T** vector lies in the plane of the ecliptic and is perpendicular to the incoming asymptote. The **B** vector is the vector from the center of the planet (center of the coordinate system) to the point where the incoming asymptote intersects the **RT** plane.

The television value vs scan clock angle is given in Fig. 55. This curve was determined by the experimenters and was based on the following considerations:

1. Coverage of the sub-solar region – shading here is dependent largely on albedo and color differences, not on surface slopes
2. Coverage of a maximum number of bright-dark regions
3. Polar cap coverage
4. Avoidance of near-terminator emission angles ε of greater than 60 deg
5. Maximization of the number of usable pictures – i.e., minimization of the number made beyond the terminator
6. Coverage of Trivium Charontis – an intrinsically interesting region

The trajectory that had been attained by *Mariner IV* after the midcourse was assigned a television value number of 0.9 or greater prior to midcourse. As indicated in Fig. 55, the *Mariner IV* value was about 0.7, based on scan-bias data. Other than the new ground rules, two factors combined to bring the *Mariner IV* value below nominal. First, there was an adverse mechanical clock angle bias between the scan and television axes. Second, the arrival time was 36 min earlier than nominal. The first factor decreased the likelihood of obtaining polar cap coverage. The two factors combined to eliminate

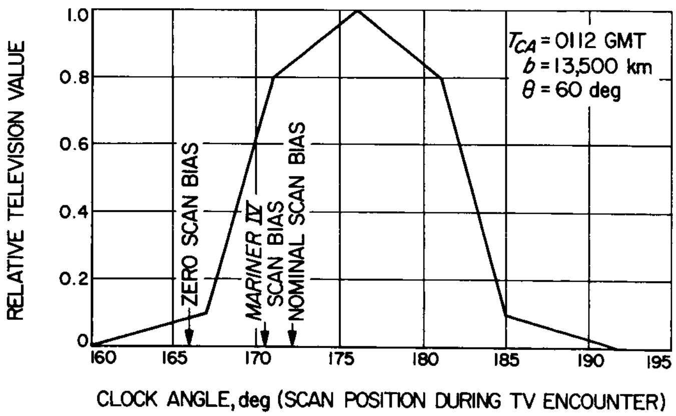


Fig. 55. Relative television values of various scan platform positions

[20] $\theta = \arctan \dfrac{\mathbf{B \cdot R}}{\mathbf{B \cdot T}}$

coverage of Trivium Charontis, which would otherwise have been obtained; hence, fewer light-dark transitions.

The value contour presented here was based on the assumption that a full set of pictures would be received at Mars encounter. If, due to an early science cover deployment and associated circumstances, only the second half of the set were received, the value of the television experiment would have been degraded by a factor of perhaps ten. Several factors entered into this evaluation including:

1. Two or more of the second ten or eleven pictures would be beyond the terminator in any case
2. No major desert region would be covered
3. Only less-distinct light-dark transitions would be covered
4. Aerial photo-interpretation in unknown territory is a difficult proposition at best; having only half the pictures greatly reduces the data sample
5. One goal of this mission was to provide knowledge to support future photographic missions; if the brightness in the sub-solar region were not sampled, it would not be possible to draw up even a relative photometric curve for Mars based on the received data
6. The best spectral-band information would be expected from the sub-solar region.
7. The trajectory and scan-position tolerances became crucial: since the scan trace would be triggered by narrow-angle acquisition, it would begin near the limb and it would become very important that it cross a wide part of the apparent disk; the number of pictures lost beyond the terminator otherwise would become a large fraction of the total number obtained.

In short, the loss of the first 11 pictures would have been very serious; however, while the relative value of the remaining pictures would have been low, their absolute value, incalculable before the fact, would have been very great.

c. Temperature control. The early science cover deployment would produce approximately an overall 4°F drop in the temperature of equipments which were thermally slaved to the bus. This was small enough not to cause concern. No change in temperature would have been expected following a normal MT-7 science cover deployment because heat dissipated by the encounter equipment nearly balances loss from the open cover. However if Channels 418 and 437, which are the TV and SPITS temperatures monitored in the instruments themselves, were sampled shortly after the cover was opened, the accompanying thermal transient would have verified the event. After the cover deployment, the TV and SPITS temperatures, because of the lower value, would positively verify cover deployment.

d. Ground command execution timing. It was obvious that understanding of the timing parameters and accuracy in command timing would be necessary to position the TV at the optimum angle.

It was determined by analysis and by testing that the ground command OSE took 53.25 + 2.24 − 0.00 sec to process a 26-bit command word in the *normal mode.* By using the automatic start feature of the OSE, the 0.2 sec reaction time of the operator was removed from the positive tolerance. It was further discovered that the paper tape tolerance of 1.00 sec could be removed by going to Mode 2 verify and then backspacing. This put the first bit of the command word adjacent to the paper tape read-head. This leaves a tolerance from 0 to 1.04 sec, which is due to the lack of synchronism between the station clocks and the subcarrier oscillator in the ground command OSE which drives the bit sync generator. For any direct command (26 bits long) only the first 10 bits are required before the command is executed in the spacecraft. Therefore, 16 sec can be subtracted (1 bit/sec) from the 53.25 sec. A ground command OSE process time of 37.3 sec was used, understanding that a 0 to 1.04 sec tolerance with uniform distribution existed on this time.

The run down time of the scan head was found to be approximately 0.75 deg or two data numbers in the scan position readout.

Armed with these data, tests were run with the PTM to demonstrate and verify the timing techniques. These tests proved the feasibility and, with minor corrections to the procedure, the scan head could be positioned within one data number (±0.36 deg) consistently. Tests were run from the SPAC area at SFOF with the PTM to develop an understanding of the two-way transmission time also. The method used to determine the transmission time for the command was based on plotting several cycles of the scan position on graph paper and predicting ahead based on these data when the sensor would be at the desired location. The two-way transmission time and the 37.3 sec ground command OSE time were then subtracted from that projected time, to arrive at the time to begin transmitting DC-24.

3. Deployment Sequence Development

Six possible command sequences were developed by the EPWG for use in the science cover deployment exercise. The total exercise was broken down into six basic yes/no type decisions, Fig. 56. The outcome of each of these decisions determined the course of action to be pursued. The six plans ranged in difficulty from Plan 1, which required no command action, to Plan 6, which required 18 direct commands to: 1) turn off the battery charger, 2) perform the encounter sequence, 3) playback, and 4) return to the cruise configuration. Table 31 lists the effects of all the commands considered. Plan 4, Fig. 3, was adopted by the *Mariner* Project Manager.

Table 31. Effects of commands considered for the science cover deployment

Phase	Command	Effect
Command verification	DC-3	The command DC-3 is the first command in the sequence, so that it will verify the correct operation of the command detector and allow DC-2, the next command, to have a measurable effect. DC-3 will switch the Data Encoder to Data Mode 3 and place an event in Register No. 4 which will not be detected until the Data Encoder is back in Mode 2.
	DC-2	DC-2 is issued as the second command to verify that it will be processed properly by the spacecraft command subsystem. This command will switch the Data Encoder to Mode 2, which is the proper data format for the cruise portion of the mission. The command DC-2 also is the only means of turning on only the cruise science for the remainder of the cruise portion of the mission. The cruise-science portion of the command operation can only be inferred by the Data Encoder switch until after DC-26.
	DC-26	The command DC-26 is issued to verify it will be processed by the spacecraft command subsystem. cruise science prior to issuing the command proper response to DC-26 under these conditions, provides almost complete functional verification. It is absolutely necessary that this command be operative, since it is the only means of turning off the encounter science before the Mars encounter.
	DC-2	The command DC-2 is issued next to turn on cruise science prior to issuing the command DC-25 in the initiate-encounter phase. Using DC-2 to turn on the cruise science will minimize the transient to the spacecraft power subsystem upon execution of DC-25.
	DC-28	The command DC-28 is issued to verify that it will be processed by the command subsystem. Battery-charger-on, the proper response to DC-28 under these conditions, provides almost complete functional verification. The command DC-28 is required to turn off the 2.4-kc power to the tape subsystem subsequent to DC-25.
Initiate encounter phase	DC-25	The command DC-25 will deploy the cover and initiate the encounter sequence. It turns on both the encounter science and the cruise science and turns off the battery charger. It also starts the scan-search sequence. The command DC-25 is required in order to record TV pictures or to initiate the scan-search.
Inhibit scan phase	DC-24	DC-24 may be sent to stop the scan platform (otherwise it will be stopped upon receipt of DC-26). DC-24 sets a relay within the scan subsystem which must then be reset at encounter for normal scan subsystem operation. If the platform can actually be stopped at the desired angle, DC-24 would be the better command to use since the double constraint (scan platform timing and TV shutter timing) on DC-26 is then removed.
Inhibit recording phase	DC-28	One of the plans offers the option of sending DC-28 here in the sequence. The command DC-28 turns off the 2.4-kc power to the video storage subsystem electronics, inhibiting the possibility of recording or playing back. The command will also turn the battery charger back on, disenabling the boost mode.
Science format phase	DC-3	Two of the plans offer the option of sending DC-3. DC-3 places the Data Encoder in Mode 3 which provides additional science data at the expense of all engineering telemetry. The additional data would provide information to assess the performance of the TV subsystem, to permit timing of DC-26 to reduce the probability of leaving the shutter open. DC-3 is not required to execute a tape playback or record sequence.
Initial playback phase	DC-4	The command DC-4 will place the recorder in the playback mode. If the tape recorder is to be exercised at all prior to encounter it should be placed in the playback mode prior to performing a record sequence. In playback the tape transport moves more slowly (by a factor of 1200:1) and greater torque is available; this minimizes any undesirable effects in the event that the tape pack has become stuck.

Table 31. Effects of commands considered for the science cover deployment (cont'd)

Phase	Command	Effect
	DC-2	The command DC-2 will shut the recorder off and restore Data Mode 2. This command should be sent after verifying correct playback mode.
Record phase	DC-16	If it is desired to exercise the tape recorder in the record mode, DC-16 must be sent. After recording 21 pictures, the subsystem will shut itself off at the second EOT or the DAS end of Picture 22.
Terminate encounter phase	DC-26	The command DC-26 may be sent to terminate the encounter sequence. This command will turn off encounter science and cruise science. If no playback of the recorder was planned the command would be postponed to the cruise condition phase.
Main playback phase	DC-4	If a playback of the tape recorder is desired following shutdown of the encounter science it could be initiated at this point with DC-4. Since there would be no data on the beginning of the tape (space before Picture 1) the Mode 4/1 logic would format the data to Mode 1.
	DC-22	After a short interval DC-22 could be sent which would switch tracks on the machine. Data would then be available from the middle of the eleventh picture recorded in the Record Phase. This would provide a functional check on the entire TV data link. A total of about 40 or 50 min, in the playback mode, could be permitted with no loss of recording capacity at encounter.
	DC-2	The command DC-2 would terminate the record phase and restore Data Mode 2 at this point. If the record and playback modes of operation were skipped and the Data Encoder is in Data Mode 3 the command DC-2 must be sent to switch to Data Mode 2. If this is not done all telemetry from the spacecraft would stop from DC-26 to DC-2.
Cruise condition phase	DC-28	In order to finally condition the spacecraft for the cruise mode, it is necessary to turn off the tape-recorder 2.4-kc power. This is accomplished by sending DC-28. This command also turns on the battery charger. Since the desire is to end this exercise with the battery charger turned off this command must be sent before DC-26.
	DC-26	The command DC-26 will turn off the battery charger, enabling the boost mode (share protector). For those sequences where the encounter science is still on this command also turns off all science (encounter and cruise) which makes the last command necessary.
	DC-2	The last command, DC-2, will turn the cruise science instruments back on and establish the required spacecraft cruise condition.

4. Contingency Plans

Plans were developed before the exercise which laid out the courses of action to follow, after careful discussion, for each envisioned exercise-induced anomalous behavior of the spacecraft. These plans are shown in Table 32.

E. Encounter Planning

1. General

The basic plan of the encounter sequence was to use ground commands to pre-position the planetary scan platform, to back up the initiation of the television record sequence, and to back up the turn-off of the video storage record motor to assure the attainment of television pictures of Mars.

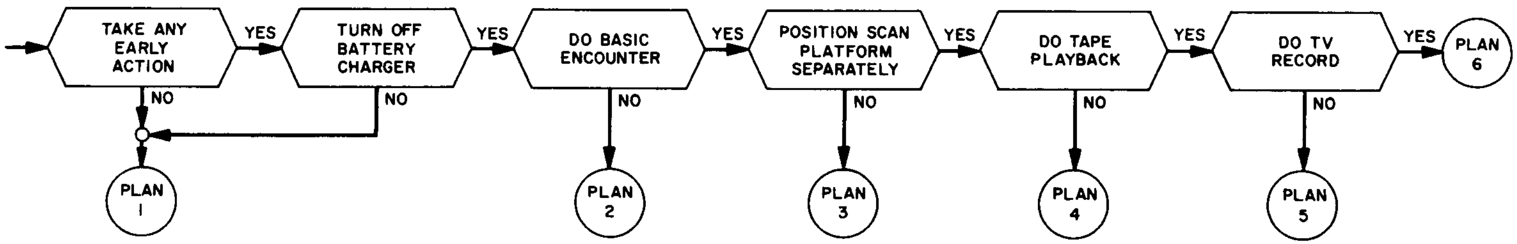
a. Initial conditions. The spacecraft was in the cruise configuration at the start of the encounter phase. No special preconditioning was required to prepare the spacecraft for the initiation of the encounter phase by ground command DC-25. Table 33 shows the configuration of all spacecraft subsystems at the start of the encounter phase.

b. Alternatives.

Received via high-gain antenna. The *Mariner* flight sequence was originally based on the transfer of the spacecraft receiver to the high-gain antenna before May 1, 1965 because the spacecraft-received carrier-power would have decreased to such an extent that reliable Earth-to-spacecraft (uplink) communications could not be maintained. However, by mid-February it became apparent that previously predicted interference patterns between the high- and low-gain antennas were affecting the uplink power level. Careful plotting of the interference pattern, Fig. 7, allowed a confident prediction that the spacecraft uplink power would be reinforced by the interference pattern, allowing reliable command capability with the DSIF 10-kw transmitters.

The spacecraft could have been operated in the receive via high-gain antenna mode if required; however, if the spacecraft lost roll control while receiving via the high-gain antenna, up to 133 hr could elapse before on-board logic would switch the receiver to the low-gain antenna and allow the receipt of ground commands.

Alternate methods of roll control. As shown in Table 33, the spacecraft would be in the DC-15 roll-control mode at the start of encounter. Two other modes of roll control were available: DC-19 and DC-18.



PLAN NUMBER	PLAYBACK	RECORD	PLAYBACK	COMMAND VERIFICATION PHASE	INITIATE ENCOUNTER PHASE	INHIBIT SCAN PHASE	INHIBIT RECORDING PHASE	SCIENCE FORMAT PHASE	INITIAL PLAYBACK PHASE	RECORD PHASE	TERMINATE ENCOUNTER PHASE	MAIN PLAYBACK PHASE	CRUISE CONDITION PHASE	COMMENTS
				DC-3 DC-2 DC-26 DC-2 DC-28	DC-25	DC-24	DC-28	DC-3	DC-4 DC-2	DC-16	DC-26	DC-4 DC-22 DC-2	DC-28 DC-26 DC-2	
PLAN 1														TAKE NO ACTION
PLAN 2				1 2									3 4	BATTERY CHARGER OFF
PLAN 3				1 2 3 4 5	6								7 8 9	BATTERY CHARGER OFF BASIC ENCOUNTER
PLAN 4				1 2 3 4 5	6	7	8	9				10	11 12	BATTERY CHARGER OFF BASIC ENCOUNTER SCAN PLATFORM POSITIONED TV SHUTTER CLOSED
PLAN 5			X	1 2 3 4 5	6						7	8 9 10	11 12 13	BATTERY CHARGER OFF BASIC ENCOUNTER TAPE PLAYBACK
PLAN 6	X	X	X	1 2 3 4 5	6	7		8	9 10	11	12	13 14 15	16 17 18	BATTERY CHARGER OFF BASIC ENCOUNTER TAPE RECORD AND PLAYBACK

Fig. 56. Science cover deployment decision diagram

Table 32. Emergency action summary

Single failures or problems	Indication	Action
1 DC-3 Wrong data mode	Data Mode 1	Do not know if this is Mode 4/1 or Mode 1 as video storage subsystem is off. Send DC-2 and stop exercise.
	Data Mode 2	If event in Register No. 4, stop exercise. If no event, check ground command OSE and decide on retransmission of DC-3.
2 DC-2 Wrong data mode	Data Mode 1	Do not know if this is Mode 4/1 or Mode 1 as video storage subsystem is off. Send DC-2 again and stop exercise.
	Data Mode 3	Check ground command OSE and retransmit DC-2.
3 DC-26 Cruise science on	Science data in format	If battery charger off: Terminate exercise. If battery charger on: Check Register No. 4 for event — if no event check ground command OSE and command lock; if normal terminate exercise, or If cause is determined early and it is safe to proceed, retransmit DC-26.
Battery charger on	Power change	If science is off: Transmit DC-2 and terminate exercise. If science is on: Check Register No. 4 for event; if no event, check ground command OSE and command lock; if normal terminate exercise, or If cause is determined early and it is safe to proceed, retransmit DC-2.
4 DC-2 Cruise science off	Mode 2 format science will be all zeros. Register No. 4 event	Check ground command OSE and command lock. Retransmit DC-2 and terminate exercise.
	Mode 2 format science will be all zeros. No Register No. 4 event	Check ground command OSE and command lock. Retransmit DC-2 and consider termination of exercise.
Wrong data mode	Data Mode 1	Check power for cruise science on. Retransmit DC-2 and terminate exercise.
	Data Mode 3 with science on	Retransmit DC-2 and terminate exercise.
	Data Mode 3 with no data	Retransmit DC-2 and terminate exercise.
5 DC-28 Battery charger off	Power indications and Register No. 4 event	Ground command OSE and command lock. If normal terminate exercise.
	Power indications, no Register No. 4 event	Check ground command OSE and command lock. If normal terminate exercise; if not consider later exercise.
6 DC-25 Loss of roll control	Canopus intensity will be many DN higher than 37 DN	Continue exercise and reacquire using DC-19 procedure after exercise.
NAA early	If it occurs within the first 6 sec of DC-25, data encoder will be in Data Mode 2 with record sequence on. DAS bit 17 will indicate NAA. DAS bits 201 to 210, scan position, will not change. Event counter 3 will give EOT indications.	Send DC-3 within 10 min to get record information. Wait until second EOT; the data mode will switch to Mode 2. Send DC-28, DC-3, DC-26 (6 sec after shutter), DC-2. At this stage it is not known whether it is light induced or electrically induced.

Table 32. Emergency action summary (cont'd)

Single failures or problems	Indication	Action
6 (Cont'd)	If it occurs after the first 6 sec of DC-25, data encoder will be in Data Mode 3 with record sequence on.	Data encoder will switch to Data Mode 2 after 21 or 22 pictures. After the switch to Mode 2 send DC-28, DC-3, DC-26 (6 sec after shutter), DC-2. At this stage it is not known whether it is light induced or electrically induced.
No encounter power	Power telemetry channels 227, 203, 205, maybe 109; temperature control can also verify this. DAS can also tell from bit 197-NRT, bits 201–210 scan position.	Send DC-25 again send DC-28 if video storage 2.4-kc power comes on. DC-3, DC-26 if the battery charger is on, DC-2 if cruise science is off.
Battery charger on	Power telemetry channel 216.	Proceed with exercise DC-24, DC-3, DC-28, DC-26, DC-2.
No NRT power-on indication	DAS bit 197 should be a "1" for on.	Evaluate power telemetry — after about 10 min send DC-3 and evaluate DAS bits 281 to 420; send DC-2, DC-24, DC-28, DC-3, DC-26, DC-2.
Cover not deployed	Temperature control can verify from Channels 414, 418, and 437. (An event in Register No. 1 indicates that current has passed through the pyrotechnics solenoid which is the primary deployment device.) After scan position has changed by 20 deg the cover lanyard should deploy the cover.	Send DC-25; since it is a solenoid the command can be recycled; send DC-24, DC-28, DC-3, DC-26, and DC-2. At some later time after study maybe a 90–180 deg pitch or a midcourse maneuver.
Scan position	Position indication DAS bits 201 to 210 unchanged from stowed position before launch. Temperature Channel 411 may indicate hotter than normal clutch slipping.	Look at DAS bit 17; if NAA see 2 above. If no NAA send DC-3, DC-2, DC-24, DC-28, DC-3, DC-26, DC-2. At some later time after study maybe a 90 — 180 deg pitch or a midcourse maneuver.
	Position indication DAS bits 201 to 210 unchanging at the low clock angle. Temperature Channel 411 should indicate hotter than normal clutch clipping.	Have gone through the low clock angle limit switch. Send DC-3 to evaluate other scan position and TR voltage; send DC-2, DC-24, DC-28, DC-3, DC-26, DC-2. Could recycle here (DC-25) and the scan platform will search out of this limit switch.
	Position indications DAS bits 201 to 210 unchanging at the high clock angle. Temperature Channel 411 should indicate hotter than normal clutch slipping.	Have gone through the high clock angle limit switch. Send DC-3 to evaluate other than scan position and TR voltage; send DC-2, DC-24, DC-28, DC-3, DC-26, DC-2. The scan platform will not search out of this limit.
Loss of downlink	No received signal from the spacecraft.	Use the standard DC-12, DC-7, DC-8 procedure. Proceed with exercise after acquiring downlink.
Loss of uplink	Verbal station report.	Make station repairs.
Loss of Sun	Power telemetry indications power switching and logic (PS&L) voltage 109 will drop to battery voltage 226; battery drain will increase; 203, 204 will increase, 222–225 will drop off. Canopus sensor 108 will be turned off; 105, 106, 107 and 112, 113, 114, will most likely be unusual.	Send DC-19 which will turn the gyros back on and allow the spacecraft to reacquire its references. Then DC-24, DC-28, DC-3, DC-26, DC-2.
Switched power amplifiers	Low-gain antenna power will drop by some amount and Channel 300 will change from present DN to around 60 or 61 DN.	Complete exercise and send DC-7 later.
Switched exciters	Channels 301 and 302 may give different indications.	Proceed with exercise. Decide later question of switching exciters back.
Switched to Data Mode 3	Science data in place of engineering.	Evaluate TV and scan. Get enough data up to 25 min to be able to send DC-26 6 sec after TV shutter. Send DC-2 and evaluate engineering—proceed with exercise.
Switched to Data Mode 4/1	Engineering data only.	Do not know if in Mode 1 or tape is moving in Mode 4 and is in no-data-present configuration. Send DC-22. If still Mode 1 then it was Mode 1; send DC-2 and proceed. If it goes to Mode 4 after DC-22 then the Mode 4/1 was present. Send DC-2 and proceed.
	Video storage subsystem playback.	Send DC-2 and proceed.

Table 32. Emergency action summary (cont'd)

Single failures or problems	Indication	Action
7 DC-24 Still searching	DAS bits 201–210, scan position changing.	Calculate DC-24 again based upon last transmission time. Send DC-24. If not successful proceed with DC-28 and DC-3. Then calculate DC-26 to arrive nominally 6 sec after TV shutter and position the scan. (DC-16 could be used but it will start the record sequence.)
Wrong position	DAS bits 201–210, scan position not the predicted value.	Proceed with exercise.
8 DC-28 Video storage subsystem 2.4-kc power on	Power telemetry Channel 227.	Send DC-28. Proceed with exercise DC-3, DC-26, DC-2.
Battery charge does not come on	Power telemetry Channel 216.	Proceed with exercise DC-3, DC-26, DC-2.
Switched data modes	Other than Data Mode 2	Video storage subsystem should be off; if getting Data Mode 4, the video storage subsystem is still running; see DC-25.
9 DC-3 Wrong data mode	Data Mode 1	Do not know if this is Mode 4/1 or Mode 1 as video storage subsystem is off. Send DC-3, DC-2, DC-26, DC-2.
	Data Mode 2; no Register No. 4 event	Check ground command OSE and command lock; send DC-3, DC-2, DC-26, DC-2, or
	Data Mode 2; Register No. 4 event	Check ground command OSE and command lock; if normal, send DC-26, DC-2.
10 DC-2 Wrong data mode	Data Mode 1	Do not know if this is Mode 4/1 or Mode 1 as video storage subsystem is off. Send DC-2, DC-26, DC-2.
	Data Mode 3	Check ground command OSE and command lock. Send DC-2, DC-26, DC-2.
11 DC-26 Encounter science on	If cruise science is off, can be seen in Mode 2 format	Evaluate command in lock and power telemetry Channels 227, 203, 205, and maybe 109; DAS can verify with bit 197 NRT DAS power on. Temperature control also can verify. Send DC-26 followed by DC-2.
	If cruise science is on, check event Register No. 4 for event.	See above.
Cruise science on	There will be cruise science in the Mode 2 format	With encounter science off — terminate exercise.
12 DC-2 Cruise science off	Mode 2 format science will all be zeros.	Send DC-2 again. (DC-25 could be used, but will turn on encounter science; this could only be done after additional careful consideration.)
Encounter science still on	DAS bit 197 should be a 0 for off. Also power telemetry Channels 227, 203, 205, maybe 109. Temperature control can also verify.	Send DC-26 again followed by DC-2.
Some other data mode	See DC-25 and DC-28 for general discussions.	

Table 33. Spacecraft configuration before the start of the encounter sequence

Subsystem	Configuration
Structures and actuators	Scan platform preset to 177.9 deg All solar pressure vanes deployed to nonoptimum positions
Radio	Exciter A Power amplifier A (traveling-wave tube) Transmit via high-gain antenna Receiver via low-gain antenna
Command	In lock
Power	Main booster-regulator on Maneuver booster-regulator off Battery charger off Battery voltage high (36.6+ v) Boost mode enabled 310 w available from solar panels
CC&S	Launch counter L-1, L-2, and L-3 relays set Master timer MT-1, MT-2, MT-3, MT-4, MT-5 and MT-6 relays set
Data encoder	Data Mode 2 at 8⅓ bps ADC/PNG A
Attitude control	Canopus gate logic overridden by DC-15 Canopus sensor cone angle at 82 deg 3.5 yr of attitude control gas remaining Y-axis solar pressure vanes inoperative in the adaptive mode
Pyrotechnics and PIPS	First motor burn capability expended Second motor burn unarmed Science cover deployed All pyrotechnics devices except second motor burn squibs expended
Video storage	Off
Data automation subsystem	RT DAS operating normally NRT DAS off
Plasma probe	Returning abnormal data
Ion chamber	Totally inoperative
Cosmic dust detector	Microphone and pulse-height analyzer operating normally Penetration film operation questionable
Cosmic ray telescope Trapped radiation detector Helium magnetometer	All returning useful data
Planetary scan subsystem Television subsystem	Off

A return to the DC-19 mode of operation would reactivate the Canopus sensor brightness gates which contributed to eight losses of star acquisition in the first 19 days of flight. Since DC-15 was used to remove the brightness-gate logic in early December 1964, there was not a single loss of star acquisition, although 37 roll transients had been observed by the end of Mission Phase I.

Although the inertial mode may have been the safest mode for encounter, it was not to be attempted unless sufficient evidence existed that picture data could not be obtained or downlink telemetry could not be maintained without it. There would be considerable unknown risk in turning on the gyros, there would be difficult operational problems in maintaining roll orientation by ground command (because of gyro drift in the inertial mode), and DC-15 had demonstrated satisfactory performance, even during heavy cosmic dust activity.

2. Encounter Sequence Development

a. Encounter sequence initiation. Two methods were available for initiating the encounter sequence: ground command DC-25 and CC&S command MT-7. After detailed study, DC-25 was chosen as the best method of initiating the sequence; MT-7 served as a backup command.

DC-25 was transmitted at 14:27:55 GMT on July 14, 1965 and initiated the encounter sequence at 14:40:33 GMT (spacecraft time) by applying 2.4-kc power to the encounter-science loads and the video storage subsystem, and simultaneously applying 52 vdc from the booster-regulator to the 400-cps, single-phase inverter that supplied power to the scan subsystem drive motor and the video storage subsystem record motor. The relative merits of DC-25 and MT-7 are shown in Table 34.

Many circumstances might occur during the encounter phase which would require that the encounter power be withdrawn and reapplied to recycle the equipment involved. The nature of some circumstances is such that the decision to recycle would be an obvious one; and, therefore, the recycle would be initiated without knowledge of whether DC-25 would function. Other circumstances might make decisions not quite so obvious. In those situations the possibility of not turning on the equipment deenergized in the initial phase of a recycle sequence must necessarily have been weighed against the advantages to be gained by recycling.

Table 34. Relative merits of DC-25 and CC&S MT-7 command

Command for initiation	Advantages	Disadvantages
DC-25	1. Allows verification of capability for possible recycle requirement. 2. Permits added time for making operational decisions should some malfunction occur at encounter science turn-on. 3. Permits added time for performing scan platform pre-positioning calculations. 4. Allows selection of transmission time to possibly optimize chances of television pictures if DC-24 and WAA are unsuccessful and the scan platform searches all the way to the planet limb intersection by a narrow-angle sensor. 5. This technique has been successfully used in flight.	1. Adds shutter pulses to the television subsystem. 2. Requires longer periods of support from key personnel. 3. Requires the use of ground command capability as a primary mode, rather than as a backup.
MT-7	Permits scan positioning immediately after occurrence of MT-7 at nearly the optimum position.	—

An apparent means of escaping this latter dilemma would be to verify that the capability of turning on encounter science by ground command really existed. Then, if such a situation should occur, it would be possible to make the recycle decision trade-offs with more confidence. The only means of verifying the existence of a DC-25 capability before a recycle would be to transmit DC-25 before MT-7 occurs.

It must be said that the more subtle situations which might require a recycle sequence would be caused by power transients changing the logic states within various subsystems or by the failure of telemetry points. No history of power transients in a normal encounter sequence existed and the failure of a telemetry point was quite unlikely. The possibilities of being confronted by one of these subtle situations were considered to be very remote.

To accomplish the scan platform pre-positioning within the existing time constraint of 2 hr after MT-7, the procedure had to be very concise and all had to go smoothly. There would be little time for retracing steps if errors were discovered. Additional time, even an hour, would provide some contingency against the indicated latest time for transmission of the DC-24 platform inhibit command. This time could be provided by earlier initiation of the scanning motion by ground command. Early turn-on of the encounter science would permit longer reaction time to any nonstandard spacecraft operation resulting from the turn-on, whatever the initiation source.

With the intersection of the planet limb and narrow-angle field-of-view used as a reference point for the pre-positioning or automatic tracking of the planet, proper initiation of scan motion would provide for scanning to NAA, with high probability of obtaining acceptable television pictures. For this approach to be meaningful the trajectory predictions had to be close to the nominal estimated at one-half day prior to closest approach and the scan period had to be the same as during science cover deployment exercise on February 10, 1965.

There were several arguments against transmitting the early DC-25 to initiate the encounter sequence. The most prominent of these was the fact that should a DC-25 initiate the encounter sequence, the scan platform pre-positioning could be effected only after approximately 2 hr of data monitoring and calculations. Permitting automatic initiation of the encounter sequence afforded the opportunity to stop the platform with a ground command as soon as the system would accept it after power turn-on. Ground commands could not arrive at the spacecraft as close to each other as was necessary to effect this approach; consequently, this mode of platform positioning had to be forfeited if the early command initiated sequence were chosen.

Lesser disadvantages of an early encounter sequence initiation were the additional shutter pulses accumulated on the television shutter (75 pulses/hr of operation), and the added time that key personnel were required to be on duty. Information available on the shutter was that a large margin existed, based on test data, in the number of pulses required before degraded performance would occur. Assuming the DC-25 to be transmitted 90 min before MT-7 occurred, the added pulses were not considered to be a factor of concern. Personnel participation for an additional 90 min was not considered too dangerous in that there would be some slack periods during which breaks could be taken. If there should be severe problems, the extra time would never be noticed until after the encounter sequence was over.

A DC-25 was to be transmitted to initiate the encounter sequence because the knowledge attained by observing the results of this command could contribute immeasurably to future action in the encounter sequence. Without a working DC-25 command, the commitment to any action that might require the DC-25 for success would have to be carefully reviewed. If a poorly positioned scan platform were the only situation requiring an encounter power recycling, the need for the DC-25 would not be so great.

The CC&S issued command MT-7 at 15:41:49 GMT on July 14, 1965. The only effect of MT-7 was an event in data encoder Event Register 2. If DC-25 were not successful in initiating the encounter sequence, MT-7 would cause 2.4-kc power to be applied to the encoder science loads and the video storage subsystem and, simultaneously, would cause 52 vdc from the booster-regulator to be applied to the 400-cps, single-phase inverter that in turn supplied power to the scan subsystem drive motor and the video storage subsystem record motor.

b. Scan platform pre-positioning. Incorporated in the design of the spacecraft was the means to position the platform for taking the television pictures without the use of ground commands. Failure of this function would leave little reaction time because its occurrence could only be observed very near to the planet. Therefore, the platform was positioned using a ground command early in the encounter phase, leaving the remaining hours free for solving any problems which might occur. Operationally, pre-positioning would be advantageous. Implementation of this capability required use of the ground command DC-24. The two major techniques for utilizing this command were as follows:

1. Advantages of DC-24 following observation of telemetry:
   a. Optimum television camera positioning would be possible.
   b. The technique was employed successfully previously in the mission during the science cover deployment exercise on February 10, 1965.
2. Advantages of DC-24 just after encounter power turn-on:
   a. The possibility of limit-switch failure would be eliminated, which in turn would prevent the platform from reversing.
   b. Pre-positioning would be possible without performing 2 hr of data calculations based upon real-time telemetry.
   c. Chances of recycling due to improper positioning would be reduced.

The scan platform position for achieving optimum pictures was determined from studies by television experimenters. Use of telemetry data to determine platform position as a function of time permitted an estimation of when the platform would be in the optimum position. A command could be timed to arrive at the spacecraft at the selected time and cause the platform to stop.

Flight experience and numerous ground tests indicated that this technique was quite accurate and could be expected to be so during the encounter phase. Stopping the platform to within 0.75 deg of the desired position was anticipated using this technique.

Circumstances were such that the preencounter scan platform position, achieved during the science cover deployment exercise, was considered to be the optimum one for television pictures. Due to changes in the prediction of the spacecraft orbit, the optimum scan position for television had changed approximately 2 deg. When the scan electronics were turned on, the scan platform moved initially in the direction which reduced the difference between the initial position and the optimum position.

The scan subsystem can react to the DC-24 command 8 sec after the power is initially applied to it. Arrival of DC-24 at this time interval after MT-7 would assure the platform is positioned to within 2 deg of the optimum position, based upon the scan search rate of 0.5 deg/sec. There was reason to expect that the platform would not move 4 deg during the initial 8 sec after turn-on. Mechanical backlash and torque buildup in the scan actuator might delay movement of the platform the equivalent of 0.5 deg. In addition there might be other forces to be overcome when the platform was initially activated, reducing the total movement prior to the receipt of DC-24.

An important aspect of the latter technique would be the prediction of the time of the occurrence of the MT-7 command. Fortunately, the flight experience had shown that the times of cyclic command (CY-1) could be accurately predicted to within 0.3 sec of the actual times. This ability to observe the time of the commands accurately to verify predictions was the result of the frequency transient which occurred on the downlink RF carrier whenever a cyclic command was issued by the spacecraft when the receiver was out of lock.

The critical operation of calculating the time for transmitting the DC-24 based upon scan position telemetry reading could be eliminated by sending DC-24 to arrive immediately after a CC&S initiated encounter sequence. This could aid the entire operation by permitting the concentration on other problems should they arise. This method had not been proved on the flight spacecraft but was successfully demonstrated on the PTM.

The advantage of the platform positioning immediately after power turn-on could not be achieved when a DC-25 initiated the encounter sequence because the flight command subsystem could not accept more than one command every 26 sec and the ground command subsystem could not issue more than one command every 55 sec in the normal mode.

The best way to position the scan platform appeared to be the use of DC-24 almost immediately (8 sec) after the CC&S MT-7 event occurred at the spacecraft. This was based upon the fact that the least demand would be placed on the scan subsystem and the operations personnel to effect such an action. However, it was felt that the knowledge gained by having transmitted the DC-25 to initiate the sequence was worth the risks and extra effort required in positioning the platform based on telemetry data.

Although the "immediate DC-24" plan would free up to 2 hr of time for studying other situations aboard the spacecraft, if the situation arose where recycle was a possibility, the assurance that the DC-25 capability still existed after many months of disuse would provide SPAC with a valuable analytic tool.

c. Science data requirements. When the scan platform was pre-positioned, the planet would not enter the field of view of the wide-angle sensor until less than 45 min before the first picture was to be taken. Only at this time would the automatic switching of the data encoder from Mode 2 to Mode 3 occur. When DC-3 was received by the spacecraft at 22:23:07 GMT July 14, 1965, it transferred the data encoder to Mode 3 operation (all science words) as soon as the transfer was acceptable to the data encoder transfer logic. The use of DC-3 permitted evaluation of the NRT DAS and television status when sufficient time would be available to recycle the sequence should it be judged necessary after reviewing the data.

In addition, the availability of Mode 3 data permitted the accurate determination of the television shutter sequence for subsequent determination of the time to transmit a DC-16 backup to the on-board initiation of the television record sequence and a DC-26 backup to the on-board termination of the television record sequence.

The lack of Mode 2 data detracted from the ability to monitor spacecraft engineering performance. A very critical feature of the all-science telemetry condition was that the RF and command conditions were unknown. In order to alleviate this situation the ground command frequency was at an appropriate offset value which would cause the automatic lockup of the command loop to begin if an inadvertent loss of lock occurred.

It was demonstrated that the successful transmission of commands to the spacecraft could be effected when the lack of engineering telemetry existed.

An operational constraint recognized for the use of DC-3 was that it was to arrive at the spacecraft in the science portion of the Mode 2 data frame. This prevented an early science gate from being generated and disturbing the DAS frame-count-vs-time computations which were being utilized in determining the times for subsequent command transmissions.

DC-3 had to be transmitted at 2 hr before the expected start of the television recording sequence and timed to arrive in the science portion of the data Mode 2 data frame.

d. Wide-angle acquisition. Wide-angle acquisition would occur at about 23:55:45 GMT (0.7 probability) on July 14, 1965 when the illuminated disk of Mars would move into the field of view of the scan subsystem wide-angle sensor. Because WAA was a backup to the DC-24 scan platform pre-positioning and the DC-3 transfer to Mode 3, the only outward effect of WAA by the scan subsystem would be the WAA event in the science data.

WAA had the following two backup functions:

1. If the scan platform were not pre-positioned by DC-24, the scan subsystem would be transferred from the search mode to the track mode about 3 hr before closest approach to Mars when the wide-angle sensor detected the illuminated disk of Mars. The DAS would then issue a command transferring the data encoder to Mode 3.

2. If the scan platform were pre-positioned but the data encoder could not be transferred to Mode 3 by ground command DC-3, the DAS would transfer the data encoder to Mode 3 when a wide-angle signal was received from the scan subsystem about 45 min before the start of the television picture recording sequence.

e. Narrow-angle acquisition. Narrow-angle acquisition would occur nominally at 0017 GMT on July 15, 1965 when the illuminated disk of Mars moved into the field of view of one of the narrow-angle sensors. When the planet is sensed by a narrow-angle sensor, the scan subsystem sends a narrow-angle signal (NAS) to the DAS, which immediately removes 400-cps single-phase power from the scan platform drive motor (backup to DC-24). The DAS starts issuing start-tape-recorder and stop-tape-recorder commands to the video storage subsystem within 60 to 204 sec after NAS. The start and stop commands continue until terminated by on-board logic or ground command.

The initiation of the television record sequence was such an important function that a ground command, DC-16, was available as a backup for the two parallel inputs from on-board sensing of the planet. DC-16 initiates an NAS and thereby conditions the DAS logic to begin the television picture recording sequence and to transfer the data encoder to Mode 3 (as a backup to WAA). This command had to be transmitted prior to the time the planet acquisition event was observed on Earth if the maximum benefit of its capability were to be attained.

Allowing the spacecraft to react normally to the stimulus of the planet in the field-of-view of the narrow-angle sensors, prior to arrival of this command at the spacecraft, was the philosophy employed in determining the command transmission time. Due to the importance of this initiation function, any unanticipated interaction was prevented between the DC-16 and a recording sequence already in progress when it arrived. This condition was so remote as to be nearly impossible, but to safeguard against any stray signal path not present before launch it was anticipated that two pictures from the latest estimated automatically-initiated record sequence would have occurred when the DC-16 arrived at the spacecraft.

Another feature of this command was its capability to inhibit the scan platform motion if this function had not already been performed. Should the command have been utilized to effect this result also, the philosophy for its use as mentioned in the previous paragraph would have to be modified to assure the proper consideration of the phasing of the scan platform position relative to the planet disk at the time of command arrival. This might result in initiation of the record sequence prior to the limb crossing and force pictures of space to be taken early at the possible expense of terminator pictures later in the sequence. The thought under these conditions was to achieve some pictures, not all nor necessarily the best pictures.

Timing required to choose the instant of DC-16 arrival at the spacecraft can be provided by the attainment of sufficient data on the NRT DAS frame sequencing from the Mode 3 data. As much as 144 sec could be saved by the reduction in tolerances which must be applied to the calculations if sufficient Mode 3 data were not available.

It was concluded that transmission of the DC-16 should take place to provide the backup to automatic record sequence initiation whenever possible.

f. Record sequence termination. The television record sequence was to be terminated about 25 min after its initiation (nominally 0049 GMT at the spacecraft) and the data encoder was to be transferred to Mode 2 by one of three on-board methods:

1. The video storage subsystem count-two-and-stop circuitry terminates the record sequence when two end-of-tape signals have been received.

2. The DAS terminates the recording sequence when it receives the first EOT after the start of Picture 19.

3. The DAS terminates the recording sequence when 22 pictures have been completed.

Ground command DC-26 would be transmitted 1 min 44 sec after NAA appeared in telemetry (normally 0038 GMT) to ensure that television data recorded on the video storage tape would not be lost due to failure of the recording sequence to terminate properly or failure of the video storage record motor to be stopped when the picture sequence ended. DC-26 removes 2.4-kc power from all science loads (video storage 2.4-kc power remains on) and removes booster-regulator 52-vdc power from the 400-cps, single-phase inverter that powers the video storage record motor, causing the tape to stop moving. DC-26 serves as a backup to the portions of the video storage subsystem and the NRT DAS that are designed to terminate the recording sequence.

Running of the tape past the record heads when no useful data were being applied to the input amplifiers would result in erasing the previously recorded data. If a condition such as this existed, eradication of the planetary data already recorded would occur with the accompanying failure of a prime mission objective. Various failure modes associated with the video storage subsystem and DAS had been determined which could have resulted in the loss of all pictures from the tape after they had been recorded.

Some failure modes could be bypassed with the use of DC-26 if the command were transmitted at the correct time. This was accomplished by allowing the DC-26 to effect various changes-of-state aboard the spacecraft which would otherwise require a number of components to function properly to accomplish the same result. Should the components aboard the spacecraft be unsuccessful in terminating the recording sequence properly, the DC-26 would arrive shortly after the sequence should end and thus provide a backup. The round-trip signal time was a critical constraint in that it prevented observation of the telemetry data to determine whether the command was really needed before the command must be initiated to be effective. Unless the DC-26 arrived shortly after the recording sequence was to automatically terminate, its effectiveness would be drastically reduced; after 10 min it would not be useful at all.

Recognizing that the number of component failures guarded against by this command action was small, compared with the great number of components which must function satisfactorily to achieve mission success, it might have been difficult statistically to become concerned with failure of these components. It must be remembered that most component failures could not be compensated for through the use of ground commands, but where component failures could be eliminated as series elements to mission success, it was reasonable and judicious to do so. This principle had merit not only in the design phase but also in the flight operations phase. The equipment which controlled the recording sequence had not been used since the launch but DC-26 had, and consequently there was uncertainty as to whether the control equipment would survive the launch environment.

In addition to terminating the recording sequence, DC-26 removed power from all science equipment as stated previously. Although a DC-2 was to be sent immediately after the DC-26 to reenergize the cruise science equipment, there had been some concern expressed that all instruments might not turn on again due to the low temperatures at which the instruments were operating. A certain amount of risk was accepted in this case to permit the backup function that was desired.

Should the DC-26 fail to turn off the encounter science equipment, CC&S command MT-8 would back up this function at 0501 GMT (spacecraft time). The objective of this reduction in spacecraft power is to alleviate the power demand during the playback phase.

Ground command DC-2 would be transmitted nominally 1 min after command DC-26 was sent. DC-2 applies 2.4-kc power to the cruise science instruments (turned off by DC-26) and transfers the data encoder to Mode 2 as a backup to the on-board logic that terminates the recording sequence. DC-2 would transmit at 5-min intervals until confirmation was received that cruise science was on.

The probability that more than one DC-2 would be required to turn on the cruise science instruments was small, but there was a small but finite possibility of dropping command lock between DC-26 and DC-2. Due to the great interest in the near-planet field-and-particles data, the periodic transmission of DC-2 would ensure that cruise science instruments would be turned on with minimum delay.

The field-and-particles instruments would be reactivated before closest approach. DC-26 would turn off the instruments about 20 min before closest approach when the spacecraft would be at a radial distance from the planet 1600 km farther than the distance of closest approach.

*g. **Occultation.*** About 70 min after the spacecraft makes its closest approach to Mars the spacecraft would pass behind Mars as seen from Earth, entering the occultation region at 0212 GMT (the RF signal would be lost at about 0224 GMT) and emerging at about 0305 (the RF signal would be reacquired at Earth at about 0317). As the spacecraft enters and exits the occultation region the 2300-Mc spacecraft tracking and telemetry signal traverses the atmosphere and ionosphere of Mars. The changes caused in the frequency, phase, and amplitude of the signal by passage through these media provide the information from which a model of the Martian atmosphere and ionosphere could be constructed.

The spacecraft would be in two-way lock as it entered and exited the occultation region in order to provide more accurate tracking data. The two-way-lock condition would be maintained until just prior to MT-9 when the DSIF transmitters would be turned off.

Command MT-8 would be issued by the CC&S at 0501 GMT on July 15, 1965. MT-8 would function as a backup to DC-26 because, if encounter science were turned off on DC-26 as planned, the only effect of MT-8 would be an event in data encoder Event Register 2.

h. Picture playback. At about 10 hr 40 min after closest approach (1141 GMT) the CC&S would issue command MT-9 which automatically would switch the data encoder to Mode 4/1, turn off the cruise science instruments, and initiate playback of the television pictures recorded by the video storage subsystem. After about 1 hr of Mode 1 (all engineering) data, the first recorded data would be received. Recorded data including the first television picture of Mars would continue for about 8 hr 35 min. Thereafter, about 2 hr of engineering Mode 1 data would alternate with 8 hr 35 min of recorded data until additional commands were received by the spacecraft. The engineering data between pictures permitted assessment of spacecraft performance and verification of the condition of the spacecraft telecommunications to aid the DSIF stations in maintaining their tracking capability.

The Mode 1 data would be obtained when on-board logic in the video storage subsystem sensed that no recorded data was on the tape between picture recordings and switched the data encoder to Mode 1. When the video storage subsystem again sensed data on the tape, the data encoder would be transferred to Mode 4 (recorded data playback). The lack of data on the tape would be due to the fact that no data inputs were applied to the tape during the starting and stopping of the tape during the record sequence. During the transition period between start and stop commands a certain length of tape passed the record heads, causing a data gap to exist between the segments of recorded data. Because the playback rate is 1/1280th of the record speed, the data gap requires about 2 hr to pass the playback heads. Many variables influence the length of tape traversed during the transitions between stop and start and consequently the length of time before the start of the first picture and between the pictures during playback could not be predicted precisely.

It was not planned to initiate playback before MT-9 was issued, if all spacecraft conditions appeared normal.

Although the spacecraft would be in two-way lock during the occultation experiment, the spacecraft would be tracked in the noncoherent (one-way) mode during picture playback because less telemetry data would be lost in the one-way mode than in the two-way (coherent) mode. Data can be lost in the coherent mode because if the ground transmitter experiences a transient, the spacecraft reacts by shifting the downlink frequency momentarily, causing the ground receiver to drop lock and the demodulator to lose telemetry data.

To ensure that the spacecraft is in the one-way mode before the start of picture playback, the tracking DSIF stations turn off their transmitters sufficiently before MT-9 to cover the contingency of initially playing back from the second video storage tape recorder channel.

The stations switch to the two-way mode during the 2-hr intervals of Mode 1 data between pictures to obtain more accurate tracking data. The period of two-way data would be timed so that sufficient margin would be left to ensure that the television pictures would be played back in one-way lock.

Two complete playbacks of the recorded data would be required. During this long span of time, the CC&S cyclic commands (CY-1) continue to be issued. The on-board logic which controls receiving antenna switching would be active during the playback period, so that the periods of two-way lock would have to be scheduled to prevent unwanted antenna switching by the on-board circuitry.

Ground command DC-4 could be used to back up MT-9 if it should become necessary. DC-4 transfers the data encoder to Mode 4/1 operation as soon as the transfer is acceptable to the data encoder transfer logic. If television picture data are available from the video storage tape recorder, television data are telemetered; if no television data are present (as between television pictures), engineering data are telemetered. DC-4 also removes 2.4-kc power from the cruise science instruments.

The time chosen to terminate the playback mode had several possibilities depending on future plans. The plan chosen was one which made it possible to record new data over only half of the Mars data and play back these new data without first playing back Mars data. This was accomplished by sending DC-28, *video storage 2.4-kc power off,* to arrive at the start of the twentieth line (Line 19) of Picture 22. This allowed all recorded data, except a little over three lines, to be played back twice. These three lines of data provided enough tape left before the EOT foil so that, when the video storage subsystem was re-energized and the tape began to move, it would reach a speed of over half the steady-state velocity before the foil passed over the EOT sensor; the tape velocity is 12.84 in./sec during the record mode. This condition of over half the steady-state velocity was a requirement to ensure that only one EOT signal was sent to the count-two-and-stop circuitry from that pass of the foil. This would then allow the tape to have new data recorded on the next side up to the second pass of the foil over the sensor and the tape would stop there. By stopping three lines before the foil, the recorded data on the tape from any future recording would be just after the foil. This would allow the desired new data to be played back without playing back Mars data first. This would save time, allowing additional cruise science information to be transmitted.

Signal DC-28 would switch the data encoder to data Mode 1 of the Mode 4/1 when the data encoder logic sensed the absence of Mode 4 playback data at video storage subsystem 2.4-kc power off; DC-28 also turned the battery charger on. Thus, DC-26 was required to turn it off again. By allowing 6 min between DC-28 and DC-26, over two commutations of the data encoder medium decks could be evaluated in that spacecraft condition.

To re-establish the cruise mode, DC-2 would have to be sent 6 min after the DC-26. This also would allow over two commutations of the data encoder medium deck for data evaluation before the Mode 2 data started arriving at Earth.

Another possibility studied for terminating the playback mode would be to send DC-28 after two complete tape passes. This would allow receipt of all data twice and put the recorder in a position to record two complete tracks of data in the future. It seemed quite likely that a special calibration recording would take place for evaluation of the recorded data of Mars which would only require five pictures. If this calibration were required, a tradeoff would have to be made between recording over all the Mars data, destroying that data, or playing back again to some point in the tape before the foil, which would leave some of the Mars data. Unless the tape were then positioned to just before the foil (which would require approximately 5 days), the tape would again have to be played back to the point of record start before all the calibration data could be played back. This plan would either require destroying the Mars data by recording over it for a television calibration, or would require from 5 to possibly 10 days of loss of cruise science while the tape was being positioned. The plan chosen did not have these disadvantages and the loss of playing back three lines of data for the second time seemed inconsequential.

i. Recycle sequence. There are situations that can develop during the encounter sequence that would require that certain aspects of the spacecraft encounter logic be reset. So that the spacecraft can be reconditioned in some optimum manner, after an undesirable state has been established by certain spacecraft encounter subsystems, a recycle sequence has been developed. After the spacecraft has been exercised using the recycle sequence, those subsystems acted upon should be in a more advantageous state to fulfill their encounter functions successfully.

The recycle sequence is used to recondition those subsystems that are associated with the picture taking sequence of the Mars encounter. The recycle sequence will most likely be used to restart the scan platform if the pre-positioning exercise fails to position the scan platform acceptably. However, other problems can occur which can make use of the recycle sequence for possible solution, e.g., early NAA or scan platform hangup at the lower limit. Certain portions of the recycle sequence can also be used in an attempt to solve problems associated with the NRT DAS, the planet sensors, the television subsystem, and the video storage subsystem.

The intent of the recycle sequence is to attempt to correct those conditions for which it has been developed in the minimum time and with the maximum obtainable advantages for the minimum number of commands. The command sequence is arranged so that each command is transmitted with the maximum confidence and with the minimum possible effect to other spacecraft circuitry. Table 35 shows the recycle sequence timing.

DC-3. DC-3 transfers the data encoder to Mode 3 (all science data). Mode 3 data is required so that the sequencing of the television camera shutter can be determined. It is desirable that the television subsystem be

Table 35. Recycle sequence

Item	Command	Time	Effect
1	DC-3	X	Transfer data encoder to Mode 3
2	DC-2	X + 83 min	Transfer data encoder to Mode 2
3	DC-26	X + 98 min	All science 2.4-kc power off and 400-cps, 1-phase power off
4	DC-2	X + 99 min	Cruise science 2.4-kc power on
5	DC-28	X + 102 min	Video storage 2.4-kc power off
6	DC-25	X + 103 min	All science and video storage 2.4-kc power on and 400-cps, single-phase power on

turned off with the camera shutter closed. This precaution is taken to eliminate the requirement that shutter resynchronization circuitry built into the television subsystem be required to operate at power turn-on. Failure of the resynchronization circuitry to operate when required could result in the catastrophic loss of all television pictures. This information obtained during the Mode 3 data will be used to calculate the time for transmission of DC-26.

DC-3 would be transmitted as soon as possible after the decision to recycle was made and at a time that would cause the data encoder to transfer to Mode 3 while the data encoder was in the science portion of the telemetry frame.

DC-2. This command is used to transfer the data encoder to Mode 2 (⅓ engineering data and ⅔ science data). The command would be transmitted so that engineering data would be transmitted from the spacecraft when the science instruments were turned off.

DC-2 would be transmitted approximately 15 min before the transmission of DC-26.

DC-26. DC-26 turns off all science 2.4-kc power and the 400-cps, single-phase power. The primary purpose of this command in the recycle sequence would be to remove the 2.4-kc power from the scan subsystem electronics. By reapplying the 2.4-kc power, the scan platform inhibit logic would be reset. This command also allows resetting the logic associated with the encounter-sensing devices; such as the wide-angle and narrow-angle sensors. The use of this command conditions the NRT DAS and the scan platform electronics so that when power is reapplied to these subsystems their logic states are the same as when their power was initially applied.

DC-26 would be transmitted so that the television subsystem would be turned off while the television camera shutter is closed.

DC-2. This command would be used in the sequence to turn on the 2.4-kc power to the cruise science loads. Since this command turns power on to only the cruise science loads, the power turn-on transient is reduced using this command as the first step in reapplying power to the science subsystem as compared to the command that turns power on to all the science loads.

DC-28. This command would be used in the sequence to remove the 2.4-kc power from the video storage subsystem. This command also would enable the charge mode of the battery charger. However, the next command in the sequence reverts the battery charger back to its boost mode. The purpose in removing the 2.4-kc power from the video storage system was to permit a reinitialization of the video storage subsystem count-two-and-stop circuitry which is designed to count two EOT signals and then inhibit any further start-record commands from the DAS. This command would be used to ensure against the possibility that the power transients associated with this sequence had not set the count-two circuitry to its inhibit-start-command condition.

This command also would reduce the chances of a catastrophic loss of all television pictures, should the automatic reset of the count-two-and-stop circuitry not occur at the first DAS start-record command.

DC-25. This command would be used in the sequence to apply 2.4-kc power to the encounter-science loads and the video storage subsystem and also apply 52 vdc to the 400-cps, single-phase inverter that supplies power to the scan drive motor and the video storage subsystem record motor. The command would also be used to revert the battery charger back to its boost mode.

DC-25 would be transmitted at a time, at least 5 min after the transmission of DC-26, such that if the scan platform remained in the search mode all the way to the planet, the scan platform would intersect the planet in the southern hemisphere.

j. Encounter sequence summary. The encounter sequence ultimately adopted and executed by SPAC is shown in Fig. 4 and Table 36.

Table 36. Encounter sequence timing

Date	Time, GMT	Event
July 14	0933/E[a]	Johannesburg rise
	1421/E	DC-25 transmitted (± 6 min)
	1433/S[b]	DC-25 received (± 6 min)
	1445/E	DC-25 seen in data (± 6 min)
	1455/E	DC-25 transmitted (if first one did not function) (± 6 min)
	1507/S	DC-25 received (± 6 min)
	1519/E	DC-25 seen in data (± 6 min)
	1529/E	DC-24 if DC-25 has been ineffective
	1541/S	MT-7
	1541/S	DC-24 (MT-7 ± 8 sec)
	1553/E	MT-7 seen in data (15:53:47)
	1553/E	Scan pre-positioned if MT-7 turned on encounter science (15:53:55)
	1753/E	DC-24 — Latest time to transmit
	1903/E	Goldstone — Echo rise
	1934/E	Goldstone — Pioneer rise
	2053/E	Johannesburg set
	2210/E	DC-3 transmitted (± 30 sec)
	2222/S	DC-3 received
	2234/E	Switch to Data Mode 3
	2339/S	Mars begins to enter scan field of view for 180-deg pre-positioned platform (E — 1 hr 24 min)
	2349/S	WAA (0.7 probability)
July 15	0001/E	WAA observed (0.7 probability)
	0017/E	DC-16 transmitted
	0021/S	NAS (E — 42 min) nominal
	0023/S	NAA (NAS + 0 to 132 sec)
	0024/S	Start TV record (NAA + 60 to 72 sec)
July 15	0029/S	DC-16 received
	0033/E	NAS observed
	0035/E	NAA observed
	0038/E	DC-26 transmitted (NAA + 1 min 44 sec)
	0039/E	DC-2 transmitted
	0049/S	TV record sequence complete; switch to Data Mode 2 (TV start + 24 min 48 sec)
	0050/S	DC-26 received
	0051/S	DC-2 received
	0101/E	Switch to Data Mode 2
	0102/E	DC-26 observed
	0103/E	DC-2 observed
	0103/S	Closest approach (nominal E — 0)
	0135/E	Tidbinbilla rise
	0153/E	Woomera rise
	0212/S	Spacecraft enters occultation region
	0224/E	Loss of RF signal at Earth
	0305/S	Spacecraft exits occultation region
	0317/E	RF signal observed at Earth
	0501/S	CC&S MT-8
	0513/E	CC&S MT-8 event received
	0624/E	Goldstone —Pioneer set
	0630/E	Goldstone — Echo set
	0930/E	Johannesburg rise
	1141/S	CC&S MT-9 at CY-1 No. 83
	1153/E	CC&S MT-9 at CY-1 (initiate tape playback)
	1241/S	Start of Mode 4 data (Picture 1)
	1248/E	Tidbinbilla set
	1253/E	Start of Mode 4 data (Picture 1)
	1352/E	Woomera set

[a]GMT time on Earth.
[b]GMT time at spacecraft.

3. Operational Considerations

a. DSIF transmitter frequency. During the encounter phase, the DSIF transmitter VCO frequencies were chosen so that one frequency would be satisfactory for a long period of time. The SPE in the spacecraft receiver VCO would not be maintained at a zero offset. Adjustment of the SPE was accomplished with the command modulation signal off. Since command modulation would be applied to the uplink signal for long periods of time during encounter, it was necessary to discard the procedure for maintaining a zero SPE in the spacecraft receiver to prevent having to drop command modulation frequently. Commands could be received by the spacecraft when the static phase error is not excessive, so a modest static phase error would be tolerated.

Doppler shift and range effects cause the SPE to move in a negative direction. Procedurally a transmitter VCO frequency was chosen initially so the spacecraft SPE telemetry indicated a positive error equivalent to several DN (approximately 30 cps). With time, the error drifted through zero and assumed negative values. The correct transmitter VCO frequency values were obtained from the Tracking Data Analyst (TDA).

Studies indicated that the doppler shifts during the Goldstone closest approach pass could be accommodated without rezeroing the spacecraft SPE using the above technique from spacecraft rise until entry into the occultation region. Figure 57 indicates this effect.

The figure shows the encounter sequence with respect to GMT time from MT-7 to MT-9. The times of occurrence of MT-7, MT-8, MT-9 and the DSIF station rise and set times were accurately predictable. Times for WAA and NAA, closest approach (E), and occultation

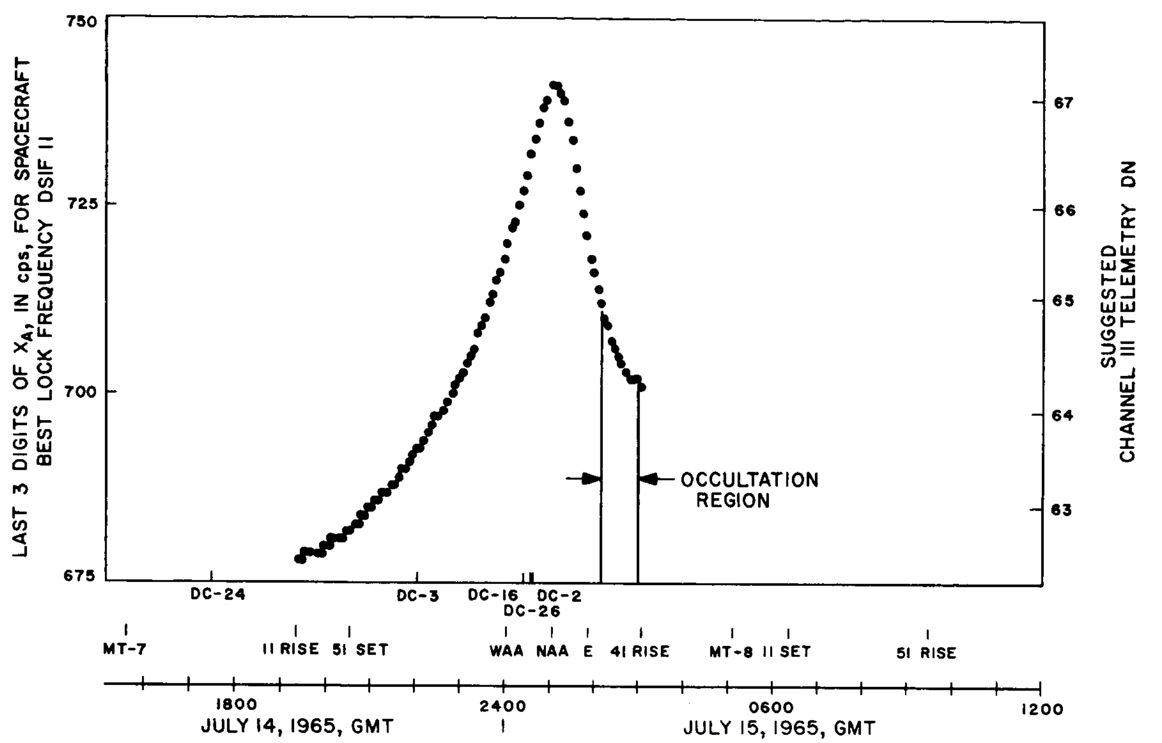


Fig. 57. Doppler shift at encounter

were subject to small uncertainties arising from uncertainties in orbit determination.

The dotted curve represents predicted doppler shift during encounter referenced to the ground transmitter VCO frequency and to the spacecraft VCO frequency as indicated by Channel 111, receiver SPE. The ground transmitter frequency was adjusted at the start of the Goldstone pass to provide a Channel 111 DN of low 63. The ground VCO frequency corresponding to a low 63 could be determined from ground transmitter VCO frequency predictions. By this procedure, the spacecraft receiver VCO frequency remained within 2 DN of the best-lock frequency for all doppler shifts from Goldstone until the spacecraft entered the occultation region, without requiring resetting of the ground transmitter VCO frequency.

b. Ground command transmission procedures. The early-mission practice of normalizing the ground command subsystem $8f_s$ after locking up the command loop was discontinued at encounter. Instead, the ground command frequency was chosen to provide an offset from the normalized frequency. The intent was to maximize the probability of regaining command lock in minimum time if a transient from the ground transmitter should cause a loss of lock. It had been judged that the ability of the spacecraft to successfully receive commands was insignificantly affected by this practice.

The usual practice of locking up to the command loop with some frequency offset from the free-running VCO frequency was continued. This offset frequency was the frequency maintained for the command modulation signal. Whenever command lock is dropped inadvertently the spacecraft command loop automatically attempts to lock up and may lock up by the time the telemetry received on the ground indicates loss of lock.

Because of the signal round-trip time consideration, after verification that the ground command subsystem was functioning properly, permission could be given to transmit a command, although command-detector telemetry indicated out-of-lock.

c. Use of DSIF 13 during encounter. Use of DSIF 13 100-kw transmitter was considered as a backup to DSIF 11 or as the command station when the spacecraft-received signal-level from DSIF 11 was unsatisfactory for providing command capability. As planning for the encounter phase progressed, the use of DSIF 13 as the primary station for transmitting to the spacecraft was considered. It was determined from a spacecraft point of view that its use as the prime transmitting station was most desirable.

In opposition to this was the fact that there was a significant degradation in the quality of the tracking data available for orbit determination when DSIF 13 was transmitting to the spacecraft. This was due to the fact that DSIF 13 was not a duplexed station[21] and the ground instrumentation and data-handling operations added degrading features to the frequency data attained when the data from another station had to be compared with the signals transmitted from DSIF 13. There did not exist a tight-closed-loop operation in this mode.

It had been estimated that the orbit near Mars could not be determined much better after encounter if DSIF 13 were transmitting than it could be determined before encounter, based solely on cruise-phase data. This consideration, and its attendant adverse effects on the occultation experiment, forced the use of DSIF 13 to be solely a backup to be used in the situation where roll orientation was lost and telemetry from the planetary experiments would be in serious jeopardy if DSIF 13 were not utilized.

A recommendation from the occultation experimenters to transmit from DSIF 13 at exit occultation still had to be considered a possibility.

4. Contingency Plans

One of the primary aims of SPAC and the EPWG was to determine the nonstandard events that were unique to the encounter phase of the mission. It is recognized that all possible nonstandard events that might have arisen during the encounter sequence were not described in this Section; however, the possible corrective actions for many of the nonstandard conditions were described. The exact conditions, applicable at the moment the nonstandard condition arose, determined the response in most cases. Figure 58 shows a portion of the encounter decision diagram developed by SPAC.

a. Command problems.

Inability to achieve command lock with 10-kw transmitter. The use of any commands under this condition would be restricted to the time when the spacecraft would be over DSIF 13, assuming that the inability to lock up the command detector sync loop could be corrected with about 12-db more spacecraft-received carrier power. Spacecraft failures could prevent any station from locking up the command loop.

Initially the effect would be that the pre-positioning of the scan platform would have to be dropped as an item in the sequence. If MT-7 turned on the encounter science, it would be too dangerous to pre-position over DSIF 13 because of the possible adverse effects of having the scan subsystem go to the track mode while the command was in transit. If DC-25 were required to turn on encounter science, a rapid DC-24 would not be possible and would result in a situation the same as an encounter-science turn-on before DSIF 13 rise.

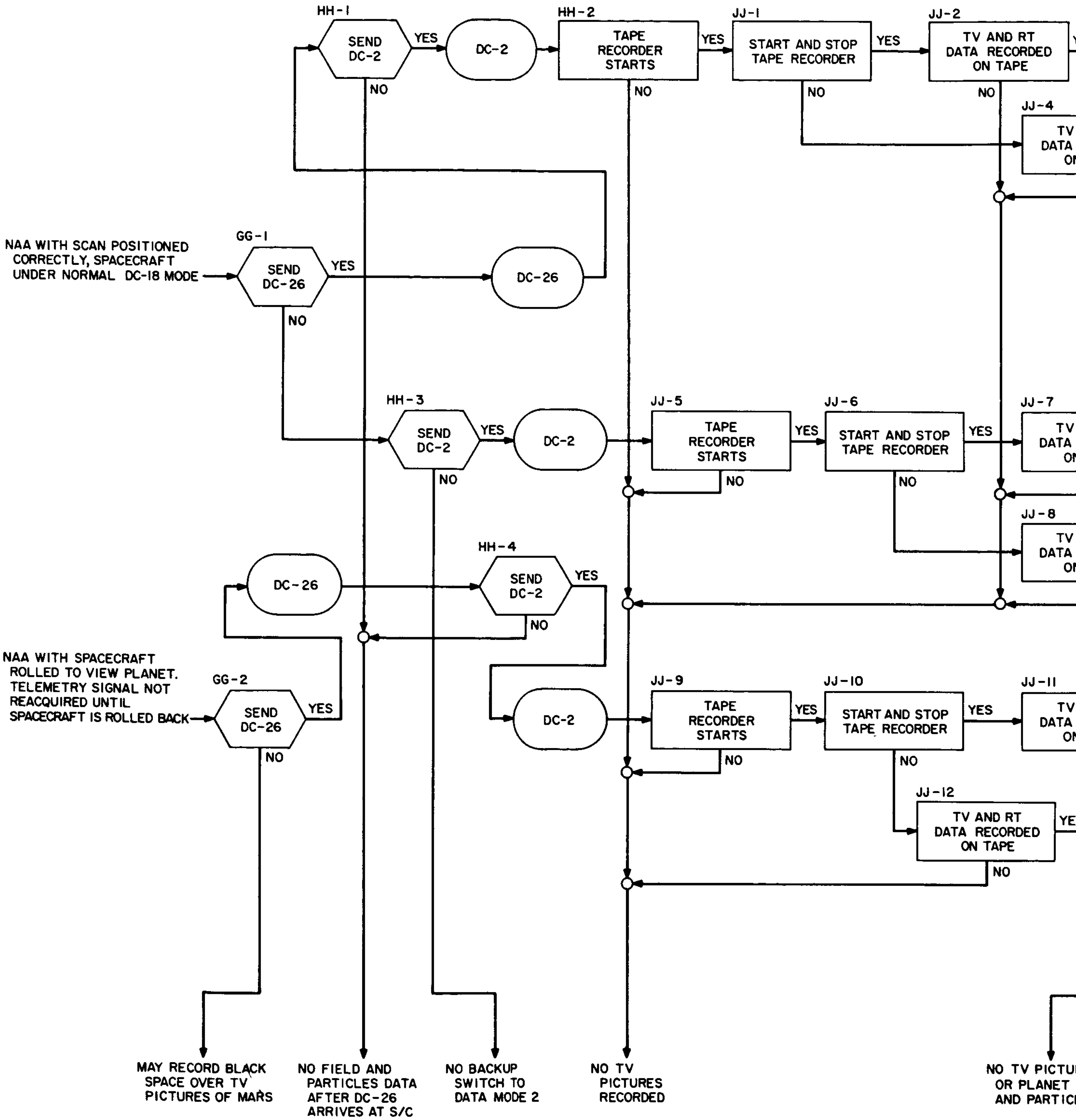
Assuming that the 100-kw transmitted power were sufficient to lock up the command loop, there would always be the possibility of transferring the spacecraft receiver to the high-gain antenna. Alternatives would have been to transmit from DSIF 13 continually at the expense of tracking data, to forego ground command participation near the picture-taking sequence, or to transmit only when needed for commands from DSIF 13 and to take the risk that conditions would not develop which required immediate command action.

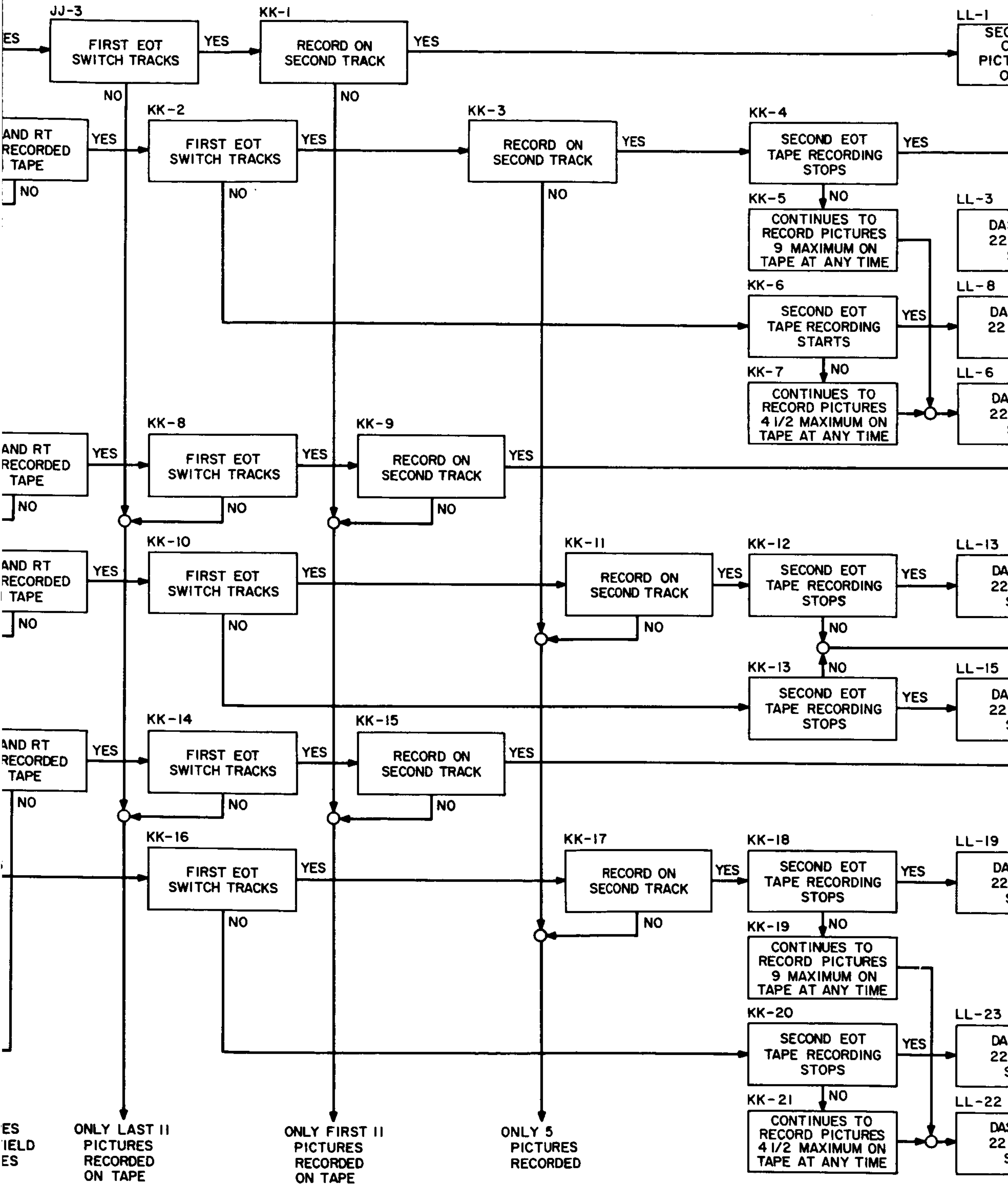
The receive-high mode, per se, had not been verified in flight, but the reinforcement of the received signal via the low-gain was due to signals coming from the high-gain antenna. Also of concern in choosing a solution was the anticipated environment at the planet based on data then available. The ability of the attitude control subsystem to maintain the proper orientation was questionable because of the unknown Martian environment.

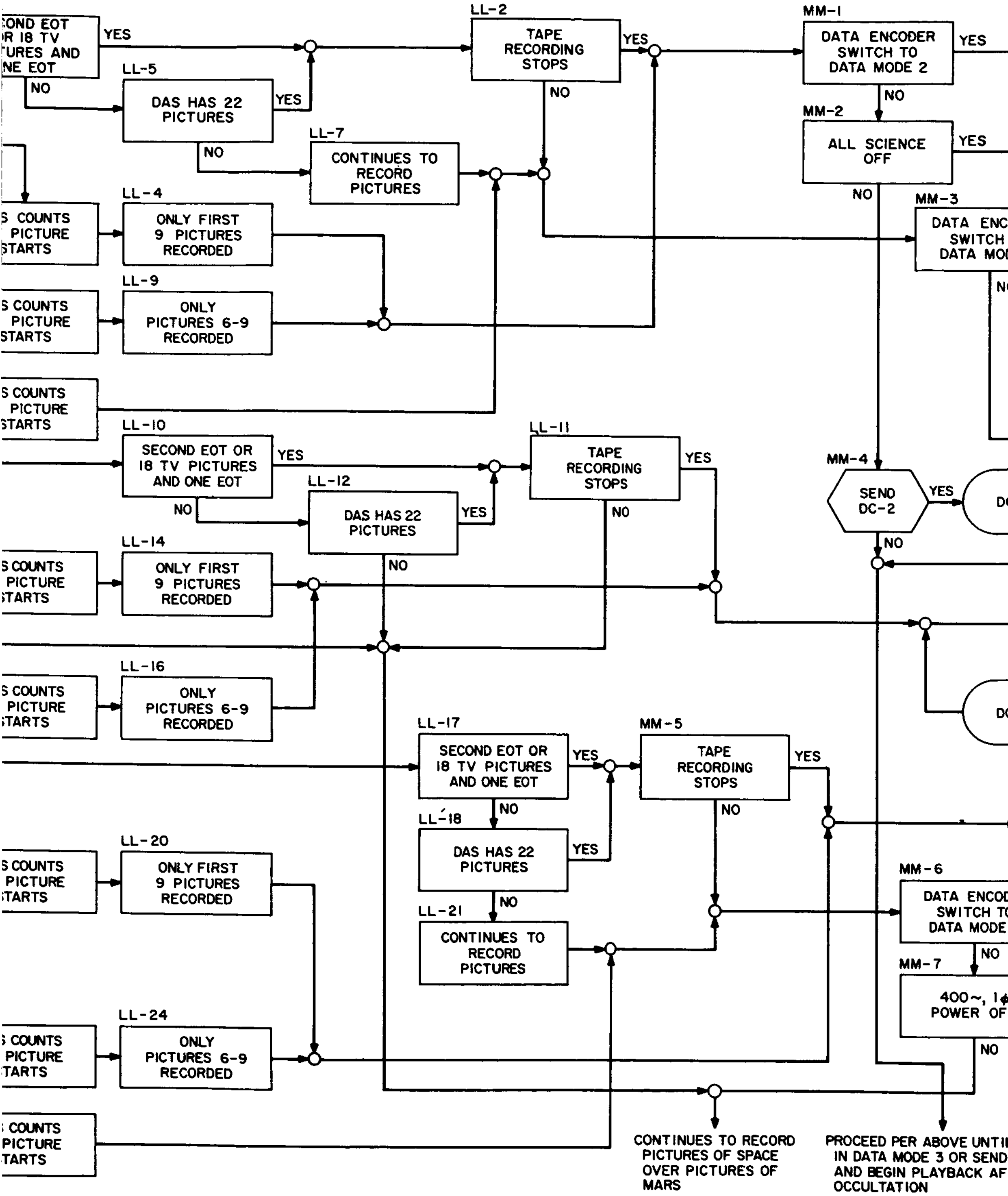
Use of DSIF 13 as a command transmitter on an as-needed basis only was highly favored. If the 10-kw stations were unable to lock up the spacecraft receiver to acquire two-way doppler data, the continuous use of DSIF 13 would be possible.

Inability to achieve command lock from any station via low-gain antenna. Existence of this condition probably would be due to a degradation or failure of the spacecraft receiver. Enough sensitivity possibly existed within the receiver to permit command lockup using a receive-high antenna configuration. Although receive-high could not be commanded from the ground, the design did provide for automatic switching of receiving antennas if two CC&S cyclics were permitted to occur without locking up two-way between them.

[21]It could not transmit and receive simultaneously.

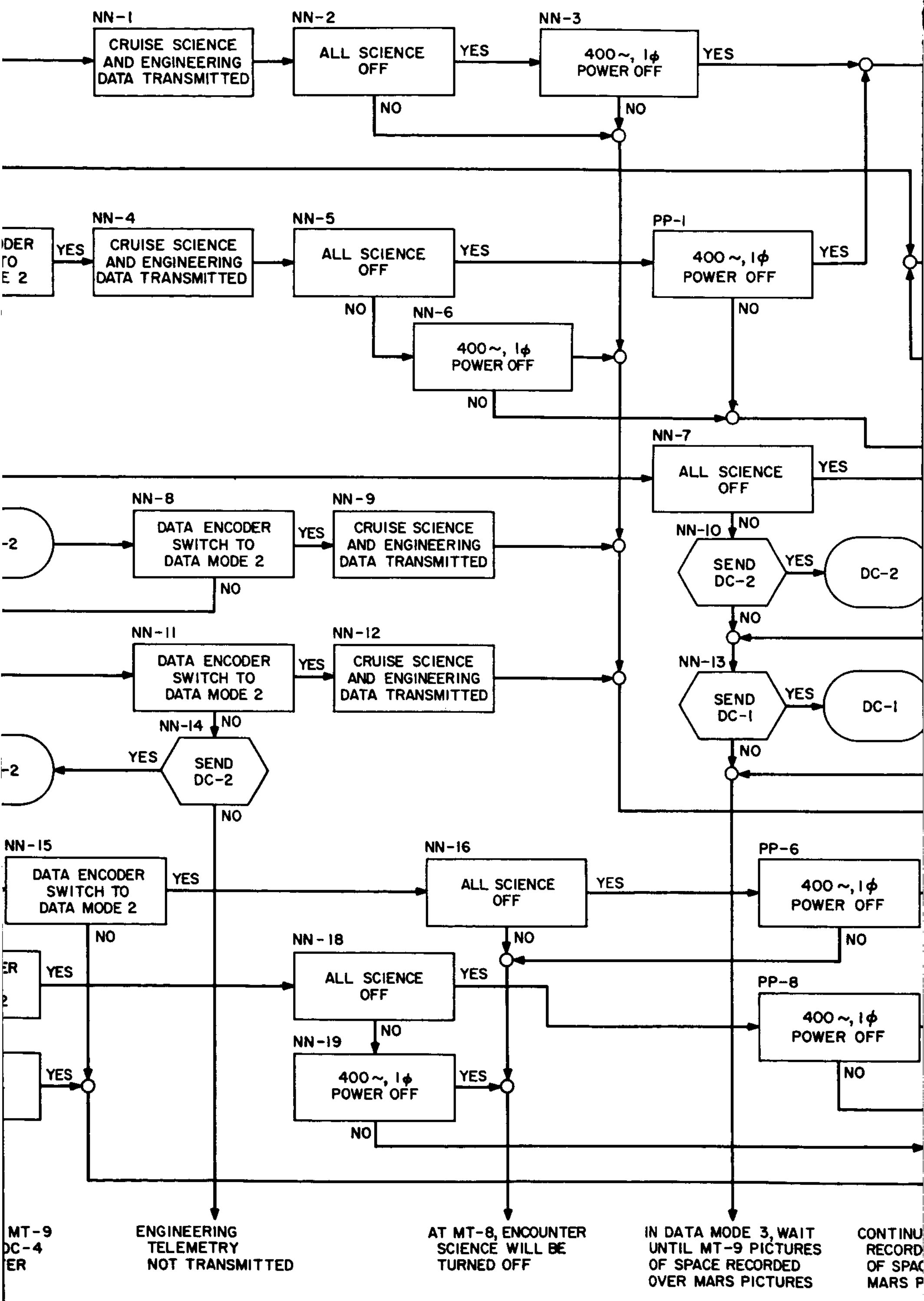

HH-1
SEND DC-2
YES
NO
DC-2
HH-2
TAPE RECORDER STARTS
JJ-1
START AND STOP TAPE RECORDER
JJ-2
TV AND RT DATA RECORDED ON TAPE
JJ-4
TV DATA
NAA WITH SCAN POSITIONED CORRECTLY, SPACECRAFT UNDER NORMAL DC-18 MODE
GG-1
SEND DC-26
DC-26
HH-3
JJ-5
JJ-6
JJ-7
JJ-8
HH-4
NAA WITH SPACECRAFT ROLLED TO VIEW PLANET. TELEMETRY SIGNAL NOT REACQUIRED UNTIL SPACECRAFT IS ROLLED BACK
GG-2
JJ-9
JJ-10
JJ-11
JJ-12
MAY RECORD BLACK SPACE OVER TV PICTURES OF MARS
NO FIELD AND PARTICLES DATA AFTER DC-26 ARRIVES AT S/C
NO BACKUP SWITCH TO DATA MODE 2
NO TV PICTURES RECORDED
NO TV PICTU
OR PLANET
AND PARTIC


JJ-3
FIRST EOT SWITCH TRACKS
KK-1
RECORD ON SECOND TRACK
LL-1
KK-2
FIRST EOT SWITCH TRACKS
KK-3
RECORD ON SECOND TRACK
KK-4
SECOND EOT TAPE RECORDING STOPS
KK-5
CONTINUES TO RECORD PICTURES 9 MAXIMUM ON TAPE AT ANY TIME
LL-3
KK-6
SECOND EOT TAPE RECORDING STARTS
LL-8
KK-7
CONTINUES TO RECORD PICTURES 4 1/2 MAXIMUM ON TAPE AT ANY TIME
LL-6
KK-8
FIRST EOT SWITCH TRACKS
KK-9
RECORD ON SECOND TRACK
KK-10
FIRST EOT SWITCH TRACKS
KK-11
RECORD ON SECOND TRACK
KK-12
SECOND EOT TAPE RECORDING STOPS
LL-13
KK-13
SECOND EOT TAPE RECORDING STOPS
LL-15
KK-14
FIRST EOT SWITCH TRACKS
KK-15
RECORD ON SECOND TRACK
KK-16
FIRST EOT SWITCH TRACKS
KK-17
RECORD ON SECOND TRACK
KK-18
SECOND EOT TAPE RECORDING STOPS
LL-19
KK-19
CONTINUES TO RECORD PICTURES 9 MAXIMUM ON TAPE AT ANY TIME
KK-20
SECOND EOT TAPE RECORDING STOPS
LL-23
KK-21
CONTINUES TO RECORD PICTURES 4 1/2 MAXIMUM ON TAPE AT ANY TIME
LL-22
YES
NO
ONLY LAST 11 PICTURES RECORDED ON TAPE
ONLY FIRST 11 PICTURES RECORDED ON TAPE
ONLY 5 PICTURES RECORDED

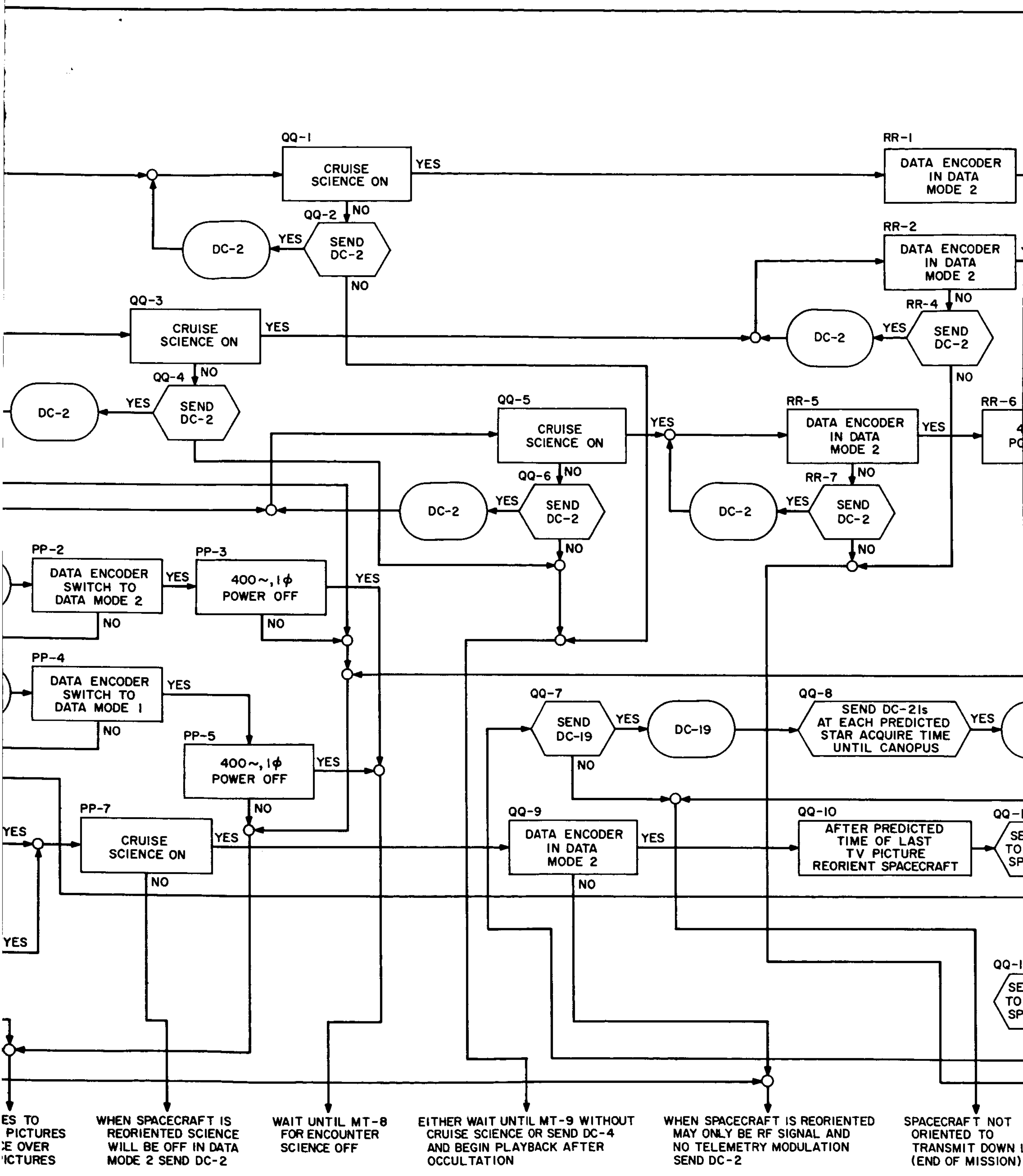


COND EOT
R 18 TV
TURES AND
NE EOT
YES
NO
LL-5
DAS HAS 22 PICTURES
YES
NO
LL-2
TAPE RECORDING STOPS
YES
NO
LL-7
CONTINUES TO RECORD PICTURES
MM-1
DATA ENCODER SWITCH TO DATA MODE 2
YES
NO
MM-2
ALL SCIENCE OFF
YES
NO
MM-3
DATA ENC
SWITCH
DATA MO
S COUNTS
PICTURE
STARTS
LL-4
ONLY FIRST 9 PICTURES RECORDED
LL-9
ONLY PICTURES 6-9 RECORDED
LL-10
SECOND EOT OR 18 TV PICTURES AND ONE EOT
YES
NO
LL-12
DAS HAS 22 PICTURES
YES
NO
LL-11
TAPE RECORDING STOPS
YES
NO
MM-4
SEND DC-2
YES
NO
LL-14
ONLY FIRST 9 PICTURES RECORDED
LL-16
ONLY PICTURES 6-9 RECORDED
LL-17
SECOND EOT OR 18 TV PICTURES AND ONE EOT
YES
NO
LL-18
DAS HAS 22 PICTURES
YES
NO
LL-21
CONTINUES TO RECORD PICTURES
MM-5
TAPE RECORDING STOPS
YES
NO
MM-6
DATA ENCOD
SWITCH TO
DATA MODE
NO
MM-7
400~, 1φ
POWER OF
NO
LL-20
ONLY FIRST 9 PICTURES RECORDED
LL-24
ONLY PICTURES 6-9 RECORDED
CONTINUES TO RECORD PICTURES OF SPACE OVER PICTURES OF MARS
PROCEED PER ABOVE UNTIL
IN DATA MODE 3 OR SEND
AND BEGIN PLAYBACK AF
OCCULTATION

182-2


NN-1
CRUISE SCIENCE AND ENGINEERING DATA TRANSMITTED
NN-2
ALL SCIENCE OFF
YES
NO
NN-3
400~, 1φ POWER OFF
YES
NO
NN-4
CRUISE SCIENCE AND ENGINEERING DATA TRANSMITTED
YES
NN-5
ALL SCIENCE OFF
YES
NO
PP-1
400~, 1φ POWER OFF
YES
NO
NN-6
400~, 1φ POWER OFF
NO
NN-7
ALL SCIENCE OFF
YES
NO
NN-8
DATA ENCODER SWITCH TO DATA MODE 2
YES
NO
NN-9
CRUISE SCIENCE AND ENGINEERING DATA TRANSMITTED
NN-10
SEND DC-2
YES
NO
DC-2
NN-11
DATA ENCODER SWITCH TO DATA MODE 2
YES
NO
NN-12
CRUISE SCIENCE AND ENGINEERING DATA TRANSMITTED
NN-13
SEND DC-1
YES
NO
DC-1
NN-14
SEND DC-2
YES
NO
NN-15
DATA ENCODER SWITCH TO DATA MODE 2
YES
NO
NN-16
ALL SCIENCE OFF
YES
NO
PP-6
400~, 1φ POWER OFF
NO
NN-18
ALL SCIENCE OFF
YES
NO
NN-19
400~, 1φ POWER OFF
YES
NO
PP-8
400~, 1φ POWER OFF
NO
YES
YES
ENGINEERING TELEMETRY NOT TRANSMITTED
AT MT-8, ENCOUNTER SCIENCE WILL BE TURNED OFF
IN DATA MODE 3, WAIT UNTIL MT-9 PICTURES OF SPACE RECORDED OVER MARS PICTURES

182-3


QQ-1
CRUISE SCIENCE ON
YES
NO
QQ-2
SEND DC-2
DC-2
RR-1
DATA ENCODER IN DATA MODE 2
RR-2
DATA ENCODER IN DATA MODE 2
RR-4
SEND DC-2
QQ-3
CRUISE SCIENCE ON
QQ-4
SEND DC-2
QQ-5
CRUISE SCIENCE ON
QQ-6
SEND DC-2
RR-5
DATA ENCODER IN DATA MODE 2
RR-6
RR-7
SEND DC-2
PP-2
DATA ENCODER SWITCH TO DATA MODE 2
PP-3
400~, 1ϕ POWER OFF
PP-4
DATA ENCODER SWITCH TO DATA MODE 1
PP-5
400~, 1ϕ POWER OFF
QQ-7
SEND DC-19
DC-19
QQ-8
SEND DC-21s AT EACH PREDICTED STAR ACQUIRE TIME UNTIL CANOPUS
PP-7
CRUISE SCIENCE ON
QQ-9
DATA ENCODER IN DATA MODE 2
QQ-10
AFTER PREDICTED TIME OF LAST TV PICTURE REORIENT SPACECRAFT
ES TO PICTURES E OVER ICTURES
WHEN SPACECRAFT IS REORIENTED SCIENCE WILL BE OFF IN DATA MODE 2 SEND DC-2
WAIT UNTIL MT-8 FOR ENCOUNTER SCIENCE OFF
EITHER WAIT UNTIL MT-9 WITHOUT CRUISE SCIENCE OR SEND DC-4 AND BEGIN PLAYBACK AFTER OCCULTATION
WHEN SPACECRAFT IS REORIENTED MAY ONLY BE RF SIGNAL AND NO TELEMETRY MODULATION SEND DC-2
SPACECRAFT NOT ORIENTED TO TRANSMIT DOWN (END OF MISSION)

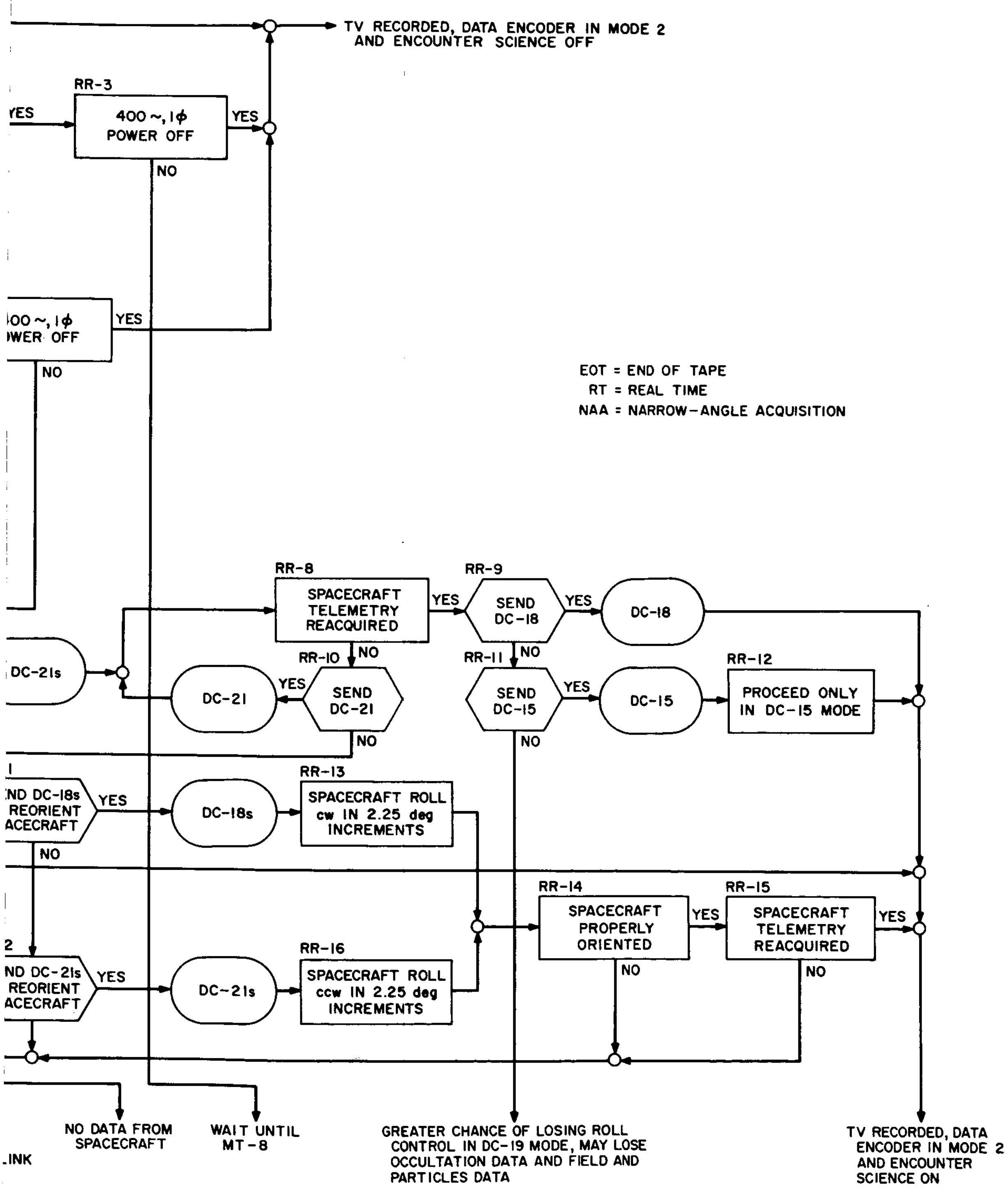


Fig. 58. Typical portion of encounter decision diagram

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The recommendation under this condition was to terminate two-way lock before CC&S MT-9, with cyclic No. 83 (1153 GMT, July 15), and resume two-way tracking after cyclic No. 84 (0634 GMT, July 18). By establishing two-way lock between cyclics, the receiver could be maintained on the high-gain antenna.

It was recognized that the competing desire for two-way tracking data for orbit determination purposes would have to be evaluated against the recommended approach should this situation arise. There was always the possibility that if significant receiver degradation had occurred, two-way doppler data via the low-gain antenna receiving configuration would not be feasible either.

b. Scan subsystem malfunction.

Search through 116-deg primary limit switch. If the scan platform searched through the primary limit switch at 116.59 deg (DAS DN 492) and the backup limit switch at approximately 115.1 deg, it would come to rest at the mechanical stop at 106°31′. There were five possible courses of action that could be taken to solve this problem.

1. The most desirable course of action was to calculate the time, taking into consideration the backlash, for the scan to search from the mechanical stop to the optimum pre-positioned clock angle. This could be done to an accuracy of within a few degrees. When the scan subsystem was turned off and then back on, it would start searching in the direction of increasing clock angle, i.e., away from the 106-deg mechanical stop. The recycle sequence had to be performed before the earliest possible WAA time to preclude the scan going to track mode before DC-24 arrived. The DC-25 of the recycle sequence would then be followed with the DC-24 at the calculated time. A second transmission time would be supplied to the ground command operator in the event of a momentary drop of lock while transmitting the first DC-24 (the second time would be after the scan had reversed at the other limit switch).

2. Another course of action was to wait until the probability of a searching scan subsystem switching to track mode at WAA was above 0.90 before performing the recycle sequence. If the scan subsystem were turned off and then back on, it would start searching toward the higher clock angles, i.e., away from this mechanical stop. For this plan, the scan PIV logic had to work, switching the scan subsystem to the track mode after searching until the planet was within the wide-angle sensor field of view.

3. The spacecraft could be rolled to point the television camera at the planet. This was undesirable because all data would be jeopardized from the time the untried gyros-on command (DC-18) reached the spacecraft, until the spacecraft was again in the normal mode. This mode unnecessarily jeopardized all orbit-determination, field-and-particles, and occultation data for the television data when there were better ways to solve the problem.

4. The spacecraft could be recycled per the recycle sequence on the hope that the scan platform would not search through these limits again. A failure that would allow the scan subsystem to search through the limit once and not the following times was conceivable, but not likely.

5. A possible course of action was to send no commands. The geometry of the scan field of view was such that the planet would come into view on the very edge of the sensor and track the planet back out. The sensor was of fairly uniform sensitivity across the field of view, but the angular semidiameter of the planet at 1800 GMT was only approximately 1.7 deg, (Fig. 59). Figure 60 shows that the Mars clock angle was 106 deg and the Mars cone angle approximately 146 deg at 1800 GMT. The edge of the sensor was at 145 deg so that, at most, 0.7 deg of the planet would be detected by the sensor. It was marginal whether the sensor would

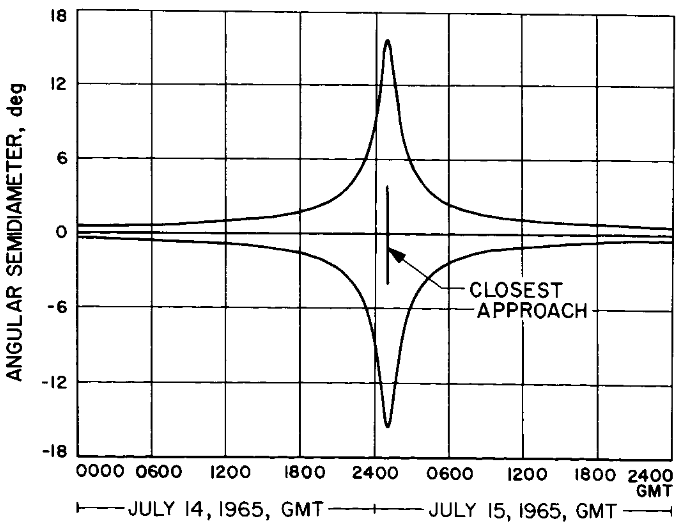


Fig. 59. Mars angular semidiameter near encounter

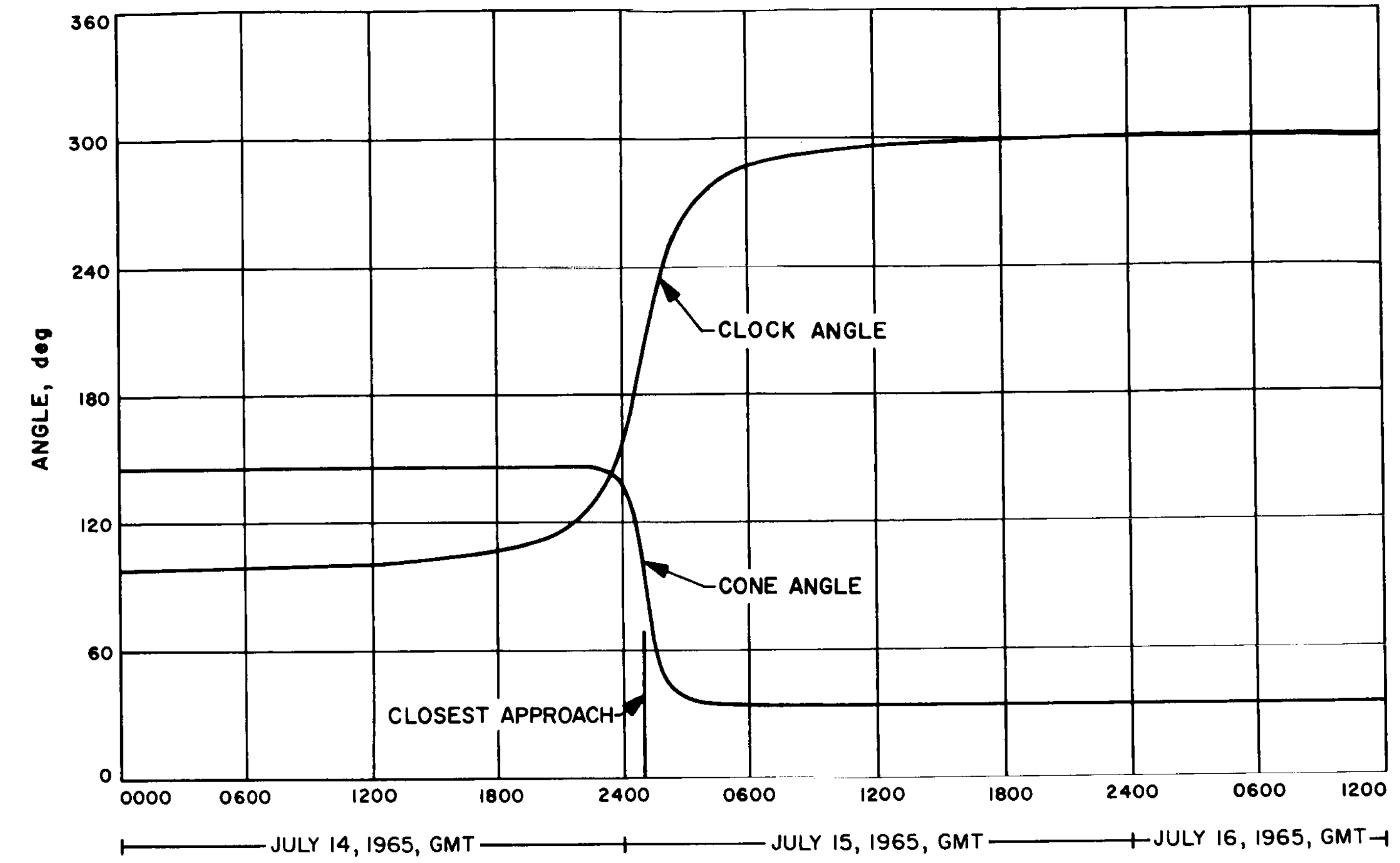


Fig. 60. Mars cone and clock angles

generate a scan planet-in-view signal and switch the scan subsystem to the track mode. Figure 61 shows the spacecraft–Mars range vs time-near-encounter.

The worst improper positioning was one in which the television subsystem would not be able to *see* the planet in the normal attitude control mode. There are three alternatives that could be followed.

Recycle. The best plan was to recycle the encounter science per the recycle sequence. Following the recycle, the scan subsystem would be allowed to search until WAA switched it to the track mode. In the event that the scan subsystem did not go to the track mode, DC-24 would be sent during the next-to-the-last scan half cycle before the planet intercepted. A backup DC-24 could also be sent on the last half-cycle.

Roll the spacecraft. Another possibility was to roll the spacecraft to allow the television to view the planet. This would require use of the gyros-on inertial mode (DC-18) and transmission of DC-21s or DC-18s, depending on the roll direction needed. This was not a desirable plan because the DC-18 mode had not been tried, and all orbit-determination, field-and-particles, and occultation data would be jeopardized by this maneuver. The RT field-and-particles information would be lost because the spacecraft would have to be rolled more than 6 deg to obtain television pictures and the spacecraft telemetry modulation could not be received after rolling more than that number of degrees.

Send no commands. The third plan, also an undesirable one, was to send no commands and suffer loss of the television pictures. The cruise science instruments were the only possible tradeoff and there was very good evidence to expect that they would be turned back on 1 min after being turned off in the recycle sequence. The DC-2 was used twice during the cover deployment exercise on February 10 to turn on cruise science, and DC-25, which actuates a redundant cruise science relay, was verified at the beginning of the encounter sequence.

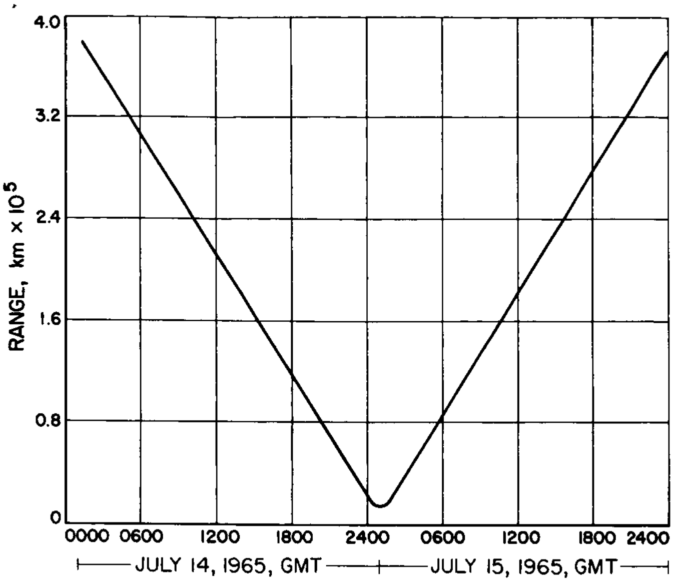


Fig. 61. Mars–spacecraft range near encounter

Search through 296.59-deg primary limit switch. If the scan platform searched through the primary limit switch located at a clock angle of 296.59 deg and the backup limit switch at approximately 298.5 deg it would search up to the mechanical stop at 13°59′. For this failure mode, recycling would not help because the scan platform would search toward this mechanical stop when recycling. The only course of action that would save the television pictures would be to roll the spacecraft, but this would jeopardize the orbit-determination, field-and-particles, and occultation data and therefore would not be done.

The logic and capability of the spacecraft made it possible to roll the spacecraft when near the planet such that the planet would come into the scan-sensor field of view and switch the scan to track mode. The spacecraft would have to be rolled in the roll search direction, DC-21s being sent either in the gyros-on inertial mode (DC-18), or at each expected star acquisition time in the DC-19 mode for the scan subsystem to track the planet away from this mechanical stop. The spacecraft could be rolled back and Canopus reacquired before the NAA time. This maneuver was risky, but did allow a tradeoff to be made. If the maneuver were successful, all objectives would be met.

Scan platform improperly positioned. There are several types of improper scan platform positioning. Figure 62 shows the relative television value vs scan platform clock angle for *Mariner IV*. Although the scan platform might not be stopped at the optimum clock angle (improperly

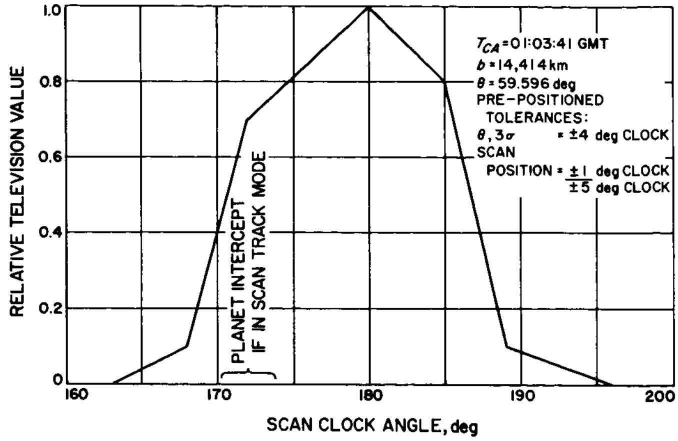


Fig. 62. Relative television values of various scan platform positions at encounter

positioned) it still might provide a higher television value than if the scan subsystem had been allowed to track the planet. The tolerance upon the scan track mode was ±1.75 deg worst-case pointing uncertainty. This was made up of a worst-case mechanical tracking error of ±0.75 deg and ±1 deg to cover the uncertainty in planet brightness distribution. This gave a worst-case television value of approximately 0.45 for the scan track mode and allowed any scan clock angle from 170.25 to 187.00 deg to provide this value or higher. However, the 3-σ uncertainty in θ was about 3 deg, equivalent to 4 deg in clock angle. The uncertainty in scan position was known to ±1 deg, giving a total equivalent-clock-angle uncertainty of ±5 deg. This provided a range from 175.25 to 182.00 deg over which the scan pre-positioning was equivalent to, or of greater relative value than, the scan track mode.

The next area of improper positioning was one in which the television camera would still be able to detect the planet, but the relative television value was lower than that of the scan track mode. Here a tradeoff had to be made between the poor quality television pictures that would result from this position and the risk associated with recycling the encounter and cruise science.

c. Video storage subsystem malfunction. Actuation of the video storage record motor at encounter-science turn-on would be indicative of some anomaly aboard the spacecraft. Several causes for this situation could be postulated.

Continuous running of the tape. As long as the EOT circuit was operative it would generate a pulse for data encoder Event Register 3. This event was the only indication that the tape was moving while in data Mode 2. In Mode 3, the event register status was not indicated, but the same indications could be obtained from the EOT and television frame bit in the DAS format. If the recorder were allowed to run it would cycle 11⅔ tape loops/hr. There were several courses of action possible for this failure mode.

1. The best procedure would be to perform a partial recycle sequence after MT-7 with a DC-24 added before DC-26. The pre-positioned angle would have to be 4 deg less than optimum in clock angle. This would allow a complete checkout of each of the other encounter subsystems before turning them off and would allow the scan to search to the optimum angle before being stopped by NAS when science would be turned back on. The last command of the recycle sequence, DC-25, would not be sent until just before the calculated normal NAA time. If the video storage subsystem operated correctly during this cycle, the normal record sequence would take place. If the video storage subsystem had operated as in the first cycle, the DC-26 at the end of the record sequence would save about 9 pictures; this was the maximum possible in this failure mode.
2. Another possible procedure would be to take no command action. As long as the EOT gave an event every 309 sec, the video storage was known to be running. If the tape were allowed to run until the DC-26 after NAA, there would be more than 110 additional tape loops, all wearing on the EOT loop foil and degrading the tape and record heads in general. If the EOT foil were worn off or became inoperative, it would no longer be possible to know the state of the record motor, which would make this course of action seem ill-advised. Without the EOT indications in this failure mode, it could not be determined whether the count-two-and-stop circuitry had stopped the tape, making recording impossible, or if the tape were still running.

Tape stopped by DAS. If the tape ran until stopped by the DAS stop command (generated periodically whenever NRT DAS power was on), this situation could not be determined until the first EOT in the record mode came up early. However, after data Mode 3 has been established, the maximum length of time from encounter-science turn-on to the first DAS stop command could be determined. This time could be any duration up to 96 sec (a maximum of less than 4 pictures could be lost). There was nothing that could be done about this failure mode, and, after the predicted early EOT had passed in the record sequence, it would be known if this failure mode existed.

Tape stopped by video storage. The tape could run continuously until the count-two-and-stop circuit had received two counts and stopped the record motor. There were two courses of action possible.

1. The best procedure was to perform a partial recycle as described previously. This removed the dependence upon the SCR which was to reset the count-two-and-stop circuit at the first start-record command from the DAS. One event in data encoder Event Register 3, about 311 sec after turn-on (and possibly a second event 309 sec later) would indicate this failure mode. In this failure mode, the recorder would record the first 9 pictures and be switched off by the count-two-and-stop circuit of video storage.
2. The other course of action was to take no command action and depend upon the SCR to reset the count-two-and-stop circuit on the first start-record command from the DAS per design.

d. High Martian radiation levels. The presence of high radiation levels in the near-vicinity of the planet would have been a significant scientific discovery. That there were radiation flux levels that could be deleterious to proper spacecraft performance was determined by tests conducted on hardware during the mission. The radiation levels required to cause malfunction of spacecraft hardware which could result in the loss of television pictures were significantly higher than the Martian radiation field predicted by leading authorities in the United States.

Most equipment, even if susceptible to high radiation fluxes, could not be prevented from experiencing the environment and affecting the spacecraft as it might. One notable exception was the Canopus sensor which could be replaced functionally with the roll gyro in the inertial mode. Studies of Canopus sensor performance under severe laboratory radiation conditions had indicated that a large margin existed between its level of susceptibility and the highest anticipated radiation flux level at Mars. On the basis of these tests it was felt that to switch to the gyro inertial mode was unwarranted because this action would subject the spacecraft to an unknown risk for which sufficient justification did not exist.

Analysis of the conditions which had warranted the turn-on of the gyro inertial mode showed that at least one of the following criteria must be satisfied:

1. The magnetic dipole moment of Mars must be at least an order of magnitude higher than that of the Earth.
2. Significant deviations from the anticipated model of a planetary trapped radiation belt must be observed during the approach to the planet.
3. Roll control via the Canopus sensor would be lost.

Criteria 1 and 2 would be reported by the SSAC encounter analysis personnel. Criterion 3 would be most readily noted by the analysis of the downlink RF signal levels by the SPAC telecommunications personnel.

e. Enigmatic loss of radio signal during encounter. The approach taken to regain the RF signal after an enigmatic loss was based on a single subsystem failure. After the *Mariner* Mars 1964 spacecraft had operated successfully for approximately 5700 hr the probability was very small that the exciter, power supply, or power amplifier for the radio subsystem would fail at the same time as the attitude control subsystem. Based on that assumption the following procedure was developed.

The station that had been transmitting to the spacecraft would transmit DC-8, DC-7, and DC-8 in that order at 2-min intervals. This procedure would take only 5 min to accomplish and would allow the determination of whether it was a radio-induced problem. If it was a radio subsystem failure associated with the exciter, the first DC-8 would transfer the redundant exciter on line in place of the failed one and the RF signal would reappear for 4 min until the second DC-8 switched the failed exciter back on. If it was a power amplifier failure, the DC-7 would transfer to the redundant power amplifier and the RF signal would be regained. The last DC-8 would switch the first exciter back on line. If nothing had been wrong with the first exciter, it would be desirable to leave the first one on line, as a high degree of confidence had been built up in this unit over the duration of the mission.

If the enigmatic loss of RF signal was caused by an attitude control loss of star reference instead of a radio subsystem problem, the first three commands would probably not have any effect on the state of the spacecraft. The spacecraft would be receiving on the low-gain antenna with a boost of 8–10 db at encounter from the interferometer effect[22] until roll control was lost. This would be desirable since, if it were a radio problem, the first three commands should provide that information; if it were an attitude control problem, the 100-kw station would be required to correct it. The 100-kw station provided the necessary increase in power to make the spacecraft capable of receiving commands in other than Canopus-referenced roll control. The procedure allowed the transmission of the first three commands followed by a station transfer to DSIF 13 immediately. After transfer, DSIF 13 would transmit for 10 min at the correct frequency and then apply command modulation for 20 min before the next command exercise would take place. During the last few minutes (two-way transmission time would be approximately 24 min) the results of the radio commands would be determined before committing to the attitude control command sequence.

Several different plans were suggested for the next sequence. The plan chosen appeared to be the most reliable and required the shortest overall time to perform: transmit DC-26 three times at 2-min intervals. This would reduce the power demand upon the solar panels to the point where the gyro turn-on transient would not throw the spacecraft into the power sharing mode even if the battery had failed. The Canopus sensor design engineer and the Guidance and Control Cognizant Engineer felt that the most likely loss of Canopus would be the result of reflected light from a dust particle. Due to the velocities with which these particles moved, the sensor could stay locked onto this reflected light for a short time before it moved out of the sensor's field-of-view. At this stage, DC-15 (Canopus gate override) would still be in effect so that the sensor would acquire many times as many objects in the DC-15 configuration as in the DC-19 configuration; therefore, the sensor would lock on a star after a few degrees of search. The Milky Way and the star Achernar are at 25–35, and 320–322 deg clock angle, respectively; each is acquirable by the sensor in the DC-19 configuration. It was very unlikely that the sensor would not acquire these stars. Due to these considerations, a sector search of ± 40 deg promised a relatively high probability of regaining the RF signal in the shortest time.

The command DC-18 would then be transmitted to turn the gyros on and place the roll gyro in inertial control. This would be followed every 30 sec for 9 min (18 commands plus the first one for a total of 19) by another DC-18. Each of these commands would move

[22]Near encounter, the bore sight of the high-gain antenna was along the Earth–spacecraft vector.

the spacecraft approximately 2.25 deg in the positive clock angle direction. Then DC-21 would be transmitted at 30-sec intervals until the RF signal reappeared, or 124 commands maximum were sent. If this sequence were used just before the picture record part of the Mars encounter, the encounter science could be turned back on with a DC-25 after 71 of the last series of DC-21s had been transmitted. This would allow the scan search mode to detect the planet without being forced through a limit switch for the remaining DC-21s. Whenever the RF signal was observed, the transmission of DC-21s would be discontinued at that time and the number of commands in transit calculated. Three would be subtracted from that number, and using the difference as the number of DC-18s required, the commands would be sent at 30-sec intervals. Approximately 24 min after the sequence was completed, the RF signal should be back on the peak of the high-gain antenna pattern, Fig. 63. If not, one DC-21 or DC-18 would be required, depending on whether or not the peak had been reached.

The spacecraft would remain on inertial roll control through the Earth occultation period and afterward, until the appropriate group could evaluate the data associated with the enigmatic loss and the new data of spacecraft performance after reacquiring the signal before committing the spacecraft to any particular attitude control mode for the duration of the mission.

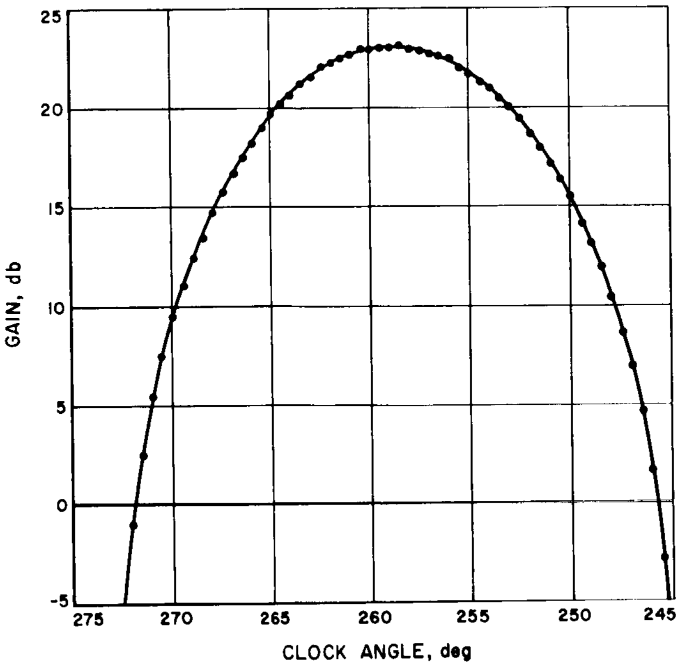


Fig. 63. *Mariner IV* high-gain antenna pattern vs clock angle for 39.5-deg cone angle

The time from the end of the Mars picture recording sequence to the beginning of the Earth occultation was calculated to be 83 min. This period would be long enough to transmit 118 DC-21s if needed,[23] plus the two-way transmission time and time to send the correcting DC-18 if needed.[24] It could take nearly 10 min in the worst case for the pre-positioned scan subsystem to find the planet.

The time from NAS to the start of picture recording would be 0–204 sec. The time required to record the pictures would be approximately 25 min. During the recording period, it would be desirable to discontinue the roll exercise. If the roll were not discontinued, after the NAS had been given, the scan head would stop its planet tracking and the television field-of-view would be rotated off the planet. This condition very possibly could produce from no pictures of Mars to a-very-unlikely four pictures of Mars. To get the four pictures, the spacecraft would have to be rolling in the correct direction across the diameter of the planet, and to start the recording sequence in the shortest time possible after NAS; the product of the three probabilities is a very small number. Thus, if the reacquiring exercise had progressed to the point where it could be completed before occultation with this added 37 min, then DC-25 would be sent at that time to start the sequence, followed 37 min later by the rest of the DC-21s. In the case where there was not sufficient time to complete reacquisition before occultation, then DC-25 would be sent at 0017 GMT and the DC-21s continued. This would only cost 30 sec for the command and might record one or more pictures of Mars which could be played back after the RF signal was reacquired.

Another approach to the problem of reacquiring in the shortest time would be to send a series of DC-21s or DC-18s without doing the sector search. This would require 160 commands or 80 min to send all the commands on 30-sec centers plus 24 min for two-way transmission time (total 104 min) to do a complete 360 deg search. This was shorter than the complete sequence time for the former plan;[25] however, it was felt that the probability of finding the RF signal during sector search was quite high and this only would take 46.5 min, a much shorter time.

[23] Started so the first one arrived at 0049 GMT, the end of the picture recording, which was transmitted at 0037.

[24] This is equivalent to 70 DC-21s and 48 DC-18s or any combination between the 118 to 70 and 0 to 49 as needed.

[25] A total of 132.5 min compared with 104 min.

Another approach studied was that of sending DC-19 and then using a series of DC-21s as roll override commands. In this plan, the spacecraft could complete a 360 deg search in as little as 44.5 min or as much as 62 min if a command arrived exactly each time the spacecraft tried to acquire a celestial object. One drawback of this plan was that because there was no telemetry being received on Earth, the roll rate would not be known, and it could be at, or any place between, these two values. This plan required observing the RF signal and sending DC-18 30 min later. This should stop the spacecraft between the extremes of where it could be. After waiting for the two-way time, it would be known which way to rotate the spacecraft. If the RF signal was passed, the number of degrees to back up could be calculated fairly accurately. However, if the signal were not reached it could be as much as 54 deg away. This would require an iterative process to again get the RF signal which over all probably would require more time than the accepted plan.

F. Post-encounter Planning

1. General

Planning for the post-encounter phase of the mission consisted of determining, evaluating, and selecting those activities which were to be attempted before the communications threshold with the spacecraft was reached. The activities under consideration were those which were needed to prolong communications with the spacecraft during 1965, to enhance the reacquisition of the spacecraft in 1967, and to exploit the utility of the *Mariner IV* as an engineering test instrument subsequent to the achievement of the primary mission objectives.

The basic ground rules utilized to make decisions relative to the post-encounter activities were that the accepted activity had to: 1) provide enhancement of the data already received; 2) protect the accumulation of a maximum amount of deep space science data in 1965, or 3) enhance the possibility of reacquiring the spacecraft in 1967. Special spacecraft tests which did not jeopardize the successful achievement of the above objectives were also permissible. The final decisions as to which activities were to be conducted rested with the *Mariner* Project Manager.

2. Operations

a. Maneuver inhibit sequence. With the acceptance of the ground rule that any operations performed on the spacecraft should enhance the possibility of acquiring the spacecraft in 1967, a maneuver inhibit sequence and a switch of the transmitter to the low-gain antenna were proposed. The maneuver inhibit sequence was designed to protect the spacecraft against a possible but unlikely failure in the CC&S that could, under normal spacecraft conditions, start a maneuver sequence. The maneuver inhibit sequence involved transmitting four commands to the spacecraft. The transmission of one of the four commands would disconnect the attitude control subsystem from the CC&S subsystem, thereby disabling the CC&S command capability to the attitude control subsystem. The other three commands would load minimum turns and motor-burn times into the CC&S maneuver shift registers as an added precaution. In this manner, even if a maneuver were somehow started by the CC&S it would have a minimum effect on the spacecraft attitude. In fact, the spacecraft could immediately reacquire the Sun after a minimum duration pitch turn was executed with no loss of solar power during the exercise.

In making the decision to execute the maneuver inhibit sequence, the advantages and risks taken in transmitting the proposed commands had to be evaluated. The risk involved in carrying out the maneuver sequence was that of the very remote possibility that the commands would be improperly interpreted by the spacecraft. The advantage was that the spacecraft would be protected from a maneuver sequence pitching the spacecraft solar panels away from the Sun which would occur if the maneuver clock were inadvertently enabled and the command sequence had not been sent. The spacecraft was designed so that the battery supplied the necessary power when the spacecraft was not Sun-oriented. However, the telemetry indications were that the battery voltage, which had been rising throughout most of the flight, was excessively high, indicating a possible battery failure. If the battery indeed had failed, a maneuver sequence would then be catastrophic. It was also anticipated that prior to reacquisition of the spacecraft in 1967, the capability of even a reasonably charged battery would have degenerated to the point where a disorientation from the Sun could not be tolerated. Therefore, the maneuver inhibit sequence of one direct and three quantitative commands was approved and successfully transmitted to the spacecraft on August 26, 1965.

b. Canopus cone-angle update. The second post-encounter command activity consisted of updating the Canopus sensor cone angle. With the spacecraft still operating 44 days after encounter, the star Canopus was about to move out of the Canopus sensor field-of-view.

So that roll stabilization, necessary for the proper high-gain antenna orientation, could be maintained and thereby enable further communications with the spacecraft, the Canopus sensor cone angle had to be updated. For, as the celestial longitude of the spacecraft changed as it orbited the Sun, the Canopus sensor field-of-view also had to be changed to keep the star Canopus in view. The required cone angle was available to the Canopus sensor only by ground command. Because the cone angle required to maintain Canopus acquisition 44 days after encounter was not required to carry out the Mars mission of *Mariner IV*, this cone angle was not included in the automatic updating sequence of the sensor by the CC&S. Therefore, on August 27, 1965, a DC-17 (change cone angle) command was transmitted to the spacecraft. The DC-17 command stepped the Canopus sensor cone angle to the next position, i.e., option number one. This allowed Canopus acquisition until March of 1966.

A continuing Canopus sensor updating sequence was proposed when an analysis of the communication requirements showed that the *Mariner IV* spacecraft could be tracked around the Sun using the DSIF 13 100-kw transmitter and the DSIF 14 210-ft parabolic antenna. With the ability to track the spacecraft around the Sun, the *Mariner IV* transmitter could be used as a planetary beacon. With the spacecraft acting as a beacon as it travelled around the Sun, periods of two-way doppler data could be used to refine the estimates of the astronomical unit (AU) and the ephemerides of Earth and Mars. Also, a better understanding of the nature and magnitude of the long term forces that act on a space vehicle such as solar pressure, meteoroid impacts, and attitude control forces could be gained by tracking the *Mariner IV* around the Sun. Since the solar pressure and attitude control forces act along the pitch, roll, and yaw axes of the spacecraft, attitude must be known to make a meaningful analysis of these forces. The forces acting on the spacecraft also affect the orbit of the spacecraft; therefore, knowledge of the spacecraft attitude is necessary before a high degree of confidence can be placed in the AU and ephemerides measurements made using the *Mariner IV*. Sun orientation presents no operational problem. But, the maintaining of Canopus acquisition continually as the spacecraft orbits the Sun requires updating the Canopus sensor cone angle. Therefore, a continuing Canopus sensor updating sequence was proposed that, if successful, would update the Canopus sensor cone angle at the proper times so that the spacecraft could maintain Canopus acquisition. Because of the knowledge that the spacecraft could be used to increase the accuracy of certain astrodynamic parameters, post-encounter planning was slanted not only toward obtaining maximum science data and reacquiring the spacecraft in 1967, but also toward using the spacecraft as a planetary beacon.

The Canopus sensor update sequence required that DC-17 (change cone angle) commands be transmitted to the spacecraft at particular points in time. To determine when the commands should be sent, a plot of Canopus cone angle as a function of time was matched with the available Canopus sensor cone angle increments, Fig. 64. Using the information from the plots, it was then possible to select the times that the Canopus sensor cone angle must be changed so that continuous lock could be maintained on the star Canopus.

The Canopus sensor update sequence along with the time schedule for the transmission of the DC-17 commands through 1966 and 1967 were presented to the *Mariner* Project Manager. The Project Manager accepted the Canopus sensor cone angle update plan and approved the transmission of the required DC-17 commands. Between August 1965 and December 1967 a total of sixty DC-17 commands will be required to step the sensor cone angle to the required values. These sixty commands would be transmitted during sixteen different time periods, with as many as seven consecutive DC-17 commands required during some updates to set the cone angle at the required value.

Because the implementation of the Canopus sensor update sequence would require the use of the experimental DSIF 13 100-kw transmitter, the proposed plan was accepted with the knowledge that conflicts could develop between normal DSIF 13 activities and maintenance and the transmission of the required DC-17 commands. Because there are overlaps between each of the Canopus sensor cone angle increments, on the order of a month or longer, the time for the transmission of the DC-17 commands can be adjusted so there is minimum interference with the regular station schedule. It should, therefore, be possible to fit the transmission of the required DC-17 commands into the DSIF schedule so that the sensor cone angle can be updated as required during the orbit of the spacecraft about the Sun.

Also of concern during Canopus sensor cone angle updates was the availability of the 210-ft antenna. Use of the 210-ft antenna made it possible to verify that two-way RF lock could be obtained with the spacecraft for the DC-17 command exercises. With the knowledge that

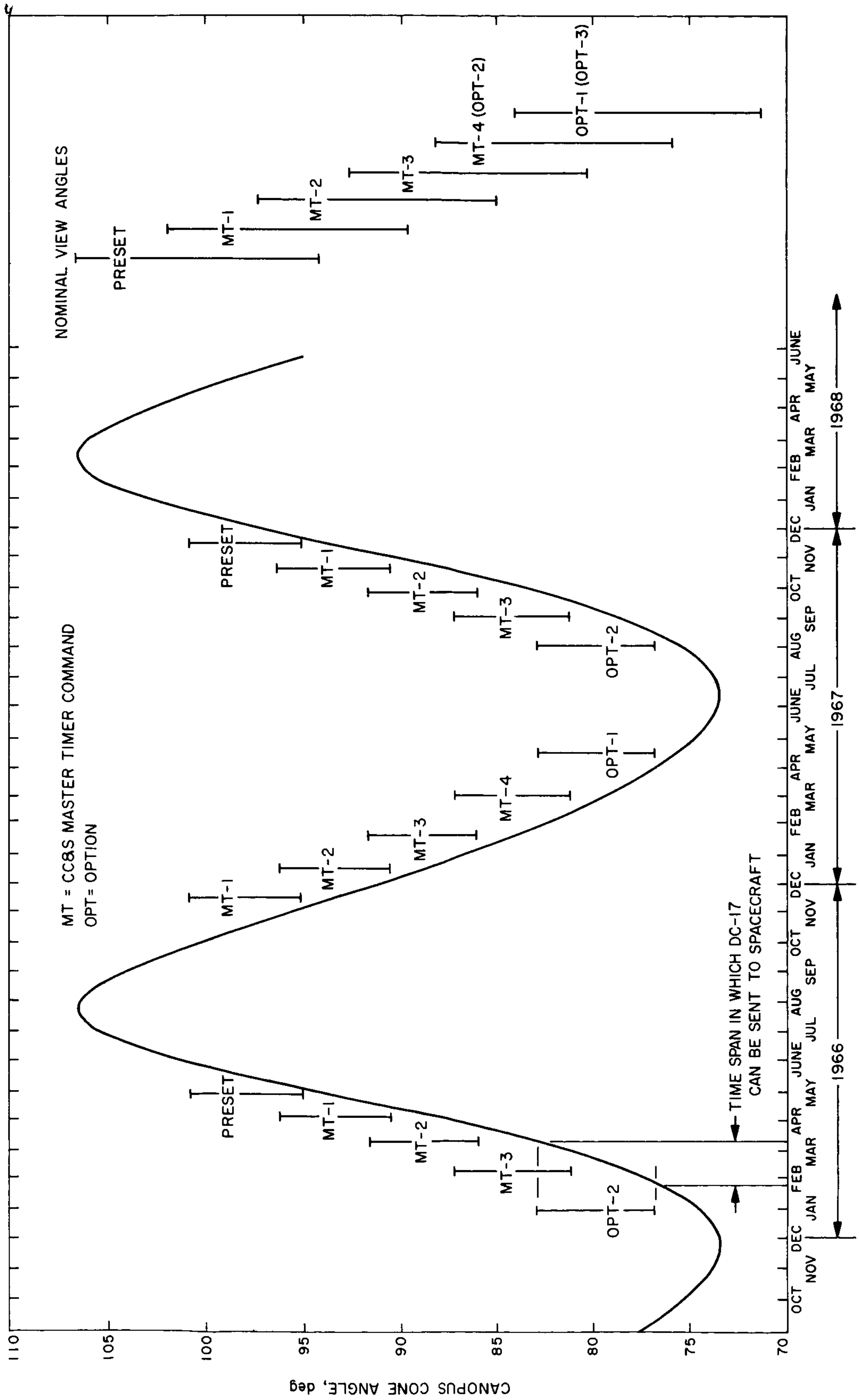


Fig. 64. Canopus sensor cone angles

two-way lock could be established, there was greater confidence in the ability of the spacecraft to receive the command messages.

c. Television haze calibration. In addition to the operational sequences proposed and carried out with the *Mariner IV* during post-encounter, a second television recording sequence was also proposed, in which would be satisfied the ground rule of enhancing data already received; it was therefore approved and carried out. The purpose of the second television recording sequence was to provide additional data on the dark response of the camera subsystem. The response of the camera to dark space and data gathered from laboratory tests was to be used in interpreting the haze or fogging observed in the *Mariner* pictures. The recording sequence required the transmission of eleven commands. The command sequence used to perform the second television record sequence, with a brief description of each command, is presented in Table 37.

Initially, the second television recording sequence was scheduled at a period in time when the television camera could be positioned to look at a first-magnitude star. The first attempt to conduct a second television recording sequence was scheduled when the star Altair (visual magnitude 0.80) was in the field-of-view of the camera. However, DSIF station problems developed early in the required command sequence and the operation had to be terminated. When the operation was finally rescheduled, no stars of sufficient brightness were in the field-of-view of the camera. Therefore, the second recording sequence television pictures were taken of black space.

Table 37. Television haze calibration

Command	Sequence
DC-25	Turns on encounter science, the television camera, video storage, and the scan platform motor.
DC-3	Switches the data encoder to Mode 3, all science data.
DC-24	Inhibits the scan drive motor.
DC-16	Starts the record sequence.
DC-2	Switches the data encoder to Mode 2 as a backup to the science data automation subsystem command to Mode 2.
DC-26	Turns off all science power and the 400-cps single-phase power to the video storage record motor and the scan subsystem drive motor.
DC-22	Switches video storage tape tracks.
DC-4	Switches the data encoder to Mode 4, the picture playback mode.
DC-28	Turns off the 2.4-kc power to the video storage playback electronics and turns on the battery charger.
DC-26	Turns off the battery charger at this time for science power is already off.
DC-2	Turns on cruise science and switches the data encoder to Mode 2.

The television recording sequence was initiated by a DC-16 command timed, using the Mode 3 data, so that each of the first five pictures would be taken at a different camera gain setting. In this way, all five of the camera gain levels could be checked by playing back just the first five pictures recorded during this second recording sequence. Because the video storage subsystem was turned off just before the EOT foil, after the second encounter data playback, only one track of data was recorded on the second television recording sequence before the second EOT was reached and the video storage record motor inhibited. Recording black space data on only one track of the recorder left encounter data available from the other track for some possible future playback requirements. A DC-22 command was required to transfer the video storage subsystem to the track containing the black space data. The playback sequence was initiated using a DC-4 command. After the first five pictures taken during the second record sequence were played back, the video storage subsystem was turned off and the spacecraft cruise configuration re-established using DC-28, -26, and -2 commands.

d. Mission Phase I termination. The final act of post-encounter was the transmission of a DC-12 (*receive low-gain* antenna, *transmit low-gain* antenna) command. The transfer of the spacecraft to the omni-antenna had a dual purpose: 1) it put the *Mariner IV* in the best condition to act as a planetary beacon; 2) it put the *Mariner IV* in the optimum condition for reacquisition of telemetry data in mid-1967. The transfer of the spacecraft from the *transmit high-* to the *transmit low-gain* antenna occurred on October 1, 1965 when the telemetry data bit-error rate was such that reasonable data acquisition was no longer feasible. The transmission of the DC-12 command was defined as the end-of-mission. For with the Earth–spacecraft distance and the high-gain antenna pointing error causing the ground received signal level to exceed the communication capability of the DSN, receipt of telemetry data will not be feasible until mid-1967 when the Earth–spacecraft distance is compatible with the reception of telemetry information using the spacecraft low-gain antenna.

3. Testing

In addition to the operational sequences proposed and carried out with the *Mariner IV* during post-encounter, twelve tests were also proposed that would use the *Mariner IV* as an engineering test instrument. Of the twelve post-encounter tests proposed, six were approved. Of the six tests approved, none required the transmission of commands to the spacecraft.

a. Approved spacecraft tests. Two of the approved tests were to evaluate the radio subsystem and two were to evaluate the command subsystem onboard the spacecraft. The primary reason these tests were approved, other than the fact they generated valuable data, was that they required no commanded change of state by the spacecraft and therefore could not jeopardize the remaining objectives of the *Mariner IV*. A discussion concerning the spacecraft radio and command tests follows.

Command threshold. This test was designed to measure, within the ability of the telemetry system of the spacecraft, the command subsystem threshold level. The information obtained would be used to determine if there had been any significant degradation in the spacecraft command subsystem since launch.

The test was performed by applying command modulation and locking-up the command receiver of the spacecraft. With the command receiver in lock, the RF carrier was modulated with a 50-kc signal. The modulation level of the signal was adjusted to give the desired RF carrier signal level. The RF carrier was decreased until the command receiver dropped lock. The RF carrier level at which the command receiver dropped lock was called the command threshold.

Command offset. This test was designed to measure the command subcarrier frequency offset that gives the minimum time to acquire command lock. The frequency offset obtained from this test could then be compared with the value obtained before launch. The comparison of these two numbers could be used to determine whether any changes had occurred in the command subsystem since launch. Also as determined by this test, if any change had occurred in the minimum lock time frequency offset the new value for the offset could be used when locking up the command subsystem during future command exercises.

This test was performed by offsetting the command carrier frequency at some predetermined value. Then command modulation was applied to the ground transmitter and the time of application noted. The spacecraft telemetry was monitored to determine when the spacecraft command subsystem obtained lock. Subtracting the two-way transmission time from the time duration between the application of command modulation and the telemetry indication of command lock gave a close approximation of the command subsystem lock time.

Spacecraft receiver threshold. This test was designed to measure, within the ability of the spacecraft telemetry system, the radio subsystem threshold level. The information obtained would be used to determine if there had been any significant degradation during the mission in the spacecraft radio subsystem.

The test was performed by modulating the RF carrier with a 10-kc signal. The spacecraft radio was phase-locked with the ground transmitter. The ground transmitter signal level at the carrier frequency was then varied by varying the modulation level of the 10-kc signal. The level of the signal at the carrier frequency was decreased until the spacecraft receiver dropped lock. The RF carrier signal level at which the spacecraft receiver drops lock was called the receiver threshold.

Spacecraft receiver SPE. This test was designed to measure the static phase error over a wider range than was obtained during normal operations. The information obtained could be used to evaluate the operation of the spacecraft receiver phase-lock-loop including the phase detector and the VCO. The data obtained during this test could be compared with similar data obtained in the laboratory to determine if there had been any degradation in the spacecraft receiver since launch.

This test was performed by offsetting the ground station transmitter frequency, so that the desired value of static phase error was developed. A plot of SPE vs frequency could then be drawn and compared with the similar plot obtained before launch.

b. Approved ground equipment tests. The spacecraft radio and command tests were the only approved tests that evaluated the spacecraft engineering subsystems, as opposed to science subsystems. The other two approved post-encounter tests evaluated ground equipment. To perform these tests, all that was required was the received signal from the *Mariner IV*. One of the tests used the received signal to evaluate an operational technique by which an increase in signal level was possible by algebraically adding the same signal received by two different systems. The low-level signal received from the spacecraft during the post-encounter period was an ideal signal to use to study and verify certain techniques for operating on signals near threshold. The other test used the low-level signal from the *Mariner IV* to evaluate an electronic signal-to-noise ratio estimator. Because the proposed tests would contribute information useful to the 1967 reacquisition and were independent of spacecraft operations, except for DSIF station scheduling, the proposed tests were approved by the *Mariner* Project Manager. A brief discussion about each of the tests follows.

Signal-to-noise ratio estimator (SNORE). The SNORE is an electronic device that could be calibrated to give direct readouts of the signal-to-noise ratio of a *Mariner* Mars 1964 type modulated signal. With the received signal from the *Mariner IV* as the input to the SNORE, signal-to-noise measurements of the spacecraft-received signal could be made. The information obtained from monitoring the *Mariner* signal with the SNORE was useful for determining operational procedures for the projected 1967 reacquisition. For example, would it be necessary to use the spacecraft high-gain antenna in 1967 to obtain useful telemetry data? If it were necessary to use the high-gain antenna for telemetry acquisition in 1967, the gyro mode of spacecraft attitude stabilization would be required so that the high-gain antenna could be pointed toward Earth. Operating the spacecraft on gyro control, the SNORE could be used to determine when the peak point of the high-gain antenna pattern was reached.

Diversity. The diversity test was designed to investigate the requirements of equipment and the logic of the technique whereby a twofold increase in signal level was possible by algebraically adding the signal, from a common source, received by two different receiver systems. The tests were conducted at the Goldstone, California station using Stations DSIF 11 and DSIF 12. Information obtained from the diversity tests could have a significant effect on the operational requirements of the *Mariner IV* 1967 reacquisition. The proposed *Mariner IV* 1967 telemetry acquisition period could be extended by proving the operational feasibility of the diversity technique. Also, the success of the diversity tests could eliminate any requirement for transferring to the spacecraft high-gain antenna for telemetry data acquisition in 1967.

c. Spacecraft tests not approved. In the early stages of post-encounter planning, before the ground rules were clearly established, the majority of the post-encounter proposals involved exercising the redundant features of the spacecraft. Also, proposals were made to activate those elements on the spacecraft, such as the ranging receiver and the receive-via-high-gain-antenna elements, that had not been activated during the pre-encounter, encounter activities.

All of the tests to check out redundancy or activate equipment aboard the spacecraft not previously used were disapproved by the Project Manager. One of the reasons for the decision not to change the state of the spacecraft, i.e., to check out redundancy or unused subsystems, was a reluctance to transfer to a unit or subsystem that had no flight history. There was the possibility that an unused subsystem might have suffered a component failure while in its standby condition, because the redundant elements of the spacecraft had not been activated previously during the mission. The possibility of a transfer from a functioning unit to a failed standby unit weighed heavily on the decision not to check out the spacecraft redundancy. Another factor to consider, when changing the spacecraft operating state, was that of power transients. The fact that power transients could cause or contribute to component failures made it desirable to keep power transients aboard the spacecraft to a minimum. Because any change of state required of the spacecraft would result in power transients and the activation of previously unused components, the disadvantages of spacecraft changes of state had to be weighed against the advantages when making a decision to check out the redundant and unused elements of the spacecraft. The decision was that a gain in knowledge sufficient to counter the risks involved would not be obtained by exercising previously unused elements of the spacecraft.

The six tests proposed to check out certain unused elements of the *Mariner IV* spacecraft, with a brief discussion about each test, are listed below.

Second trajectory-correction maneuver. The purpose of this test was to investigate the storage and restart capability of a liquid propellant propulsion system after long term storage in a space environment. A second motor burn using the *Mariner IV* would represent the only opportunity to gain experience in firing a rocket engine after it had been stored a long time in space, prior to the *Voyager* missions. It was possible that for a slight risk to the spacecraft, information about degradation of components due to radiation, micrometeoroid penetrations, 0-*g*, thermal gradients, and space vacuum could be obtained by igniting the midcourse motor a second time during post-encounter. Because monopropellant hydrazine propulsion systems similar to that on *Mariner IV* undoubtedly have future space applications, e.g., *Voyager*, the information obtained during a second trajectory correction could contribute to the design philosophy of future space missions.

Another fact considered when disapproving this test was that a second maneuver would disturb the spacecraft orbit. Although a second maneuver could reduce the Earth–probe distance, at closest approach, by 5 million km, and thereby could increase the communication tolerance in 1967, the disturbance to the trajectory would degrade the tracking data accumulated from launch to the second maneuver. As mentioned earlier data generated by tracking the spacecraft around the Sun could be used to improve the estimates of the ephemerides of Earth and Mars and the AU; it was decided that this possibility was more important to future missions such as *Voyager* than the information gained by a second maneuver.

Redundancy exercises. This test was to determine if long term storage in a space environment affected the operations of electronic components. By activating the redundant elements of the spacecraft that had not been activated during most of the flight, such as ADC/PNG B, the RF cavity amplifier, and RF exciter B, some of the effects, if any, of long term storage in deep space could be detected.

The test involved transferring the spacecraft to those redundant elements, listed above, that could be switched by ground command. The plan was to transfer to a backup unit, evaluate the spacecraft operation, and then to transfer back to the primary unit. By comparing the operational characteristics of the unit before launch with the characteristics determined during this test, changes could be detected in the characteristics of the unit resulting from long dormant periods in space.

Transfer of receiver to high-gain antenna. This test was to verify the operational status of the spacecraft when it was receiving by the high-gain antenna. Because the spacecraft was never operated during the mission in its *receive high* mode, those elements associated with this mode were never activated. To perform this test required the transmission of a command to the spacecraft that would transfer the radio receiver from the low-gain antenna to the high-gain antenna. By transferring to *receive high*, the antenna control circuitry and those states of the circulator switches associated with receiving by the high-gain antenna would be verified.

A point considered when making the decision concerning this test was that the final operation to be performed on the spacecraft was a transfer to *receive low, transmit low*. Because the spacecraft was already in the *receive low* mode as required to fulfill the remaining mission objectives, the decision was not to transfer the spacecraft to the *receive high* mode. Therefore, this test was disapproved.

Ranging. Activation of the ranging receiver was proposed so that ranging data could be accumulated and used to refine the estimates of the AU. The ranging receiver was not used during the mission because of a false-lock condition that could develop in the transponder when the ranging receiver was activated. This anomaly was discovered late in the program. A possible effect of the false-lock condition was that ground commands could not be transmitted to the spacecraft. Operating under the assumption that the post-encounter activities represented the end of the mission, the risks of a false-lock developing in the transponder were felt to be countered by the accumulation of data that could be used to refine the AU. However, in light of the ground rules that were finally decided upon, this proposal was disapproved.

High data rate. The proposal to switch the spacecraft from 8⅓ to 33⅓ bps was prompted by a 1-db drop in signal level when MT-6 (change bit rate) occurred. Investigations into the reasons for the decrease in signal level have been inconclusive. With the development of the SNORE a more accurate evaluation of any changes in signal level was possible.

This test would require the spacecraft to be switched to the 33⅓-bps data rate by ground command. The signal-to-noise ratio would then be evaluated using the SNORE. Then the spacecraft would be switched back to 8⅓ bps where any change in the signal-to-noise level would be detected by the SNORE. Using the more accurate measurements made by the SNORE, a better evaluation of the nature and the magnitude of any change in signal level that occurs with a change in bit rate could be made.

Because the change to 33⅓ bps would increase the bit error rate of the telemetry data and the 8⅓-bps rate was required to fulfill the remaining mission requirements, this test was disapproved.

Inertial control mode. Initial studies concerning the reacquisition of *Mariner IV* in 1967 determined that the Earth cone angle and the communications distance would be compatible with an acquisition attempt in the early part of that year. But, because of the communication distance, telemetry information could not be received from the spacecraft unless it were transmitting through its high-gain antenna. To use the high-gain antenna in 1967 will require rolling the spacecraft so that the antenna pattern will be pointed toward the Earth for, although the spacecraft has an acceptable cone angle, it does not have the required clock angle. To change the spacecraft clock angle requires gyro control. A DC-18 command could be used to turn on the gyros and turn off the Canopus sensor. Succeeding DC-18s or DC-21s could then be transmitted to roll the spacecraft 2.25 deg cw or ccw, respectively, and thereby position the spacecraft to the required clock angle.

An inertial mode control test was proposed, to gain operational experience in gyro control of the spacecraft attitude. Performance of this test during post-encounter before the communication tolerances were exceeded would provide information and experience that could be used during the 1967 acquisition attempt. The test would consist primarily of transferring the spacecraft to gyro control and then determining the spacecraft attitude using the change in the ground-received signal level. After the direction and magnitude of the gyro drift rates were determined, the spacecraft attitude could be controlled using DC-18 and DC-21 (2.25-deg roll turn) commands.

Further investigation into the feasibility of the inertial mode test determined that the Sun–probe distance in early 1967 will be so great that the gyro turn-on transient will, very probably, exceed the available solar panel power. For not only will the power-producing capability of the solar panels be reduced because of the large Sun–probe distance, but also the solar panels will be exposed to possible radiation damage between post-encounter and a reacquisition attempt in early 1967 that will decrease the available solar panel power. Also, the battery life expectancy will be exceeded by 1967 and, therefore, no energy storage capability can be counted on from the battery for any power transients that exceed the solar panel capability. If the gyro turn-on transient exceeds the available power, the spacecraft power subsystem will be forced into a low-voltage state from which the spacecraft cannot recover. Therefore, inertial control in early 1967 will not be attempted. Thus, it was not necessary to perform an inertial mode control test during post-encounter in preparation for inertial control in early 1967.

The desirability of controlling the spacecraft with the gyros presents itself again in mid-1967. At this time the spacecraft will be close enough to the Sun so that sufficient solar panel power will be available to support a gyro turn-on transient, if the panels have not suffered extensive radiation damage. Controlling the spacecraft roll attitude during the 1967 acquisition attempt, if successful, can extend the acquisition period and also provide operational experience in controlling spacecraft attitude using gyros and ground commands. The use of gyro control, if the spacecraft is successfully acquired in 1967, will depend on the facts available and the evaluation of the telemetry information received from the spacecraft at that time.

APPENDIX A

Chronology

Table A-1. Flight history

Date	Time, GMT	Event
November 28, 1964	14:22:01	Liftoff
	14:24:15	Mark 1
	14:24:17	Mark 2
	14:27:00	Mark 3
	14:27:18	Mark 4
	14:27:21	Mark 5
	14:27:23	Mark 6
	14:28:14	Mark 7
	14:30:39	Mark 8
	15:02:53	Mark 9
	15:04:28	Mark 10
	15:07:09	Mark 11: RF power up, cruise science on, plasma probe high voltage on, video storage launch mode off, CC&S relay hold off; Channel F telemetry off.
	15:07:10	Mark 12: pyrotechnics armed, separation-initiated timer started, isolation amplifier for Channel F telemetry disabled, attitude control on.
	15:10:10	Deploy solar panels and unlatch scan platform via separation-initiated timer.
	15:15:00	CC&S L-1 command (deploy solar panels); observed in data at 15:15:06.
	15:17:35	Spacecraft left Earth's shadow.
	15:19:00	CC&S L-2 command (turn on attitude control subsystem): start Sun acquisition; observed in data at 15:19:05 GMT.
	15:30:57	Sun acquisition complete, start of magnetometer calibrate roll.
	16:59:00	Data encoder rate 3/4 skipped 409, 410, then reset and advanced one count. During the next few minutes (until 1716) the plasma probe and cosmic dust detector also exhibited anomalous behavior.
	17:00:28	Data encoder rate 1 skipped at word 119 which made science PN code start one word early.
	17:00:40	Data encoder rate 2 reset from 208, skipping word 209.
	17:00:53	Data encoder rate 3/4 skipped from word 411 to 400.
	17:04:00	Data encoder rate 2 skipped position 206 from 207.
	17:04:13	Data encoder rate 2 reset to 200 from 207, skipping 208 and 209.
	17:04:25	Data encoder rate 3/4 reset to 400 from 401, skipping 402 through 419.
	17:04:37	Data encoder rate 1 sync occurred one word early. *Probably* 100 deck reset at one of the two 8 positions in the science deck. Data encoder rate 2 reset to 200 from 201, skipping 202 through 209.
	17:04:50	Data encoder rate 3/4 reset to 410 from 400, skipping 401 through 409.
	17:05:15	Data encoder rate 1 sync occurred one word early. Again, *probably* 100 deck reset at 8 position. Data encoder rate 2 reset to 200.
	17:05:27	Data encoder rate 3/4 reset to 400.

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
November 28, 1964	17:06:05	Data encoder rate 1 sync occurred one word early. *Probably* 100 deck reset at position 8. Data encoder rate 2 reset to 200 from 202, skipping 203 through 209.
	17:06:17	Data encoder rate 3/4 reset to 410 position.
	17:06:42	Data encoder rate 1 sync occurred one word early. *Probably* 100 deck reset at position 8. Data encoder rate 2 reset to 200 from 202, skipping 203 through 209.
	17:07:06	Data encoder rate 1 skipped 109 causing science PN code to occur one word early. Data encoder rate 2; 200 sync not observed, but 220 occurred at 110.
	17:07:18	Data encoder rate 3/4 reset to 410 position.
	17:07:43	Data encoder rate 1 sync occurred one word early. *Probably* 100 deck reset at position 8. Data encoder rate 2 reset to 200 from 202, skipping 203 through 209.
	17:07:55	Data encoder rate 3/4 reset to 400 position.
	17:09:28	Data encoder rate 2 skipped 208, going from 207 to 209. Data encoder rate 3/4 reset to 410 position.
	17:11:21	Data encoder rate 2 skipped 208, going from 207 to 209. Data encoder rate 3/4 reset to 400 position.
	17:15:33	Data encoder rate 2 reset to 200 from 208, skipping 209.
	17:16:36	Data encoder rate 2 skipped 205, going from 204 to 206. Data encoder rate 3/4 reset to 410 position.
	17:18:29	Data encoder rate 2 reset to 200 during first 9 words of engineering frame since 205 is correct but 220 occurred during 225 time indicating a reset from position 5 to 0 on rate 2. Data encoder rate 3/4 reset to 400 position.
	17:19:20	Data encoder rate 2 skipped position 204 but not 214 and 224. It read out 205, 214, 224. Indicates three rate-2 decks out of step.
	17:19:33	Data encoder rate 2 read out 206, 215, 224. Indicates three rate-2 decks out of step.
	17:19:45	Data encoder rate 2 read out 207, 216, 226. Indicates three rate-2 decks out of step.
	17:19:58	Data encoder rate 2 read out 208, 217, 227. Indicates three rate-2 decks out of step.
	17:20:11	Data encoder rate 2 reset to 200, skipping 209.
	17:20:23	Data encoder rate 2 reset to 200, i.e., 200 deck sync again.
	17:20:35	Data encoder rate 3/4 reset to 410 position.
	17:21:26	Data encoder rate 2 skipped position 205 and succeeding channels were incorrect until the deck was reset at Channel 209. Channels 219 and 229 were skipped as a result of the reset.
	17:22:28	Data encoder rate 3/4 reset, 410 skipped.
29	06:59:00	CC&S L-3 command (transfer solar pressure vanes from erect to operate and turn on Canopus sensor): spacecraft into roll search. Observed in data at 06:59:00 GMT.
	07:10:03	Gyros off, spacecraft roll acquired in the vicinity of the star Markab.
	13:12:34	Roll acquisition lost, gyros on.
	13:26:15	Gyros off, spacecraft roll acquired the star Regulus.
	16:59:00	CC&S CY-1 No. 1 on schedule. Observed in data at 16:59:14 GMT.
30	09:13:00	DC-21 (roll override) transmitted, gyros on and spacecraft in roll search.
	09:20:56	Gyros off, spacecraft roll acquired the star Naos.

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
November 30, 1964	10:45:00	DC-21 transmitted. Gyros on but spacecraft telemetry did not show roll search.
	10:46:00	Gyros off. Spacecraft had rolled slightly, acquiring star γ Velorum.
	10:57:09	DC-21 transmitted. Gyros on and spacecraft in roll search.
	10:59:38	Gyros off, spacecraft roll acquired the star Canopus.
	13:41:00	A sharp transient was noted in the roll position telemetry. Some cross-coupling into pitch and yaw was observed. The transient was not severe enough to cause the loss of Canopus acquisition.
December 1	02:04:41	Absorptivity standard white sample stepped as expected.
2	10:09:00	Canopus acquisition lost, gyros on. Telemetry indicated a clockwise (cw) roll. Star intensity increased to the high gate level, then dropped immediately to about 0.3 times Canopus. Reacquisition of Canopus occurred and the gyros went off normally.
	11:39:05	CC&S CY-1 No. 2 on schedule. Observed in data at 11:39:10 GMT.
	11:45:42	Absorptivity standard polished-aluminum sample stepped as expected.
4	13:05:00	QC-1-1 (pitch turn duration and polarity) transmitted.
	13:10:00	QC-1-2 (roll turn duration and polarity) transmitted.
	13:15:00	QC-1-3 (motor burn duration) transmitted.
	13:45:00	DC-29 (first maneuver arm) transmitted.
	14:05:00	DC-14 (release maneuver inhibit) transmitted.
	14:35:00	DC-27 (start midcourse maneuver) transmitted.
	14:36:31	Approximately 52 sec after the execution of DC-27 by the spacecraft, roll acquisition was lost and the spacecraft went into roll search. As a result, the maneuver attempt was aborted by ground command.
	14:47:31	DC-13 (inhibit maneuver) transmitted. Spacecraft in normal roll search.
	14:55:52	Gyros off, spacecraft roll acquired a non-Canopus star.
	15:22:00	DC-21 (roll override) transmitted. Gyros on.
	15:26:32	Gyros off, spacecraft acquired another non-Canopus star after an estimated 3-deg roll.
	15:32:00	DC-21 transmitted. Gyros on and spacecraft in roll search.
	15:35:08	Aborted pitch-turn start indication. Observed in data at 15:35:11 GMT.
	15:36:12	Gyros off, spacecraft acquired a third non-Canopus star. Virtually no position change.
	15:39:19	Aborted pitch-turn stop indication. Observed in data at 15:39:24 GMT.
	15:57:08	Aborted roll-turn start indication. Observed in data at 15:57:15 GMT.
	16:02:00	DC-21 transmitted. Gyros on and spacecraft in roll search.
	16:06:39	Gyros off, spacecraft acquired another non-Canopus star after about 4-deg roll.
	16:11:20	Aborted roll-turn stop indication. Observed in data at 16:11:32 GMT.
	16:19:08	Aborted motor-burn start indication. Observed in data at 16:19:18 GMT.
	16:19:29	Aborted motor-burn stop indication. Observed in data at 16:19:31 GMT.
	16:25:08	CC&S maneuver relay reset and Sun reacquisition started. Observed in data at 16:25:11 GMT.
	17:54:08	Midcourse counter overflow. Observed in data at 17:54:15 GMT.
	22:40:00	DC-21 (roll override) transmitted. Gyros on and spacecraft in roll search.

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
December 4, 1964	22:44:32	Gyros off. Non-Canopus star acquired after 2-deg roll.
	23:04:00	DC-21 transmitted. Gyros on and spacecraft in roll search.
	23:05:00	DC-21 transmitted. Spacecraft still in roll search.
	23:06:00	DC-21 transmitted. Spacecraft still in roll search.
	23:28:28	Gyros off, spacecraft roll acquired to the star Regulus.
	23:39	Roll transient found during nonreal-time data analysis.
	23:40:00	DC-21 transmitted. Gyros on and spacecraft in roll search.
	23:52:07	Gyros off, spacecraft roll acquired to the star Naos.
	23:57:00	DC-21 transmitted. Gyros on and spacecraft in roll search.
	23:58:00	DC-21 transmitted. Spacecraft still in roll search.
5	00:02:44	Gyros off. Spacecraft roll acquired the star Canopus.
	06:19:09	CC&S CY-1 No. 3 on schedule. (Not observed in data.)
	13:05:00	QC-1-1 (pitch turn duration and polarity) transmitted.
	13:10:00	QC-1-2 (roll turn duration and polarity) transmitted.
	13:15:00	QC-1-3 (motor burn duration) transmitted.
	13:45:00	DC-29 (first maneuver arm) transmitted.
	14:05:00	DC-14 (release maneuver inhibit) transmitted.
	14:25:00	DC-27 (start maneuver) transmitted. Gyros on and data encoder to Mode 1.
	15:25:10	All axes to inertial control, start of ccw pitch turn. Observed in data at 15:25:10 GMT.
	15:28:53	End of ccw pitch turn. Observed in data at 15:28:54 GMT.
	15:47:11	Start of cw roll turn. Observed in data at 15:47:12 GMT.
	16:01:20	End of cw roll turn. Observed in data at 16:01:21 GMT.
	16:09:11	Start of motor burn. Observed in data at 16:09:12 GMT.
	16:09:31	End of motor burn. Observed in data at 16:09:34 GMT.
	16:15:11	CC&S maneuver relay start of Sun reacquisition. Observed in data at 16:15:12 GMT.
	16:21:07	Sun acquisition complete, spacecraft in roll search.
	16:44:36	Gyros off. Spacecraft roll acquired the star γ Velorum.
	16:52:00	DC-21 (roll override) transmitted. Gyros on and spacecraft in roll search.
	16:55:00	Gyros off. Spacecraft roll acquired Canopus.
	17:44:11	Midcourse counter overflow. Observed in data at 17:44:21 GMT.
6	02:03:04	Data encoder rate 2 reset, skipped 209; 400 deck not affected. The cosmic dust detector and the plasma probe also showed anomalous behavior. Later analysis showed a degradation in the performance of the plasma probe beginning approximately 2 hr before the first deck skip and continuing until after the last skip observed on December 7, at 06:48:36 GMT. All engineering measurements were normal.
	06:21:34	Data encoder rate 3/4 reset, skipping from 416 to 401. Implied reset before 06:21:22 GMT.
	09:02:14	Data encoder rate 2 skipped 207, read out 208, 217, 227; 400 deck reset. Next reading 201 = 90. Skipped 418, 419, and 400.

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
December 6, 1964	13:29:46	Data encoder rate 3/4 reset, skipped 408, 409, and 410.
	13:42:10	Data encoder rate 3/4 and rate 2 reset, skipped 209.
7	03:41:08	Data encoder rate 1 reset, skipped 119 and 206.
	06:48:36	Data encoder rate 3/4 reset.
	12:29:41	Gyros on, spacecraft in roll search. Telemetry prior to the loss of Canopus indicated that the brightness was increasing and the tracker was seeing a large ccw error signal. Automatic reacquisition to the star γ Velorum was normal.
8	00:59:13	CC&S CY-1 No. 4 on schedule. Observed in data at 00:59:14 GMT.
	10:43	Roll transient found during NRT data analysis.
9	05:35:36	Gyros on, spacecraft in roll search during one sample of the roll position channel. Reacquisition to γ Velorum was normal. Apparent on-time of the gyros was 200 sec.
	22:14:36	Acquisition of the star γ Velorum lost, gyros on. The star intensity channel showed one sample of 8-times-Canopus intensity rather than the normal 0.12 times Canopus for γ Velorum. Reacquisition to γ Velorum was immediate, but the total on-time of the gyros was far shorter than normal (165 sec compared with a normal 200 sec). Later analysis showed this to be the case whenever reacquisition occurred within 2 sec of the initiation of search.
10	05:13:10	Absorptivity standard black sample stepped as expected.
	19:39:17	CC&S CY-1 No. 5 on schedule. Observed in data at 19:39:25 GMT.
13	14:09:00	DC-7 (switch power amplifiers) transmitted. TWT amplifier into warmup mode and DSIF receiver out-of-lock.
	14:10:38	DSIF receiver back in lock; TWT in standard mode at 40.2 dbm output.
	14:19:21	CC&S CY-1 No. 6 on schedule. Observed in data at 14:19:33 GMT.
	20:48:27	Acquisition of the star γ Velorum lost, gyros on. Reacquisition was immediate, and the total on-time of the gyros was 75 $\pm$ 2 sec.
14	22:51	Roll transient found during nonreal-time data analysis.
16	08:59:26	CC&S CY-1 No. 7 on schedule. Observed in data at 08:59:30 GMT.
17	05:36:15	Brightness transient noted on the Canopus intensity channel. The transient was not severe enough to produce a violation of the high intensity gate, hence no loss of acquisition occurred.
	07:16:43	Acquisition of the star γ Velorum lost, gyros on. Reacquisition of γ Velorum was normal. Total gyro on-time was 203 sec.
	16:00:00	DC-21 (roll override) transmitted. Gyros on and spacecraft in roll search.
	16:06:22	Gyros off, spacecraft roll acquired the star Canopus (the roll reference had been γ Velorum as a result of a loss of Canopus and an automatic star reacquisition sequence).
	17:30:00	DC-15 (Canopus gate override) transmitted.
19	03:39:30	CC&S CY-1 No. 8 on schedule. Observed in data at 03:39:39 GMT.
21	22:19:34	CC&S CY-1 No. 9 on schedule. Observed in data at 22:19:35 GMT.
24	16:59:38	CC&S CY-1 No. 10 on schedule. Observed in data at 16:59:48 GMT.
26	20:04:04	Recovery transient observed in roll position telemetry. The star intensity channel was not sampled at the appropriate time to observe a possible brightness transient, but all indications were that one occurred.

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
December 26, 1964		Changes in TWT helix current were noted throughout the day. In general the changes involved an increase of 0.4 to 0.5 ma. Discussion with the vendor and with the cognizant personnel for the Telstar satellite indicated that this was a normal, though little understood, phenomenon of TWT amplifiers.
27	11:39:42	CC&S CY-1 No. 11 on schedule. Observed in data at 11:39:53 GMT.
30	06:19:46	CC&S CY-1 No. 12 on schedule. Observed in data at 06:19:50 GMT.
31	00:53:46	Recovery transient observed in roll position telemetry. No brightness transient observed. Recovery was normal.
January 1, 1965	15:32:55	Absorptivity standard black sample stepped as expected.
2	00:59:51	CC&S CY-1 No. 13 on schedule. Observed in data at 00:59:59 GMT.
3	16:59:54	CC&S MT-6 command (switch bit rates) transferred, telemetry rate to 8⅓ bits/sec (bps).
4	08:00:11	Absorptivity standard black sample stepped as expected.
↓	18:49:14	Recovery transient observed in roll position telemetry. No brightness transient was observable.
	19:39:56	CC&S CY-1 No. 14 on schedule. Observed in data at 19:40:03 GMT.
5	06:31:46	Absorptivity standard aluminum-silicon sample stepped.
6	11:16:27	Recovery transient observed in roll position telemetry. No brightness transient was observable.
↓	21:35:25	Absorptivity standard white sample stepped as expected.
7	14:20:01	CC&S CY-1 No. 15 on schedule. Observed in data at 14:20:17 GMT.
8	05:01:15	Recovery transient observed in roll position telemetry. No brightness transient was observable.
9	16:55:53	Absorptivity standard aluminum-silicon sample stepped as expected.
↓	21:53:30 to 24:00:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
10	00:00:00 to 03:42:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
↓	09:00:06	CC&S CY-1 No. 16 on schedule. Observed in data at 09:00:26 GMT.
	21:48:08 to 24:00:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
11	00:00:00 to 03:37:14	No telemetry data; DSIF 51 assigned to *Ranger* Project.
↓	21:42:10 to 24:00:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
12	00:00:00 to 03:36:24	No telemetry data; DSIF 51 assigned to *Ranger* Project.
↓	06:24:27	Absorptivity standard black sample stepped as expected.
	21:36:00 to 24:00:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
13	00:00:00 to 03:27:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
↓	03:40:11	CC&S CY-1 No. 17 on schedule. Observed in data at 03:40:56 GMT.

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
January 13, 1965	14:13:51	A large roll position transient was observed, time correlated with science activity. Minor transients were observed in pitch and yaw which appear to be cross-coupling effects. A change of 1 DN was observed in the Canopus brightness channel.
15	22:20:16	CC&S CY-1 No. 18 on schedule. Observed in data at 22:20:46 GMT.
16	05:44:55	Absorptivity standard polished-aluminum sample stepped as expected.
18	17:00:21	CC&S CY-1 No. 19 on schedule. Observed in data at 17:00:55 GMT.
20	23:46:00	Recovery transient observed in roll telemetry. No brightness transient was observable.
21	05:59:24	Absorptivity standard white sample stepped as expected.
	11:40:27	CC&S CY-1 No. 20 on schedule. Observed in data at 11:41:07 GMT.
23	00:00:45	Absorptivity standard black sample stepped as expected.
	05:45:09	Absorptivity standard polished-aluminum sample stepped as expected.
24	06:20:32	CC&S CY-1 No. 21 on schedule. Observed in data at 06:21:16 GMT.
27	01:00:38	CC&S CY-1 No. 22 on schedule. Observed in data at 01:01:28 GMT.
	07:44:28	Absorptivity standard aluminum-silicon sample stepped as expected.
29	19:40:44	CC&S CY-1 No. 23 on schedule. Observed in data at 19:40:48 GMT.
February 1	14:20:51	CC&S CY-1 No. 24 on schedule. Observed in data at 14:21:00 GMT.
	16:24	Absorptivity standard white sample stepped as expected.
3	02:01:10	Absorptivity standard black sample stepped as expected.
↓	12:00	Spacecraft command loop locked up. No commands were transmitted to the spacecraft. The spacecraft command subsystem operated normally throughout the exercise.
↓	18:22:19	An increase was observed in the midcourse propellant pressure. It was believed that the pressure increase was due to incompatibility of the bladder with the hydrazine propellant. All other indications were normal.
4	09:00:57	CC&S CY-1 No. 25 on schedule. Observed in data at 09:01:11 GMT.
	13:55	Roll transient found during NRT data analysis.
5	04:16	Roll transient found during NRT data analysis.
↓	07:21:39	Absorptivity standard polished aluminum sample stepped as expected.
↓	18:00:00	A Class 2 solar flare was detected by the science instruments on board the spacecraft. There was no indication of any effect upon the engineering subsystems.
7	03:41:04	CC&S CY-1 No. 26 on schedule. Observed in data at 03:41:23 GMT.
↓	05:35:00	A large transient was observed in roll position. Disturbances were also seen in pitch and yaw. Recovery was normal.
↓	18:24:05 to 24:00:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
8	00:00:00 to 01:29:04	No telemetry data; DSIF 51 assigned to *Ranger* Project.
↓	02:32:56	Absorptivity standard aluminum-silicon sample stepped as expected.
↓	18:19:50 to 24:00:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
February 9, 1965	00:00:00 to 01:16:46	No telemetry data; DSIF 51 assigned to *Ranger* Project.
	03:37:25	Absorptivity standard black sample stepped as expected.
	18:15:40 to 24:00:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
	22:21:12	CC&S CY-1 No. 27 during DSIF 51 pass. Not observed in data.
10	00:00:00 to 03:18:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
	18:11:15 to 24:00:00	No telemetry data; DSIF assigned to *Ranger* Project.
11	00:00:00 to 01:08:41	No telemetry data; DSIF 51 assigned to *Ranger* Project.
	02:54:15	After an apparently normal command-loop lockup before the start of the science cover drop exercise, the command subsystem dropped lock. Command came back in lock at 03:00:08 GMT and performed normally for the balance of the exercise.
	03:29:29	DC-3 (switch to Mode 3) transmitted. Spacecraft telemetry to Mode 3.
	03:36:13	DC-2 (switch to Mode 2) transmitted. Spacecraft telemetry to Mode 2.
	03:53:15	DC-26 (encounter science off) transmitted. Spacecraft cruise science off, battery charger off, boost mode enabled.
	03:53:15	Data encoder rate 2 reset to 200 from 206 and rate 3/4 reset to 400 from 416.
	04:15:51	DC-2 transmitted. Spacecraft cruise science on.
	04:32:39	DC-28 (battery charger on) transmitted. Spacecraft battery charger on, boost mode disabled.
	06:54:43	DC-25 (encounter science on) transmitted. Spacecraft encounter science on, science cover deployed, scan platform started, battery charger off, boost mode enabled.
	06:54:43	Data encoder rate 1 reset, rate 2 reset from 203 to 200, and rate 3/4 reset from 407 to 410.
	08:59:23	DC-24 (inhibit scan) transmitted. Spacecraft scan platform stopped at 177.94 deg clock angle.
	09:13:51	DC-28 transmitted. Spacecraft video storage 2.4-kc power off, battery charger on.
	09:30:56	DC-3 transmitted. Spacecraft telemetry to Mode 3. Television camera shutter normal.
	10:21:20	DC-2 transmitted. Spacecraft telemetry to Mode 2.
	10:27:08	DC-26 transmitted. Spacecraft encounter science off, television shutter positioned, battery charger off, boost mode enabled, cruise science off.
	10:27:08	Data encoder rates 1, 2, and 3/4 reset.
	10:49:35	DC-2 transmitted. Spacecraft cruise science on.
	18:07:30 to 24:00:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
12	00:00:00 to 01:04:20	No telemetry data; DSIF 51 assigned to *Ranger* Project.
	02:53:22	Recovery transient observed in roll telemetry. No brightness transient was observable.
	17:01:18	CC&S CY-1 No. 28 on schedule. Observed in data at 17:01:47.
	18:02:49 to 24:00:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
February 13, 1965	00:00:00 to 01:00:55	No telemetry data; DSIF 51 assigned to *Ranger* Project.
	13:12:54	The absorptivity standard black sample stepped unexpectedly. This step was not anticipated for about 4 wk.
	17:58:00 to 24:00:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
14	00:00:00 to 00:57:20	No telemetry data; DSIF 51 assigned to *Ranger* Project.
	17:45:00 to 24:00:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
15	00:00:00 to 00:55:40	No telemetry data; DSIF 51 assigned to *Ranger* Project.
	10:00:57	Absorptivity standard white sample stepped as expected.
	11:41:27	CC&S CY-1 No. 29 on schedule. Observed in data at 11:41:59.
	17:50:31 to 24:00:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
16	00:00:00 to 00:48:38	No telemetry data; DSIF 51 assigned to *Ranger* Project.
	13:47:24	Roll transient; spacecraft recovery normal.
	17:46:48 to 24:00:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
17	00:00:00 to 00:44:58	No telemetry data; DSIF 51 assigned to *Ranger* Project.
	17:38:00 to 24:00:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
18	00:00:00 to 01:41:53	No telemetry data; DSIF 51 assigned to *Ranger* Project.
	06:21:35	CC&S CY-1 No. 30 on schedule. Observed in data at 06:22:13.
	17:39:43 to 24:00:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
19	00:00:00 to 00:37:42	No telemetry data; DSIF 51 assigned to *Ranger* Project.
	17:35:47 to 24:00:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
20	00:00:00 to 00:07:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
	02:10:27	Absorptivity standard polished-aluminum sample stepped as expected.
	17:35:47 to 24:00:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
21	00:00:00 to 00:30:17	No telemetry data; DSIF 51 assigned to *Ranger* Project.
	01:01:44	CC&S CY-1 No. 31 on schedule. Observed in data at 01:02:25.

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
February 21, 1965	17:28:30 to 24:00:00	No telemetry data; DSIF 51 converting from L-band to S-band.
22	00:00:00 to 00:29:26	No telemetry data; DSIF 51 converting from L-band to S-band.
23	11:13:25	Absorptivity standard white sample stepped as expected.
	11:41:53	CC&S CY-1 No. 32. Exact time of event not known because of DSIF 51 problem.
26	14:21:02	CC&S CY-1 No. 33 on schedule. Observed in data at 14:22:54.
27	13:56	Attitude control roll transient observed.
↓	17:02:05	CC&S command MT-1 (Canopus sensor cone-angle update) on schedule. Canopus sensor cone-angle updated from 100.2 to 95.7 deg. Observed in data at 17:02:19.
↓	17:02:19	Data encoder skipped Channels 300, 400, and 420 coincident with MT-1.
March 1	09:02:13	CC&S CY-1 No. 34 on schedule. Apparent momentary loss of signal strength observed simultaneously at DSIF 42 (later found to be the result of a frequency shift caused by spacecraft radio design characteristic). Observed in data at 09:02:18.
2	02:49:46	Absorptivity standard aluminum-silicon sample stepped as expected.
4	03:42:23	CC&S CY-1 No. 35 on schedule. Apparent 2-db, 5-sec loss-of-ground received signal observed simultaneously (later found to be due to frequency shift caused by spacecraft radio subsystem design characteristic). Observed in data at 03:42:32.
5	13:02:25	CC&S MT-5 transfer spacecraft transmitter to high-gain antenna on schedule. Observed in data at 13:02:37.
6	22:22:34	CC&S CY-1 No. 36 on schedule. Observed in data at 22:22:47.
7	20:02:58	Absorptivity standard polished-aluminum sample stepped as expected.
9	17:02:44	CC&S CY-1 No. 37 on schedule. Observed in data at 17:03:03.
10	01:06:41	Absorptivity standard black sample stepped as expected.
↓	16:34:00 to 23:31:50	No telemetry data; DSIF 51 assigned to *Ranger* Project.
11	02:17:56	Absorptivity standard white sample stepped as expected.
↓	04:47	Attitude control roll transient observed.
↓	16:30:38 to 23:39:21	No telemetry data; DSIF 51 assigned to *Ranger* Project.
12	11:42:55	CC&S CY-1 No. 38 on schedule. Observed in data at 11:43:18.
↓	16:29:22 to 23:23:41	No telemetry data; DSIF 51 assigned to *Ranger* Project.
13	16:25:39 to 23:24:26	No telemetry data; DSIF 51 assigned to *Ranger* Project.
14	02:17	Attitude control roll transient observed.
↓	16:23:07 to 23:20:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
15	06:23:06	CC&S CY-1 No. 39 on schedule. Observed in data at 06:23:34.
↓	16:19:36 to 23:15:30	No telemetry data; DSIF 51 assigned to *Ranger* Project.

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
March 16, 1965	16:16:48 to 23:26:46	No telemetry data; DSIF 51 assigned to *Ranger* Project.
17	16:14:59 to 23:08:58	No telemetry data; DSIF 51 assigned to *Ranger* Project.
	—	Ion chamber and associated GM 10311 tube failed during DSIF 51 pass.
18	01:03:18	CC&S CY-1 No. 40 on schedule. Observed in data at 01:03:58.
↓	01:03:50	Attitude control roll transient observed.
↓	16:11:23 to 23:07:26	No telemetry data; DSIF 51 assigned to *Ranger* Project.
19	00:42:28	Absorptivity standard white sample stepped as expected.
↓	16:09:02 to 23:03:30	No telemetry data; DSIF 51 assigned to *Ranger* Project.
20	16:06:00 to 23:01:25	No telemetry data; DSIF assigned to *Ranger* Project.
↓	19:43:29	CC&S CY-1 No. 41 during DSIF 51 pass. Not observed in data.
21	16:03:00 to 22:58:20	No telemetry data; DSIF 51 assigned to *Ranger* Project.
22	16:01:13 to 22:55:40	No telemetry data; DSIF 51 assigned to *Ranger* Project.
23	14:23:41	CC&S CY-1 No. 42 on schedule. Observed in data at 14:24:24.
↓	15:58:26 to 22:51:30	No telemetry data; DSIF 51 assigned to *Ranger* Project.
24	15:55:00 to 22:50:19	No telemetry data; DSIF 51 assigned to *Ranger* Project.
25	15:53:00 to 22:48:00	No telemetry data; DSIF 51 assigned to *Ranger* Project.
26	09:03:54	CC&S CY-1 No. 43 on schedule. Observed in data at 09:04:40.
	09:16:14	Absorptivity standard polished-aluminum sample stepped as expected.
29	03:44:06	CC&S CY-1 No. 44 on schedule. Observed in data at 03:44:07.
30	14:03:42	Absorptivity standard aluminum-silicon sample stepped 2½ days later than expected.
31	22:24:19	CC&S CY-1 No. 45 on schedule. Observed in data at 22:24:25.
April 2	14:24:25	CC&S command MT-2 (Canopus sensor cone-angle update) on schedule. Canopus sensor cone angle updated from 95.7 to 91.1 deg. Observed in data at 14:25:15.
3	17:04:32	CC&S CY-1 No. 46 on schedule. Observed in data at 17:04:42.
4	17:07:42	Absorptivity standard white sample stepped as expected.
6	11:44:45	CC&S CY-1 No. 47 on schedule. Observed in data at 11:45:01.
7	11:10	Small roll transient observed.
9	06:24:58	CC&S CY-1 No. 48 on schedule. Observed in data at 06:25:19.
11	06:18	Small roll transient observed.
	08:00	Class 2 flare detected on east limb of Sun by AGIWARN[a] network.

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
April 12, 1965	01:05:12	CC&S CY-1 No. 49 on schedule. Observed in data at 01:05:38.
13	22:21:18	Absorptivity standard black sample final step observed as expected.
14	19:45:26	CC&S CY-1 No. 50 on schedule. Observed in data at 19:45:55.
15	09:55	Extremeley small roll transient observed.
	16:20:35	Absorptivity standard white sample stepped as expected.
16	09:47	Class 2 flare observed on east limb of Sun by AGIWARN network.
	13:00	Increased magnetic and plasma activity detected by spacecraft instruments. Possibly due to solar storm resulting from Class 2 flare observed on Sun on April 11.
17	14:25:40	CC&S CY-1 No. 51 on schedule. Observed in data at 14:26:14.
20	09:05:54	CC&S CY-1 No. 52 on schedule. Observed in data at 09:06:33.
23	03:46:09	CC&S CY-1 No. 53 on schedule. Observed in data at 03:46:53.
24	23:47	Roll transient observed.
25	22:26:23	CC&S CY-1 No. 54 on schedule. Observed in data at 22:27:11.
28	17:06:38	CC&S CY-1 No. 55 on schedule. Observed in data at 17:06:41.
	—	DSIF 51 performed blind command detector lockup.
	—	DSIF 11 performed blind command detector lockup.
30	—	DSIF 41 performed blind command detector lockup.
May 1	11:46:54	CC&S CY-1 No. 56 on schedule. Observed in data at 11:47:02.
3	—	Spacecraft received carrier power with command modulation applied decreased below −139 dbm, the command link sum of the negative tolerances.
	—	Decreasing temperature, in conjunction with the component failure incurred on December 6, 1964, caused plasma probe data to be declared uninterpretable. Formerly they were considered partially recoverable.
4	06:27:09	CC&S CY-1 No. 57 on schedule. Observed in data at 06:27:21.
7	01:07:24	CC&S CY-1 No. 58 on schedule. Observed in data at 01:07:41.
	17:27:25	CC&S command MT-3 (Canopus sensor cone-angle update) on schedule. Canopus sensor cone angle updated from 91.1 to 86.5 deg. Observed in data at 17:28:15.
9	19:47:39	CC&S CY-1 No. 59 on schedule. Observed in data at 19:48:01.
11	05:43:15	Absorptivity standard aluminum-silicon sample stepped as expected.
12	14:27:55	CC&S CY-1 No. 60 on schedule. Observed in data at 14:28:20.
	—	DSIF 13 100-kw transmitter established command lock with spacecraft.
15	09:08:09	CC&S CY-1 No. 61 on schedule. Observed in data at 09:08:41.
17	15:44:41	Absorptivity standard polished-aluminum sample stepped earlier than expected.
18	03:48:24	CC&S CY-1 No. 62 on schedule. Observed in data at 03:49:01.
19	22:53:40	DSIF 13 100-kw transmitter locked up with the spacecraft command detector and maintained command lock until 23:33:09 GMT. The modulation index was apparently set incorrectly. Lockup time: 12 min. Modulation loss: 2.2 db.
20	21:42:57	DSIF 13 100-kw transmitter locked up with the spacecraft command detector and maintained command lock until 22:04:47 GMT. Lockup time: 6 min. Modulation loss: 1.7 db.

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
May 20, 1965	22:28:40	CC&S CY-1 No. 63 on schedule. Observed in data at 22:27:22.
	22:52:40	DSIF-13 100-kw transmitter locked up with the spacecraft command detector and maintained command lock until 23:38:53 GMT. Lockup time: 6 min. Modulation loss: 3.5 db.
23	17:08:55	CC&S CY-1 No. 64 on schedule. Observed in data at 17:09:42.
25	23:00	A solar flare was detected by spacecraft instruments.
26	11:49:11	CC&S CY-1 No. 65 on schedule. Observed in data at 11:49:12.
29	06:29:27	CC&S CY-1 No. 66 on schedule. Observed in data at 06:29:33.
June 1	01:09:43	CC&S CY-1 No. 67 on schedule. Observed in data at 01:09:53.
2	22:07:06	DSIF-13 100-kw transmitter locked up with the spacecraft command detector and maintained command lock until 22:54:59 GMT. The modulation index was apparently adjusted incorrectly. Lockup time: 4 min. Modulation loss: −4.7 db.
3	19:49:59	CC&S CY-1 No. 68 on schedule. Observed in data at 19:50:14.
5	18:07	An unusual type of solar flare was optically observed by the AGIWARN network.
	19:30	The unusual solar flare was detected by the spacecraft trapped radiation detector.
6	14:30:15	CC&S CY-1 No. 69 on schedule. Observed in data at 14:30:35.
9	08:00	Increased magnetic activity detected by the magnetometer, probably caused by the unusual solar flare on June 5, 1965.
	09:10:31	CC&S CY-1, No. 70 on schedule. Observed in data at 09:10:56.
	11:41:05	A vigorous roll transient was observed coincident with a large change in Canopus brightness and a visible impulse on the spacecraft yaw axis.
10	19:54	A small roll transient was observed.
11	15:40	A roll transient was observed coincident with slight impulses in the pitch and yaw axes.
12	03:50:47	CC&S CY-1 No. 71 on schedule. Observed in data at 03:51:16.
	18:49	Small roll transient found during nonreal-time data analysis.
14	15:50:57	CC&S command MT-4 (Canopus sensor cone-angle update) on schedule. Canopus sensor cone angle updated from 86.5 deg to 82 deg. Observed in data at 15:51:45.
	15:53	A roll transient was observed.
	22:31:02	CC&S CY-1 No. 72 on schedule. Observed in data at 22:31:36.
15	07:36	Class 2 solar flare detected by AGIWARN network.
	10:40	Class 2 solar flare detected by the pn junction detectors of the spacecraft trapped radiation detector.
16	11:38	A small roll transient was observed.
16	21:29:47	DSIF 13 100-kw transmitter locked up with the spacecraft command detector and maintained command lock until 22:26:25 GMT. Lockup time: 4 min. Modulation loss: −3.8 db.
17	17:11:18	CC&S CY-1 No. 73 on schedule. Observed in data at 17:11:57.
19	04:50	A small roll transient was observed.
20	11:51:34	CC&S CY-1 No. 74 on schedule. Observed in data at 11:52:17.
23	06:31:49	CC&S CY-1 No. 75 on schedule. Observed in data at 06:31:49.
	21:34	DSIF 13 100-kw transmitter locked up with the spacecraft command detector and maintained command lock until 22:08 GMT. Lockup time: 2 min. Carrier suppression: 3.2 db.

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
June 26, 1965	01:12:05	CC&S CY-1 No. 76 on schedule. Observed in data at 01:12:09.
27	07:14	Roll transient observed.
	18:47	Roll transient observed.
28	15:28	DSIF 51 10-kw transmitter locked up with the spacecraft command detector and maintained command lock until 17:45 GMT. Lockup time: 10 min.
	19:52:21	CC&S CY-1 No. 77 on schedule. Observed in data at 19:52:29.
	22:20	DSIF 11 10-kw transmitter locked up with the spacecraft command detector and maintained command lock until 23:05 GMT. Lockup time: 16 min.
30	18:32	DSIF 51 10-kw transmitter locked up with the spacecraft command detector and maintained command lock until 19:55 GMT. Lockup time: 16 min.
July 1	14:32:36	CC&S CY-1 No. 78 on schedule. Observed in data at 14:32:49.
3	09:56	A minor roll transient was observed.
4	09:12:52	CC&S CY-1 No. 79 on schedule. Observed in data at 09:13:10.
5	09:27	A minor roll transient was observed.
6	03:10	A minor roll transient was observed.
	23:05	Class 1+ solar flare detected by AGIWARN network.
7	03:32	Class 1 solar flare detected by AGIWARN network.
	03:53:08	CC&S CY-1 No. 80 on schedule. Observed in data at 03:53:30.
8	08:05	Class 1+ solar flare detected by AGIWARN network.
9	22:33:23	Not observed in data.
12	05:40:56	Minor roll transient observed.
	06:23:46	Small roll transient observed.
	17:13:07	CC&S CY-1 No. 82 on schedule. Observed in data at 17:13:39.
	22:11	DSIF 11 10-kw transmitter locked up with the spacecraft command detector and maintained command lock until 22:45 GMT. Lockup time: 10 min.
	22:38	DSIF 13 100-kw transmitter locked up with the spacecraft command detector and maintained command lock until 00:09 GMT on July 13. Lockup time: 3 min.
13	00:27	DSIF 13 100-kw transmitter locked up with the spacecraft command detector and maintained command lock until 00:59 GMT. Lockup time: 7 min.
	18:06	DSIF 11 10-kw transmitter locked up with the spacecraft command detector and maintained command lock until 18:45 GMT. Lockup time: 3 min.
	21:28	DSIF 51 10-kw transmitter locked up with the spacecraft command detector and maintained command lock until 22:43 GMT. Lockup time: 3 min.
14	13:34	DSIF 51 10-kw transmitter locked up with the spacecraft command detector and maintained command lock until 19:40 GMT. Lockup time: 4 min.
	14:27:55	DC-25 (encounter science on) transmitted.
	14:40:32.8	DC-25 received at spacecraft. Encounter science turned on and scan platform in search mode.
	14:40:32.8	Data encoder rate 3/4 skipped from 409 to 401.
	15:41:49	CC&S command MT-7 (encounter science on) issued at spacecraft. No effect because encounter science already turned on by DC-25 at 14:40:32 GMT.

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
July 14, 1965	14:52	DC-25 observed on Earth.
	15:53:40	CC&S command MT-7 observed on Earth. Observed in data at 15:54:05.
	17:10:18	DC-24 (inhibit scan) transmitted.
	17:22:55	DC-24 received at spacecraft. Scan platform stopped with television camera at 178.45-deg clock angle.
	17:34:55	DC-24 observed on Earth.
	21:54:19	DSIF 11 10-kw transmitter locked up with the spacecraft command detector and maintained command lock until 02:19 GMT on July 15. Lockup time: 30 min.
	22:10:29	DC-3 (transfer data encoder to Mode 3) transmitted.
	22:23:07	DC-3 received at spacecraft. Data encoder transferred to Mode 3.
	22:35:08	DC-3 initiated Mode 3 data observed on Earth.
	23:24:12	Earliest possible 3-σ WAA.
	23:42:00.3	WAA (scan planet-in-view) signal initiated at spacecraft ($\pm$10.8 sec) when Mars moved into the scan platform wide-angle sensor field of view. No effect because scan platform pre-positioned by DC-24 at 17:22:55 GMT and data encoder transferred to Mode 3 by DC-3 at 22:23:07 GMT.
	23:55:45	Probability of WAA by spacecraft equals 0.7.
15	00:11:57	DC-16 (NAA) transmitted.
	00:16:50.1	Narrow-angle sensor first detected Mars' presence ($\pm$12.6 sec).
	00:17:21.1	NAA at spacecraft when Mars moved into field-of-view of NAMG. Television picture recording sequence started.
	00:18:29.6	First DAS start tape command sent to video storage subsystem at spacecraft. No. 1 Shutter 00:18:31.1 No. 2 Shutter 00:19:21.1 No. 3 Shutter 00:20:57.1 No. 4 Shutter 00:21:45.1 No. 5 Shutter 00:23:21.1 No. 6 Shutter 00:24:09.1
	00:22:30.3	False shutters or EOT signal initiated at spacecraft ($\pm$25.2 sec).
	00:24:36	DC-16 received at spacecraft. No effect because NAA already initiated by NAMG at 00:17:58 GMT. No. 7 Shutter 00:25:45.1 No. 8 Shutter 00:26:33.1 No. 9 Shutter 00:28:09.1 No. 10 Shutter 00:28:57.1
	00:28:23.1	False shutter or EOT signal initiated at spacecraft ($\pm$25.2 sec).
	00:29:22.5	NAA observed on Earth.
	00:30:31	First DAS start tape command observed on Earth. No. 11 Shutter 00:30:33.1
	00:30:54.3	First EOT signal issued by video storage subsystem at spacecraft ($\pm$25.2 sec). No. 12 Shutter 00:31:21.1
	00:31:42	DC-26 (all science off) transmitted.
	00:32:40	DC-2 No. 1 (transfer data encoder to Mode 2 and turn on cruise science) transmitted. No. 13 Shutter 00:32:57.1 No. 14 Shutter 00:33:45.1

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
July 15, 1965	00:34:31.7	False shutter or EOT signal observed on Earth. No. 15 Shutter 00:35:21.1 No. 16 Shutter 00:36:09.1
	00:36:37	DC-16 observed on Earth.
	00:37:00	DC-2 No. 2 transmitted. No. 17 Shutter 00:37:45.1 No. 18 Shutter 00:38:33.1 No. 19 Shutter 00:40:09.1
	00:40:24.5	False shutter or EOT signal observed on Earth. No. 20 Shutter 00:40:57.1
	00:42:00	DC-2 No. 3 transmitted. No. 21 Shutter 00:42:33.1
	00:42:55.7	First video storage EOT signal observed on Earth. No. 22 Shutter 00:43:21.1
	00:43:45.1	Tape recorder stopped and data encoder transferred to Mode 2 by Picture 22 stop command from DAS.
	00:44:21.5	DC-26 received at spacecraft. All science turned off.
	00:45:19.6	DC-2 No. 1 received at spacecraft. Cruise science turned on. Data encoder already transferred to Mode 2 by DAS Picture 22 stop command at 00:43:45 GMT.
	00:47:00	DC-2 No. 4 transmitted.
	00:49:39	DC-2 No. 2 received at spacecraft. No effect.
	00:52:00	DC-2 No. 5 transmitted.
	00:54:39	DC-2 No. 3 received at spacecraft. No effect.
	00:55:46.6	Transfer of the data encoder to Mode 2 by Picture 22 stop command observed on Earth.
	00:56:23.0	DC-26 observed on Earth.
	00:57:00	DC-2 No. 6 transmitted.
	00:57:20	DC-2 No. 1 observed on Earth.
	00:59:39	DC-2 No. 4 received at spacecraft. No effect.
	01:00:57	Closest approach of spacecraft to Mars: 6118 mi.
	01:01:40	DC-2 No. 2 observed on Earth.
	01:04:39	DC-2 No. 5 received at spacecraft. No effect.
	01:06:40	DC-2 No. 3 observed on Earth.
	01:09:39	DC-2 No. 6 received at spacecraft. No effect.
	01:11:40	DC-2 No. 4 observed on Earth.
	01:16:40	DC-2 No. 5 observed on Earth.
	01:21:40	DC-2 No. 6 observed on Earth.
	02:19:11	Spacecraft entered Earth occultation region.
	02:31:12	Spacecraft RF signal lost on Earth.
	03:13:04	Spacecraft emerged from Earth occultation region.
	03:25:06	Spacecraft RF signal reacquired on Earth.

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
July 15, 1965	05:01:49	CC&S command MT-8 (encounter science off) issued at spacecraft. No effect because encounter science already turned off by DC-26 at 00:44:21 GMT.
	05:13:52	CC&S command MT-8 observed on schedule on Earth. Observed in data at 05:14:39.
	11:41:49.8	CC&S command MT-9 and CC&S CY-1 No. 83 issued on schedule at spacecraft. Data encoder transferred to Mode 4/1, cruise science turned off, and video storage in playback mode.
	11:53:53.3	Mode 1 engineering data initiated by MT-9 command observed on Earth; CC&S CY-1 No. 83 observed simultaneously. Two events were observed in data encoder event Register 2 instead of the single event normally associated with MT-9. Observed in data at 11:53:53.
	13:01:58	Mode 1 engineering data ended and Picture 1 data observed on Earth.
	21:38:07	Picture 1 data ended and Mode 1 engineering data observed on Earth.
	23:32:27	Mode 1 engineering data ended and Picture 2 data observed on Earth.
16	08:08:00	Picture 2 data ended and Mode 1 engineering data observed on Earth.
	10:04:28	Mode 1 engineering data ended and Picture 3 data observed on Earth.
	18:39:54	Picture 3 data ended and Mode 1 engineering data observed on Earth.
	20:35:12	Mode 1 engineering data ended and Picture 4 data observed on Earth.
17	05:10:12	Picture 4 data ended and Mode 1 engineering data observed on Earth.
	06:18	DSIF 42 10-kw transmitter locked up with the spacecraft command detector and maintained command lock until 07:09 GMT. Lockup time: 18 min.
	07:07:45	Mode 1 engineering data ended and Picture 5 data observed on Earth.
	15:43:18	Picture 5 data ended and Mode 1 engineering data observed on Earth.
	17:40:32	Mode 1 engineering data ended and Picture 6 data observed on Earth.
18	02:16:10	Picture 6 data ended and Mode 1 engineering data observed on Earth.
	04:13:25	Mode 1 engineering data ended and Picture 7 data observed on Earth.
	06:34:08	Predicted time for observation of CC&S command CY-1 No. 84 on Earth. The data encoder event normally associated with CY-1 could not be observed because television picture data (Mode 4) was being transmitted by the spacecraft; however, the frequency shift associated with CY-1 when the spacecraft is in one-way lock was reported by the DSIF station.
	12:49:03	Picture 7 data ended and Mode 1 engineering data observed on Earth.
	14:46:13	Mode 1 engineering data ended and Picture 8 data observed on Earth.
	23:21:51	Picture 8 data ended and Mode 1 engineering data observed on Earth.
19	00:30	DSIF 11 10-kw transmitter locked up with the spacecraft command detector and maintained command lock until 01:19 GMT. Lockup time: 23 min.
	01:19:32	Mode 1 engineering data ended and Picture 9 data observed on Earth.
	09:55:35	Picture 9 data ended and Mode 1 engineering data observed on Earth.
	10:55	DSIF 51 10-kw transmitter locked up with the spacecraft command detector and maintained command lock until 11:03 GMT. Lockup time: 17 min.
	11:52:52	Mode 1 engineering data ended and Picture 10 data observed on Earth.
	20:28:29	Picture 10 data ended and Mode 1 engineering data observed on Earth.
	22:25:33	Mode 1 engineering data ended and Picture 11 data observed on Earth.

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
July 20, 1965	03:15:32	First EOT signal and normal tape recorder track change occurred at Line 111 of Picture 11.
	07:01:26	Picture 11 data ended and Mode 1 engineering data observed on Earth.
	08:57:42	Mode 1 engineering data ended and Picture 12 data observed on Earth.
	17:33:15	Picture 12 data ended and Mode 1 engineering data observed on Earth.
	19:28:51	Mode 1 engineering data ended and Picture 13 data observed on Earth.
21	01:14:23	Predicted time for observation of CC&S command CY-1 No. 85 on Earth. The data encoder event normally associated with CY-1 could not be observed because television picture data (Mode 4) was being transmitted by the spacecraft; however, the frequency shift associated with CY-1 when the spacecraft is in one-way lock was reported by the DSIF station.
	04:04:32	Picture 13 data ended and Mode 1 engineering data observed on Earth.
	06:01:22	Mode 1 engineering data ended and Picture 14 data observed on Earth.
	14:37:00	Picture 14 data ended and Mode 1 engineering data observed on Earth.
	16:34:43	Mode 1 engineering data ended and Picture 15 data observed on Earth.
22	01:10:21	Picture 15 data ended and Mode 1 engineering data observed on Earth.
	03:07:13	Mode 1 engineering data ended and Picture 16 data observed on Earth.
	11:42:53	Picture 16 data ended and Mode 1 engineering data observed on Earth.
	13:40:59	Mode 1 engineering data ended and Picture 17 data observed on Earth.
	22:16:37	Picture 17 data ended and Mode 1 engineering data observed on Earth.
23	00:15:04	Mode 1 engineering data ended and Picture 18 data observed on Earth.
	08:50:41	Picture 18 data ended and Mode 1 engineering data observed on Earth.
	10:48:08	Mode 1 engineering data ended and Picture 19 data observed on Earth.
	19:23:48	Picture 19 data ended and Mode 1 engineering data observed on Earth.
	19:35:05	Absorptivity standard white sample stepped.
	19:54:38	CC&S CY-1 No. 86 on schedule. Observed in data at 19:54:40.
	21:21:10	Mode 1 engineering data ended and Picture 20 data observed on Earth.
24	05:56:50	Picture 20 data ended and Mode 1 engineering data observed on Earth.
	07:55:01	Mode 1 engineering data ended and Picture 21 data observed on Earth.
	16:30:40	Picture 21 data ended and Mode 1 engineering data observed on Earth.
	18:28:33	Mode 1 engineering data ended and Picture 22 data observed on Earth.
	19:26:33	Picture 22 data ended and Mode 1 engineering data observed on Earth.
	21:21:53	Mode 1 engineering data ended and Picture 1 data observed on Earth.
25	05:57:54	Picture 1 data ended and Mode 1 engineering data observed on Earth.
	07:53:36	Mode 1 engineering data ended and Picture 2 data observed on Earth.
	16:29:15	Picture 2 data ended and Mode 1 engineering data observed on Earth.
	18:24:31	Mode 1 engineering data ended and Picture 3 data observed on Earth.
26	03:00:09	Picture 3 data ended and Mode 1 engineering data observed on Earth.
	04:56:24	Mode 1 engineering data ended and Picture 4 data observed on Earth.

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
July 26, 1965	13:32:03	Picture 4 data ended and Mode 1 engineering data observed on Earth.
	14:34:52	CC&S CY-1 No. 87 on schedule. Observed in data at 14:34:59.
	15:27:47	Mode 1 engineering data ended and Picture 5 data observed on Earth.
27	00:03:39	Picture 5 data ended and Mode 1 engineering data observed on Earth.
	02:00:57	Mode 1 engineering data ended and Picture 6 data observed on Earth.
	10:36:39	Picture 6 data ended and Mode 1 engineering data observed on Earth.
	12:33:01	Mode 1 engineering data ended and Picture 7 data observed on Earth.
	21:08:41	Picture 7 data ended and Mode 1 engineering data observed on Earth.
	23:04:57	Mode 1 engineering data ended and Picture 8 data observed on Earth.
28	07:40:41	Picture 8 data ended and Mode 1 engineering data observed on Earth.
	09:37	Mode 1 engineering data ended and Picture 9 data observed on Earth.
	18:12	Picture 9 data ended and Mode 1 engineering data observed on Earth.
	20:09:57	Mode 1 engineering data ended and Picture 10 data observed on Earth.
29	04:45:32	Picture 10 data ended and Mode 1 engineering data observed on Earth.
	06:43:10	Mode 1 engineering data ended and Picture 11 data observed on Earth.
	09:15:06	Predicted time for observation of CC&S command CY-1 No. 88. The data encoder event normally associated with CY-1 was observed when Mode 1 data appeared at 15:19:13 GMT.
	11:33:27	First EOT signal of second playback and normal track change observed. Extra data encoder event was observed that was probably due to "dirty" tape foil.
	15:19:13	Picture 11 data ended and Mode 1 engineering data observed on Earth.
	17:16:43	Mode 1 engineering data ended and Picture 12 data observed on Earth.
30	01:51	Picture 12 data ended and Mode 1 engineering data observed on Earth.
	03:46:24	Mode 1 engineering data ended and Picture 13 data observed on Earth.
	12:21	Picture 13 data ended and Mode 1 engineering data observed on Earth.
	14:20	Mode 1 engineering data ended and Picture 14 data observed on Earth.
	22:55:28	Picture 14 data ended and Mode 1 engineering data observed on Earth.
31	00:52	Mode 1 engineering data ended and Picture 15 data observed on Earth.
	09:27	Picture 15 data ended and Mode 1 engineering data observed on Earth.
	11:24	Mode 1 engineering data ended and Picture 16 data observed on Earth.
	20:02:15	Picture 16 data ended and Mode 1 engineering data observed on Earth.
	21:59:57	Mode 1 engineering data ended and Picture 17 data observed on Earth.
August 1	03:55:21	Predicted time for observation of CC&S command CY-1 No. 89. The data encoder event normally associated with CY-1 was observed when Mode 1 data appeared at 06:31 GMT.
	06:31	Picture 17 data ended and Mode 1 engineering data observed on Earth.
	08:28	Mode 1 engineering data ended and Picture 18 data observed on Earth.
	17:09:00	Picture 18 data ended and Mode 1 engineering data observed on Earth.
	19:07:10	Mode 1 engineering data ended and Picture 19 data observed on Earth.

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
August 2, 1965	03:42:33	Picture 19 data ended and Mode 1 engineering data observed on Earth.
	05:40:00	Mode 1 engineering data ended and Picture 20 data observed on Earth.
	14:15:27	Picture 20 data ended and Mode 1 engineering data observed on Earth.
	16:14:48	Mode 1 engineering data ended and Picture 21 data observed on Earth.
3	00:39	Picture 21 data ended and Mode 1 engineering data observed on Earth.
	02:18	DSIF 13 100-kw transmitter locked up with the spacecraft command detector and maintained command lock until 04:37 GMT. Lockup time: 21 min.
	02:36	Mode 1 engineering data ended and Picture 22 data observed on Earth.
	03:08:33	DC-28 (turn off video storage 2.4-kc power and turn on battery charger) transmitted.
	03:14:33	DC-26 (turn off battery charger) transmitted.
	03:20:33	DC-2 (transfer data encoder to Mode 2 and turn on cruise science) transmitted.
	03:22:37	DC-28 received at spacecraft. Video storage 2.4-kc power turned off and battery charger turned on. Data encoder transferred to Mode 1.
	03:28:37	DC-26 received at spacecraft. Battery charger turned off.
	03:28:37	Data encoder rate 2 skipped from 209 to 200, then reset to 201. Rate 3/4 skipped from 408 to 410 and 308 to 300.
	03:34:37	DC-2 received at spacecraft. Data encoder transferred to Mode 2 and cruise science turned on.
	03:36:02	DC-28 observed on Earth.
	03:42:02	DC-26 observed on Earth.
	03:48:02	DC-2 observed on Earth.
	22:35:36	CC&S CY-1 No. 90 on schedule. Observed in data at 22:35:57.
4	03:32	Minor roll transient observed.
	03:42	Minor roll transient observed.
	16:17	Minor roll transient observed.
6	17:16:15	CC&S CY-1 No. 91 on schedule.
9	11:56:04	CC&S CY-1 No. 92 on schedule.
12	06:36:53	CC&S CY-1 No. 93 on schedule.
	22:01	DSIF 13 100-kw transmitter locked up with the spacecraft command detector and maintained command lock until 23:35 GMT. Lockup time: 2 min.
15	01:16:32	CC&S CY-1 No. 94 on schedule. Observed in data at 01:17:12 GMT.
17	19:56:46	CC&S CY-1 No. 95 on schedule. Observed in data at 19:57:30 GMT.
20	14:36:08	CC&S CY-1 No. 96 on schedule.
21	20:04:13	DSIF 13 100-kw transmitter locked up with the spacecraft command detector and maintained command lock intermittently until 21:21 GMT.
	22:16:00	DSIF 12 10-kw transmitter locked up with the spacecraft command detector and maintained command lock until 00:15:00 GMT, August 22. Lockup time: 6 min.
	22:22:00	DC-25 (turn on encounter science) transmitted.
	22:37:21	DC-25 received at spacecraft. Encounter science turned on and scan platform in search mode.

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
August 21, 1965	22:37:21	Data encoder deck reset from 413 to 400.
	22:52:04	DC-25 observed on Earth. Shortly after DC-25 was transmitted, the spacecraft AGC began to fluctuate more widely than normal and several out-of-lock indications were observed in the data. The cause of the anomalous behavior was a malfunction of the DSIF 12 transmitter frequency synthesizer.
	23:20:00	DC-28 No. 1 (turn off video storage 2.4-kc power and turn on battery charger) transmitted.
	23:22:00	DC-26 No. 2 transmitted.
	23:28:13	DC-26 No. 1 (turn off battery charger and all science) transmitted.
	23:30:13	DC-26 No. 2 transmitted.
	23:32:13	DC-2 No. 1 (turn on cruise science) transmitted.
	23:34:13	DC-2 transmitted.
	23:35:21	DC-28 No. 1 received at spacecraft. Video storage 2.4-kc power turned off and battery charger turned on.
	23:37:21	DC-28 No. 2 received at spacecraft. No effect.
	23:39:00	DC-2 No. 3 transmitted.
	23:43:34	DC-26 No. 1 received at spacecraft. Battery charger and all science turned off.
	23:44:00	DC-2 No. 4 transmitted.
	23:45:34	DC-26 No. 2 received at spacecraft. No effect.
	23:47:34	DC-2 No. 1 received at spacecraft. Cruise science turned on.
	23:49:00	DC-2 No. 5 transmitted.
	23:49:34	DC-2 No. 2 received at spacecraft. No effect.
	23:50:05	DC-28 No. 1 observed on Earth.
	23:52:05	DC-28 No. 2 observed on Earth.
	23:54:00	DC-2 No. 6 transmitted.
	23:54:21	DC-2 No. 3 received at spacecraft. No effect.
	23:58:18	DC-26 No. 1 observed on Earth.
	23:59:00	DC-2 No. 7 transmitted.
	23:59:21	DC-2 No. 4 received at spacecraft. No effect.
22	00:00:18	DC-26 No. 2 observed on Earth.
	00:02:18	DC-2 No. 1 observed on Earth.
	00:04:18	DC-2 No. 2 observed on Earth.
	00:04:21	DC-2 No. 5 received at spacecraft. No effect.
	00:09:05	DC-2 No. 3 observed on Earth.
	00:09:21	DC-2 No. 6 received at spacecraft. No effect.
	00:14:05	DC-2 No. 4 observed on Earth.
	00:14:21	DC-2 No. 7 received at spacecraft. No effect.
	00:19:05	DC-2 No. 5 observed on Earth.
	00:24:05	DC-2 No. 6 observed on Earth.

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
August 22, 1965	00:29:05	DC-2 No. 7 observed on Earth.
23	09:17:12	CC&S CY-1 No. 97 on schedule. Observed in data at 09:17:15 GMT.
25	21:41:01	DSIF 13 100-kw transmitter locked up with the spacecraft command detector and maintained command lock until 22:11:05 GMT. Lockup time: 7 min.
	03:57:26	CC&S CY-1 No. 98 on schedule.
26	20:32:49	DSIF 12 10-kw transmitter locked up with the spacecraft command detector and maintained command lock until 22:08 GMT. Lockup time: 2 min.
	21:06:52	DC-13 (inhibit midcourse maneuver) transmitted.
	21:15:16	QC-1-1 (cw 0.18-deg pitch turn) transmitted.
	21:22:32	DC-13 received at spacecraft.
	21:23:40	QC-1-2 (cw 0.18-deg roll turn) transmitted.
	21:30:56	QC-1-1 received at spacecraft.
	21:32:04	QC-1-3 (0.08-sec motor burn duration) transmitted.
	21:37:42	DC-13 observed on Earth.
	21:39:20	QC-1-2 received at spacecraft.
	21:46:06	QC-1-1 observed on Earth.
	21:47:44	QC-1-3 received at spacecraft.
	21:54:32	QC-1-2 observed on Earth.
	22:02:54	QC-1-3 observed on Earth.
27	19:21:19	DSIF 12 10-kw transmitter locked up with the spacecraft command detector and maintained command lock until 20:28 GMT. Lockup time: 1 min.
	19:40:00	DC-17 (update Canopus sensor cone angle) transmitted.
	20:11:01	DC-17 observed on Earth.
28	22:37:39	CC&S CY-1 No. 99 observed on schedule. Observed in data at 22:37:52 GMT.
30	20:02	DSIF 12 10-kw transmitter locked up with the spacecraft command detector and maintained command lock until 02:25 GMT on August 31. Lockup time: 4 min.
	20:30:00	DC-25 (encounter science on) transmitted.
	20:45:56	DC-25 received at spacecraft (encounter science and scan platform power on).
	21:01:33	DC-25 observed on Earth.
	21:10:24	DC-3 (transfer data encoder to Mode 3) transmitted.
	21:26:20	DC-3 received at spacecraft (data encoder transferred to Mode 3).
	21:41:53	DC-3 observed on Earth (Mode 3 data).
	22:48:33	DC-24 (inhibit scan) transmitted.
	23:04:29	DC-24 received at the spacecraft (scan platform motion inhibited).
	23:21	DC-24 observed on Earth (scan platform inhibited at DN 406 = 148.43 deg).
	23:35:26	DC-16 (NAA) transmitted.
	23:51:22	DC-16 received at the spacecraft (NAA logic actuated).

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
August 31, 1965	00:05:00	DC-2 (transfer data encoder to Mode 2)) transmitted.
	00:06:52	DC-16 observed on Earth.
	00:08:19	First EOT observed in data.
	00:20:15	Second EOT observed in data.
	00:33:05	Mode 2 data observed on Earth. Data encoder transferred to Mode 2 by DAS after Picture 22.
	00:36:50	DC-2 observed in data.
	00:44:00	DC-26 transmitted.
	00:49:00	DC-22 transmitted.
	00:59:56	DC-26 received at the spacecraft. (All science power and scan platform and video storage 400-cps power off.)
	01:04:56	DC-22 received at the spacecraft (video storage change tracks).
	01:15:14	DC-26 observed on Earth.
	01:20:14	DC-22 observed on Earth.
	01:25:00	DC-4 transmitted.
	01:40:57	DC-4 received at the spacecraft (data encoder to Mode 4/1).
	01:56:16	DC-4 observed on Earth (Mode 1 data).
	02:00:46	Mode 4 data observed on Earth (Picture 1).
	10:36:38	End of Picture 1; start of Mode 1 data.
	12:30:40	Start of Picture 2.
	17:17:52	Predicted time for observation of CC&S command CY-1 No. 100. The data encoder event normally associated with CY-1 was observed when Mode 1 data appeared at 21:06:30 GMT.
	21:06:30	End of Picture 2; start of Mode 1 data.
	23:01:03	Start of Picture 3.
September 1	07:37:10	End of Picture 3; start of Mode 1 data.
	09:31:40	Start of Picture 4.
	18:07:30	End of Picture 4; start of Mode 1 data.
	20:03:20	Start of Picture 5.
2	04:39:23	End of Picture 5; start of Mode 1 data.
	05:39:49	DSIF 41 10-kw transmitter locked up with the spacecraft command detector and maintained command lock until 07:12:00 GMT. Lockup time: 4 min.
	06:17:00	DC-28 transmitted.
	06:23:00	DC-26 transmitted.
	06:29:00	DC-2 transmitted.
	06:33:05	DC-28 received at the spacecraft (video storage 2.4-kc power off, battery charger on).
	06:34:46	Start of Picture 6.
	06:39:05	DC-26 received at the spacecraft (battery charger off).
	06:45:05	DC-2 received at the spacecraft (cruise science power on, data encoder to Mode 2).

Table A-1. Flight history (cont'd)

Date	Time, GMT	Event
September 2, 1965	06:48:32	DC-28 observed on Earth; Mode 1 data observed on Earth.
	06:54:32	DC-26 observed on Earth; data encoder high-rate deck skip and medium- and low-rate deck reset observed on Earth.
	07:00:32	DC-2 observed on Earth; Mode 2 data observed on Earth.
3	11:58:03	CC&S CY-1 No. 101 on schedule; observed in data at 11:58:25 GMT.
6	06:38:16	CC&S CY-1 No. 102 on schedule; observed in data at 06:38:16 GMT.
9	01:18:28	CC&S CY-1 No. 103 on schedule; observed in data at 01:18:59 GMT.
11	19:58:40	CC&S CY-1 No. 104 on schedule; observed in data at 19:59:16 GMT.
14	14:38:51	CC&S CY-1 No. 105 on schedule; observed in data at 14:38:54 GMT.
17	09:18:58	CC&S CY-1 No. 106 on schedule; observed in data at 09:19:49 GMT.
20	03:58:24	CC&S CY-1 No. 107 on schedule; observed in data at 03:59:15 GMT.
22	22:38:40	CC&S CY-1 No. 108 on schedule; observed in data at 22:39:31 GMT.
25	17:19:37	CC&S CY-1 No. 109 on schedule; observed in data at 17:19:47 GMT.
28	—	CC&S CY-1 No. 110. No data during this time because DSIF 61 was not operating. Signal would have been observed at 11:59:47 GMT.
October 1	06:39:58	CC&S CY-1 No. 111 on schedule. Observed in data at 06:40:17 GMT.
	21:17:42	DSIF 13 100-kw transmitter locked up with the spacecraft command detector and maintained command lock until 22:23 GMT. Lockup time: 1 min.
	21:30:17	DC-12 transmitted.
	21:48:01	DC-12 received at the spacecraft (switch transmitter to low-gain antenna).
	22:05:07	DC-12 observed at Earth; loss of RF signal as expected.

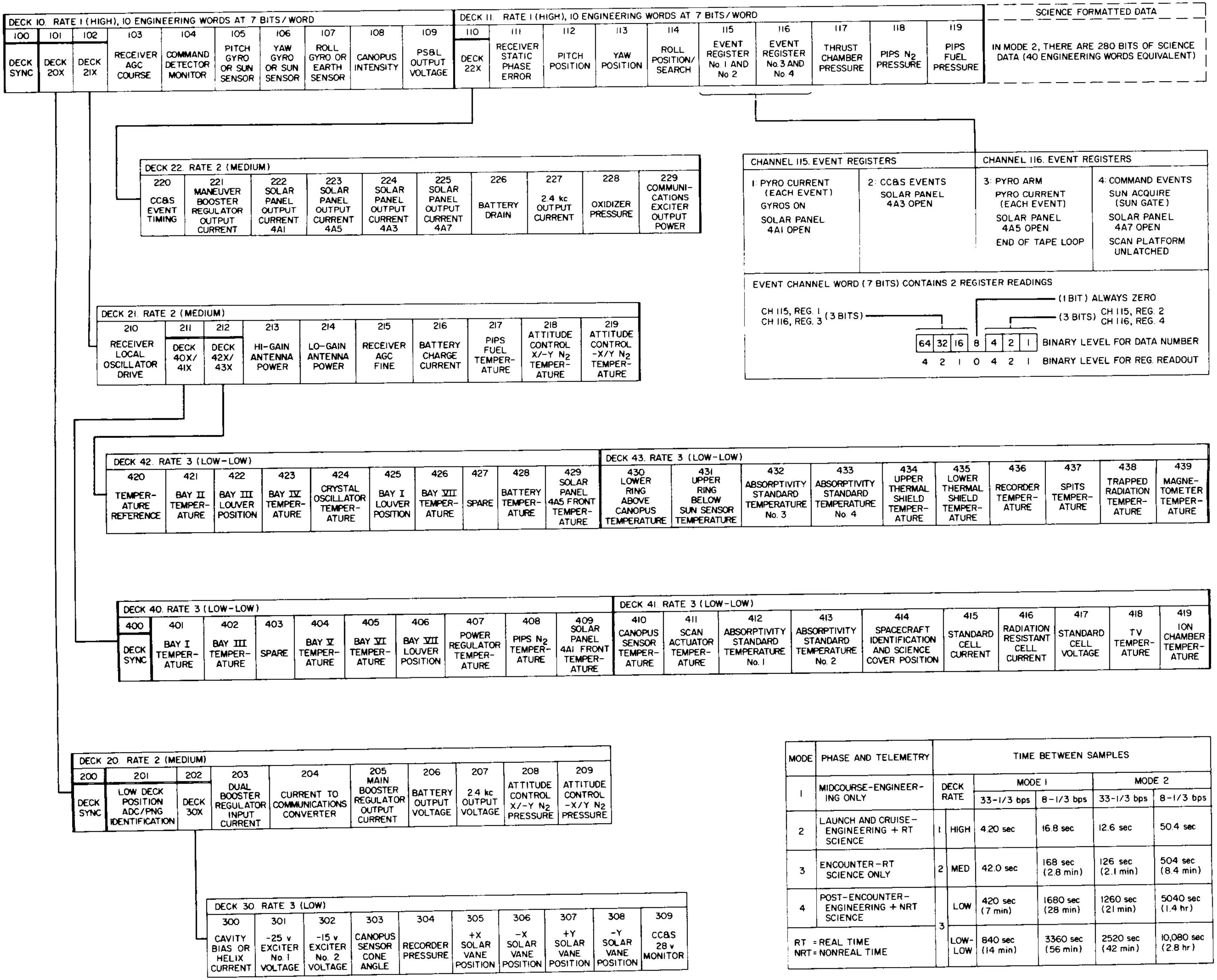
APPENDIX B

Telemetry Data

Figures B-1 and B-2 illustrate the engineering and real-time science telemetry data commutation schemes. Figures B-3–B-79 show 77 of the 100 individual engineering telemetry channels. Table B-1 lists the 23 engineering channels shown in Fig. B-1 that are not shown as individual channel graphs in Fig. B-3–B-79.

Table B-1. Engineering channels not plotted

Channel	Reason not shown in Fig. B-3–B-79
100	Data encoder Deck 100 synchronization channel
101	Data encoder Deck 100 subcommutation channel for Deck 200
102	Data encoder Deck 100 subcommutation channel for Deck 210
105	Typical sample of pitch Sun sensor output shown in body of Report as part of Fig. 8
106	Typical sample of yaw Sun sensor output shown in body of Report as part of Fig. 8
110	Data encoder Deck 110 subcommutation channel for Deck 220
114	Typical sample of roll position output shown in body of Report as part of Fig. 8
115	Sources of events in data encoder Registers 1 and 2 shown in Fig. B-1
116	Sources of events in data encoder Registers 3 and 4 shown in Fig. B-1
118	Shown in body of Report as Fig. 36
119	Shown in body of Report as Fig. 37
200	Data encoder Deck 200 synchronization channel
202	Data encoder Deck 200 subcommutation channel for Deck 300
206	Shown in body of Report as Fig. 27
211	Data encoder Deck 210 subcommutation channel for Deck 400
212	Data encoder Deck 210 subcommutation channel for Deck 420
228	Shown in body of Report as Fig. 38
400	Data encoder Deck 400 synchronization channel
403	Unused telemetry channel
415	Shown in body of Report as Fig. 24
416	Shown in body of Report as Fig. 25
417	Shown in body of Report as Fig. 23
427	Unused telemetry channel



MODE	PHASE AND TELEMETRY	DECK RATE		TIME BETWEEN SAMPLES: MODE 1, 33-1/3 bps	MODE 1, 8-1/3 bps	MODE 2, 33-1/3 bps	MODE 2, 8-1/3 bps
1	MIDCOURSE-ENGINEERING ONLY						
2	LAUNCH AND CRUISE-ENGINEERING + RT SCIENCE	1	HIGH	4.20 sec	16.8 sec	12.6 sec	50.4 sec
3	ENCOUNTER-RT SCIENCE ONLY	2	MED	42.0 sec	168 sec (2.8 min)	126 sec (2.1 min)	504 sec (8.4 min)
4	POST-ENCOUNTER-ENGINEERING + NRT SCIENCE	3	LOW	420 sec (7 min)	1680 sec (28 min)	1260 sec (21 min)	5040 sec (1.4 hr)
	RT = REAL TIME; NRT = NONREAL TIME	3	LOW-LOW	840 sec (14 min)	3360 sec (56 min)	2520 sec (42 min)	10,080 sec (2.8 hr)

Fig. B-1. Engineering telemetry commutation

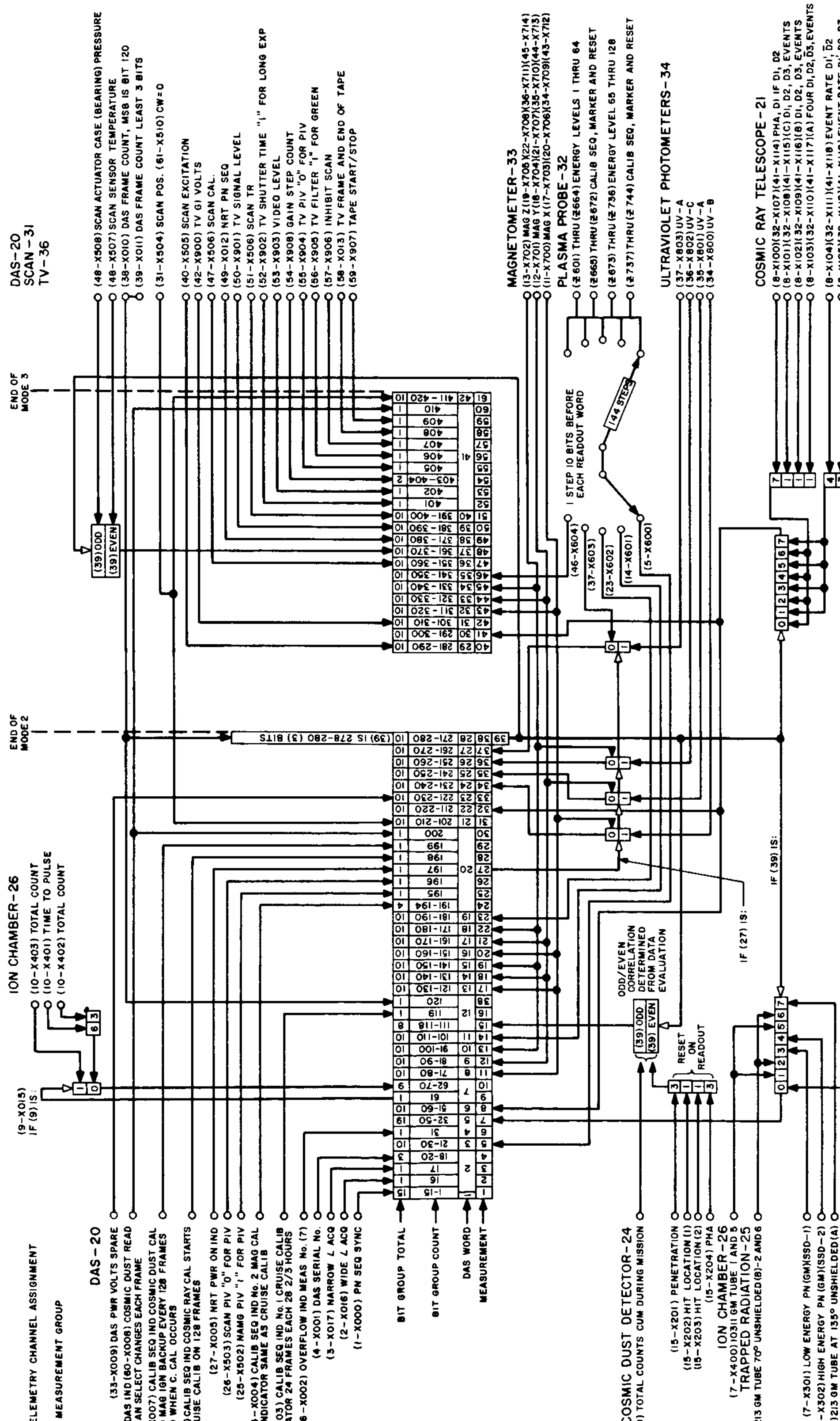


Fig. B-2. Real time science telemetry commutation

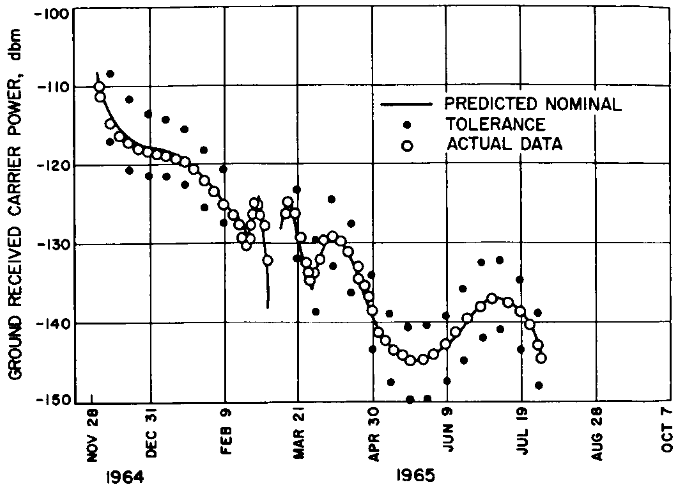


Fig. B-3. Channel 103, radio receiver coarse AGC

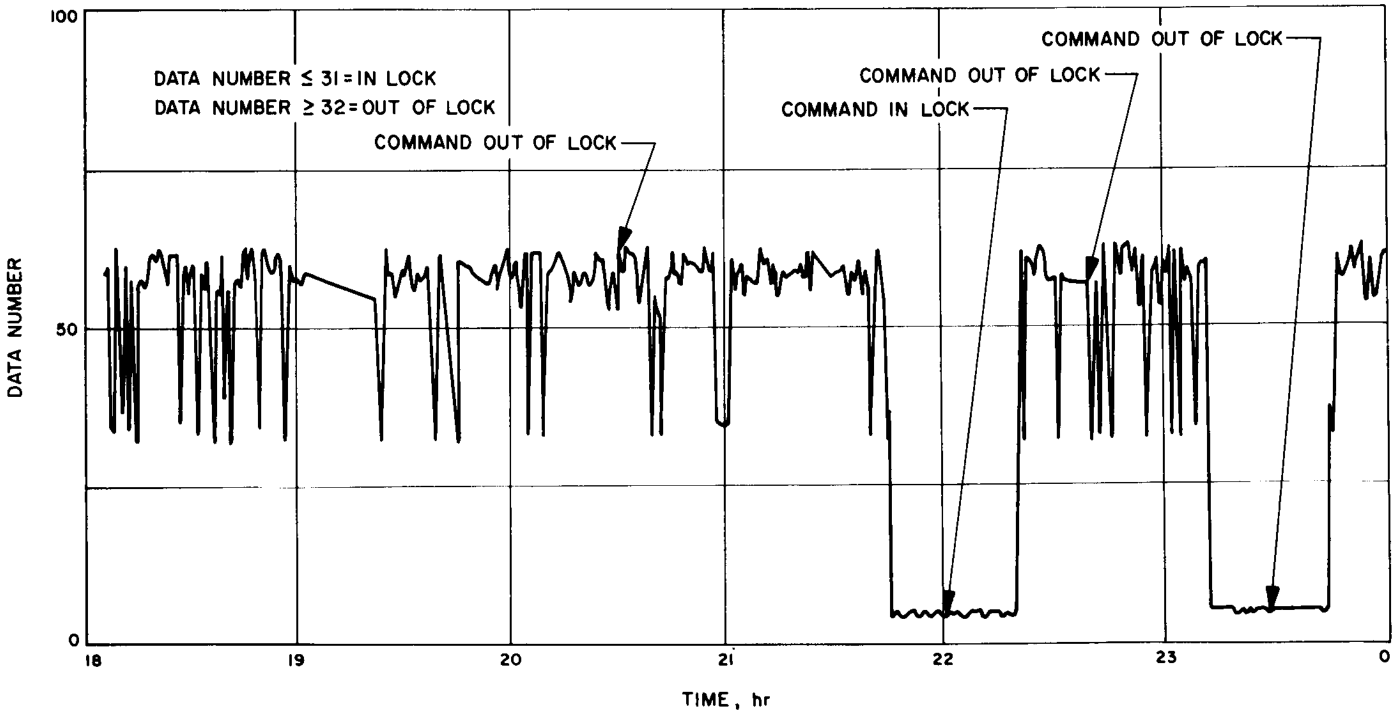


Fig. B-4. Channel 104, command detector monitor showing typical in-lock and out-of-lock indications

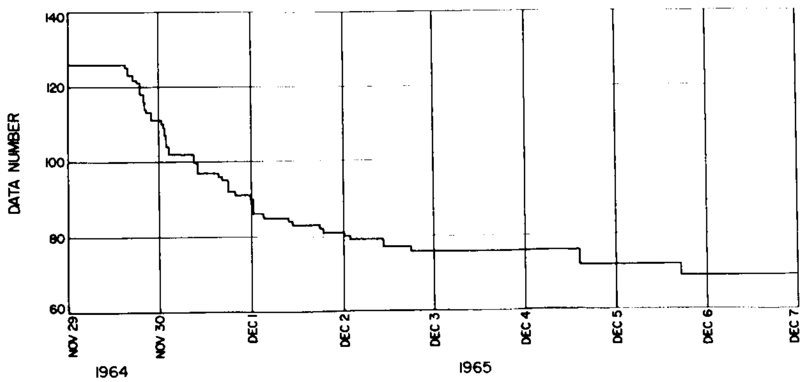


Fig. B-5. Channel 107, Earth sensor output

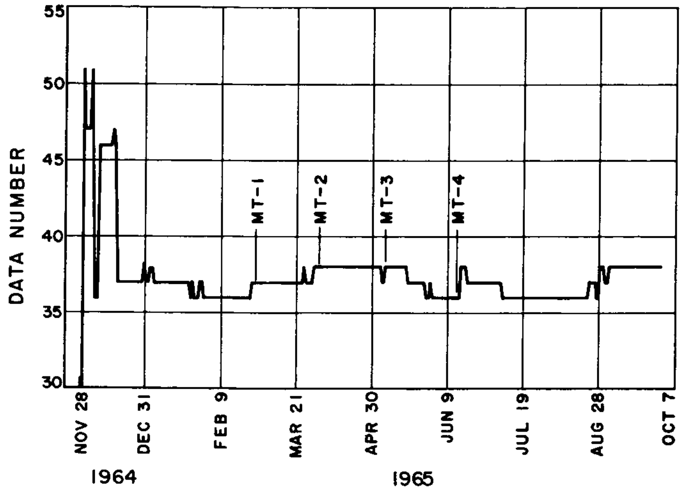


Fig. B-6. Channel 108, Canopus intensity

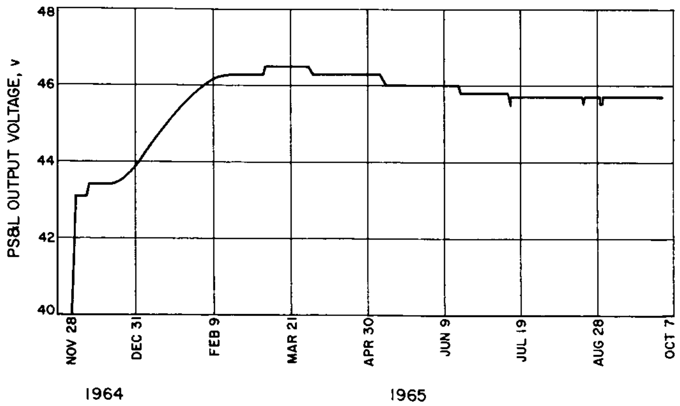


Fig. B-7. Channel 109, power switching and logic output voltage

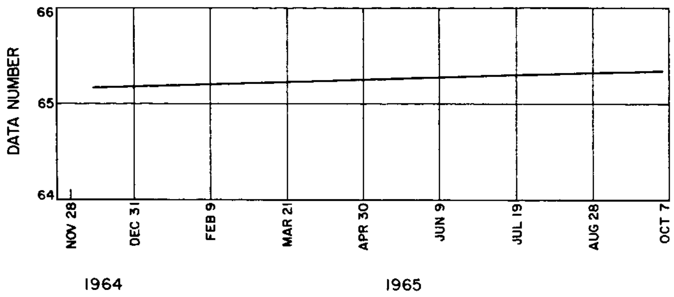


Fig. B-8. Channel 111, radio receiver static phase error

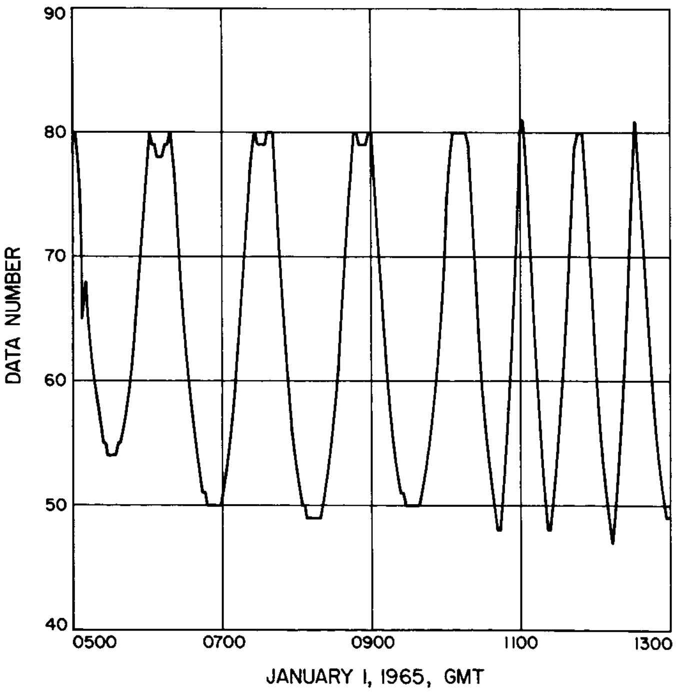


Fig. B-9. Channel 112, typical pitch position output

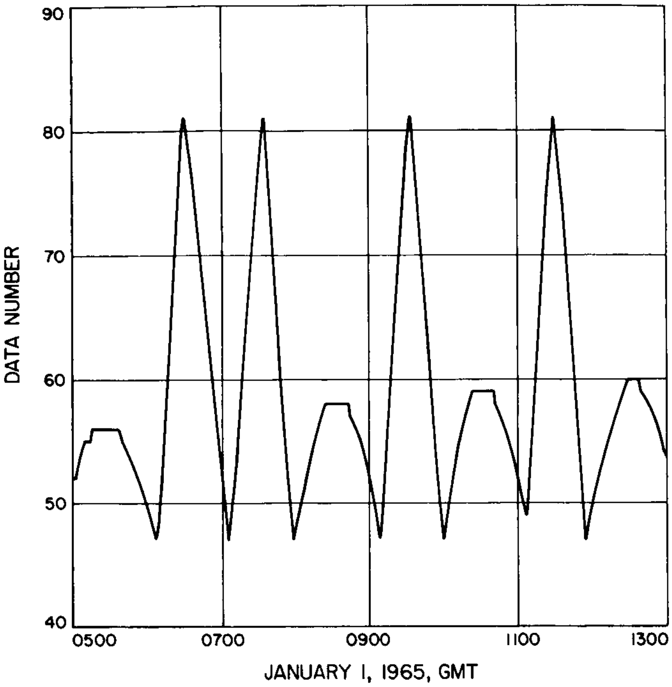


Fig. B-10. Channel 113, typical yaw position output

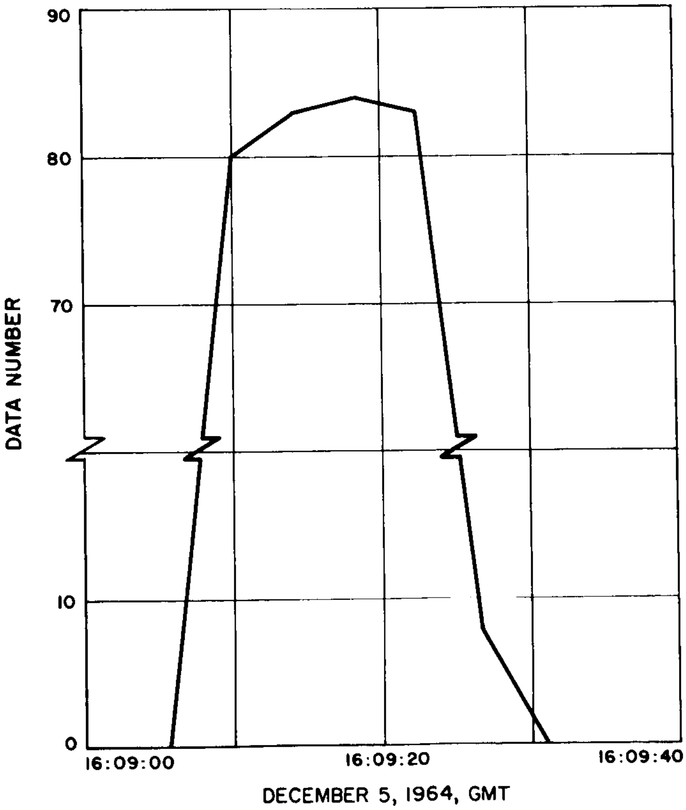


Fig. B-11. Channel 117, PIPS thrust chamber pressure during trajectory-correction maneuver

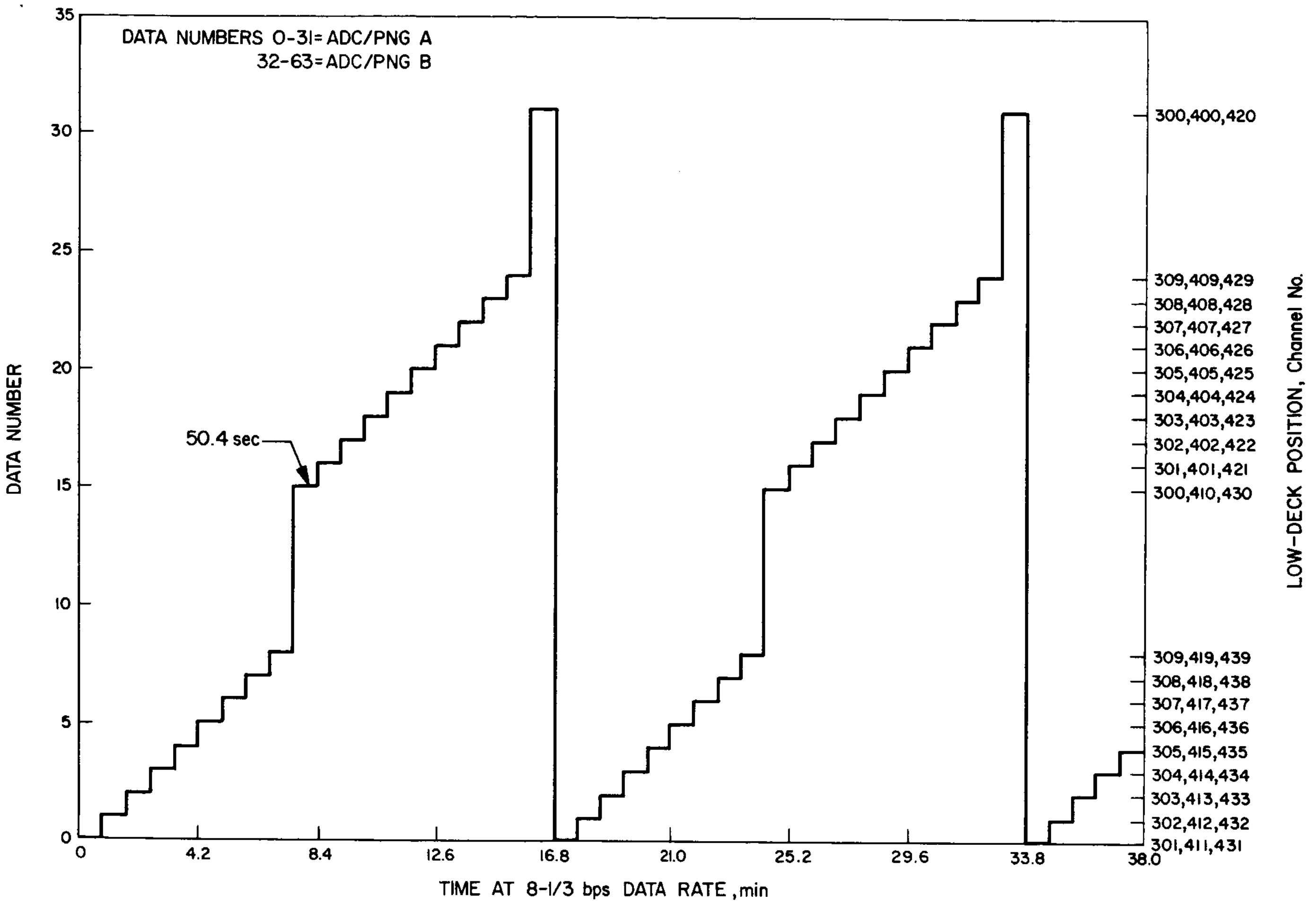


Fig. B-12. Channel 201, low-deck position indicator and ADC/PNG identification

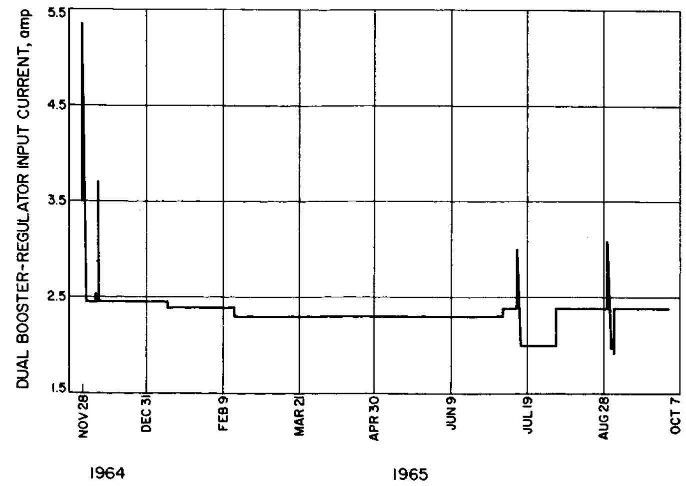


Fig. B-13. Channel 203, dual booster-regulator input current

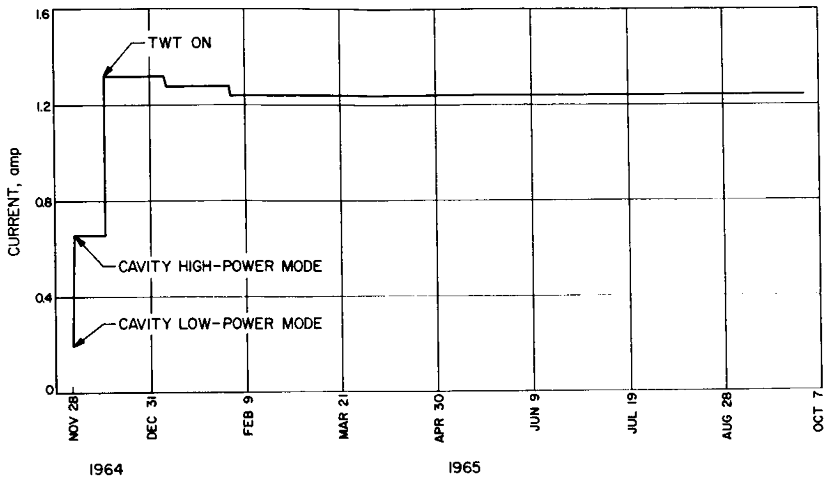


Fig. B-14. Channel 204, power switching and logic current to communications converter

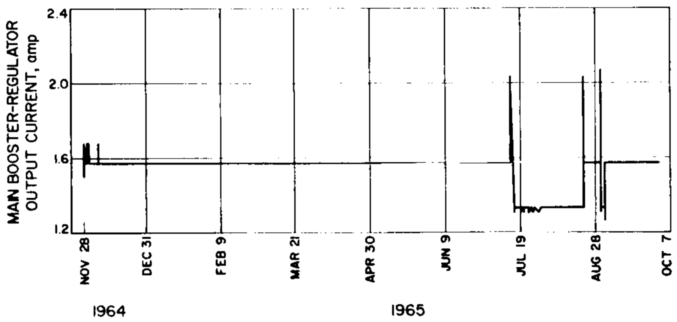


Fig. B-15. Channel 205, main booster-regulator output current

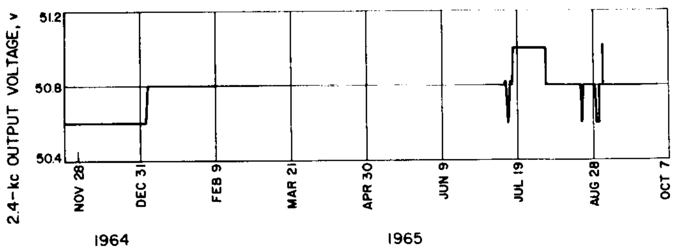


Fig. B-16. Channel 207, 2.4-kc inverter output voltage

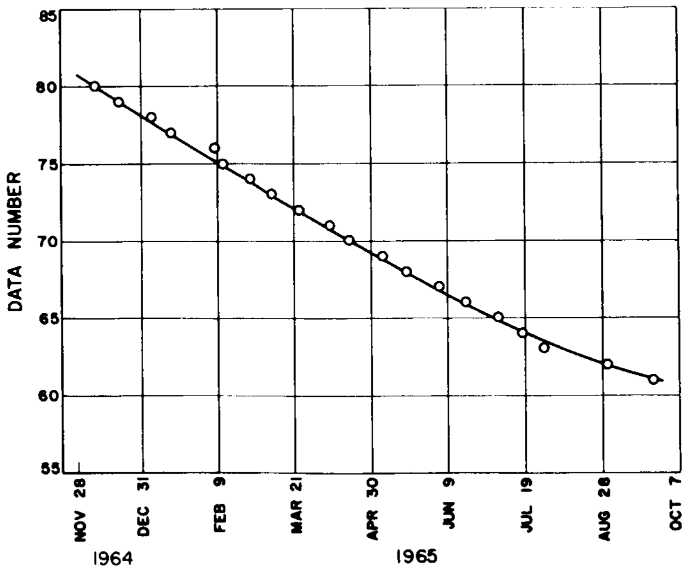


Fig. B-17. Channel 208, attitude control +X/−Y gas assembly nitrogen pressure

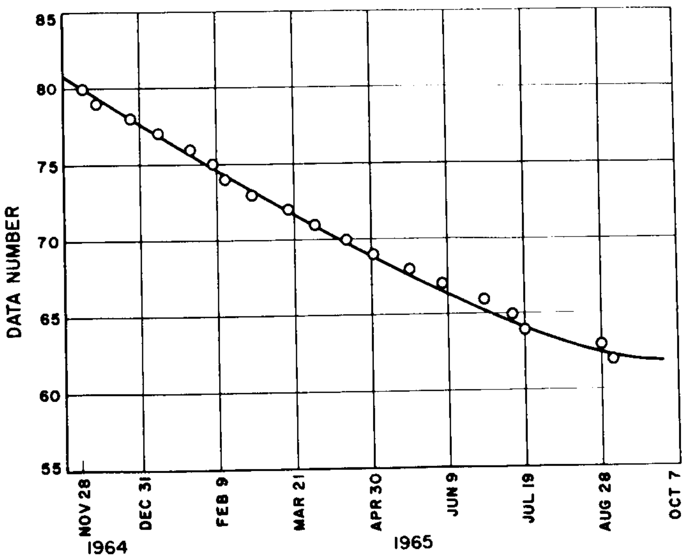


Fig. B-18. Channel 209, attitude control −X/+Y gas assembly nitrogen pressure

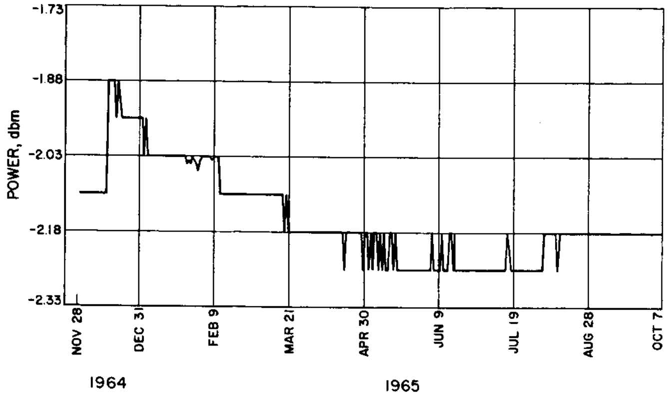


Fig. B-19. Channel 210, radio receiver local-oscillator drive

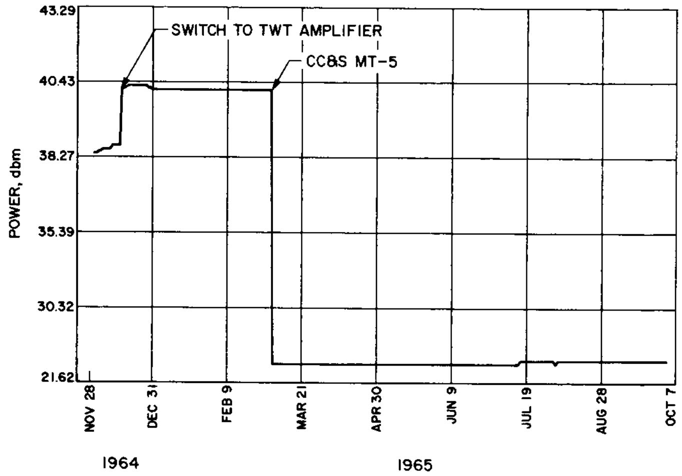


Fig. B-21. Channel 214, low-gain antenna power

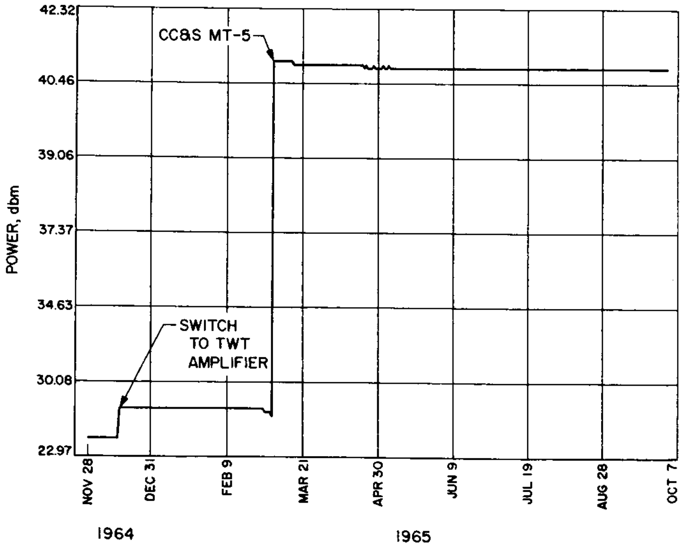


Fig. B-20. Channel 213, high-gain antenna power

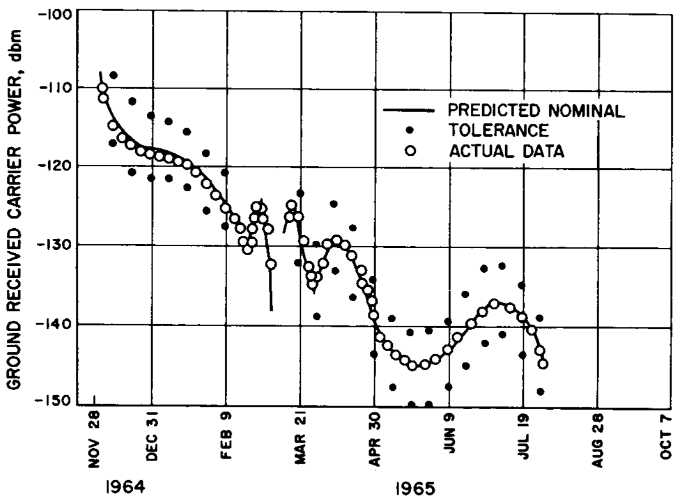


Fig. B-22. Channel 215, radio receiver fine AGC

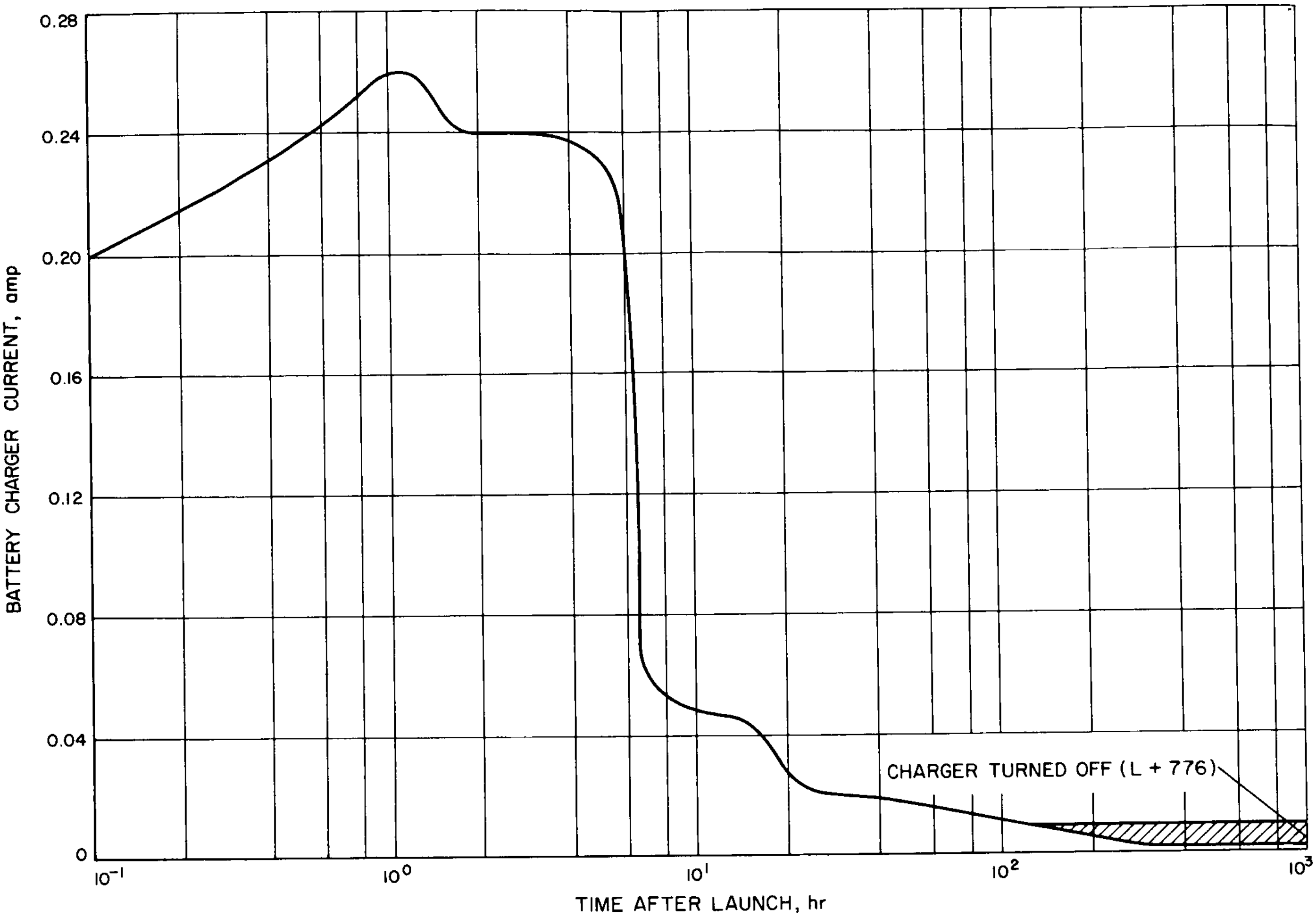


Fig. B-23. Channel 216, battery charge current until battery charge turn-off

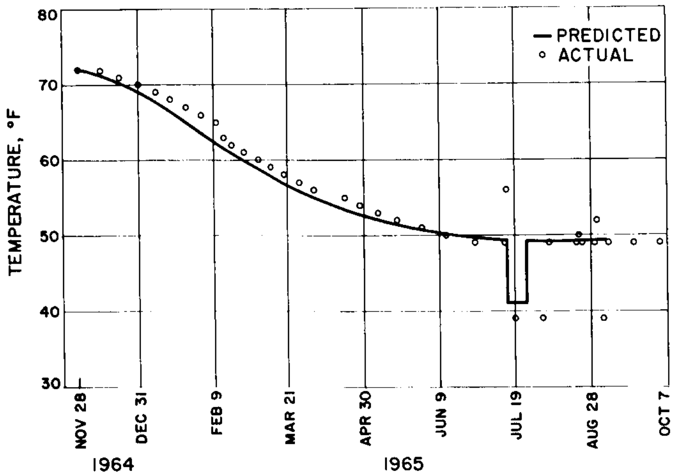


Fig. B-24. Channel 217, PIPS fuel temperature

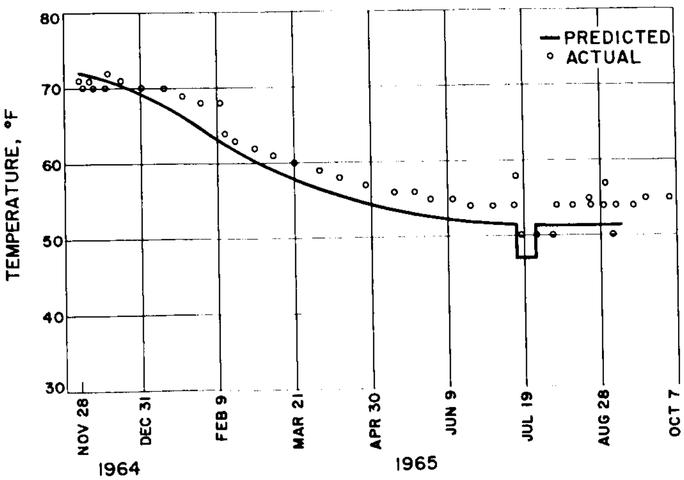


Fig. B-25. Channel 218, attitude control +X/−Y gas assembly nitrogen temperature

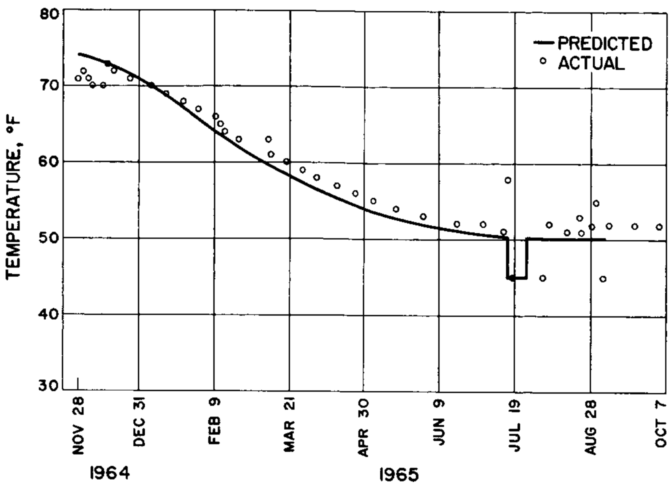


Fig. B-26. Channel 219, attitude control −X/+Y gas assembly nitrogen temperature

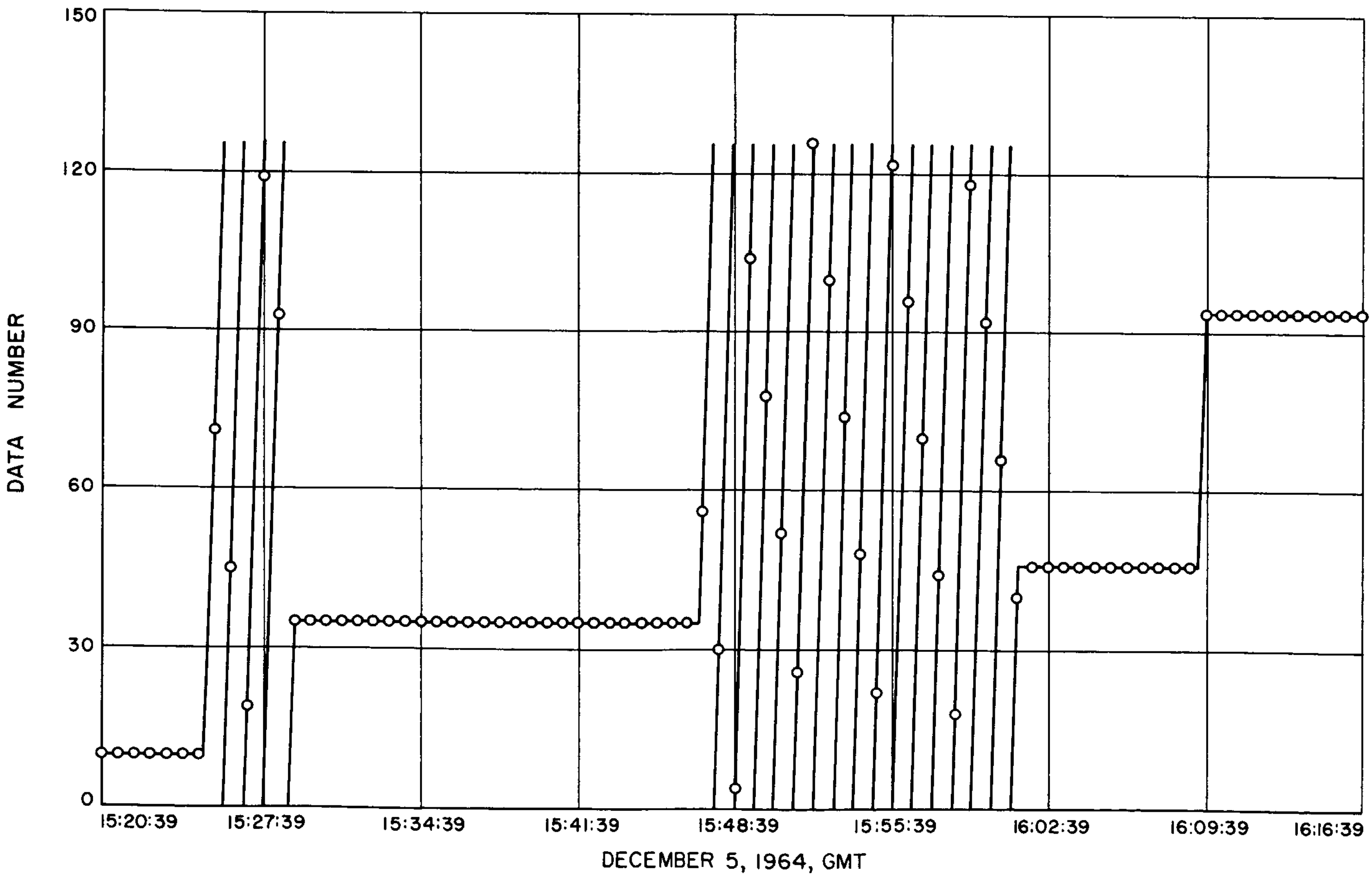


Fig. B-27. Channel 220, CC&S event timing during trajectory-correction maneuver

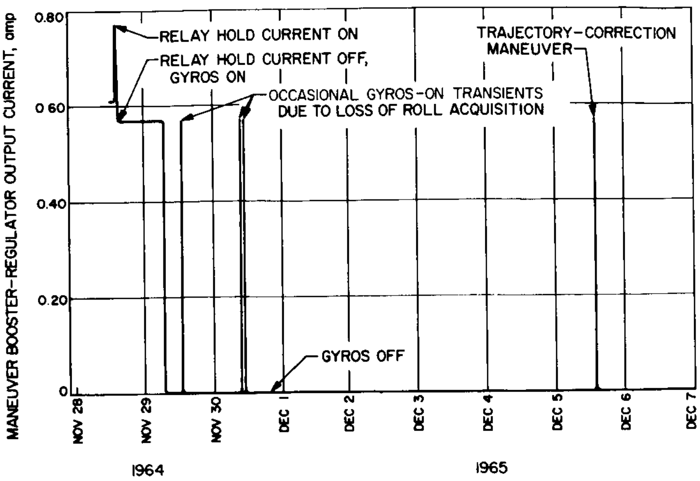


Fig. B-28. Channel 221, maneuver booster-regulator output current during launch and trajectory-correction maneuver

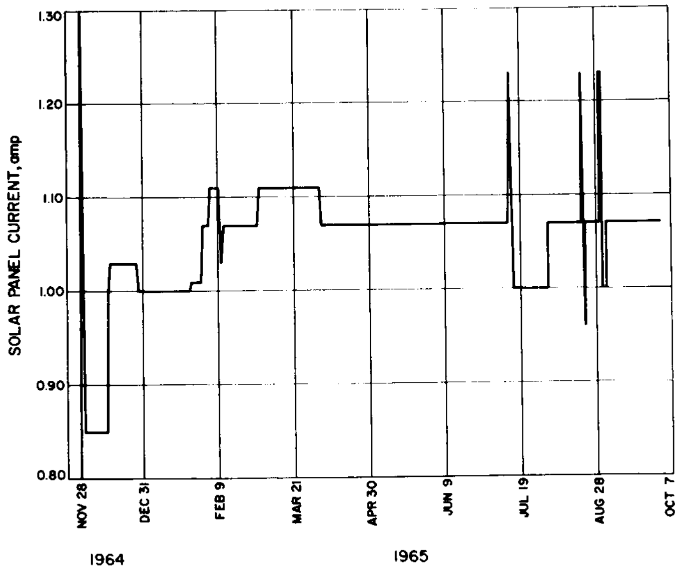


Fig. B-29. Channel 222, solar panel 4A1 output current

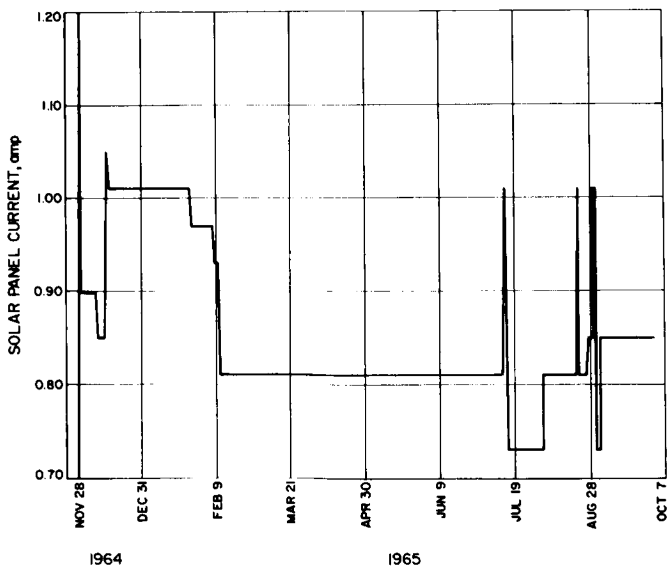


Fig. B-30. Channel 223, solar panel 4A5 output current

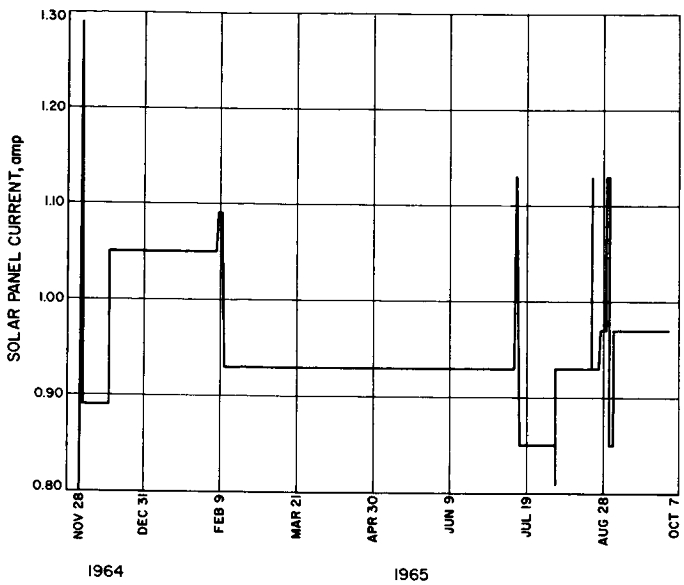


Fig. B-31. Channel 224, solar panel 4A3 output current

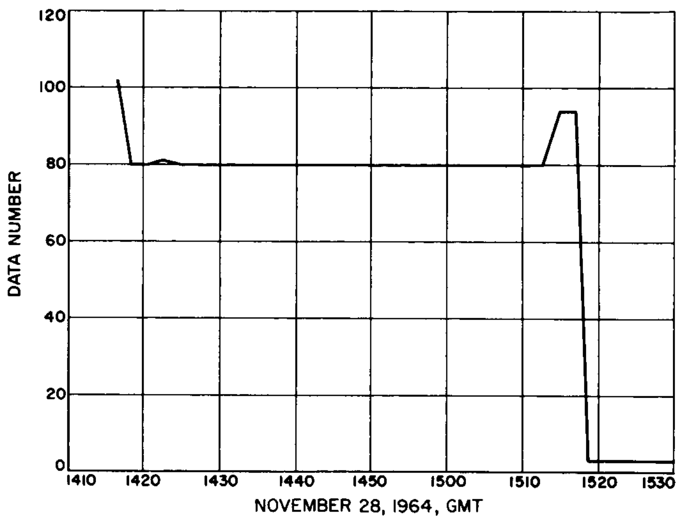


Fig. B-33. Channel 226, battery drain

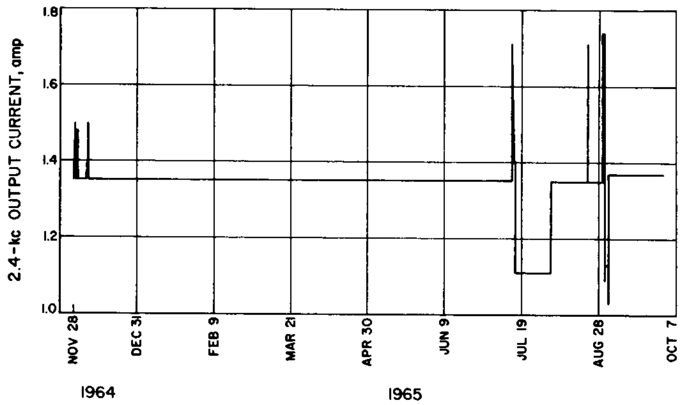


Fig. B-34. Channel 227, 2.4-kc inverter output current

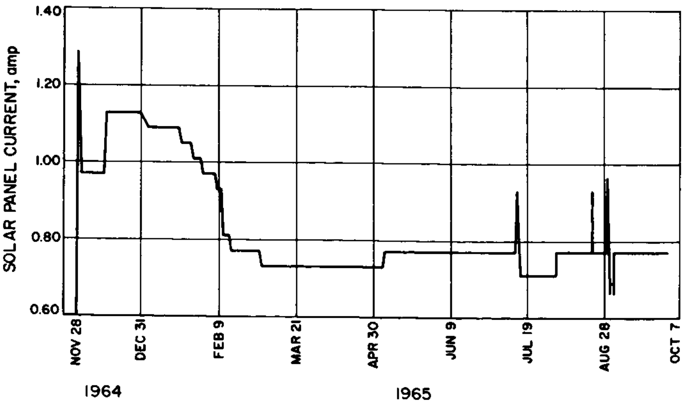


Fig. B-32. Channel 225, solar panel 4A7 output current

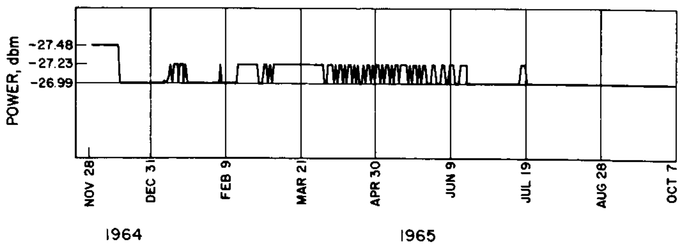


Fig. B-35. Channel 229, radio exciter output power

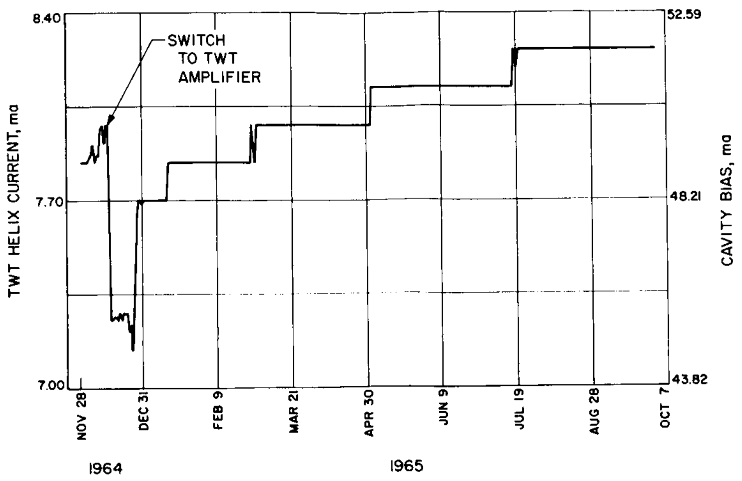


Fig. B-36. Channel 300, radio cavity amplifier bias or TWT helix current

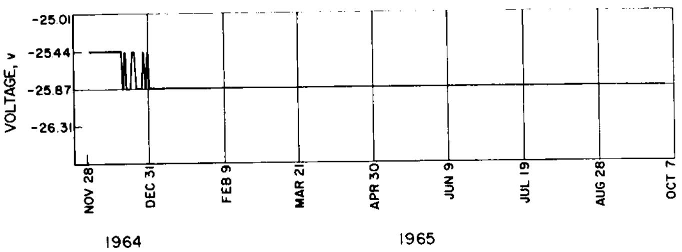


Fig. B-37. Channel 301, radio exciter voltage No. 1

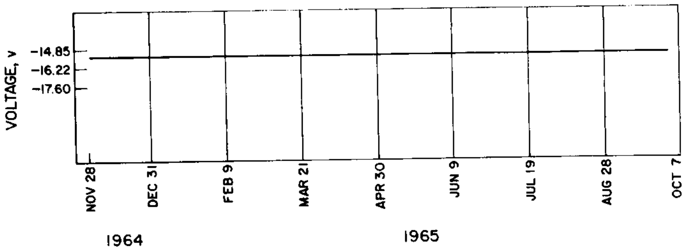


Fig. B-38. Channel 302, radio exciter voltage No. 2

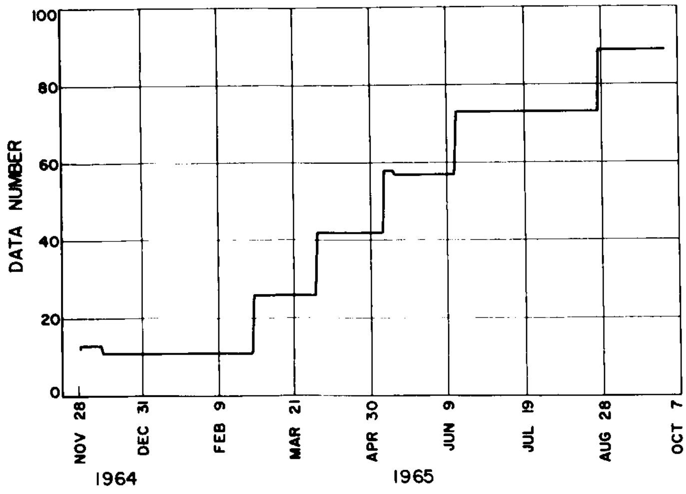


Fig. B-39. Channel 303, Canopus sensor cone angle

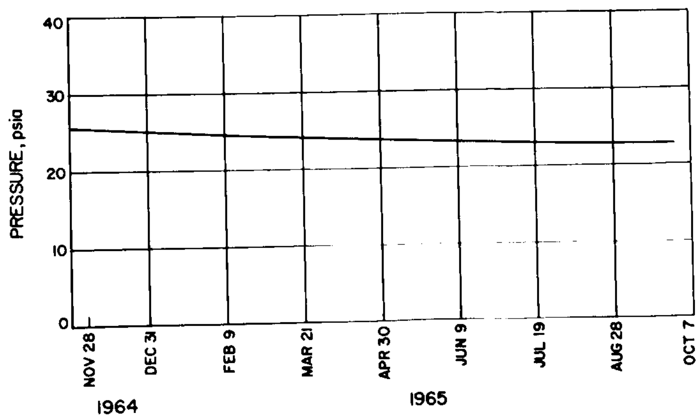


Fig. B-40. Channel 304, video storage tape recorder pressure

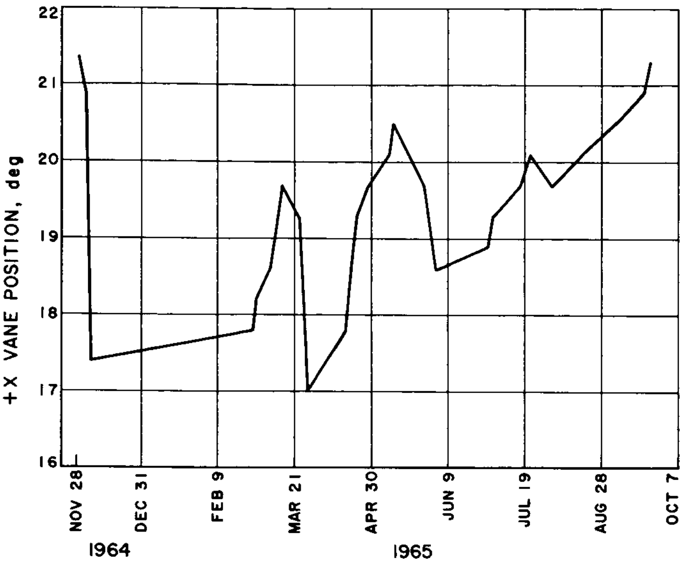


Fig. B-41. Channel 305, +X solar pressure vane position

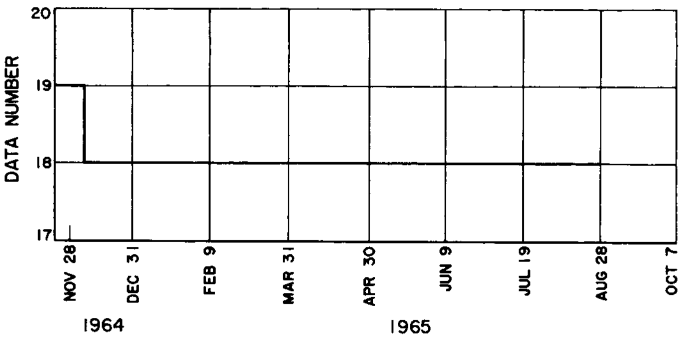


Fig. B-43. Channel 307, +Y solar pressure vane position

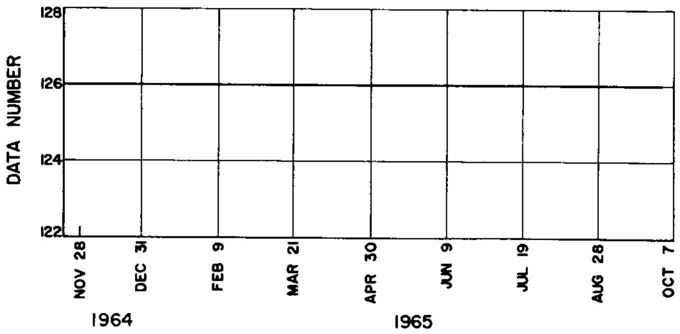


Fig. B-44. Channel 308, −Y solar pressure vane position

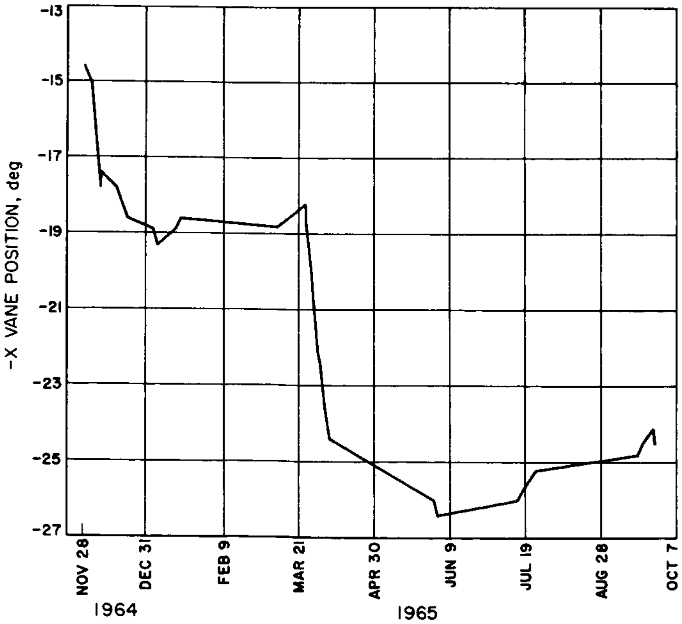


Fig. B-42. Channel 306, −X solar pressure vane position

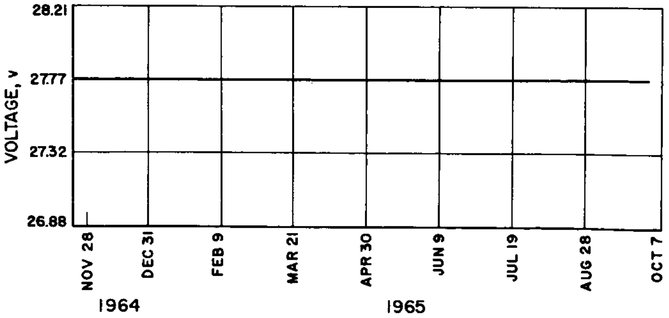


Fig. B-45. Channel 309, CC&S 28-v monitor

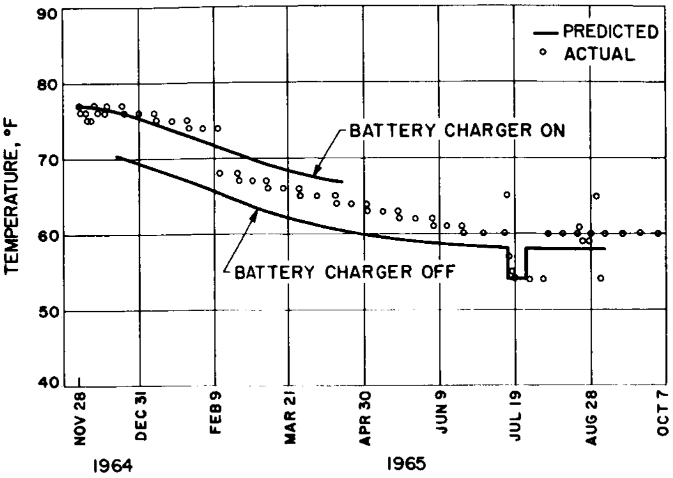


Fig. B-46. Channel 401, Bay I temperature

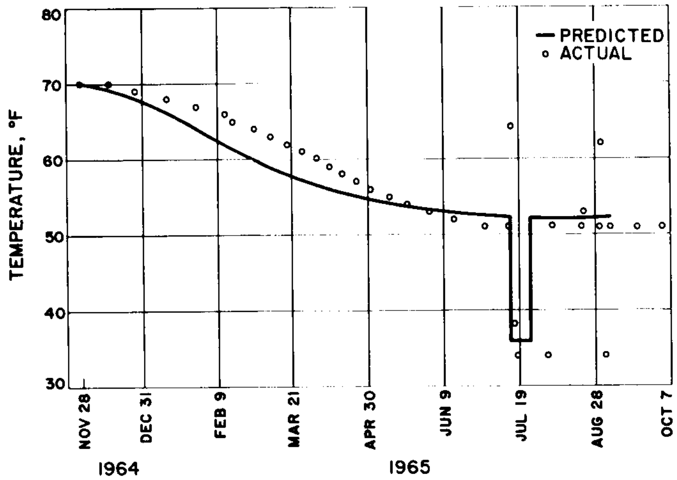


Fig. B-47. Channel 402, Bay III temperature

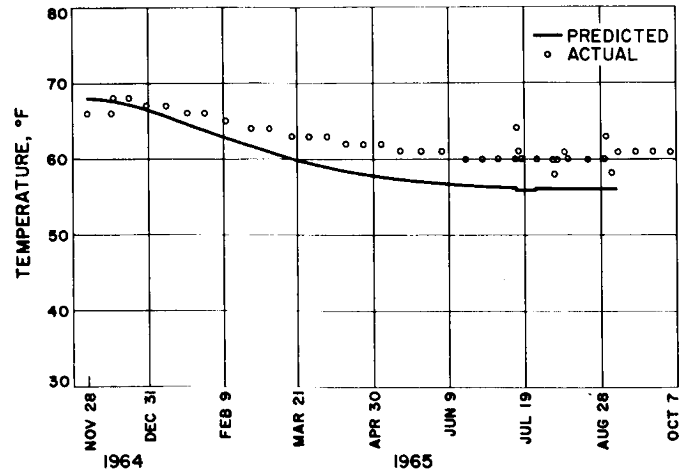


Fig. B-48. Channel 404, Bay V temperature

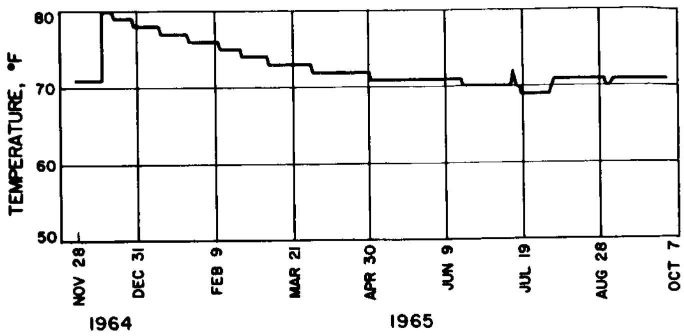


Fig. B-49. Channel 405, Bay VI temperature

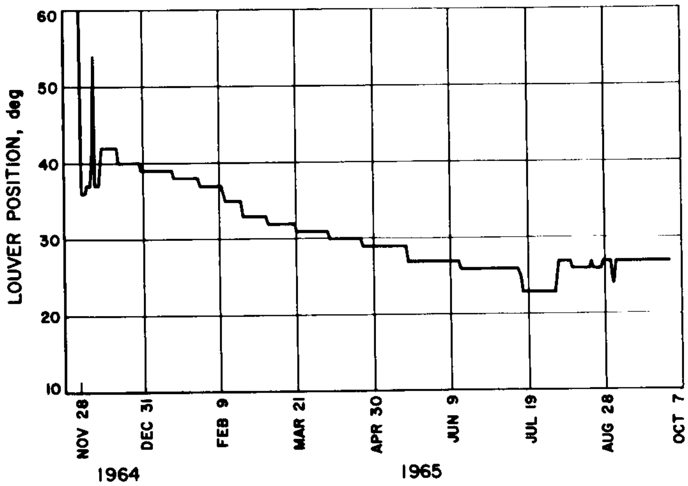


Fig. B-50. Channel 406, Bay VII temperature control louver position

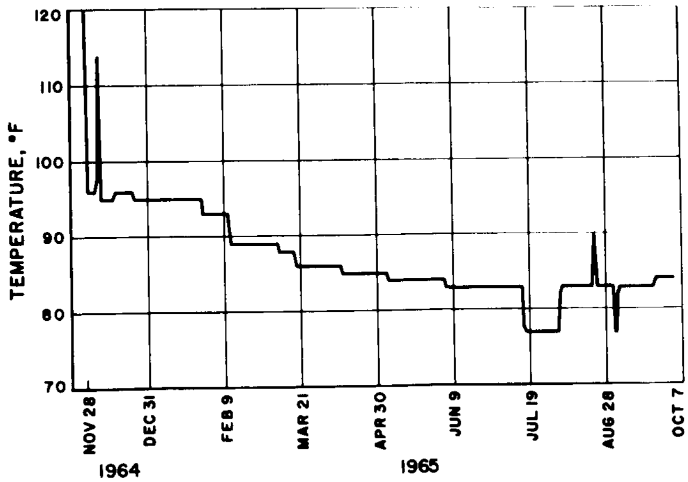


Fig. B-51. Channel 407, power regulator temperature

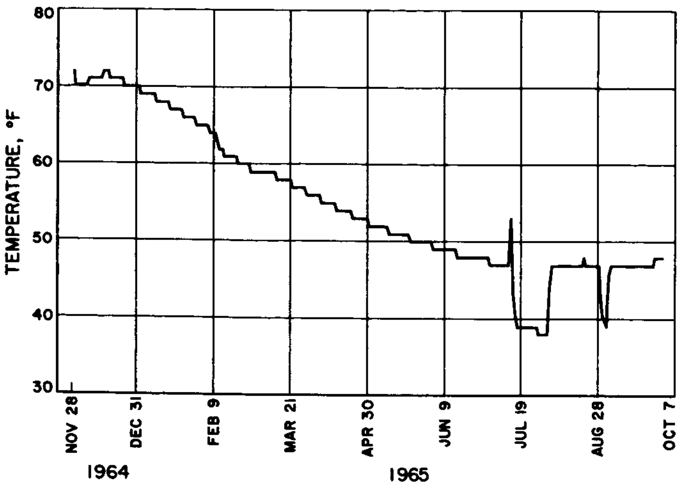


Fig. B-52. Channel 408, PIPS nitrogen temperature

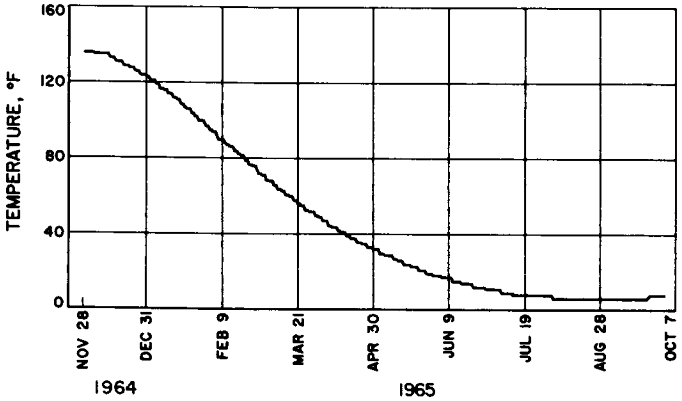


Fig. B-53. Channel 409, solar panel 4A1 front temperature

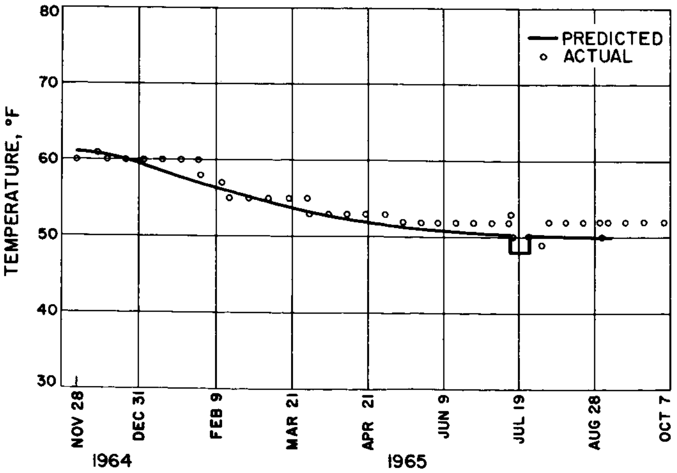


Fig. B-54. Channel 410, Canopus sensor temperature

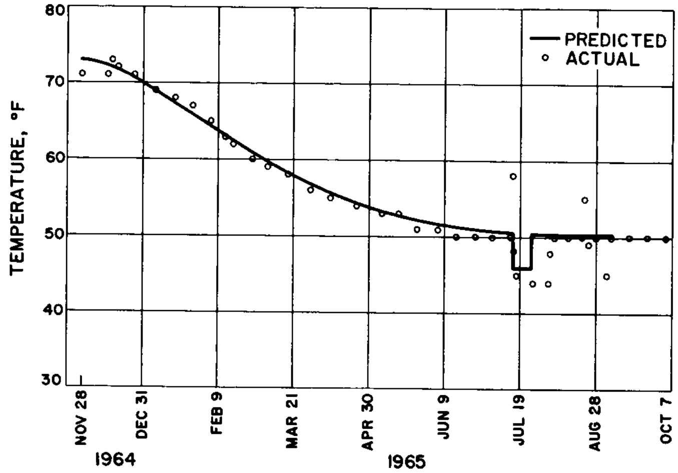


Fig. B-55. Channel 411, scan actuator temperature

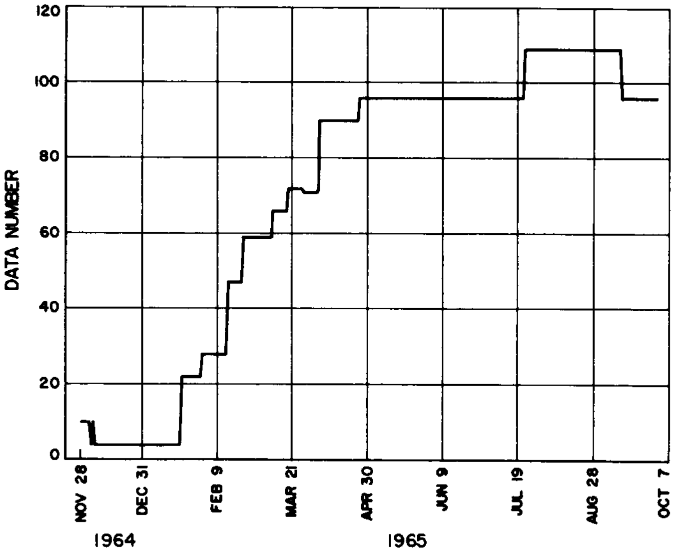


Fig. B-56. Channel 412, absorptivity standard white sample temperature

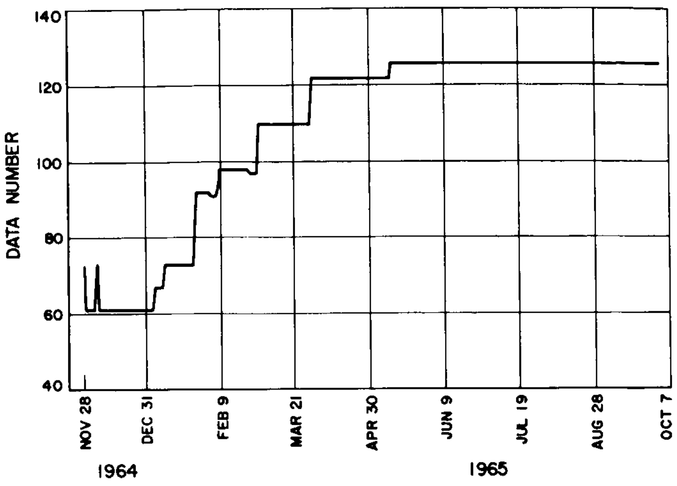


Fig. B-57. Channel 413, absorptivity standard aluminum-silicon sample temperature

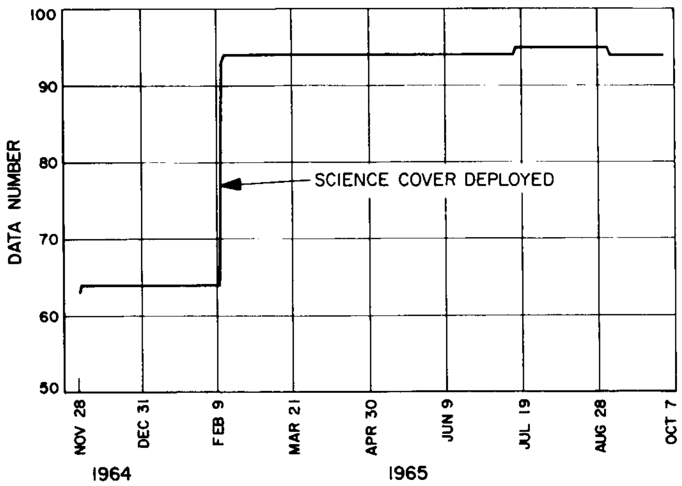


Fig. B-58. Channel 414, spacecraft identification and science cover deployment indication

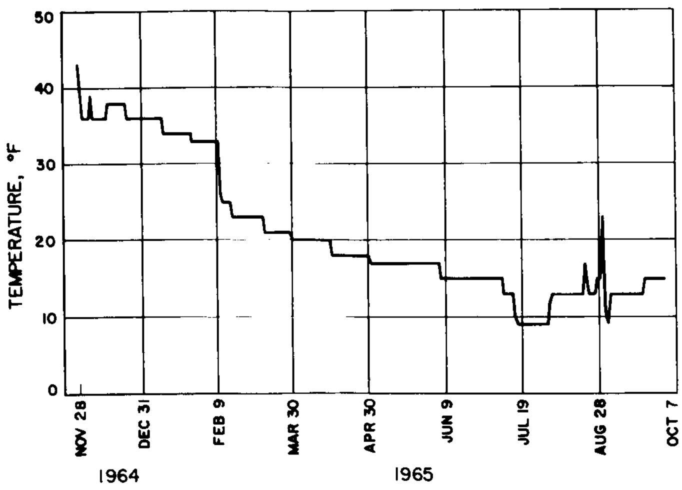


Fig. B-59. Channel 418, television temperature

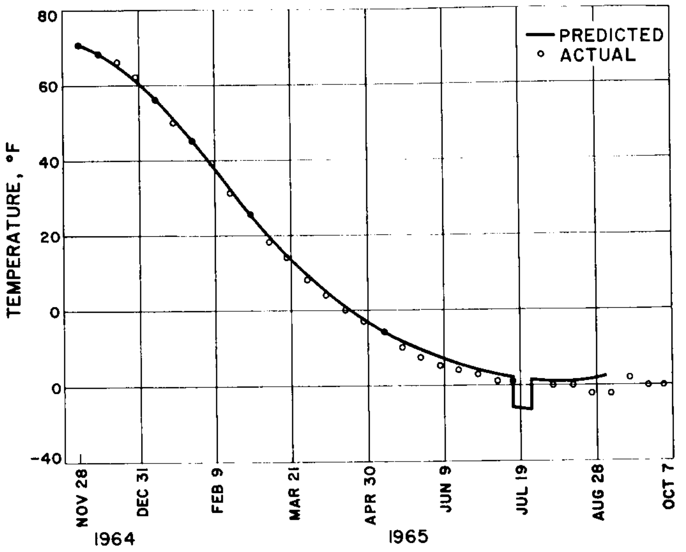


Fig. B-60. Channel 419, ionization chamber temperature

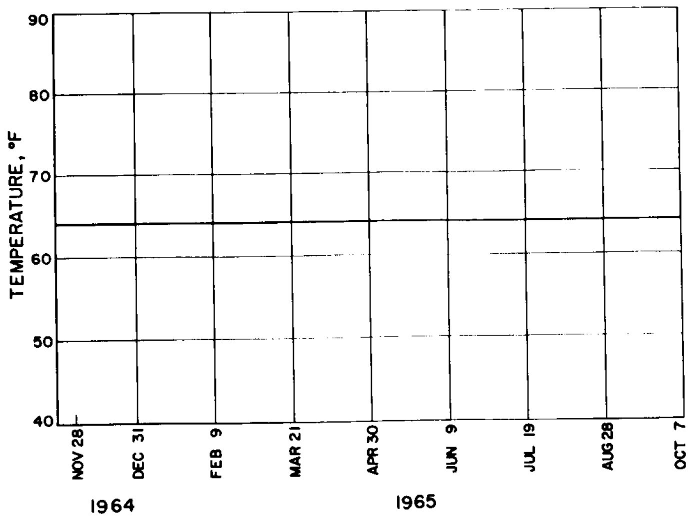


Fig. B-61. Channel 420, temperature reference

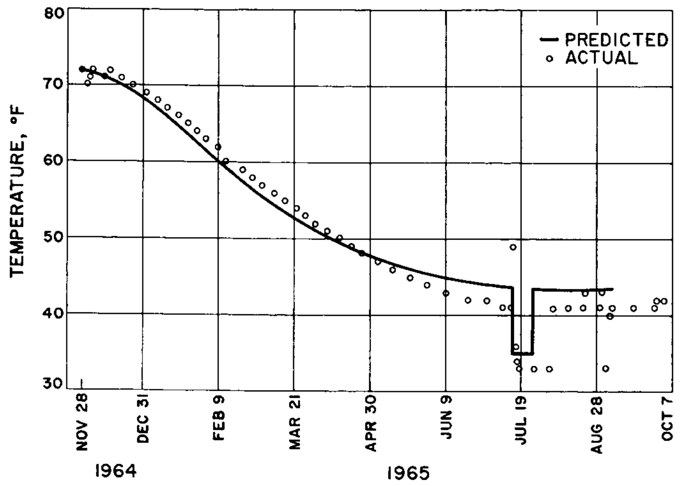


Fig. B-62. Channel 421, Bay II temperature

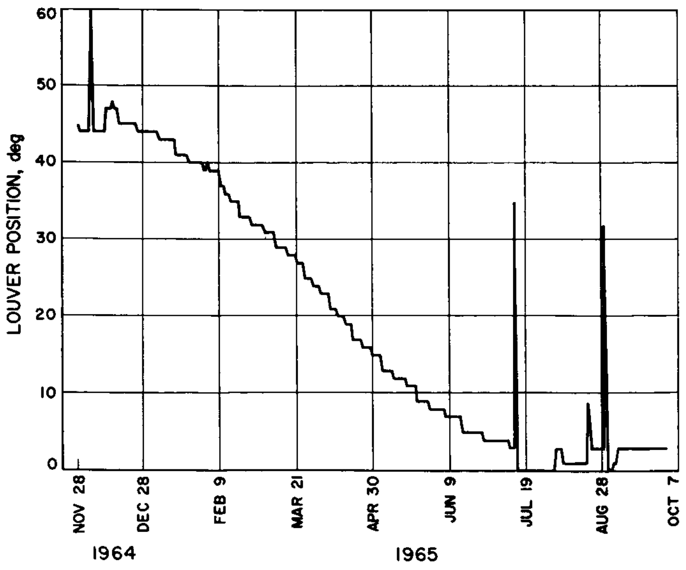


Fig. B-63. Channel 422, Bay III temperature control louver position

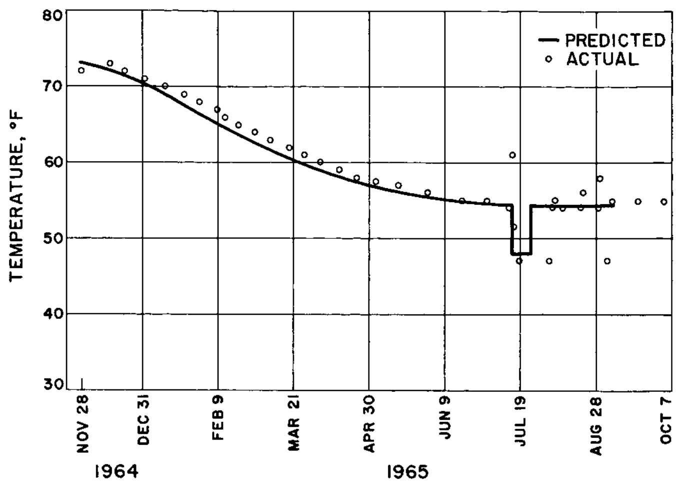


Fig. B-64. Channel 423, Bay IV temperature

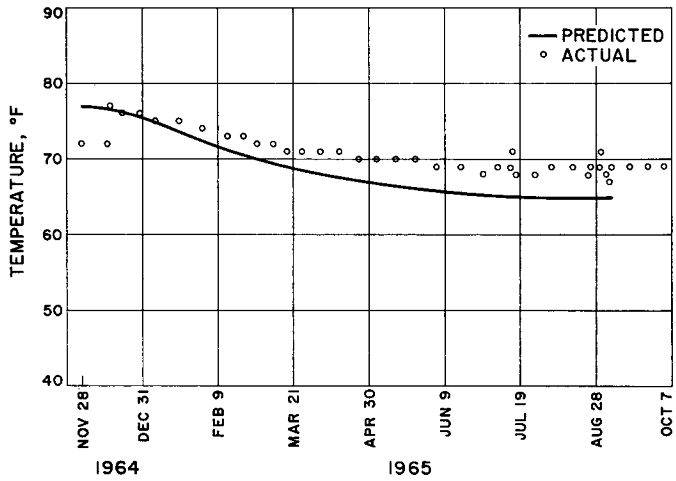


Fig. B-65. Channel 424, radio crystal oscillator temperature

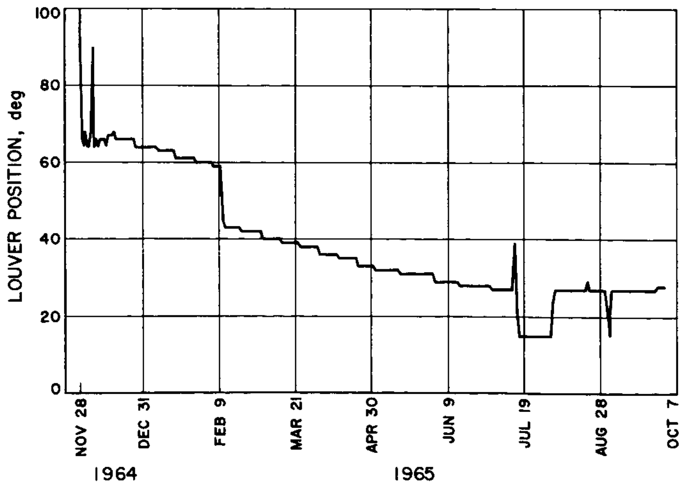


Fig. B-66. Channel 425, Bay I temperature control louver position

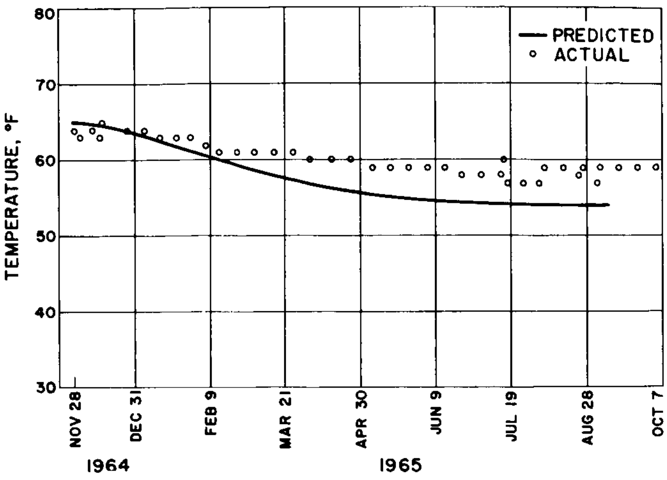


Fig. B-67. Channel 426, Bay VII temperature

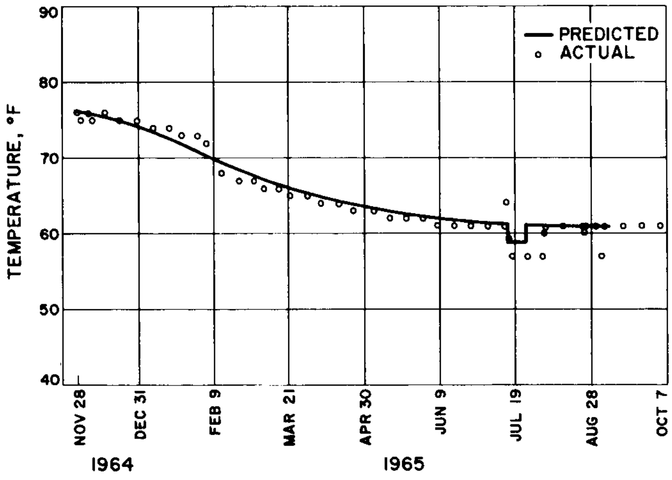


Fig. B-68. Channel 428, battery temperature

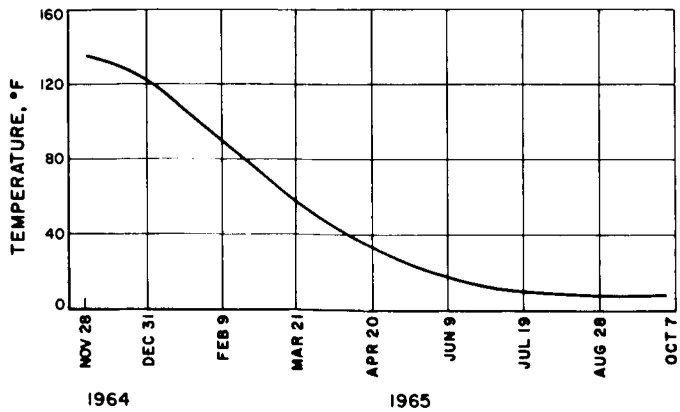


Fig. B-69. Channel 429, solar panel 4A5 front temperature

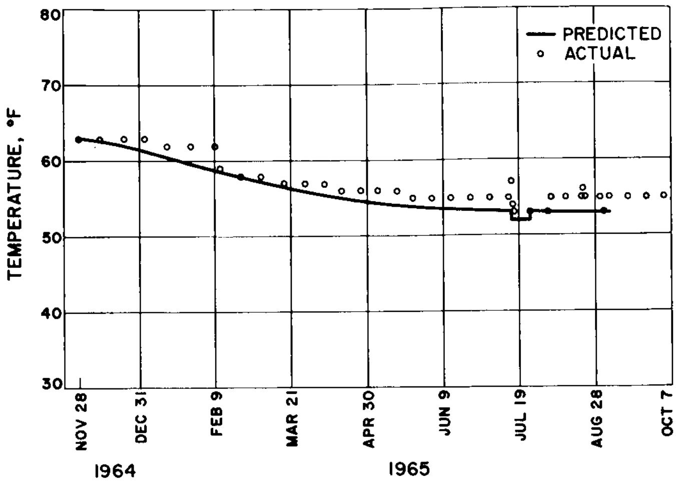


Fig. B-70. Channel 430, lower ring temperature above Canopus sensor

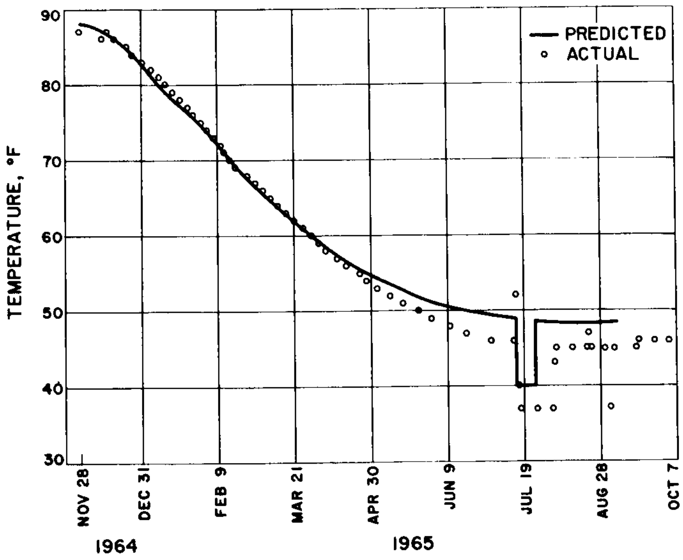


Fig. B-71. Channel 431, upper ring temperature below Sun sensor

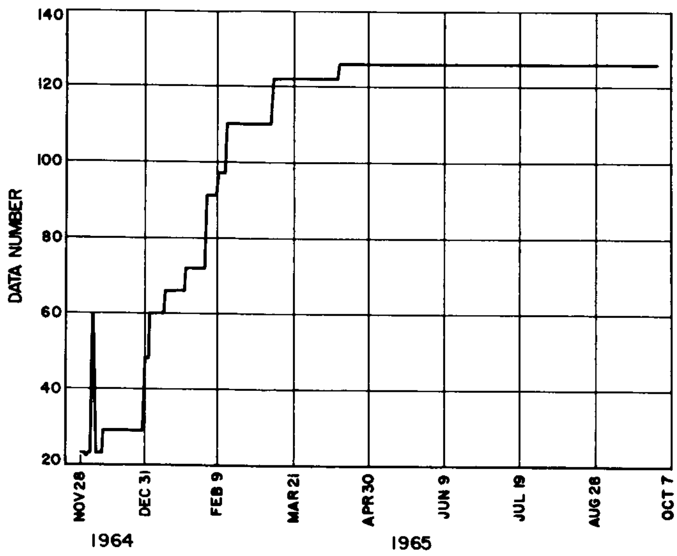


Fig. B-72. Channel 432, absorptivity standard black sample temperature

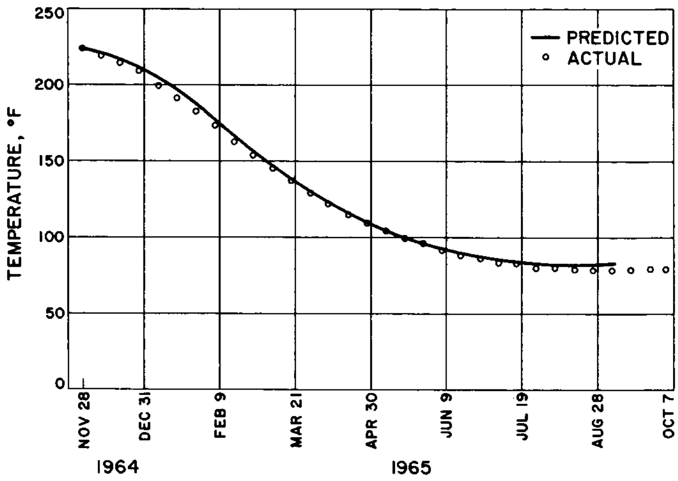


Fig. B-74. Channel 434, upper thermal shield temperature

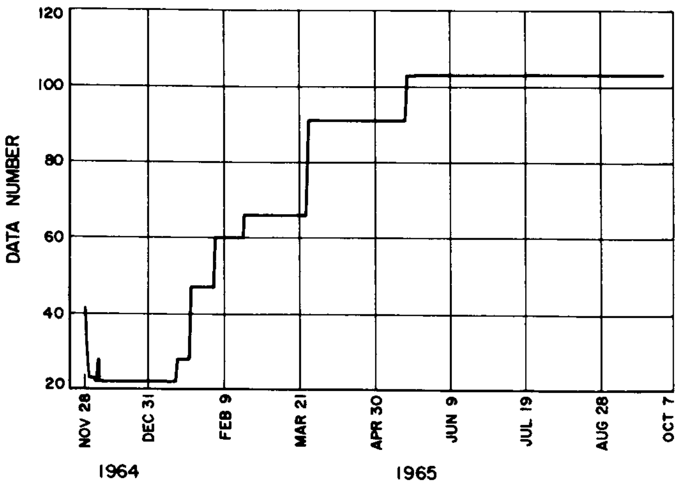


Fig. B-73. Channel 433, absorptivity standard polished aluminum sample temperature

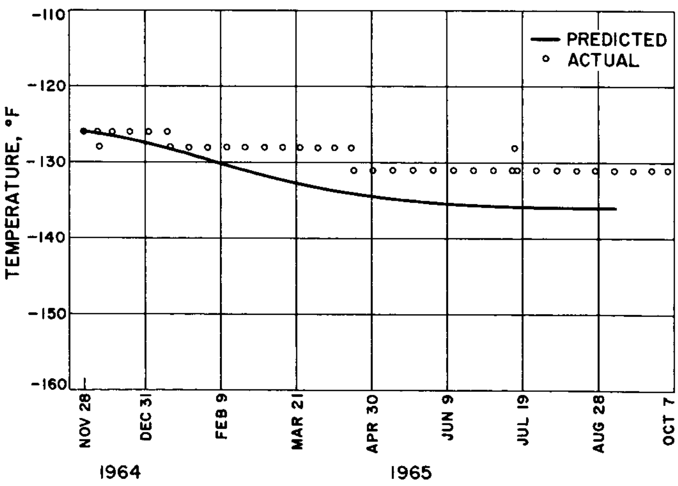


Fig. B-75. Channel 435, lower thermal shield temperature

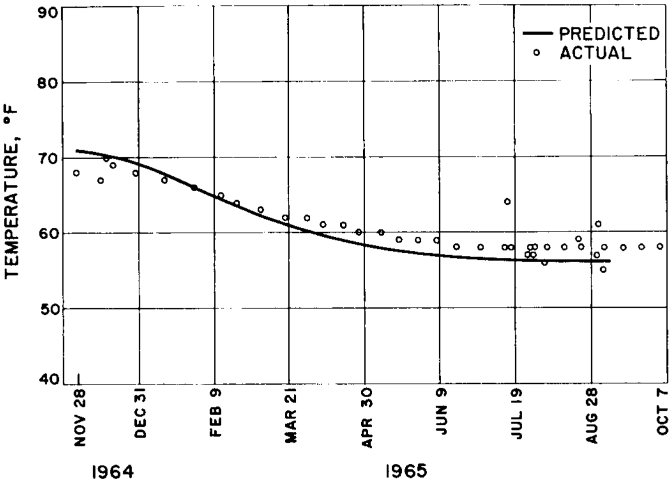


Fig. B-76. Channel 436, video storage tape recorder temperature

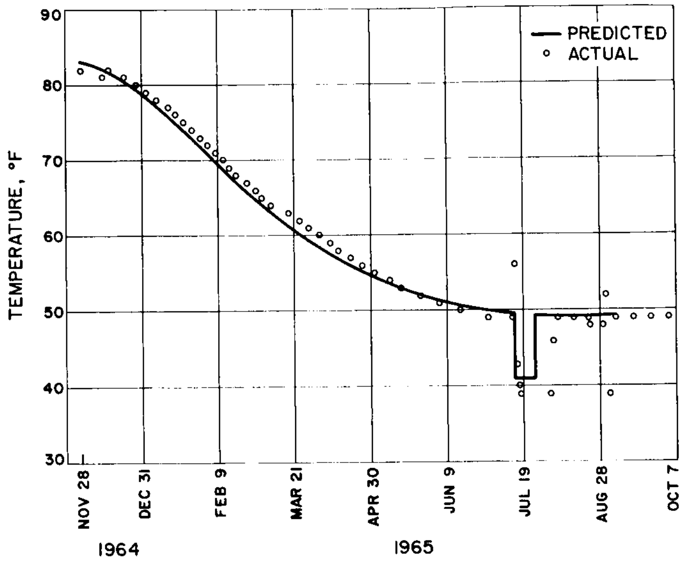


Fig. B-78. Channel 438, trapped-radiation detector temperature

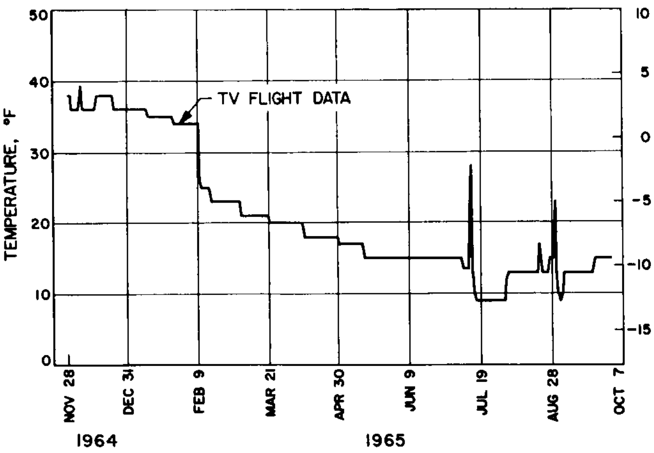


Fig. B-77. Channel 437, scan platform inertial and thermal simulator temperature

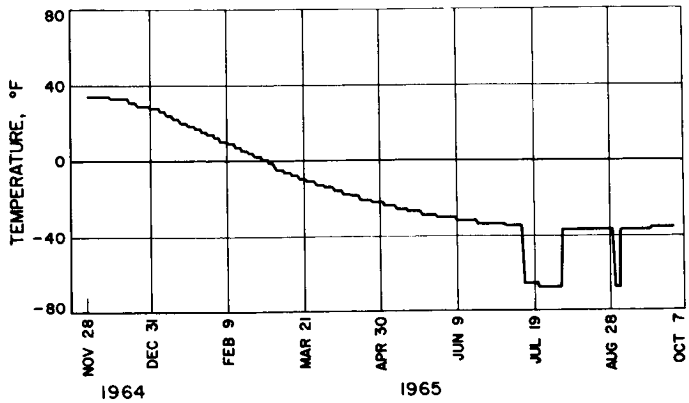


Fig. B-79. Channel 439, magnetometer sensor temperature

APPENDIX C

Flight Problems and Failures

A total of 73 flight Problem/Failure Reports (P/FRs) were written during the 307 days of the *Mariner IV* mission. Three of these were concerned with verifiable failures: solar vane electrical lockup, plasma probe degradation, and ion chamber failure. Several others reported possible or suspected failures such as the battery voltage rise. The majority[26] of the Problem/Failure Reports written, however, involved only minor departures from the nominal flight sequence or observed phenomena not known previously to be normal spacecraft characteristics. Many of these were written to document spacecraft behavior known to be probable, but not considered normal. A total of 24 were written to document roll disturbances, and 7 to document data encoder deck skips and resets.

A summary of the problems and failures reported during the *Mariner IV* mission is presented in Table C-1.

[26]Sixty-three of the total of 73 Reports.

Table C-1. Problem/Failure Reports

SPAC log number	P/FR number	Subsystem	Nature of problem	Problem date
001	10133	008	Extra pyro events at L-1	November 28, 1964
002	10134	007	Solar vane deployment	
003	10135	006	Data encoder deck skips	
004	10136	011	Temperatures low	
005	10137	007	Roll transient	30
006	10138	007	Canopus loss	December 2
007	10139	007	Canopus loss after DC-27	4
008	10140	006	Data encoder deck skips	6
009	10141	006	Data encoder deck skips	7
010	10142	007	Canopus loss	
011	10143	007	Loss of γ Velorum	9
012	10144	007	Loss of γ Velorum	9
013	10145	007	Loss of γ Velorum	13
014	10146	007	Loss of γ Velorum	17
015	10147	007	Roll transient	26
016	10113	002	TWT helix current rise	26
017	10112	011	Black absorptivity standard no switch	28
018	10115	002	MT-6 RF power drop observed at DSIF 41	January 3, 1965
019	10150	002	Interferometer effect	February 22
020	10158	002	CY-1 RF dropouts	March 1, 1965
021	10159	003	Command drop of lock	February 11
022	10160	007	Roll transient	March 11
023	10122	007	Roll transient	February 17
024	10155	026	GM 10311 tube failure	9
025	10156	011	Black absorptivity standard early switch	13
026	10149	007	Roll transient	16
027	10117	004	Battery voltage increase	22
028	10157	006	Data encoder deck skip at MT-1	27
029	10111	024	Cosmic dust detector film indicators	March 5
030	10161	002	Spacecraft AGC change at MT-5	5
031	10162	002	DSIF 11 failure to lock up	11
032	10163	002	DSIF 42 failure to lock up	12
033	10165	007	Roll transient	14
034	10166	002	DSIF 11 failure to lock up	15
035	10103	026	Ion chamber	17
036	10167	007	Roll transient	18
037	10174	004	PS&L voltage drop	April 1
038	10173	002	Circulator isolation	March 26

Table C-1. Problem/Failure Reports (cont'd)

SPAC log number	P/FR number	Subsystem	Nature of problem	Problem date
039	10171	007	Solar pressure vane activity	Long term
040	10170	007	Gas valve excessive torques	Long term
041	10175	007	Minor roll transient	April 7, 1965
042	10116	007	Solar pressure vane lockup	January 11
043	10121	197	Ground command OSE dumped lock during DC-15	January 4
044	10176	002	1-db drop at MT-2	April 2
045	10101	024	Cosmic dust detector transients November 28, 1964	December 3
046	10102	032	Plasma transients November 28, 1964	3
047	10132	032	Plasma probe failure	21
048	10177	007	Minor roll transient	April 10
049	10119	005	Change in CC&S clock frequency incorrectly calculated	21
050	10178	007	Minor roll transient	25
051	15654	011	Aluminum sample did not step	15
052	10184	002	DSIF 11 and 42 dropout	May 9
053	10183	002	DSIF 51 could not lock up in 2 min	13
054	10182	002	DSIF 41 lost lock	May 13, 1965
055	10181	002	Multiple ground receiver losses	14
056	10187	007	Minor roll transient	25
058	10188	002	Telemetry signal decrease in 2-way lock	25
059	10190	007	Roll transient	June 9
060	10191	007	Roll transient	22
061	10192	007	Roll transient	
062	10193	007	Roll transient	
063	10195	007	Roll transient	
064	10196	016/020	Extra EOTs	15
065	10199	—	Extra event at MT-7	
066	10401	006	Data encoder deck skip at DC-25	
067	10402	016	Extra track change event	August 11
068	10403	020	Cyclic noise every 16 min	July 14
069	10404	006	Data encoder deck reset at DC-26	August 3
070	10405	007	10 roll transients	NA
071	10407	024	Cosmic dust penetrations	September 15
072	10408	006	Data encoder deck skips	15
073	10409	021	Cosmic ray telescope shift	21
074	10088	010	PIPS fuel tank bladder incompatibility	February 3

APPENDIX D

Command Description

Table D-1. *Mariner* Mars 1964 commands

Command	Effect
DC-1	Transfers the data encoder to Mode 1 operation (all engineering words) as soon as the transfer is acceptable to the data encoder transfer logic.
DC-2	Transfers the data encoder to Mode 2 operation (20 engineering words, 40 science words) as soon as the transfer is acceptable to the data encoder transfer logic; applies 2.4-kc power to the cruise science instruments.
DC-3	Transfers the data encoder to Mode 3 operation (all science words) as soon as the transfer is acceptable to the data encoder transfer logic.
DC-4	Transfers the data encoder to Mode 4/1 operation (television picture data or engineering data) as soon as the transfer is acceptable to the data encoder transfer logic. If television picture data is available from the video storage tape recorder, television data is telemetered; if no television data is present (as between television pictures), then engineering data is telemetered; removes 2.4-kc power from the cruise science instruments.
DC-5	Transfers the data encoder from one bit rate to the other. The data encoder can operate at either 8⅓ or 33⅓ bps.
DC-6	Transfers the data encoder from one analog-to-digital converter/pseudonoise generator (ADC/PNG) to the other. The data encoder has two ADC/PNGs, A and B.
DC-7	Transfers the radio from one power amplifier to the other. The radio subsystem has two power amplifiers, A (TWT) and B (cavity).
DC-8	Changes the radio from one exciter to the other. The radio subsystem has two exciters, A and B.
DC-9	Turns the spacecraft radio ranging-receiver either on or off.
DC-10	Causes the radio subsystem circulator switches to be conditioned so that the spacecraft transmits via the high-gain antenna and receives via the low-gain antenna.
DC-11	Causes the radio subsystem circulator switches to be conditioned so that the spacecraft transmits and receives via the high-gain antenna.
DC-12	Causes the radio subsystem circulator switches to be conditioned so that the spacecraft transmits and receives via the low-gain antenna.
DC-13	Removes the attitude control excitation power from the CC&S control lines so that the attitude control functions that are controlled by the CC&S are disabled. DC-13 also prevents the pyrotechnics control circuitry from firing the motor start and stop squibs.

Command	Effect
DC-14	Reverses the state of all the relays acted upon by DC-13. DC-14, therefore, is a reset for DC-13 and reverts the attitude control and pyrotechnics subsystems back to CC&S control.
DC-15	Causes the Canopus sensor roll error signal to be applied to the roll gas jet electronics at all times, regardless of whether or not the roll acquisition logic is satisfied and also prevents the roll search signal from being applied to the roll channel, and roll acquisition logic violations from turning on the gyros.
DC-16	Initiates a narrow-angle acquisition signal and thereby conditions the data automation subsystem (DAS) logic to begin the television picture taking sequence and to transfer the data encoder to Mode 3.
DC-17	Causes a step change in the Canopus sensor cone angle by changing the voltage on the deflection plates of the Canopus tracker's image dissector.
DC-18	Turns on the gyros (in the inertial mode) and the Canopus sensor Sun shutter, and turns off the Canopus sensor. DC-18 also turns on the turn command generator and conditions the attitude control circuitry for commanded roll turns. Succeeding DC-18s cause cw 2.25-deg roll turns.
DC-19	Serves as the reset for DC-15, DC-18, and DC-20.
DC-20	Turns off the Canopus sensor and turns on the Canopus sensor Sun shutter; also inhibits the roll acquisition logic from turning on the gyros.
DC-21	Simulates a Canopus acquisition logic violation, turns on the gyros, and applies a negative roll search signal to the roll gas jet electronics; thereby causing the spacecraft to ccw roll search to acquire a new target. DC-21 will also cause the spacecraft to roll turn 2.25-deg ccw if preceded by a DC-18.
DC-22	Changes video storage tape tracks by applying power to a record head and gating the output of the playback amplifiers.
DC-23	Sets relays in the pyrotechnics subsystem so that CC&S commands M-6 and M-7 are routed to the squibs allotted to the second motor burn.
DC-24	Removes 400-cps single-phase power from the scan platform drive motor.

Table D-1. *Mariner* Mars 1964 commands (cont'd)

Command	Effect
DC-25	Causes 2.4-kc power to be applied to encounter science loads, the video storage subsystem, and also the cruise science loads if 2.4-kc power was off to cruise science; while at the same time applying 52 vdc from the booster-regulator to the 400-cps single-phase inverter which in turn supplies power to the scan subsystem drive motor and the video storage subsystem record motor. DC-25 also enables the battery charger boost mode and causes the pyrotechnics subsystem to energize the solenoid that releases the scan platform science cover.
DC-26	Removes 2.4-kc power from all of the science loads (video storage 2.4-kc power remains on) and 52-vdc power from the 400-cps single-phase inverter; also enables the battery charger boost mode.
DC-27	Starts the maneuver sequence by issuing the CC&S command M-1 (turn on gyros), by applying power to the maneuver clock, and by removing the maneuver clamp and a flip-flop reset signal from the CC&S maneuver circuitry.
DC-28	Removes 2.4-kc power from the video storage subsystem and enables the charge mode of the battery charger.
DC-29	Sets relays in the pyrotechnics subsystem so that the CC&S commands M-6 and M-7 are routed to the squibs allotted to the first motor burn.
QC-1-1	Sets pitch turn polarity and preloads the CC&S pitch shift register so that at a counting rate of 1 pulse/sec (pps) the register will fill in the required time interval for the attitude control subsystem to pitch turn the spacecraft the amount required for a given midcourse maneuver.
QC-1-2	Sets roll turn polarity and preloads the CC&S roll shift register so that at a counting rate of 1 pps the register will fill in the required time interval for the attitude control subsystem to roll turn the spacecraft the amount required for a given midcourse maneuver.
QC-1-3	Preloads the CC&S velocity shift register so that at a counting rate of 20 pps the register will fill in the time interval necessary for the midcourse motor to burn so that the spacecraft obtains the required velocity change for a given midcourse maneuver.

APPENDIX E

Mariner SPAC Group Encounter Test Plan

I. INTRODUCTION

During the preparation leading to the planetary encounter phase of the *Mariner IV* mission, it was recognized that the *Mariner* SPAC would have some special test requirements in order to adequately support the encounter. These requirements were enumerated, and a special SPAC Test Plan specifically designed to satisfy them was generated.

A. Purpose

The SPAC group test plan was prepared by the *Mariner* Mars 1964 SPAC Director and his staff, and was intended to serve as the basis for all future testing and test planning directly associated with the participation of the *Mariner* SPAC in *Mariner IV* flight operations. It was intended to be the current, basic description of the test philosophies, objectives, procedures, acceptance criteria, and methods of evaluation to be used in all SPAC tests.

B. Scope

The SPAC Test Plan described all Space Flight Operations Facility (SFOF) tests which required direct participation of the *Mariner* SPAC personnel or function, starting with the planetary encounter tests scheduled for April 1965 and ending with the termination of the *Mariner IV* mission.

C. Summary of Mariner SPAC Test Operations

1. General

The SPAC test operations were intended to exercise the engineering and science performance analysis personnel in the performance of their duties with the objective of developing and maintaining the proficiency required to participate properly in such critical and complex spaceflight operations as the planetary encounter and post-encounter phases of the *Mariner IV* mission. Inasmuch as the personnel involved had been actively engaged in the interpretation of spacecraft telemetry data in order to determine the state of the *Mariner IV* from the time of launch, it was not intended to test toward the objective of orientation and familiarization with spacecraft operations, but rather toward the objective of exercising all of the interfaces within the DSN which directly affect SPAC, of providing a thorough understanding of the spacecraft logic which had heretofore been unexercised during the flight, and of developing in detail the procedures that would be used during the actual flight operations.

2. SPAC Encounter Tests

The following summary outlines the SPAC tests that were to be performed in support of the planetary encounter activities.

a. Encounter sequence plan (ESP) tests. The objective of the ESP tests was to exercise SPAC in a nominal encounter sequence in order to allow the generation of ESP procedures in sufficient detail to permit their use in flight operations.

b. Backup mode tests. The backup mode tests were basically nominal encounters with induced failures or variations that required corrective action or the adoption of some alternate encounter mode.

c. Logic flow verification tests. The objective of logic flow verification tests was to establish and verify the telemetry criteria required to properly determine the logical state of the spacecraft and to aid in operational decisions.

d. Scan position test. The scan position test was a proficiency test for the science operations personnel involved in positioning the scan platform via ground command.

e. Scan search-to-planet tests. The objective of scan search-to-planet tests was to verify the procedures for determining the intersection of a searching scan platform with the limb of the planet.

f. Special tests. A number of special tests were to be conducted to test various phases of the encounter under special conditions. One test would involve pre-positioning the scan platform using data from Johannesburg to test both the procedure and the ability to use operational-quality data.

g. Readiness tests. Readiness tests were real-time encounter tests of the encounter phase procedures involving the bulk of the SFO organization with the objective of achieving and confirming a full operational readiness state prior to actual encounter.

II. TEST DESCRIPTION

A. ESP Tests

1. Objective

The objective of the ESP tests was to exercise SPAC and the operations areas with which it interfaces in the execution of the ESP encounter sequence to confirm the feasibility of the ESP as an operations plan, to aid in the development of any special equipment or techniques required to support SPAC and the operations personnel from SSAC, and to allow the generation of detailed ESP procedures for the actual flight operation.

2. General

The ESP tests were designed to exercise the analysis area operations personnel in all of the activities required in the standard ESP. In order to provide an adequately realistic and flexible data source, the *Mariner* Mars 1964 PTM spacecraft was to be used. Participation in the test operations was to include the PTM test team [located at the Spacecraft Assembly Facility (SAF)], SPAC, SSAC operations personnel, such SFO personnel as normally complete the operations link with the DSIF, and such data processing personnel as would be required to provide the projected ESP processing capability in SPAC and SSAC. The MMSA was to be maintained as closely as possible in the encounter configuration with all special equipment and display or output devices installed and operational.

3. Procedure

The ESP tests were to be controlled by the SPAC Director and to consist of a nominal encounter sequence including all ground commands scheduled as primary, or, if applicable, backup per the ESP. Since the ESP allowed for alternate modes in the event of nonstandard occurrences, it was permissible for SPAC to deviate per the ESP from the nominal sequence; however, there was no intention of scheduling induced failures or anomalies in the data source.

The tests could be performed in collapsed time in order to allow the completion of all test events within a standard 8-hr working day. The test events to be included would nominally be those in the ESP occurring between CC&S command MT-7 and the state of video playback after CC&S command MT-9, although the test might be scheduled to terminate as early as the end of the record sequence at the discretion of the test conductor.

4. Acceptance Criteria

The acceptance criteria for the ESP test were to be threefold:

1. The SPAC and SSAC personnel were to have attained a demonstrable proficiency in performing standard events required in the ESP.
2. The detailed procedure used was to have been verified to be adequate for use in nominal flight operations.
3. All equipment associated with the test was to have operated within tolerance and to have furnished adequate support to its respective users.

B. *Backup Mode Tests*

1. Objective

The backup mode tests were to test the capability of SPAC and the SSAC operations personnel to respond adequately to failures and/or anomalies occurring during the encounter phase of the *Mariner* mission.

2. General

The general framework of the backup mode tests was to be the same as for the ESP tests, with the exception that during the backup mode tests the PTM test team, at their discretion, might induce failures or anomalies during the performance of the ESP. It was SPAC's responsibility to determine the nature of these induced problems and to recommend the corrective action required.

3. Procedure

The backup mode tests were to be scheduled and controlled in the same manner as the ESP tests, and were to differ from them only insofar as the introduction of nonstandard conditions was concerned.

4. Acceptance Criteria

1. SPAC was to have shown itself to be proficient in recognizing spacecraft failures and in taking immediate, proper corrective action.
2. The emergency procedures generated for the encounter phase of the *Mariner* mission were to have been verified as being both correct and sufficient.

C. *Logic Diagram Verification Tests*

1. Objective

The logic diagram verification tests were to be used to establish and verify a set of telemetry criteria which determined the logical state of the spacecraft and which served as a basis for decision which concerned nonstandard ground command action and deviations from the nominal ESP.

2. General

The logic diagram verification tests were designed as a calibration rather than a simulation. The *Mariner* Mars 1964 PTM spacecraft was configured into one of the many logical states possible during the encounter and the telemetry indications were recorded. A change of state would then be induced, and the telemetry indications would again be noted. The change in telemetry should enable the SPAC personnel monitoring the test data to formulate a set of telemetry criteria which would determine as uniquely as possible the state of the spacecraft.

Participation in these tests would extend only to the data analysis and the SFO support necessary to operate the communications to SAF and the data output devices in the MMSA.

3. Procedure

All logic diagram verification tests were to be run in collapsed time without the simulated transmission lag normally employed in the SPAC testing program. No formal procedure would be generated for these tests; instead the PTM Test Director would respond directly to the test conductor (SPAC Director). The tests, in general, would be performed using the *Mariner* Mars 1964 flight sequence flow diagrams. The tests would be planned for 8 hr, and as many decision points on the diagrams as possible would be investigated.

4. Acceptance Criteria

A sufficiently detailed and accurate set of telemetry criteria was to have been generated and verified to allow the publication of a formal set of encounter telemetry criteria for use during the actual *Mariner IV* encounter.

D. Scan Position Test

1. Objective

The objective of the scan position test was to establish and maintain the proficiency of the SSAC operations personnel in properly positioning the scan platform under simulated encounter condition by means of ground command.

2. General

The scan position tests would be conducted jointly by the SPAC Director and the SSAC Operations Director. Participation would be required only from the SSAC operations personnel, the PTM test team, and such SFO personnel as would be required to furnish communications and data processing. There would be no SPAC participation in the scan position tests other than the SPAC Director function.

3. Procedure

The data source was to be the *Mariner* Mars 1964 PTM spacecraft, and encounter conditions were to be simulated as closely as possible. Encounter science should be turned on with CC&S command MT-7, and the scan platform should begin operating in the search mode. The SSAC operations personnel were responsible for determining the proper time for command action in order to stop the platform from searching at a predetermined position.

4. Acceptance Criterion

The SSAC operations personnel should have demonstrated their proficiency in positioning the scan platform with sufficient accuracy to assure a high degree of confidence in any possible scan positioning operation to be undertaken during the *Mariner IV* planetary encounter.

E. Scan Search-to-Planet Tests

1. Objective

The objective of this test was to develop and verify the procedures for determining the intersection of a searching scan platform with the limb of the planet during the *Mariner IV* encounter.

2. General

The scan search-to-planet tests were conducted by the SSAC Operations Director. Participation was the same as in the scan position test, except that trajectory information supplied by FPAC was required.

3. Procedure

The test procedure was to be the same as in the scan position test, with the exception that the simulated time scale of the test would extend to the acquisition of the planet by the television subsystem.

4. Acceptance Criterion

The SSAC operations personnel were to have demonstrated the capability to determine accurately the position and time of intersection of a searching scan platform with the limb of the planet.

F. Special Tests

1. Objective

The special tests were a series of tests which, for the most part, were generally designed to provide a more adequate simulation of the encounter conditions in the areas where sufficient confidence in the ability of SPAC and SSAC to perform reliably did not exist.

2. General

One special test which had been defined was a scan position test using a data tape played over the normal communications lines from DSIF 51. These communications channels had proved the weakest during the *Mariner* mission before encounter, yet they had to be relied upon for so critical a function as positioning the scan platform.

3. Procedure

The test was to be conducted per standard scan positioning procedures.

4. Acceptance Criterion

SSAC was to have demonstrated the ability to position the scan platform accurately using DSIF 51 as a data source.

G. *Readiness Tests*

1. Objective

The objective of the readiness tests was to provide verification of the operational readiness of the combined DSN prior to *Mariner IV* planetary encounter.

a. Preliminary readiness test. The objective of this test was to be to verify the adequacy of encounter procedures and to verify the smooth operation of all of the interfaces among the various components of the DSN.

b. Second readiness test. The objective of this test was to verify the operational status of all equipments, procedures, techniques, and programs required during the *Mariner IV* encounter.

c. Final readiness test. The objective of this test was to verify that the analysis areas and the DSN were fully prepared to commit to support the *Mariner IV* encounter.

2. General

The readiness tests were real time simulated encounter tests using the *Mariner* Mars 1964 PTM as a data source. The test conductor for the readiness tests should be the Space Flight Operations Director, and participation in the test should be required of all analysis areas and all operations areas and facilities. The test was to encompass all of the events from an arbitrary time prior to CC&S command MT-7 until approximately 20 lines of the second TV picture had been played back.

3. Procedure

The readiness tests were to be conducted per the ESP.

4. Acceptance Criterion

All of the participating areas and facilities should have demonstrated their complete operational readiness to such a degree that the *Mariner* Project Manager was prepared to accept their commitment to support the *Mariner IV* encounter.

APPENDIX F

Table F-1. Day conversion chart

	November 1964			December 1964																		
Date	28	29	30	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19
DAY OF YEAR	333	334	335	336	337	338	339	340	341	342	343	344	345	346	347	348	349	350	351	352	353	354
DAY OF MISSION	0	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21
	December 1964												January 1965									
Date	20	21	22	23	24	25	26	27	28	29	30	31	1	2	3	4	5	6	7	8	9	10
DAY OF YEAR	355	356	357	358	359	360	361	362	363	364	365	366	001	002	003	004	005	006	007	008	009	010
DAY OF MISSION	22	23	24	25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40	41	42	43
	January 1965																					Feb.
Date	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26	27	28	29	30	31	1
DAY OF YEAR	011	012	013	014	015	016	017	018	019	020	021	022	023	024	025	026	027	028	029	030	031	032
DAY OF MISSION	44	45	46	47	48	49	50	51	52	53	54	55	56	57	58	59	60	61	62	63	64	65
	February 1965																					
Date	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23
DAY OF YEAR	033	034	035	036	037	038	039	040	041	042	043	044	045	046	047	048	049	050	051	052	053	054
DAY OF MISSION	66	67	68	69	70	71	72	73	74	75	76	77	78	79	80	81	82	83	84	85	86	87
	February 1965					March 1965																
Date	24	25	26	27	28	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17
DAY OF YEAR	055	056	057	058	059	060	061	062	063	064	065	066	067	068	069	070	071	072	073	074	075	076
DAY OF MISSION	88	89	90	91	92	93	94	95	96	97	98	99	100	101	102	103	104	105	106	107	108	109
	March 1965														April 1965							
Date	18	19	20	21	22	23	24	25	26	27	28	29	30	31	1	2	3	4	5	6	7	8
DAY OF YEAR	077	078	079	080	081	082	083	084	085	086	087	088	089	090	091	092	093	094	095	096	097	098
DAY OF MISSION	110	111	112	113	114	115	116	117	118	119	120	121	122	123	124	125	126	127	128	129	130	131
	April 1965																					
Date	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26	27	28	29	30
DAY OF YEAR	099	100	101	102	103	104	105	106	107	108	109	110	111	112	113	114	115	116	117	118	119	120
DAY OF MISSION	132	133	134	135	136	137	138	139	140	141	142	143	144	145	146	147	148	149	150	151	152	153
	May 1965																					
Date	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22
DAY OF YEAR	121	122	123	124	125	126	127	128	129	130	131	132	133	134	135	136	137	138	139	140	141	142
DAY OF MISSION	154	155	156	157	158	159	160	161	162	163	164	165	166	167	168	169	170	171	172	173	174	175
	May 1965									June 1965												
Date	23	24	25	26	27	28	29	30	31	1	2	3	4	5	6	7	8	9	10	11	12	13
DAY OF YEAR	143	144	145	146	147	148	149	150	151	152	153	154	155	156	157	158	159	160	161	162	163	164
DAY OF MISSION	176	177	178	179	180	181	182	183	184	185	186	187	188	189	190	191	192	193	194	195	196	197

Table F-1. Day conversion chart (cont'd)

	June 1965																	July 1965				
Date	14	15	16	17	18	19	20	21	22	23	24	25	26	27	28	29	30	1	2	3	4	5
DAY OF YEAR	165	166	167	168	169	170	171	172	173	174	175	176	177	178	179	180	181	182	183	184	185	186
DAY OF MISSION	198	199	200	201	202	203	204	205	206	207	208	209	210	211	212	213	214	215	216	217	218	219

	July 1965																					
Date	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26	27
DAY OF YEAR	187	188	189	190	191	192	193	194	195	196	197	198	199	200	201	202	203	204	205	206	207	208
DAY OF MISSION	220	221	222	223	224	225	226	227	228	229	230	231	232	233	234	235	236	237	238	239	240	241

	July 1965				August 1965																	
Date	28	29	30	31	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18
DAY OF YEAR	209	210	211	212	213	214	215	216	217	218	219	220	221	222	223	224	225	226	227	228	229	230
DAY OF MISSION	242	243	244	245	246	247	248	249	250	251	252	253	254	255	256	257	258	259	260	261	262	263

	August 1965													September 1965								
Date	19	20	21	22	23	24	25	26	27	28	29	30	31	1	2	3	4	5	6	7	8	9
DAY OF YEAR	231	232	233	234	235	236	237	238	239	240	241	242	243	244	245	246	247	248	249	250	251	252
DAY OF MISSION	264	265	266	267	268	269	270	271	272	273	274	275	276	277	278	279	280	281	282	283	284	285

	September 1965																					Oct.
Date	10	11	12	13	14	15	16	17	18	19	20	21	22	23	24	25	26	27	28	29	30	1
DAY OF YEAR	253	254	255	256	257	258	259	260	261	262	263	264	265	266	267	268	269	270	271	272	273	274
DAY OF MISSION	286	287	288	289	290	291	292	293	294	295	296	297	298	299	300	301	302	303	304	305	306	307

	October 1965					
Date	2	3	4	5	6	7
DAY OF YEAR	275	276	277	278	279	280
DAY OF MISSION	308	309	310	311	312	313

APPENDIX G

Personnel

People who supported SPAC activities during its lifetime are listed by Division and function, in the hope that future SPAC groups may benefit from the knowledge and experience which each contributed individually to the *Mariner IV* mission.

I. DIVISION 29, PROJECT ENGINEERING

A. G. Conrad, *Mariner* SPAC Director

SPAC Direction Team

R. K. Case
R. F. Draper
R. F. Miles, Jr.
R. A. Neilson

SPAC Support

R. E. Enfeld
L. Holmes
J. B. Ingles
D. M. Neumann
F. C. Smith
P. H. Steinbrook

SAF Test Team

G. W. Earle
L. J. Fullmer
G. Ostheller
T. W. Shain
H. H. Weaver
V. A. Wirth, Jr.
W. R. Teague (Comprehensive Designers, Inc.)
H. W. Clary (Northrop Space Laboratories: NSL)
W. Davidson (NSL)
T. L. Edwards (NSL)
T. J. Laney (NSL)
K. V. Laverty (NSL)
R. T. Miller (NSL)

II. DIVISION 33, TELECOMMUNICATIONS

J. A. Hunter, SPAC Division Representative

Telecommunications Analysis

S. A. Aranyi
E. N. Gordon
J. A. Hunter
L. Nalaboff
J. S. Omahen
V. Z. Viskanta
J. Votaw

Data Encoder Subsystem

W. Allen
W. Apel
C. Carl
W. Johnson
W. D. Sheldon

Command Subsystem

L. Chandler
M. Crawford
R. De Santis
R. Duvall
D. Olsen
R. Rakunus
T. Spear

Radio Subsystem

T. Cocca
D. D. Dahms
C. Hill
M. Holmes

Video Storage Subsystem

D. Appleton
H. Bell
R. Grumm
J. Jurisich
W. Trester

Ground Telemetry Subsystem

R. Balluff
L. A. Couvillon
R. David
D. Drogosz
M. Graus
M. Kameya
R. P. King
R. Merritt
W. J. Rousey
P. Sandidge

III. DIVISION 34, GUIDANCE AND CONTROL

D. R. Thomas, SPAC Division Representative

Power

S. Abramowitz
D. Bowen
G. C. Cleven
K. M. Dawson
C. L. Edwards
C. D. Fredrickson
J. V. Goldsmith
A. Krug
D. R. Lancaster
W. L. Long
L. A. Packard
H. C. Primus
D. Prisbrey
W. K. Shubert
R. L. Toomath
R. C. Moy (Space Technology Labs, Inc: STL)
T. Rigas (STL)
A. M. Youngren (STL)

CC&S

P. Fogg
E. Greenberg
N. Herman
M. Hoppe
P. Lecoq
S. Lingon
D. D. Lord
C. Newman
R. Otamura
D. Ross
R. Smith
G. Wardwell
G. Murphy (STL)

Attitude Control

H. K. Bouvier
L. J. Brimmer
D. Carpenter
T. A. Casad
W. D. Charlan
W. E. Crawford
E. S. Davis
T. J. Donlin
P. J. Hand
G. E. Hooper
E. S. Ivie
B. Johnson
J. M. Kent
T. Kerner

W. K. Moore
J. F. Petralia
L. C. Pless
L. F. Schmidt
G. Shima
R. Cooper (Northrop Nortronics: NTX)
G. Gupta (NTX)

Division and General

J. D. Acord
R. Bardwell
R. P. Castro
R. E. Curry
R. V. Morris
D. H. Newell
G. Pace
L. D. Runkle
J. L. Savino
E. E. Suggs
S. Szirmay

IV. DIVISION 35, ENGINEERING MECHANICS

D. W. Lewis, J. D. Schmuecker — SPAC Division Representatives

Thermal Control

L. N. Dumas
D. C. Miller
T. O. Thostesen

Structures and Actuators

G. G. Coyle
E. L. Floyd
P. T. Lyman
R. P. Thompson
H. D. von Delden
J. N. Wilson

V. DIVISION 37, ENGINEERING FACILITIES

SFO/SAF Operations

L. S. Paul

VI. DIVISION 38, PROPULSION

T. A. Groudle, B. W. Schmitz — SPAC Division Representatives

Propulsion

H. G. Hartung
J. H. Kelley
R. F. Mattson
T. R. Metz

Pyrotechnics

J. E. Earnest
M. L. Moore

SFO/SAF Operations

M. E. Maine

VII. DIVISION 32, SPACE SCIENCES

W. G. Fawcett, SPAC Division Representative

Operations

W. G. Hodges, W. L. Momsen, L. G. Parker — alternate SPAC representatives
R. K. Sloan, *Mariner* SSAC Director

The following Division 32 personnel were SSAC members who were called upon for direct support to the EPWG and to the SPAC operations.

Data Automation Subsystem

W. Baumer
A. P. Calabrese
D. L. Nay
W. J. Schneider

Cosmic Ray Telescope

R. J. Holman

Trapped Radiation Detectors

R. Lockhart
D. K. Schofield

Cosmic Dust Detector

B. V. Connor
D. K. Schofield

Ion Chamber

H. R. Anderson
L. G. Despain

Scan Subsystem

R. F. Denning
R. Y. Wong

Plasma Probe

C. W. Snyder

Magnetometer

D. D. Norris
E. J. Smith

Television Subsystem

J. D. Allen
D. E. Bell
A. G. Herriman
C. C. LaBaw

Science Data

S. Foster
S. Gunter
L. Jahr
M. Sander
A. Snyman

SFO/SAF Operations

J. H. Lucks
F. A. Tomey

VIII. DIVISION 31, SYSTEMS

SPAC Operations Assistants

D. Backofen
M. D. Johnson
L. Morgan
T. M. Taylor
D. Tustin

SPAC Data Handling

J. K. Hickey
F. House

FPAC Liaison

N. R. Haynes, *Mariner* FPAC Director

H. J. Gordon

D. A. Tito

Programmer Support

H. W. Alcorn

D. Casperson

T. Cox

D. Jacobs

E. Oliver

W. Paul

L. Pyle

J. Witt

NOMENCLATURE

ac	alternating current
A/D	analog-to-digital
ADC	analog-to-digital converter
AFETR	Air Force Eastern Test Range
AGC	automatic gain control
APC	automatic phase control
A/PW	analog-to-pulse width converter
AU	astronomic unit
bps	bits per second
CC&S	central computer and sequencer
cg	center of gravity
ccw	counterclockwise
cw	clockwise
CY-1	CC&S cycle command
DAS	data automation subsystem
DAP	SFO computer program
DC	direct command
dc	direct current
DIS	data input system
DN	data number
DPS	data processing system
DSIF	Deep Space Instrumentation Facility
DSN	Deep Space Network
ev	electron volts
EOT	end of tape
EPWG	Encounter Planning Working Group
FPAC	Flight Path Analysis and Command Group
F/PR	Failure/Problem Report
G1	television subsystem grid number 1 voltage
GM	Geiger-Mueller (tube)
GTA	Ground Telemetry Analyst
GTS	ground telemetry subsystem
IR	infrared
LCE	launch complex equipment
L-1 to -3	CC&S launch counter commands
MDL	Master Data Library
MMSA	Mariner Mission Support Area
MT-1 to -9	CC&S master-timer commands
NAA	narrow-angle acquisition
NAMG	narrow-angle Mars gate
NASA	National Aeronautics and Space Administration
NRT	nonreal time
OSE	operational support equipment
PAM	pulse-amplitude modulation
PAS	pyrotechnics arming switch
PCA	pyrotechnics control assembly
PCM	pulse code modulation
P/FR	Problem/Failure Report
PHA	pulse-height analysis
PIPS	postinjection propulsion subsystem
PLL	phase-lock loop
PN	pseudonoise
PNG	pseudonoise generator
PS&L	power switching and logic
PSK	phase-shift keyed
PTM	proof test model
QC	quantitative command
RF	radio frequency
RFP	Request for Programming
RT	real time
RTDT	real time data translator

NOMENCLATURE (Cont'd)

SCR silicon-controlled rectifier
SFO Space Flight Operations
SFOD Space Flight Operations Director
SFOF Space Flight Operations Facility
SIT separation-initiated timer
SNORE signal-to-noise ratio estimator
SPAA Spacecraft Performance Analysis Area
SPAC Spacecraft Performance Analysis and Command Group
SPE static phase error
SSAC Space Science Analysis Command Group
SSDM SFO computer program
STC system test complex
TDA Tracking Data Analyst
TTY teletype
TWT traveling-wave tube
UV ultraviolet
VCO voltage-controlled oscillator
VSWR voltage standing-wave ratio
WAA wide-angle acquisition