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APOLLO EXPERIENCE REPORT - DEVELOPMENT FLIGHT INSTRUMENTATION

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16. Abstract

Development flight instrumentation was delivered for 25 Apollo vehicles as Government-furnished equipment. The problems and philosophies of an activity that was concerned with supplying telemetry equipment to a space-flight test program are discussed. Equipment delivery dates, system-design details, and flight-performance information for each mission also are included.

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SUMMARY

In this report, the development flight instrumentation systems that were supplied as Government-furnished equipment for 25 Apollo vehicles are discussed. These systems, which were used to flight-test and qualify the earlier Apollo vehicles, were designed, integrated, and tested at the NASA Lyndon B. Johnson Space Center (formerly the Manned Spacecraft Center) and shipped to the prime contractors or launch sites for vehicle installation. The 8-year activity supported several types of missions varying from the first pad-abort launch (boilerplate 6) from White Sands Missile Range to the Apollo 13 lunar flight from the NASA John F. Kennedy Space Center. During this time, techniques were developed to provide for flexible and reliable systems. These techniques involved primarily the use of relatively low-cost off-the-shelf equipment and an accompanying qualification program that ensured the necessary degree of reliability for the flight test program.

Component qualification and acceptance testing was devised to place emphasis on the environmental testing of all flight components. This testing was considered a major factor in obtaining the overall 98 percent successful data retrieval for the Apollo development flight instrumentation systems. Components were standardized between vehicles because the development flight instrumentation used a "building block" approach that easily accommodated any variation in quantity or types of measurements. A maximum degree of standardization was also used for component input/output characteristics, common component test levels, component range selection lists and test procedures, and so forth.

The significant problems encountered in the component areas involved electromechanical devices. Tape recorders were of particular concern and required considerable attention to achieve improved reliability. Some of the most serious problems occurred with failures in the vehicle wiring harnesses, which affected the integrity of the development flight instrumentation systems. One lesson that was learned was the importance of performing a proper system installation design within the vehicle. Poor component accessibility and high-density wiring harness design were major factors in system failures.

INTRODUCTION

In late 1961, the NASA Manned Spacecraft Center (MSC) (now the Lyndon B. Johnson Space Center) began an activity to provide the development flight instrumentation (DFI) for the Apollo vehicles. The instrumentation involved in this activity was primarily used to provide data during the flight testing phases of the operational subsystems and consisted of transducers, amplifiers, modulation packages, tape recorders, transmitters, and so forth. The systems were flexible (variable in capacity) and could accommodate high-bandwidth analytical parameters; furthermore, the systems were designed to be phased out of the Apollo Program as the vehicles became operational. The DFI equipment was generally dispersed over all parts of the vehicle and frequently occupied locations that were destined later to be taken by the operational systems. Transducers were located in such positions as the inside of rocket thrust chambers, the outside skin of service modules, the pipes on hydraulic systems, and the surface of structural members. The larger elements of the system (modulation packages, tape recorders, etc.) were generally centrally sited on shelves (if available) for accessibility. In contrast to the DFI, the "operational" instrumentation system was designed as a permanent entity and was later phased into the Apollo Program to monitor the vehicle operational "housekeeping" parameters. The operational instrumentation was essentially a low-bandwidth system and was not concerned with analytical-type measurements that required high frequency response rates. Although the original intention was to provide the DFI systems for the developmental part of the Apollo Program, the equipment was used throughout the operational phases, including the early lunar missions. All DFI systems were supplied as Government-furnished equipment (GFE) from MSC. A DFI group managed the MSC in-house activities for the fabrication and delivery of the systems. These activities involved system design, component procurement, component qualification, system integration, and so forth.

In all, 25 flight systems were designed and constructed. Eighteen of these were actually flown on missions; the remaining seven, because of program changes, were used in ground test vehicles or were reassigned for use as spares. Additional equipment (not full systems) was flown on several other vehicles. The first flight system was used on the boilerplate (BP) 6 pad-abort mission at White Sands Missile Range (WSMR) in November 1963. The DFI was supplied to the prime contractors for installation in the flight vehicles. A total of 8000 replaceable units (transducers, amplifiers, transmitters, etc.) was qualified and supplied as DFI to support the Apollo Program. Some of the hardware was designed and fabricated at MSC; the majority, however, was obtained under contract with 45 equipment vendors.

The DFI performed well throughout the flight program: no major equipment failures occurred and the overall average data retrieval was 98 percent. A secondary benefit derived from the DFI activity was the engineering training of personnel in spacecraft equipment. Engineers received a practical indoctrination in the design and operation of various equipment and in the development of general testing requirements and specifications. The gained knowledge, especially the general testing requirements and specifications, became an asset to the Apollo Program and helped in the preparation of the operational equipment. Later, DFI specifications were reflected in the operational specifications and procedures.

The first DFI requirement originated from aerodynamic and structural measurements on Apollo/Saturn missions 101 and 102, which carried both service modules and boilerplate command modules. The initial system design was predicated on large quantities of measurements using radio-frequency (rf) transmission as the means of data retrieval (nonrecoverable mission). Development of the instrumentation design was based on systems with approximately 1200 data channels and 7 transmitters; however, the initial requirement for DFI on these vehicles was postponed because of the BP-6 mission at WSMR. The objectives of that test were to demonstrate the launch escape system and to test various parts of the boilerplate; 94 measurements were made with the DFI system on the BP-6 vehicle. The system performed satisfactorily and provided confidence to continue with the basic block design for future Apollo DFI systems. Flight performance information for each mission is given in appendixes A to D.

INITIAL PROJECT CONSIDERATIONS

The decision to supply the DFI as GFE was made for two prime reasons: to permit NASA to maintain schedule and system flexibility during the flight test program and to relieve the prime contractor workload to allow greater concentration on the operational systems. During the initial system design, the fact became apparent that meeting schedule deadlines would preclude excessive component development and require that systems be designed to use reliable off-the-shelf equipment.

To determine the best selection of telemetry hardware, considerable research was done with the NASA George C. Marshall Space Flight Center, the NASA Langley Research Center, and other institutions. The resulting information revealed that more research and component evaluation would be necessary in the general areas of low-level commutation, signal conditioning, and frequency modulation (FM). Because of the extent of the projected activity, the decision was made to begin on a long-term design with a building-block approach. In the beginning, neither weight nor size was a problem because the early boilerplate vehicles were not equipped with operational equipment. However, as the activity continued, weight and size became increasingly important as operational equipment was introduced and less room was left for the DFI. The later requirement for DFI in the lunar module (LM) necessitated a weight and size reduction of the individual components. A comparison of the resultant reduction in DFI component characteristics between the command and service module (CSM) and the LM is indicated by the following statistics.

1. Average LM component volume = 20 percent of CSM counterpart

2. Average LM component weight = 32 percent of CSM counterpart

3. Average LM component power = 27 percent of CSM counterpart

The DFI was not subject, as was the operational equipment, to the more stringent reliability and quality assurance (R&QA) control measures at vendor plants. To remove most reliability risk factors associated with off-the-shelf equipment, applicable R&QA standards were adopted and a qualification and test program was performed at MSC on all components. Considerable emphasis was placed on acceptance testing to reveal fabrication and quality control weaknesses in prospective flight hardware. Most of the DFI ground-support equipment used by the prime contractors was supplied by MSC. This was the most economical approach because it ensured a maximum standardization of test equipment and test procedures for the project.

One significant factor in ensuring the integrity of the DFI systems was the adoption of the breadboard integration tests. In these predelivery tests, the actual flight components of a particular system were strapped to a large, metal-covered breadboard. The entire system was energized, checked, and subjected to electromagnetic-interference conditions. The cabling on the breadboard was similar to that used on the flight vehicle. The test method ensured, as far as possible, that the system would work after it was delivered and that it also would function within the electromagnetic-interference environment inside the vehicle. Both the system and the breadboard were shipped to the prime contractor so that the system could be checked and accepted before installation. Furthermore, if problems occurred in the vehicle, the contractor could use the breadboard for additional system checking.

Project engineers were assigned the responsibility for specific vehicle systems. For instance, a project engineer at MSC would be responsible for a system and the testing of that system. He would then monitor and approve the installation and testing performed at the prime contractor facility, ultimately following through to the NASA John F. Kennedy Space Center or WSMR and the launch. For backup, resident teams of DFI-trained contractor personnel were assigned by MSC to provide daily liaison with Houston, to implement modifications, and generally to expedite the solutions to system problems.

DESIGN CONSIDERATIONS

In the general system design, certain salient items helped to standardize the approach and to provide a sound basis for the systems. One such item and one of the first to be decided on was the operating signal voltage for subcarrier oscillators, amplifiers, commutators, and transducers. A cross section of the various transducer types that would have had to be used could have produced output voltages varying from millivolts to volts in combination with various polarities and bias offsets. Although subcarrier oscillators that accommodated these varying sensitivities and conditions of signals could be acquired, the decision was made to standardize all subcarrier oscillators at a signal input of 0 to 5 volts. Therefore, the amplifier or transducer would have to supply 0 to 5 volts regardless of the polarity or input swing of the parameter. Apparently, in many cases, tailoring the subcarrier oscillators to suit the varying parameter conditions would have been easier. However, considering that each subcarrier oscillator within a given package was of a different frequency, the fact became readily apparent that an additional variable, the subcarrier oscillator input voltage, would compound the problems of spares and interchangeability.

Consideration then was given to the low-level part of the system in which commutation apparently would be most economically advantageous because of the required low frequency response and the profusion of measurements (thermal surveys, strain measurements, etc.). The original systems were designed in accordance with a

10-millivolt signal input standard. No range from any of the low-level devices was expected to give less than this value, considering any possible dynamic range of the forcing functions. Previous experience with noise problems at these levels indicated that the systems should be designed with balanced signal-conditioning networks having output impedances of 100 ohms. The fact that the required low-level data were generally less than 10 hertz permitted filtering in the frequency spectrum above which most electrical interference occurred. This approach permitted fairly long line lengths (approximately 40 feet) for remote sensor locations and kept noise levels to approximately 1 percent of full scale. Low-level transducers requiring high frequency response were fed directly into high-gain direct-current amplifiers and converted to a high level (0 to 5 volts) for insertion into the subcarrier oscillators. The output and input impedances of amplifiers and conversion devices were designed to be approximately 1000 and 50 000 ohms, respectively. High-level transducer output impedances were generally designed to be a maximum of 1000 ohms. By establishing fairly rigid rules on system impedances and voltages, component engineers (who had to deal with 45 vendors) were provided with fairly specific standards for procurement and interface specifications.

Before the individual design of many components could be established, the matter of automatic calibration had to be decided. Automatic calibration was considered a desirable system feature, providing it did not compromise the system in cost and reliability. One of the benefits to be derived from this approach is that it provides a fast and easy way to check a system before launch and in flight. However, many valid arguments exist for not using such a method. In many cases, calibration functions do not exercise telemetry channels completely from end to end; therefore, the system is only partly checked. Many high-level transducers were not equipped to accommodate automatic calibrations and would have had to be specially modified. This modification would have destroyed the integrity already built into off-the-shelf components. Certain transducers (resistance thermometers, thermocouples, strain gages, etc.) can be checked effectively and easily by minor circuit additions into signal-conditioning networks. Others, such as variable-reluctance pressure transducers and crystal accelerometers, cannot be checked as easily and are scarcely worthy of the attempt because of the complexity involved. Also, when an ambient condition (temperature, pressure, etc.) falls within the dynamic range of a sensor, a good channel-verification point is provided. Furthermore, certain units are so reliable that to check them is probably superfluous. It also must be remembered that when a system is commanded into a calibration mode, the system must necessarily switch from the true data position. Therefore, the risk of inflight malfunction exists; the system may either shift or remain in the calibration position and return only calibration information instead of data. Although there were many reasons for not using such a feature, the convenience of a fast built-in test arrangement overrode the objections and an automatic calibration system was designed into the DFI. This system included the transducer elements, where practical, and all signal amplifiers except vibration-charge amplifiers. The subcarrier oscillators also were switched into either five- or two-step calibration levels, and the commutators had fixed-reference voltages impressed on certain channels. This general philosophy of system calibration was followed throughout the DFI activity because it provided a quick check for faultfinding and reduced unnecessary probing into already functioning system areas.

The DFI system design was not founded on a redundant concept as were the later operational CSM and LM systems. At the time, redundancy, in the true sense of duplication, seemed to be extremely complex and expensive; furthermore, the DFI equipment was not subject to the stringent mission-essential requirements that governed the later operational equipment. Certain aspects of the design compensated somewhat for the lack of redundancy. Where practical, most units had internally regulated power supplies, which, in effect, made them independent of any commonly regulated supply (excluding the prime battery) and thus free from most single-point power failures. Units designed with this characteristic consisted of high-level transducers, signal amplifiers/converters, subcarrier oscillators, mixer amplifiers, transmitters, and, to some extent, low-level conditioning units. This concept also made possible the use of isolated power-to-signal lines for most of the front-end instrumentation, thus assisting the overall design to achieve a single-point grounding system with a minimum of potential loops.

Another feature that helped to compensate for no redundancy was the cross loading of batches of measurements to different modulation packages and transmitters. In other words, if a given area of the spacecraft was subjected to a specific survey of measurements, where practical, the measurements would be distributed across various rf links. Thus, in the case of the partial or entire loss of a link, a percentage of the pertinent measurements would still be preserved.

The central part of the system was the signal-conditioning unit, which channeled most of the signals. This unit served as a central checking point and was capable of quick access for channel and range changes when the system was already installed in the vehicle. Furthermore, this unit permitted programed inflight time sharing for the purpose of maximum use of high frequency channels. This unit and the central power unit contained all the system fusing, which was used in both single and grouped component powerlines. Fusing arrays were optimally designed into the DFI system because a high possibility of system damage existed during a mission. The equipment was to be used in the initial proving stages of the Apollo Program, during which high stresses and violent maneuvers were imposed on the vehicles with some possible catastrophic effects. One such case was BP-12, which was launched from WSMR. Parts of the service module rammed the bottom of the CSM boilerplate and severed some DFI cables feeding several pressure transducers. Some fuses were blown in the affected system area, but the short-circuiting effect on the powerlines was not transferred to the remainder of the system, which continued to function.

Components of the system were designed and specified to include polarity-reversal circuitry with one exception: the power and control unit, through which all prime power was fed and switched. This overall precautionary design concept saved schedules and equipment because many of the systems were subjected to inadvertent power-polarity reversal during testing.

The overall selection of the DFI system, regarding modulation transmission and tracking, was controlled by the existing telemetry ground stations and radars in the Manned Space Flight Network (MSFN) and at WSMR. The requirements for instrumentation called for wide-band multichannel systems capable of catering to data rates from 2 kilohertz to steady states. The choice was a pulse amplitude modulation (PAM)/pulse duration modulation (PDM)/FM/FM system; the PDM part was used exclusively for multiplex tape recording. Briefly, this part of the system conformed to the following design parameters.

1. Commutation

   a. 90 channels × 10 samples/sec (PAM/PDM)

   b. 90 channels × 1-1/4 samples/sec (PAM/PDM)

   c. 3 channels × 0.8 sample/sec (PAM) special high frequency data commutator (2 kilohertz)

2. Modulation

   a. Proportional bandwidth (PBW) systems interrange instrumentation group (IRIG) channels 2 to 16 and channel E

   b. Constant bandwidth (CBW) systems Aerospace Industries Association channels 1c to 10c

3. Transmission: Very high frequency (vhf) 228.2 to 257.3 megahertz (FM)

4. Tracking

   a. C-band transponders

   b. 5690 to 5765 megahertz

5. Recording

   a. 14 tracks on 1-inch tape at 15 in/sec

   b. Commutated data — 90 × 10 PDM and 90 × 1-1/4 PDM

   c. Continuous data — PBW channels 2 to 16

   d. Wide-band data — FM record 14.5 kilohertz (2-kilohertz data)

The low-speed commutation rate of 90 channels × 1-1/4 samples/sec was previously used in Project Mercury and, although suitable for many quasi-static parameters, was not considered fast enough to resolve discrete events adequately for the Apollo Program; hence, a rate of 90 channels × 10 samples/sec was additionally adopted. The commutators that had a rate of 3 channels × 0.8 sample/sec were specially developed to time-share high frequency data (as much as 2 kilohertz) effectively on single continuous channels. Various makes and sizes of PBW modulation packages, the larger ones containing 16 channels, were used throughout the DFI project. Channel E was generally used for the loading of high-rate commutators. The CBW packages, which were used in the LM phase of the project, were introduced to accommodate the large amount of high frequency data and to reduce the number of rf links. These packages could absorb as many as 10 channels of 2-kilohertz information and were, in fact, the first CBW systems ever flown. The systems, although having relatively large capacities for high frequency data, had certain drawbacks. First, the information loading on the subcarriers resulted in a modulation index (MI) of 1 (2-kilohertz data).[1] Second, because of the relatively high number of channels occurring at the high end of the spectrum (compared to PBW IRIG), the transmitter MI per channel was down to 0.3, the overall permissible vhf transmitter deviation being 128 kilohertz (ground-receiver bandwidth limitations).[2] This loading, with its resultant low values of modulation index, gave reduced FM improvement with consequently lower signal-to-noise ratios than the PBW systems. Altogether, the systems were very useful as convenient large-capacity tools to obviate the need for additional rf links but had to be used within the sense of their limitations.

During the boilerplate stages of the DFI activity, 12-watt tube-type transmitters were used. Later, these transmitters were superseded by 5-watt and then by 10-watt solid-state units as the DFI equipment was phased into the CSM and LM vehicles. Early missions generally conformed to short-range ballistic or 100-nautical-mile-altitude Earth-orbital paths; in these cases, the radiated power was more than sufficient to give high positive circuit margins to MSFN and WSMR. From this viewpoint, the rf links appear to be conservatively designed. With the inception of the LM Earth-orbital vehicles, which used as many as five 10-watt DFI transmitters, the systems were designed to meet a worst-case 3-decibel margin in conjunction with the MSFN helical vhf antennas at a required slant range of 1200 nautical miles. Data-channel noise under these conditions generally conformed to less than 2 percent of full scale for the PBW systems. However, the CBW systems, for the reasons previously discussed, gave better performance near the stations and assumed channel-noise levels of approximately 3.5 percent full scale for slant ranges of 600 nautical miles.

FLIGHT SYSTEM BREADBOARDING

The block diagram of the basic DFI systems (fig. 1) illustrates a design approach. The only differences in the various systems are in the number and types of measurements made and in the number of pieces of telemetry equipment (commutators, modulation packages, and transmitters) used to accommodate these measurements. The photographic cameras and onboard tape recorders that were used on the CSM boilerplate and production vehicles, however, are not shown in figure 1.

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[1] It is preferable to use an MI value of 2 (or greater) in this part of the system, since lower values produce poorer signal-to-noise ratios; $\mathrm{MI} = \frac{\text{carrier deviation}}{\text{modulation frequency}}$.

[2] A higher value of MI (say 1) would be preferable at this part of the system to permit a better signal-to-noise ratio. (See also footnote 1.)

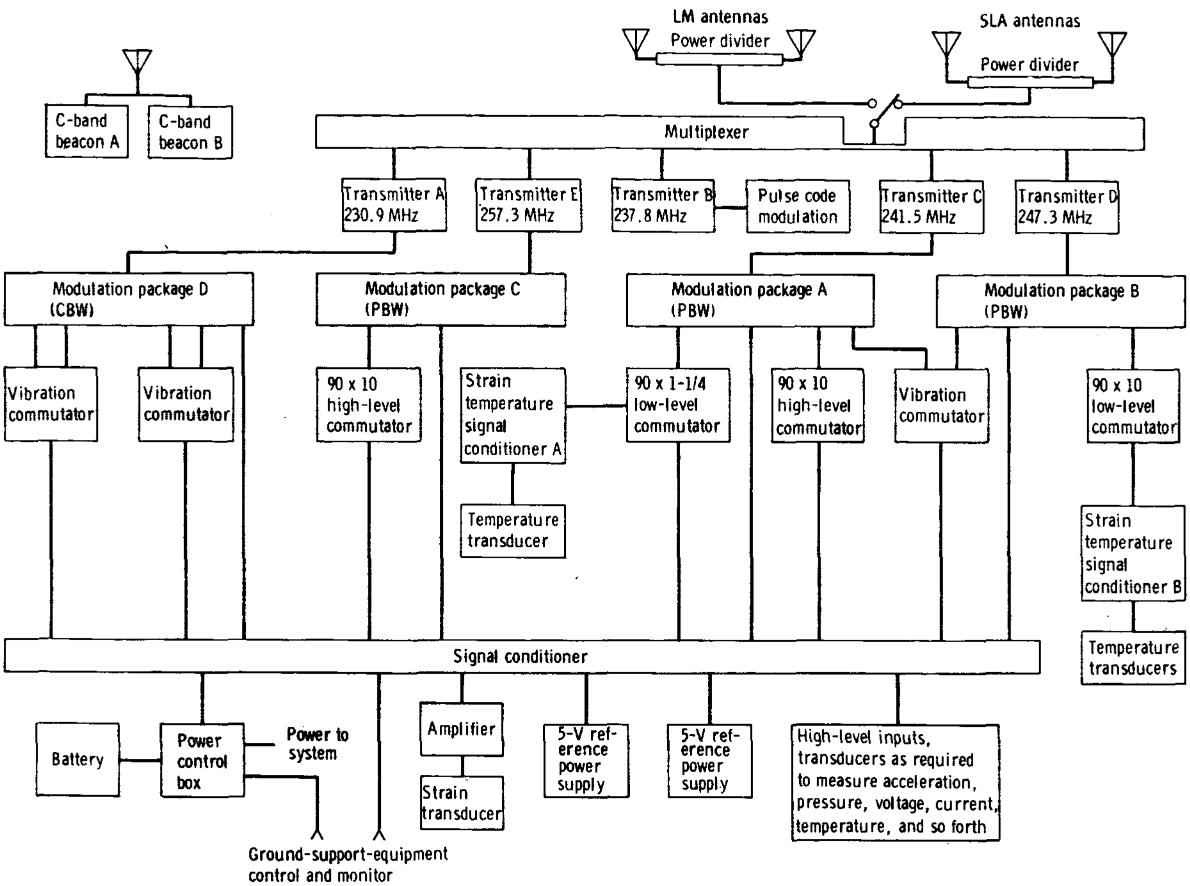


Figure 1.- The DFI system block diagram.

After acceptance testing, each component was mounted on a breadboard system (fig. 2). The breadboard consisted of a movable A-frame-shaped structure on which the equipment could be easily mounted and electrically interconnected. The breadboard proved to be a convenient, useful, and necessary tool for verifying an entire system operation. The breadboard also served as a handy shipping container.

After breadboard drawings had been prepared, the flight-qualified components were installed on the breadboard. The components were interconnected according to the schematics and prepared for system tests. The DFI test console (fig. 3) was developed to verify the operation of the system and components and to isolate system wiring malfunctions and other operating problems. A typical DFI subsystem installation in an LM is seen in figure 4. (This particular example is from LM-2.)

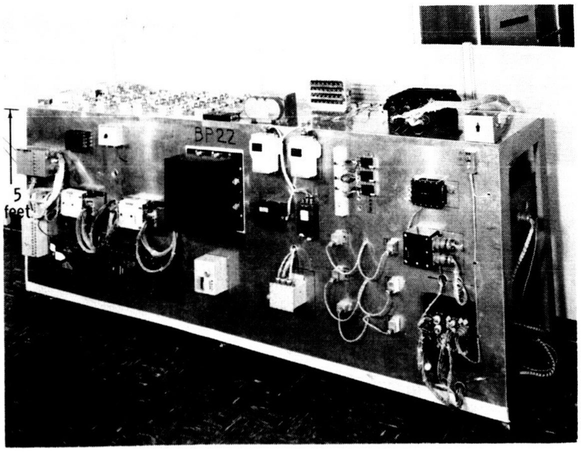


Figure 2. - A typical breadboard system.

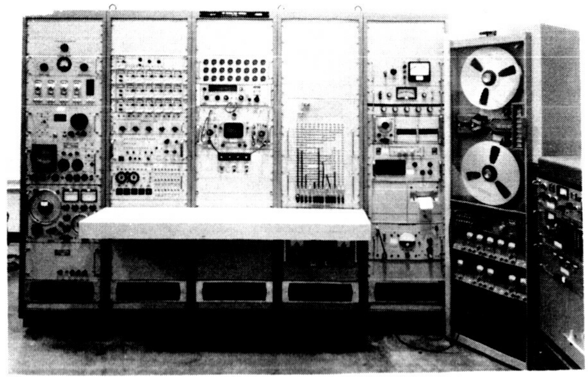


Figure 3. - The DFI test console.

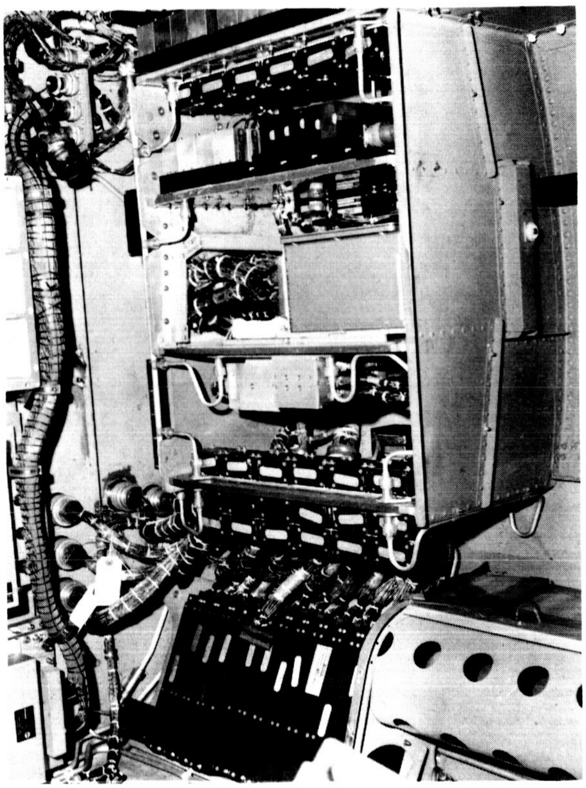


Figure 4. - The DFI system installed in LM-2.

ACCEPTANCE AND QUALIFICATION TESTING

As early as 1961, a general plan of component qualification was outlined for the DFI. Although at that time the main concern was selecting and evaluating certain components, the ground rule was established that the overall project would include environmental testing for acceptance as well as for qualification. The DFI was based on off-the-shelf equipment, and, to detect manufacturing defects and provide confidence for flightworthiness, the decision was made to subject each flight unit to the most critical flight environments. The selected environments were vibration, temperature, vacuum, and acceleration. Each unit also was bench tested to electrical specifications. This mode of acceptance testing was pursued throughout the DFI activity and was considered to be the major contributor in ensuring reliable flight component operation.

The qualification testing for each component type that was evolved contained 12 separate environmental tests. However, the number of tests varied slightly throughout the activity. The peak test levels were founded on values above the maximum design limits; that is, any level expected in any vehicle area that might contain DFI equipment. This was a major difference between the DFI and operational qualification philosophy. Whereas most DFI components were qualified at one level, the operational equipment was tailored for specific environmental zones within the vehicles. The standardized concept was used on the DFI to ensure that most equipment could be used and installed in any part of a vehicle without incurring different or additional qualification. Use of this concept not only permitted general mounting flexibility but also simplified procedures, procurement, and paperwork.

In a few cases, components (generally transducers) were exposed to high-intensity environments. Examples of this exposure occurred in the service modules, where components were mounted on thin shelving and were subjected to high vibrational g-forces. Other components were mounted through the service module skin where both high acoustic noise and vibration levels were imposed. To cater to these types of conditions, a part of the general qualification document was written to include a section for "space-exposed and/or extreme conditions." This section allowed the imposition of especially high levels where applicable. The use or partial use of the latter procedure was more the exception than the rule.

An example of the difference of levels between the general and extreme qualifications is shown by a comparison of the vibration specifications. The former required 12.3g; the latter required 45g. In general, the extreme qualification section was a "catchall" that satisfied all high environmental requirements and, at the same time, standardized the qualification testing.

COMPONENT FAILURES

The failure rates and problem areas associated with equipment after delivery to the prime contractor are shown in table I. The survey involves the preinstallation and postinstallation component failures on seven LM-type vehicles. The following facts were derived from the information in table I.

TABLE I. - DFI EQUIPMENT FAILURES AFTER DELIVERY TO PRIME CONTRACTOR[a]

[Failures reported for LTA-1,[b] LTA-2R, LTA-8, LTA-10R, LM-1, LM-2, and LM-3]

Phase	Equipment failures						Induced failures		
	Major			Minor					
	Design	Fabrication	Parts	Design	Fabrication	Parts	External design	Personnel	Miscellaneous
Postinstallation	3	1	1	--	--	--	12	42	17
Preinstallation	4	4	3	2	7	20	1	29	5
Total	16			29			106		

[a]An additional 43 failures reported were determined to be incorrect (i.e., no failures occurred).

[b]LM test article (LTA) 1.

Most of the failures were induced by some external means into the DFI. Of the 106 induced failures, most (71) were caused by personnel action; that is, handling, installation, and so forth. It is also interesting to note in the induced-failure area that postinstallation failures outweigh preinstallation failures by approximately 2 to 1; this ratio indicates that most component damage occurred after vehicle installation.

Of a total of 45 equipment failures, 16 were major. The remaining 29 were minor and, although attributable to various causes, were primarily out-of-tolerance calibration conditions. Failures of this kind would warrant replacement only if replacement was convenient. The 43 incorrectly reported DFI failures (table I footnote (a)) are the inevitable results of premature diagnosis and preliminary test investigations.

In summary, the information in table I indicates that the highest failure rate on the equipment was caused by handling. There are approximately 2.4 induced failures for every 1 equipment failure. Some of the problems were prompted by inaccessibility, by vehicle cabling design, and by the fact that some equipment was within the personnel "traffic" flow through the vehicle. Also, the fact that only four major failures occurred because of component parts defects (resistors, capacitors, etc.) is significant. The DFI used mature hardware; hence, most of the poorer component parts were already screened from the equipment. This factor, together with the additional screening effected by the acceptance testing at MSC, reduced the parts problems to minor proportions. Although involving a different prime contractor, the trends displayed were similar for the DFI systems supplied for the boilerplate vehicles and the command and service modules.

Some actual flight failures are listed in table II, which contains information from a survey including 10 DFI flight systems on vehicles BP-6 to spacecraft 011 (SC-011). These vehicles afforded a better opportunity to investigate flight failures than the LM vehicles because most of the systems were recovered and postflight analysis was possible. From the information in table II, it is apparent that the causes and types of failures covered are varied. The number of static surface-pressure failures (8) seems relatively high; but this can be explained on the basis that this area sometimes included more than 50 percent of all system measurements. However, vibration and acoustic failures did prompt a revision of qualification levels to accommodate extreme environments as mentioned previously. The relatively high number of wiring failures (10) accounts for nearly 50 percent of all failures and stresses the criticality of this area.

TABLE II. - DFI FAILURES IN FLIGHT[a]

(a) Measurement failures

Measurement type	Number of failures
Pressure (motor combustion)	1
Pressure (static surface)	8
Pressure (fluctuating surface)	3
Calorimeter (surface)	3
Thermocouple (surface)	1
Vibration (surface)	3
Strain (beam)	3
Acoustic (interior)	1
Total	23

(b) Causes

Probable cause of failure	Number of failures
Vibration	4
Constricted pressure lines	3
Acoustics	3
Wiring (short or open circuit)	10
Unknown	3
Total	23

[a]A total of 1104 measurements was made; 9 complete and 14 partial measurement failures occurred. Failures were reported for BP-6, BP-12, BP-13, BP-15, BP-22, BP-23, BP-23A, SC-002, SC-009, and SC-011.

SPECIAL SYSTEMS

One interesting part of the program was the supplying of some special GFE instrumentation systems. These systems were small compared with the average DFI vehicle systems, but each one had to be supplied on a compressed schedule. Some were identified as being required 2 or 3 months before the vehicle launch time. In fact, the vehicle was frequently on the launch pad when the initial requirement arose.

One such camera system, which was installed on CSM-020, generated stereographic mapping photographs across the width of the United States; it additionally provided photographs of the plasma phenomenon during reentry. The entire system was self-contained because it was not to interfere with, or impact in any way, the operational and DFI equipment already on board the spacecraft. The equipment consisted of two cameras, a timing and delay unit, a battery, and an accelerometer start mechanism that was triggered by the ascent g-force of the vehicle. The system was designed, fabricated, and qualified within a 3-month period. The system was used successfully and provided, for the first time, a set of overlapping stereographic photographs across the United States. One of the photographs is shown in figure 5.

Another camera system, supplied on a previous spacecraft (CSM-017), produced a sequence of color pictures of the Earth from a 10 000-nautical-mile altitude. These pictures, which were the first taken of the Earth from that distance, were published worldwide. An example is shown in figure 6.

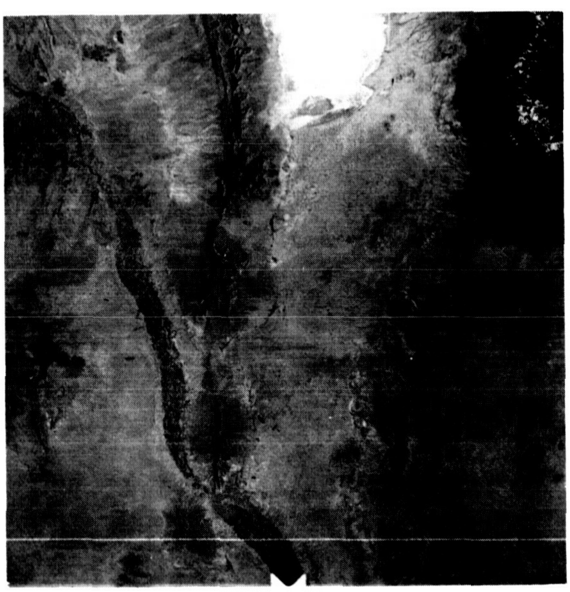


Figure 5. - A photograph of White Sands, New Mexico; one of 370 Earth-mapping pictures taken during the Apollo 6 mission.

Figure 6. - The first photograph of the Earth taken from an altitude of 10 000 nautical miles.

SCHEDULES AND MANPOWER LEVELS

The manpower peak for the DFI activity was reached between December 1965 and June 1966. During that time, the NASA and contractor support was approximately 40 and 125 men, respectively. This peak support was prompted by the requirement to deliver seven LM systems within an 8-month period. Curves representing the manpower applied to the DFI program are shown in figure 7. All DFI systems were delivered to the contractors on schedule. A list of the delivery dates is included in table III.

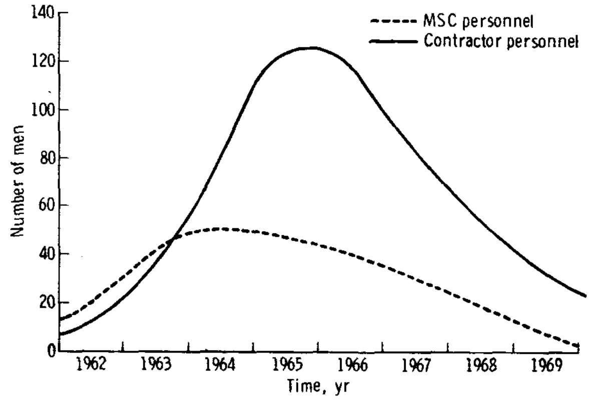


Figure 7. - Variation of DFI manpower levels with time.

TABLE III. - APOLLO DFI PROGRAM MILESTONES

Vehicle	Apollo mission	Delivery date to prime contractor	Launch date
		Boilerplate	
BP-6	PA-1	December 1962	November 7, 1963
BP-12	A-001	March 1963	May 13, 1964
BP-13	AS-101	April 1963	May 28, 1964
BP-15	AS-102	July 1963	September 18, 1964
[a]BP-14	--	February 1965	--
BP-23	A-002	May 1963	December 8, 1964
BP-23A	PA-2	January 1965	June 29, 1965
BP-22	A-003	July 1964	May 19, 1965
		Command module	
SC-002	A-004	December 1964	January 20, 1966
[b]SC-010	--	January 1965	--

[a]Ground test vehicle.

[b]Backup to SC-002.

TABLE III. - APOLLO DFI PROGRAM MILESTONES - Concluded

Vehicle	Apollo mission	Delivery date to prime contractor	Launch date
Command module - Concluded			
[a]SC-006	--	December 1964	--
[a]SC-008	--	February 1965	--
SC-009	AS-201	December 1964	February 26, 1966
SC-011	AS-202	March 1965	August 25, 1966
SC-017	4	August 1965	November 9, 1967
SC-020	6	November 1965	April 4, 1968
SC-101	7	June 1966	October 11, 1968
SC-103	8	August 1966	December 21, 1968
Lunar module			
[a, c]HS-1	--	February 1966	--
[a]LTA-8	--	May 1966	--
LTA-2	6	October 1966	April 4, 1968
LTA-10	4	June 1966	November 9, 1967
LM-1	5	June 1966	January 22, 1968
[a]LM-2	--	August 1966	--
LM-3	9	October 1966	March 3, 1969
LTA-B	8	August 1968 (to Kennedy Space Center)	December 21, 1968

[a]Ground test vehicle.

[c]House spacecraft (HS) 1.

PROBLEM AREAS

The major component problem areas were in electromechanical devices and involved attitude gyros, cinecameras, and tape recorders. Although all these components were troublesome in regard to reliability and delivery schedules, tape recorders were the major concern.

Tape Recorders

Tape recorders were key components in the DFI systems because they were a prime source of data. Most DFI was selected on a proven off-the-shelf basis. The first recorders were selected similarly because they were involved in production runs totaling more than 100 units and were proved in high-linear-g environments on sled tests. Despite their apparent ruggedness, they failed in the required 12.5g vibration environment and had to undergo major redesign modifications on the main housing castings to meet this requirement.

Various troubles continued to plague the recorders during the DFI activity: motors burned out, tape jammed or would not start, tape curled on reel edges, and, periodically, certain tracks would not record because of improper tape-to-head tension. After various modifications, however, they were made operable for flight usage. One of the contributing factors to the successful flight operation of these recorders was the exclusive handling and tape loading by a few trained personnel who stayed with them until launch. Later in the program, a higher capacity recorder was introduced; this recorder was a complete departure from the former type and used a co-belt system with flangeless tape reels. In effect, this design was used to compensate for the previous problems of flutter and tape packing. However, a new set of problems arose regarding belt tension and belt materials. Various modifications to the capstans and belts had to be made to make the recorder operational for the later boilerplate and airframe series of launches.

The problem of recorder qualification was in the vibration areas. It was difficult to make a recorder with a relatively high tape capacity conform to "wow and flutter" specifications under a 12.5g random environment. One factor that alleviated the situation and permitted easier qualification was the introduction of shock mounts. The problems that involved bearings, motors, and tape are standard problems that continue in present-day flight tape recorders. The reason for this situation is probably twofold. First, the fundamental principle of pulling tape is still the same. Furthermore, the fact that flight recorders are frequently redesigned or greatly modified for various programs tends to negate the maturity or reliability that might be established with a given type.

Commutators

Another area of concern that can be categorized as electromechanical is commutators. In 1963, low-level commutators were not available in a reliable solid-state configuration. At that time, the DFI used mechanical devices for multiplexing the low-level measurements. Initially, these devices proved very reliable, having lifetimes

(800 hours) approaching those of the electronic units. However, the reliability was compromised when the devices were modified to program channel and rate selection. This modification increased the number of switching poles and caused heavier motor loading. This was a classical case of improving an item that was highly reliable to the point that it became complicated and unreliable. The resultant lifetime was reduced to approximately 250 hours, and the units were phased out of the LM systems and replaced by solid-state devices, which by then had become available. All high-level-type commutators used solid-state techniques throughout the project and proved to be reliable components.

Vehicle System Cabling

The area of concern that seriously affects the integrity of the entire system is the vehicle cabling. Neglect in the proper design, accessibility, and installation of cables can offset all the reliability engineered into the components of the system. The fact that the DFI harness was completely replaced twice on BP-6 and once in LM-1 is evidence of a problem area. In the BP-6 harness, the problems were design and fabrication errors; for instance, all the signal shields were inadvertently short-circuited because uninsulated instead of insulated shields were used. This type of cable problem was not as detrimental to the system as the LM-1 problem.

In LM-1, the scarcity of available space and the consequent miniaturization of certain DFI components led to the design of a central signal-conditioning unit that had a density of 1600 connector pins over a 45-square-inch faceplate. This unit was demonstrated on the system breadboard at MSC as being operational but requiring care in installation to provide for cable accessibility. The unit was installed initially in LM-1 in a relatively inaccessible position, and the mating cable harness consisted primarily of no. 26 AWG wire. After a series of requirements changes and troubleshooting procedures that involved moving and opening the signal-conditioning unit, some of the wires in the harness became fatigued and broken. This problem was also manifested in the harness in other areas where cable movement was excessive. The situation deteriorated to the point at which attempts to rectify certain cable breakages precipitated further breakages in adjacent areas. The Apollo Program was forced into the major decision of reinstalling the signal-conditioning unit with a new swing-out shelf (for accessibility) and refabricating and installing a new DFI harness with heavier gage wire (24 AWG).

Handling Damage

Handling failures were induced by such things as excess power voltage, reverse power polarity, overpressure, riveting, and physical damage during and after installation. It is impossible to avoid handling failures at field sites despite all the precautions and procedures. However, any equipment installation should be delayed to the last possible moment to avoid vehicle manufacturing operations as much as possible. The overall provision of spares for components must necessarily take into account the attrition resulting from handling damage; this point is amplified by the comments already made in the "Component Failures" section, which shows the relatively high incidence of handling failures.

CONCLUDING REMARKS

The building-block approach used on the DFI project permitted a standardized system that was basically common to the Apollo vehicles. This feature, together with the programable signal-conditioning units, allowed flexibility in program changes and new requirements. Measurement changes were implemented on flight vehicles sometimes within a matter of hours; some small systems were designed, qualified, and installed in periods of 3 months.

Certain features of the activity helped to permit this versatility. The project had access to fast purchasing through the NASA Manned Spacecraft Center procurement chain, and the hardware that was purchased was invariably off-the-shelf type and readily available. Moreover, the DFI was not subject to the "mission critical" standards as were the operational systems and consequently had greater latitude in reliability and quality control. This feature eliminated the necessity for the inclusion of "high reliability" parts and generally excluded the equipment from any developmental programs. The fact that all flight hardware was acquired on "fixed cost" contracts (rather than "cost plus" type contracts generally incurred by developmental contracts) is an indication of the small amount of development work involved throughout the project.

As in most flight test activities, the DFI requirements were generally defined late. In some cases, this late definition only permitted a period of approximately 12 months between the go-ahead and the actual system delivery to the prime contractor. Certain measures that were taken helped alleviate these scheduling problems; one of these involved the standardization of measurement ranges. Transducers comprised the largest area of the system, contained the greatest variety of component types, and were subject to the greatest number of changes (because of varying measurement requirements). To simplify and expedite procurement, transducer ranges were standardized to cut down the variety of instruments. In effect, the project offered a standardized list of transducer measurement ranges instead of responding to specific range requests. The standardized levels that were established for component qualification ensured that DFI equipment generally could be installed in any part of a vehicle without incurring additional testing. This not only permitted mounting flexibility but also simplified procedures, procurement, and paperwork. Similarly, standardization of electrical specifications for components, regarding input/output voltages, impedances, and so forth, ensured simplified system interfaces and set specific controls and standards for procurement with the component vendors. This latter point was significant because it provided clear guidelines to component engineers who had to interface with approximately 45 equipment vendors.

The DFI project obtained an overall 98-percent successful data retrieval. Two factors were major contributors in achieving this status. First, off-the-shelf mature hardware was selected; this hardware had already been screened for poor parts by usage. (Only minor parts problems were experienced throughout the activity). Second, all flight hardware was subjected to acceptance procedures that emphasized environmental testing and effectively weeded out the weaker components before commitment for flight purposes. Certain precautionary design features were instrumental in minimizing system problems during the testing and operational phases of the DFI activity. The adoption of internally regulated power supplies (where practical) for individual units made most parts of the system independent of single-point power supply failures and also permitted signal-to-powerline isolation, which prevented ground loop problems and gave greater immunity from electrical interference. The DFI was not designed with true redundancy, and such features as independent power regulators precluded the possibility of losing large sections of the system because of single power unit failures. The technique of cross loading measurements from a given area onto different radio-frequency transmitters and modulation packages provided some functional redundancy and ensured that a percentage of pertinent parameters could be preserved if a major system link were lost. Most units of the system were designed to include polarity-reversal circuitry to prevent the gross destruction of components during inadvertent system power reversals. The reversal of power polarity on complete systems (and components) actually occurred many times during the project.

Various problems are discussed in the report, but one of the most significant areas from which some very important lessons were learned concerned system cabling. From the cabling problems cited, three conclusions can be drawn. First, high-density wiring configuration should be avoided. Second, signal conditioning should be decentralized or made remote so that low-density connector configuration can be achieved to permit easy access and repair and not result in inflexible bundles of cables. Third, the DFI system involved frequent equipment changes; therefore, it should use a heavier gage wire than the more permanently situated, operational-type equipment. For general signal wiring, no. 22 AWG wire is recommended, although no. 24 AWG wire is suitable if weight becomes a constraining factor. Finally, the installation design with respect to accessibility should be given proper and timely attention. Neglect of this area will promote needless damage and generally will lead to a more unreliable system. In the component area, tape recorders proved to be the most problematical because they occupied key positions and were relatively unreliable. Failure in a recorder would have meant a significant or total loss of data; consequently, these devices were given exclusive personnel handling treatment throughout their test and flight phases.

The benefits gained from keeping the DFI independent are worth mentioning. Obviously, the DFI had many interfaces with other systems, which were, in fact, the sources of the measurement parameters. The point to be made, however, is that the control, power, wiring, and calibration functions generally were independent of other onboard systems. This fact meant that modifications (particularly late ones) required of the DFI because of measurement changes could be implemented with little or no impact on the vehicle operational systems. It also meant that the DFI could be checked out without disturbing other systems. The DFI was used frequently to take advantage of unscheduled vehicle downtime for additional testing because of its overall independence of operation. It could be quickly energized and checked with its own support equipment. The net result was that testing of the system was easily dovetailed into the vehicle master test plans and provided a convenient means for schedule optimization during the vehicle test operations at the prime contractors' plants.

Lyndon B. Johnson Space Center
National Aeronautics and Space Administration
Houston, Texas, November 16, 1973
914-50-50-06-72

APPENDIX A

BOILERPLATE PHASE

INTRODUCTION

The boilerplate (BP) vehicles were constructed to simulate the weight, shape, and center of gravity of the man-rated Apollo spacecraft. The flight test series consisted of five launches from White Sands Missile Range and two from the NASA John F. Kennedy Space Center (KSC). In addition to the rocket launches, boilerplate vehicles were dropped from aircraft, impacted into land and water, and used during parachute-evaluation tests. The boilerplate vehicle, originally quite simple, became a complex, functioning unmanned spacecraft as the program progressed. The test program during this phase gathered data about basic spacecraft design, operation of subsystems, and compatibility of spacecraft and launch vehicles.

APOLLO MISSION PA-1, BP-6 VEHICLE

A total of 94 measurements was required on BP-6 during Apollo mission PA-1. The development flight instrumentation (DFI) flight equipment used is listed in table A-I. The BP-6 during launch escape system test firing is shown in figure A-1.

The system operated satisfactorily except for two pressure-measurement channels. One incurred a 4-second interruption and the other a 1-psi calibration displacement.

TABLE A-I. - BP-6 FLIGHT EQUIPMENT

Component	Quantity
Telemetry modulation package	1
Telemetry radio-frequency (rf) package	1
Timer	1
Tape recorder	1
Signal conditioner	1
Power control	1
Junction box	1
Main battery	1
Pyrotechnic battery	2
Rate gyroscope package	1
Attitude gyroscope	3
Linear accelerometer	6
Vibration system	6
Amplifier rack	3
Pressure transducer	42
Temperature system	4
Amplifier	5
Resistance thermometer	4

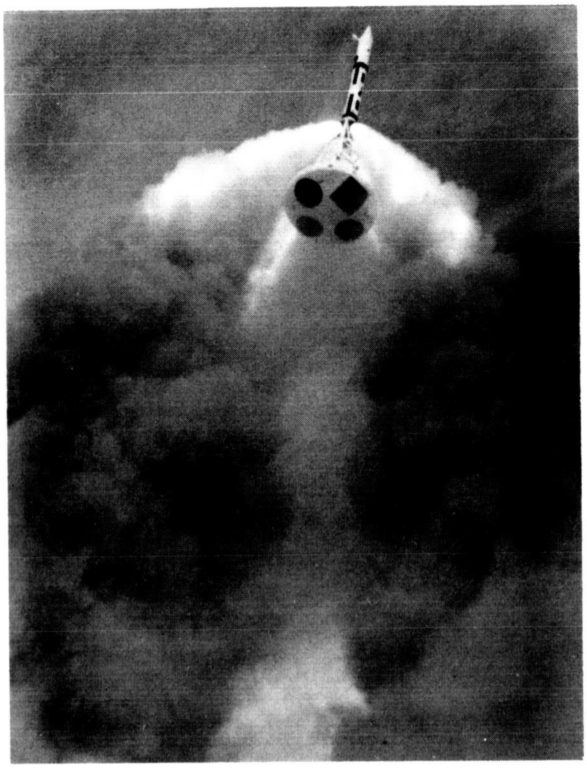


Figure A-1. - The BP-6 vehicle during launch escape system test firing.

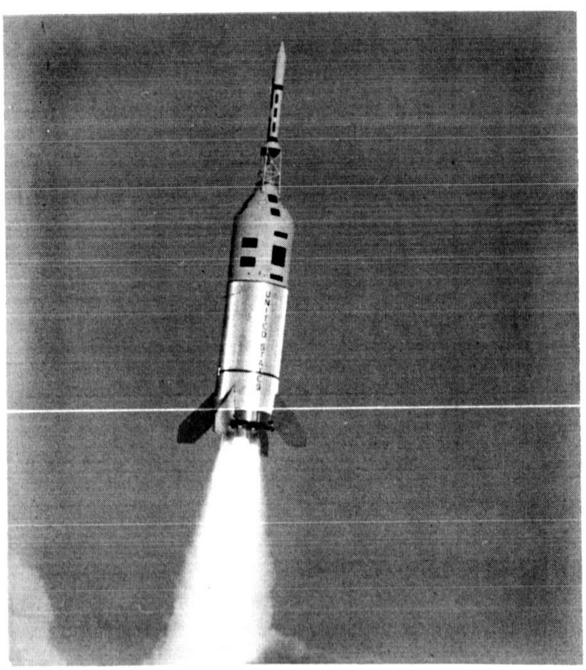


Figure A-2. - The launch of BP-12.

APOLLO MISSION A-001, BP-12 VEHICLE

Apollo mission A-001, the second abort test, used BP-12 to test the launch escape system under high dynamic pressure and transonic speed conditions. The test vehicle consisted of the spacecraft (launch escape system, BP-12 command module, and boilerplate service module) and the launch vehicle (Little Joe II (LJ-II) booster) (fig. A-2).

A total of 138 measurements was required on BP-12 during mission A-001. The DFI equipment complement was similar to that flown on BP-6. The equipment used is listed in table A-II.

The telemetry system performed satisfactorily except for a brief disturbance at 28.47 seconds into the flight and the failure of three measurements to provide data. The three 16-millimeter cameras provided flight data, but the service module camera evidently was damaged by the explosive force from the thrust termination of the LJ-II booster.

TABLE A-II. - BP-12 FLIGHT EQUIPMENT

Component	Quantity
Telemetry modulation package	1
Telemetry rf package	1
C-band transponder	2
Tape recorder	1
Timer	1
Signal conditioner box	1
Junction box	1
Main battery	1
Pyrotechnic battery	7
Power control box	1
Rate gyroscope package	1
Attitude gyroscope	3
Linear accelerometer	6
Relay box	1
Pressure transducer	26
Beacon line filter	2
Amplifier rack	9
Amplifier	15
Resistance thermometer	3
Temperature signal conditioner box	1
Temperature simulator box	1
Commutator	1
Camera	3

APOLLO MISSION AS-101, BP-13 VEHICLE

Apollo mission AS-101 was the first flight of an Apollo spacecraft configuration with a Saturn booster vehicle. The test vehicle consisted of the booster (Block II Saturn I and an S-IVB) and the spacecraft (launch escape system, BP-13 command module, and boilerplate service module). The DFI equipment is listed in table A-III.

Performance of the instrumentation and communications systems throughout the mission was satisfactory. Analysis revealed that 106 of the 112 measurements provided reliable data.

TABLE A-III. - BP-13 FLIGHT EQUIPMENT

Component	Quantity
Telemetry modulation package	3
Telemetry rf package	3
C-band beacon	2
Power control box	1
Junction box	1
Battery	6
Signal conditioner box	1
Amplifier rack	15
Amplifier, direct current	22
Pressure transducer	25
Accelerometer	7
Vibration system	3
Strain gage	48
Commutator	1
Resistance thermometer	16
Calorimeter	20
Zone box	20
Temperature signal conditioner box	1
Beacon line filter	2

APOLLO MISSION AS-102, BP-15 VEHICLE

Apollo mission AS-102, the second flight of an Apollo spacecraft configuration with a Saturn launch vehicle, was launched from KSC on September 18, 1964. The BP-15 spacecraft was used for this test. Figure A-3 is a block diagram of the instrumentation and communications subsystems. A total of 133 measurements was required on BP-15 during mission AS-102. The DFI equipment used to obtain the required measurements is listed in table A-IV.

The flight performance of the BP-15 instrumentation system was satisfactory. Of a total of 133 measurements, 131 provided continuous data. The two C-band beacons provided good tracking information throughout the flight.

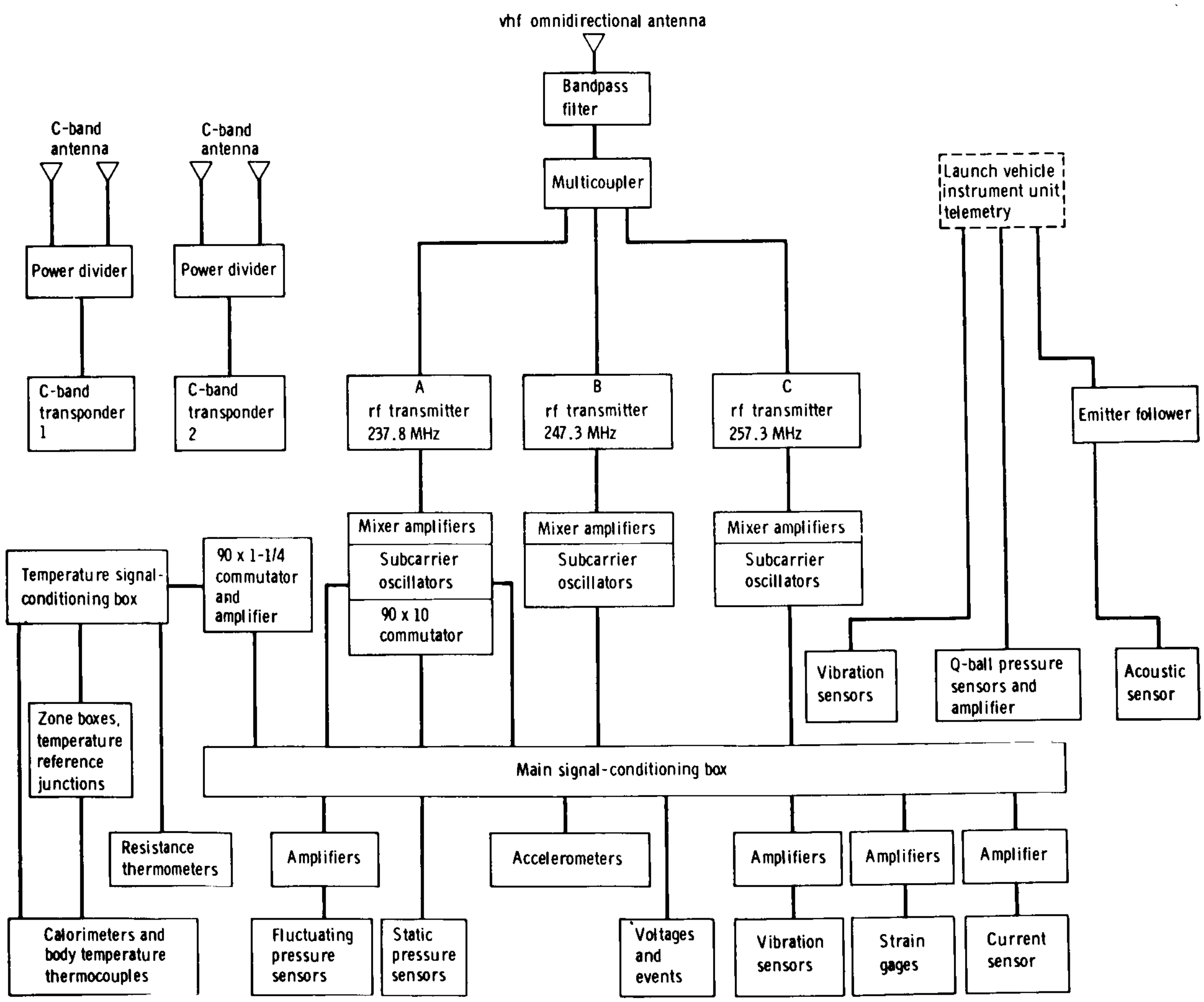


Figure A-3. - The instrumentation and communications subsystems used on BP-15.

TABLE A-IV. - BP-15 FLIGHT EQUIPMENT

Component	Quantity
Telemetry modulation package	3
Telemetry rf package	3
90 × 10 commutator	1
90 × 1-1/4 commutator	1
Main signal conditioner box	1
Temperature signal conditioner box	1
Amplifier rack	11
Amplifier	21
Vibration system	6
Pressure transducer	24
Accelerometer	7
Strain gage	48
Resistance thermometer	20
Thermocouple	9
Calorimeter	20
Zone box	20
Microphone	1
Ball assembly	1

APOLLO MISSION A-003, BP-22 VEHICLE

The BP-22 spacecraft, which was launched on an LJ-II booster, contained 230 DFI measurements. The flight equipment is listed in table A-V. The flight performance of the BP-22 instrumentation system was proved satisfactory by the proper function of all 230 measurements during Apollo mission A-003. However, significant information was not obtained from some of the instrumentation because of an early abort of the flight.

TABLE A-V. - BP-22 FLIGHT EQUIPMENT

Component	Quantity
Telemetry modulation package	2
Tape modulation package	10
Commutator	1
Telemetry rf package	2
C-band beacon	2
Beacon line filter	2
Tape recorder transport	2
Tape recorder electronics	2
Tape recorder line filter	2
Camera battery	3
Logic battery	2
Coded timer	1
Signal conditioner box	1
Signal distribution box	1
Power control box	1
Main battery	2
Junction box	1
Rate gyroscope package	1
Attitude gyroscope	2
Junction box	1
Inverter	1
Pressure transducer	44
Accelerometer	6
Amplifier	49
Accelerometer	34
Pressure transducer (acoustic)	9
Amplifier rack	2
Breakwire measurement adapter	1
Frequency detector	1

TABLE A-V. - BP-22 FLIGHT EQUIPMENT - Concluded

Component	Quantity
Voltage monitor, alternating current	1
Temperature signal conditioner box	1
Zone box	19
Thermocouple	3
Calorimeter	19
Resistance thermometer	6
Camera	3
Camera mount	3
Camera control box	3
Pulse generator	3
Strain gage	8
Displacement transducer	1

APOLLO MISSION A-002, BP-23 VEHICLE

The BP-23 spacecraft, which was launched on an LJ-II booster, contained 190 DFI measurements. The flight equipment is listed in table A-VI.

Of the 132 tape-recorded channels, 128 provided satisfactory data during Apollo mission A-002. The C-band beacons were interrogated by three PPS-16 radars during flight. The command module was tracked throughout the flight, and satisfactory return signals were received from both transponders. The tower and command module cameras performed satisfactorily. The service module camera provided only 4 seconds of usable film before it jammed because of mechanical malfunction.

TABLE A-VI. - BP-23 FLIGHT EQUIPMENT

Component	Quantity
Amplifier	3
Resistance thermometer	3
Pressure transducer	53
Linear accelerometer	2
Junction box	1
Transponder	2
Beacon line filter	2
Relay box	1
Pressure transducer	11
Amplifier	16
Telemetry modulation package	1
Telemetry rf package	1
Onboard timer	1
Tape recorder	1
Signal conditioner box	1
Rate gyroscope	1
Attitude gyroscope	3
Power control box	1
Main battery	1
Pyrotechnic battery	7
Linear accelerometer	4
Strain gage	8
Displacement transducer	1
Camera	3
Camera mount	3
Camera control box	3
Camera pulse generator	3

APOLLO MISSION PA-2, BP-23A VEHICLE

The vehicle DFI used on BP-23A (pad-abort mission PA-2) was the refurbished system that was used on BP-23 for Apollo mission A-002. The instrumentation subsystem provided good quality data from all measurements on the vehicle. Complete telemetry data coverage was obtained from the ground stations.

The two onboard cameras, one mounted in the launch escape system tower and one in the command module, operated as programed by the subsystem control box. The tower camera showed launch escape system motor plume impingement on the command module until the lens was sooted during tail-off of the launch escape motor. The command module camera, started at 5 seconds after launch, gave excellent coverage of apex-cover jettison, tower jettison, and all parachute deployments.

APPENDIX B

COMMAND AND SERVICE MODULE PHASE

INTRODUCTION

The command and service module (CSM) phase of the test program used vehicles that were constructed to the production model configuration of the man-rated Apollo spacecraft. These vehicles were used to confirm the structural integrity and the operational capabilities of the Apollo subsystems. All Apollo vehicles listed in this appendix, with the exception of spacecraft 002 (SC-002), were launched from the NASA John F. Kennedy Space Center on Saturn boosters.

APOLLO MISSION A-004, SC-002 VEHICLE

A total of 242 onboard measurements was required on Apollo mission A-004. Satisfactory data were obtained from 237 of the 242 onboard measurements. Failure of the single operational scimitar antenna (because of damage by protective-cover jettison) following abort caused a loss of both radio-frequency telemetry links at 74.7 seconds after lift-off. However, data transmitted by these links were recorded on the onboard tape recorder and recovered. The spacecraft was launched on a Little Joe II booster. The SC-002 flight equipment is listed in table B-I.

TABLE B-I. - SC-002 FLIGHT EQUIPMENT

Component	Quantity
Telemetry package	2
Tape modulation package	10
90 × 10 high-level commutator	2
Main instrument battery	2
Tape transport	2
Tape electronics	2
C-band transponder	2
Line filter C-band transponder	2
Signal distribution box	1
Signal conditioner box	1
Power control box	1

TABLE B-I. - SC-002 FLIGHT EQUIPMENT - Concluded

Component	Quantity
5-volt reference supply	1
25-volt reference supply	1
Time reference generator	1
Logic battery	2
Amplifier, direct current	98
Microphone assembly	1
Amplifier, alternating current	1
Rate gyroscope package	1
Attitude gyroscope system	1
Gyroscope package	2
Inverter	1
Junction box	1
Pressure transducer	51
Amplifier, alternating current	31
Pressure transducer	2
Angular displacement	2
Resistance thermometer	13
Thermocouple	3
Calorimeter	6
Zone box	9
Accelerometer	29
Linear accelerometer	6
Strain gage	164
Camera	1
Camera mount	1
Camera control box	1
Camera pulse generator	1
Camera battery	1

APOLLO MISSION AS-201, SC-009 VEHICLE

An instrumentation package in the CSM interfaced with the communications subsystem and processed 37 measurements from the structural subsystem, the instrumentation system, the service propulsion system, and the reaction control system during Apollo mission AS-201. An instrumentation package in the service module interfaced with the communications subsystem and processed 93 measurements from the structural subsystem. The SC-009 and service module flight equipment is listed in table B-II. The performance of all development flight instrumentation (DFI) equipment was satisfactory.

TABLE B-II. - SC-009 AND SERVICE MODULE FLIGHT EQUIPMENT

Component	Quantity
Telemetry package	1
Modulation assembly	2
Transmitter	2
90 × 10 high-level commutator	2
5-point calibrator	3
90 × 10 low-level commutator	3
Time-code generator	1
Command receiver	1
Auxiliary decoder	1
Command receiver voltage regulator	1
Modulation kit pulse amplitude modulation (PAM)/frequency modulation (FM)/FM system	1
5-point calibrator	1

SPACECRAFT 010

A DFI system for SC-010 was designed, procured, checked, calibrated, and installed on a breadboard. The integrated systems checkout of the breadboard was completed successfully, and the breadboard was shipped to the contractor facility. However, because of the success of the SC-002 mission, SC-010 was not flown. Components were returned to bonded storage areas at the NASA Manned Spacecraft Center to be used in other instrumentation systems.

APOLLO MISSION AS-202, SC-011 VEHICLE

The DFI system for SC-011 was composed of the items listed in table B-III. An instrumentation package interfaced with the communications subsystem and processed 60 measurements from the structural subsystem, the operational instrumentation system, the service propulsion system, and the reaction control system. A second package processed 93 additional measurements from the structural subsystem. The four camera systems were used to record panel and apex-cover deployment. All DFI equipment operated satisfactorily during Apollo mission AS-202.

TABLE B-III. - SC-011 FLIGHT EQUIPMENT

Component	Quantity
Telemetry package	1
Modulation assembly	2
FM transmitter	2
90 × 10 high-level commutator	2
5-point calibrator	3
Tape recorder support equipment	3
90 × 10 low-level commutator	3
Time-code generator	1
Modification kit PAM/FM/FM system	1
Camera	4
Camera power distribution	1
Timing pulse generator	1
Battery	2
Baroswitch	1
Control box	1

APOLLO 3 MISSION, SC-012 VEHICLE

Instrumentation equipment was supplied for the Apollo 3 mission (SC-012) but was never required because of the accident and fire within the spacecraft.

APOLLO 4 MISSION, SC-017 VEHICLE

The DFI equipment used on the Apollo 4 mission is listed in table B-IV. A self-contained camera system was included for purposes of photographing the Earth from a 10 000-nautical-mile apogee. Overall performance of the SC-017 telemetry system and camera system was satisfactory.

TABLE B-IV. - SC-017 FLIGHT EQUIPMENT

Component	Quantity
Modulation package	1
90 × 10 high-level commutator	2
90 × 10 low-level commutator	2
Time-code generator	1
Camera system	1
Camera	1
Lens	1
Camera control box	1
"G" switch	1
Battery	1

APOLLO 6 MISSION, SC-020 VEHICLE

The SC-020 vehicle contained instrumentation and camera systems (table B-V) similar to those used on the Apollo 4 mission. Equipment was provided to acquire Earth-mapping and CSM plume-impingement photographs. Performance of these systems was satisfactory, and useful data were obtained throughout the Apollo 6 mission.

TABLE B-V.- SC-020 FLIGHT EQUIPMENT

Component	Quantity
Modulation package	1
90 × 10 high-level commutator	2
90 × 10 low-level commutator	2
Time-code generator	1
Boost/entry camera	1
Boost/entry-camera control box	1
Earth-mapping camera	1
Earth-mapping-camera control box	1
"G" switch	1
Battery	1

APPENDIX C

LUNAR MODULE PHASE

INTRODUCTION

In addition to the flight vehicles, several ground test vehicles were constructed for the lunar module (LM) phase. The LM test article (LTA) vehicles contained only development flight instrumentation (DFI) systems and were designed to gather information on vibration and strain and to obtain acoustical data. Later configurations of the LM vehicle contained additional subsystems and more sophisticated instrumentation. The instrumentation measured additional parameters such as temperature, pressure, and acceleration. A C-band radar was also included.

HOUSE SPACECRAFT 1

A total of 241 DFI measurements was initially installed on house spacecraft 1 (HS-1). The equipment was similar to that used on LM-1. The HS-1 vehicle was constructed for ground test purposes and for in-house training.

LUNAR MODULE TEST ARTICLE 1

The DFI system installed in HS-1 was later reinstalled in LTA-1. The LTA-1 vehicle was constructed for ground test and training purposes.

APOLLO 4 MISSION, LTA-10 VEHICLE

A total of 38 DFI measurements was made on the LTA-10 vehicle for the Apollo 4 mission (table C-I). The instrumentation system performed satisfactorily except for three accelerometers that exhibited temporary bias offsets during maximum dynamic pressure conditions. This vehicle also carried the first constant bandwidth (CBW) modulation package to be launched into space.

TABLE C-I. - LTA-10 FLIGHT EQUIPMENT

Component	Quantity
Modulation assembly, CBW	1
Modulation assembly, proportional bandwidth (PBW)	1
90×10 high-level commutator	1
25-volt power supply	1
Signal conditioner box	1
Power control box	1
Temperature resistance thermometer	6
Vibration system	9
Linear accelerometer	5
10-watt transmitter	2
Acoustic pressure transducer	2
Amplifier	22
5-volt reference supply	1
Strain gage	64

APOLLO 6 MISSION, LTA-2 VEHICLE

The DFI system for the LTA-2 vehicle was very similar to the one installed on LTA-10 and performed satisfactorily throughout the Apollo 6 mission.

APOLLO 5 MISSION, LM-1 VEHICLE

A total of 195 DFI measurements was installed on the LM-1 vehicle for the Apollo 5 mission. The flight equipment used is listed in table C-II. The instrumentation system performed satisfactorily, with the exception of seven measurements that did not respond as planned. Four thermocouple measurements malfunctioned (probably became unbonded), and three low-pressure measurements failed (probably because of constricted ports from engine firing).

TABLE C-II. - LM-1 FLIGHT EQUIPMENT

Component	Quantity
10-watt transmitter	5
Telemetry modulation assembly, PBW	3
Telemetry modulation assembly, CBW	1
Signal conditioner	1
90 × 10 high-level commutator	2
90 × 10 low-level commutator	1
Vibration commutator	3
90 × 1-1/4 low-level commutator	1
Power control box	1
5-volt reference supply	1
C-band beacon	2
C-band filter	2
25-volt reference supply	1
Amplifier	38
Strain gage	80
Pressure transducer	45
Vibration system	18
Temperature resistance thermometer	36
Temperature thermocouple	25
Zone box	41
Strain/temperature system	2

LUNAR MODULE 2

The DFI system on LM-2 was nearly identical to that installed on LM-1. The system was completely checked and accepted at the contractor facility before the decision was made to place LM-2 in a nonflight status.

APPENDIX D

MANNED FLIGHT VEHICLE PHASE

INTRODUCTION

The launch of the Apollo 7 spacecraft on October 11, 1968, marked the beginning of the manned Apollo vehicle flights. Development flight instrumentation (DFI) was provided by NASA for three manned missions: Apollo 7, Apollo 8, and Apollo 9.

APOLLO 7 MISSION, COMMAND AND SERVICE MODULE 101 VEHICLE

A total of 167 DFI measurements was made on command and service module 101 (CSM-101) during the Apollo 7 mission. The flight equipment is diagramed in figure D-1 and listed in table D-I. The telemetry system performed satisfactorily except for one high-level commutator that became erratic during the reentry phase.

APOLLO 8 MISSION, CSM-103 VEHICLE

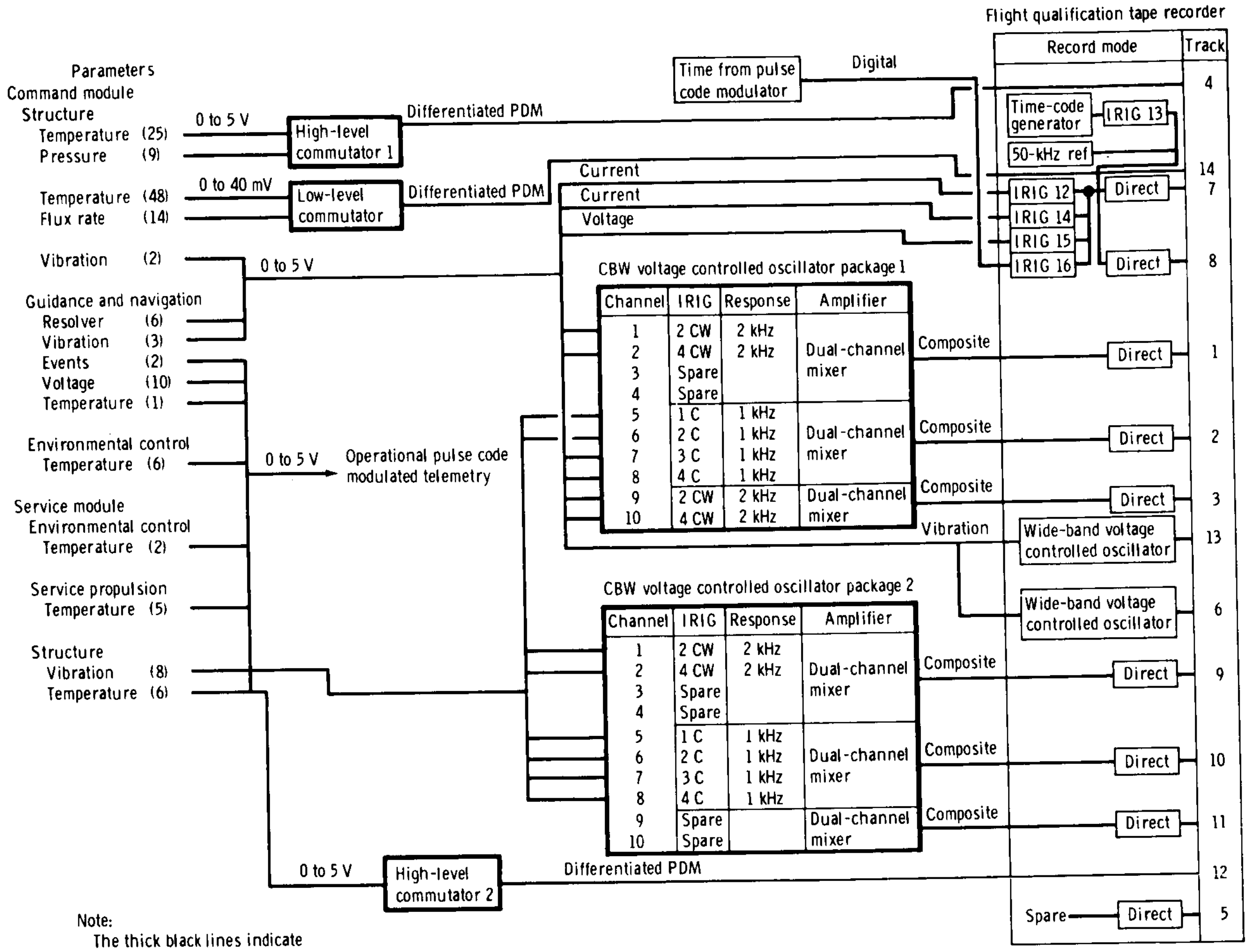
A total of 36 DFI measurements was made on CSM-103 during the Apollo 8 mission. Equipment similar to that used for CSM-101 (table D-I) was used for CSM-103 (table D-II). The telemetry system performed satisfactorily during the mission.

APOLLO 8 MISSION, LTA-B VEHICLE

A total of six DFI measurements was used on LTA-B for the Apollo 8 mission. The instrumentation performed satisfactorily during the mission.

APOLLO 9 MISSION, LM-3 VEHICLE

A total of 248 DFI measurements was made on LM-3 during the Apollo 9 mission. The flight equipment used is listed in table D-III. The overall evaluation of the total 248 measurements indicated an average return of 98.7 percent of the data over the 10-hour period of system operation.



Note:
The thick black lines indicate DFI equipment.

Key:
CBW - constant bandwidth
DFI - development flight instrumentation
IRIG - interrange instrumentation group
PDM - pulse duration modulation
ref - reference

Figure D-1. - The CSM-101 flight-qualification instrumentation diagram.

TABLE D-I. - CSM-101 FLIGHT EQUIPMENT

Component	Quantity
90 × 10 high-level commutator	2
90 × 10 low-level commutator	1
Modulation package, constant bandwidth (CBW)	2
Time-code generator	1

TABLE D-II. - CSM-103 FLIGHT EQUIPMENT

Component	Quantity
Modulation package, proportional bandwidth (PBW)	1
Time-code generator	1

TABLE D-III. - LM-3 FLIGHT EQUIPMENT

Component	Quantity
10-watt transmitter	5
Telemetry modulation assembly, PBW	3
Telemetry modulation assembly, CBW	1
Signal conditioner	1
90 × 10 high-level commutator	2
90 × 10 low-level commutator	1
Vibration commutator	3
90 × 1-1/4 low-level commutator	1
Power control box	1
5-volt reference supply	1
C-band beacon	1
C-band filter	2
25-volt reference supply	1
Strain gage	80
Pressure transducer	30
Temperature resistance thermometer	35
Temperature thermocouple	25
Zone box	48
Strain/temperature system	2
Vibration system	18
Pressure transducer	4
Amplifier	33
Power control timer assembly	1
90 × 10 low-level commutator	1

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