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EFFECTIVENESS OF SPACECRAFT TESTING PROGRAMS

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ABSTRACT

This paper departs from the usual subject of the Space Simulation Conference and concerns itself with the need for testing under simulated mission operational conditions and reviews the results of such tests from the point of view of the user. It presents a brief overview of the usual test sequences for high reliability, long life spacecraft and will analyze the effectiveness of the testing program in terms of the defects which are discovered by such tests. The need for automation, innovative mechanical test procedures and design for testability will be discussed.

INTRODUCTION

As our spacecraft have become progressively more complex and their missions have become more sophisticated and of longer duration, it has become increasingly important to improve the methods for verifying design and workmanship prior to making the decision to launch. Such verification is based on an extensive testing program which duplicates operation over a range of worst case mission scenarios and simulates exposure to launch, boost and space flight environments.

Since a typical spacecraft may cost $40 million, contain 60,000 electronic piece parts and must last 7 to 10 years in orbit, it clearly is wise to learn as much as possible about how well it can be expected to perform in space. A comprehensive test program must progress in a logical manner from development testing, through qualification and acceptance testing to flight readiness checkout at the launch site. In addition, the test program must include development, qualification and acceptance tests at progressively higher levels of assembly starting with parts and continuing through unit level (i.e., component, black box or assembly) to subsystem, spacecraft and system level tests. The real challenge lies in choosing the most perceptive and cost effective test procedures and environmental exposures at each assembly level. These must be tailored to the specific spacecraft and mission at hand but can be summarized for purposes of this paper as described below.

PARTS TESTING

Current procedures for piece-part testing are based on a large body of engineering data and various analyses of the physics of failure for the different kinds of parts. Standardization of test procedures has been fostered by the government and the electronic parts industry. For example, MIL-STD-883 defines screening tests for micro-electronic devices and is intended to yield an in-equipment failure rate of less than .004% per thousand hours for high reliability (Class S) parts. Detailed screening, sampling and lot qualification tests are specified, including temperature cycling, burn-in, particle impact noise detection (PIND), radiographic inspection and life tests, among others. Comparable specifications and procurement standards exist for other electronic parts and are reflected in the design manuals and procedures used by individual aerospace contractors. These procedures have resulted in a significant drop in part failures during subsequent unit level testing.

COMPONENT TESTS

Unit or component-level test procedures are designed to verify all functional performance requirements over a range of environmental exposures without disassembly or change in the configuration of the component as installed in the spacecraft. For electronic components this usually requires that test points be brought out to a special test connector; this makes it possible to inject test signals and observe the resulting response, waveforms, logic levels, etc. Computer controlled special test equipment is required for functional checkout of digital components and complicated electronic components which have several modes of operation and process a variety of signals. Mechanical and electromechanical components also require special purpose test equipment and test fixtures for functional checkout but generally do not require automated testing.

The environmental tests for component-level qualification and acceptance are specified in MIL-STD-1540A "Test Requirements for Space Vehicles". The newer spacecraft projects, especially those for important military missions, use the approach spelled out by this document. It involves functional testing before, during and after exposure to pyroshock, random vibration or acoustics, thermal cycling, vacuum and EMI. Depending on the nature of the component, thermal cycling is performed at ambient pressure or in a vacuum chamber. In either case at least eight temperature cycles should be applied to expose any workmanship or design defects before acceptance. Test levels and duration vary from unit to unit but should be selected to accelerate detection of inherent defects without inducing damage or degradation of good equipment.

SPACECRAFT TESTS

A simplified version of a typical spacecraft-level test flow is shown in Figure 1. There are three separate phases, namely an integration phase, an environmental qualification or acceptance phase and a prelaunch verification phase. During the integration phase the various components, subassemblies and harnesses which have undergone testing at the component and part level are installed on the spacecraft structure and interconnected to form subsystems and related equipment groups. Physical inspection and functional tests are performed under ambient conditions to verify interfaces and correctness of the assembly procedures. The environmental qualification or acceptance phase consists of functional performance tests under various environmental conditions to demonstrate that the full range of performance requirements is met. The usual approach is to conduct end-to-end tests which closely simulate the actual in-orbit mission. Large deployable appendages such as solar arrays and antennas are usually tested separately and may be removed during various electrical functional tests. The prelaunch verification tests are performed at the launch site and vary greatly from program to program. On some spacecraft a full functional test is performed prior to installing the spacecraft on the booster. In other cases only on-stand interface and command compatibility tests are conducted in addition to ordnance circuit checkout and fueling operations.

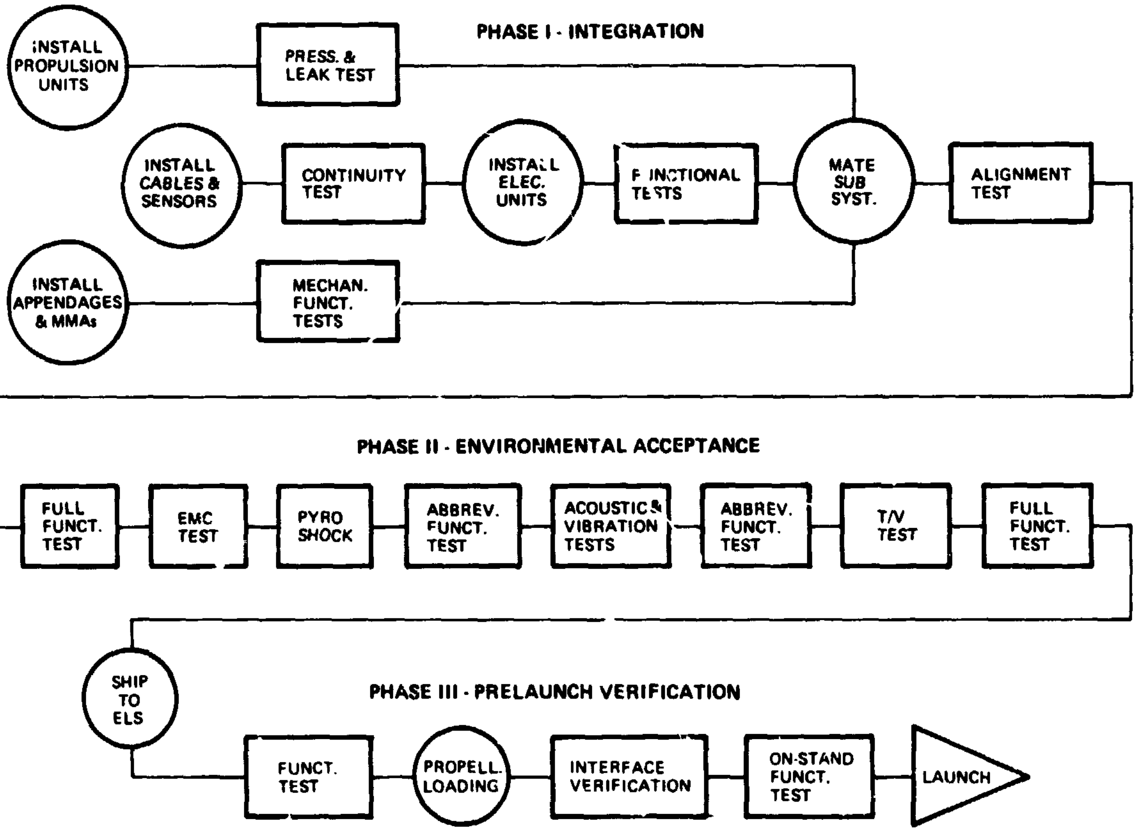

Figure 1 - Spacecraft Test Flow

EVALUATION OF TEST RESULTS

The ultimate objective is to launch spacecraft which will operate flawlessly in orbit for their specified lifetime or longer. We therefore strive for failure-free acceptance testing under conditions which closely simulate orbital conditions. Discrepancies which are discovered during acceptance testing are corrected prior to launch. In Figures 2 and 3 we have attempted to correlate orbital performance to performance during acceptance testing prior to launch. The average number of on-orbit and acceptance test discrepancies for four different projects representing a total of 25 spacecraft was normalized with respect to spacecraft complexity as measured in terms of part count and the average number of defects per 1000 parts was plotted for each project. Note that lower defect rates during acceptance testing yield better on-orbit performance. This is also borne out by Figure 3 which plots normalized on-orbit discrepancies against spacecraft level acceptance test discrepancies fo 12 individual spacecraft comprising three separate projects.

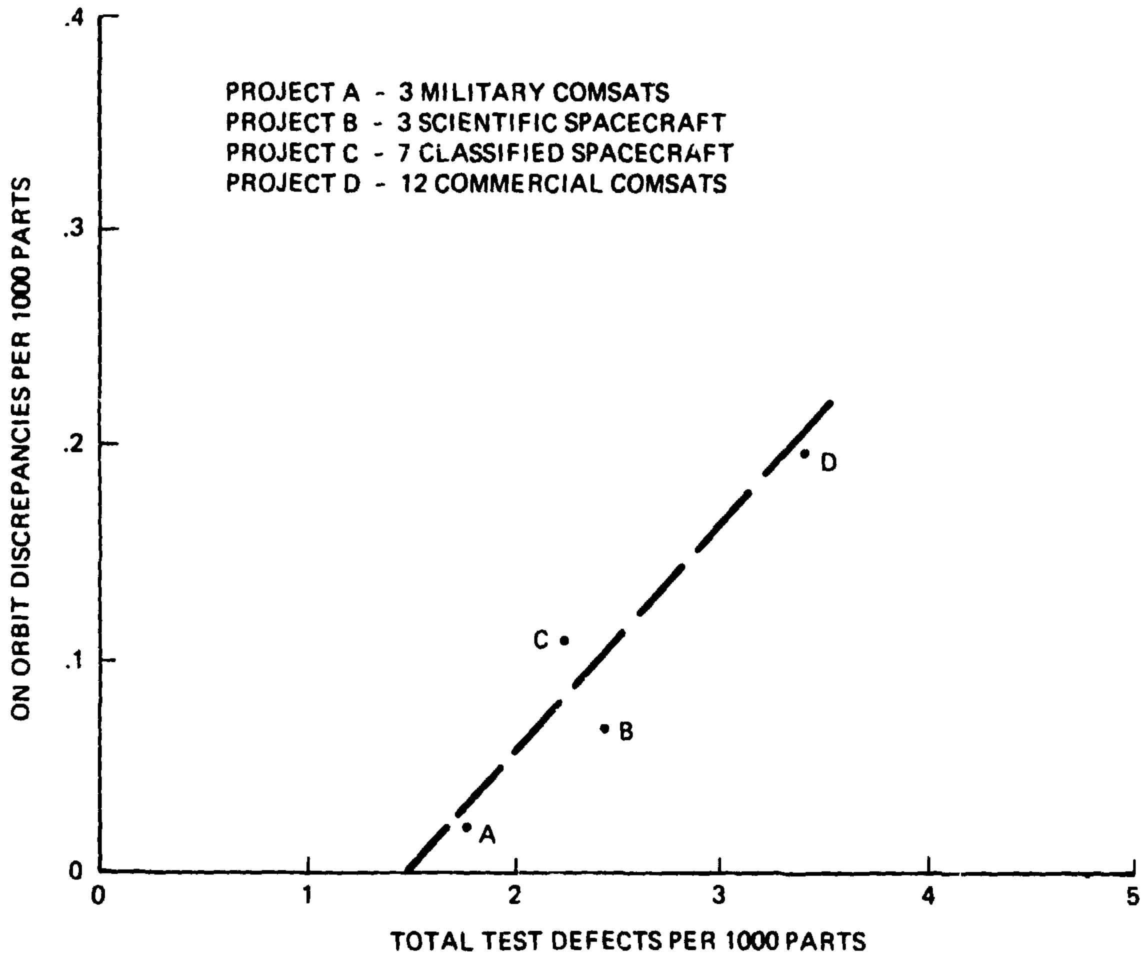

Figure 2 - Orbital Performance Correlation by Project

ANALYSIS OF TEST PROGRAM

In order to determine whether the current approach of duplicating anticipated operating modes and simulating space environments during the qualification and acceptance test program is really effective in locating all potential defects, we reviewed the actual test history for two recent spacecraft projects in more detail. Tables 1 ar 2 give the number and type of discrepancies which were found during successive phases of the testing program and during orbital operation. Project A consists of a qualification test spacecraft and three identical flight spacecraft all of which are currently operating in orbit. Each satellite contains approximately 58,000 electronic parts including about 5500 integrated circuits. Project B consists of three scientific satellites, all of which have been launched. The first of the three spacecraft served as both a prototype and flight vehicle since there was no separate qual spacecraft. Not including the scientific payload, each satellite contains approximately 30,000 electronic parts. The data of Table 2 does not include discrepancies found in the scientific instruments payloads since these were GFE.

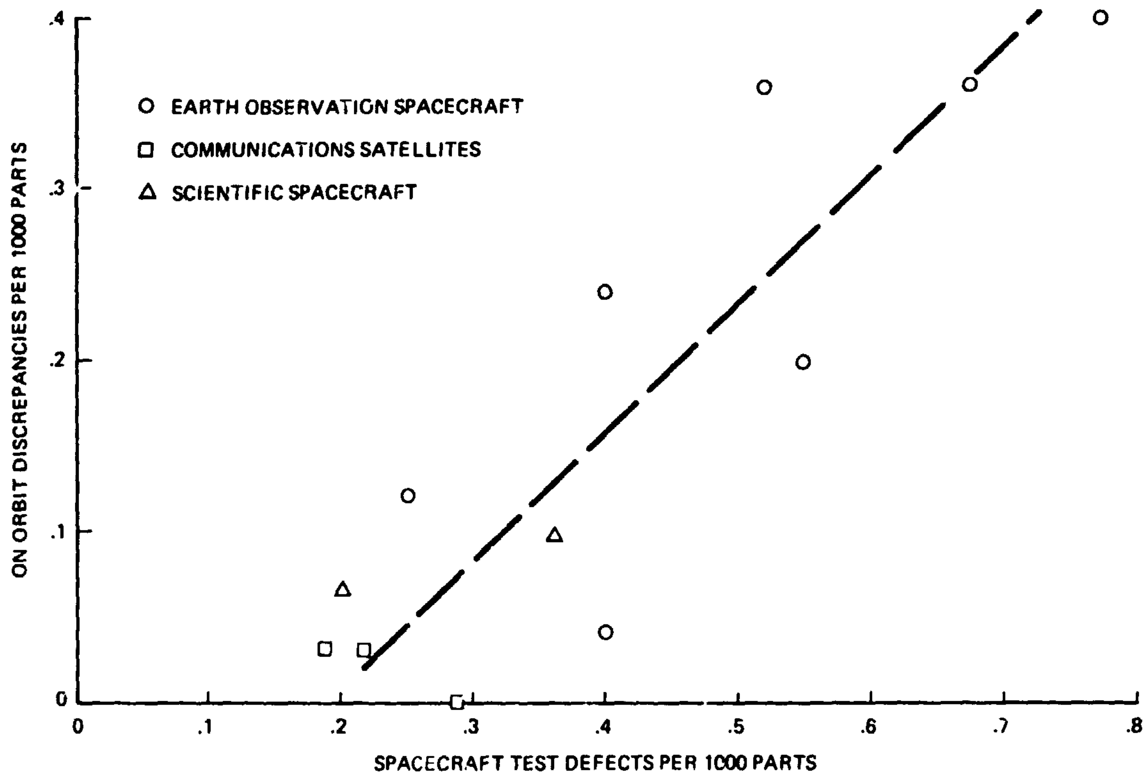

Figure 3 - Orbital Performance vs. System Test Results

Both Tables 1 and 2 provide the number of test discrepancies due to defects in the equipment under test, due to defects in test procedures and due to causes which are not related to any defect. Note that 367 and 174 product defects respectively were found during component level qualification and acceptance testing which were corrected prior to start of spacecraft level testing. Additional component defects were found during spacecraft level testing in the factory and at the launch site and during on-orbit operation. These additional discrepancies are called "escapes" because they escaped detection during component qual or acceptance testing. The escape ratio E_C is simply the ratio of component test escapes to total component defects. Similarly, the on-orbit discrepancies or performance anomalies are escapes from the ground test program if it is assumed that there are no spontaneous failures due to wearout or other causes during the mission life of the spacecraft. The escape ratio E_O then is the ratio of on-orbit discrepancies to total discrepancies. The purpose of these ratios and the data of Tables 1 and 2 is to provide a broad overview of what results can be expected from typical testing programs.

Table 1. PROJECT A - CAUSE OF TEST DISCREPANCIES

DEFECT/CAUSE OF DISCREPANCY	TEST LEVEL				ESCAPE RATIO	
	COMPONENT	SPACECRAFT	LAUNCH BASE	ON ORBIT	E_c	E_o
PRODUCT DEFECTS						
PART FAILURE	68	10	1		13.9	
COMPONENT MANUFACTURING	112	19		1	14.5	0.7
COMPONENT DESIGN	174	28			13.8	
SPACECRAFT ASSEMBLY		14				
SPACECRAFT DESIGN		16	1	3		15
TEST INDUCED	13	5				
UNDETERMINED		11				
TOTAL	367	103	2	4	14.3%	0.04%
PROCEDURE DEFECTS						
DOCUMENTATION	110	34				
TEST EQUIPMENT	41	23	1			
OPERATOR ERROR	35	16	1			
TEST SETUP	18	3	1			
TOTAL	204	76	3			
NON-DEFECTS						
WITHIN TOLERANCE/WAIVER	78	21	2	4		
UNABLE TO REPEAT PROBLEM	29	13		1		
TOTAL	107	34	2	6		

NOTES: 4 spacecraft, 3 in orbit, 3 years of orbital performance

To gain further insight into the effectiveness of environmental simulation for spacecraft level testing we have broken down the product defect discrepancies as shown in Tables 3 and 4 for Projects A and B respectively. Note that in both projects about 40% of the spacecraft acceptance test defects were found during the integration phase (Figure 1). Project A spacecraft received a temperature cycling test in a thermal chamber at ambient pressure during which 15 discrepancies were observed. During the subsequent T/V test four component failures occurred, all on the qualification spacecraft. These failures most likely were not induced by the vacuum environment since each of the failed components had previously passed a T/V test as part of the component acceptance sequence. Also note that on Project B only one component discrepancy was found during spacecraft T/V whose discovery cannot be credited to vacuum exposure. It appears therefore that exposure to vacuum as part of spacecraft level acceptance testing is not as profitable as testing over the widest possible temperature range in a thermal chamber. The merit of temperature cycling at the spacecraft level has also been demonstrated on a classified project as described in Reference 1.

Careful analysis of the discrepancy data for Projects A and B and several other projects leads to the following additional conclusions:

a) A majority of on-orbit discrepancies are due to subtle design defects which were not or could not be discovered during the qualification and acceptance test program.
b) Repetitive room ambient testing over a variety of mission scenarios and using different test methods will disclose more defects than simple environmental exposure without electrical stress.

Although it is impossible to achieve reliable operation in orbit by testing alone, it seems axiomatic that the more testing and the more variation in testing which is conducted, the better the orbital performance. This justifies the use of automated test equipment and requires design of spacecraft for testability as described below.

Table 2. PROJECT B - CAUSE OF TEST DISCREPANCIES

DEFECT/CAUSE OF DISCREPANCY	TEST LEVEL				ESCAPE RATIO	
	COMPONENT	SPACECRAFT	LAUNCH BASE	ON ORBIT	E_c	E_o
PRODUCT DEFECTS						
PART FAILURE	38	4		1	13.6	2.3
COMPONENT MANUFACTURING	79	5	1	-	17.0	-
COMPONENT DESIGN	45	19	4	5	38.3	6.8
SPACECRAFT ASSEMBLY	-	18	-	-		
SPACECRAFT DESIGN		11	2	1		
TEST INDUCED	6	-	2			
UNDETERMINED	6	4	1			
TOTAL	174	61	11	7	19.8	2.9%
PROCEDURE DEFECTS						
DOCUMENTATION	55	23	1			
TEST EQUIPMENT	62	4	1			
OPERATOR ERROR	50	13				
TEST SETUP	35	7	3			
TOTAL	202	47	5			
NON-DEFECTS						
WITHIN TOLERANCE/WAIVER	21	21	2			
UNABLE TO REPEAT PROBLEM	35	7	1	1		
TOTAL	56	28	3	1		

NOTES: 3 spacecraft, 3 on orbit, 4 years of orbital performance
Experiment TDRs not included

AUTOMATED TESTING

An objective of the spacecraft acceptance test sequence is to exercise all equipment and verify all the operating modes which the spacecraft will encounter during its space mission. To get an idea of the complexity of such a comprehensive spacecraft test (CST) consider the Tracking and Data Relay Satellite (TDRS) shown in Figure 4, which is currently under development at TRW. The TDRS system provides two-way communications with a large ground station for up to 20 spacecraft users in low earth orbits including the shuttle orbiter. In addition, it provides twelve 18 MHz repeater channels at C band for commercial communication satellite service. Altogether, there are more than 60 switchable RF communications links. Each TDRS contains 172 active electronic units which contain approximately 55,000 parts of which 4500 are integrated circuits. One thousand seventy-six discrete and 58 serial ground commands are available for configuration control, redundancy switching and power management during on-orbit operation. One thousand two hundred eighty-eight performance and status parameters are telemetered to the ground.

Table 3. PROJECT A - LOCATION OF TEST DISCREPANCIES

TEST/DEFECT	ELECTRONIC COMPONENTS			PROPULSION/STRUCTURE			HARNESS		
	Part	Workmanship	Design	Part	Workmanship	Design	Workmanship	Design	TOTAL DEFECTS
UNIT ACCEPTANCE	68	100	167		12	7			354
SPACECRAFT ACCEPTANCE	11	15	25	2	23	2	7	2	87
Integration	(3)	(6)	(12)	(1)	(11)	(1)	(3)	(1)	(38)
First Functional	(3)	(2)	(5)						(10)
Temperature	(2)	(2)	(7)		(1)		(2)	(1)	(15)
EMC/IM	(1)	(1)	(1)		(4)		(1)		(8)
Dynamics									
Post Dynamics					(4)		(1)		(5)
T/V Test	(2)	(2)							(4)
Preship Functional		(2)			(3)	(1)			(6)
Launch Base				(1)					(1)
ON ORBIT			1		1	2			4

Table 4. PROJECT B - LOCATION OF TEST DISCREPANCIES

TEST/DEFECT	ELECTRONIC COMPONENTS			PROPULSION/STRUCTURE			HARNESS		
	Part	Workmanship	Design	Part	Workmanship	Design	Workmanship	Design	TOTAL DEFECTS
UNIT ACCEPTANCE	35	79	43	3		2			162
SPACECRAFT ACCEPTANCE	5	6	23		7	1	11	12	65
Integration			(9)		(5)	(1)	(2)	(9)	(26)
EMC		(2)	(1)				(1)	(1)	(5)
First Functional	(1)	(2)	(4)		(1)		(3)		(11)
Dynamic Environment	(1)	(1)	(1)				(1)		(4)
Post Dynamic Functional	(1)		(2)				(2)		(5)
T/V Environment			(1)						(1)
Preship Functional	(1)		(1)		(1)		(2)		(5)
ETR	(1)	(1)	(4)					(2)	(8)
ON ORBIT	1		5			1			7

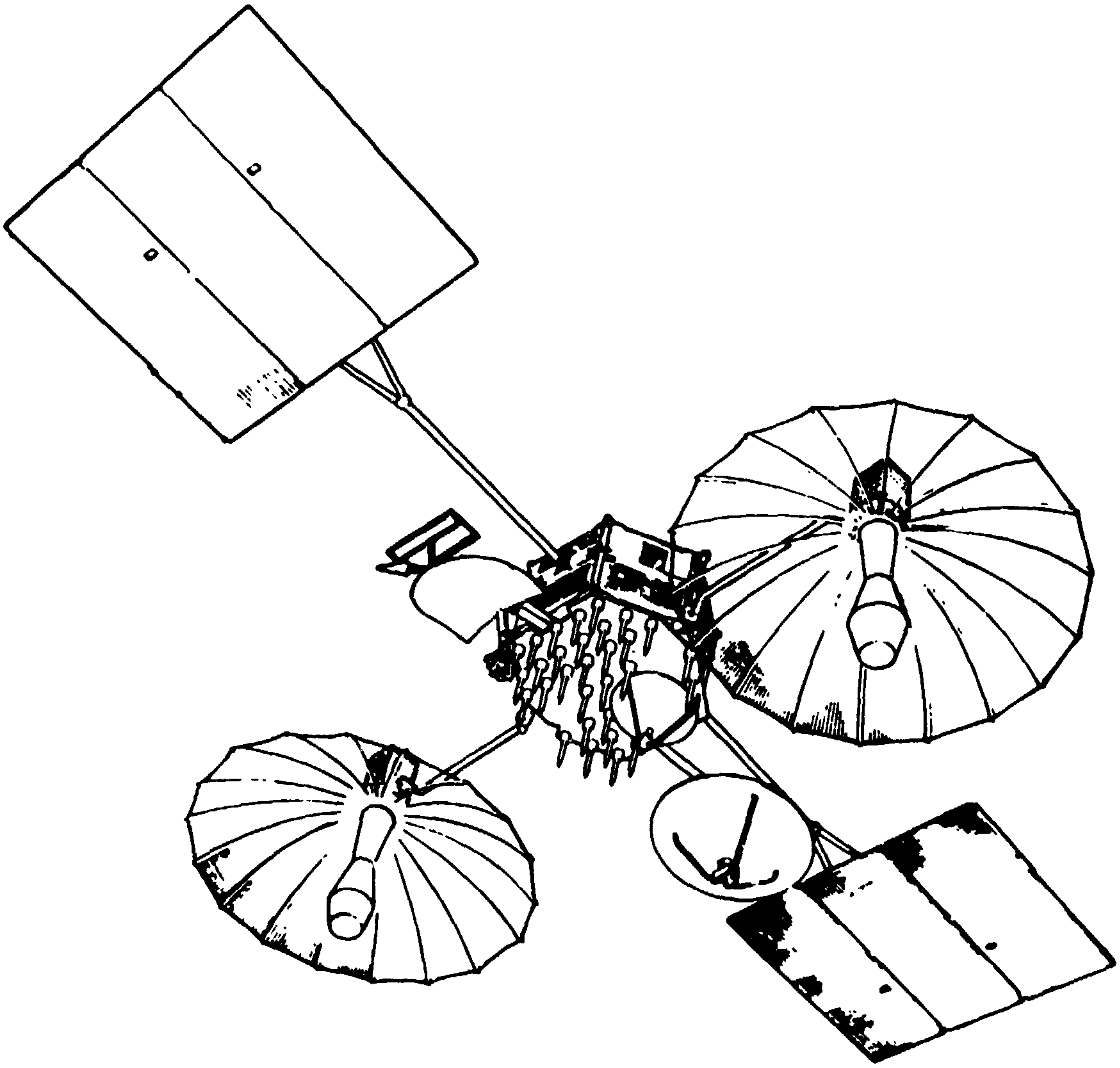

Figure 4. TDRS Spacecraft Configuration

Because of this complexity, a computer is required for generating command sequences, processing of telemetry data, conversion of RF measurements to performance parameters appearing in requirements specifications and for recording test results. In addition, a computer is needed to control the test equipment which supplies the RF signals, sensor stimuli and electrical power which are required for operation of the spacecraft.

The significance of computer based automated testing must be recognized by program managers and test facility managers. Although computers with sufficient speed and memory capacity to handle almost any spacecraft system test requirement are readily available, the software needed for conducting the test and processing the data must be designed for the specific spacecraft and costs significantly more than the computer hardware. Standardized software and distributed processors are being used wherever possible to reduce this high cost, but a lot more effort and innovation is required to arrive at optimized automated test systems.

TESTABILITY

Another issue which is of major concern to system test organizations is the basic testability of complex spacecraft. Over the last 10 to 15 years we have seen a change in the characteristics of spacecraft so that today we have more and more digital equipment using LSI and on-board processing and much larger and more complicated mechanical structures. In addition, the cross-strapping and reconfiguration capability of the electronic equipment is so great that it is very difficult to validate all anticipated operating modes prior to flight. It is, therefore, very important for test people to participate in the design of a spacecraft from the start in order to influence the following design features:

- selection of test points and location of test connectors
- equipment layout and clearances to enable replacement of faulty units
- provision of accelerated command capability for automated testing
- selection of telemetry measurements
- design of on-board computers for reprogrammability and interfacing with ground test equipment
- accessability for x-ray and visual inspection of critical connectors
- mechanical design which enables functional testing of mechanisms
- location of hard points to allow safe support during testing and transportation
- use of test-only sensors vs. flight telemetry

Several of these points may appear trivial or obvious, but experience has shown that their importance must not be underestimated. Another observation which should influence future activities is the fact that through automation and improved test equipment we have kept pace with the increasing complexity of electronic and electrical equipment aboard a spacecraft. On the other hand, our mechanical and structural test procedures have not changed significantly during the last 10 years. Acoustic and vibration test facilities are available but methods for conducting deployment tests under simulated zero gravity conditions are still relatively crude. On-orbit testing using the shuttle as a base may prove to be a viable option in the future and deserves further study.

CONCLUSIONS AND RECOMMENDATIONS

Performance during acceptance testing affects performance in orbit. Comprehensive testing over a wide range of usage and environmental conditions has been a major contributor to the excellent success record of spacecraft which have been launched.

Severe thermal thermal cycling over the maximum allowable temperature range rather than extended thermal vacuum exposure should be used to detect workmanship and design defects at the spacecraft level.

Test engineers and project managers must be aware of the large cost and schedule requirements for generating the software which controls the automated test procedures.

Methods for testing the deployment of appendages and/or verifying mechanical characteristics of large structures will have to be improved as structures become larger and more flexible and dynamic interactions which influence alignment and pointing accuracies become more critical.

Test engineers must participate in early spacecraft design decisions and exert greater influence to insure that testability is achieved.

REFERENCES

1. C. E. Nelson, "System Level Reliability Thermal Cycling", Proceedings of the 5th Aerospace Testing Seminar, Institute of Environmental Sciences, 19-21 September 1979, p. 69.