# VOLUME 1

# PROCEEDINGS OF THE SYMPOSIUM ON LONG LIFE HARDWARE FOR SPACE

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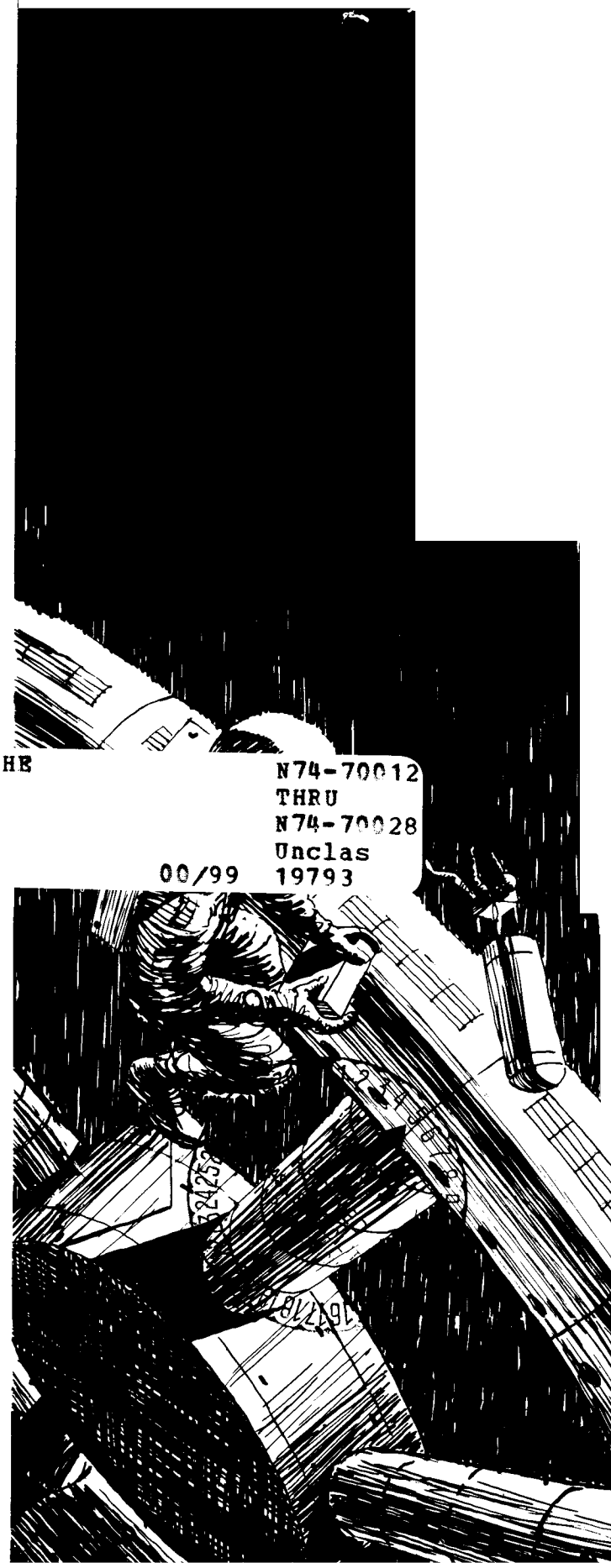

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# INTRODUCTION

The age of space has reached the time when longer missions are being performed than ever before, and even more extensive ones are in the planning stage. Because the life of the astronaut crew is completely dependent upon the operation of the hardware in the spacecraft, we must give devoted attention to achieving all that we can by way of proper hardware performance. The Marshall Space Flight Center and the Huntsville Section of the American Society for Quality Control agreed that a Symposium which would attempt to bring together some of the best thinking in the nation on this subject would be of considerable benefit at this time. After a selection was made of the major topics for discussion, a very considerable effort was put into determining who should speak on what topics. The speakers for the Symposium were then requested on an individual basis to present topics on particular subjects that lay within their specialty areas. Because the speakers are recognized experts in their field and because they were requested to present their own ideas in their particular area of specialty, it should be understood that no attempt has been made to review the material to be presented by the speakers on the part of either the Huntsville Section of the American Society for Quality Control or the Marshall Space Flight Center. The speakers are serving without any compensation whatever in the interest of the entire space program. The ideas which they present are their own and should not be taken to represent the official opinions of either the American Society for Quality Control or the Marshall Space Flight Center. We nevertheless feel that the material presented will serve its intended purpose of focusing much attention on this area where a great deal of work is needed and of provoking the thought, discussion, and effort necessary to achieve a large magnitude of improvement.

Dieter Grau
General Chairman

# ELECTRONIC PARTS AND LONG DURATION SPACE MISSIONS

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## Abstract

The prognosis for a second decade of spacecraft development indicates an ever increasing spacecraft complexity and in certain operations such as the Grand Tour, the demand for vastly increased successful operational time.

It is the intent of this document to consider, from the electronic parts viewpoint, the longevity preparation philosophy associated with 12 year outer planet missions.

This paper considers one of the major differences between current methods of electronic parts selection, control, qualification and screening, and those methods that may be applied in extra long life missions.

Certain of the long life environments are identified and their relationships to electronic parts problems are discussed. Possible solutions and/or methods of problem attack are submitted for consideration. Recognition is also given to the unique parts difficulties associated with the Grand Tour design and fabrication time schedule.

The discussion investigates the current electronic parts approach being planned and implemented at JPL in extra long life projects. The details for approved parts lists, parts review criteria and application notes concerning long duration missions are also included in the paper. Problems involving radiation, screening and qualification are presented and the Laboratory approach to these problems is discussed.

The paper finally presents some conclusions based on the study and investigation work being done at the Laboratory.

## Introduction and Philosophy

Current spacecraft require successful operation for a few months of combined ground tests and flight mission functions (typically eight months or 6000 hours). Outer Planet Spacecraft, contemplated for use

in such missions as the Grand Tour*, will be required to operate successfully for about twelve years (105,000 hours). The increase demanded in successful operational time, by a factor of about 18, will certainly require the development and application of unique techniques throughout the many spacecraft disciplines. Whereas a complete consideration of these multidisciplines would be beneficial to the understanding of the needed techniques, it is the intent of this paper to consider only the relationship of electronic parts and the long duration space mission. Specifically, the rationale supporting the requirement for new and unique techniques will be discussed. The current approach and recognized inadequacies of their application to long duration space missions will be considered. Certain problems will be identified and possible solutions offered, particularly those approaches being pursued at JPL.

It is readily apparent that the consideration of long duration space missions require a brief discussion of the interval involved. Today, the term long duration commonly means a period of time not exceeding fourteen months, (10,000 hours). One recent electronic parts symposium even defined long duration as twelve months. The term long duration in this paper refers to a period of at least ten years of trajectory time. In addition, pre-flight test time and preparations are added to make long duration in some considerations extend for nearly twenty years. This will be more fully expanded later in the discussion.

The approach to achieving electronic parts reliability through the years makes interesting study material.

The records indicate that the first significant realization of the importance of parts reliability resulted from Navy and Air Force experience during and immediately after World War II. In 1949 it was reported that about 70 percent of Navy electronic equipment was not operating properly. Another statistic indicated the Navy was supplying a million replacement parts a year to keep a total of 160,000 pieces of equipment in operation. The Air Force reported the trouble-free operation of electronic gear on bombers to be only about 20 hours.[1]

A broad attack on the problem by industry and government began to take shape in 1950-51 with the realization that it was costing more to maintain military equipment than to design and build it. Several

* The Grand Tour mission is a combined "fly by" of the planets Jupiter, Saturn, Uranus and Neptune. Two classes of trajectories exist. Low energy trajectories pass outside Saturn's rings and require about 11 years to Neptune encounter. High energy trajectories pass within Saturn's rings requiring more than 8 years to Neptune.

programs were undertaken including the work of the Advisory Group on Electronic Tubes (AGET) and the Arinc tube failure analysis study. One of the first attempts to obtain statistical data on component failures other than tubes was done by the Vitro Corporation in 1950. It was demonstrated that parts in wide production and use were more reliable than parts in small production and use.

Later efforts such as the Advisory Group on Reliability of Electronic Equipment (AGREE), dealing with all types of electronic component parts, did much to propagate interest and activities that would become the parts reliability discipline of the space program era.

The entrance into the first decade of space flights and operations, some ten years ago, witnessed a remarkable upswing in electronic parts reliability interest. Several schemes became available with mutual intent to force the parts reliability up and the specific part failure rate down. Some of the most notable approaches included the Ad Hoc Study Group on Parts Specifications Management for Reliability culminating in the Darnell Report; the early Minute Man techniques; the NASA Hi-Rel specifications; the Space Parts Working Group 38000 series specifications and the more current DoD Extra Reliability (ER) and Testing Extra (TX) documents.[2] There were problems, even severe problems, with some of these plans; however, they all contributed to parts reliability improvement. True, certain improvements were limited to only a betterment in reliability philosophy but even this helped. The general trend of this period was toward statistical qualification and 100% screening. Indeed the effects of the developed approaches and techniques of statistical qualification are largely with us today even though their total practice has been severely curtailed or limited in NASA circles by funding constraints. It is certain these austerity programs have precluded some reliability studies and procedure applications; however, the fund limitation also has forced reevaluation of reliability techniques in electronic parts. The effects and merit or degradation of the modified and/or simplified techniques have yet to be fully evaluated.

Certainly the recent parts reliability techniques, either of the full statistical philosophy or the more austere test-limited approach include at least two points in common. First, the duration of the reliability tests have been closely connected with and governed by the flight trajectory times. Secondly, the parts involved in the reliability considerations have been restricted to representatives of the specific devices of interest in a particular spacecraft project or mission.

The current entrance into the second decade of space missions yields a prognosis for electronic parts reliability techniques and procedures preempted by one word---change. This must occur due to the

magnitude of the differences in several factors. For example, the aforementioned time scale demanding more than 100,000 hours of trouble free operation between failure of electronic parts and/or assemblies. The vastness of this time factor is difficult to comprehend. However, it is fact that a great many of the current NASA parts personnel who today begin work on the Grand Tour electronic part devices will reach retirement age prior to the conclusion of that mission. The implication of this philosophy is certainly unique in the experience of most contemporary parts engineers.

The time scale will result in another interesting fact. Parts will become obsolete, manufacturers will cease to even make some devices, other parts bearing given EIA identification will grossly change, all within the Grand Tour project time scale. Yet even with such obsolescence, the availability and ability to supply appropriate replacement parts during preflight activities and the required engineering interest in the use of the out-moded devices must be extended through the mission. It will be an era of contradiction; very short lifetime to obsolescence in the case of microelectronics and extremely long lifetime trajectories.

The extremes in differences between the currently accepted MTBF expressions and time to obsolescence of electronic parts and the requirements for the long duration missions must cause corresponding extremes to occur in the philosophy of reliability as applied to the parts. It is apparent that the majority of the current procedures are inadequate. Those techniques employed to satisfy spacecraft reliability in the decade just concluded are unacceptable for long duration applications. There must be developed a new approach to parts reliability to meet the new demand. The outer planet equipments simply cannot be extensions of current assemblies. Rather, a new family of equipments are required, together with the processes, procedures, techniques and management controls for their construction and test.

## The Current Approach

The current approach to securing electronic parts of superior reliability for a given spacecraft is principally regulated by two factors: engineering requirements and economic constraints. Since these two factors are gross variables from center to center, specific examples experienced at the Jet Propulsion Laboratory will be considered in the examination of this point.

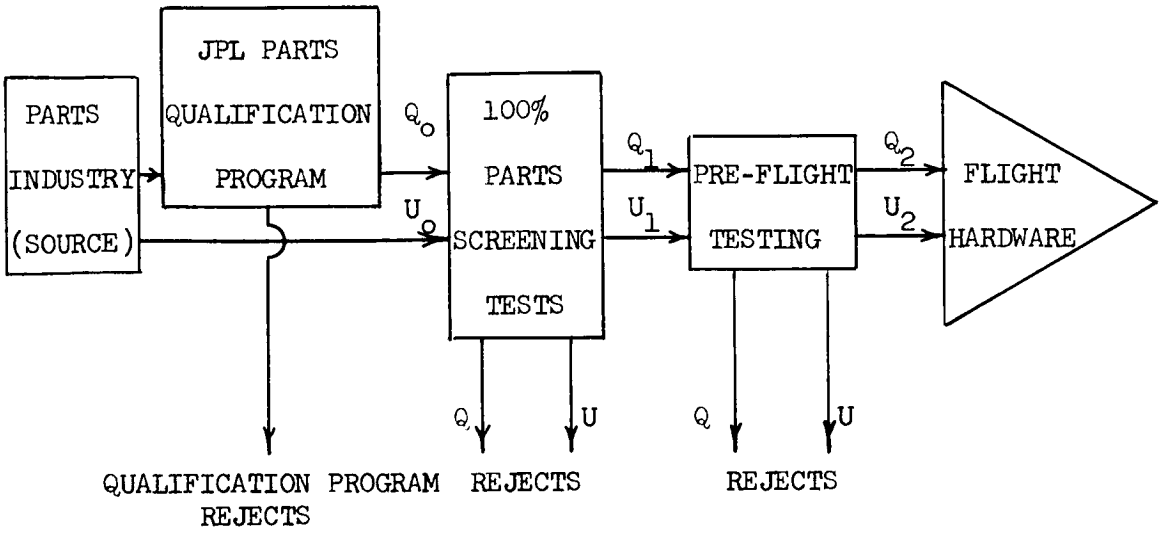
There are two general ways the effective reliability of a part may be improved above that of the parent population of unprocessed parts: (1) by improvement of the part at the procurement source through actual design and fabrication changes and (2) by screening procured parts and rejecting devices which are out-of-specification or anomalous to the given lot.

Actual improvement of the part at source involves the cultivation and development of procurement sources for certain selected parts. These are usually the high-use devices. The improvement process includes the standardization of selection, use of rigid specifications and their proof tests, careful failure analysis and appropriate quality control. Parts procured under these methods are designated qualified devices (Q parts). Unfortunately, qualified parts are not always sufficient to satisfy the requirements or preferences of equipment designers; therefore, the remainder of the procurement for a given spacecraft must be made up of unqualified parts (U parts).

The use of various screening procedures to improve the reliability of a part above that of the containing lot have received broad acclaim among spacecraft parts users. The demonstrated effectiveness of the available screens, coupled with such factors as the obsolescence of qualification data on Q parts and economic constraint, frequently result in the screening of all parts to be employed in a given project.

The screened parts, both Q rated parts and the parts of lesser reliability or U parts are usually subjected to the same screening acceptance criteria. This increases the reliability of both Q and U parts above the lot population but does not necessarily raise them to the same level of reliability. It has been observed at the Laboratory that the reject-rate of Q parts employed in the Mariner spacecraft are significantly below the reject-rate of U parts. This has not been proven directly by in-flight records since there have been no identifiable in-flight relevant failures prior to mission termination. However, the lower reject-rate ratio of Q to U parts has been demonstrated to correlate between the incident of pre-flight testing failures and actual screening activity measurements. Consideration of the qualification and screening approaches applied to the Mariner Programs are presented in Figure 1 for illustration. Details of the observations on the Ma '67 are specifically shown.

Mariner Spacecraft Qualification
and Screening Techniques




| | Q | U | Q | U | |
|---|---|---|---|---|---|
| Ma '67 | 22% | 56% | .1% | .23% | No known In-flight Failures |
| | Δ = 2.5 | | Δ = 2.3 | | |

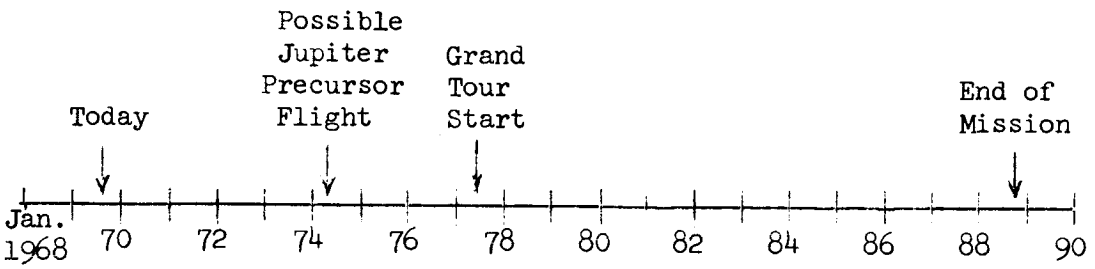
FIGURE 1

It can be seen that there was a factor of 2.5 more U parts rejected than Q parts during the Mariner '67 screening operations. During the Pre-Flight Tests, actual part failures provided additional Q of U yield demonstration in that there were 2.3 times more U failures than Q failures.

The comparison of the screening and pre-flight test data empirically indicates that the favorable ratio of Q to U existed throughout the spacecraft test history. Apparently the applied screening functions, coupled with the pre-flight tests raised the overall reliability of the parts to a level adequate to satisfy mission demands. The observed Q to U ratio for the Mariner '67 related to 12715 Q parts and 13071 U parts in the spacecraft. A different mix of Q and U parts would have modified the ratio. It is possible to envision a sufficient increase in the number of U parts over Q parts to a point where unsatisfactory reliability would definitely result. It is also true that mission requirements could be increased in such factors as trajectory time so that the currently available Q to U ratio would not produce an acceptable spacecraft.

## The Inadequacy of Current Practices

It has been stated that the present procedures employed to provide electronic parts for spacecraft are inadequate for long duration missions. This can be appreciated as the tentative time schedule for the Grand Tour mission is considered.




There are approximately eight years from January 1969 until launch of the Grand Tour flight in 1977. A precursor Jupiter flight is often considered as a developmental test of a Grand Tour spacecraft in 1974. It is anticipated that the two spacecraft would be very similar in design and function. The long expanse of time prior to launch coupled with the desire to fly similar craft presents difficult electronic parts problems.

Most contemporary spacecraft are fabricated from electronic parts possessing some degree of evaluation pedigree. This knowledge is obtained from test information obtained on parts essentially identical to those that will be fabricated into the flight equipments. Consideration of the time scale of the Grand Tour mission clearly indicates a discrete part pedigree, secured in the early 70's, will be most difficult to maintain over the years until launch.

As pointed out, it is a must to improve the reliability of the parts but what parts are the specific areas of interest? Many types of discrete devices have been manufactured for several years in their present forms; however, there is less than a satisfactory guarantee that these particular parts will be available for the Grand Tour build or in certain cases even the earlier 1974 precursor flight. Microelectronics presents a worse problem in that this discipline, due to it's extremely dynamic development, ensures the microelectronic devices of even next year to be different than the current models.

It is currently unreasonable to initiate any appreciable stockpile of parts for future use either actually or in the form of electronic part design documents. Yet, the flight parts, when the appropriate time comes, must surely have been subjected to qualification and screening tests. To stockpile is to err from obsolescence, to fail to

select parts in a timely manner will preclude the marked improvement in reliability that is mandatory for the success of a long duration mission.

The microelectronic medium scale (MSI) and large scale (LSI) integrated circuits appear particularly attractive to long duration operation. Demand for redundant circuits, minimum power consumption, compact equipment size, decreased weight and improvement in reliability potential are important relevant factors. However, the desire to apply MSI and LSI circuitry poses a huge economic problem. The selection and qualification tests of these complex circuits are expensive. Coupled with this expense today, is the very likely demand to also require considerable research and/or development support to a given type circuit prior to securing the required device characteristics for a particular need. The total cost of the test, research and development program may easily become so great that commitment to a specific circuit involving these costs may occur years before the technology of the microelectronic discipline and the spacecraft requirement are ready for this freeze.

## Possible Problem Solutions

The Thermoelectric Outer Planet Spacecraft (TOPS) R&AD Advanced System Technology Project, currently operating at the Jet Propulsion Laboratory, is a system design and development effort. TOPS is intended to provide an understanding of outer planet spacecraft subsystems and their interactions, for use in anticipated spacecraft designs. TOPS is primarily concerned with advanced development of those disciplines particularly related to long duration spacecraft missions. It can be appreciated that electronic parts engineering, including the selection, testing, application and acquisition of parts, is one of these disciplines. The major emphasis to be focused on the parts effort and activities are well defined in the TOPS Program Policy Document. Three distinct types of part activities, consistent with available resources, are included. These are: the discrete devices, standard or off-the-shelf integrated circuits and non-standard or custom-built integrated circuits of the MSI and LSI complexity. The "standard" and "non-standard" IC's have been separated into two distinct categories. This split recognizes the differences existing between IC's built for the market by the manufacturer and IC's custom-built specifically for a given system. The division also recognizes the dependence of subsystem design on part characteristics and manufacturing processes.

The Discrete Component Parts include the more conventional devices such as diodes, capacitors, transistors and resistors. In order to guide parts selection consistent with mission constraints, a TOPS discrete parts list is being developed at the Laboratory. Initially the existing JPL Preferred Parts List is being used as the guide for

acceptable candidates for this parts list. The candidates have been compiled into a TOPS Recommended Parts List (RPL). The initial list (see representative page from TOPS/RPL, Table 1) presents three types of information for each listed candidate. The threshold of radiation damage is listed; derating information estimates for five and ten year missions are presented and a best estimate of the availability of the part ten years in the future is detailed. It is fully recognized that the initial information must be superceded as improved data and closer estimates become available. Therefore, the TOPS/RPL will be continually revised and updated. Appreciating the RPL is very limited in the detail it can contain, appropriate application notes relating to the RPL listings will be prepared. The application notes will include rationale for derating factors, recommended circuit applications and restrictions and environmental operating limits critical to performance of part types.

Preliminary work is presently underway on application notes pertaining to extra long life observations on some of the listed RPL candidate parts.

The RPL and its supporting application notes are being produced to serve as guide documents to the TOPS subsystem design engineer. It is important to understand that two phases of TOPS investigation and development are progressing simultaneously at the Laboratory. First, there are the basic endeavors intended to ultimately produce a spacecraft capable of executing a long duration mission of the Grand Tour type. The electronic parts work associated with this intent includes the production of the RPL of tomorrow. The list of today is but the forerunner of that improved and updated version. Secondly, considerable circuit feasibility and characteristics study is being conducted by the design engineer. The electronic parts discipline must also support this current activity, seeking to guide yet not unduly restrict; seeking to select yet allowing adequate freedom; and all the while promoting consultation and appropriate record keeping for all parts used in experimental work.

The TOPS Policy Document requires the design engineer and the parts specialist to work closely together in production of a parts list for the subsystem in question. The design engineer shall provide a list of the parts employed in the feasibility experiments. The specialist shall rate these parts as: (1) approved or acceptable, (2) inferior, wherein usage constitutes high risk or (3) parts with insufficient supporting information. Appropriate controls, planned to prevent the use of any inferior parts, will be enforced. Use of parts with insufficient supporting information will also be minimized in the preliminary studies. Prior to the beginning of any flight-rated equipment fabrication, all inferior parts and those with insufficient support information must necessarily be eliminated through appropriate tests, substitution and/or parts control.

Candidates for the
Thermoelectric Outer Planet Spacecraft
Recommended Parts List

CAPACITORS

| Part No. | Vendor | Type | Radiation Limits (Dose for 11 years) | Derating Factors 5 years | Derating Factors 10 years | Availability 1972-1977 | Comments |
|---|---|---|---|---|---|---|---|
| | | | nvt | Percent | Percent | | |
| BSF | AVX | Ceramic | $1x10^{18}$ | 50 | 60 | Yes | |
| BLF | AVX | | $1x10^{18}$ | 50 | 60 | Yes | |
| BLS | AVX | | $1x10^{18}$ | 50 | 60 | Yes | |
| BSS | AVX | | $1x10^{18}$ | 50 | 60 | Yes | |
| FLA305 | AVX | | $1x10^{18}$ | 50 | 60 | Yes | |
| 287 | MAI | | $1x10^{18}$ | 50 | 60 | Yes | |
| CJ | CEN | | $1x10^{18}$ | 60 | 70 | Yes | |
| CKR | AVX | | $1x10^{18}$ | 50 | 60 | Yes | |
| VKR | VIT | | $1x10^{18}$ | 50 | 60 | Yes | |
| HMC | AVX | | $1x10^{18}$ | 50 | 60 | Yes | |
| CYFR | CGW | Glass | $1x10^{18}$ | 40 | 50 | Yes | |
| HRDM | ELM | Mica | $1x10^{18}$ | 40 | 50 | Yes | |
| CTM | GEC | Mylar | * | 50 | 60 | Yes | Possible Moisture Problems |
| 617G | TRW | | * | 50 | 60 | Yes | |
| 118P | SPR | Met Paper | * | 50 | 60 | Yes | Testing at Higher Flux Desirable |

* Not available at this time

TABLE 1

The Standard or off-the-shelf integrated circuits will be treated in the same manner as discrete electronic parts except that additional emphasis will be placed on common procurement for TOPS hardware and inprocess QA monitoring at source. In subsequent editions of the TOPS Recommended Parts List, standard IC's will be listed. The initial RPL does not include IC's or magnetic items due to limitation of available information.

Medium Scale and Large Scale Integration technology application of either the monolithic or hybrid approach raises anew the question... what is the optimum relationship of the circuit design engineer to the parts specialist? There are those functions connected with the circuit feasibility, application and operational characteristics that are certainly the unquestioned prerogative of the design engineer. There are also those functions of the complex integrated circuit which pertain to the geometric layout, mask design, fabrication processes and methods and reliability tests that are the responsibility of the parts specialist. The functions of either the design engineer or the parts specialist are of sufficient magnitude to demand full attention without dilution from the other. To further complicate the situation, it is recognized that the line of demarcation between many other responsibilities relating to IC's is not clear but demands the combined attention of both the parts specialist and the circuit designers.

Through the era of the discrete parts and extending into the conventional or off-the-shelf IC period of today, the functions of the parts specialist and design engineer have tended to be well separated. The design engineer has conceived the electronic circuit, utilizing the assistance of the parts specialist to suggest parts providing given operating characteristics. The parts specialist has been engaged in supplying this support with parts of optimum reliability, within the project constraints, for the assembly in question. Even under the best conditions both parts specialist and design engineer have been constrained to employ devices totally controlled by the manufacturer. Close team effort between the design engineer, the parts specialist and the manufacturer, although very desirable, has existed on a far less than optimum basis.

The advent of the customized complex integrated circuit demands a revolutionary change in design engineer/parts specialist relationship to each other and to the parts manufacturer. The MSI and LSI require combined team work of the design engineer, parts specialist and manufacturer if circuits are to be produced in an economical and timely manner and in the anticipated limited quantity required by the NASA spacecraft market.

Recognizing the need for team action in the TOPS MSI/LSI effort, JPL has instituted a design team charged with the responsibility to successfully interrelate the three functions. The team members, each representing segments of technical divisions plus a member from the parts engineering activity, are working together to select circuits of common interest in order to facilitate maximum standardization of usage compatible with the subsystem requirements. Once selected, the common interest circuits, as well as the more unusual devices, will be subjected to an intense probe as to their availability, degree of development yet required, manufacturing processes and operating reliability in the environments of concern. It is appreciated that operating characteristics data of the type produced during actual long duration life test will not be available on most customized circuits. Furthermore, special test information involving long life studies and unique environments such as radiation will not be available. It is the plan of the JPL team to utilize design appraisal techniques, the indepth investigation detail available with the Scanning Electron Microscope and other special studies and tests to arrive at a position of recommendation for circuit types, manufacturers acceptable circuit fabrication procedures, procurement methods, qualification approaches and screening controls.

The MSI/LSI devices of today incorporate a remarkable variety of microelectronic fabrication techniques. These include bi-polar semiconductors, metal oxide semiconductors (MOS), junction field effect transistors (J FET), micropower devices and hybrids. With this present array of techniques, certain approaches will have a greater ability to provide long duration operation than will others. The TOPS design team is concerned about this problem particularly as radiation and long life pertain to the various types of MSI/LSI construction. The long duration spacecraft currently on the design boards depend on radioistope thermoelectric generators (RTG) for the power source. The integrated dose from the RTG over the long duration trajectory is viewed as a primary reliability problem to some microelectronic devices, depending on their fabrication processes and/or procedures. The magnitude of the problem is not yet fully understood; however, it is apparent that a margin of safety or guard band has not been demonstrated to exist between the potential MSI/LSI acceptable total dose limits and the RTG integrated dose over the intended trajectory period. It is the intent of the TOPS parts team to pursue the radiation problem through analysis of existing data, special tests as required and subsequent control and selection of appropriate discrete and microelectronic devices.

It is appropriate to consider one other problem being encountered by the TOPS team. The pursuit of circuits for application in breadboard and feasibility assemblies tends to result in a problem capable of far reaching consequences. In the meeting of schedules requiring the fabrication of demonstration equipment in the near future,

inappropriate amounts of development effort can be conducted on a given MSI/LSI. Later indepth study may indicate the development produced a circuit which is incapable of reliable performance in the long duration mission environment. The TOPS team is continually confronting this problem wherein the team must decide the appropriate degree of development in the light of other supporting information concerning the given circuit.

## Conclusions

Consideration has been given to the vastly extended time scale and requirements of the outer planet missions. It was pointed out that the current electronic parts philosophy just does not include many of the operational aspects of long duration flights. The reliability requisites of longevity coupled with the demonstrated value and need for parts qualification and screening plus the rapidly changing nature of the microelectronic industry demand that new and unique approaches to parts reliability be developed.

The fact that changes in the current approaches are mandatory certainly does not preclude the continued use of contemporary methods for current requirements. However, new and improved approaches must evolve because the parts engineer is aware of and sensitive to the need for superior methods for tomorrow.

Many of the reliability problems associated with long duration missions are certainly not new and can be identified. Screens need continual improvement; the old statistical qualification effort needs reevaluation in view of the more dynamic design appraisal approaches; and parts control methods need appropriate application to preclude reinventing-the-wheel on each new project. Achievement has been made in these problem areas but more work needs to be done.

The problems that are more unique to outer planet missions, such as those affiliated with radiation and longevity, tend to be insidious, as many of the factors are not yet well understood. Radiation problems arising from a combination of the natural environments and the flight RTG will be troublesome. The amplitude of these difficulties are not understood due to variables in spacecraft design, parts to be employed, the RTG size and location and duration of the trajectory.

The period of time prior to and including the outer planet flights present still other special problems to parts engineers. These pertain to part obsolescence, unidentified manufacturing process changes and procurement control.

It certainly is not claimed that solutions to many of these problems are in sight at the Laboratory or else where for that matter.

However, effort is underway at JPL to understand the magnitude of the parts problems relating to long duration space missions. Perhaps the most encouraging achievement to date at JPL, in this difficult area, is the demonstrated fact that engineers with grossly diverse interests can and will work closely together as a team to solve mutual problems.

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# CONSTRUCTION ANALYSIS - A TOOL TO EVALUATE PARTS INTEGRITY

By William R. Rumpza
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Huntsville, Alabama

## Introduction

Ten years ago the high frequency of design inadequacies and manufacturing defects in parts led to the institution of 100% screen testing. While the value of parts screening has been conclusively established, a significant quantity of defective parts escape detection and result in higher test level or mission failure[1]. For new spacecraft with mission durations near or in excess of one year, it is imperative that all manufacturing and design defects leading to part failures be removed. Part Construction Analysis has the potential of contributing as much to the part "defect removal" goal as did screening during its inception and growth to maturity.

"Construction Analysis" is defined as the detail analysis of a representative sample of parts from each procurement lot for the purpose of identifying part design, process, and workmanship defects which may degrade the functional parameters of the part or cause catastrophic failure.

This paper presents an assessment of construction analysis as a tool to identify potential failures. The analysis functions performed on each part type; the laboratory equipment needed to perform an adequate job; and the analysis observations made on diodes, microcircuits and transistors during research and project support functions are discussed. The observations are recorded in terms of the types of defects observed and then categorized into areas that the part manufacturer should improve. An overview of the effectiveness, limitations and cost effectiveness of the construction analysis tool is also presented.

## Approach

The use of construction analysis on the Saturn S-IC program was a result of evolution rather than revolution. That is, an Engineering Quality Assurance (EQA) program was in existence where a predetermined sample of equipments or assemblies were completely "torn down" to evaluate their quality. The success of EQA in reducing hidden defects and obtaining corrective action from suppliers led to the present parts construction analysis program.

Parts were selected at random from the parts storage area for construction analysis per the flow chart shown in Figure 1. This chart shows analysis functions necessary to detect the majority of part design,

process or workmanship defects. The typical or more common defects are listed to the right of each functional block. A more complete list is available in other references[2,3,4]. The analysis sequence was altered to accommodate unique features of the various parts analyzed. For example, package gas analysis was not done on parts where the cavity was extremely small (certain types of diodes). When destructive test or functions were performed, the construction analysis part lot or sample was divided into sub lots so that the main flow of functions could always be accomplished.

State-of-the-art developments in equipment and methods have expanded the construction analysis approach already defined. These developments were not used in establishing the data in this report, but because of their potential and immediate impact on improved defect detection, it is felt they should be mentioned.

The first development may be appropriately called "construction/circuit analysis" and is limited to microcircuits. This analysis examines the relationship of electronic parameter behavior to surface cosmetics, circuit layout and fabrication techniques. It includes a voltage probe of circuit nodes; isolation of all circuit elements (diodes, resistors, transistors); electrical characterization of each circuit element (25°C, +150°C); electrical characterization of each parasitic element (isolation junctions, and bulk resistances); construction of a computer model using measured characteristics; computer analysis of non-measurable electrical characteristics (branch currents, hot spots, etc.); and current analysis effects of burn-in circuit operation (power distribution, voltage bias, current distribution). This type of analysis gives an excellent insight into microcircuit weak points and their response to actual operating conditions.

The second development "construction/microprobe analysis" is still in the experimental stage because of the newly introduced scanning electron microprobe instrument. Visits to NASA Centers, such as Goddard and ERC, and Research labs, such as JPL, indicate that the ability to observe defects has outstripped the ability to interpret the impact of what is seen[5]. The very broad depth-of-field available when operating in the scanning electron microscope mode, plus the probes capability to quantitively and qualitatively analyze material as well as perform many other functions beyond the scope of this paper will advance the construction analysis technology greatly. Preliminary data indicates that the use of the instrument may only increase the cost of the original approach used by fifty percent.

## Laboratory Requirements

Most of the equipment required for construction analysis is found in a basic failure analysis laboratory. Figure 2 lists this equipment and the approximate dollar investment. The asterisk items are additional

equipment which would be found in a well equipped laboratory. It must be emphasized that all the items on the basic equipment list are absolutely necessary. An inadequate laboratory can introduce analysis errors. For example, lack of sectioning equipment can easily result in an induced cracked die or completely cutting through of a defect site. Inadequate microscopy will completely overlook major defects.

Potential users of construction analysis should evaluate carefully whether a laboratory is cost effective based on their programs and volume of work. In many cases more reliable analysis are obtained from an established laboratory because it has the experienced technologists needed to properly use the equipment and to correctly interpret the often subtile test results.

## Analysis Results

Construction analysis was performed on forty-three lots of diodes, microcircuits and transistors. Thirty-two lots will be discussed in this section with the remaining eleven lots discussed in the "effectiveness section" because the parts have a different history and were procured in a different time frame. "Lot" as used here means a sample of parts subjected to construction analysis. The definition of data presented here was collected by the failure/construction analysis specialists in support of company sponsored research, contracted research programs, and the Saturn S-IC project.

### Part Types Analyzed

The types of parts analyzed included a cross-section of the semiconductor industry. No part manufacturer produced all defect free lots; however, four manufacturers supplied at least one lot free of defects. Item A of Figure 3 shows that thirty-two lots consisting of fifteen lots of diodes, eight of microcircuits, and nine of transistors were analyzed. Most of the parts analyzed were selected for use three or four years ago. These lots contained a total of one hundred eighty-eight parts with the lot size varying from two to twenty. The median lot size was about five. Fifty percent of the diode lots were made up of devices previously screened to a specification which included reverse bias burn-in and internal inspection.

### Defect Quantities

The origin of defects is categorized into three general areas - design, process or workmanship. Figure 4 shows twenty-six types of defects classified into these areas. The design defects are those which are expected to have an extremely high incidence of reoccurrence because the basic design of the part is at fault. Process defects are those which can result from lack of adequate process definition, process

control or inspection. Workmanship defects are those defects resulting from human error.

A summary of the defect types reveal that 15.3% are attributable to design, 38.4% to process and 46.3% to workmanship. These data emphasize where part procurement controls should be improved to maximize corrective action to prevent defects. It should be noted that corrective action should be based on the types of defects rather than the total number of defective parts.

Item B of Figure 3 summarizes the defects observed during these analyses. A total of one hundred fifty-nine defects were recorded, in some cases there were as many as six defects per part. As would be expected, the highest number of defects occurred in microcircuits because of their complexity. One hundred fifteen parts (61%) revealed at least one defect and twenty-eight lots (87%) analyzed contained at least one defective condition. Of the screened diodes analyzed, 28% contained a major defect.

Classification of Defects

To gage the reliability impact, each defect was classified as shown in Item C of Figure 3. Those parts with a degree of defect which would cause emminent part failure were classified as critical. Eleven (9.5%) of the defective parts fell into this category. Typical defects classified as "critical" are shown in Figures 5A, B and C. The critical microcircuit defect shown in Figure 5A is a corroded aluminum interconnect. It can be safely assumed that the corrosion will continue and eventually result in device failure. The transistor (SCR) defect in Figure 5B shows a bond overhanging the cathode gate junction. The slightest accumulation of contaminants will cause a short across the junction. Figure 5C depicts a severely misplaced whisker on a diode making it very susceptable to mechanical failure. Incidentially it was a screened part.

Defects classified into the "major" category were those severe enough to cause probable early part failure under normal operating conditions. Figures 6A, B and C contain pictures of typical major defects. Sixty-five parts (56.5%) were classified as having major defects. Figure 6A is a photograph of severe metallization tool damage making the device very susceptable to a "mass transport"[6-7] failure in its normal operating condition. The major transistor defect in Figure 6B shows a grossly misplaced emitter bond - weakening the bond plus creating a potential short. In Figure 6C a diode whisker is misaligned to the point of creating a major hazzard to its operating life.

The degree of part defect which would probably not result in failure unless used in severe environments was classified into the minor

category. Figure 7A contains a picture of a resistor mask defect on a microcircuit. Figure 7B shows a transistor with a severe mask defect; however, no apparent parameter degradation was noted during analysis. Figure 7C depicts small voids in the die piece parts - little impact. Thirty-six parts (31.3%) exhibited this degree of defect.

The only defects which were classified into the no impact category were extremely small "voids under the die" which would slightly interfere with the heat dissipation of the devices.

## Effectiveness

A review of the types of defects and their high frequency of occurrence indicates that their prevention, where possible, would greatly improve reliability. That the prevention of defects is possible can be illustrated by comparing analysis data on parts where the manufacturer was not aware that construction analysis would be employed, to the analysis data on parts where the manufacturer was so informed at the time of part procurement. This comparison is made in Figure 8 where fifteen lots of diodes (these lots comprised a portion of the original thirty-two lots discussed in Figure 3) are compared to eleven lots of diodes. The data shows that where the manufacturer (15 lots) did not know that construction analysis would be performed, there were defects in 100% of the lots and in 52% of all parts with .53 defects per part. In the case where this information was given to the manufacturer (11 lots) at the time the parts were ordered, only 27% of the parts had defects with only .30 defects per part. This indicates that defining to the part vendor the detail part construction analysis procedure that will be performed by the user, gives the vendor the incentive to improve his quality control. It should be noted that the eleven construction analysis lots were procured approximately two years after the original fifteen lots were analyzed. Therefore, some of the reduction in defects can be attributed to more mature manufacturing lines. However, the impact on the conclusion is considered minimal because of continuous process changes occurring in the semiconductor industry which result in the introduction of new defects.

The ability to identify a defect through construction analysis is largely dependent on where it is located within the specimen being analyzed. The three most common categories of failure mechanisms are; bulk, surface and mechanical defects. Of these, mechanical defects are the major contributor[8].

Because bulk defects are beneath the surface, they are extremely difficult to isolate by construction analysis and are not discussed here.

Most surface defects, such as oxide pin holes, faulty oxide growth, poor mask registration, and use of a damaged mask, are readily

identifiable. More difficult to observe are over or under etch and incomplete removal of oxide at the contact windows. Contaminants left under the oxide during device manufacturing are almost impossible to detect.

Mechanical defects are those that manifest themselves by visual observation before and after mechanical and electrical stress and are ideal for construction analysis detection. These defects include bond weakness, metallization adherence, chip adherence to the header, integrity of package seal, metallization scratches, cracked chip, intermetallic formation, mass transport of metallization, corrosion, cracked oxide, wire dressing and a host of others too numerous to list.

Determining the exact degree to which a part defect is detrimental to its performance is very subjective and difficult to define. The answer lies largely in discussing with the part manufacturer the details of his part design, the degree of defect he intentionally passes through his line (because they theoretically have no impact) and the kind of inspection tools he uses.

It must be emphasized that construction analysis will only identify the defects, not remove them. In order to be effective, corrective action must be instituted with the part manufacturer by mutually identifying the kind of design, process, or workmanship standard changes required for correction. Where standard changes are uneconomical, effective selection criteria must be imposed to remove the observed defects from part lots intended for spacecraft use.

Construction analysis should be applied to the product coming off of the production line prior to screening, since the reduced incidence of partially screenable defects may prevent their detection. The one exception is where the manufacturer is performing a rigid preseal inspection and screening. In this case, construction analysis is being used to monitor the inspection and screening being performed. The observation has been made[1] that the quantity of defects will vary with each lot procured, even from the same manufacturer, therefore, construction analysis must be performed on a sample of every production lot. Since construction analysis is destructive and must always be performed on a lot sampling basis, the probability of detecting a defect is limited to the statistics governing the sample size selected. The sample size selected is dependent on the lot size purchased and the cost of the part. Lots should be rejected if any defects are detected in the critical or major classification.

## Cost Effectiveness

Based on experience, the cost of performing construction analysis is broken down by basic functions in Figure 9. It indicates that typical

construction analysis will cost between thirty-six and sixty manhours based on a sample size of five parts. With proper planning of new projects the application of mechanical stresses could be done simultaneously on several lots. The time charted in Figure 9 could be reduced by one-third if construction analysis lots were submitted in groups.

The cost impact of failures after the parts have been incorporated into higher assemblies can be evaluated using Figures 10A and B. Relative cost figures for the removal and repair of hardware at each progressive test level were published at the 67 Annual Reliability Symposium[9]. Using these cost figures as a guide, relative cost figures per failure were estimated to be fifty hours at component qualification test level, five hundred hours at component flight acceptance test level and five thousand hours at spacecraft flight acceptance test level. Figure 10A shows the costs of removing failures at each of the defined test levels. In Figure 10B are the cost figures for performing construction analysis for a spacecraft using between one hundred and six hundred part types. This assumes that construction analysis will be performed once per part type.

Applying figure 10A to the Lunar Orbiter failures of five electrical/electronic parts failures during component qualification testing; fifty-three during component flight acceptance testing; and seven during spacecraft flight acceptance testing; the total cost for identifying, removing, analyzing and correcting the failures would be 61,750 manhours. Assuming that construction analysis was only ten percent effective in removing failures, construction analysis could have been performed on one hundred twenty-three part types without any additional costs. Putting it another way, if the spacecraft used three hundred part types, construction analysis would have to be only twenty-four percent efficient in removing defects to be cost effective. This is not unreasonable because a review of the construction analysis data shows that sixty-seven percent of all defective parts fall into the critical and major defect classifications. If this data is representative of parts being produced on typical commercial and military lines, a twenty-four percent improvement in the number of failures at higher component test levels could readily be achieved. The only exception wherein construction analysis may not be cost effective would be if very rigid preseal inspection were instituted at the part vendor. However, even in this case, it must be remembered that a complete construction analysis includes many destructive tests and, therefore, provides a much higher order of confidence in establishing that all defects have been detected. Bond strength, metallization adherence, oxide cracks, contamination in contact windows, and microcracks in the die are only a few items which cannot be evaluated by preseal inspection.

## Conclusions

The number and gravity of defects identified in this review indicate that construction analysis is definitely capable of evaluating part quality and design integrity, and if implemented into a project early, it can substantially improve system reliability with little or no cost impact.

Construction analysis presents some specific design advantage in that it actually changes hardware quality by identifying and demanding the removal of defects; it is cheap for what can be learned; it is very fast and has little or no impact on schedules; and it makes available part manufacturing data which is normally not available to the user.

When procuring parts for long life spacecraft, all lots of parts which exhibits serious defects detected by construction analysis must be rejected until (a) the part manufacturer institutes corrective action to prevent recurrence of defects at which time a new lot is procured or (b) a selection criteria is established which will remove the noted defects or (c) a screen is defined which will remove the defects with 100% confidence.

Construction analysis is an effective tool in making part procurement decisions. It can be used to:

a. Identify the weak points in part design or manufacturing process.
b. Identify stresses which will accelerate the weak point so that correct part screening can be defined.
c. Define the design margin of the parts.
d. Assess risk by identifying, when a failure occurs, probability of the remainder lot having the same defect.
e. Evaluate the integrity and compatability of materials used in a device.
f. Monitor vendor production so that you know when processes have gone out of control.

Recommendations:

Construction analysis as a tool for evaluating part integrity is effective but is still in its infancy. Much of the data used in this paper is subjective. It is considered essential to the success of long term space missions that research be sponsored by both Government and industry to:

1. Better define the degree of defects and their relationship to impact on reliability.
2. Create a construction analysis/failure analysis handbook which documents methods, procedures, and reject criteria associated with identifying individual types of defects.

## Acknowledgements

Appreciation is extended to Mr. Leo Buldhaupt for his encouragement in writing this article and to Failure/Construction Analysis Specialists, Norman Bean and Len Roth, for their technical assistance.

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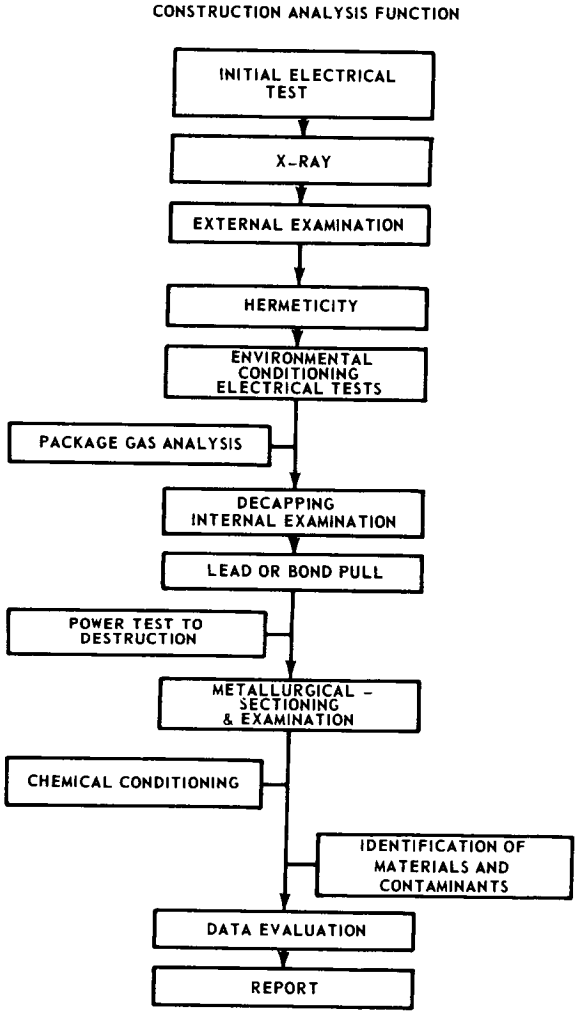


TYPICAL DEFECTS SOUGHT OR EXAMINED FOR

Voltage drop, leakage current & other pertinent parameters based on part specifications

Piece part alignments, voids in piece part construction, foreign particles

Cracks, pitts, crazing in glass seals, external contaminants, defects in external joints and/or welds

Determine leak rate-gross and fine

Test materials compatability anf interface strength, mechanical integrity

Analyze internal gas composition for corrosive or detrimental contents

Die cracks, alignment, contamination, welds joints, metallization, etc. (See Supplement No. I)

Determine lead/bond electrical/mechanical strength

Power design margin

Cracks, voids, piece part, alignment, bond/weld integrity, contaminants, , diffusion depths, material thickness, etc.

Used for: msterial cleaning, removal and enhancement. (See Supplement II)

Identification of materials and contaminants, composition

SUPPLEMENT

I. VISUAL EXAMINATION

masking: registration, residue inclusion, exclusions
oxide defects
smeared interconnects
corrosion (installation)
bonds - formation, placement
particles
lead dressing

II CHEMICAL CONDITIONING
1. Metallization Removal
2. Glass passivation ($Si^{O}$ 2) removal
3. Silicon etch for a) identifying junctions and depth
b) stacking faults
c) other bulk defects

FIGURE 1. CONSTRUCTION ANALYSIS FLOW DIAGRAM

| EQUIPMENT | ESTIMATED COST |
|---|---|
| Digital Voltmeter | 2K |
| Low and High Voltage Supplies | 2K |
| Patchboard & Test Fixtures | .5K |
| Low & High Current Meters | 1.5K |
| X-Ray Equipment | 23K |
| Film Processing Equipment | 1K |
| Stero Wide Field Microscope | .5K |
| Photographic Equipment | 2K |
| Fine - Helium Leak Tester | 5K |
| * Radiflo Leak Tester | 10K |
| Gross - Hotplate Oil & Containers | .5K |
| Environmental Chamber (2) -65 +300°C Capability | 3K |
| *Vibration | 20K |
| Shock | 5K |
| Centrifuge | 20K |
| *Package Gas Analysis System | 22K |
| Microscope (25-1000X) | 3K |
| Photographic Capability | 1.5K |
| Metallograph | 13K |
| Bond & Lead Pull/Bond Fixtures | 1K |
| Metallurgical Grinders, Polishers, Potting Components, Cutoff Machines, etc. | 15K |
| Supply of Chemicals, Containers, Fume Hoods, Sink, etc. | 3K |
| *Electron Microprobe | 130K |
| Spectrographic Equip. | 10K |
| Delidding Tools, Recorders, etc. | 2K |
| TOTAL COST | 296.5K |
| Basic laboratory cost is | 114.5K |

FIGURE 2. CONSTRUCTION ANALYSIS LABORATORY EQUIPMENT REQUIREMENTS

A. PART TYPES ANALYZED

| | |
|---|---|
| Number of lots subjected to construction analysis | 32 |
| Total parts analyzed | 188 |
| Number of part numbers | 26 |
| Number of part vendors | 16 |

B. DEFECT QUANTITIES

| | |
|---|---|
| Number of lots with defects | 28 |
| Total number of defects observed | 159 |
| Total types of defects | 26 |
| Number of parts defective | 115 |
| Defects per Part | .85 |

C. CLASSIFICATION OF DEFECTIVE PARTS

| | |
|---|---|
| Critical (part failure emminent) | 11 |
| Major (part failure probable) | 65 |
| Minor (part failure not probable) | 36 |
| No impact (part acceptable with defect) | 3 |

D. PERCENTAGES

| | |
|---|---|
| Lots defective | 87% |
| Parts defective | 61.1% |
| Parts with critical defects | 9.5% |
| Parts with major defects | 56.5% |
| Parts with minor defects | 31.3% |
| No impact defects | 2.6% |
| Defects per Defective Part | 1.3% |

FIGURE 3. ANALYSIS RESULTS

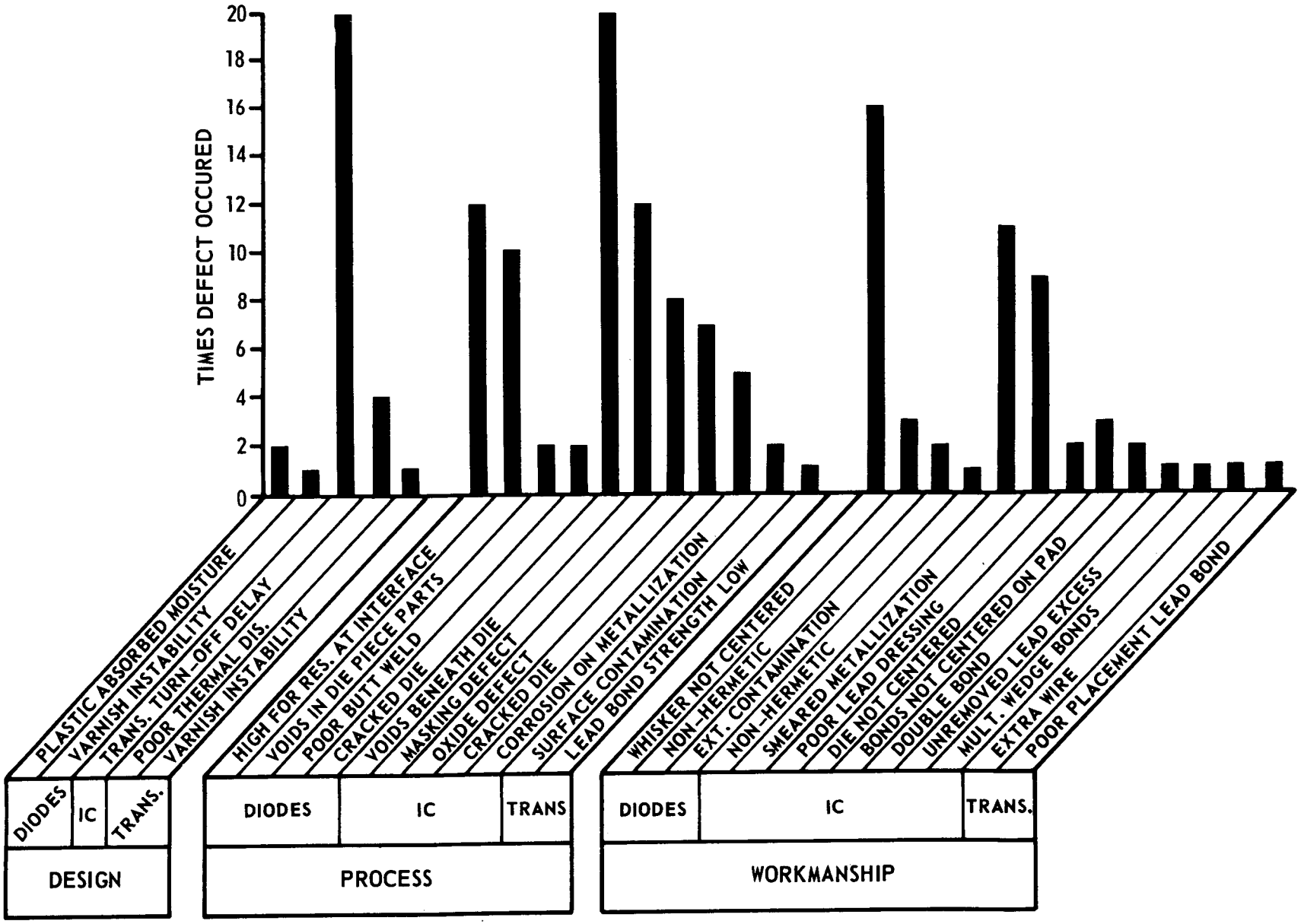


FIGURE 4. CATEGORIZATION OF DEFECTS

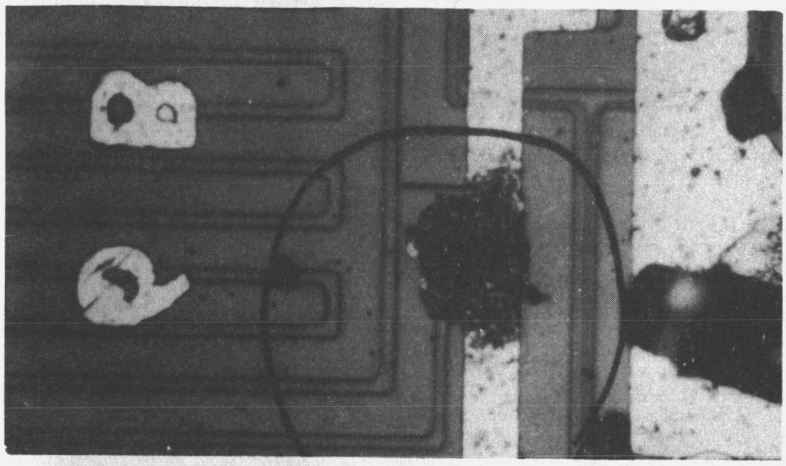

Figure 5A: CRITICAL MICROCIRCUIT DEFECT - CORRODED ALUMINUM INTERCONNECT

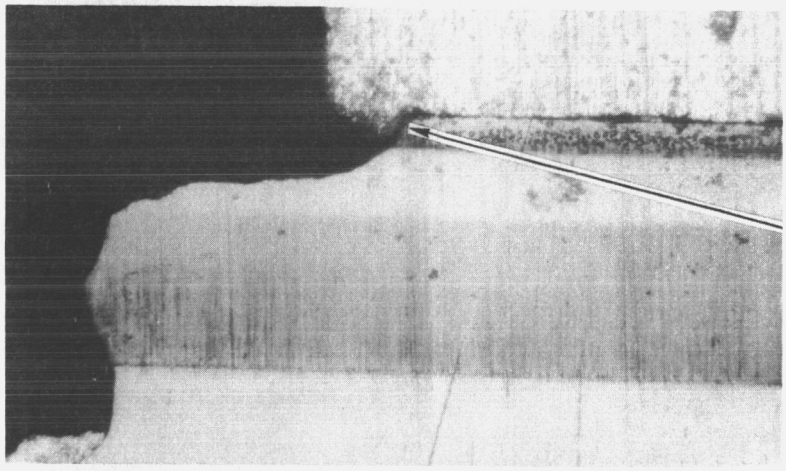

Figure 5B: CRITICAL SCR DEFECT BOND OVERHANG

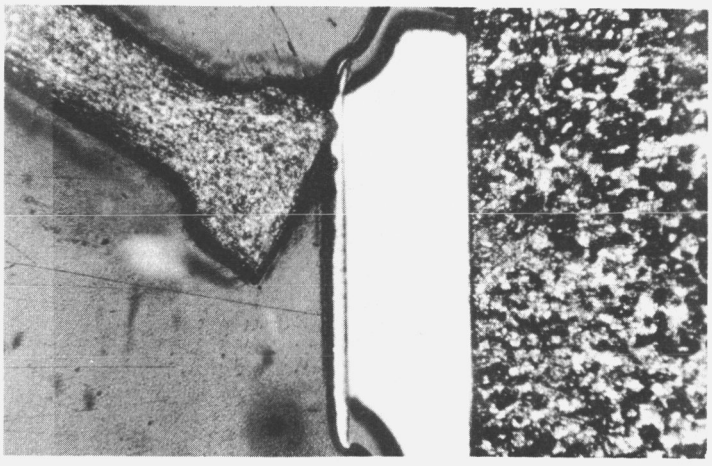

Figure 5C: CRITICAL DIODE DEFECT - SEVERELY MISPLACED WHISKER

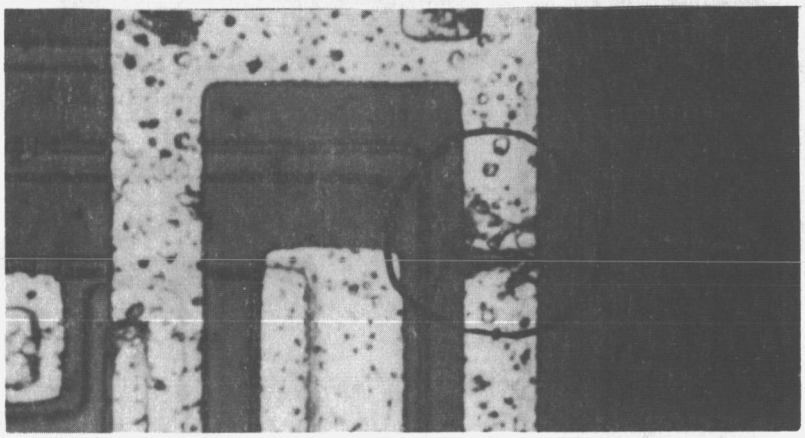

Figure 6A: MAJOR MICROCIRCUIT DEFECT - SEVERE METALLIZATION TOOL DAMAGE

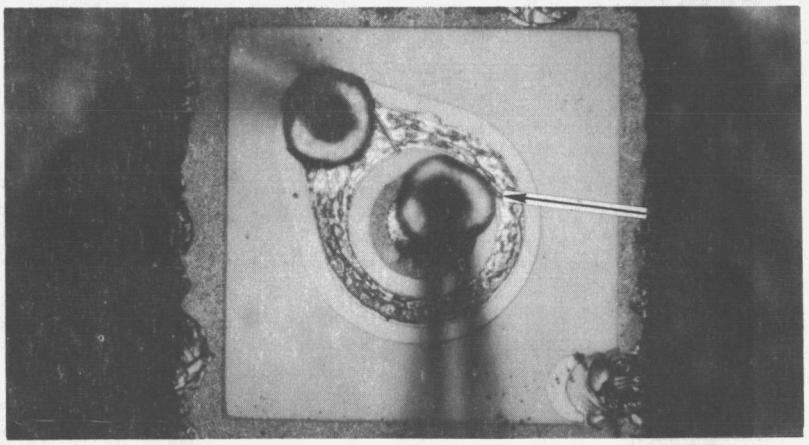

Figure 6B: MAJOR TRANSISTOR DEFECT - GROSSLY MISPLACED BOND

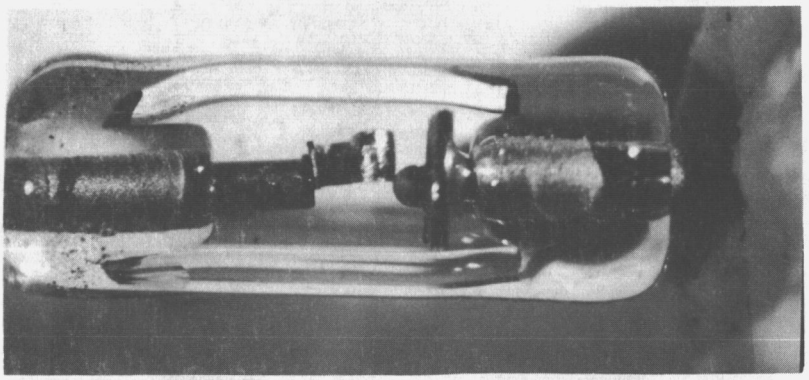

Figure 6C: MAJOR DIODE DEFECT - WHISKER MISALIGNED

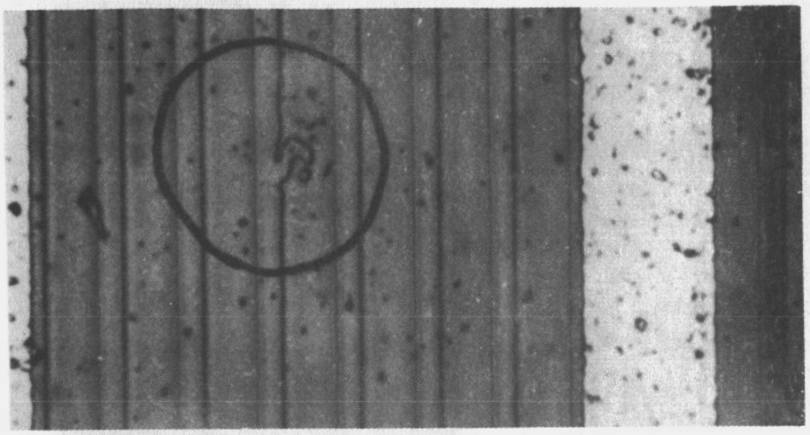

Figure 7A: MINOR MICROCIRCUIT DEFECT - RESISTOR MASKING DEFECT

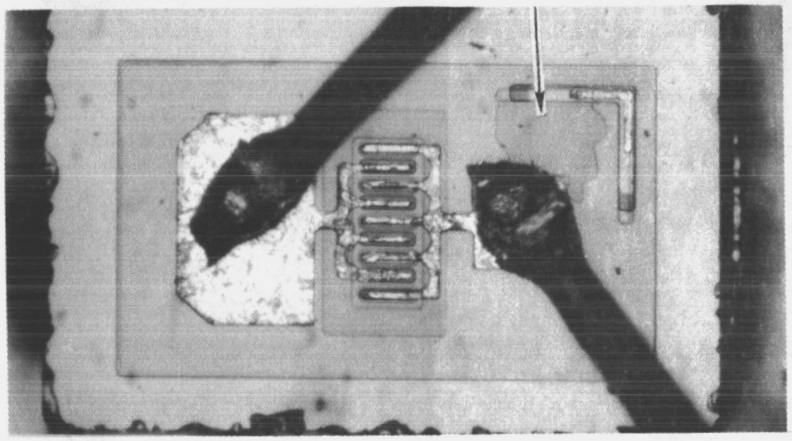

Figure 7B: MINOR TRANSISTOR DEFECT - MASK DEFECT

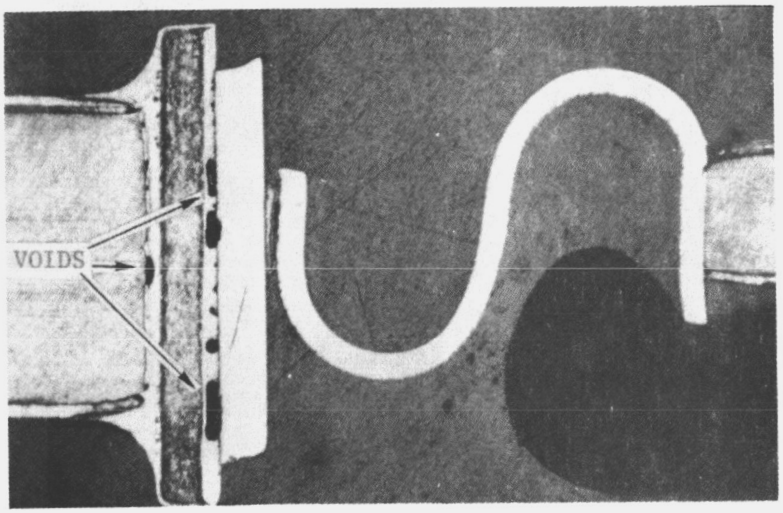


Figure 7C: MINOR DIODE DEFECT - VOIDS IN THE DIE PIECE PARTS

| | MFR. UNAWARE OF C.A. REQUIREMENT | MFR. AWARE OF C.A. REQUIREMENT |
|---|---|---|
| A. PART TYPES ANALYZED | | |
| Number of lots subjected to construction analysis | 15 | 11 |
| Total parts analyzed | 94 | 55 |
| Number of part numbers | | 11 |
| Number of part vendors | | 5 |
| B. DEFECT QUANTITIES | | |
| Number of lots with defects | 15 | 8 |
| Total number of defects observed | 50 | 17 |
| Total types of defects | 9 | 7 |
| Number of parts defective | 49 | 15 |
| Defects per part | .53 | .30 |
| C. CLASSIFICATION OF DEFECTIVE PARTS | | |
| Critical (part failure emminent) | 5 | 6 |
| Major (part failure probable) | 36 | 2 |
| Minor (part failure not probable) | 6 | 7 |
| No impact (part acceptable with defect) | 3 | 0 |
| D. PERCENTAGES | | |
| Percent of lots defective | 100% | 72.7% |
| Percent of parts defective | 52% | 27% |
| Percent of parts with critical defects | 5.3% | 10.8% |
| Percent of parts with major defects | 38.3% | 3.6% |
| Percent of parts with minor defects | 6.3% | 12.7% |

FIGURE 8. COMPARISON OF DIODE DATA

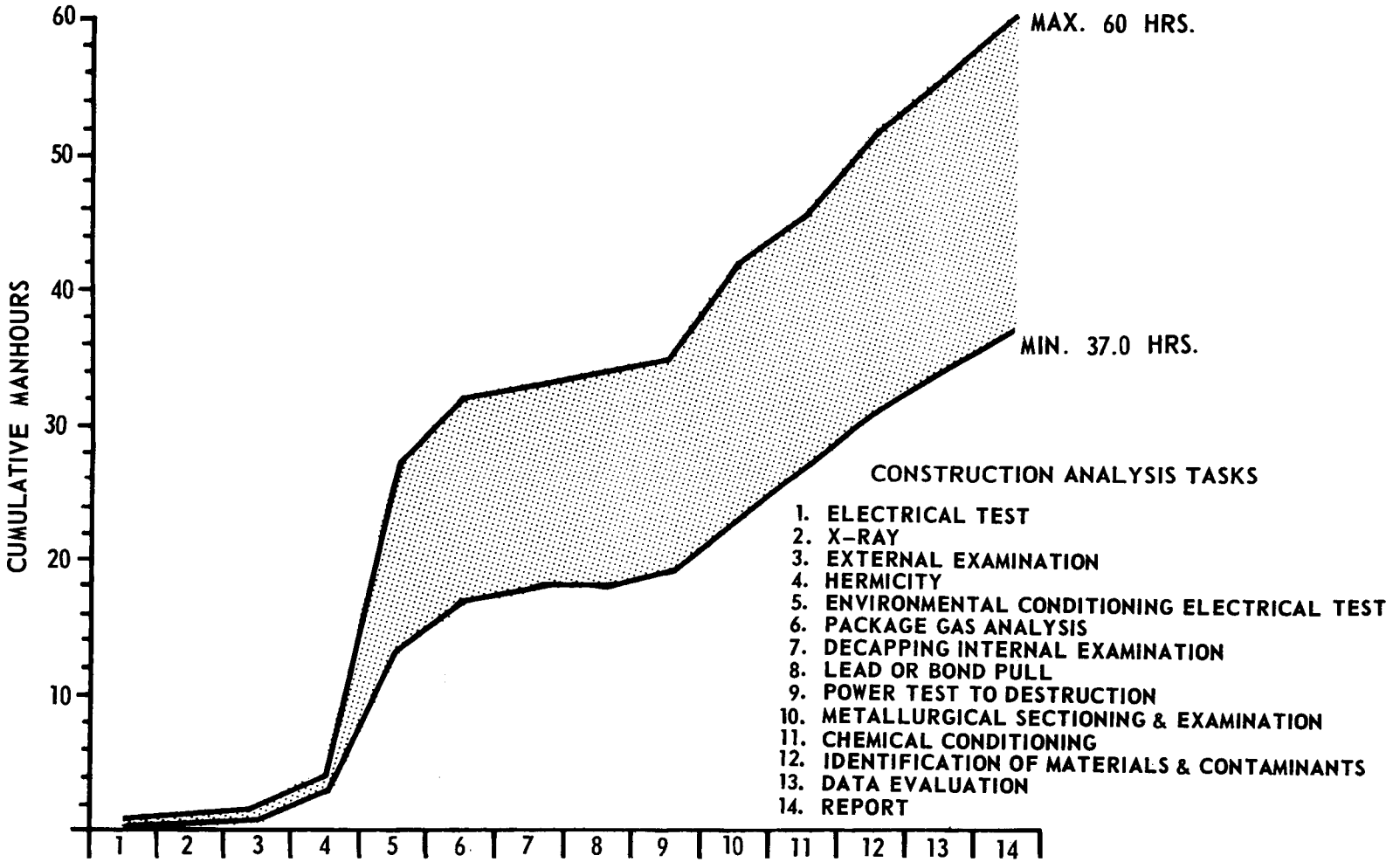


FIGURE 9

ANALYSIS MANPOWER ESTIMATE BASED ON A SAMPLE SIZE OF FIVE PARTS

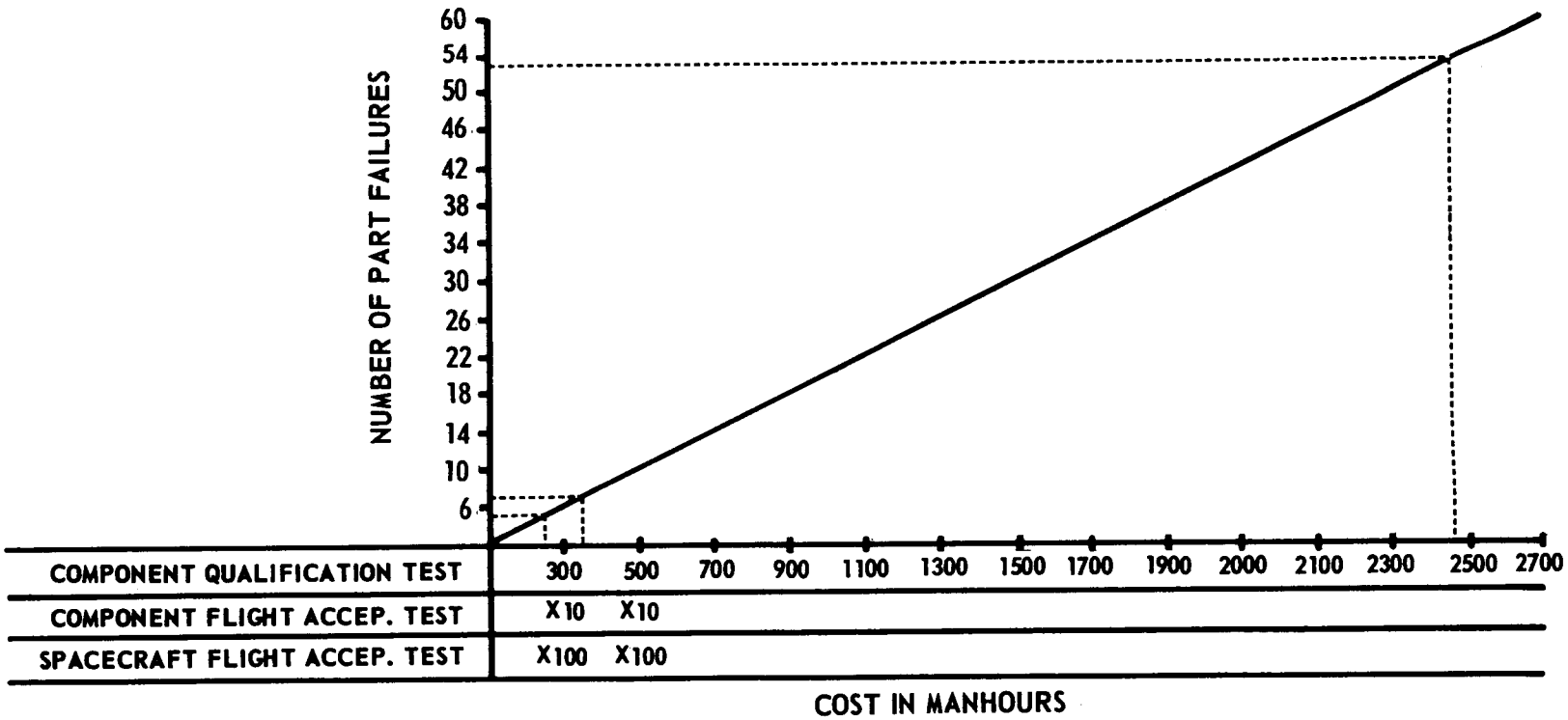


FIGURE 10A COST OF REMOVING FAILURE AT THREE LEVELS OF TEST

| NO. OF CA (5 PART LOTS) | 100 | 150 | 200 | 250 | 300 | 350 | 400 | 450 | 500 | 550 | 600 |
|---|---|---|---|---|---|---|---|---|---|---|---|
| NO. OF HOURS REQUIRED | 5000 | 7500 | 10000 | 12500 | 15000 | 17500 | 20000 | 22500 | 25000 | 27500 | 30000 |

FIGURE 10B CONSTRUCTION ANALYSIS COST CHART

# ELECTRICAL CONTACT MATERIALS FOR SPACE APPLICATIONS

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## Abstract

Factors influencing the performance and reliability of electrical contacts are largely determined by reactions at the contact surface. These reactions depend on the contact material, the chemical environment in which the contact device operates, and the mode of operation of the contact device. The significance of surface reactions relative to performance and reliability is often established by electrical requirements of the contacts. Friction, wear, sticking, tarnishing and the formation of insulating surface films, and reactions with organic vapors are most often responsible for contact failure. In this paper, performance and failure phenomena for several precious-alloy electrical contact materials are considered.

## Introduction

The functions of electrical contacts are to make, maintain, and break electrical circuits, and to allow passage of electrical information or power throughout their service life. Except for these functions, the circuit should not be influenced by the presence of electrical contacts. The importance of stability and reliability of electrical contacts varies according to application--in some cases failure is merely inconvenient, in other cases failure can have extreme consequences. For space applications maximum reliability is required in all systems, including electrical contact devices. Thus, detailed knowledge of factors influencing the performance of electrical contacts is of particular importance to space endeavors.

There are two broad categories of electrical contact devices. These may be defined as low-energy devices serving primarily to transmit electrical information, and higher energy devices for electrical power transmission circuitry. Because of differences in device construction, operations, contact materials, and problems involved in their operation, the topics are quite different. This paper will deal with electrical contact materials as related to low-energy applications.

In low-energy circuits, very slight changes that often occur on the surface of electrical contact materials can seriously impair the

quality of the electrical signal being transmitted. When dealing with "micro-watt" signals, changes in contact resistance of more than a few milliohms may seriously distort or cancel the signal. For this reason, the microscopic surface phenomena that occur on contacts are of major importance. It is toward this subject that this paper is directed.

## Electrical Contacts

### Mode of Operation

Basically, electrical contacts operate in one of three modes. These are sliding as in slip rings, normal opening and closing for relays, or static, such as connector pins. Some devices incorporate two or even all three of these modes either by design or unintentionally. For example, static connector pins may require periodic removals and insertions, and vibrations during service may cause sliding or arcing on a micron scale. In low-energy circuits, mechanical wear and surface film formation are important in establishing life. Because of the low voltages and amperages involved, contact resistance must generally remain at levels of a few milliohms for satisfactory service.

The presence of non-conducting tarnish films of more than a few tens of angstroms may result in failure of the contact to function properly. Likewise, abrasive wear caused by the presence of thin oxide films, for example, may generate very fine debris in the wear track of sliding contacts and increase contact resistance values to several hundred milliohms. Thus, surface reactions directly and indirectly control in large measure contact performance.

### Contact Materials

Copper and silver, when they can be used, are the most economical of contact materials. However, rapid tarnishing in earth-ambient environments has generally precluded the use of copper and silver-base alloys as low-energy contacts.

More precious alloys of gold, palladium, and platinum form the basis of most contacts for service in low-energy applications. Of these alloying bases, gold is the most noble, and its surface properties are unaffected by common inorganic environments. Surface reactivity of harder, more durable gold alloys depends upon amounts and types of alloying additions.

In contrast to gold, platinum and palladium form stable surface oxides. Although these are too thin to directly influence contact resistance significantly, the thin surface oxides do have important indirect effects on performance of platinum and palladium-base alloy contacts, as will be described later.

Electroplated composites are often used as contact materials for both technical and economic reasons. Gold and gold alloys are the most commonly used electrodeposits. These are applied over copper alloys, or nickel or rhodium base plates, for example, and find particular application as connector pins for low-energy circuits. This is a logical and often successful method of achieving the surface nobleness of gold at a price far less than that of a solid gold item. However, by so doing, one is apt to introduce problems that might be expected of a composite system, such as plating defects and selective substrate corrosion reactions.

Although the bulk properties of contact materials, such as resistivity, hardness, and elastic constants determine to some extent the usefulness of the material for a given application, these material properties are not nearly so important as the "surface properties" of the material in response to specific environments.

## Importance of Environment

Reactions between the environment and the surfaces of electrical contacts are all-important to performance and reliability of contacts. Physical adsorption, tarhishing or chemical reaction, and mechanical disruption of surface films are processes by which the behavior of contacts is altered. These processes and reactions are further complicated by mechanical wear occurring on contact surfaces. Furthermore, these reactions and their effects vary according to the mode of operation and electrical requirements of the contacts.

In the earth's environment, common atmospheric contaminants, such as sulfur-bearing gases, water vapor, and organic vapors, as well as oxygen, strongly influence the performance, life, and reliability of electrical contacts by forming insulating surface films, for example. Some of the natural surface phenomena can be beneficial, but most are detrimental. In the hard vacuum of space, the absence of active environmental species can radically affect the performance of moving contact surfaces. Wear mechanisms, for example, might be altered from abrasive wear to adhesive transfer, and in the extreme, pressure welding might occur under light load conditions.

Thus, considerations of the performance of electrical contacts in space must recognize the importance of surface reactions, which strongly depend upon both environment and materials. Because materials for space are of earth origin, environmental reactions during manufacture, proof testing, and storage prior to actual application in space remain very important. Once in space, the effects of prior earth-atmosphere exposure may be rapidly lost, or they may decay quite slowly.

For numerous contact devices, controlled environments to insure reliable performance and sufficient life are provided by hermetically sealing the device.

## Environmental Reactions

Broadly speaking, there are two types of environmental reactions that most frequently affect the performance of electrical contact materials. These are tarnishing reactions and organic vapor or organic film induced reactions.

### Tarnishing

Copper and silver are ingredients of contact materials that are most readily attacked by air. Although airborne sulfur-containing impurities have long been known to effect tarnishing of these materials, lack of adequate understanding of the fundamentals of tarnish behavior has caused uncertainties concerning the applicability of contact alloys containing copper and silver. For example, commonly used accelerated laboratory tarnish environments using $H_2S$, $SO_2$, or sulfur vapors result in tarnish ratings that are completely unrealistic by comparison with longer-term field test data. Without good correlation between laboratory and field tarnish behavior, evaluation of materials containing copper and silver for specific service is extremely tedious at best, and at worst may be completely false where accelerated tarnishing is used to predict service behavior.

Recent studies at Battelle have led to improved understanding of tarnish behavior of copper and silver, and a basis for realistic accelerated tarnish testing has been developed.[1] Figure 1 shows the typical tarnish behavior of silver and copper in natural earth environments. Both metals tarnish more rapidly in polluted city environments than in cleaner rural air. However, to a first approximation, the relative tarnish comparing copper to silver is the same in either natural environment.

Typical results of accelerated laboratory tarnish tests using $H_2S$, $SO_2$, or sulfur vapor carried in simulated clean air are illustrated in Figure 2. Although appreciable acceleration of tarnishing can be achieved the results are not at all realistic considering natural tarnish behavior. In no case does copper tarnishing approach the lower rate of natural tarnishing. Chemical analyses of tarnish films also showed gross differences between natural tarnish films and laboratory tarnish films. For example, the tarnish films formed in accelerated $H_2S$ testing are predominantly (copper) or exclusively (silver) sulfides. Natural corrosion products on copper are $Cu_2O$, $CuO$, and $Cu_2S$; on silver, $Ag_2S$ with 5-10 percent $Ag_2O$ or $Ag_2SO_4$ forms naturally.

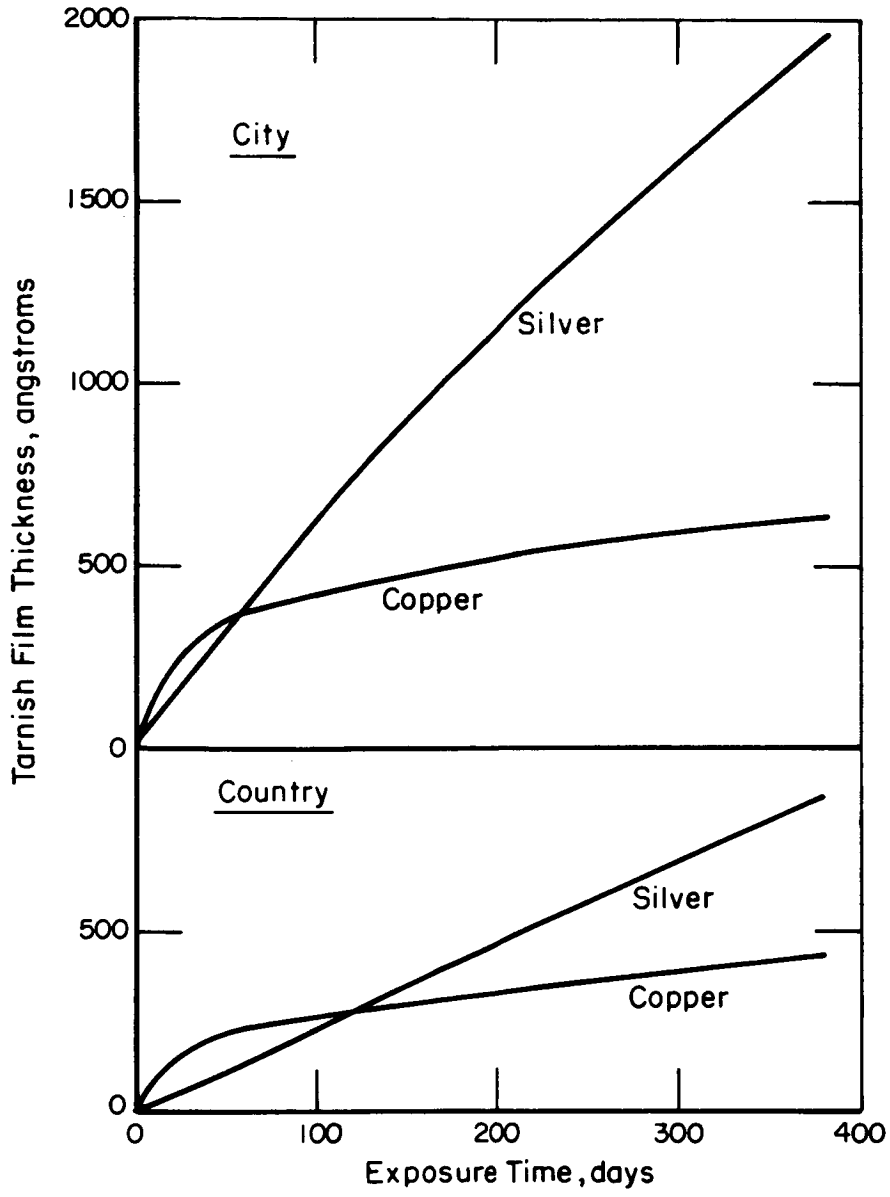



FIGURE 1. TYPICAL TARNISH BEHAVIOR OF SILVER AND COPPER IN NATURAL ENVIRONMENTS[2]

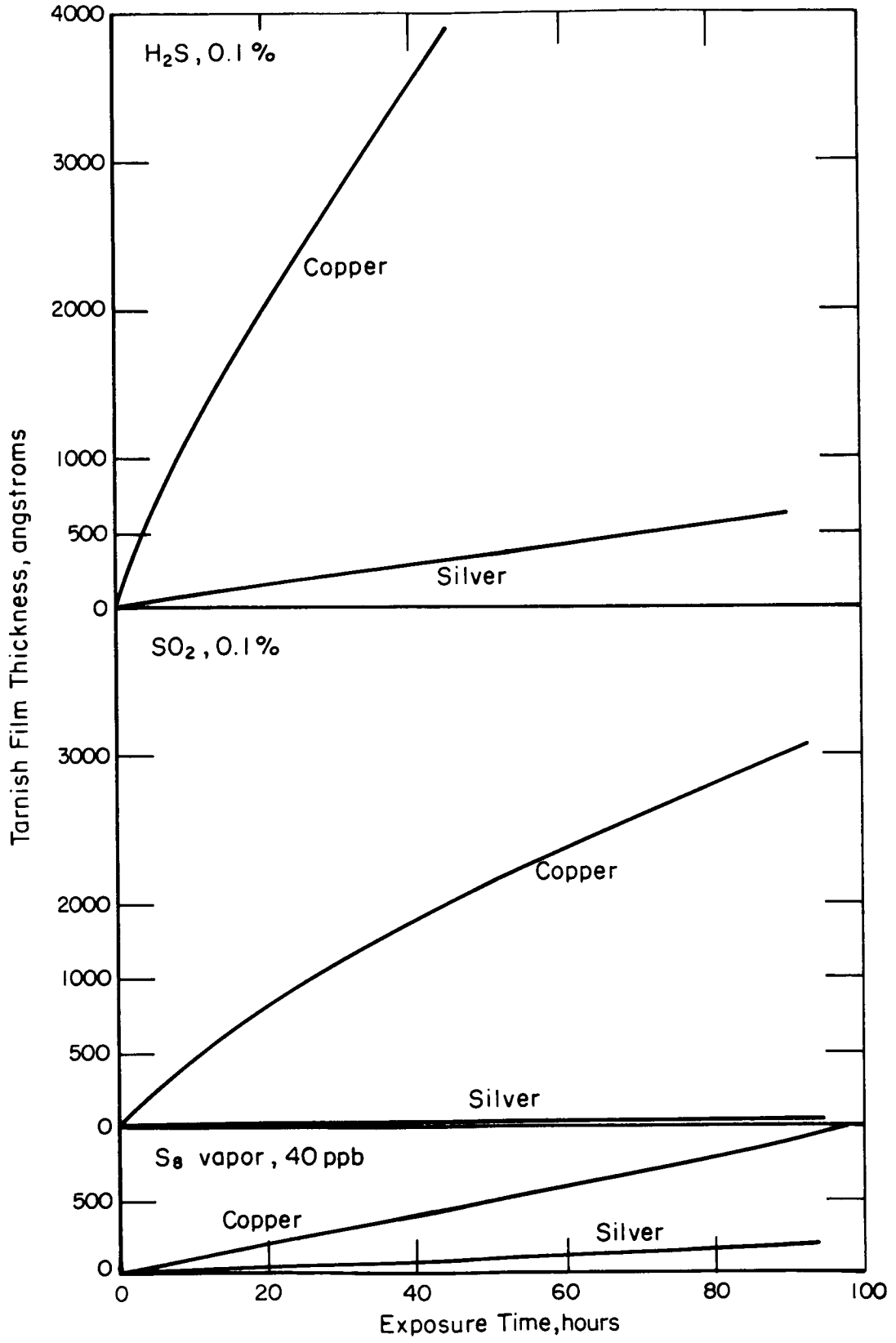



FIGURE 2. ACCELERATED TARNISH TESTS ON SILVER AND COPPER IN $N_2$-20 $O_2$ CONTAINING SULFUR-BEARING IMPURITIES

Battelle studies have found that both the film compositions and relative rates of formation of tarnish films formed on copper and silver are sensitive to the amounts of combined species of impurities common to natural earth environments. Either sulfur vapor or $H_2S$ in a $N_2$-20 $O_2$ mixture forms suitable basis for tarnish testing, sulfur vapor being the easier to control. Both water vapor and $SO_2$ are very important additional test-environment ingredients.

Figure 3 illustrates the powerful effects of environment on the tarnish behavior, particularly for copper. The data were taken from carefully controlled laboratory tarnish tests at Battelle using a base environment of $N_2$-20%$O_2$-20ppb$S_8$. By comparison with Figure 1 (the abscissa units are hours, rather than days), considerable acceleration is apparent. In such testing, the rates of tarnishing of both silver and copper are sensitive to the amount of $S_8$ vapor and flow rate of the test environment. The rate of tarnish of silver is relatively insensitive to $SO_2$ or water vapor additions to the basis test environment. However, the tarnish behavior of copper is very sensitive to these variables, as shown in Figure 3. Both water vapor and $SO_2$ increase the initial rate of tarnishing of copper, but also, in the illustrated cases, result in a transition to much lower tarnish rates after several hours testing. This behavior is quite similar to that shown in natural environments (Figure 1).

Analysis of compositions of tarnish films similarly indicated the importance of the proper combination of sulfur vapor, $SO_2$, and $H_2O$ in simulating natural-environment tarnishing. For silver, inclusion of small amounts (e.g., 100 ppb)of $SO_2$ vapor in the test environment results in the formation of the minor $Ag_2O$ or $Ag_2SO_4$ constituent in the $Ag_2S$ tarnish film, as commonly observed in natural tarnishing. For copper, both $SO_2$ and water vapor are necessary additives to the basis $N_2$-$O_2$-$S_8$ test environment to achieve realistic tarnish film chemistries in the laboratory. For example, in a $N_2$-20%$O_2$-20ppb$S_8$-100ppb$SO_2$-60% relative humidity test environment, the tarnish film on copper comprised about 70 percent $Cu_2O$, 20%CuO, and 10%$Cu_2S$ after 200-400-hour exposures. This composition is considered quite similar to those reported for natural tarnishing of copper.[3]

Figure 4 illustrates how these films influence contact performance. Here thickness of films formed under various conditions are related to static contact resistance. If, for example, the maximum tolerable static contact resistance were 20 milliohms, tarnish film thicknesses of several thousand angstrom on silver would be acceptable. For copper the indicated allowable film thickness depends upon the tarnish environment, and may be as low as 170 A or as high as 700 A according to the data. The curve resulting from exposure to a balanced $N_2$-$O_2$-$S_8$-$SO_2$-$H_2O$ environment to result in film compositions very similar to those found

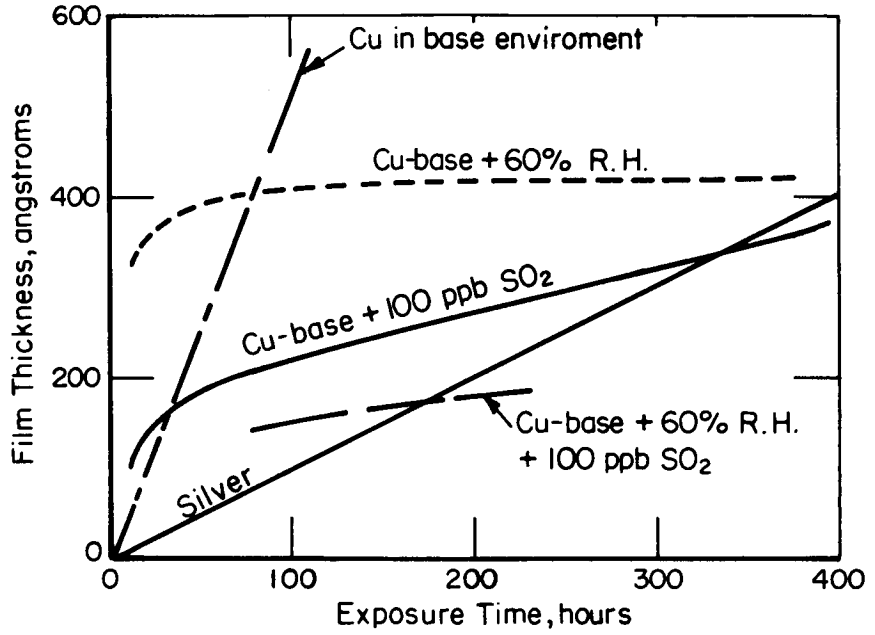



FIGURE 3. TARNISHING OF COPPER AND SILVER IN $N_2$-20 $O_2$ WITH 20 ppb $S_8$ VAPOR FLOWING AT 0.8 L/MIN

Shows influence of $SO_2$ and water vapor on tarnish kinetics of copper.

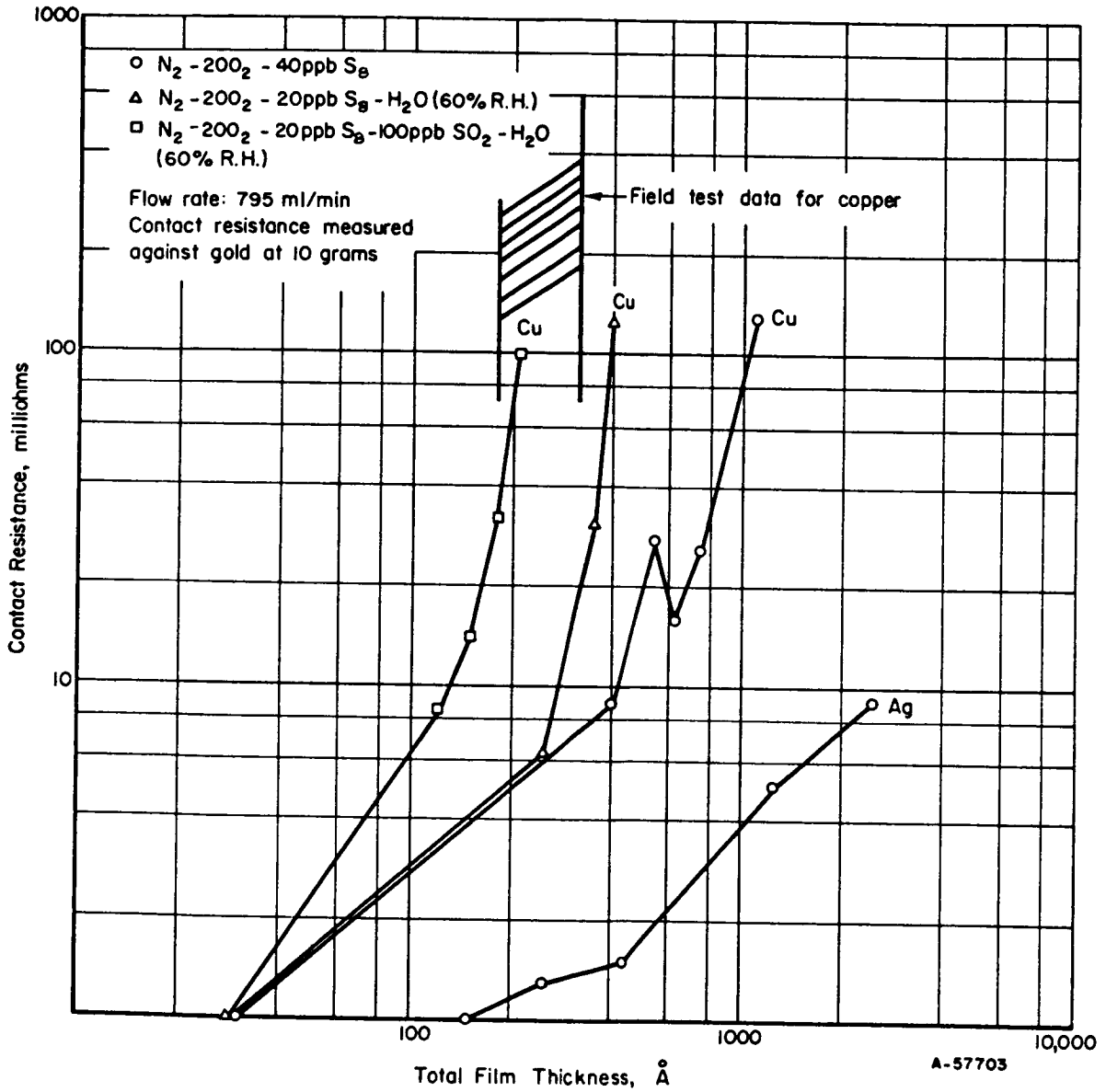



FIGURE 4. CONTACT RESISTANCE OF COPPER AND SILVER VERSUS TOTAL FILM THICKNESS IN THREE ENVIRONMENTS AT 30C

in field exposures. The thickness-resistance relation between lab and field exposures in this realistic test also agreed well, as indicated. Laboratory tests in $N_2$-$O_2$-$S_8$-$H_2O$ (no $SO_2$) gave different reaction kinetics, important differences in reaction products, and poorer agreement with field exposure results. Tests conducted in an $N_2$-$O_2$-$S_8$ environment were quite unrealistic in all regards. If one were to use an alloy whose tarnish behavior were controlled by high copper content, an accelerated tarnish test in a "nominal" laboratory tarnish environment of $N_2$-$O_2$-$S_8$ indicating a service life of, say five years, field service might well result in only a few months actual life because of the important relationships indicated here. (Note: This comparative discussion contains pitfalls, and is not meant to imply, for example, that tarnish films of several thousand angstroms thickness on silver are acceptable for contact performance.)

As a result of these Battelle studies, it is apparent that by proper attention to details, reliable accelerated tarnish testing procedures are available or can be developed which simulate behavior expected in service. The above discussion is centered upon environmental reactions with unalloyed silver and copper. Implications of the importance of detailed knowledge of environmental corrosion or tarnish affects are much broader, however, as both copper and silver are apt to be common alloying ingredients in electrical contact alloys. In this regard, recent Battelle studies for NASA have examined the tarnish behavior of gold alloys containing up to 20 percent silver and copper.[4] In an unrealistically agressive laboratory tarnish test environment of $N_2$-20 $O_2$-40ppm$S_8$ for 100 hours at 30 C, the thickest tarnish films were only about 50 angstroms compared with several hundred angstroms for pure silver or copper. The contact resistance of the alloys was not impaired by these thin films and remained at the values from 2 to 7 milliohms which were measured prior to tarnish testing. In this case, it was concluded that up to about 20 percent copper or silver as alloying additions to gold would not cause problems from natural tarnish phenomena that might be expected to result from earth storage prior to use in space.

## Oxidation

Storage of components prior to space service may involve a wide range of environmental conditions. One such condition could include modestly elevated temperatures such that normal oxidation could be a consideration.

Air-oxidation exposures for up to 1000 hours at 200 C is being studied at Battelle for NASA-Electronics Research Center.[4] Both wrought gold alloys and selected gold-alloy electrodeposits are being evaluated.

Table 1 summarizes some typical oxidation-test data, and shows the response of contact resistance to oxidation exposure and surface oxide growth. The binary Au-Ag alloy is stable against oxidation, but the addition of indium resulted in the growth of indium oxide that in some measurements increased contact resistance to nearly 100 milliohms. Copper as an alloying addition to gold resulted in relatively thick film formation, and gross deterioration of contact properties. The wrought Au-1Co alloy developed a visible film during oxidation exposure, but this film, presumably cobalt oxide, was easily disrupted by a contact force of less than 1 gram (10 grams were used in these measurements).

The Au-0.1Co electroplate on a copper substrate showed high and erratic contact resistance after oxidation for slightly more than 100 hours at 200 C. This was attributed primarily to development of copper oxides on the surface as a result of diffusion of copper through the electroplate. Use of nickel as a substrate somewhat improved the oxidation stability, although a significant increase over the initial value of 1.9 milliohms typical of these laboratory-prepared plates were observed. Because thinner (100 μin.) Au-0.1Co plates on nickel showed resistance values as high as 183 mΩ, it is reasonably certain that 200-microinch plates of this Au-Co alloy over nickel would not be sufficiently stable at 200 C for exposure periods approaching 1000 hours.

## Influence of Oxide Films on Contact Behavior

Although the noble metals (gold, platinum, palladium) and their alloys resist gross corrosion, the contact performance of platinum and palladium is influenced by the very thin, stable oxide films formed in air. These effects are perhaps best typified by contacts operating in a sliding mode. In "standard" sliding tests at Battelle, specimens of contact materials are fabricated from 0.110-inch-diameter wire, and are tested in a sliding crossed-rod configuration. Tests are conducted in a sealed chamber containing known, and closely controlled environments. Specimens are prepared asceptically to prevent spurious surface contamination effects. The equipment used is designed and instrumented for a range of contact forces, sliding velocity, and to permit measurement of static and dynamic contact resistance and coefficient of sliding friction. As preface to surface oxide effects, Figure 5 describes the performance of gold sliding on gold. The criteria here are sliding friction and the range of static contact resistance, both as a function of operating time. During sliding in either nitrogen or $N_2$-$O_2$, sliding friction increases continuously. Static contact resistance first decreases, then increases slightly as sliding and wear progress. From studies at Battelle, it is known that the presence of oxygen does not influence the performance of sliding, low-energy gold contacts. The presence of water vapor appeared to have some lubricating value, but did not influence the electrical phenomenon of gold.

TABLE 1. EFFECT OF OXIDATION ON CONTACT RESISTANCE

| Alloy (wt %) | Type | Static Contact Resistance, m Ω | | | |
|---|---|---|---|---|---|
| | | Initial | After 1000 Hr, 200 C Air Exposure | | Oxide Thickness, From Exposure, A[1] |
| | | | Maximum | Minimum | |
| Au-19Ag | Wrought | 6.4 | 4.2 | 2.6 | 0 |
| Au-19Ag-2In | Wrought | 9.8 | 98 | 4.1 | Not measured |
| Au-1Co | Wrought | 10.2 | 5.0 | 2.3 | Not measured |
| Au-12Cu | Wrought | 7.3 | 2800 | 12 | 1400 |
| Au-0.1Co | 200 μin. plate on copper | 1.9 | 153[2] | 8.3[2] | 61 (Copper oxides) |
| Au-0.1Co | 200 μin. plate on nickel | 1.9 | 6.8[2] | 3.2[2] | Not measured |

(1) Determined by electrolytic reduction. Indium and cobalt oxides are not reducible in the techniques used.

(2) After 100 to 120-hour exposure.

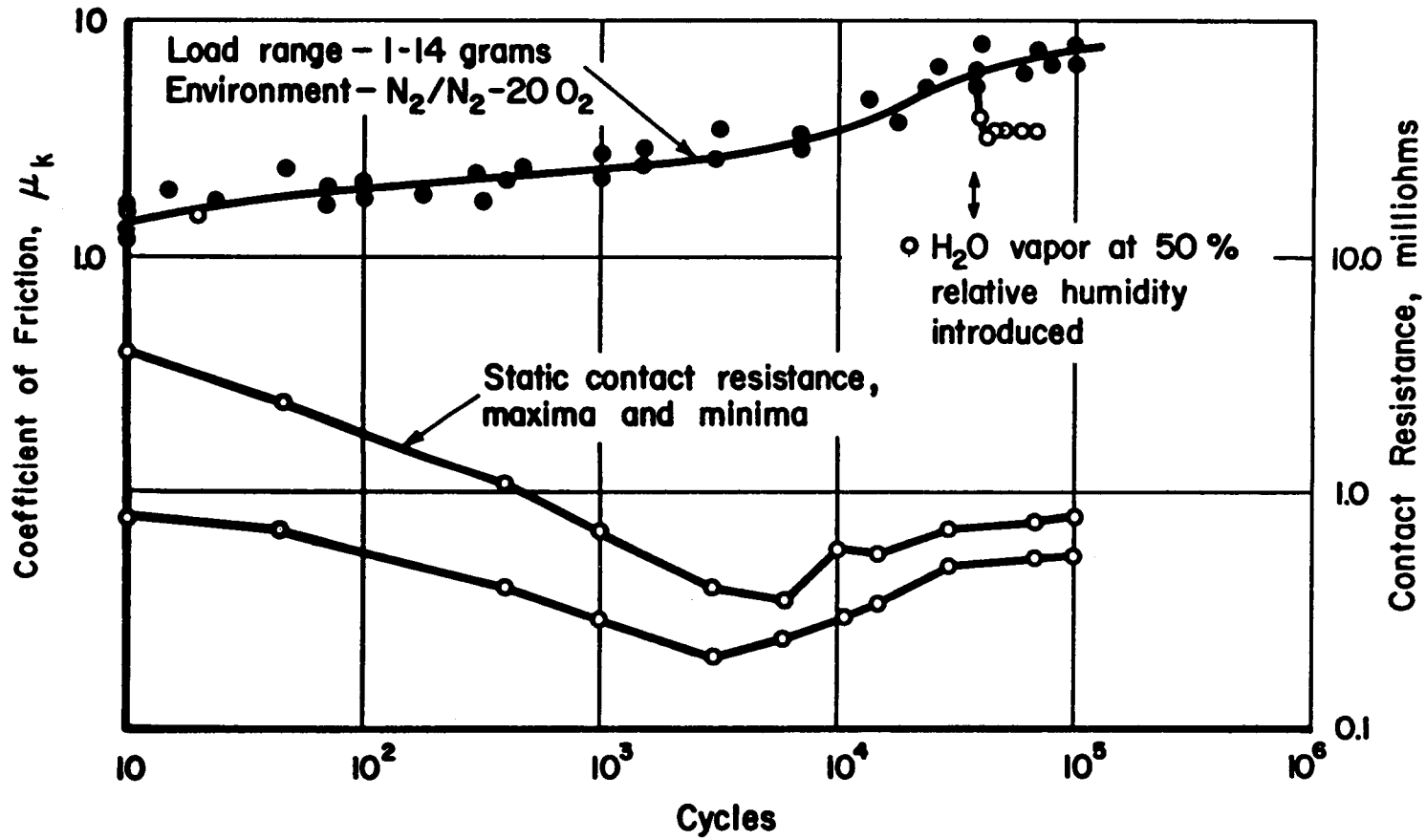



FIGURE 5. EFFECTS OF ENVIRONMENT AND LOAD ON THE FRICTION AND CONTACT RESISTANCE OF GOLD

Wear of gold contacts occurs largely by adhesive transfer. Alloying of gold can radically affect contact performance, but the affects are generally predictable and are related to expected changes in surface reactions.

Platinum- and palladium-base materials as sliding low-energy contacts are different in many respects from gold-base alloys. The differences in contact performance are largely explained by environmental reactions at the surface. Figure 6 shows typical results of sliding tests for platinum and palladium. In nitrogen, sliding friction shows a modest drop, followed by an increase during contact wear-in. Here, processes such as disruption of a thin surface film, initial wear pattern development, and work hardening at the surface explain the behavior. Thereafter, sliding friction remains relatively constant under nitrogen. Contact resistance of platinum and palladium is 1 to 1-1/2 orders of magnitude greater than for gold, and does not decrease during formation of the wear pattern but increases continuously. The electrical resistance behavior is associated largely with development of wear particle debris that is absent in sliding gold contacts. When oxygen is introduced to the test environment, radical changes are seen. Sliding friction decreases, and contact resistance and particularly the spread between maximum and minimum contact resistance increases drastically, as shown. Oxygen results in a much greater number of finer wear particles in sliding platinum and palladium contacts compared with wear debris in the absence of oxygen. The constriction resistance associated with the wear debris explains in large measure the drastic rise in contact resistance of platinum and palladium sliding in the presence of oxygen.

From this discussion, it is apparent that unalloyed gold, which is chemically inert with respect to normal inorganic environmental reactions in air, is not affected by oxygen in electrical contact applications. In contrast, the very thin, but stable oxides on palladium strongly influence behavior. These films are too thin to have much direct effect on resistance, but either residual films or films generated during sliding in air promote an abrasive wear mode and fine wear debris in the wear track, which does, by added constriction resistance, significantly affect contact performance.

## Organic Film

In addition to the effects of the usual environmental components (O, S, $H_2O$, $SO_2$, etc.), the influence of organic vapors on both sliding and relay contacts is extremely important. Organic vapors may be locally introduced to otherwise clean environments by adsorbed films residual from prior degreasing operations, or by slow volatilization from insulating plastics in vacuum or in asceptic, hermetically sealed devices. Under dry circuit (nonarcing) low-energy conditions, the

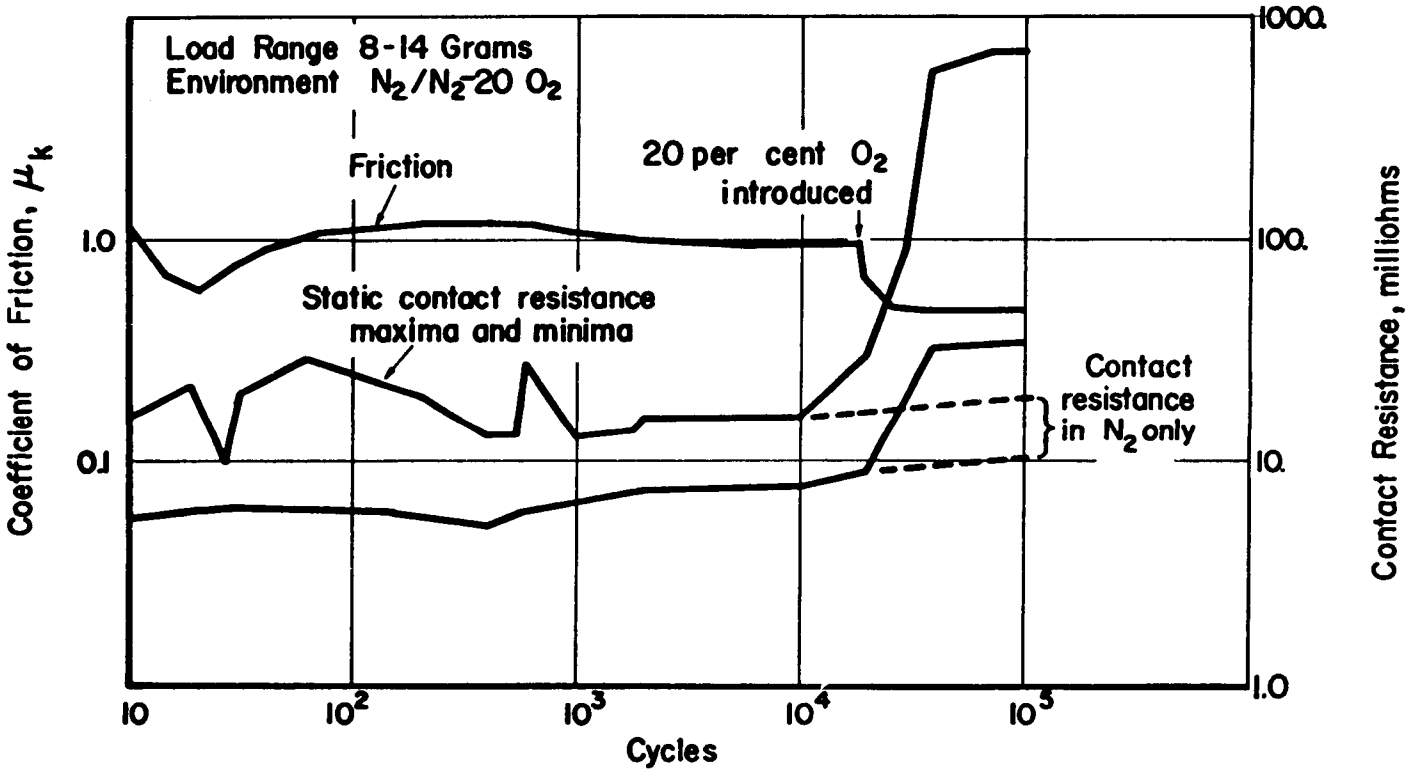



FIGURE 6. EFFECT OF ENVIRONMENT ON THE FRICTION AND CONTACT RESISTANCE OF PALLADIUM AND PLATINUM

presence of organic vapors, such as benzene, results in the formation of an insulating polymer that opens or prevents closure of the electrical circuit. This is called frictional polymer formation.

Figure 7 illustrates the behavior of gold under dry sliding conditions as evaluated in Battelle studies.[5] Benzene vapor was introduced at about 100 cycles. Instead of rising gradually as is usual in $N_2$ or $N_2$-$O_2$, the friction decreased slightly through about 7000 cycles, then dropped more noticeably between about 7000 and 12,000 cycles. Commonly, an incubation period is required to form polymer. The modest decrease in friction from 100 to 7000 cycles reflects modest lubrication value of adsorbed benzene. The drop at about 10,000 cycles is associated with polymer formation. This polymer "lubrication" markedly decreases the wear rate of the sliding gold contacts. Contact resistance is not affected in sliding dry circuit tests of gold-on-gold. Dynamic resistance measurements (e.g., measurements during sliding) of gold sliding on gold in Battelle's test facility shows frequent open circuit bounce when tested in $N_2$ or $N_2$-$O_2$ environment. Electrical noise, defined as the dynamic resistance envelop, is accordingly extreme for gold. With the development of frictional polymer and its incorporation in the wear track, open circuit bounce is eliminated and contact noise is reduced to very respectable values. Figure 8 shows the wear track of a gold specimen after testing under polymer forming conditions. The dark polymer that forms is adherent, easily sheared, and not continuous. Although it is most likely nonductive, its morphology in the wear track is such that gold-gold contact is retained. Frictional polymer of this type formed on pure gold under dry sliding conditions is very beneficial to contact performance.

Dry sliding tests of palladium in the presence of organic vapor show quite different behavior from that of gold. Figure 9 illustrates this behavior. After an incubation period approaching 100,000 cycles, polymer forms. This results in an apparent decrease in friction, but with a relatively large increase in contact resistance from a value of 10-20 milliohms to values of several hundred milliohms. In comparison with tests on gold, the amount and type of polymer are quite different. Figure 10 shows a rather fluffy polymer, containing metallic wear debris, largely, but not completely swept from the wear track.

Platinum studies in benzene showed polymer formation tendencies similar in some respects to those for palladium--large amounts of fluffy, nonadherent polymer are formed after rather long incubation periods. However, as shown in Figure 11, the polymer is completely swept from the wear track, and as a result, there is no affect on friction nor on contact resistance. Most surprisingly, the wear rate of platinum, and even of hard platinum alloys is drastically increased to values greater than those for soft gold under polymer forming

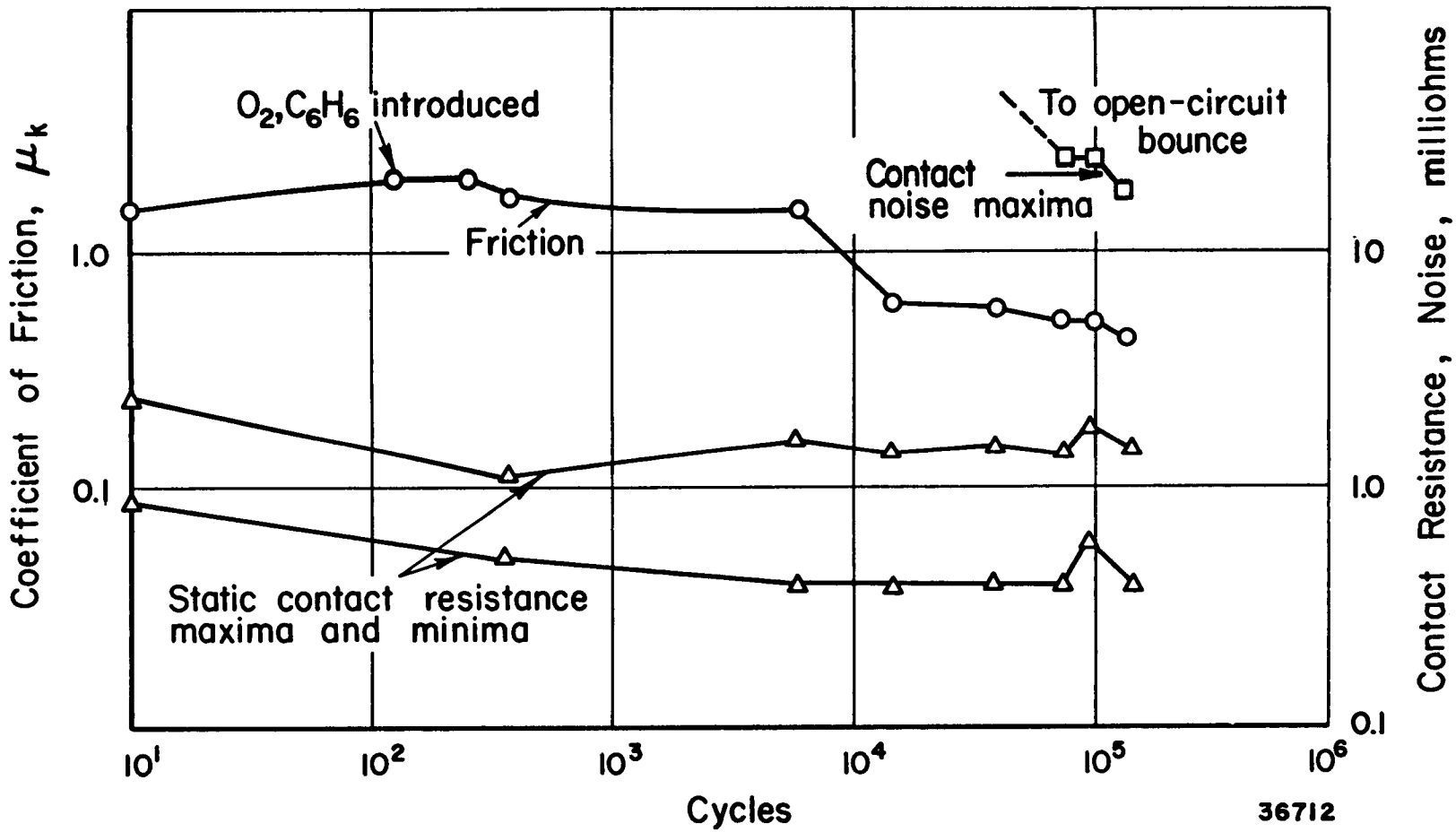
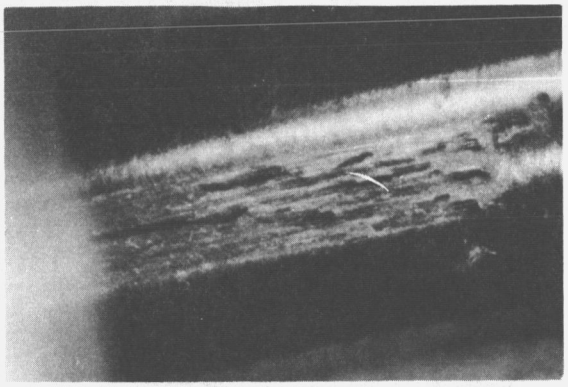
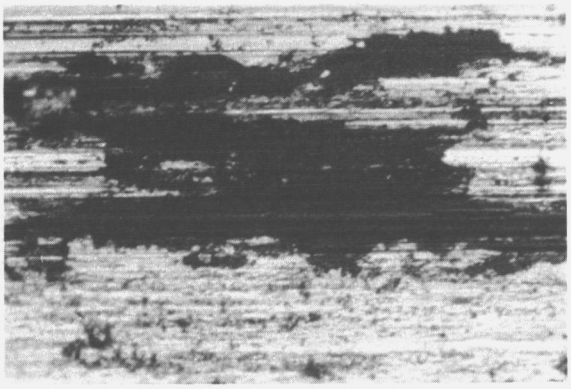



FIGURE 7. EFFECT OF OXYGEN AND BENZÉNE VAPOR ON THE FRICTION AND CONTACT RESISTANCE OF GOLD AT 14.0 GRAMS - I



a. 20X



b. 500X

FIGURE 8. FRICTIONAL POLYMER FORMED ON GOLD

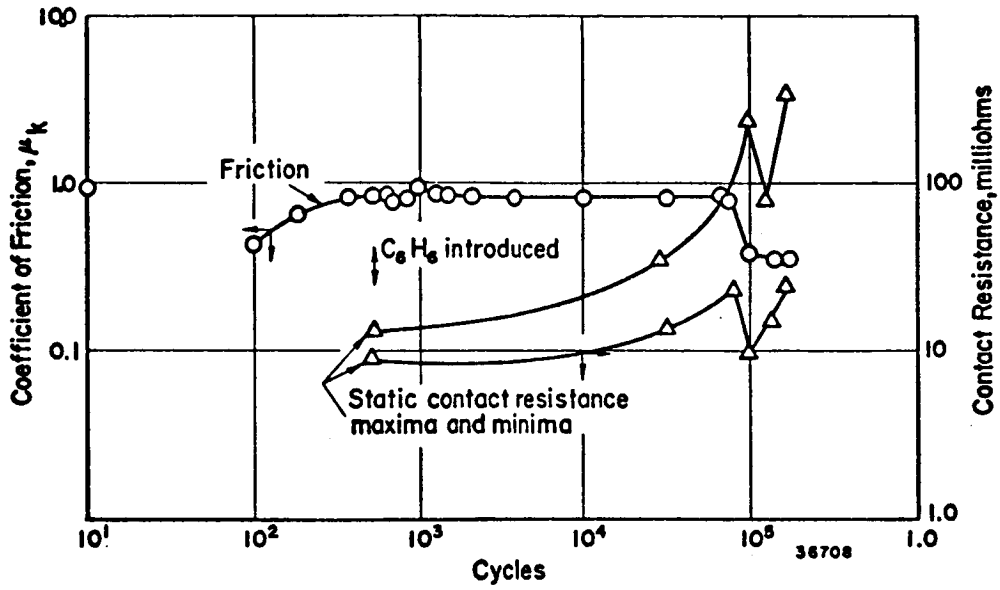



FIGURE 9. EFFECT OF BENZENE VAPOR ON THE FRICTION AND CONTACT RESISTANCE OF PALLADIUM AT 14.0 GRAMS

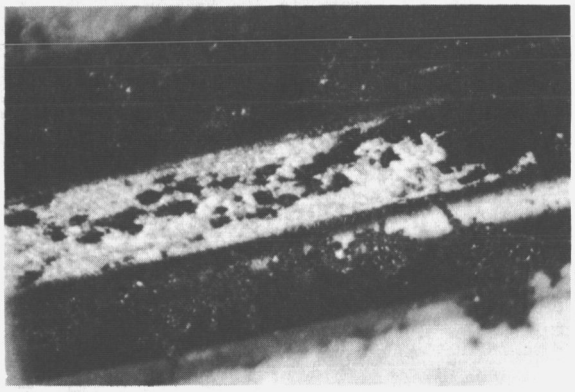


FIGURE 10. FRICTIONAL POLYMER ON PALLADIUM

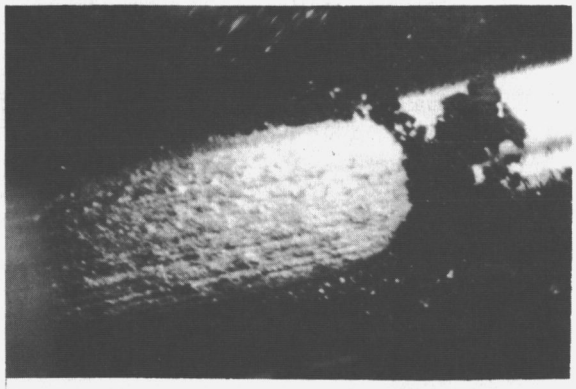


FIGURE 11. FRICTIONAL POLYMER ON PLATINUM

conditions. The increased wear is illustrated in Figure 12. As yet, there is no explanation for this behavior. In contrast with the beneficial effects of polymer formation on gold, polymer formation during sliding of platinum, palladium, and their alloys, is highly detrimental to contact performance.

Effects of Vacuum

The absence of oxygen or other environmental contaminants in space would be expected to influence the performance of contact materials. For example, the effect of oxygen in generating profuse amounts of wear debris on palladium alloys would be absent, and generally lower values of contact resistance might be expected in space than in earth service. On the other hand, the lack of foreign surface films (e.g., oxides) increases the tendency for adhesive metal-to-metal bonding, as is well known[6], and kinetic friction values, stick-slip behavior, and contact noise values will increase as a result. Specific effects of vacuum on contact performance involve complex considerations of contact material, device design, mode of operation, and service life.

In evaluation of gold alloys for possible use as connector materials, Battelle under NASA sponsorship[7] has performed sliding tests in air and vacuum which serve to illustrate the influence of vacuum on material performance. Table 2 summarizes some data obtained in this program. Measurements of kinetic friction, static contact resistance, and sliding contact noise after the contact surfaces had been cycled through 100 sliding cycles are listed. This sliding service is reasonable for connector applications, but barely within the region of "wear-in" for sliding applications such as slip rings.

As described previously, the binary Au-19Ag alloy is resistant towards the formation of oxide films, while the other alloys do contain oxidizable ingredients and the surface films formed do influence behavior. As indicated in Table 2, the friction coefficient of the less noble alloys, although lower than that of Au-19Ag in air operation, is measurably increased by 10 to 30 percent to the range exhibited by the Au-19Ag alloy during cycling in vacuum.

Measurements of static contact resistance after 100 sliding cycles in vacuum show the values to be from about 3 to 4 milliohms ($\sim$ 40 percent) lower than after comparable treatment in air for alloys containing stable oxide formers. For the Au-Ag alloy, this difference was only 1.1 milliohm on the average. The differences in static contact resistance are attributed to differences in wear modes and the amount and size of wear-particle debris present in the wear track where measurements were made. Sliding in air favors the generation of many fine wear particles for alloys containing copper, cobalt, or

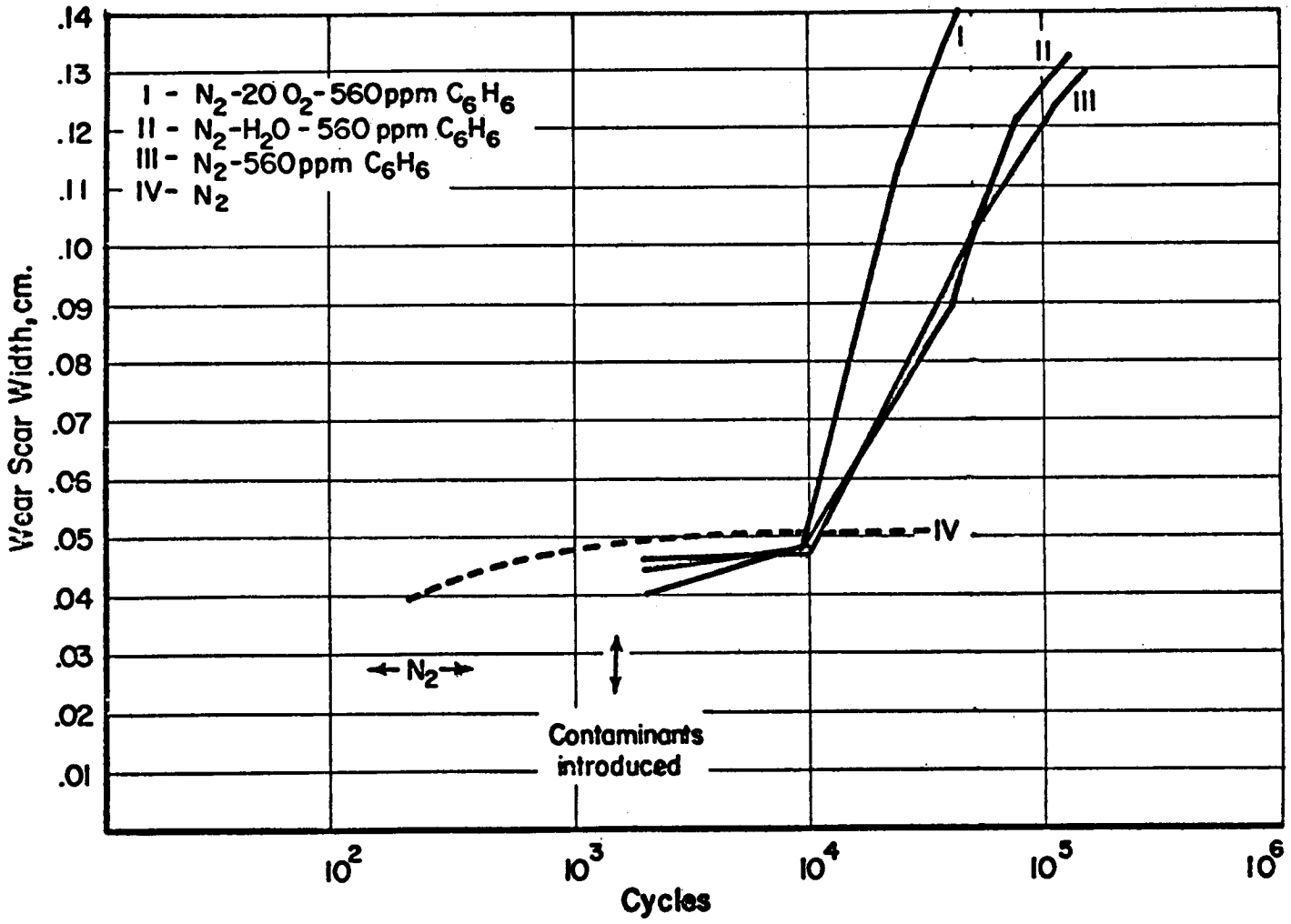



FIGURE 12. EFFECT OF OXYGEN AND WATER VAPOR ON THE WEAR OF PLATINUM SLIDING IN $N_2$-560 PPM $C_6H_6$ AT 14 GRAMS

TABLE 2. SUMMARY OF SLIDING CONTACT BEHAVIOR AFTER 100 SLIDING CYCLES IN $N_2$-20 $O_2$ AND VACUUM

| Alloy (wt %) | Coefficient of Sliding Friction | | Median Static Contact Resistance, m Ω | | Sliding Contact Noise (Peak-To-Peak Res.), m Ω | |
|---|---|---|---|---|---|---|
| | Air | Vacuum | Air | Vacuum | Air | Vacuum |
| Au-19Ag | 2.34 | 2.18 | 3.4 | 2.3 | 4.7 | 8.6 |
| Au-12Cu | 1.70 | 2.06 | 7.2 | 4.5 | 7.0 | 10.2 |
| Au-19Ag-2In | 1.61 | 2.07 | 7.2 | 3.6 | 5.6 | 6.4 |
| Au-12Cu-2In | 1.65 | 1.77 | 9.0 | 5.5 | 9.5 | 11.4 |
| Au-1Co | 2.15 | 2.44 | 8.1 | 5.0 | 9.0 | 9.2 |
| Au-1Co-1In | 1.99 | 2.14 | 9.5 | 6.3 | 8.0 | 8.7 |

Note: Test vacuum was $\sim 10^{-7}$ to $10^{-8}$.
All data are averages of at least two tests.

indium. In vacuum, tendencies are towards adhesive transfer rather than abrasion as the dominant wear process, and wear particle debris is much reduced. Accordingly, constriction resistance, hence static contact resistance, associated with wear debris is lower after sliding in vacuum.

Sliding contact noise, a measurement of dynamic contact resistance peak-to-peak variation made while contacts are sliding, is highly dependent upon the mechanical and operating characteristics of the contact device (contact load, sliding speed, contact mounting rigidity, etc.). The indicated results in Table 2 show that in every case, noise is greater in vacuum sliding than in air sliding. This also is attributed to the increased "stick-slip" behavior associated with the predominant adhesive wear made under vacuum.

In general, differences observed between air and vacuum sliding were small in the cited examples, although the indicated trends were quite definite. In absolute terms, values of sliding friction and contact noise were not as low as is desirable for low-energy contact materials. Theseresults, however, do suggest the types of evaluations that are critical to evaluations of performance of electrical contact materials for space application.

## Electroplated Contacts

As mentioned previously, gold or gold alloy electroplates are often used in contact devices. Implicit in their use is that service conditions be within the limits of wear and breakthrough to the inexpensive but undesirable (from electrical contact aspects) substrate surface. Life and reliability of performance depend on wear rates, thickness, and impermeability of the electrodeposites, as well as the electrical contact behavior such as contact resistance and its stability, and sliding contact noise.

Despite at least moderate use of electroplated contact materials, much remains to be learned about performance as related to plating quality requirements and electroplate-substrate interactions.

In a NASA-sponsored program at Battelle[7], numerous performance aspects of electroplated gold-alloy contacts are being evaluated. Briefly, some of the major findings to date are summarized in the following paragraphs.

Porosity of gold-alloy electrodeposits is very sensitive to substrate cleanliness and geometry, and to plating thickness. Data for Au-0.1Co electrodeposites on chemically polished copper substrates

having initial roughness of 5 to 7-microinches CLA, illustrate the effect of plating thickners:

| Nominal Coating Thickness, microinches | Pore Density (No. of Pores/sq cm), (5 Determinations) |
|---|---|
| 50 | 10, 21, 4, 11, 4 |
| 100 | 3, 7, 4, 2, 0 |
| 200 | 1, 2, 1, 0, 1 |

Data were obtained by electrographic printing.

Performance of electroplates may differ markedly from that of similar wrought gold alloys. Table 3 compares some sliding contact material evaluations of solid, wrought versus electroplated Au-Co alloys. From these data, it is apparent that the electroplated material is markedly superior to the wrought material in both $N_2$-20 $O_2$ and vacuum environments. The wear mechanism for the wrought alloy was predominantly adhesive transfer, but that of the electroplated material showed no evidence of either adhesive transfer of abrasive wear. Wear tracks of electroplates were smooth with only very few wear particles. This wear mode was reflected in low contact resistance, stability, and contact noise. Exact reasons for these pronounced differences are not yet known, but surface chemistry differences are believed most important.

Laboratory tarnish tests on Au-Co electroplates over copper showed no evidence of substrate bleed-through. As discussed previously, oxidation tests performed in air at 200 C for times up to 1000 hours showed that both copper and nickel diffused through even 200-microinch electroplates in as little as 100 hours, resulting in high resistance values.

In summary, electroplated contact materials exhibit performance characteristics that are not necessarily predictable based on knowledge of solid wrought material of similar nominal composition. Much remains to be learned concerning the reasons for these differences to allow design and development of optimum electroplated composite materials for service as electrical contacts.

## Summary

Studies at Battelle have demonstrated the extreme importance of surface reactions and phenomena to the performance and reliability of materials used for low-energy electrical contacts. Reactions characteristic of earth-ambient environments and their influence on contact behavior are reasonably well understood. These are important to space

TABLE 3. COMPARISON OF CONTACT PROPERTIES OF WROUGHT AND ELECTROPLATED Au-Co ALLOYS

| | | Properties Measured After 100 Sliding Cycles[1] | | | | |
|---|---|---|---|---|---|---|
| Material | Test Environment | Friction Coefficient | Static Contact Res., m Ω | | Sliding Contact Noise, m Ω | Wear Scar Width, cm[2] |
| | | | Median | Range | | |
| Wrought | $N_2$-20 $O_2$ | 2.15 | 8.1 | 3.9 | 9.0 | 0.0172 |
| 100 μin. Electroplate | $N_2$-20 $O_2$ | 1.39 | 2.0 | 1.0 | 6.0 | 0.0056 |
| Wrought | Vacuum[3] | 2.44 | 5.0 | 4.7 | 9.2 | 0.035[4] |
| 100 μin. Electroplate | Vacuum[3] | 1.31 | 2.4 | 2.1 | 6.3 | 0.012[4] |

(1) Results are averages of at least two tests.

(2) On 0.110-inch-diameter rod specimens.

(3) $10^{-7}$ torr.

(4) After 500 sliding cycles.

activities in that earth manufacture, testing and qualification, and storage remain necessary functions. Where electrical contact materials are to be used in space without environmental control, performance is governed by the hard vacuum of space. As indicated by simulated-air versus vacuum tests, life and performance may depend strongly on environment. For long-duration space missions, some of the problems associated with long-life, high-reliability electrical contact materials are known or can be predicted. Others, however, will require more detailed knowledge of specific mechanisms governing contact performance, and the relation between these mechanisms and behavior of contact materials to allow formulation of new and improved electrical contact materials for space application.

## Acknowledgment

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# MANAGEMENT DECISIONS FOR SYSTEM DEVELOPMENT

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## ABSTRACT

The advent of long duration space flights magnifies existing historical problems and imposes new ones which must be solved if those flights are to achieve their missions. Because of the high equipment operating times involved, the opportunity for failures to occur increases.

Our task then is to reduce the impact of equipment failure.

Historically, we have been plagued by failures of the lowest order equipments which have in turn resulted in loss of mission or objective. Concentrated R&D efforts to improve those equipments has resulted and will continue to result, in significantly increased success.

Another method to reduce the occurrence of failures is to optimize equipment operating time.

Still another method to reduce the impact of equipment failure is to apply the concept of redundancy logic and its extensions to systems designs.

Similar effort needs to be expended in the R&D conceptual area to provide techniques for fault detection and restoration of failed equipment to operating condition within the time constraint imposed.

## INTRODUCTION

The advent of long duration space flights magnifies existing historical problems and imposes new ones if those flights are to achieve their missions. Because of the high equipment operating time involved, the opportunity for failures to occur increases. Examination of a system whose mandatory functions are satisfied by 300 equipment components in simplex array can serve to demonstrate that point. If each of the 300 components has a failure rate of .62 x $10^{-6}$ failures per hour, the probability of successfully operating that system for 48 hours is greater than .99. That two days of continuous operation can be related to a Mercury flight. If, however, the same system were to be used in a Mars/Venus flyby, circa 1975, the probability of its success becomes .086. The reduction is due to the fact that the components must operate for 13,200 hours. It is of interest to note that the failure rate used coincides with that of the combined electrical connector types used in the Boeing 707 and 720 aircraft. It is also significant to point out that the failure rate of connectors compares favorably to that of other components.

It is obvious that a probability of success as low as .086 cannot be tolerated. The task then becomes twofold. On one hand, the incidence of failure must be reduced. On the other, the impact of individual component failure on the system must be lessened. Both of these considerations must also recognize their effect on each other and any mission or system constraints which are imposed. It is possible, as shall be shown, to degrade one factor while introducing a gain in another with a resultant net gain in their cumulative effect. Systems analysis to establish trends and optimize the possible combinations of component array and actions should be utilized to synthesize a simulation model in order to predict system behavior. It should be noted that the desired condition is one of no failure, but in actual practice failures occur. It is that "real life" condition which must be treated.

Historically, the plague of failures of the lowest order of equipment has been the cause of loss of mission or objective. Concentrated effort to improve those equipments has resulted in a significant capability to achieve success. If the 300 components previously treated for a Mars/Venus flyby were to have a failure rate of .048 x $10^{-6}$ failures per hour, the probability of success becomes .827.

The gain is significant. Use of .048 x $10^{-6}$ failures per hour was deliberate in that it is the failure rate attributed to MIL-C-26500 connectors in the same environment as that previously cited for combined electrical connectors.[1]

## COMPONENT RELIABILITY IMPROVEMENT

The established reliability ratings of many of today's lowest order standard equipments dictate the need for design improvements. Items like connectors, switches, fasteners, pipe joints, oil seals, electric motor and generators are cited. Fortunately, unlike the weather, something can be done about component reliability.

In many cases a new or improved material is required, or possibly a change in design concept. Whichever the case, the fact that some meaningful component R&D efforts are needed is inescapable.

To be really effective, these R&D efforts are needed in the conceptual phase of program development. A component "Most Wanted" list should be established and R&D efforts undertaken to satisfy these requirements.

After the system designer has laid his pencil to the board and begun component selection, it is frequently too late to initiate component R&D efforts. It is well known how drawing release schedules constrain prototype development and testing.

An example of how a component R&D effort can pay-off is Boeing's experience with electro-mechanical switches. For some time, these switches have been the standard component selected by system designers for indicating changes in the relative position between two parts. The switches have been and continue to be a problem in critical applications due to their relatively high failure rate. The problem has been design concept. The fact that electro-mechanical switches have internal and external parts which must move for switch operation dictates inevitable failure due to mechanical wear or contaminants entering the switch housing through mechanical access holes. Actuating levers are easily bent or broken. Electrical contacts become contaminated and shorts occur. A switch was needed which is hermetically sealed, has no moving parts and a significantly improved reliability.

An aggressive R&D effort was undertaken by selected switch design manufacturers at Boeing's urging. The result was the solid state proximity switch. This switch is hermetically sealed, has no moving parts and is demonstrating a major improvement in reliability. The trouble rate of electro-mechanical switches for a cross section of applications on Boeing 727 aircraft averaged 58 x $10^{-6}$ failures per hour of switch operation. A similar cross section of applications of the proximity switch on Boeing 737 aircraft reveals a trouble rate of 3.8 x $10^{-6}$ failures per hour of operation. In the cited like applications, electro-mechanical switches averaged failing approximately fifteen times more often than proximity switches.

Another example of how component R&D efforts can pay significant reliability dividends is the Integrated Drive Generator (IDG).[2]

The IDG concept was originated by Sundstrand Aviation in 1966. The objectives were to:

1. Improve significantly the overall reliability of aircraft electrical power systems.

2. Achieve weight reductions and improved maintainability where possible, yet maintain or increase electrical power output ratings.

The IDG has taken the constant speed drive and a generator and merged them in a single housing. Prior to the IDG these items were installed and maintained as individual components.

The IDG design resolves many of the problems associated with the individual component approach. The drive and generator are contained in a single housing and share a common bearing and oil supply. The oil acts as a lubricant for all bearings and a generator coolant by being sprayed directly onto the generator windings. There are no rotating oil seals between the drive and generator, thereby eliminating a major source of possible oil leaks. All bearings are uniformly lubricated during operation.

Several fallout improvements have resulted from IDG prototype testing. Two are a reduction in payload penalty

and increased safety.

Another advantage of the oil cooling is that the generator can run at 150% of rated load continuously without damage whereas air cooled generators can only sustain overloads for a few minutes without damage to insulation.

The big payoff in the IDG R&D effort has been the achievement of the reliability objective. The initial Mean Time Between Failure (MTBF) of the combined drive/generator package is predicted to be 10,000 hours or approximately three times the MTBF now obtained from previous designs. With design maturity and experience the MTBF is expected in increase to 15,000 hours or more. Component R&D can and does pay-off.

## REDUCING IMPACT OF FAILURE

Thus far the introduction of high reliability components has increased the systems' probability of success from .086 to .827 for a Mars/Venus flyby mission. The gain is laudable, but the risk is still high. The second part of the task is now at hand. It is one of reducing the impact of individual component's failure. The first method that comes to mind is the concept of redundancy. If half of the 300 components in the previously shown high reliability system are redesigned so that parallel redundancy exists, then the mission probability of success becomes .909. For that increase, the payload penalty in weight and space is roughly equal to one-and-a-half times the original value. If all 300 components were to be made duplex redundant, the probability of success becomes .999. The increased probability of spurious or unwanted signals or functions being introduced into the system by the redundant legs has been ignored.

Up to this point, restoration actions relating to the continuously operating system have not been considered. With the introduction of redundant legs, restoration of failed components to proper operation becomes feasible if the failures are detected. Failure of one leg of a duplex array does not cause loss of function. If the second leg fails while the first is being restored, then system failure occurs since we have originally stated the requirement for continuous operation. Effectively, the requirement has been

that there is no allowable system downtime. In the system where half of the components were made redundant, the probability of success remains .909 if failed legs can be restored to operational use in an hour. Under the same conditions, the all-duplex system's probability of success remains .999. In addition to doubling the number of installed components and adding connection requirements, the need for spares, trouble detection devices, and other logistics resources has been added without a detectable third decimal place improvement in probability of success. In the system described, there appears to be no justification for providing an in-flight restoration capability.

Miniaturization and micro-miniaturization have made valuable contribution toward reducing the redundancy payload penalties. In the case of the all-duplex system, if miniaturization were to increase the failure rate of each component to the original value of $.62 \times 10^{-6}$, the probability of system success becomes .980.

It is conceivable that a concept of standby or switchover redundancy could be used whereby a single component could act as the redundant leg for more than one in-line component. Standby or switchover redundancy imposes a requirement for the capability to detect impending failure of the component for which the redundant component will be substituted. Application of this technique requires marginal diagnosis and the setting of pre-control limits at which the use of the redundant component is made. Once having determined when to use the redundant leg, it could be switched in by manual action or it could be automated by having the impending failure sensor act as a trigger. Two significant contributions to reduce the probability of success (when compared with the all-redundant system) have just been made. One is the reduction in the number of equivalent redundant legs and the other is the introduction of added components by the sophistication of troubleshooting and switch-in requirements. On the positive side, a payload gain may be achieved and the probability of inadvertently introducing a signal or function has been reduced.

Thus far the system considered had to operate continuously with no downtime allowed. If the original simplex system (or one with similar characteristics) is considered, but one hour of downtime can be tolerated at any time during the mission, then its probability of success becomes .991 if we can restore the failed component to operational use in 20 minutes. No redundancy has been built into the system so

there is no payload penalty on that score. There is a payload penalty involved by the requirement to troubleshoot and effect restoration rapidly. The luxury of allowing equipment downtime can sometimes be afforded by the introduction of a capability to accumulate, store, and use on demand the function it provides. In that manner, continuity of function is provided during the time the equipment which normally provides it is being restored to an operating condition. Use of this technique could provide the one hour of equipment downtime necessary to obtain the probability of success of .991 just cited. Note that the requirement for continuous function has been retained and satisfied by a type of redundancy related to the function rather than the equipment components themselves.

Figure 1 compiles the predicted availability of the example system's function for two component failure rates. Some variety in component functional arrangement and conditions of restoration action is included. The numbers on the chart are not considered absolute within themselves. They have been derived for use in establishing trends which result from the changes in configuration and actions allowed.

| CONDITIONS OR RESULTS / COMPONENT FUNCTIONAL ARRANGEMENT | AVAILABILITY OF SYSTEMS FUNCTION | | ALLOWED EQUIPMENT DOWNTIME | MEANTIME TO REPAIR | APPROXIMATE PAYLOAD PENALTY FACTOR |
|---|---|---|---|---|---|
| | $.62 \times 10^{-6}$ FAILURES/ HR. | $.045 \times 10^{-6}$ FAILURES/ HR. | | | |
| SIMPLEX SYSTEM | .086 | .827 | 0 | NO REPAIR | 0 |
| | .885 | .991 | 1 HR* | 20 MIN. | > 0 |
| HALF DUPLEX REDUNDANT SYSTEM | .290 | .909 | 0 | NO REPAIR | 1.5+ |
| | .293 | .909 | 1 HR.** | 1 HR. | >1.5+ |
| ALL DUPLEX REDUNDANT SYSTEM | .980 | .999 | 0 | 0 | 2+ |
| | .999 | .999 | 1 HR.** | 1 HR. | > 2+ |

* STORED FUNCTION USED DURING DOWNTIME
** FOR FAILED REDUNDANT LEGS ONLY

FIGURE 1 EXAMPLE SYSTEM'S AVAILABILITY

## CONSIDERATION OF CUMULATIVE EFFECTS

There are other mission conditions which can be considered. Each system must be examined for its own requirements. A system may have to operate continuously to provide a function. Another system may have some equipment downtime allowed if the function can be stored and used on demand. Still another system may operate to provide a function which is used intermittently to commence either at a discrete time or during an envelope of time. In the last case, properly scheduled pre-use checks can be of value. The check won't assure that the system will work on demand, but it will provide opportunity for restoration action. If the system is down and no pre-use check is made, then the time constraint within which to effect restoration is limited to zero or the envelope of time during which it must commence functioning if it has failed during its dormant period. Thus, the pre-use check can serve to provide an amount of time, in addition to the envelope, during which restoration action can be performed.

Models which consider and simulate the functional arrangement of system components and the operating demands are extremely valuable tools which should be used to optimally approach goals or requirements for performance. One such model was used to predict the trend of the Saturn V Launch Vehicle availability for launch during a lunar window. The first step in the process of simulation was to describe the functions performed during pre-launch test and checkout and to optimize the arrangement of the functions so that the series path of functions would consume as little time as possible. Once having optimized the pre-launch flow, simulation runs were made to determine the probability of successfully launching the vehicle during a lunar window. That probability of no failure was approximately .20. When that probability was allowed to include not only the "no failure" runs but also those during which failure occurred, and the failures were restored to operational use within the time constraint imposed, it became .45. Holds were introduced into the operational flow and simulation runs made. The holds were inserted so that they could be used all at once or bit by bit during designated phases of the countdown. If all of a phase's hold time were not used for restoration during a simulation run, the remainder was simply used by a dwell at the end of that phase. Equipment operating time was charged to the entire phase. The elapsed time

for a phase then became the sum of the critical time path and the planned hold. Hold time was varied until an optimum probability of launch within the lunar window occurred. The time constraint within which to effect the restoration action now became not only the launch window and the period of time equipment was not on the critical time path, but those two times plus the hold time. More failures were experienced during the simulation runs because of the increased equipment operating time, but the loss was more than offset by the increased capability to recover from failure.

The net effect of opening up the time constraint within which to effect restoration action resulted in a probability of successfully meeting the launch window of approximately .65.

## CONCLUSIONS

From the limited model shown in Figure 1 and the launch vehicle availability experience stated some general conclusions can be reached. These are:

1. The use of high reliability components makes a significant mission success contribution in itself.

2. If component redundancy is used, little gain is made by introducing the capability to restore failed legs to an operating condition.

3. Redundancy, with its payload penalty, can be used to solve the problem of high reliability parts' unavailability. Where payload penalty is not a factor, as in ground support equipment, the economics of using redundant low reliability components should be considered.

4. Where equipment downtime can be tolerated or introduced, a restoration capability can be used extensively as a contributor toward availability requirements.

5. Reliability and restoration capability can, and should, be traded with respect to each

other to achieve required availability in a safe, cost-effective manner.

In addition, it can be concluded that more sophisticated models utilized in computerized systems simulation should be used during the advanced design phase of a system to predict availability. The logic of the model should consider both reliability and the capability for restoration actions as possible contributors to availability. The product of the model should be an optimized arrangement of components to achieve the availability requirement within any constraints which are imposed. The task at hand is to reduce the impact of equipment failure on mission success.

Note that the task is "to reduce the impact of equipment failure on mission success". It follows, then, that an equipment failure which does not cause a function to go out of tolerance during the time when that function is required, has no impact. The ideal path is to eliminate equipment failures entirely. Concentrated R&D efforts to improve the reliability of equipment have resulted in a significantly increased success pattern. Similar amounts of R&D effort must be expended to provide techniques which recognize or introduce allowable equipment downtime and the capability to restore those equipments to an operating condition within the time constraint imposed.

Mission success can be thought of in terms of its complement - mission failure. Mission failure occurs when there is loss of a mandatory function. Mandatory function loss occurs when there is an equipment failure which is not restored within the time constraint imposed. Equipment failure is dependent upon its inherent reliability and its operating time. Restoration of equipment failure depends upon its inherent restoration characteristics and the allowable time within which to restore.

Some actions which enhance one factor may tend to degrade another. The use of a plug-in unit may allow quick component replacement but degrade reliability by the introduction of a connection point. The use of "B" nuts instead of welded tubing has a similar effect. It is only when the cumulative effect is considered that a logical decision can be reached. There must be no cultism or tunnel vision practiced. Cumulative effect implies a requirement to consider the variable effects of all contributors in order to establish a trend which can be optimized to approach the goal - MISSION SUCCESS.

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Mr. Gralow graduated from the U. S. Naval Academy and was assigned to the cruiser force with duties in gunnery and engineering. Designation as a Naval Aviator and qualification as a Special Weapons Bombardier/Navigator followed together with activities as squadron maintenance officer. He taught at the Nuclear Warfare School and has lectured on all phases of nuclear weapons at the Naval War College, the Armed Forces Staff College and the Naval Academy, as well as presenting courses at the R. A. F. Staff College. After leaving the Navy, he was employed by Avco and supervised the training services area for the Operational Missiles Subdivision. A period as a consultant performing the Human Factors advanced design for aircraft simulators followed. He returned to Avco in the Human Factors area and was engaged in preventing reliability degradation through human error in manufacturing. Mr. Gralow came to Boeing in January 1965 to work in Maintenance Engineering on the Saturn V Launch Vehicle. Principal program contributions include being author or co-author of Boeing and NASA maintenance analysis methods documents, maintenance concepts, Human Engineering exhibits, and Launch Decision Criteria rationale and methods. He is currently supervisor of the Systems Analysis Group performing Technical Integration and Evaluation of the Apollo Program at Washington, D. C.

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Mr. Varunok graduated in 1949 with a degree in Aeronautics from St. Louis University. He joined Boeing in 1955 after one year with Republic Aviation and three years with Curtiss-Wright as Field Service Engineer. He has served in a variety of technical and management positions on aircraft, missile and space programs. They include Field Service Engineer on the B-52 program, Service Engineering Manager at Malmstrom and Whiteman Air Force Base on the Minuteman program, Integrated Support System/Reliability Chief on Minuteman, and Logistics Manager for the Washington, D. C., Apollo Technical Integration and Evaluation operations. His assignments have involved the development and application of logistics concepts, techniques for evaluation of reliability and integrated logistics support systems performance, mission effectiveness assessment techniques, and techniques for evaluation and assessment of on-going logistics programs.

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Clinton A. Wilkinson

Mr. Wilkinson is currently manager of Boeing's Apollo Technical Integration and Evaluation support activity in Washington, D. C. and has been with the company since 1941.

His first assignment was as a structures engineer. Subsequent positions included work in materials and process development, propulsion, ramjet engine propulsion, and pneumatic accessories in the B-52 bomber program. In 1960 he was appointed project engineer for the Bomarc Missile Program, moving from there to director of engineering for the S-IC program when Boeing formed its Saturn team in 1961. He was named Michoud Manager, heading Boeing's New Orleans rocket assembly and testing effort, in August, 1965. He was named Technical Integration and Evaluation manager in June, 1967.

Mr. Wilkinson is a graduate of Washington State University, receiving his B. S. degree in mechanical engineering in 1936 with highest honors. He is a native of Billings, Montana, a member of the American Society of Metals and the American Institute of Aeronautics and Astronautics.

# TECHNICAL MANAGEMENT OF THE GEMINI PROGRAM

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## Program Objectives

The objectives established for the Gemini Program were varied and technically quite demanding as shown in Figure 1. They included the original objectives which were established at program initiation as well as several key objectives which were established during the program in response to evolving manned spaceflight requirements. This latter group generally required an extremely rapid reaction by the Gemini Program team to meet the schedule required.

A major objective was rendezvous and docking. This required the inclusion of sophisticated electronics for establishing relative vehicle positions and velocities. Sophisticated dynamic analyses and mission planning techniques were also required. The Agena target vehicle was furnished by the Air Force. A docking adapter was developed by McDonnell to mount at the forward end of the Agena.

Another major program objective was the control of the reentry to achieve landing at a preselected touchdown site which required an offset of the vehicle CG to achieve aerodynamic lift and some means of controlling the direction of the lift trajectory. This was accomplished by use of much of the same electronics hardware required by the rendezvous and docking objectives.

Another major objective which is probably the one of most interest to this particular symposium involved a mission duration of 14 days. Although many of our unmanned spacecraft had completed missions of much longer duration, a manned flight of 14 days was a considerable increase over previous manned spacecraft experience at the time and has not been exceeded to date. The major spacecraft feature which resulted from this requirement was the use of a fuel cell for electrical power utilizing hydrogen and oxygen reactants. In addition, more stringent time-dependent design and qualification requirements were implemented.

In addition to the objectives which were established at program initiation, there were several opportunities during the program to achieve more sophisticated goals. These opportunities were seized upon with vigor and accomplished in a timely manner. An example is the achievement of extravehicular activities on several missions. This capability was provided through minor modifications to the basic spacecraft along with

extensive preflight crew training and carefully planned mission operation activities. Another example is the Gemini 7/6 mission following the failure of the target vehicle intended for use with spacecraft 6 to achieve orbit. A plan was quickly formulated to achieve a rendezvous between two manned spacecraft. Gemini 7 was modified for rendezvous target use and launched on its 14 day mission. It was joined by Gemini 6 in orbit for several hours of station keeping. Accomplishment of this dual mission required some fast reaction changes to the Gemini 7 spacecraft as well as an all out effort by the launch operations team. Another example of an objective or requirement which was developed during the program was an Augmented Target Docking Adapter (ATDA) which was developed in a short time span as a backup to the baseline Agena target vehicle. It utilized Gemini subsystem hardware packaged into an orbital vehicle which was used as a target on the Gemini 9 mission.

It is apparent that the original Gemini Program objectives were challenging and required a well conceived technical management approach. This approach had a built-in quick reaction capability which also allowed the achievement of several more objectives which were created as program evolution demanded.

## Program Management

The Gemini organization was established in such a way that it brought high level McDonnell management attention to bear on all elements of the program while providing the NASA management team with access at any level needed to assure timely coordination on problems as well as routine matters. This organization is shown in Figure 2. In addition to the normal project line functions, the Vice President and General Manager was supported by a Technical Director who gathered into a closely knit group the program elements who were more heavily involved in technical aspects. This permitted the normal program direction loop from NASA to the Vice President and General Manager to be augmented by a rapid-response technical coordination loop through the Technical Director. It also permitted easy movement of key personnel between engineering, acceptance tests, and launch operations as required by the evolving program demands. In addition, the Technical Director had a staff of Spacecraft Managers with companywide authority to bring any required resources to bear on problems affecting each specific spacecraft as it made its way through the manufacturing, acceptance test, and launch operation phases of the program.

For a research and development program like Gemini where the objectives were sophisticated, the schedule was demanding, and the attention of the general public always present, routine management was obviously unsatisfactory. Some of the key reasons for this are noted in Figure 3. It should be noted that the program was under a well-formulated incentive

contract and had the advantage of excellent team morale stemming from its high national priority and prestige. NASA and McDonnell capitalized on these advantages and established many specific procedures which allowed the program to receive the detail attention which it required. Some examples of the procedures used are shown in Figure 4. A daily meeting of key contractor personnel provided a means of instant communications between various staff functions and gradually welded a common approach to decision making which permitted delegation of daily problems so that no time was lost due to lack of on-the-spot decisions. Control rooms containing schedule and financial information, as well as facts and status on outstanding problem areas were maintained in St. Louis and Houston to permit rapid access to program status by both McDonnell and NASA executives. Monthly management meetings gave a regular forum for presentation of data related to specific problems and resulting NASA/McDonnell management review and direction. A configuration management plan was established which achieved careful control of the evolving configuration of each spacecraft without being unduly burdened by paperwork. Subsystem managers were established for each major subsystem to assure that across-the-board coordination between NASA, the prime contractor, and the subcontractors was constantly achieved. This also assured that the proper balance between engineering requirements, available funding, schedule requirements, etc. was achieved. Another key management technique was the establishment of regular executive meetings including key personnel from all of the major contractors. These meetings were called by NASA Headquarters and proved useful in achieving top management participation in the program.

## Design Approach

The design approach used on Gemini was typical of that employed in most major research and development programs. However, in many ways the design approach used had a strong influence on many of the program decisions made later since the spacecraft configuration was deliberately structured to permit operational flexibility.

A series of weekly NASA/McDonnell technical reviews was held throughout the first year of the program. These involved across-the-table discussions of configuration details as they evolved. This series of meetings essentially accomplished, in a less formal way, the Phase B/C program definition effort now included in more recent programs.

Some examples of the configuration tradeoffs made are shown in Figure 5. One example was the choice of ejection seats rather than an escape tower for launch abort. This minimized the dynamic interaction between the spacecraft launch configuration and the booster as well as providing a backup landing capability.

Many of the configuration tradeoffs had a profound influence on the later capability to react to evolving program requirements. It was decided to locate the bulk of the subsystem equipment external to the pressure vessel. Although this required qualification for a severe vacuum environment, it yielded great rewards in improving the capability to manufacture, inspect, modify, and checkout the subsystem installation. It also greatly reduced any potential in-flight fire hazards. The spacecraft configuration was separated into several discrete modules. This greatly enhanced access to any individual subsystem area and again allowed efficiencies in manufacturing, inspection and operational activities. Another significant design concept change from our previous experience on the Mercury Program was adoption of manual operation rather than automatic operation for key mission functions. This tended to simplify many of the subsystem design requirements. It also helped assure that key mission sequences would not be inadvertantly actuated prematurely by an automatic system.

Figure 6 gives an overview of the spacecraft configuration which resulted from the original design tradeoffs. It illustrates the modular construction which was used. It also points out that independent systems were used to achieve maximum reliability wherever required. For example, the reaction control system used for reentry was packaged separately and remotely located from the orbital attitude and maneuvering system. This assured that the reentry control system would be available and unused when required for the extremely important reentry portion of the mission. Ease of checkout and resulting reliability were greatly enhanced by the provision of separate test points on each black box used to verify proper operation and/or troubleshoot any anomalies without the necessity of disconnecting previously checked electrical connectors. Subsystem equipment was located so that each piece could be individually installed or removed without disturbing adjacent pieces. This again enhanced reliability in that it was unnecessary to disturb another piece of equipment in order to work on a troublesome piece.

Figure 7 illustrates one of the key design features which we believe contributed to the reliability of the spacecraft. Wire bundles were preassembled in three-dimensional fixtures on the bench so that uncluttered and comfortable access to each operation was available. The spacecraft's structure was so arranged that these premanufactured wire bundles could be installed without undue bending or threading through structural holes. This greatly minimized inadvertant breakage or damage to the wire bundle.

Crew safety was always paramount in choosing the design approach for a specific area. Figure 8 illustrates one example of this. At the time of firing the solid retro rockets, it could not be proven conclusively that the volatiles which remained in the jettisoned portion of the adapter section would not react violently and cause some damage to the spacecraft. Although it was felt that the probability of a problem was small, a blast resistant shield was developed to prevent impingement of the retrograde rocket exhaust on the storage tanks in the jettisoned portion of the spacecraft.

The normal complement of development and qualification tests was performed to assure achievement of full mission duration. These are highlighted in Figure 9. Probably the most significant was the full-scale thermal vacuum test of a complete vehicle which achieved orbital environment stabilization.

## Sustaining Engineering

As noted in Figure 10, Gemini design and system engineers participated in all phases of the program from the initial establishment of requirements through post-mission redesign required by mission anomalies. Typically, an engineer might be involved sequentially in design efforts, factory checkout, KSC checkout, real-time mission support, and post-mission analysis for his particular area of specialization. This flow of personnel from one program activity to another assured availability of competent, technically qualified people for all program phases. This personnel flow is illustrated in Figure 11. A design engineer would begin his project duties on the design team. He might then be assigned to a subcontractor's plant to assist in and monitor the development of his subsystem. He would then be assigned to the acceptance test team during the factory checkout activities. He would then continue to KSC to participate as a member of the launch operations team. Similarly, an operations engineer who had participated in the Mercury launch operations would come to St. Louis to monitor project design activities and/or participate in the acceptance test operations. He would then proceed to the launch site to again become a member of the launch operations team. These design and operations engineers were augmented with St. Louis personnel experienced in general test operations. These people would also join the acceptance test team and later flow into the launch operations team. Finally, the remaining members of the team were made up of qualified vendor representatives who had been involved in subsystem development at the subcontractor's plant, then came to St. Louis to participate in acceptance testing, and eventually to KSC as part of the launch operations team. This gathering of technically qualified people from all program activity areas insured a well-balanced test team who could do all necessary tasks in checkout, troubleshooting, and correction of anomalies. This flow of people resulted in staffing time histories as shown in Figure 12. It can be seen that as St. Louis staffing passed its peak, the KSC staffing was beginning to build up and achieved the required level of effort throughout the launch operations period with many of the people coming from the original St. Louis team.

Of equal importance with properly supporting acceptance testing and launch operations, is providing highly skilled people for direct support of the spacecraft and crew during the mission. This support was offered at Houston in direct coordination with Mission Control Center while

backup support was offered by a team located at KSC and by another team of design specialists located in St. Louis. This is illustrated in Figure 13. Figure 14 summarizes some of the contributions that such a team of technical specialists can make. A particular example involved the failure of a cryogenic tank heater on the Gemini 5 mission. This heater failed early in the mission and resulted in inability to provide the desired pressure of oxygen to the fuel cell. After detailed analyses of the circuitry failed to reveal any likely areas which could be repaired by operational methods, attention was brought to bear on determining the acceptability of the available pressure. A team in St. Louis immediately began operation of a ground test fuel cell system with the supply of pressure reduced to the level available in the Gemini 5 spacecraft in orbit. This test and other analyses contributed to a "go" decision which permitted the full 8-day mission to be accomplished. Thus, an immediate payoff was achieved from the capability provided in the mission support team.

Another key technical management feature utilized on Gemini was the capability to make changes in the field in an orderly and quick-reaction manner. This was done through preparation of a Field Change Notice as illustrated in Figure 15. The change could be prepared either at the launch site or at St. Louis. It was then cycled through a KSC change board consisting of both contractor and NASA personnel. It would then be fabricated and installed at the launch site, while being concurrently reviewed in St. Louis for application to subsequent vehicles. This capability for launch site changes permitted an extremely rapid turn around for repairs of anomalies found during checkout.

This ability to react quickly to required changes was given the ultimate test when it was decided to modify Gemini 7 so that it could be used as a rendezvous vehicle in conjunction with Gemini 6. This was prompted by the failure of the Gemini 6 target vehicle to achieve orbit. Figure 16 shows the time phasing of the various activities required. While Gemini 7 was in its factory checkout flow, spacecraft 5 mission was accomplished. This mission had some anomalies which initiated modifications to spacecraft 7. These mods were incorporated prior to the delivery date in early October. About three weeks after Gemini 7's arrival at KSC, the spacecraft 6 target vehicle failed to achieve orbit, and the launch of Gemini 6 was scrubbed. The decision was made a few days later to rendezvous the two manned spacecraft and this required another round of modifications to spacecraft 7 to permit it to be used as a target vehicle for spacecraft 6. By then it had reached its final checkout phase at KSC. In the remaining 5 weeks before launch, these changes were designed, tested, fabricated, installed and verified, and launch accomplished. Figure 17 lists the two types of modifications which were made; those due to spacecraft 5 anomalies, and those due to the Gemini 7/6 mission.

## Failure Analysis and Corrective Action

In order to assure that each mission had full benefit of the knowledge gained on previous missions, it was necessary to have a tightly closed loop for following up on failures and correcting anomalies. A system was established which assured thorough failure analysis and formal close out of any failure. Figure 18 illustrates the flow of activities involved in this closed loop failure analysis.

## Applications to the Airlock Module Program

As many of you know, the Airlock Program was conceived to make maximum use of previously developed hardware, particularly from the Gemini Program, to produce a subsystem core vehicle for the Apollo Applications Program. Since this involved a mission duration of up to 8 months, it was necessary to develop a cost-effective methodology for qualifying Gemini equipment for this greatly extended mission duration. This methodology is illustrated in Figure 19. It results in a systematic review of each component from the standpoint of its expected lifetime capability. This review can result in decisions ranging from use of the equipment without further testing, to modifying or replacing the part with one of increased lifetime capability. A sample review form is shown in Figure 20. For a given component each environment is considered individually. The part's previous qualification limits are compared to the Airlock qualification requirement. This detail comparison results in a decision to either accept the part, extend its test history, modify the part, replace it with a better part, move it to a less severe environment, make it redundant, etc.

To generate basic information needed to support the acceptability methodology described above, a limited number of Orbital Weathering Tests have been initiated on the Airlock Program. These tests are summarized in Figure 21. The vacuum facilities used in these tests are illustrated in Figure 22. Radiation testing was accomplished with the equipment shown in Figure 23. Results to date indicate that previous screening of materials for vacuum usage was valid. Attempts to accelerate vacuum exposure through an increase in test temperature have not yielded a correlation with real time test data. Certain equipment features, such as thermal coatings, lubricants, and surfaces subject to contamination, have indicated some degradation and require further attention.

## Summary

When considered as part of a symposium on long life hardware for space, the Gemini Program flight duration of 14 days seems modest.

However, it did achieve a major extension of previous manned flight duration and the management techniques and technical approaches utilized are felt to be valid for similar research and development programs.

Many of the subsystem components utilized on Gemini are currently planned for use in the much more ambitious 8-month flight duration of the Apollo Applications Program. Careful management of test resources supplemented by rigorous analysis of features affecting equipment lifetime provide a cost-effective avenue for development of the Airlock Module for this program.

FIGURE 1

# GEMINI PROGRAM OBJECTIVES

- RENDEZVOUS AND DOCKING
- CONTROLLED TOUCHDOWN SITE
- 14-DAYS DURATION
- REAL-TIME OBJECTIVES
  - EVA
  - GEMINI 7/6
  - AUGMENTED TARGET DOCKING ADAPTER

FIGURE 2

# GEMINI ORGANIZATION

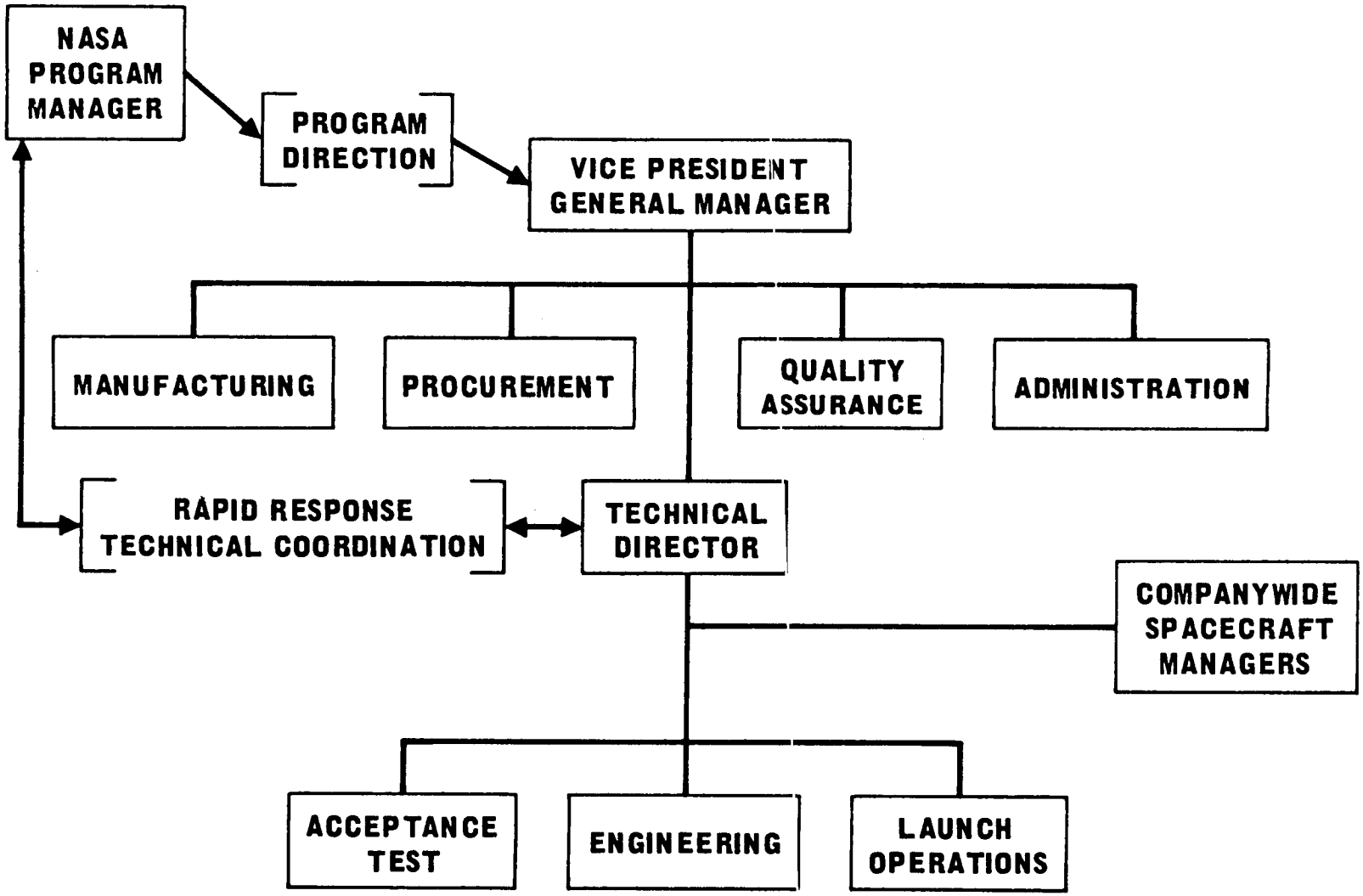


FIGURE 3

# ROUTINE MANAGEMENT UNSATISFACTORY

- PRODUCTION NUMBERS COMPARATIVELY SMALL
- EACH MISSION HAS UNIQUE REQUIREMENTS
- EACH FLIGHT MUST CORRECT PREVIOUS PROBLEMS
- COSTS MOUNT RAPIDLY DURING DELAYS
- CREW SAFETY A PARAMOUNT REQUIREMENT
- WHITE HEAT OF PUBLICITY EVER PRESENT

FIGURE 4

# MANAGEMENT PROCEDURES

- WELL-FORMULATED INCENTIVE CONTRACT
- STRONG CENTRAL POINTS OF CONTACT AT NASA AND MCDONNELL
- DAILY MANAGEMENT MEETINGS
- DELEGATED RESPONSIBILITY AND AUTHORITY
- COMPANYWIDE SPACECRAFT MANAGERS
- COMPANYWIDE SUBSYSTEM MANAGERS
- MONTHLY NASA/MCDONNELL MANAGEMENT MEETINGS
- NASA – SPONSORED EXECUTIVE MEETINGS

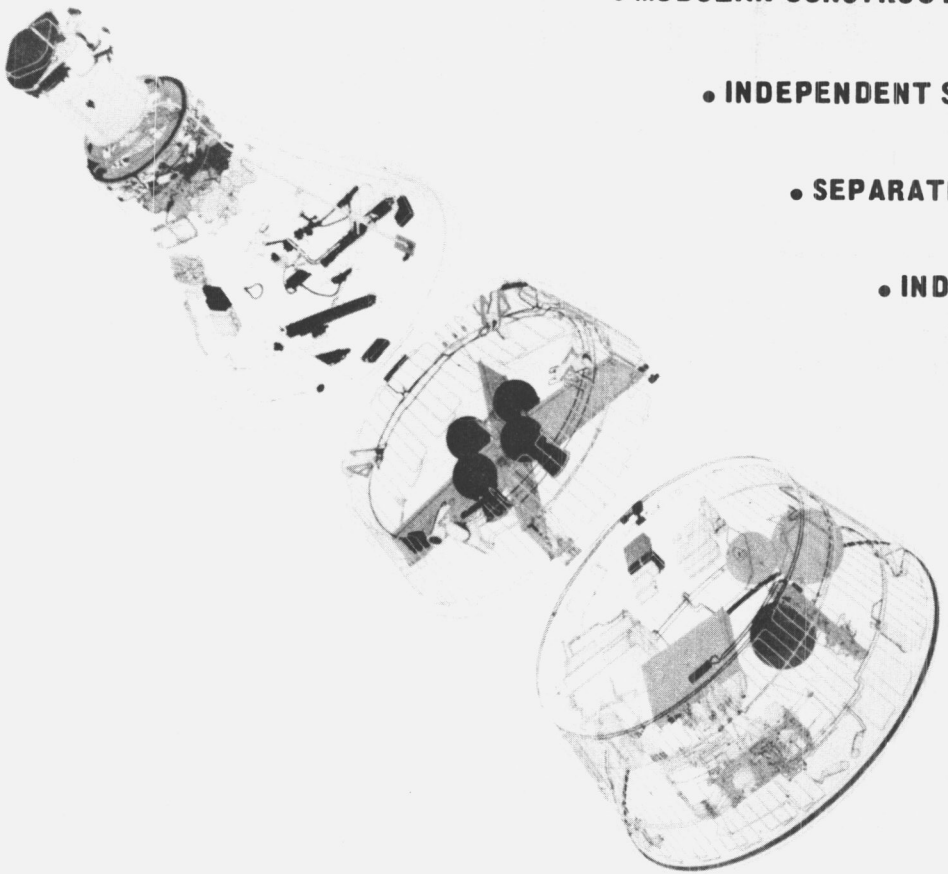
FIGURE 5

# TYPICAL CONFIGURATION TRADE OFFS

- ESCAPE TOWER VS EJECTION SEATS
- PURE OXYGEN VS MIXED-GAS ECS
- SOLID ROCKET VS LIQUID ROCKET RETROGRADE
- INTERNAL VS EXTERNAL EQUIPMENT LOCATION
- INTEGRAL VS MODULAR CONSTRUCTION
- AUTOMATIC VS MANUAL OPERATION

FIGURE 6

# OPERATIONAL DESIGN CONFIGURATION

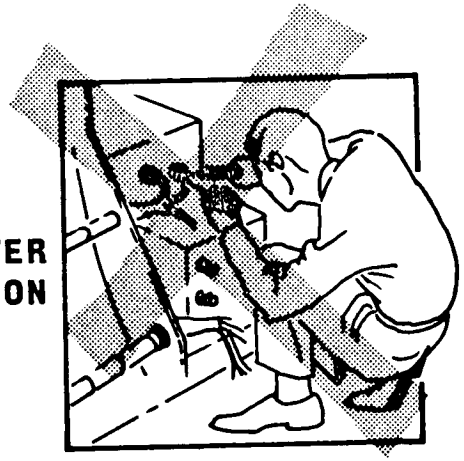


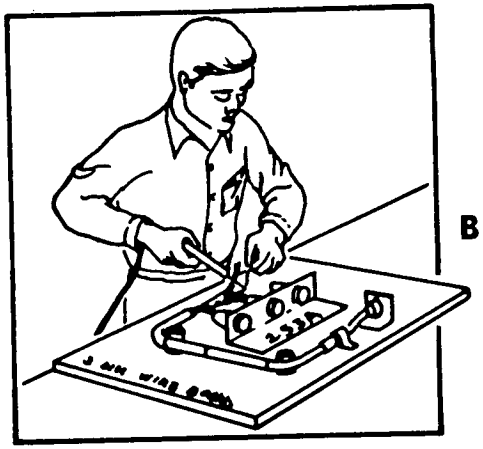
- MODULAR CONSTRUCTION
- INDEPENDENT SYSTEMS
- SEPARATE TEST POINTS
- INDIVIDUAL ACCESSIBILITY
- RELIABLE PROCESSES

FIGURE 7

# EASY INSTALLATION MINIMIZES ERRORS

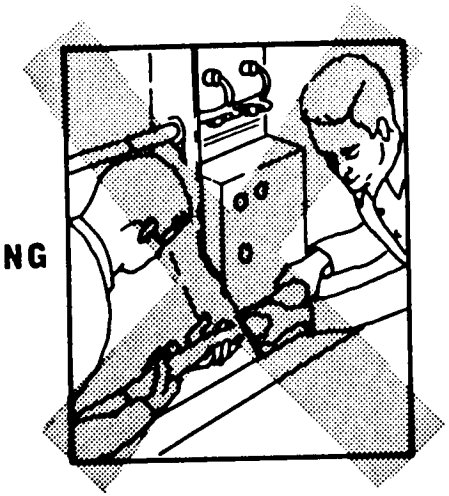
SOLDERING AFTER INSTALLATION

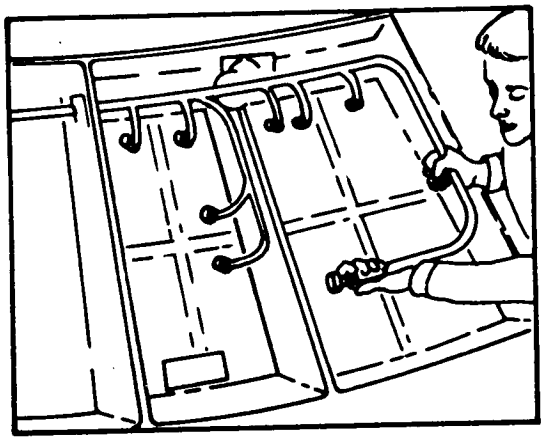




BENCH ASSEMBLY

CAPTIVE WIRING





FREE INSTALLATION

FIGURE 8

# TYPICAL DETAIL DESIGN FOR SAFETY

## PREVENTING RETRO IMPINGEMENT ON VOLATILES

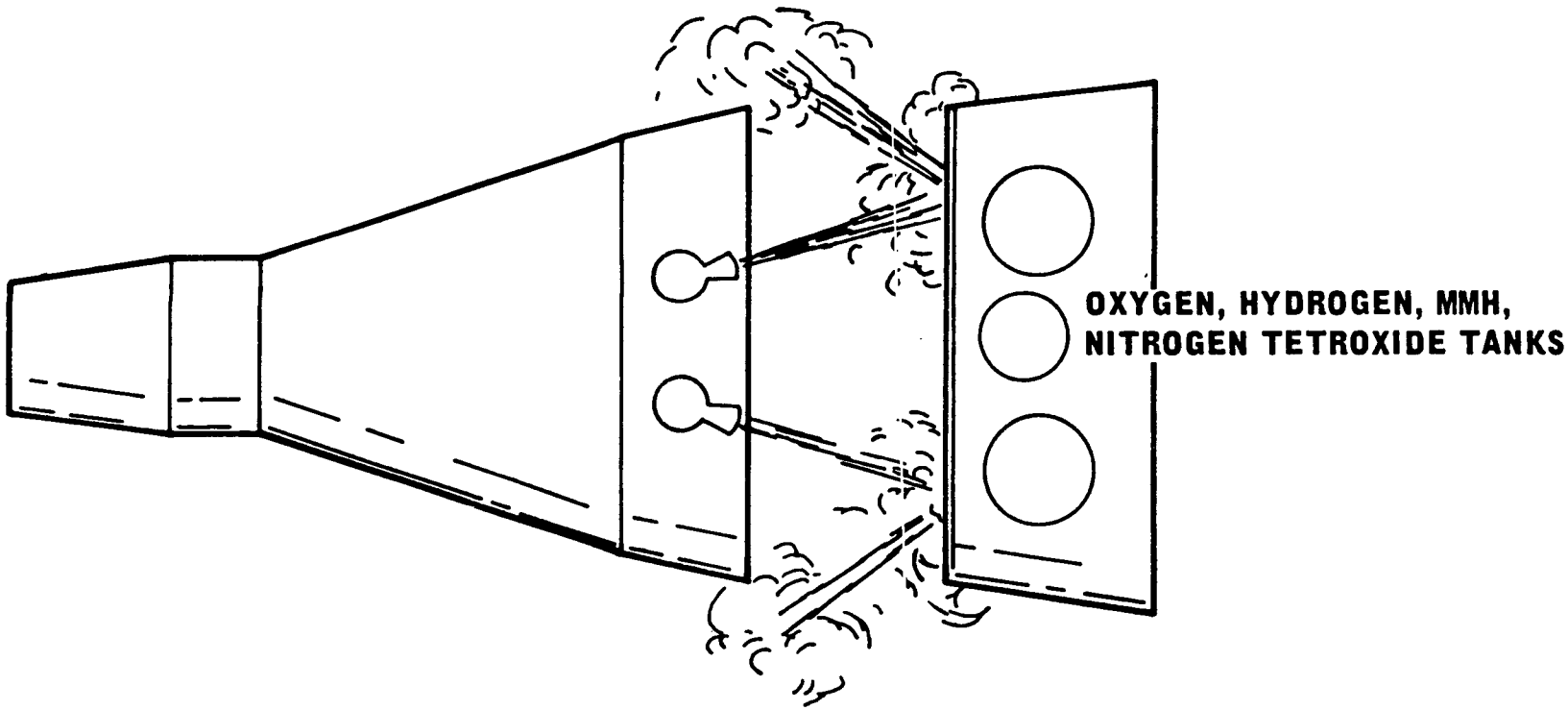


FIGURE 9

# MAJOR TESTS TO ENSURE MISSION DURATION

- FULL-SCALE THERMAL VACUUM
  - COMPLETE SPACECRAFT WITH ALL SUBSYSTEMS
  - 420 THERMAL MEASUREMENTS
  - $1 \times 10^{-6}$ TORR
  - 14 DAY DURATION
  - RESULTED IN INSTALLATION OF WATER & PROPELLANT HEATERS
- SUBSYSTEM MISSION SIMULATIONS
  - PROPELLANT STORAGE & THRUSTER ENDURANCE CYCLING
  - FUEL CELL ENDURANCE
  - CRYOGENIC BOIL-OFF
  - BATTERY WET STORAGE LIFE
  - ENVIRONMENTAL CONTROL MISSION SIMULATIONS

FIGURE 10

# GEMINI SUSTAINING ENGINEERING

- GEMINI DESIGN AND SYSTEM ENGINEERS PARTICIPATED IN:
  - ESTABLISHMENT OF REQUIREMENTS
  - DESIGN
  - FACTORY CHECKOUT
  - KSC CHECKOUT
  - REAL TIME MISSION SUPPORT
  - POST MISSION ANALYSIS
  - REDESIGN DUE TO MISSION ANOMALIES

FIGURE 11

# PERSONNEL FLOW

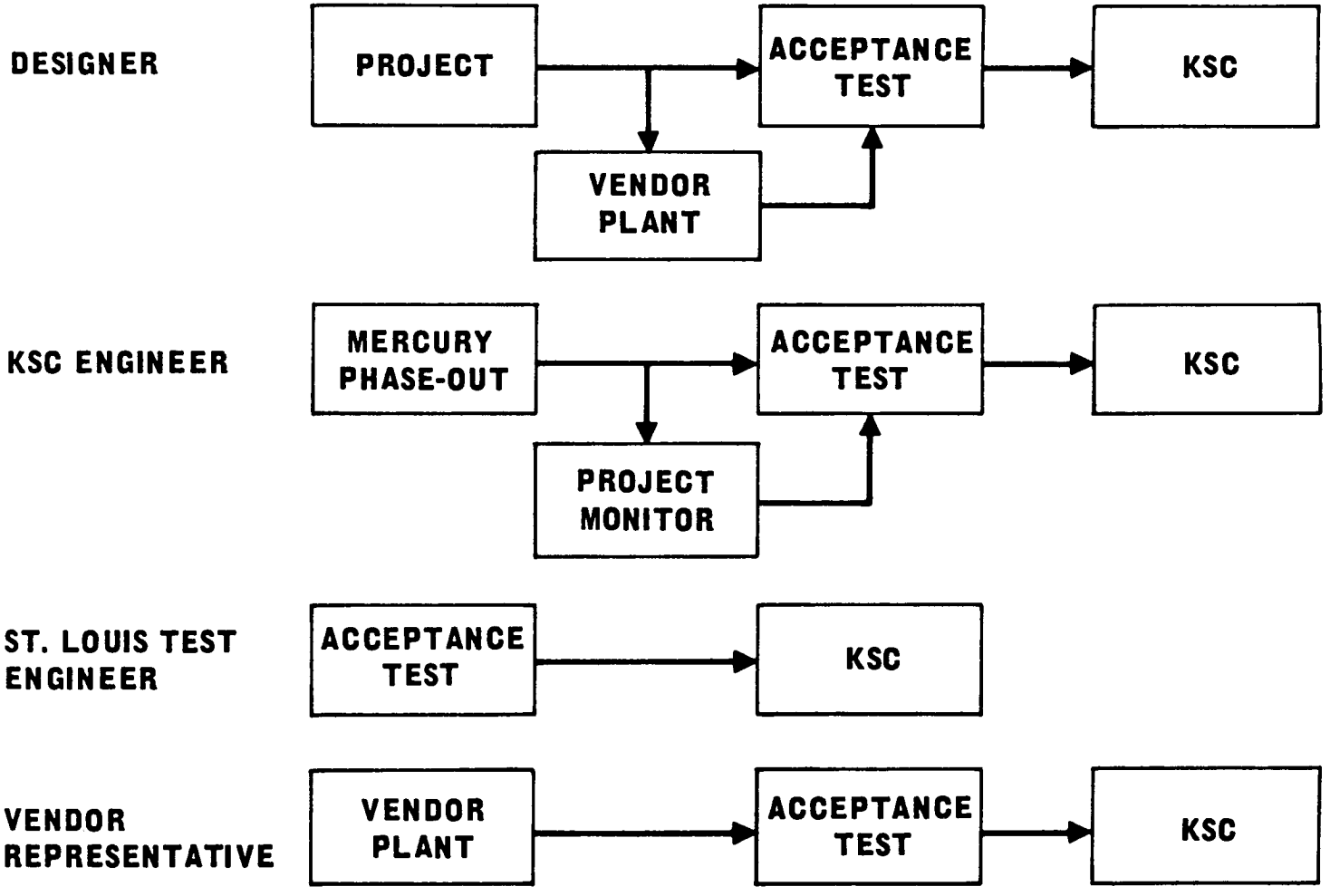


FIGURE 12

# GEMINI ENGINEERING MANPOWER

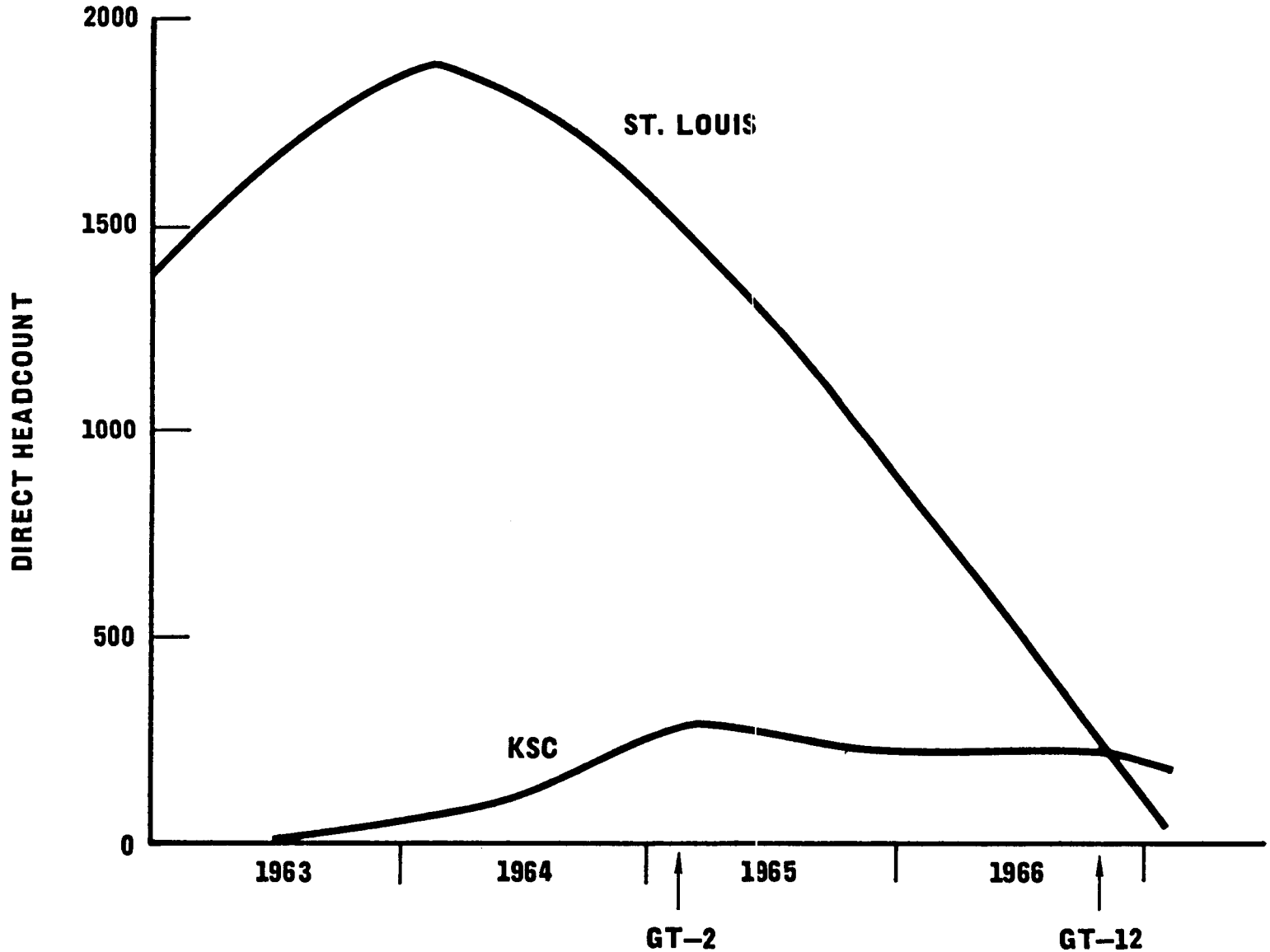


FIGURE 13

# MISSION SUPPORT

| | HOUSTON | KSC | ST. LOUIS |
|---|---|---|---|
| KSC TEAM → | KEY MANAGEMENT → | FULL TEAM | |
| ST. LOUIS TEAM → | KEY SPECIALIST → | | BACK-UP TEAM |
| HOUSTON MCC TEAM → | MCC CONSOLE SUPPORT | | |

FIGURE 14

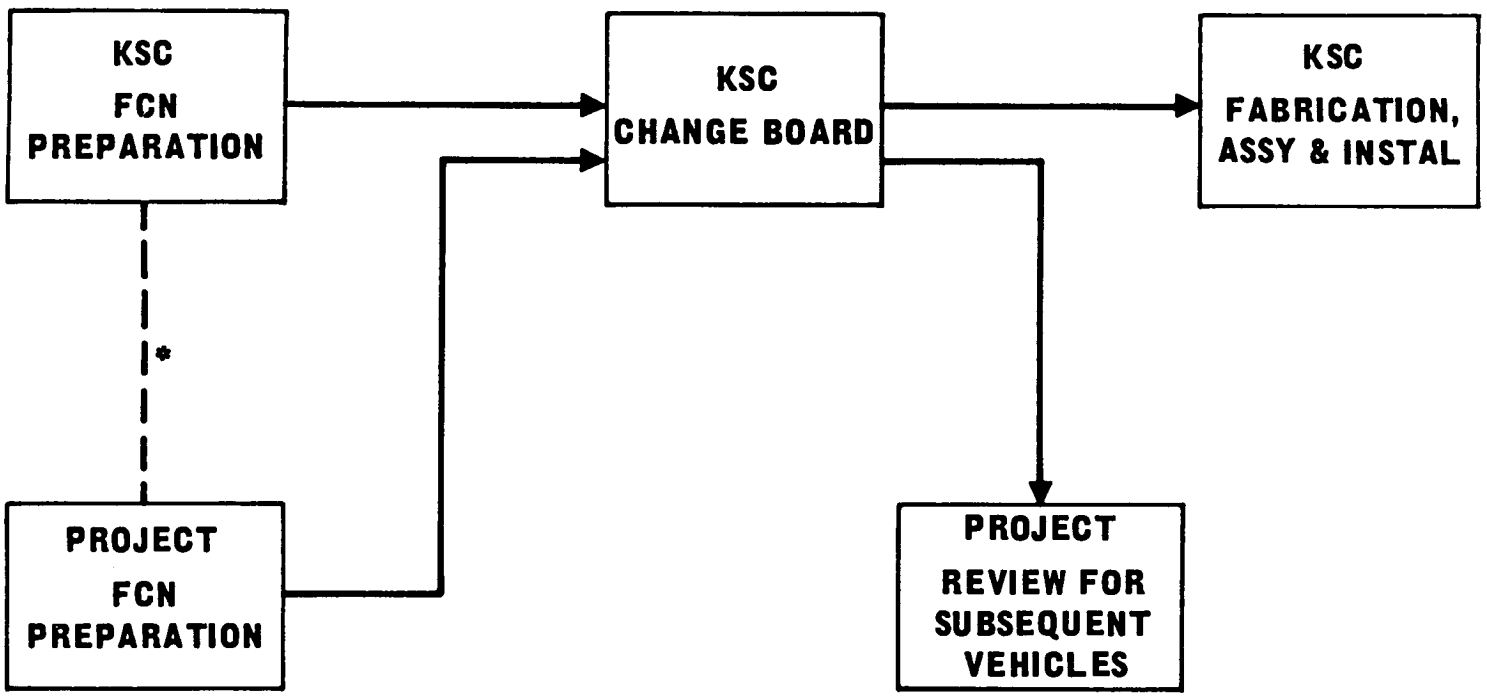
# SUPPORTING THE MISSION

- FULL TEAM CAN MAKE CRITICAL CONTRIBUTIONS
- GEMINI 5 CRYOGENIC HEATER IS AN EXAMPLE:
  - OXYGEN HEATER FOR FUEL CELL SUPPLY FAILED EARLY
  - CIRCUITRY TRACED TO ISOLATE PROBLEM
  - GROUND TEST FUEL CELL SYSTEM PUT INTO IMMEDIATE OPERATION TO CONFIRM SAFETY OF CONTINUING
  - MANY PEOPLE CONTRIBUTED TO VALID "GO" DECISION

FIGURE 15

# POST DELIVERY CHANGES

[FIELD CHANGE NOTICE (FCN)]




*OBTAIN DWG CONTROL NO. FROM KSC

FIGURE 16

# GEMINI 7 MODIFICATION PHASING

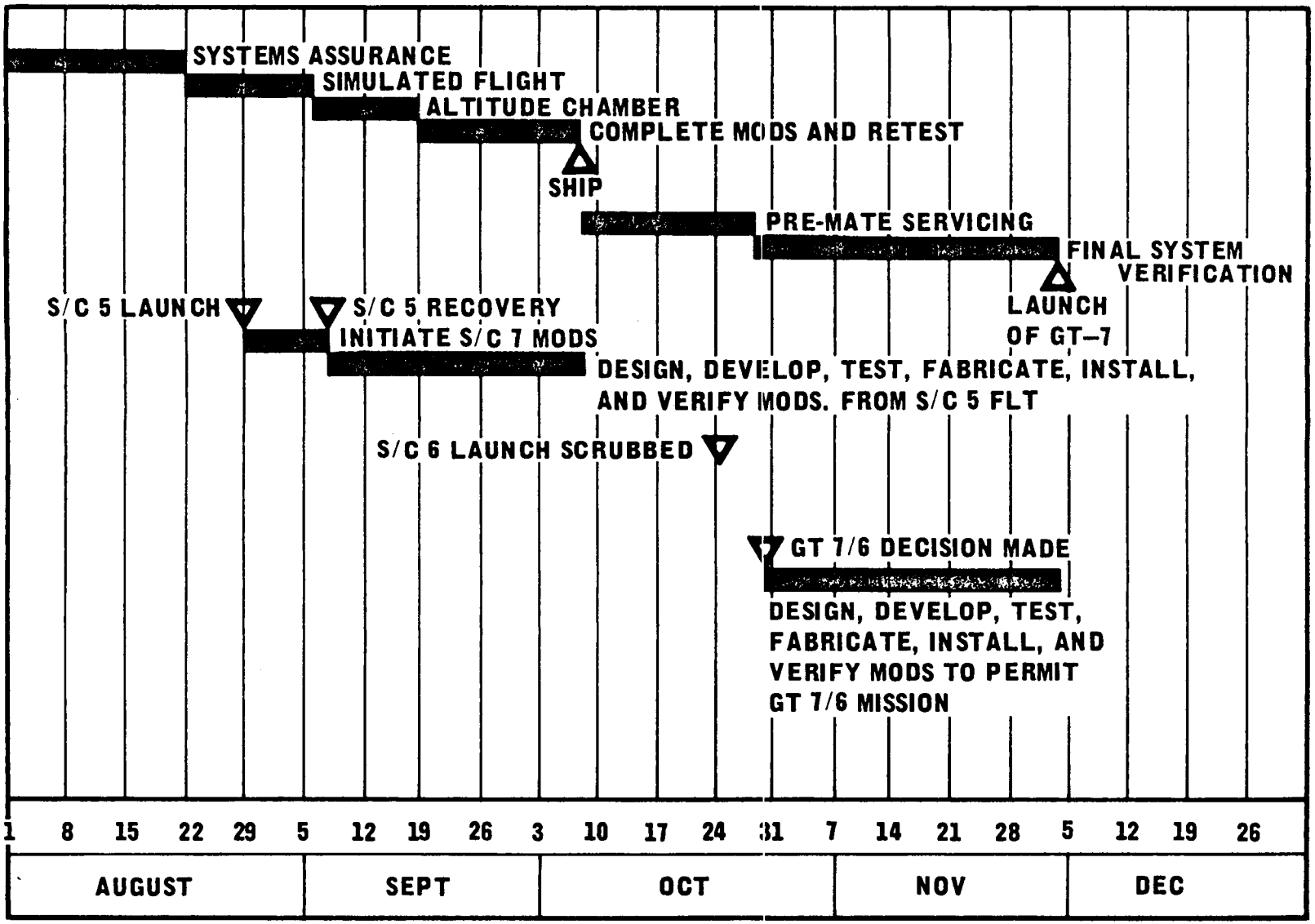


FIGURE 17

# GT-7 LATE MODIFICATIONS

- DUE TO S/C 5 FLIGHT
  - ADD URINE MEASUREMENT SYSTEM
  - MODIFY STOWAGE PROVISIONS
  - REVISE WATER SYSTEM
  - ADD RESERVE FUEL TANK
  - INTERCONNECT OXYGEN BOTTLES
  - REDESIGN $H_2$ BOTTLE MOUNTS
  - ADD $H_2$ REGENERATIVE COOLING
  - ADD CRYO ANNULUS VENTING
  - ADD REDUNDANT OAMS HEATERS
  - ADD LOW POWER COOLANT INVERTERS
  - REWORK WINDOW ENCLOSURES
  - MODIFY PCM RECORDER
- DUE TO GT 7/6 MISSION CHANGE
  - DESIGN TRANSPONDER INSTALLATION
  - ADD BOOST REGULATOR
  - ADD SPIRAL ANTENNA
  - ADD FLASHING LIGHTS
  - ADD FLASHING LIGHT BATTERY
  - ADD RELAY PANEL
  - ADD SWITCHES AND WIRING
  - CHANGE "C" BAND BEACON CODING
  - REINFORCE FLASHING LIGHT SCREEN
  - MODIFY TRANSPONDER ANTENNA SWITCHING CIRCUITRY INTERNALLY

FIGURE 18

# FAILURE ANALYSIS AND CORRECTIVE ACTION CONTROL

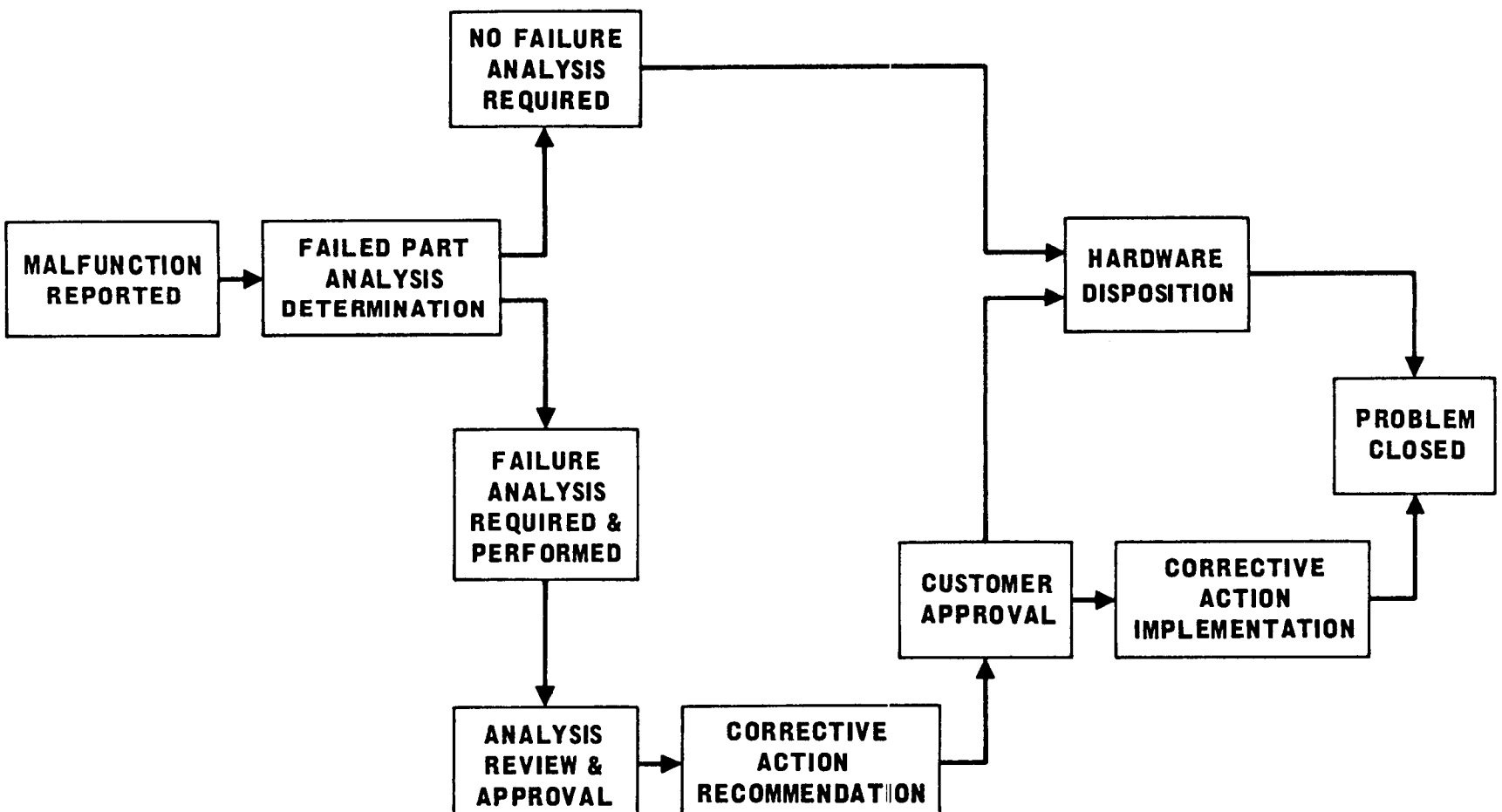


FIGURE 19

# ACCEPTABILITY METHODOLOGY

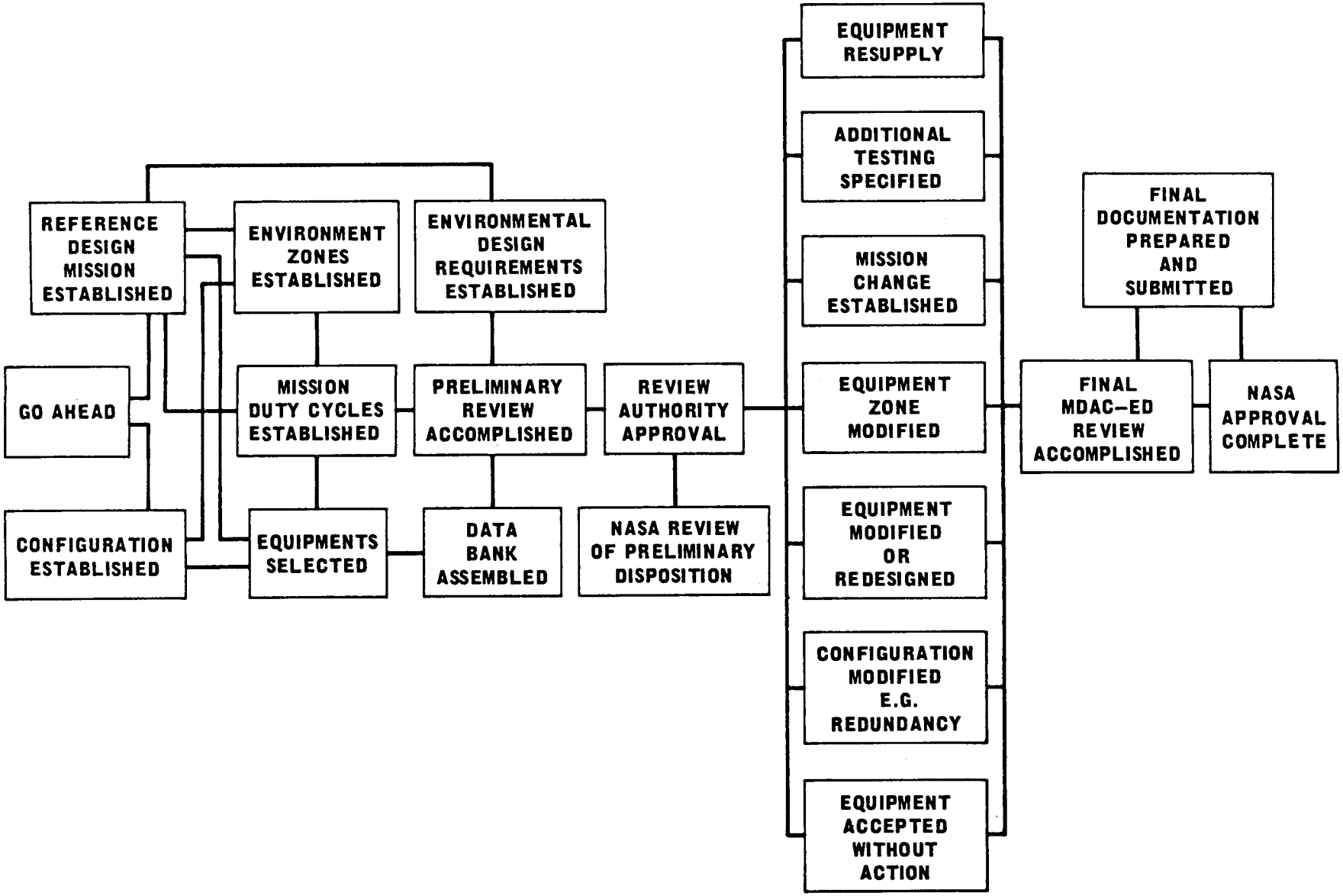


FIGURE 20

# AIRLOCK EQUIPMENT ACCEPTABILITY REVIEW

## SYSTEM: ENVIRONMENTAL CONTROL SYSTEM

PART NUMBER 52-83700-245
PART NAME SELECTOR VALVE
ORIGINAL PROGRAM GEMINI
REFERENCED DOCUMENTS USED IN EVALUATION

(SAMPLE)

MDAC-ED REPORT NO.: G499
DATE: 2 JANUARY 1969
REVISED: _____

SCD 52-83700
AIRESEARCH QUAL TEST RPT SS-2206, SS-2186, SS-2215, SS-2216
AIRESEARCH RELIABILITY DEMONSTRATION TEST RPT SS-2217 & SS-2218
MCDONNELL RPT. F319
MCDONNELL RPT 8433
GEMINI MALFUNCTION SUMMARY REPORT – EQUIPMENT FOR AIRLOCK

SUMMARY OF SIMILARITY:
PART TESTED 52-83700-245

GEMINI S/C FLOWN ON: 3 THRU 12

| 1. TEST ENVIRONMENT | 2. PREVIOUS QUALIFICATION LIMITS | 3. PREVIOUS GEMINI OR OTHER QUALIFICATION TESTS | 4. AIRLOCK QUALIFICATION LIMITS (PER MDAC-ED REPORT F319) | 5. AIRLOCK QUALIFICATION TESTS |
|---|---|---|---|---|
| VIBRATION | SINUSOIDAL EXCITATION TO BE APPLIED TWICE IN EACH OF 3 MUTUALLY PERPENDICULAR AXES. INPUT FREQUENCY VARIED LOGARITHMICALLY FROM 5 TO 2000 TO 5 CPS IN 15 MIN @ DOUBLE AMPLITUDE (±3G TO 100 CPS, ±5G TO 500 CPS, ±8G TO 2000 CPS). | TESTED PER COLUMN #2 REFERENCE: AIRESEARCH TEST REPORT NO. SS-2186, SECTION 11. UNIT S/N 302261 | LAUNCH: RANDOM VIBRATION AS FOLLOWS: FROM .02 TO .06$G^2$/CPS FROM 20 TO 200 CPS .06$G^2$/CPS FROM 200 TO 500 CPS. FROM .06 TO .01$G^2$/CPS FROM 500 TO 1000 CPS. .01$G^2$/CPS FROM 1000 TO 2000 CPS. OVERALL ACCEL LEVEL = 7.0G RMS. ORBIT: RANDOM VIBRATION AS FOLLOWS: .008$G^2$/CPS FROM 20 TO 500 CPS. OVERALL ACCEL LEVEL = 2.0G RMS. ALL VARIATIONS LINEAR ON LOG SCALES. | NOT REQUIRED. |
| SHOCK | LANDING SHOCK: 30G, HALF SINE PULSE FOR 11 MILLISECONDS. ULTIMATE SHOCK: 41G, HALF SINE PULSE FOR 11 MILLISECONDS. | TESTED PER COLUMN #2 REFERENCE: AIRESEARCH TEST REPORT NO. SS-2186, SECTION 11. UNIT S/N 302261 | ±2.5G's LONGITUDINAL COMBINED WITH 2.6G's ANY LATERAL FOR 100 MILLISECONDS. | NOT REQUIRED. |
| TEMP ALT | 160°F AT 15.5 TO 5 x $10^{-4}$ PSIA | TESTED PER COLUMN #2 REFERENCE: AIRESEARCH TEST REPORT NO. SS-2215 AND SS-2216. UNIT S/N 302263 | TEMPERATURE TESTS CONSTITUTE A SUFFICIENT TEST OF THIS PARAMETER. | NOT REQUIRED. |

(FORM 1) PAGE _____

FIGURE 21

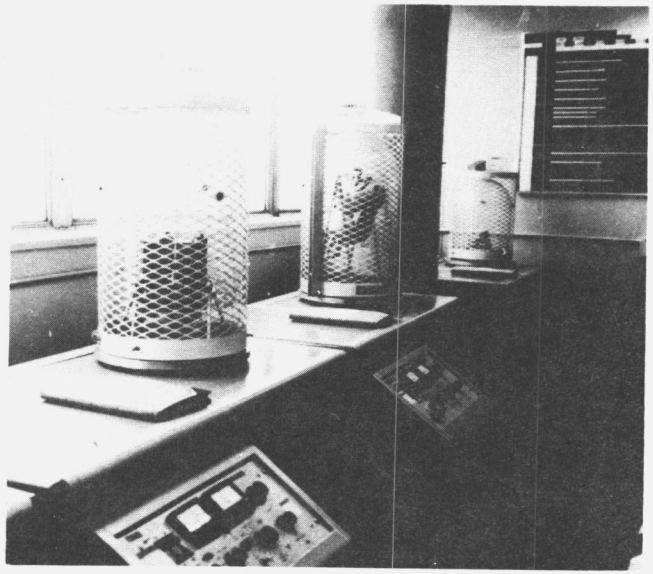
# AIRLOCK ORBITAL WEATHERING TESTS

- CORROSION (95% R.H., 100% $O_2$, 120°F, 90 DAYS)
- VACUUM (4 x $10^{-9}$ TORR., 75°F, 12 MONTHS)
- ACCELERATED VACUUM (1 x $10^{-6}$ TORR., 155°F, 30 DAYS)
- RADIATION (300 N.M. EQUIVALENT, PROTONS, ELECTRONS, UV, 1 YEAR)
- CONCLUSIONS:
  - CORROSION ANOMALIES (LUBRICANT DEGRADATION, SURFACE CONTAMINATION) REQUIRE ATTENTION
  - PRE-SCREENED MATERIALS WERE VALIDATED IN VACUUM TESTS
  - VACUUM ACCELERATION GOOD FOR INITIAL SCREENING BUT NOT CORRELATED COMPLETELY WITH REAL TIME
  - THERMAL COATINGS REQUIRE ATTENTION

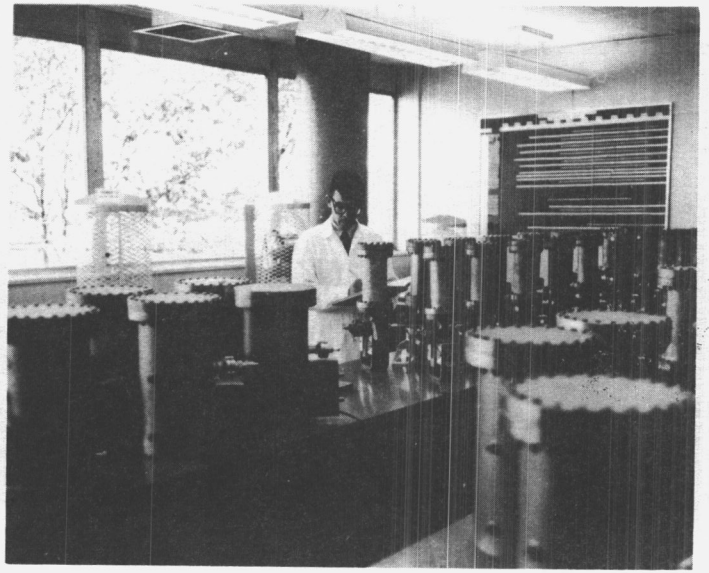
FIGURE 22

# AIRLOCK
# Extended Mission Testing

VACUUM TEST FACILITIES
- REAL TIME OR ACCELERATED
- SMALL CHAMBERS – 40
- MEDIUM CHAMBERS – 28
- LARGE CHAMBERS – 5



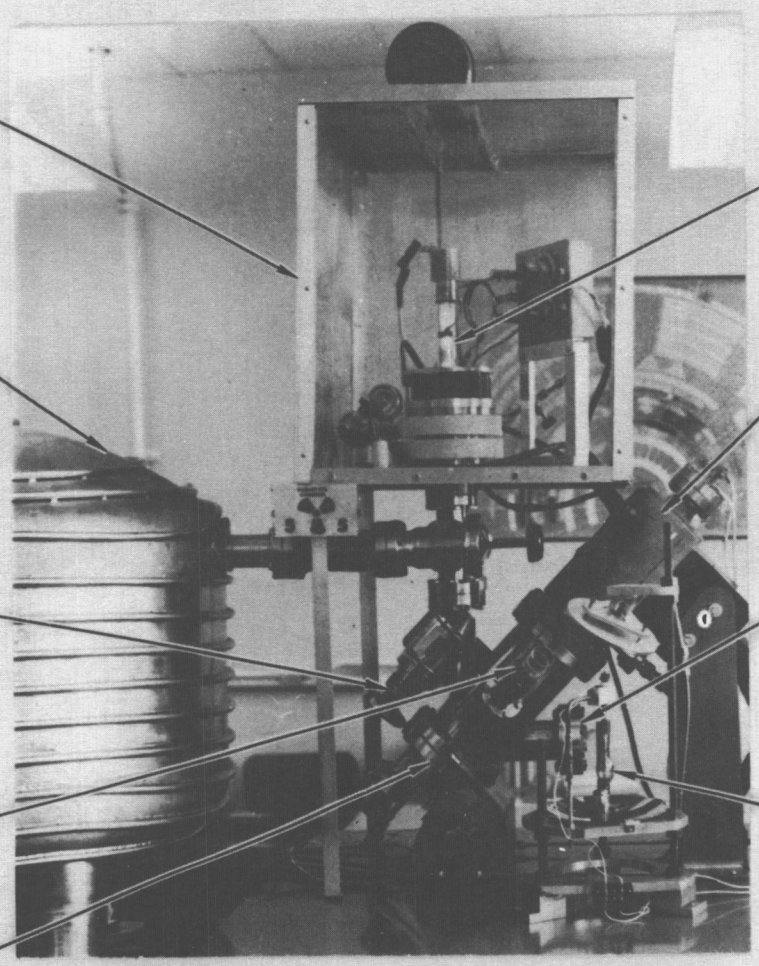
LARGE BELL JAR CHAMBERS



ION PUMPED, SMALL & MEDIUM
STEEL CHAMBERS

FIGURE 23

# AIRLOCK RADIATION TEST FACILITY



# TIROS AND ESSA SATELLITE DESIGN AND TEST FOR MISSION SUCCESS

by

A. Schnapf
TIROS/TOS Project Manager
RCA Astro-Electronics Division
Princeton, New Jersey

The TIROS and the ESSA meteorological satellites have been one of this country's most eminently successful unmanned satellite programs. During this first decade of space, TIROS has progressed steadily from a research and development program in 1960 to a semi-operational meteorological satellite system between 1961 and 1965, and to the world's first global operational system with the launching of the TIROS Operational System (TOS) spacecraft, ESSA 1 and 2 in February 1966. The TOS program will continue to provide operational satellite systems with additional TOS spacecraft, and in the near future the second generation Improved TIROS Operational System (ITOS) will be introduced to provide further improvements in the operational system.

During these nine years of TIROS and TOS space operations, a total of 18 consecutive mission successes have been achieved by 10 TIROS and 8 ESSA satellites, as shown in Figure 1. These spacecraft were built by RCA Astro-Electronics Division under the technical management of NASA's Goddard Space Flight Center. The operational satellites are funded and operated by the U.S. Department of Commerce's Environmental Science and Services Administration.

The TIROS and ESSA programs objectives have been met and exceeded by these satellites, assuring the scientists and weather forecasters of this country, as well as the users and researchers from many nations abroad, continuous daily global satellite observation without interruption. Typical weather data are shown in Figure 2.

The TIROS and TOS programs have demonstrated how satellites can be employed for peaceful use by all of mankind on this ever-shrinking planet. These programs have also shown the reliability and the potential longevity of unmanned satellites, as shown in Table 1, and their ability to provide

routinely, vital operational data that can readily be interleaved with earth-bound conventional data. Table 2 shows the useful life of the TIROS and ESSA satellites. More importantly, the programs have demonstrated that these objectives can be realized on a cost effective basis. Table 3, extracted from reference 1, compares various NASA unmanned spacecraft programs as to cost.

The success of the TIROS and ESSA satellites and the achievements of this project can be attributed mainly to the following factors:

a. A sound system concept, with a baseline design that permitted growth and accommodation for improved sensors.

b. An engineering reliability and quality program to assure the adequacy of the space hardware.

c. Detailed development tests and prototype and flight acceptance tests at component, black box, and spacecraft level to demonstrate the capability of meeting the expected launch and mission environment and life.

d. A detailed spacecraft calibration and measurement program to enable evaluation of the satellite's performance in orbit and to provide key data to the operational centers.

e. A spacecraft launch and post-launch support team to assure proper spacecraft handling, testing, and post-launch evaluation.

f. The reliable Delta launch vehicle.

g. A Project Team effort by government and industry personnel.

The TIROS Research and Development spacecraft design was initiated in 1958, a point in time when the space frontiers were barely penetrated and the U.S. space program was still in its infancy. During this period, the TIROS I satellite, launched on April 1, 1960, was probably one of the more complex and weightier spacecraft.

The primary objective of the program was to demonstrate the feasibility of observing the earth's cloud cover from an earth-orbiting satellite by means of slow-scan miniaturized television camera systems. The secondary mission objective was to demonstrate the spacecraft design in the following areas:

a. Spin attitude stability.

b. Satellite dynamics control, including despin, spin-up, and nutation control.

c. Power subsystem consisting of solar array, voltage regulator, and rechargeable power supply.

d. Communication subsystem including command receiver, telemetry beacon, primary data, transmitter, and RF antennas.

e. Command and control, covering programming for direct and remote operations, timing of camera and recorder events, and switching two complete chains of subsystems.

f. Compatibility between spacecraft and the TIROS Command and Data Acquisition Stations and Control Center.

g. Spacecraft and launch vehicle compatibility.

h. Spacecraft passive thermal design.

i. Satellite capability of surviving the launch environment and operations in the space environment at a 400-n. mi. near-circular earth orbit.

These objectives were successfully demonstrated by the mission of TIROS I. The TIROS I spacecraft design, shown in Figure 3, was constrained in diameter and weight by the capability of the Thor-Able launch vehicle in the 1958-59 period. The spacecraft weight limit was set at 275 pounds in order to enable the rocket to place the spacecraft into a 400-nautical mile near-circular orbit with a 48-degree inclination angle to the equator. The satellite diameter of 42 inches was set by the maximum space available within the bulbous, low-speed nose fairing of the Thor-Able vehicle. The solar array height and 18-sided polygon hat box shape were governed by the solar array area and the storage capacity of the NIckel-Cadmium cells needed to support the day and nighttime spacecraft loads. Additionally, considerations were given in the array design to permit a limited amount of degradation due to the space environment and still provide adequate power output to perform the mission.

The flat drum-like baseplate was designed to provide a large clear surface for mounting the black boxes and a structure that was capable of meeting the static and dynamic loads. (During this early phase, the Thor-Able load factors were not too well known since the 3-stage rocket had not made its maiden flight.) Very conservative prototype design loads of 2 to 3 times

the expected flight loads were established. Additionally, the large flat mounting surface permitted flexibility in layout of boxes and harness, suitable thermal interface of units to the baseplate (which in orbit became the thermal heat dissipator), and accurate alignment of the primary camera sensors and other optical devices with the spacecraft spin axis and missile mating surface.

The limitations of weight and size contrained the design of the sensors and the associated electronics to be lightweight. Additionally, these components were designed to consume minimal power since the total daytime average load the solar array was able to support was limited to 35 watts.

The television subsystem utilized a 1/2-inch slow-scan vidicon television camera with a two-second readout. This unit was designed for low power and minimum size, and was compatible with the video tape recorder used for storage of picture data.

Spin stabilization, because of its simplicity and reliability, was selected to keep the spin axis oriented in the orbit plane. The spacecraft spin was maintained between 8 and 12 rpm. This speed was sufficiently high to keep the spin axis stable and low enough to avoid picture smear.

The two TV cameras, one wide-angle (104 degrees) and the second narrow-angle (12 degrees), were mounted with their optical axes parallel to the spin axis; the lens of each camera projected through the underside of the baseplate. In this configuration the cameras only viewed the earth for one-quarter of each orbit since the spin axis was inertially fixed in space. Figure 4 (upper view) shows the orbital coverage of the early TIROS I to VIII satellites. A continuous strip consisting of 32 overlapping pictures was taken covering a scene 800 miles wide and 6,000 miles long. During a 24-hour period of operations, a 14-orbit sequence would scan approximately one-quarter of the sunlit earth.

With this brief description of the TIROS I mission objectives and design constraints and the initial system configuration, it would be appropriate at this time to examine a few of the system design details to show how the design of a decade ago provided the foundation for a spacecraft that was not only able to meet its primary and secondary objectives, but also provided a baseline design that permitted expanded mission capability and growth, and evolved into a highly reliable global operational system. Figure 5 shows the spacecraft evolution during this period. Despite the increased requirements from the standpoint of data output in quantity and quality, each of the new blocks of spacecraft provided increased length of life and continued to maintain cost effectiveness.

The design philosophy and techniques employed on the TIROS program were the key to the development of a long life and flexible spacecraft. The approach used was the application of sophisticated systems concepts translated into relatively simple and very reliable hardware.

The selection of spin stabilization to provide spin-axis attitude control was well within the constraints of television camera optical requirements from the standpoint of pointing and picture smear. The maintenance of the satellite's spin rate within the 8 to 12 rpm limits was initially achieved by the employment of 5 pairs of thumb-size solid rocket motors that were mounted on the periphery of the baseplate. Figure 6 shows these spin-up rockets. Whenever the satellite's spin rate decayed to a value below 8 rpm (primarily due to magnetic drag), a pair of rockets were fired to increase the spin rate approximately 3 rpm. The 5 pairs of rockets were capable of keeping the speed of the satellite above the 8 rpm limit for a period in excess of two years.

One of the more significant design innovations to the TIROS design was implemented on TIROS II, launched November 23, 1960. The new design was a result of the post-launch evaluation performed on TIROS I. It was observed and later computed that the TIROS I satellite's spin axis was slowly precessing sinusoidally over a 60-day period. This phenomenon was computed to be the interaction between the earth's magnetic field and the satellite's magnetic dipole moment. From the mathematical model established as a result of the phenomenon, a very simple lightweight coil was installed on the periphery of the satellite in a plane normal to the spin axis. Provision was made to program electric currents of varying magnitude and polarity through the coil so that a controlled magnetic field was generated in the vicinity of the spacecraft (see Figure 7). This technique resulted in a net magnetic torque which, in turn, precessed the satellite spin axis to precalculated positions. The method was used successfully on all subsequent TIROS spacecraft.

Further refinement to the satellite's attitude control techniques was made on TIROS IX, the first wheel-mode satellite, in which additional and programmable magnetic coils were added to provide very precise spin-axis control such that stabilization of ±1 degree was established in all three axes. This same system permitted the satellite's spin axis to be maneuvered by magnetic torquing from the initial injection position in the plane of the orbit to a position 90 degrees or normal to the orbit plane. In the wheel-mode attitude the spinning spacecraft with its radially mounted television cameras was able to observe the earth with each revolution of the satellite about its spin axis, or approximately every 6 seconds. (Figure 4 compares the standard versus the wheel-mode configuration.)

Furthermore, on TIROS IX and all ESSA satellites, the addition of redundant magnetic spin control coils permitted more precise control of the satellite spin rate to a precision of ±25 milliseconds throughout the life of the satellite. This improvement not only resulted in a very long life and a more reliable device, but also permitted precision control of spin-up or spin-down of the satellite spin speed, thereby permitting the spin rate of the spacecraft to be used for timing or clocking key operational functions and simplifying the processing of the operational data by the users. The spin control technique is also employed on ITOS to adjust the spacecraft momentum of the system by commutating spin control coils.

TIROS IX, the first wheel-mode satellite, shown in Figure 3, was launched in January 1965 with the objective of expanding the capability of the TIROS satellites to provide complete global weather observation on a daily basis. This coverage represented an increase of four times the daily global observation provided by the predecessor TIROS satellites.

With its new design, TIROS IX differed from its predecessors in many aspects, and was the forerunner of the ESSA satellites now used in the TIROS Operational System. Figure 8 depicts the TOS mission.

The primary differences that resulted in daily global readout of the earth's weather are the reconfiguration of the satellite's sensors and sensor triggering system, the addition of enhanced magnetic torquing for maneuvering the spin axis and spacecraft spin rate, and finally the utilization of a sun-synchronous near-polar orbit. In TIROS I through TIROS VIII, the two TV cameras were mounted on the baseplate of the satellite with the optical axes parallel to the inertially stabilized spin axis. Hence, the camera axes were parallel to the orbit plane and viewed the earth for about 25 percent of each orbit. In the TIROS IX configuration and in the TOS spacecraft, the TV cameras were mounted diametrically opposite one another and looked out through the sides rather than through the baseplate of the satellite. The satellite was injected into orbit with the spin axis in the orbital plane; however, the spin axis was then maneuvered by an improved magnetic-torquing system to an attitude normal to the orbit plane. Figure 6 depicts the coils used to orient the spin axis; Figure 9 shows the orientation technique. Thus, the spinning satellite "rolled" along its orbital path and the field of view of each camera passed through the local vertical once during each spin or "roll". At the proper interval in the picture-taking sequence, the camera shutter was triggered to take a photo of the local scene when the camera was looking down at the earth. Hence, throughout the sunlit portion of the orbit, the earth below the satellite could be observed by means of a sequence of overlapping photos.

By placing the wheel satellite in a near-polar, sun-synchronous orbit, the entire earth could be observed on a daily basis. Figure 10 shows the first photo of earth taken by TIROS IX in a 24-hour period.

Spin stabilization and the magnetic torquing and spin control technique have been extended in the second generation Improved TIROS Operational System (ITOS). TIROS M, the forerunner of the ITOS system, utilizes Stabilite, a momentum flywheel system, in which the spinning flywheel coupled through a bearing to a despun platform is used to maintain the required pitch axis attitude stabilization. Figure 11 shows the application of this mode of operation. The flywheel is colinear with the pitch axis and contains a scanning mirror that enables fixed IR bolometers to detect sky-earth transitions in this field of view. The satellite pitch control system regulates the speed of the motor-driven flywheel based upon position and rate signals derived from the IR bolometer.

TOS-type QOMAC* coils are used to correct roll and yaw errors, as well as to perform the initial orientation maneuver. MBC** coils are utilized to correct for the residual magnetic dipole and provide the one degree-per-day precession rate required to track the orbit regression of a sun-synchronous orbit. A magnetic spin-control coil will provide for the adjustment of momentum about the pitch axis, and liquid dampers will reduce satellite nutation.

The TIROS M/ITOS design is a further extension of the existing spin stabilization technology developed and proven on the earlier TIROS and ESSA satellites. As a result of the stabilization scheme, there is a significant improvement in the ability now to provide a large earth-oriented platform for a combination of Automatic Picture Transmission (APT) and Advanced Vidicon Camera System (AVCS) daytime television cameras, day and night dual channel scanning radiometer, solar proton monitor, and a flat plate radiometer. Figure 12 shows the TIROS M/ITOS configuration. Not only will the single TIROS M spacecraft be capable of performing the mission of two of the present ESSA satellites, one APT and one AVCS, it will also provide for the first time operational day and night infrared and visible radiometric data. Global observation of the earth's cloud cover will now be provided every 12 hours with a single spacecraft instead of every 24 hours with two of the present ESSA satellites. Here again, the predominant objective is to upgrade the operational data in quantity and in quality and still permit cost-effective operation, and continue a baseline that can accommodate further growth and enhancement of the system.

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* QOMAC, acronym for quarter orbit magnetic attitude control

** MBC, magnetic bias coil

The mechanical nutation dampers are another example of a relatively simple and reliable device that was capable of performing a sophisticated assignment. When the TIROS or ESSA spacecraft is separated from the third-stage rocket, the spacecraft is spinning at 125 rpm due to the ballistics requirements of the rocket. At the time of separation, the spacecraft is coning slightly either due to unsymmetrical separation forces or due to initial coning imparted by the third stage during its coast phase after burnout and prior to separation. At separation, the spacecraft separation switch activates a power circuit to fire a pair of squibs, removing a pin that uncages a bearing-mounted-weight which rolls down on a gently curved monorail. (If the automatic circuit fails to uncage the dampers, a back-up command can be sent up to the spacecraft to uncage the dampers.) This weight or cart is caused to oscillate on the monorail due to the wobble or nutation of the spin axis. The oscillation of the cart along the rail dissipates the nutational or kinetic energy and quickly reduces the nutation of the spin axis to less than 1/4 degree. This device, termed a TEAM (Tuned Energy Absorption Mechanism) damper, was used on the TIROS series of satellites; it is shown in Figure 6. In the ESSA series, the system was refined by the use of a simple, but efficient, redundant liquid damper that removes the nutation by the friction of the fluid within the inner tubular wall. The spin axis nutation with the liquid dampers is held below 0.1 degree. Both devices described are simple and highly reliable and capable of long life.

The design of the attitude and camera trigger infrared sensors on TIROS IX and ESSA wheel-mode spacecraft again exemplifies the utilization of simple devices to perform a critical and complex function. The IR horizon sensors, after scanning the cold sky, detect the warm leading edge of the earth's horizon and trigger the shutter of the radially mounted camera at the time the camera's optical axis is passing through local vertical. Figure 13 shows schematically this IR trigger. This precise camera pointing and triggering effectively permits the spin-stabilized platform to perform as well as a three-axis earth-stabilized platform. The triggering accuracy is better than 99 percent and the pointing of the camera to local vertical is within one degree in pitch, roll, and yaw. The same pair of redundant horizon sensors used to trigger the camera shutter are also used to commutate the magnetic spin control coil. With each revolution the diametrically opposed IR sensors reverse the current flow in the coil, thereby making the spacecraft act like a synchronous motor in space.

A pair of infrared horizon sensors are also utilized to provide attitude information. In Figure 14 it can be seen how the measured pulses between the leading edge and trailing edge of the earth's horizon for each of the infrared

detectors in the V-head sensor provide an error signal to CDA* stations when the spacecraft axis is off the orbit normal. The magnetic attitude control coil (QOMAC) is turned on by the CDA station to remove the error and, hence, restore the spin axis to orbit normal. The combination of satellite magnetic spin control of the body, and spin axis attitude control by the QOMAC, provides a very reliable, low power, long-life attitude and spin control system. With no consumable products to be dissipated the satellite can be kept under three-axis control to within ±1 degree for extremely long-term operation. This performance matches that of more sophisticated systems which are more complex and heavier, require greater power, and utilize life-limiting stored-gas control jets.

One of the more basic design philosophies for assuring long life on the TIROS and ESSA satellites is the employment of redundancy throughout the spacecraft. Figure 15 is the spacecraft block diagram showing this redundancy. In addition, the design permits cross-coupling of units so that elements from one redundant chain can be brought on line with other selected elements of the second redundant chain. In other words, if the camera on system No. 1 fails, then camera 2 can be selected for use with the remaining units in camera 1's chain, such as recorder, transmitter, decoder, programmer, etc.

This feature has permitted many of the TIROS and ESSA spacecraft to fulfill the mission objectives of daily global observations despite failure or degradation of some components in orbit.

The design for long life on the TIROS and ESSA satellites is further assured by the application of design techniques that facilitate spacecraft testing. As will be discussed later in this paper, the testing, calibration, and handling of the spacecraft are important operations in the TIROS and TOS programs. Anywhere from 6 to 10 months of effort go into the build-up, integration, debugging, test, and calibration of a satellite. With this important phase of the program in mind, the TIROS M/ITOS spacecraft was designed to be a simple box-like structure that can be readily disassembled or buttoned up for the various phases of the above-mentioned tasks. Figures 16 & 17 show the ability to work initially with the primary equipment panels separately, then in an open or horizontal configuration. As required the side walls are easily raised to button up the spacecraft for final tests. The large flat equipment panels permit flexibility in configuring and mounting new mission equipment. The two side panels at present are configured to carry the primary television sensors, their associated electronics and data storage recorders. The baseplate houses the

* Command and Data Acquisition

equipment for the bus functions, such as power, telemetry, communications, command, and control. The "bus" concept has been one of the key design features of TIROS M/ITOS; it will permit growth and alternate missions with minimum redesign.

During the spacecraft system and subsystem and black box design, a series of design reviews are held. The first is at the conceptual level; the second is a major design review at the time of design release; and the third, a follow-up review after the prototype tests are completed. The designers are required to utilize components listed in RCA's established spacecraft standards. These standards reflect the data available from NASA and other government agencies. All nonstandard parts must be given special tests to be certified as acceptable for space use.

A parts application program is established at the beginning of the design and is one of the items studied during the design review. Particular emphasis is placed on the derating of components and the actual application of each component in the circuit. Since the ESSA spacecraft are placed into a near-polar, 790 n.mi. sun-synchronous orbit, the radiation effect from the Van Allen belt and the Starfish belt is carefully analyzed for all components and materials utilized on the TOS spacecraft. Components are radiation tested to assure that there is an understanding of how they are affected by radiation. The circuit designer reviews his design for beginning and end-of-life conditions, taking into account the possible radiation degradation effect and temperature variations. This conservative design approach has permitted long-term operation of the TOS spacecraft in an orbit that is relatively hostile by virtue of the presence of radiation.

Although there is some loss of power due to radiation damage or degradation of solar cells, the array is sized with sufficient reserve to provide ample end-of-life power for the mission. The rechargeable nickel-cadmium batteries are operated with a shallow depth of discharge of 5 to 15 percent to assure long life.

All circuits and boxes are analyzed for potential failure modes. The design precludes that a single failure can result in a catastrophic loss of the mission.

Fuse protection of key circuits for each chain is included in the design to assure isolation of a unit or subsystem in the event of a high current drain or short, thus permitting safe operation of the balance of the system.

The procurement and test phase is another important phase of the program that assures that the basic design and the workmanship are adequate for achieving mission success. The extensive test program permits the weeding out of early life failures and provides a record of the spacecraft signature and calibration which is utilized as a baseline to evaluate the spacecraft performance once in orbit.

The test program for the TIROS and TOS spacecraft systems starts at the component level and continues through various levels of assembly such as board levels, box level, subsystem level, and, finally, spacecraft level. Through each of these levels, the type and duration of the tests vary. For each black box there is a series of tests that cover the breadboard phase, mechanical and electrical test models, the prototype, and, finally, the flight units.

Similarly, at the spacecraft level, a series of development tests are performed on a mechanical test model, thermal test model, antenna test model, electrical test model, and, finally, the flight spacecraft. Figure 18 shows test models used on TOS. Table 4 lists the tests that are typically performed at the different assembly levels and different classes of models.

It is this composite test program that covers the satellite hardware from the component to the final flight article and provides the assurance that:

a. The intended system and detail design has been met in the assembled hardware by its ability to meet performance specifications.

b. The equipment is adequately designed with a sufficient margin to meet the electrical and mechanical requirements, and environmental levels during prototype tests are in excess of the expected launch and orbital environments. (Examples of prototype spacecraft testing are shown in Figures 19 and 20.)

c. Sufficient time has been applied to the equipment to weed out early life catastrophic failures, or workmanship defects.

d. The equipment under test is totally compatible with ground station hardware, and the data format is compatible with the communications links and the processing equipment of the users.

e. The electrical and optical signatures are established at various temperatures and operational situations to be useful for evaluation of the spacecraft once placed into orbit.

f. Finally, the satisfactory performance of the spacecraft under test establishes the readiness of the spacecraft for launch.

The TIROS and ESSA satellites successes have not only been the by-product of sound system concepts, engineering design, high quality workmanship, and an extensive test program, but are a result of a team effort. The TIROS team, the people who made this program a very important part of their lives, the Program Management and Project Teams both in the government and at RCA have been the catalysts that kept order, overcame program setbacks, motivated the personnel, pulled the resources together whether they were people, facilities, or material, provided program guidance, kept the level of effort at the required pace to meet schedules and to assure continuous uninterrupted operational service, provided interface between various organizations and agencies, and kept the program costs at a level that could be afforded. The Program management at NASA Headquarters, NASA Goddard Space Flight Center, ESSA National Environmental Satellite Center, and RCA Astro-Electronics Division all worked toward one common goal -- mission success. The results speak for themselves. From April 1, 1960, a total of ten TIROS and eight ESSA satellites were successfully orbited. The composite useful life time in orbit for this group exceeds 30 years in space, averaging 20 months per spacecraft in orbit for this group.

## References


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3. A. Schnapf, "The Performance of the TIROS IX Wheel Satellite", presented at the XVI International Astronautical Congress, International Astronautical Federation, Athens, Greece, September 1965.

4. A. Schnapf, "TIROS I, II, III Design and Performance", Aerospace Engineering, June 1962.

TABLE 1. TIROS/ESSA LONGEVITY — USEFUL LIFE IN ORBIT AS OF FEBRUARY 25, 1969

| ORBITAL LIFE | NO. OF SATELLITES | SPACECRAFT |
|---|---|---|
| 1 YR OR MORE | 12 | TIROS II, VI, VII, VIII, IX, X<br>ESSA 1, 2, 3, 4, 5, 6 |
| 2 YRS OR MORE | 7 | TIROS VII, VIII, IX, X<br>ESSA 1, 2, 3 |
| 3 YRS OR MORE | 4 | TIROS VII, VIII, IX<br>ESSA 2 |
| 4 YRS OR MORE | 1 | TIROS VII |

TABLE 2. TIROS AND ESSA SATELLITES USEFUL LIFE

| SATELLITE | LAUNCH DATE | USEFUL LIFE (DAYS) | PICTURES TRANSMITTED |
|---|---|---|---|
| TIROS I | 4/1/60 | 89 | 22,952 |
| TIROS II | 11/23/60 | 376 | 36,156 |
| TIROS III | 7/12/61 | 230 | 35,033 |
| TIROS IV | 2/8/62 | 161 | 32,593 |
| TIROS V | 6/19/62 | 321 | 58,226 |
| TIROS VI | 9/18/62 | 389 | 68,557 |
| TIROS VII | 6/19/63 | 1,809 | 125,331 |
| TIROS VIII | 12/21/63 | 1,287 | 102,463 |
| TIROS IX | 1/22/65 | 1,238 | 88,892 |
| TIROS X | 7/1/65 | 730 | 78,874 |
| ESSA 1 | 2/3/66 | 861 | 111,144 |
| ESSA 2 (APT) | 2/28/66 | 1,093* | 93,600* |
| ESSA 3 (AVCS) | 10/2/66 | 738 | 92,076 |
| ESSA 4 (APT) | 1/26/67 | 465 | 27,129 |
| ESSA 5 (AVCS) | 4/20/67 | 677* | 69,800* |
| ESSA 6 (APT) | 11/10/67 | 473* | 47,000* |
| ESSA 7 (AVCS) | 8/16/68 | 193* | 26,600* |
| ESSA 8 (APT) | 12/15/68 | 72* | 7,000* |

* Still Operational as of February 25, 1969.

TABLE 3. AUTOMATED SPACECRAFT DEVELOPMENT EXPERIENCE

| Spacecraft | Number of Spacecraft | Total Cost Including Development ($ millions) | Cost Per Spacecraft ($ millions) | Weight, (lb) Dry | Cost Per Pound ($ thousands) | Development Time to First Flight (mo)[a] |
|---|---|---|---|---|---|---|
| **Scientific and geodetic satellites** | | | | | | |
| IMP | 10 | 57 | 6 | 140 | 43 | 26 |
| Radio Astronomy Explorer | 2 | 14 | 7 | 415 | 17 | 35[b] |
| Geos | 2 | 13 | 7 | 385 | 18 | 18 |
| Biosatellite | 6 | 135 | 22 | 950/1250 | 20 | 32 |
| Orbiting Solar Observatory | 8 | 80 | 10 | 600 | 17 | 29 |
| Orbiting Geophysical Observatory | 6 | 220 | 37 | 1300 | 28 | 44 |
| Orbiting Astronomical Observatory | 4 | 280 | 70 | 4200 | 17 | 63 |
| **Meteorological satellites** | | | | | | |
| Tiros Operational Satellite (ESSA) | 9 | 26 | 3 | 300 | 10 | 19 |
| Nimbus | 4 | 200 | 50 | 930/1160 | 48 | 46 |
| **Communications and technology satellites** | | | | | | |
| Syncom | 3 | 21 | 7 | 80 | 87 | 18 |
| Applications Technology Satellite | 5 | 130 | 26 | 800 | 33 | 31 |
| **Lunar probes** | | | | | | |
| Ranger | 9 | 170 | 19 | 675/800 | 26 | 20 |
| Surveyor | 7 | 483 | 69 | 596/624 | 113 | 62 |
| Lunar Orbiter | 5 | 167 | 33 | 656 | 58 | 29 |
| **Planetary and interplanetary probes** | | | | | | |
| Mariner 2 | 3 | 30 | 10 | 450 | 22 | 12 |
| Mariner 4 | 3 | 84 | 28 | 575 | 49 | 24 |
| Pioneer | 5 | 67 | 13 | 145 | 90 | 28 |

[a]From contract go-ahead. [b]First flight scheduled for Nov. 1967.

TABLE 4. TIROS/TOS TEST PROGRAM

I. Component Test Flow

A. Part Specification (Spacecraft Standard)
   1. Non-Standard Part Approval
B. Part Procurement
   1. High-Reliability Tests
   2. 300-Hour Burn-In
   3. Field Inspection
   4. Incoming Part Inspection
   5. Storage in Controlled Area
C. Radiation Exposure Tests
D. Life Testing of Batteries, Bearings, Drives, etc.

II. Board Level Test Flow

A. Board Assembly and Inspection
B. Prepotting Test
C. Box Prepotting Assembly and Test
   1. Electrical
   2. Thermal Test
D. Board Potting and Inspection
E. Board Post-Potting Test

III. Box-Level Test Flow

A. Post-Potting Box Assembly and Inspection
B. Box Bench-Electrical Tests
C. Box Calibration
D. Box Thermal-Tenney Tests
E. Box Vibration - Prototype or Flight - Random and Sine Vibration in Three Axes
F. Box Post-Vibration Electrical Test
G. Box Thermal-Vacuum Tests - Prototype or Flight
H. Box Post-Thermal Vacuum - Electrical and Calibration Measurements
I. Box Acceleration Tests - Prototype Only
J. Box to Controlled Stock - Ready for Spacecraft Inspection

TABLE 4. TIROS/TOS TEST PROGRAM (Continued)

IV. Spacecraft Level Tests

A. Development Tests
1. Mechanical Test Model
a. Mock-up Layout of All Equipment
b. Integration of Prototype Units and Mechanical Dummies for Prototype Qualification Tests
1) Spin Balance
2) Moment of Inertia and Center of Gravity Measurement
3) Instrumentation - Accelerometer and Strain Gages
4) Vibration in Three Axes - Sine and Random - 1-1/2 times flight levels, 4 minutes in each plane
c. Fit-Check with Launch Vehicle Attach Fitting
1) Mating Operations
2) Alignment
3) Shock and Separation Operations
d. Despin Test
e. Dynamic - Simulation Tests
f. Antenna Deployment Tests
g. Solar Panel - Deployment Tests
h. Spacecraft Handling Procedures and Checkout of Special Test Equipment and Test Facilities

B. Thermal Test Models
1. Thermal Tests to Verify Analytical Results
2. Thermal Tests to Simulate Injection and Operational Modes
3. Solar Tests to Verify Passive Thermal Fence Design
4. Subsystem Tests on Actuators to Verify Life of Actuation System

C. Antenna Test Model
1. Passive RF Elements - Check RF Isolation
2. Antenna Pattern Measurements
3. RF Interference Tests
4. Command Sensitivity

D. Solar Array Tests
1. Q-Board Tests - Check of Solar Array Module under Extreme Thermal Cycles
2. Total Solar Panel - Mechanical Vibration Tests
3. Solar Panel Thermal/Vacuum - Cycling Tests

TABLE 4. TIROS/TOS TEST PROGRAM (Continued)

IV. Spacecraft Level Tests (Continued)

4. Solar Panel Solar Simulation Tests
5. Radiation Degradation Tests of Sample Cells

E. Dynamic Tests - with ETM Spacecraft

1. Moment of Inertia
2. Balance and Center of Gravity
3. Attitude Control System - Dynamic Suspension Test
4. Yo-Yo Despin Tests
5. Coning - Despin Tests
6. Pendulum Tests for Liquid Dampers

F. Electrical Test Model

1. Spacecraft Subsystem - Electrical Integration
2. Debugging of Subsystems and Systems
3. Power-Systems Measurement
4. Command Systems Measurement
5. Communication Systems Measurement
6. Control Systems Measurement
7. Telemetry Systems Measurement
8. Primary Sensors Systems Measurement
9. Secondary Sensors Systems Measurement
10. Spacecraft RFI Tests
11. Spacecraft Command Sensitivity Tests
12. Alignment Measurements
13. Calibration
14. Spacecraft and Ground Support Equipment Compatibility
15. Spacecraft System Electrical Performance Evaluation Tests
16. Spacecraft RF Tests

V. Flight Spacecraft - Flight Acceptance Tests

A. Spacecraft Electrical and Mechanical Integration
B. Spacecraft Debugging
C. Parameter Measurements, All Subsystems

1. Power
2. Command
3. Communications
4. Control
5. Telemetry
6. Primary Sensors
7. Secondary Sensors
8. RFI
9. Command Sensitivity
10. Alignment
11. Calibration

D. Spacecraft Balance
E. Moment of Inertia and Center of Gravity

TABLE 4. TIROS/TOS TEST PROGRAM (Continued)

V. Flight Spacecraft - Flight Acceptance Tests (Continued)

F. Standard Electrical Performance Evaluation Test
G. Flight Vibration - 3 Axes Random and Sine
H. Abbreviated SEPET*
I. Thermal-Vacuum Tests
J. SEPET
K. Final Alignment
L. Magnetic Measurement
M. RFI
N. Command Sensitivity
O. Camera Calibration
P. SEPET
Q. Pack and Ship to Launch Site

VI. Launch Site Operation

A. Unpack and Inspect
B. Mechanical Compatibility Checks with Rocket Mating Hardware
C. Spacecraft SEPET
D. Spacecraft Mating to Upper Stage
   1. If 3rd stage solid rocket motor - spin balance at special spin facility
   2. Mate Spacecraft to Rocket at Launch Stand
E. Spacecraft Final Calibration Check on Stand
F. Daily - Spacecraft System Checkout
G. T-1 Day - Special Pre- and Post-Fairing Inspections and Tests
H. T-0-Spacecraft Tests

VII. Post-Launch

A. 1st Orbit - Initiation of Orientation Maneuver
B. Orientation on Next 13 to 18 Orbits
C. Orbits 19 to 100 - Start Subsystem-by-Subsystem Checkout
D. After spacecraft is fully checked out, it is turned over to ESSA for operational control

*Standard Electrical Performance Evaluation Test

TABLE 4. TIROS/TOS TEST PROGRAM (Continued)

VIII. Spacecraft Storage

In the event that launching of the flight qualified spacecraft is not required:

A. Store spacecraft in special container with dry nitrogen atmosphere
B. Remove spacecraft every two weeks for Go/No-Go check
C. Every six weeks perform the full SEPET
D. Repeat above sequence until spacecraft is called up for launch
E. If spacecraft is called up, additional environmental tests and calibration will be repeated depending on length of time in storage. Age sensitive equipment, such as batteries, may be replaced.

LIST OF ILLUSTRATIONS


Figure 1. TIROS, TOS, ESSA Performance Highlights

Figure 2. TOS Weather Photographs

Figure 3. TIROS I Configuration (Upper View) and TIROS IX Wheel (Lower View)

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Figure 5. Evolution of TIROS, TOS, ESSA Satellites

Figure 6. TIROS Dynamic Control Components

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Figure 8. TOS System

Figure 9. Initial Orientation Maneuver by Means of QOMAC and MBC Coils

Figure 10. First Complete View of the World's Weather from TIROS IX Satellite, February 13, 1965

Figure 11. Two Methods of Spin Stabilization -- Wheel Mode vs. Stabilite Mode

Figure 12. TIROS M Spacecraft

Figure 13. MASC Coil and Infrared Trigger

Figure 14. Attitude Horizon Crossing Sensor

Figure 15. TOS APT Spacecraft, Block Diagram

Figure 16. Buildup of TIROS M Spacecraft Structure

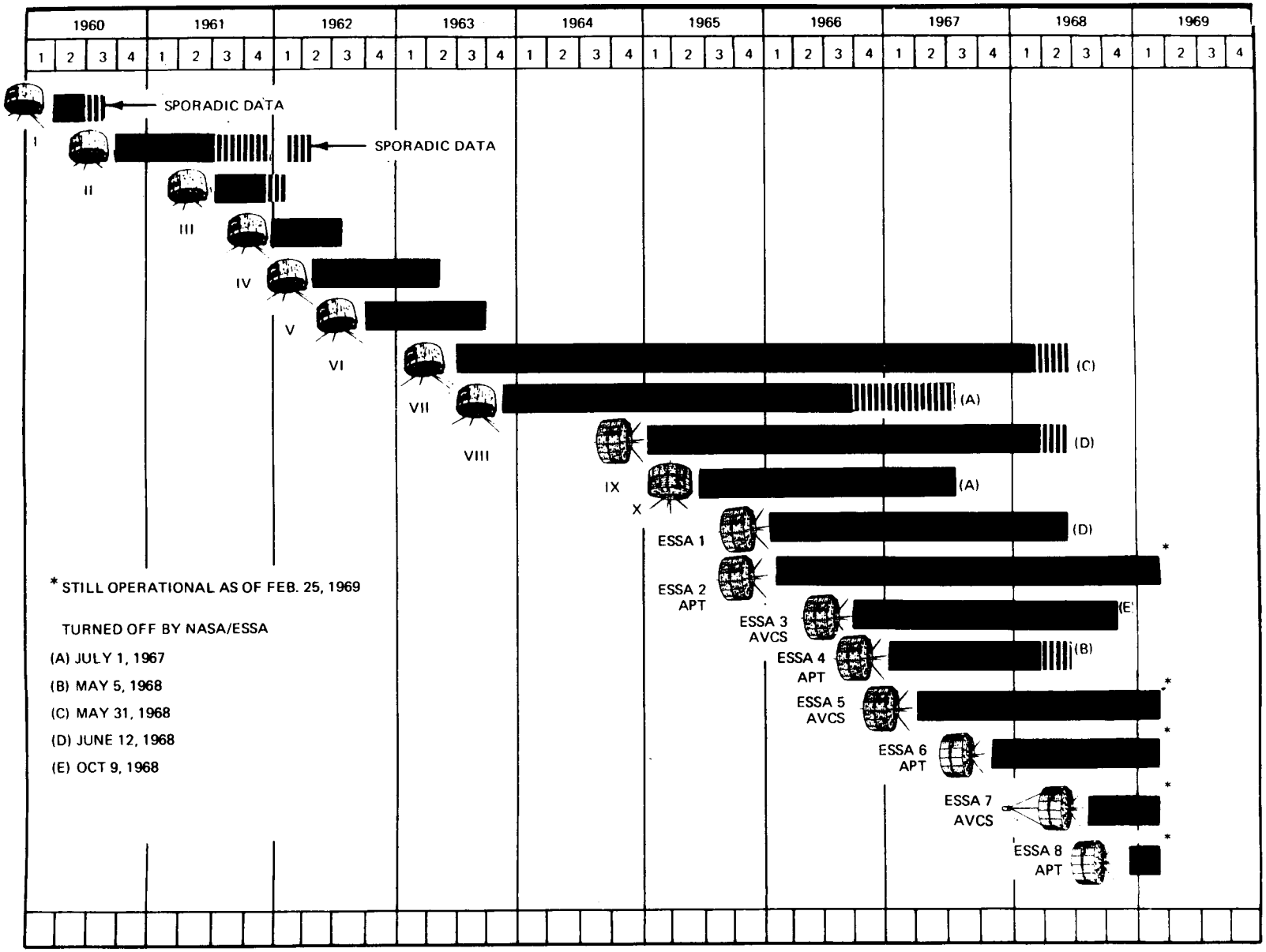
Figure 17. TIROS M Spacecraft, Showing (a) Panels Open for Installation of Components and (b) Completely Assembled Spacecraft

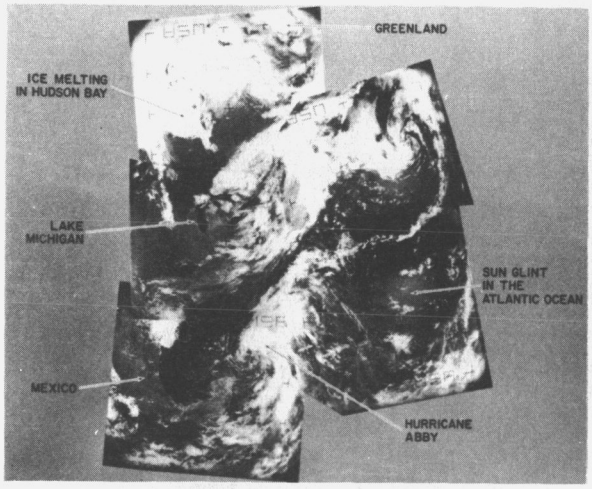
Figure 18. TOS Spacecraft Test Models

Figure 19. TIROS M Mechanical Test Model in Vibration Test

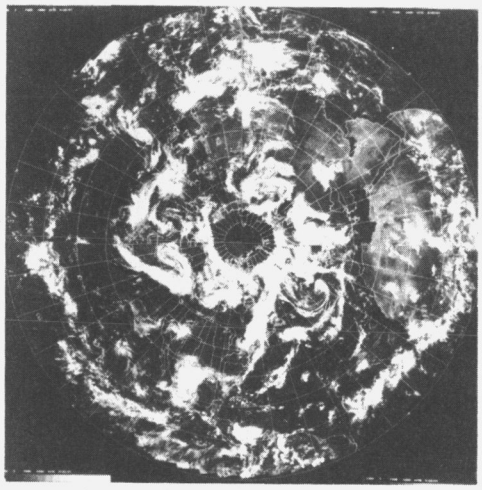
Figure 20. TIROS M Dynamic Simulation

FIGURE 1. TIROS, TOS, ESSA Performance Highlights

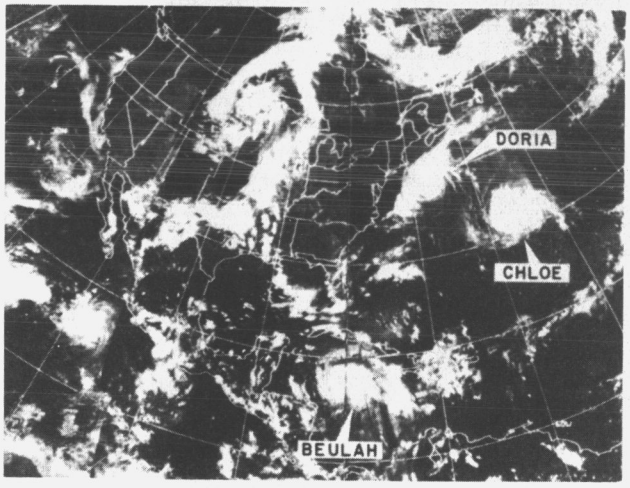




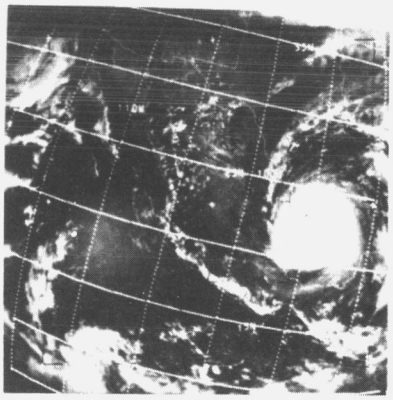

**ESSA 6 APT PHOTOGRAPH OF HURRICANE ABBY 6/3/68**
**FIRST HURRICANE OF 1968 SEASON SOUTHWEST OF FLORIDA**



**ESSA 3**
**GLOBAL WEATHER PHOTOMOSAIC NORTHERN HEMISPHERE, SEPTEMBER 22, 1967**




**ESSA 5**
**1967 HURRICANE WATCH PHOTOMOSAIC SEPTEMBER 1967**



**ESSA 3**
**AVCS PHOTOGRAPH OF HURRICANE INEZ NEAR MEXICO, OCTOBER 8, 1966**

FIGURE 2. TOS Weather Photographs

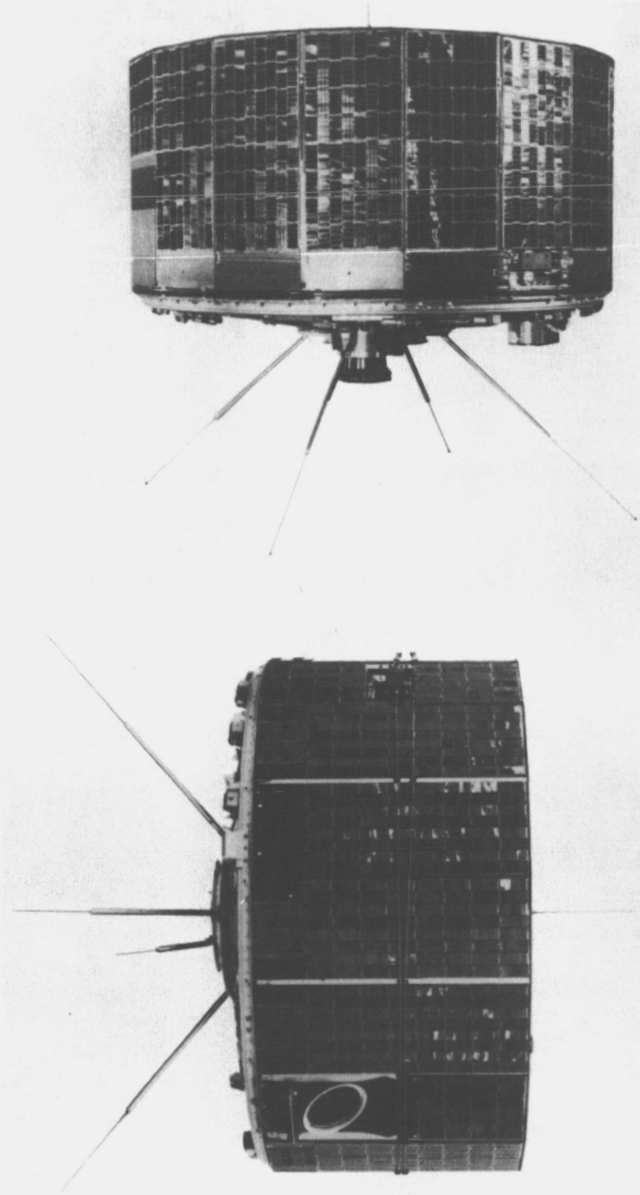
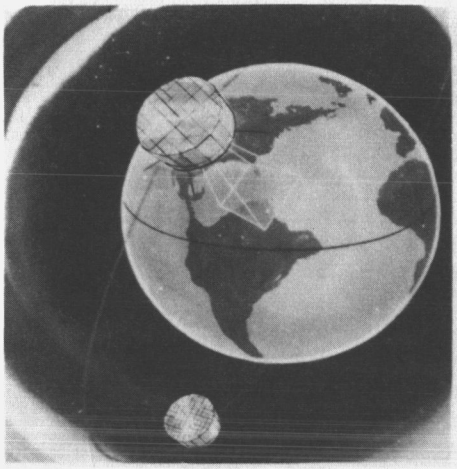
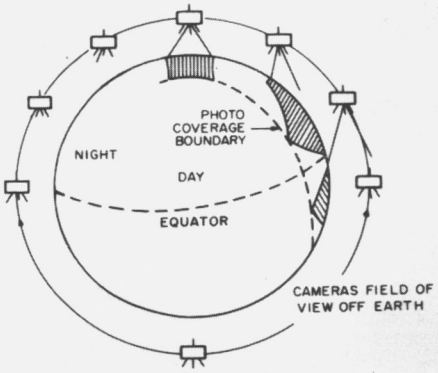
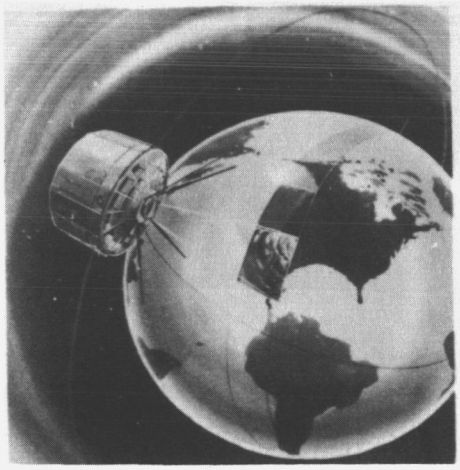


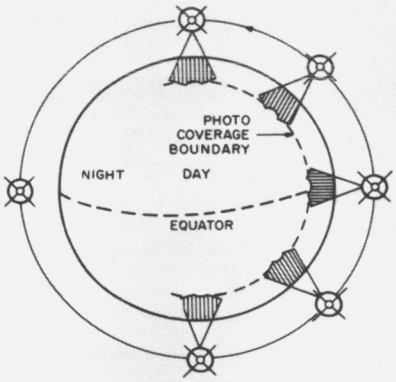
FIGURE 3. TIROS I Configuration (Upper View) and TIROS IX Wheel (Lower View)






Standard-TIROS Orientation

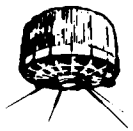





**Wheel-TIROS Orientation**

FIGURE 4. Comparison of Photo Coverage, Standard Mode and Wheel-Mode TIROS Satellites

1960-63



TIROS R&D MISSION

TIROS I
- Television Feasibility
- Infrared Experiments

TIROS II
- Magnetic Torquing

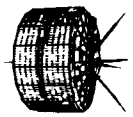
TIROS III
- Limited Operational Use
- University of Wisconsin Radiometer
- NASA Radiometer

TIROS IV through VI
- Limited Operational Use

TIROS VII
- Limited Operational Use
- Ion Temperature Probe

TIROS VIII
- First Use of APT Real-Time Television Readout

1965-66

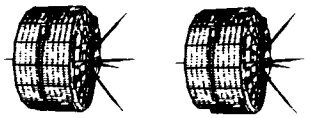


TIROS WHEEL

TIROS IX
- First Wheel Mode Spacecraft
- Near-Polar Sun-Synchronous Orbit For Daily Global Observation
- Forerunner of the TIROS Operational System

ESSA 1
- Operational Satellite

1966-69



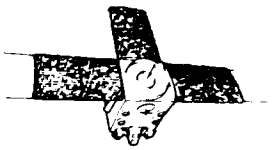
APT AVCS

TIROS OPERATIONAL SYSTEM

ESSA 2, 4, 6 and 8
- APT Spacecraft For Real-Time Global Readout

ESSA 3, 5, and 7
- AVCS Spacecraft For
  - Daytime Global TV Observation
  - Remote Sensing and Storage for U.S. Readout

1969-72

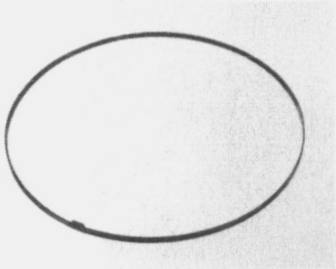


THE IMPROVED TIROS OPERATIONAL SYSTEM

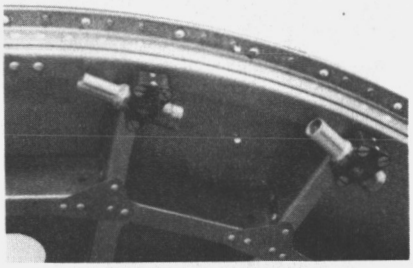
TIROS M AND 5 ITOS SPACECRAFT

- Daytime Global APT Readout
- Daytime Global AVCS Readout
- Day and Night Real-Time and Remote IR and Visual Channel Scanning Radiometer
- Solar Proton Monitor
- Flat Plate Radiometer
- Earth-Stabilized Platform With Growth Capability For Future Sensors

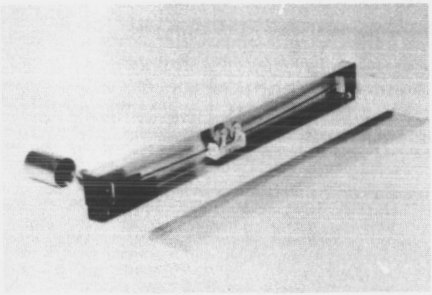
FIGURE 5. Evolution of TIROS, TOS, ESSA Satellites



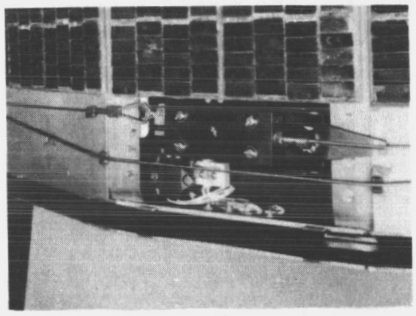
QOMAC and MBC Coils



Spin-Up Rockets



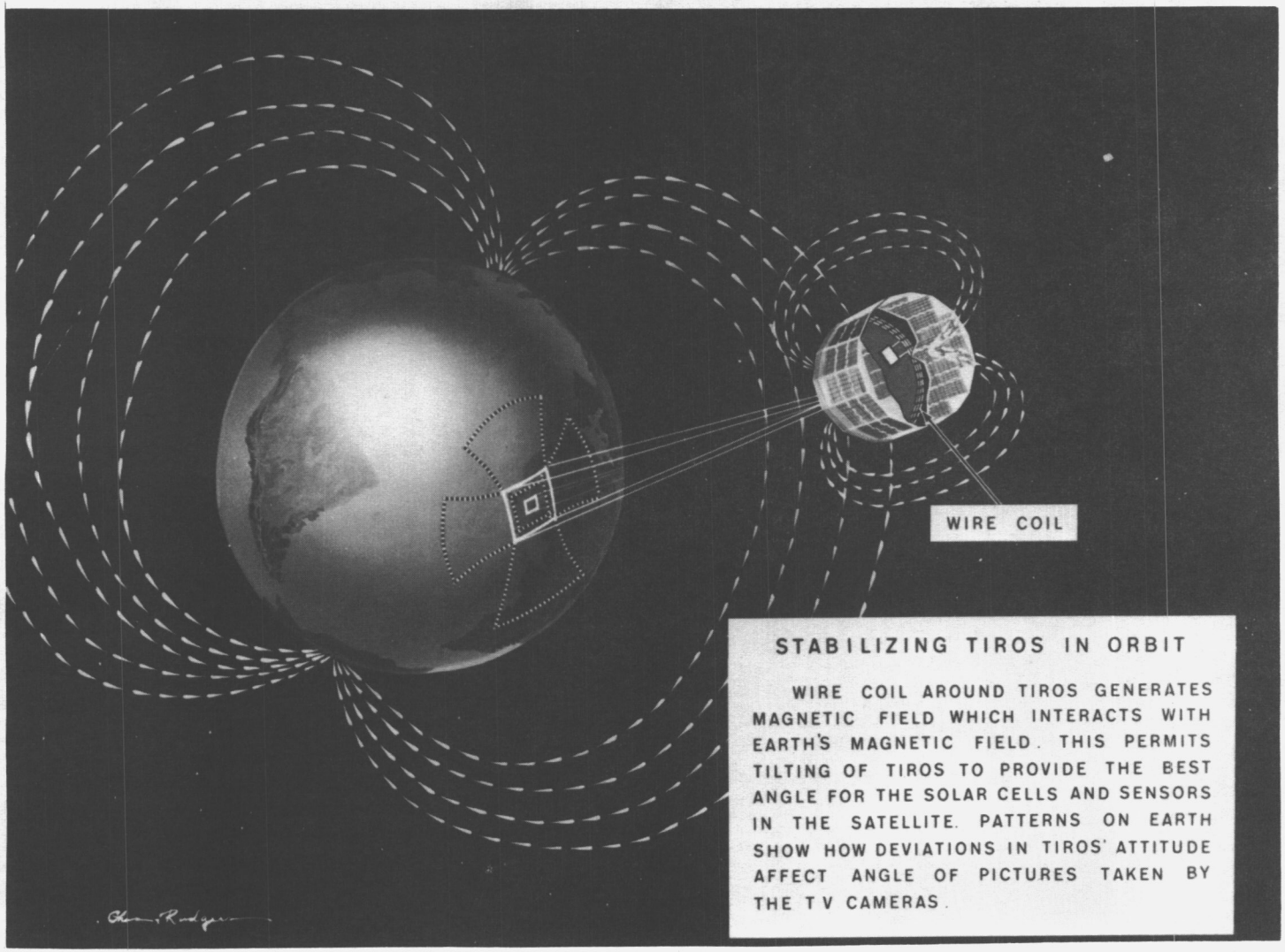
TEAM Damper



De-Spin Mechanism

FIGURE 6. TIROS Dynamic Control Components

FIGURE 7. Dynamic Torquing



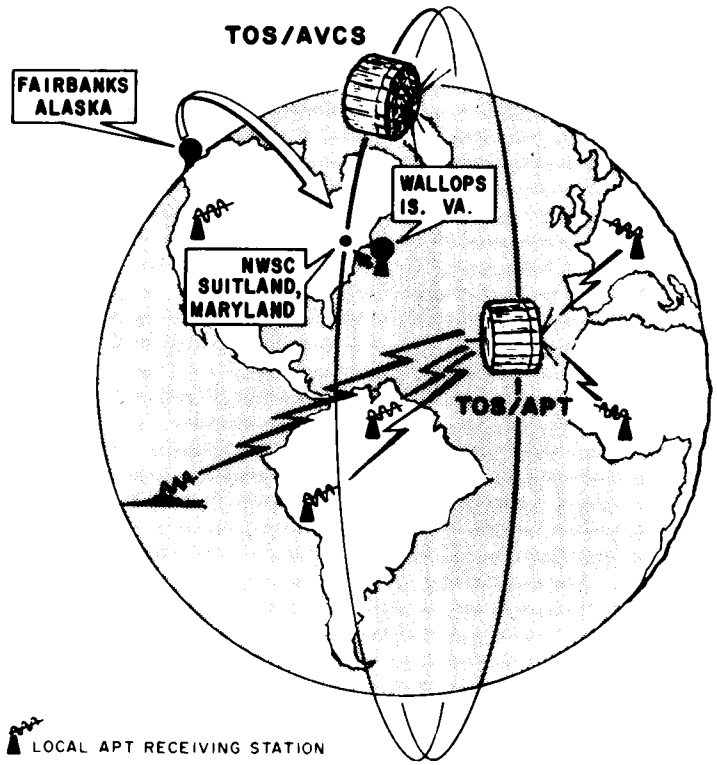
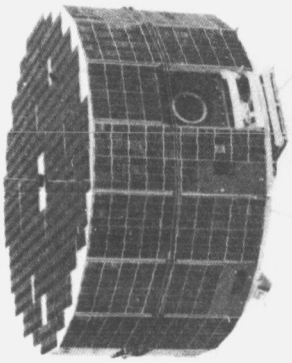
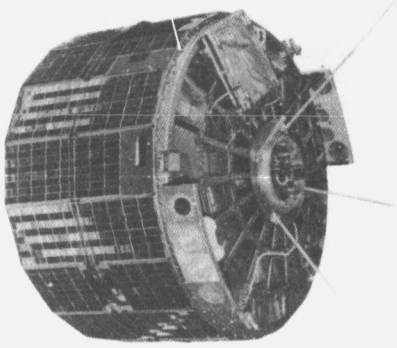


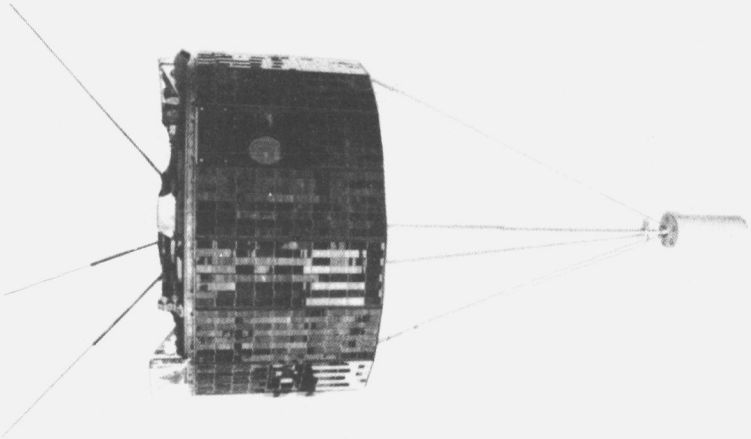

FIGURE 8. TOS System (Sheet 1)



TOS APT Satellite

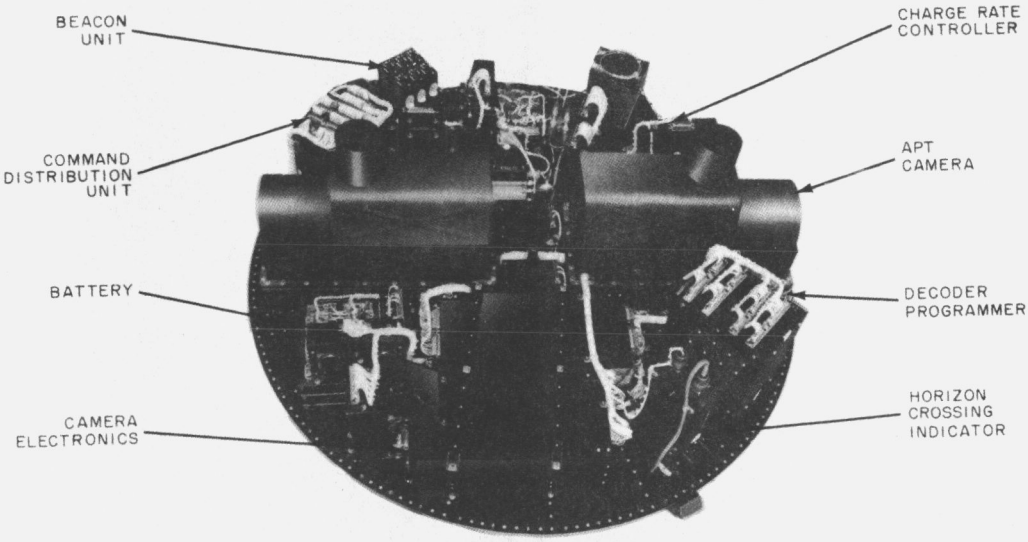


TOS AVCS Satellite

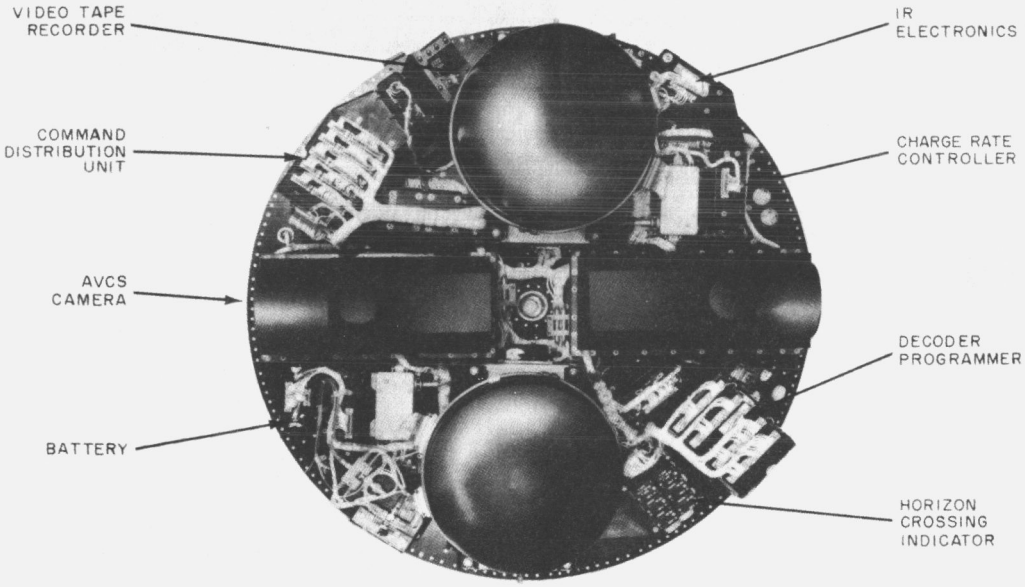


TOS AVCS Satellite with S-Band

FIGURE 8. TOS System (Sheet 2)



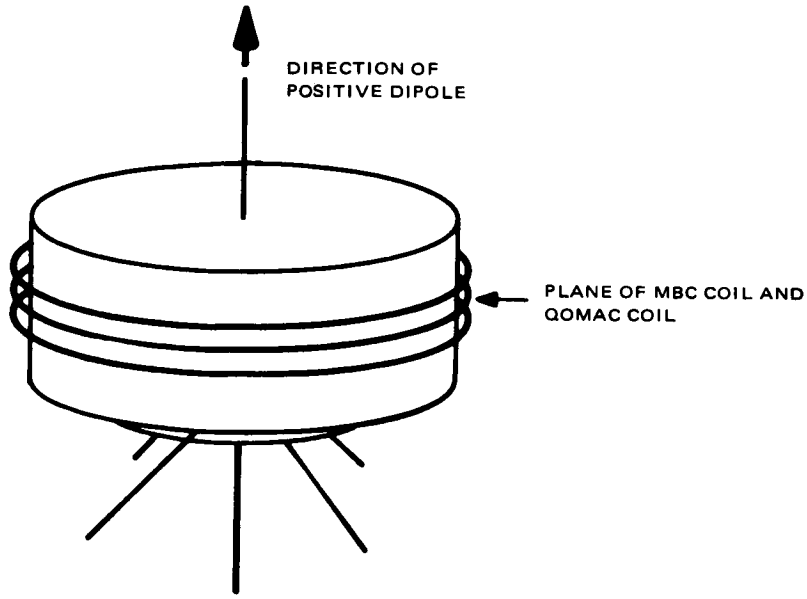

TOS APT SPACECRAFT



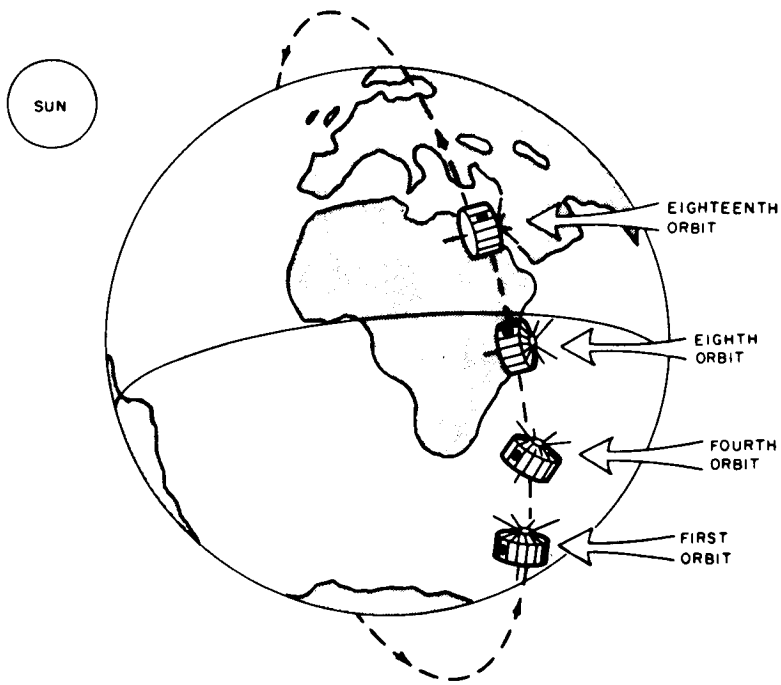

TOS AVCS SPACECRAFT

TOS APT and TOS AVCS Equipment Mounted on Satellite Baseplates

FIGURE 8. TOS System (Sheet 3)




a. Magnetic Dipole Moment Generated by QOMAC and MBC Coils




b. Initial Orientation Maneuver for TOS Spacecraft

FIGURE 9. Initial Orientation Maneuver By Means of QOMAC and MBC Coils

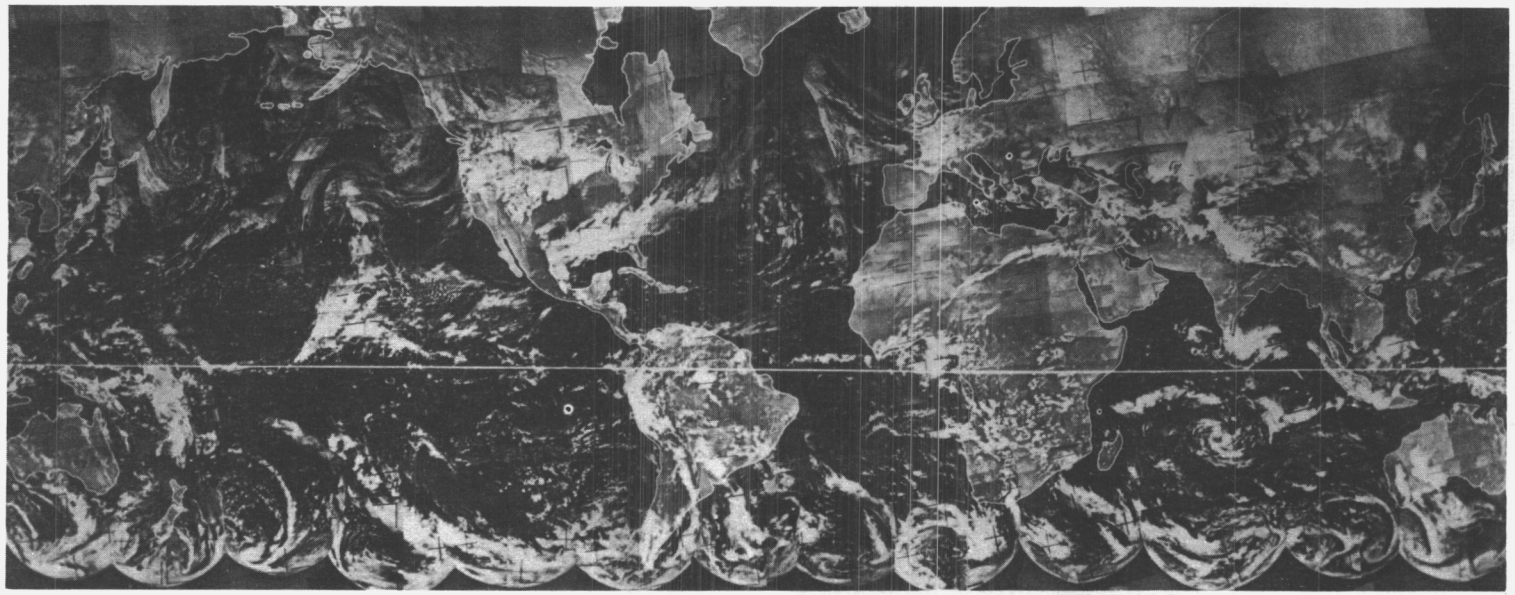
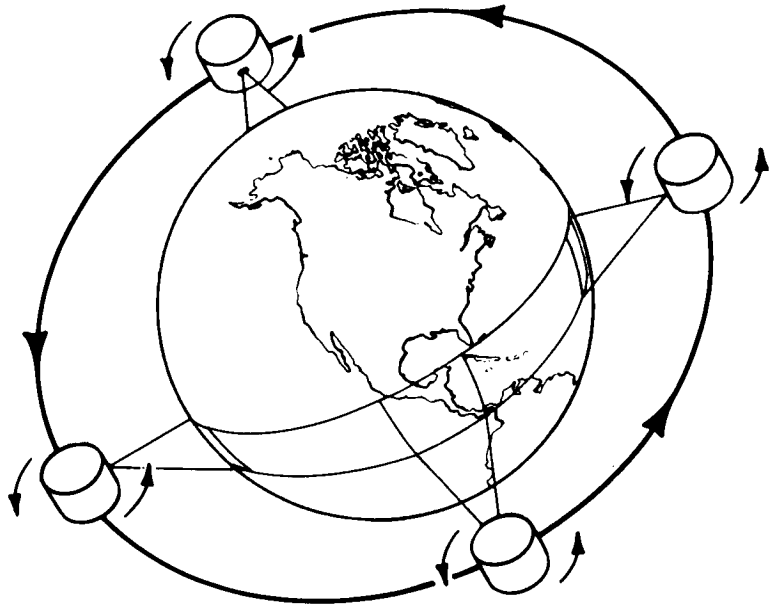
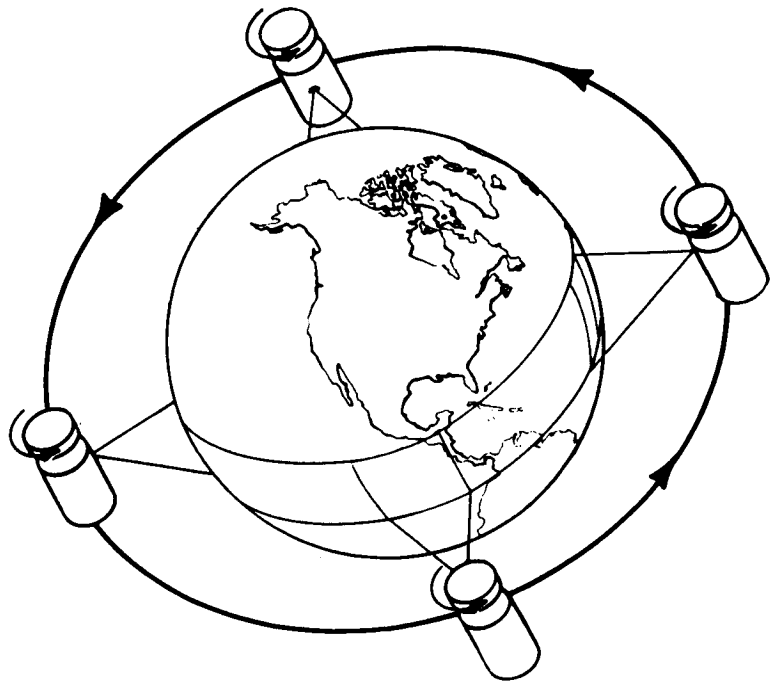


FIGURE 10. First Complete View of the World's Weather From TIROS IX Satellite, February 13, 1965

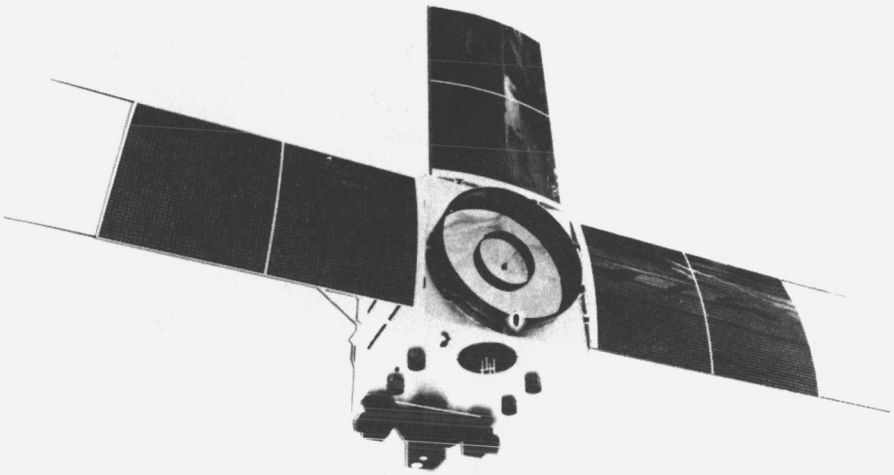


a. Wheel Mode - TIROS Spacecraft at Approximately 10 RPM



b. Stabilite - Flywheel at Approximately 150 RPM; Main Body (Earth Oriented) Spinning at 1 Revolution per Orbit

FIGURE 11. Two Methods of Spin Stabilization Wheel Mode vs Stabilite Mode



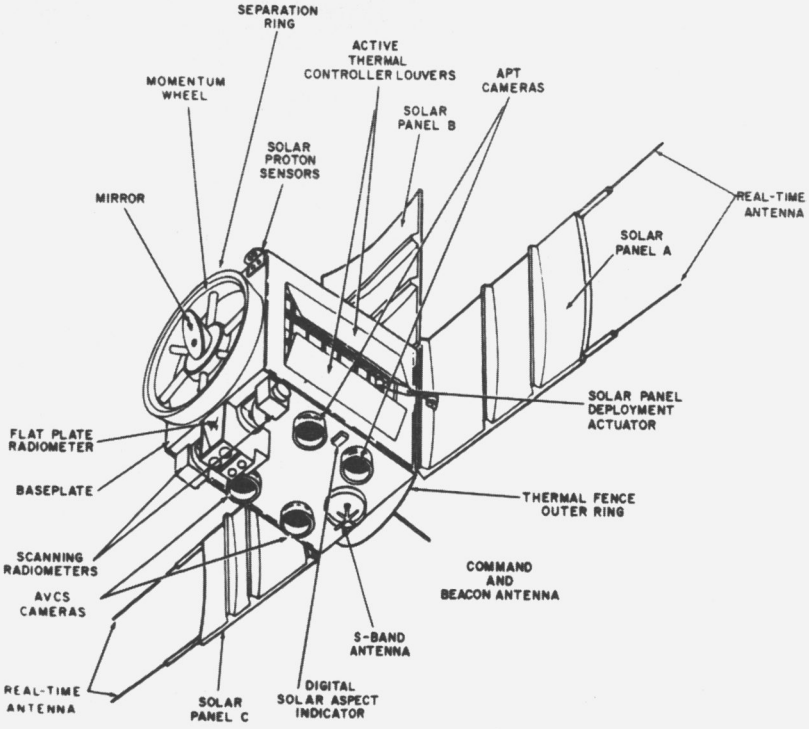
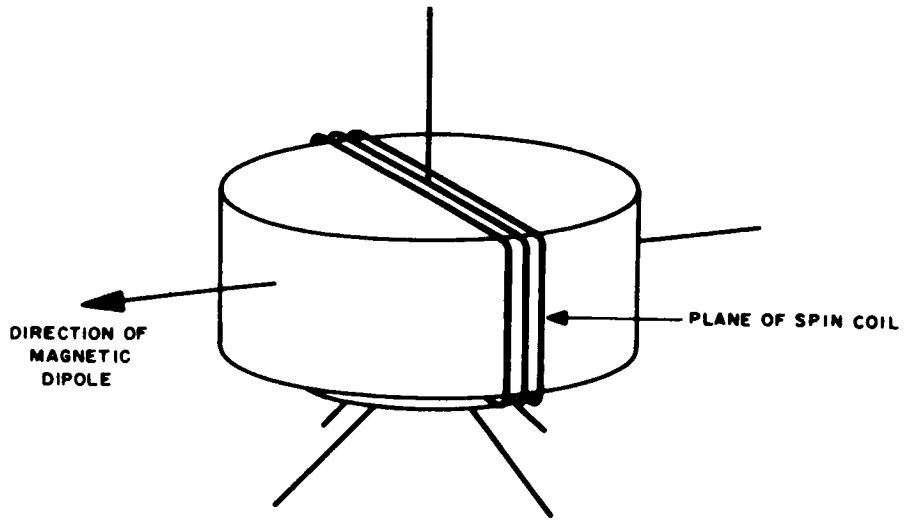
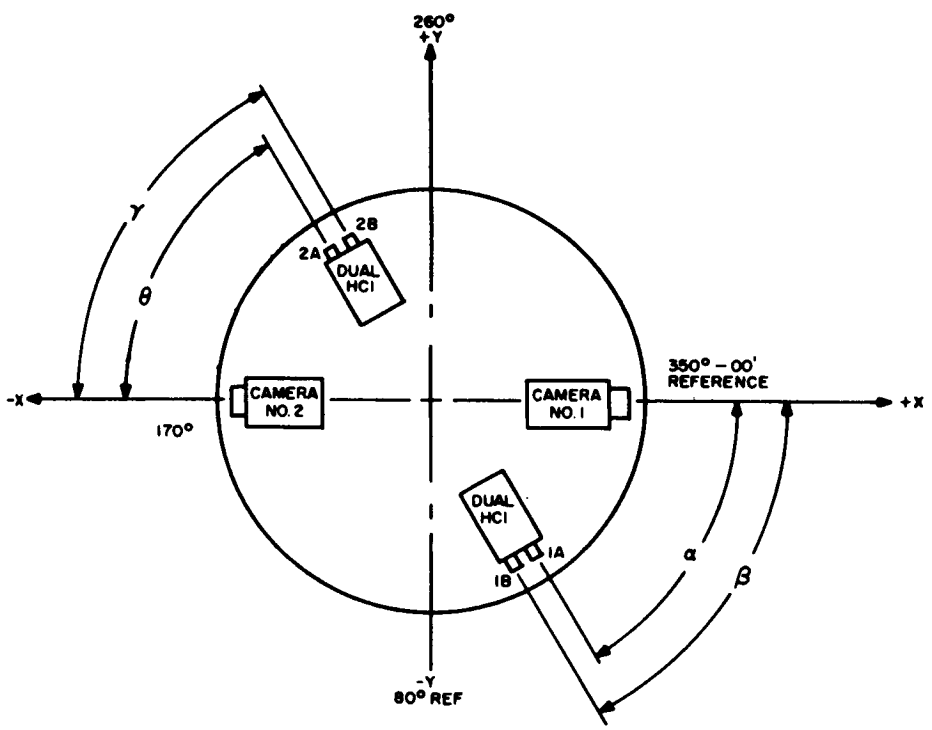



FIGURE 12. TIROS M Spacecraft




a. Magnetic Dipole Moment Generated by MASC Coil




b. Locations of Dual Horizon Crossing Indicators with Respect to Locations of TV Cameras.

FIGURE 13. MASC Coil and Infrared Trigger

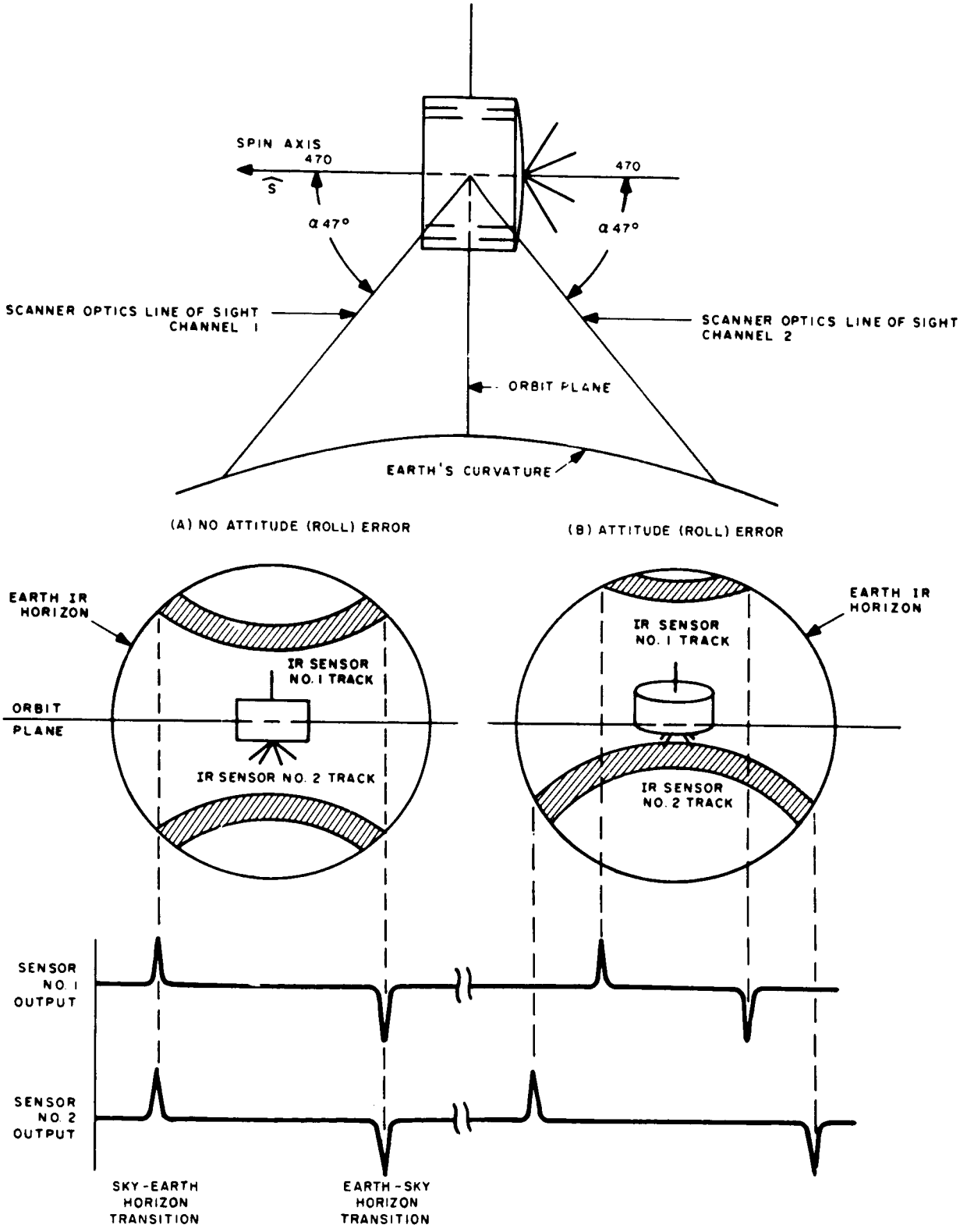
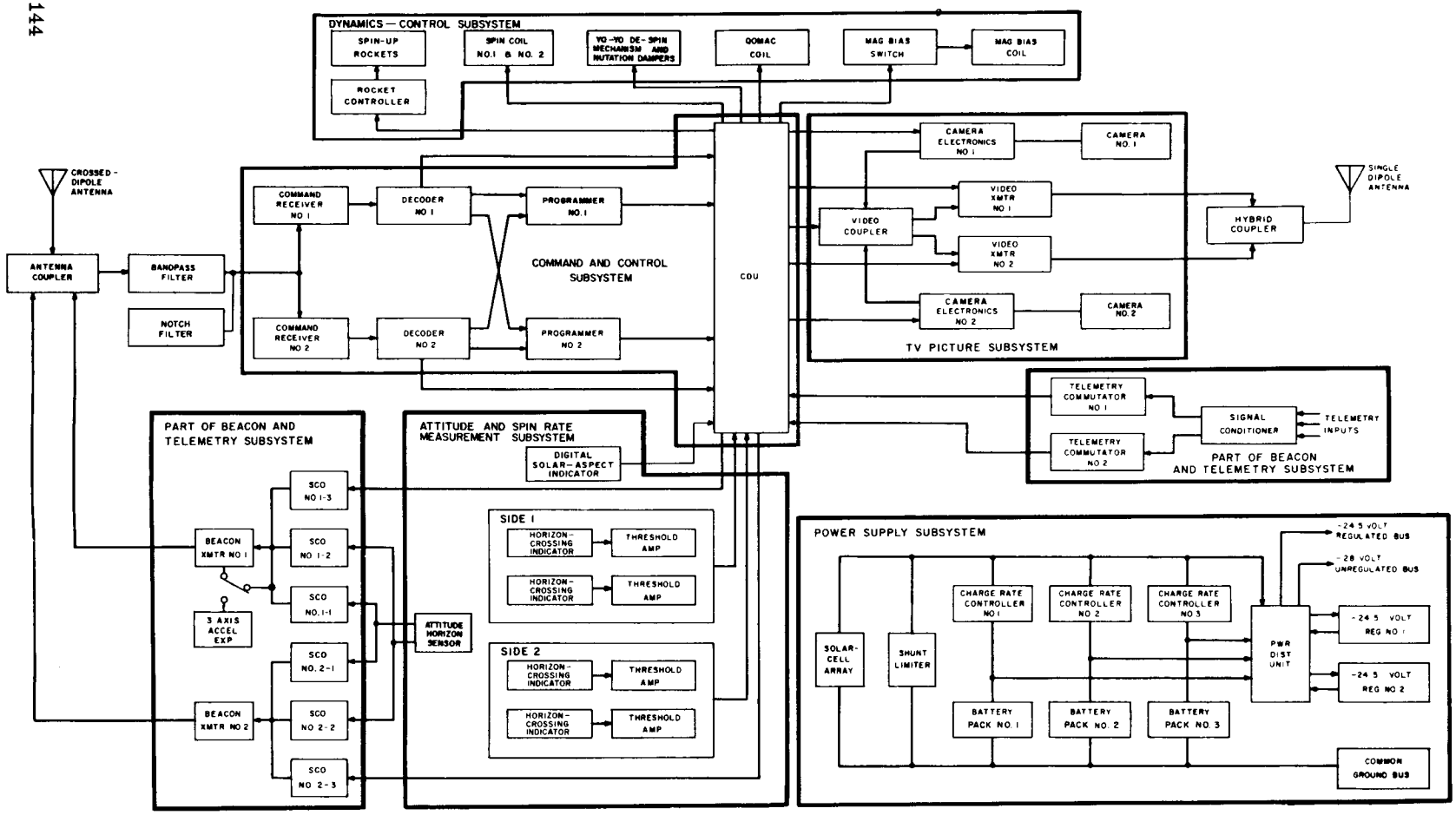



FIGURE 14. Attitude Horizon Crossing Sensor




NOTE: ELEMENTS NOT ENCLOSED BY A HEAVY LINE ARE PART OF THE ANTENNA SUBSYSTEM

FIGURE 15. TOS APT Spacecraft, Block Diagram

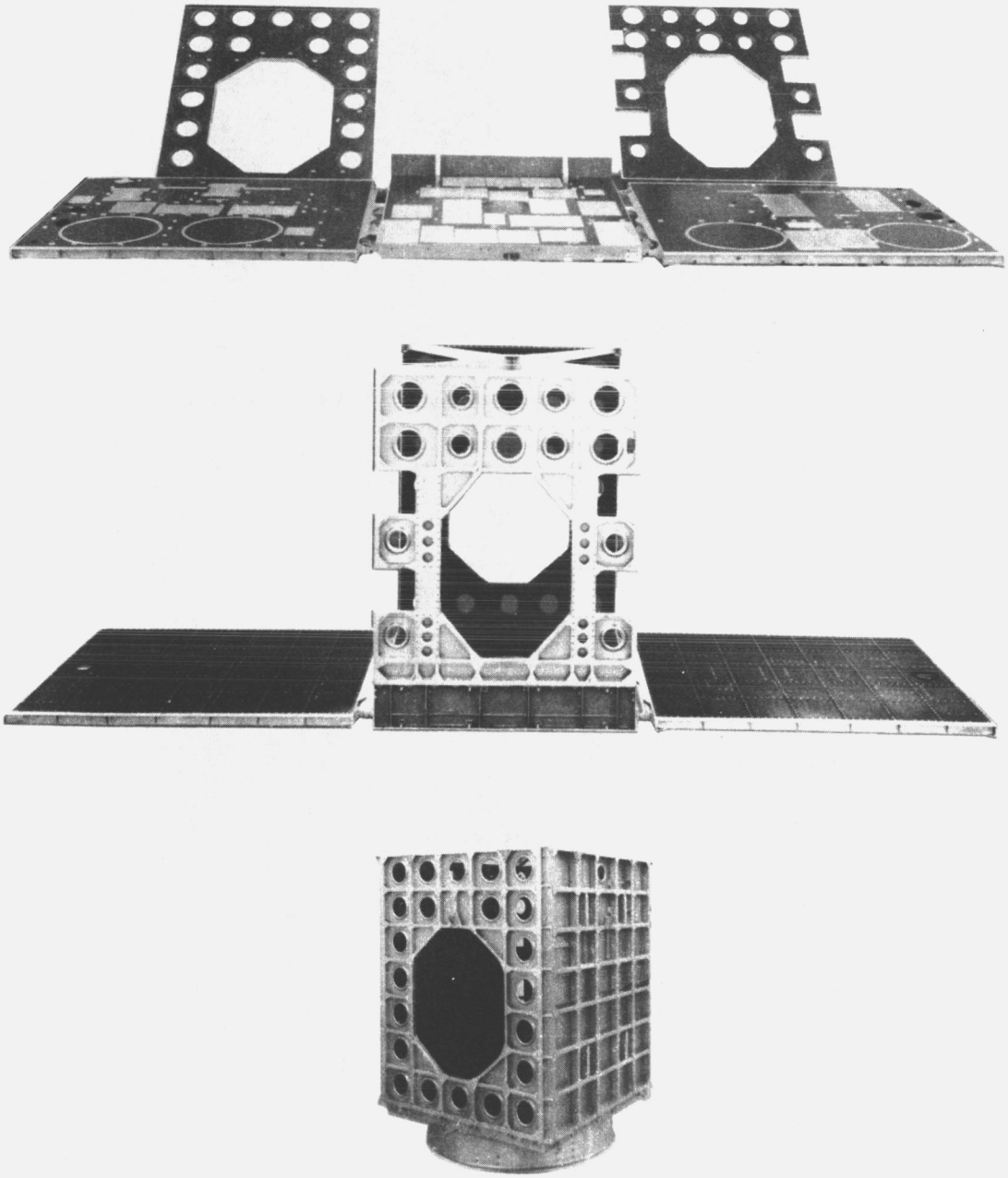


FIGURE 16. Buildup of TIROS M Spacecraft Structure

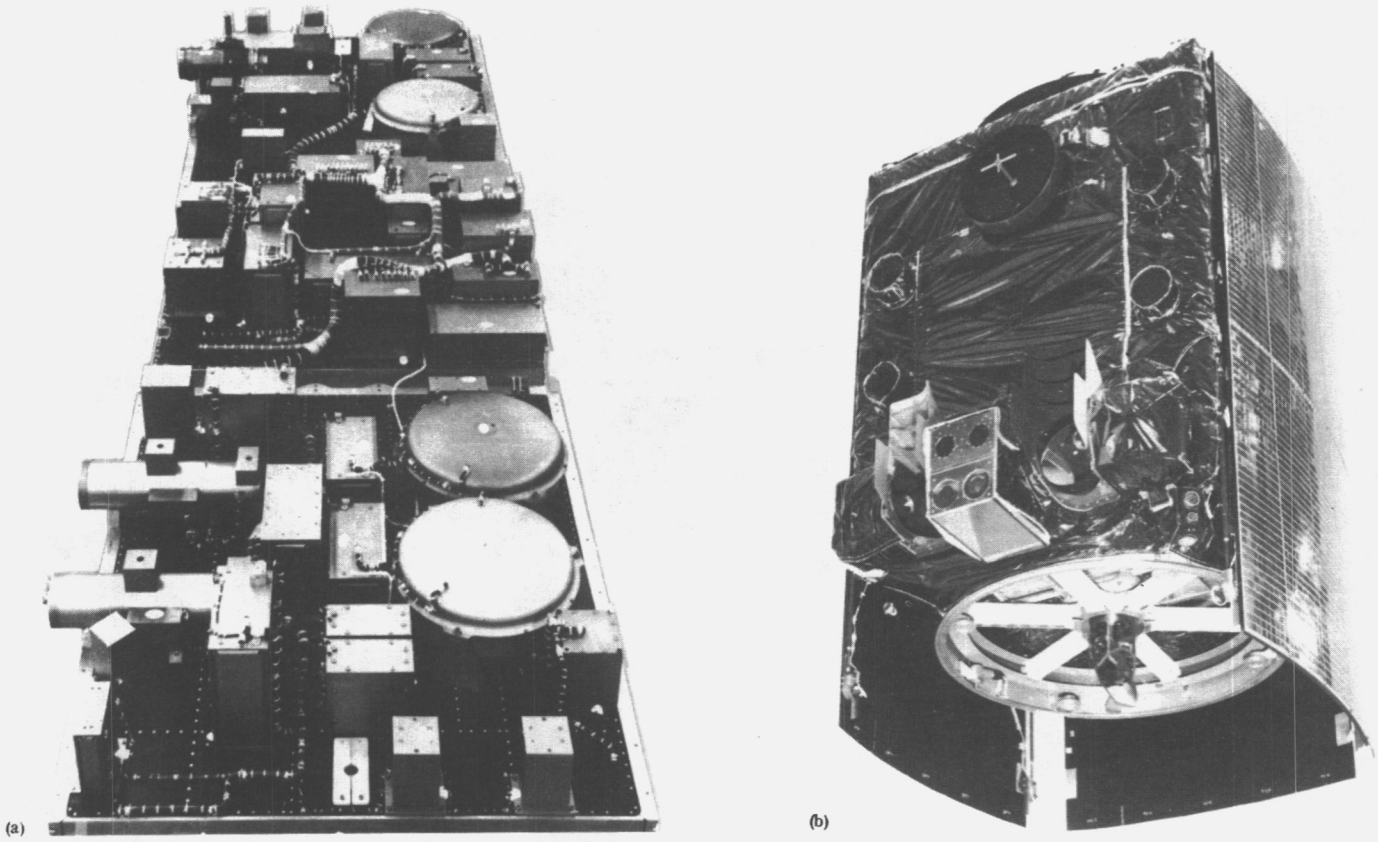



FIGURE 17. TIROS M Spacecraft, Showing (a) Panels Open For Installation of Components and (b) Completely Assembled Spacecraft

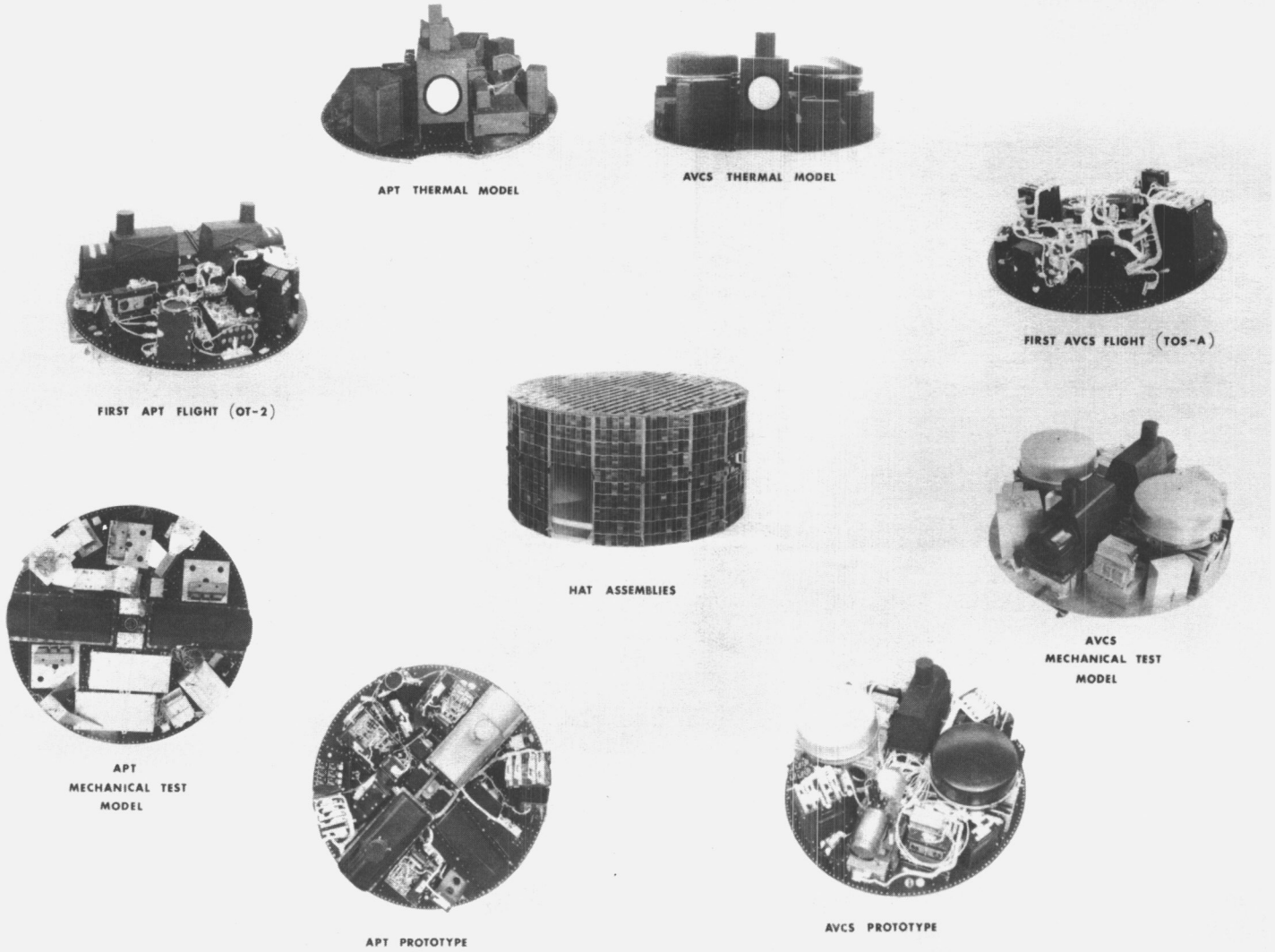



FIGURE 18. TOS Spacecraft Test Models

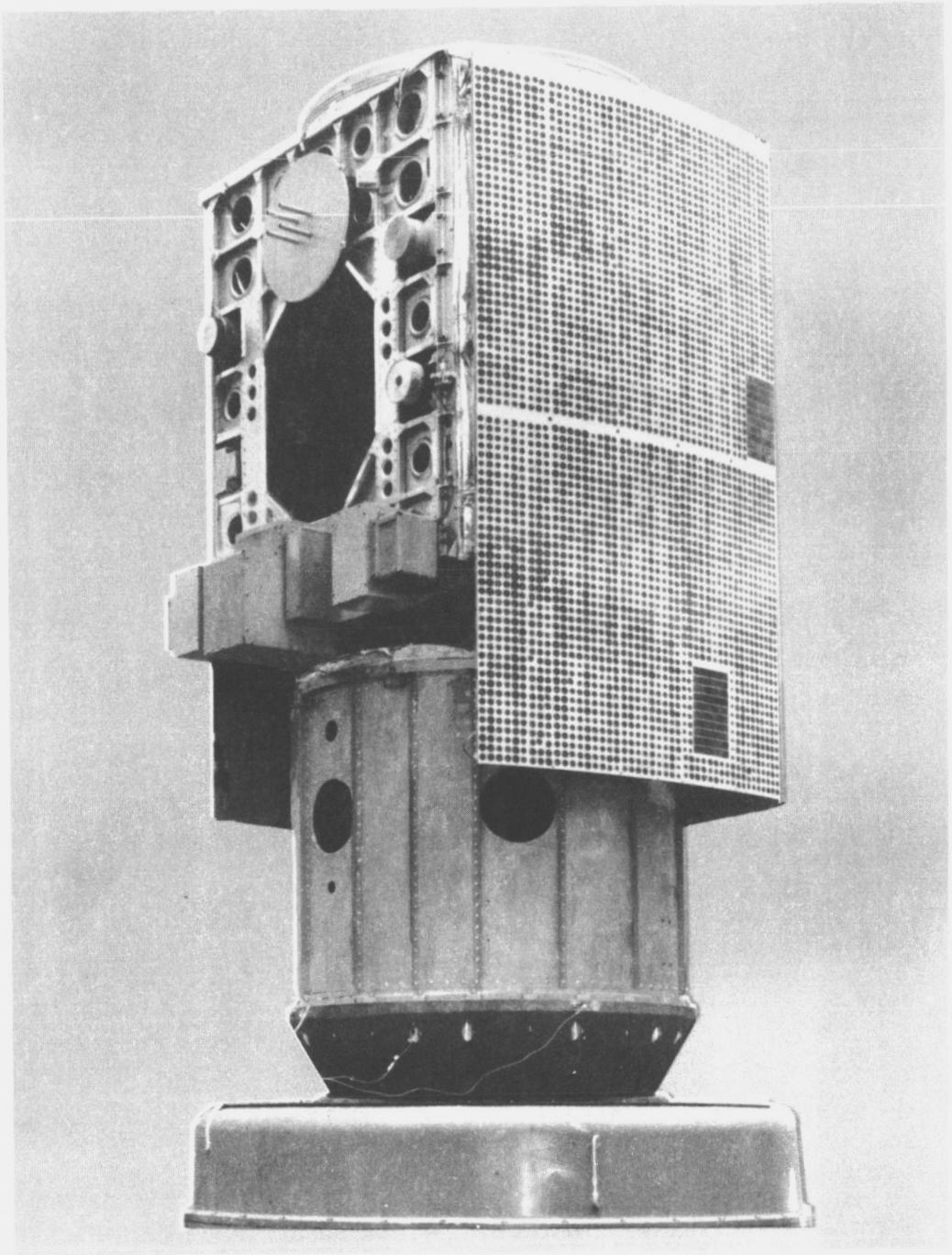


FIGURE 19. TIROS M Mechanical Test Model In Vibration Test

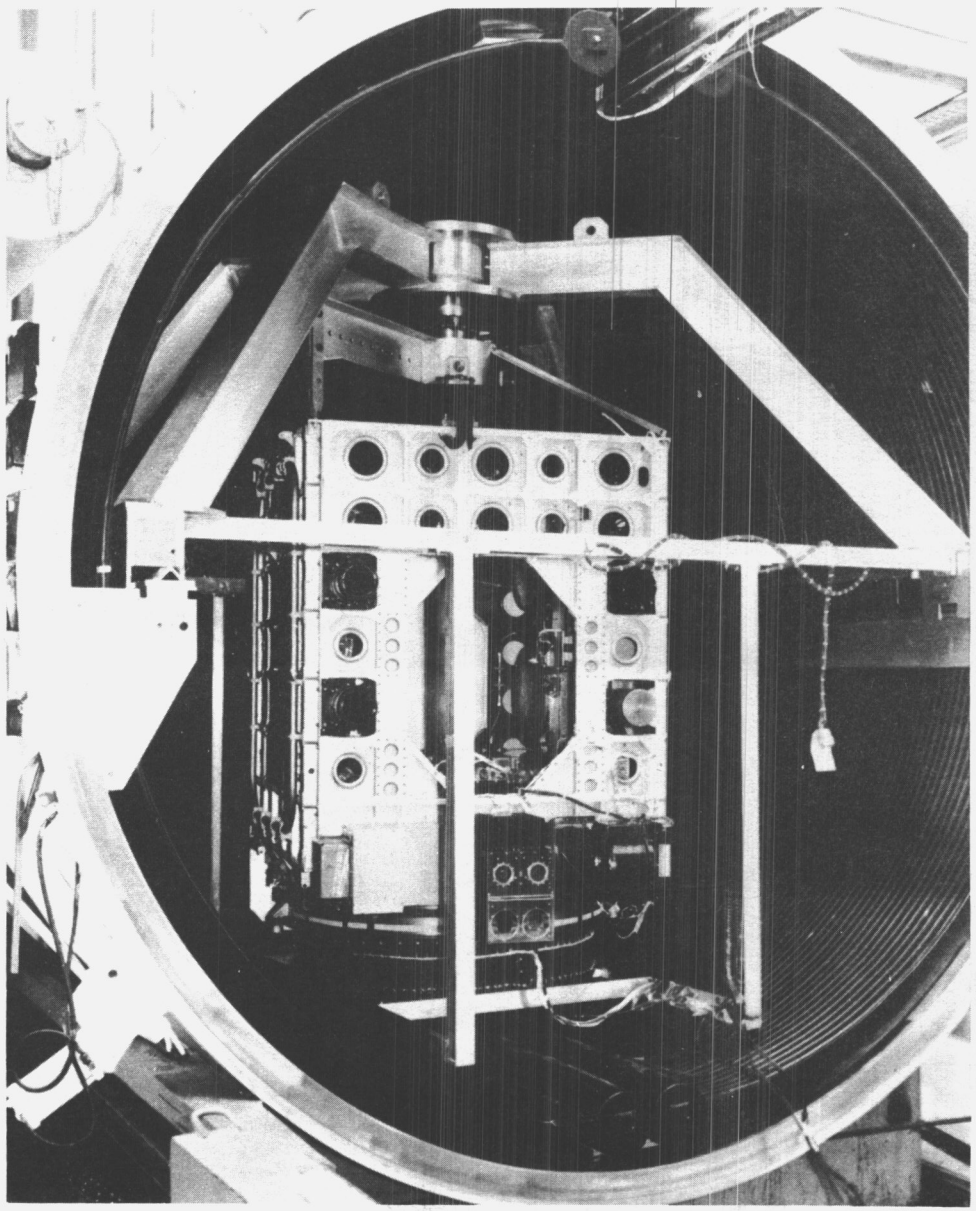


FIGURE 20. TIROS M Dynamic Simulation

# LUNAR ORBITER MANAGEMENT - CASE HISTORY

Thomas R. Costello
Aerospace Group
The Boeing Company
Washington, D. C.

## Introduction

All five Lunar Orbiter missions were successful. The program achieved results beyond the primary objective which was to photograph in detail potential Apollo manned landing areas to aid in the selection and certification of specific landing sites. Surface data from the Surveyor Program were used also in the site selection and certification process.

The first three Orbiter missions essentially satisfied the photographic requirements of the Apollo Program. It was therefore possible to reorient missions four and five to emphasize the acquisition of photography desired by the scientific community. As a result, the entire earth-side hemisphere of the moon has been photographed at a resolution at least ten times better than was previously available through the best earth based telescopes; and 99.5 percent of the far side hemisphere has been photographed in greater detail than earth based observations have been able to provide of the near side. Figure 1 is an oblique view into the crater Copernicus. The central peaks and the far rim are clearly visible. Fauth is the keyhole shaped crater in the foreground. Views such as this are aiding in the scientific study of the moon.

Lunar Orbiters obtained valuable non-photographic data as well. Each spacecraft flew with 20 pressurized cell detectors to measure the micrometeoroid flux in cislunar space and in the vicinity of the moon. They all carried counters to record radiation dosage encountered during the missions. Precise tracking of the spacecraft in their various orbits provided a great improvement in the definition of the moon's gravitational field. A very accurate gravitational model is now being used for the tracking, guidance, and navigation of Apollo spacecraft.

Figure 1

Oblique View of the Crater Copernicus

It is not the intent of this paper to report on the vast amount of data returned by Lunar Orbiter missions, but rather to review the various phases of the program with emphasis on the pertinent management aspects which heavily influenced the manner in which the program was executed. This review covers the program definition, organization, design, test, and mission operation phases.

## Program Definition

The requirement for Lunar Orbiter was based almost entirely on providing wide area, high resolution topographic data to support Apollo lunar landing site selection. The feasibility of such a system was established through NASA sponsored studies conducted in 1962 and early 1963. Funding for the program was approved in 1963 and in the summer of that year NASA released to industry a Request for Proposal for an "Agena Class Lunar Orbiting Photographic Project." The name was later shortened to "Lunar Orbiter." This request came from the Langley Research Center which had been named the managing center for the project.

The request for proposal defined program objectives and basic system criteria which set the stage for the program as it evolved.

### Objectives

Important to the entire program were the clearly defined objectives which remained unchanged. These were:

#### Photographic

To photograph wide areas of the moon to aid in landing site selection and verification. Also to aid in the verification of landing vehicle design.

#### Environmental

Radiation Environment Measurements
Micrometeoroid Flux Measurements

Selenodetic

Size, shape, and gravity gradient data.

Basic System Criteria

NASA specified a spacecraft system capable of accomplishing the established objectives while observing the following basic criteria:

a. be launched by the Atlas-Agena D

b. each spacecraft to be able to photograph 40,000 square kilimeters to a resolution of 8 meters and, 8,000 square kilimeters to a resolution of one meter

c. be launched from the Kennedy Space Center using existing NASA and AMR facilities

d. be compatible with the existing Deep Space Network

e. make maximum use of qualified components

f. be complete in time to support Project Apollo

g. have an operating life of 30 days in lunar orbit to accomplish the photographic mission with a goal to operate an additional eleven months for the extended mission (environmental and selenodetic data)

Specific system designs to meet these criteria were left to the discretion of the individual proposers. The Boeing Company offered a three-axis stabilized spacecraft designed around a photographic payload, the heart of the system. In December 1963, The Boeing Company was chosen to negotiate with NASA on this program. Negotiations were completed in March 1964, and a contract was signed. The contract was NAS1-3800, a CPIF (Cost Plus Incentive Fee) type. RCA and Eastman Kodak were the major subcontractors to Boeing and assisted in the preparation of the proposal. RCA had the

responsibility for providing the electrical power subsystem and the major elements of the communications subsystem. Kodak provided the photographic subsystem.

Boeing's proposed system became the basis for the negotiated Statement of Work. As the prime contractor, Boeing was responsible for the following services and deliverable items:

Services

a. Overall Project Management

b. Planning and Integration

c. Design and fabrication of eight spacecraft with ground support equipment and spares

d. Qualification, environmental, and performance demonstration testing.

e. Training planning

f. Support of launch and flight operations

g. Photographic and performance data collection

h. Performance data analysis

Deliverable Items

a. Five flight spacecraft

b. Over 200 items of documentation

c. Reassembled lunar photos

d. Acquired scientific data

A work breakdown structure was also negotiated. This structure became the basis for program control. The breakdown divided Boeing's activity into more than fifty 'work packages.' Program reporting to NASA on status, cost, and

schedule were based on these work packages. NASA PERT and Companion Cost methods were employed. This system proved to be a very viable working tool providing good visibility throughout the program.

## Organization

Several NASA organizations in addition to the Langley Research Center had Lunar Orbiter responsibilities. All were under the overall direction of the NASA Headquarters Lunar Orbiter Program Office in the Office of Space Science and Applications. Interagency agreements were negotiated between Langley and the other organizations for the services, hardware, and facilities each was to provide. The agreements also established the working interfaces and relationships.

### Lewis Research Center

Lewis provided the Atlas-Agena launch vehicles, the spacecraft adapters and the nose fairings.

### Kennedy Space Flight Center

The Kennedy Space Flight Center provided the launch facility and launch operations support.

### Jet Propulsion Lab

JPL provided the Space Flight Operations Facility and the Deep Space Instrumentation Facilities, and performed 'mission independent' flight operations.

Working Groups were established between the Langley Center and each of these organizations. Boeing was represented on each of the working groups which met as needed to provide continuity as the various activities progressed.

### Boeing Organization

Provisions were included in the Boeing Lunar Orbiter Organization to accommodate certain features of the program.

Ambitious Schedule To insure the availability of adequately experienced personnel for each of the rapidly progressing phases of the program, the organization provided for the timely transition of personnel from design to test and later into operational functions.

All Up First Flight The first spacecraft to fly was to be fully capable of meeting all performance objectives. There were no test flights planned. Also, the fact that only a few sets of hardware were being produced made it highly probable that each spacecraft, regardless of the controls exercised, would possess subtle differences. A test team concept was therefore employed; one team assigned to each spacecraft during the assembly phase who performed all test and handling operations on their respective spacecraft. The existence of a test team thoroughly familiar with their spacecraft and its idiosyncrasies is felt to have contributed much to the maintenance of difficult schedules and to a confidence in spacecraft readiness.

Uncertainties Several uncertainties had to be accounted for in designing the first American spacecraft to orbit the moon. What were the moon's gravitational properties? Would micrometeoroid dust pit the lenses? Were there concentrations of high energy radiation in the vicinity of the moon which could fog the film? These and other questions were considered by a technical review board made up of prominent scientists who met periodically to consult with the spacecraft designers and mission planners.

Another feature of the Boeing organization was the direct reporting of the reliability and quality assurance managers to the program manager. They were responsible for implementing and managing all reliability and quality assurance provisions specified by contract and as required by internal Boeing policy. The Lunar Orbiter contract specified the application of NPC 250-1 and NPC 200-2.

The size of the Boeing organization was small enough, peaking at about 1100, to allow the design, technical staff and planning personnel to be co-located in one facility. A consensus today of the involved personnel reveals that this was a very important factor which enhanced the

successful coordination of the various activities.

## Design

Spacecraft design activities were based on the proposed Boeing concept. The resulting design was very close to the proposed system, attesting to the validity of the initial concept. The final design did have almost 50% more film capacity than NASA had established in the original system criteria. This is one of the reasons the program was able to exceed original objectives.

Mission design progressed concurrently with the spacecraft design. Close coordination and attention to the interactions between these two functions prior to any design freeze accounts in large measure for the success of the mission operations. Mission designers were able to get the most out of the spacecraft and the spacecraft designers were aware of demands the missions would impose on the hardware.

Rigid configuration control and attention to generic problems were emphasized throughout the design phase. A very effective design review program was conducted at the component, subsystem, and spacecraft levels. Design freezes were sought as early as possible to support the ambitious schedule.

Ground support equipment design, both test and operational, proceeded along with the spacecraft and mission design. Ground support equipment was available and certified in time to test the first assembled spacecraft.

### Spacecraft Design

A brief description of the spacecraft follows:

The spacecraft as flown weighed 850 pounds. (Figure 2) Its lower equipment mounting deck is five feet in diameter. The height from this deck to the rocket engine on top is five and one-half feet. The solar panels when extended span 12 feet. It is 18½ feet across the entended antennas.

The equipment mounting deck holds the photographic,

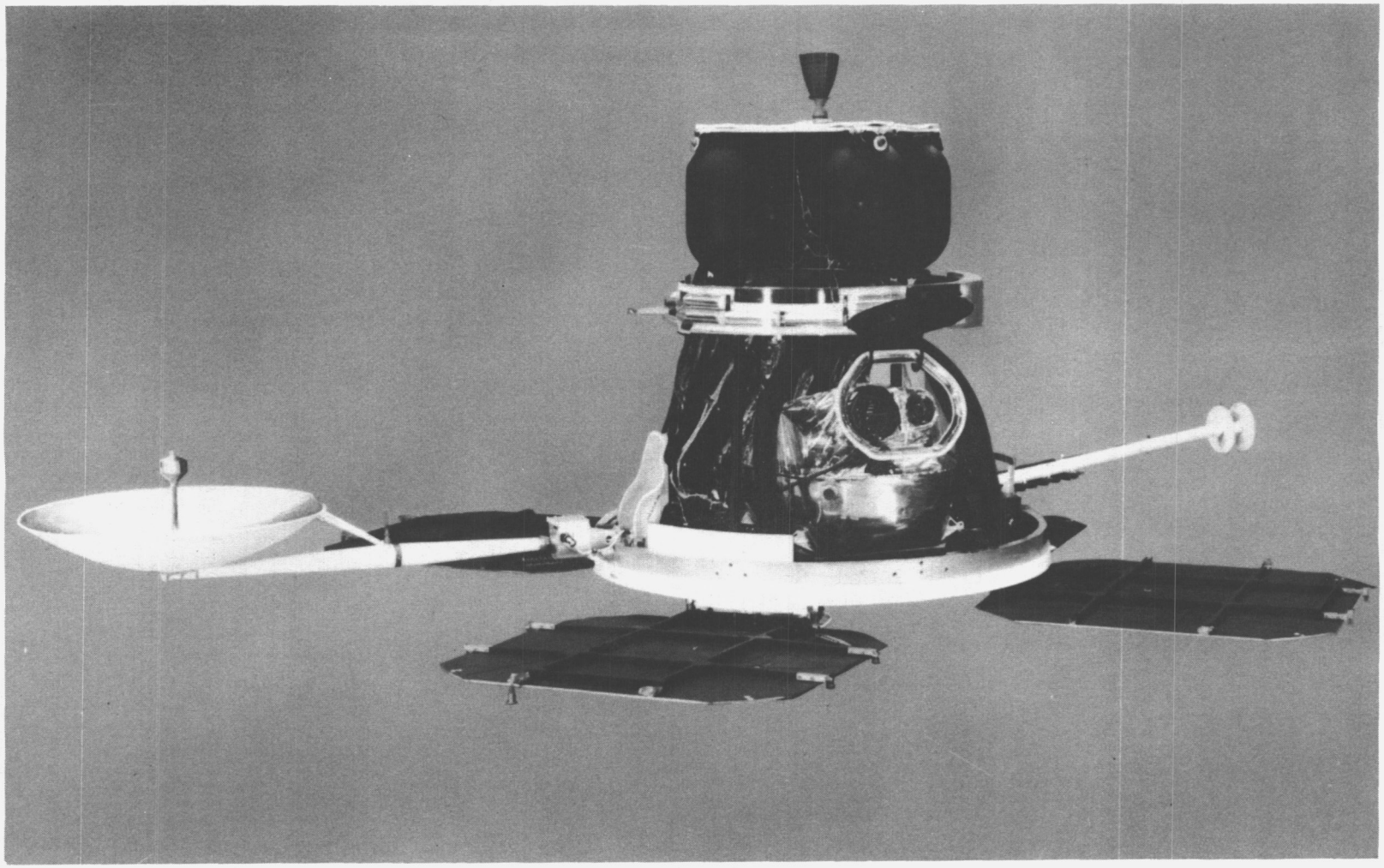


Figure 2. Lunar Orbiter Flight Configuration

communications, power, and command subsystems. Attached to the equipment mounting deck through a truss structure is the propulsion module. The propulsion module contains the velocity control engine with its associated tankage, plumbing and controls, and also attitude control components.

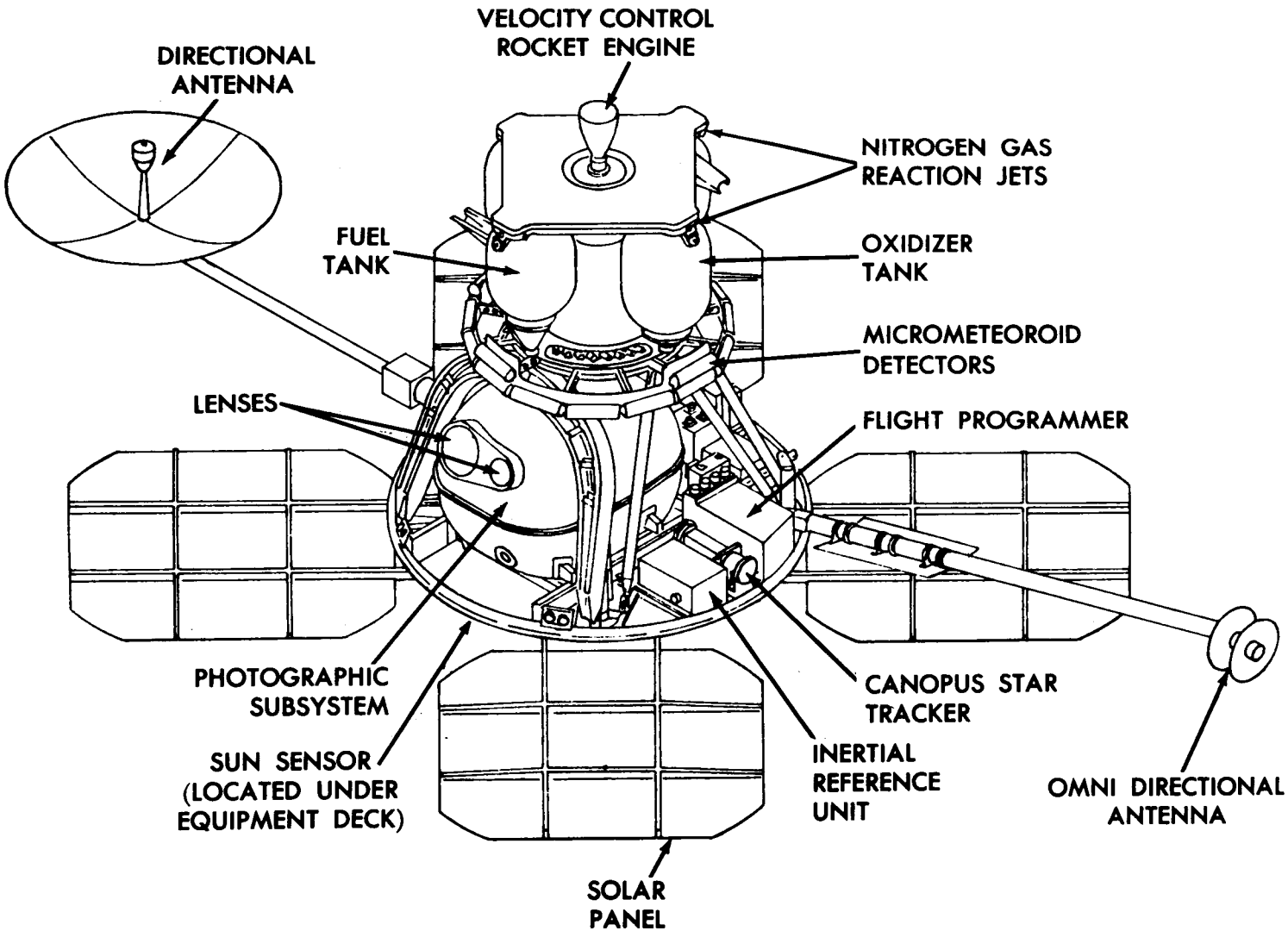
Spacecraft electrical power was provided by four solar panels. (Figure 3) In direct sunlight they produced about 360 watts, enough to operate all spacecraft systems and to charge batteries which handled the load when the spacecraft was maneuvered off-sun or when it was in the moon's shadow.

The velocity control system was used to perform midcourse corrections, retro into lunar orbit, change orbit altitude, change orbit plane and also to crash the spacecraft on the lunar surface - positively preventing them from interferring with future missions. This system used a 100 pound thrust, bi-propellant, gimbal mounted, liquid rocket engine. It was successfully operated 29 times, in some cases after periods of inert storage of up to four months. Lunar Orbiter II's engine was fired for an impact maneuver after almost a year in lunar orbit.

Spacecraft attitude was controlled by cold gas nitrogen thrusters mounted on the upper corners of the propulsion module. Attitude references during the cruise mode of operation were the sun for pitch and yaw and the star Canopus for roll. When either the sun or Canopus were not visible to the spacecraft, or when it was necessary to maneuver off either or both references - as when pointing the camera - an inertial reference unit with three gyros provided the reference.

Communications were S-band--compatible with the Deep Space Network. An up-link for command, and a down-link for command verification and telemetry operated through the omnidirectional antenna and a 0.5 watt transponder adapted from the Mariner Program. For picture data transmission a 10 watt TWTA (Travelling Wave Tube Amplifier) was used to transmit the signal over the three foot directional antenna.

The photographic subsystem weighed 150 pounds. This was a complete photo laboratory containing a dual lens camera,




LUNAR ORBITER SPACECRAFT

Figure 3

film supply, processor, dryer and a readout device to convert the film images into corresponding electrical signals for transmission to earth.

The last major component to be defined for the Lunar Orbiter was the flight programmer. It was last because it was the one element which had a major interface with all other spacecraft subsystems. Because the schedule provided only a short time for its development, and because these were no existing units which could be adapted, it was decided to build the programmer in-house. The flight programmer was a serial data processing computer which employed integrated circuit microelectronics, and contained the precision spacecraft clock and the necessary input-output electronics. It also used a magnetic core memory with a capacity of 128 words of 21 bit length. The programmer sequenced all spacecraft commands which were either stored in its memory prior to launch or reprogrammed into it during flight. Commands to the spacecraft could be executed in real time or stored in the programmer memory for execution at some future time. All spacecraft functions on the lunar far side were controlled from programmer stored commands. These units performed flawlessly on all missions.

## Test Program

The test program was conducted in three major phases: Engineering Development Tests, Component Tests, and Spacecraft Tests.

### Engineering Development Tests

These tests were performed during the design phase. Thermal, static and dynamic spacecraft models, and subsystem breadboards and functional models were used. Specifications and final designs were based on the results of these tests.

### Component Tests

A Lunar Orbiter spacecraft was composed of 35 separate components. Components were fabricated to the specifications and designs which evolved from the development phase. Three complete sets of components were devoted to the component

tests. These were Reliability Demonstration Tests, Qualification Tests, and System Design Verification Tests. Reliability, performance and failure data accumulated during these tests were fed back into the various component designs.

### Spacecraft Tests

Three spacecraft were devoted to spacecraft level ground tests. One spacecraft was used for Qualification testing. Another was used for Performance Demonstration tests and the third was used for Mission Simulation tests. These tests all involved static, dynamic and environmental (thermal-vacuum) operations. Results of these tests were fed back into the design.

### Flight Spacecraft

The components for each flight spacecraft had to pass Flight Acceptance tests before assembly into the spacecraft. Each flight spacecraft underwent spacecraft level Flight Acceptance tests before delivery to the launch site. These tests were also static, dynamic and environmental.

## Mission Operations

Mission operations were centered at the Space Flight Operations Facility (SFOF) of JPL in Pasadena, California. Deep Space Stations located at Goldstone, California; Madrid, Spain; and Woomera, Australia were the link between the Pasadena facility and the spacecraft.

The first thirty-five days of each mission were very busy, requiring round-the-clock operations. Ten to fourteen days of lunar orbit were required for the photography and another fifteen to twenty days were needed to readout the picture data.

Boeing had about one hundred people working a three shift operation at the SFOF during each mission to support the NASA mission directors. An additional thirteen people were stationed at each of the Deep Space Stations to handle spacecraft communications and to receive and process picture data, telemetry and performance data.

Good software, a meaningful training program and personnel who had prior program experience were major factors contributing to the success of the missions.

Operational work-arounds were successfully employed to overcome several inflight difficulties. One good example occured during the mission of Lunar Orbiter I. Early in the translunar flight when an attempt was made to acquire the star Canopus with the star tracker to establish the roll reference, it was discovered that an interfering light source (glint) was saturating the tracker. (The interference was later determined to be reflections from the omni antenna which was white - this antenna was painted black for the subsequent missions). When a 360 degree roll was executed by the spacecraft, the telemetered star tracker data showed a bright object. This was probably the moon but to be sure another 360 degree roll was programmed with the directional antenna oriented such that its transmitted beam would intersect the earth. This maneuver verified that the bright object seen by the tracker was the moon. The moon therefore became the roll reference during the translunar flight permitting accurate mid-course and lunar orbit insertion maneuvers to be performed. As soon as the spacecraft entered the moon's shadow, the source of interfering glint disappeared, verifying that the star tracker had not malfunctioned.

## Summary

Lunar Orbiter made a significant contribution to the American program of lunar exploration. There will very probably be future space programs which could benefit not only from the technical developments of this program, but also from the application of certain of its management features.

It is difficult to single out specific management actions as being more important than others to the success of the program. It can be said however, that the accumulated management actions were successful in creating a motivated team which performed well together, was dedicated, and which succeeded in flying the first spacecraft twenty-eight and one-half months after program go-ahead.

# GODDARD SPACE FLIGHT CENTER TEST PHILOSOPHY AND RESULTANT RECORD

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## GSFC Test Philosophy

Because spacecraft are not only "one-shot" but are also "one-of-a-kind", statistical approaches to testing developed for mass production of consumer goods and weapons are not applicable. Actual flight hardware must be exposed to the expected operating environment to assure a high probability of successful performance in space. In addition, several goals must be achieved by the spacecraft test program.

These goals are:

1. To verify that designs (system, subsystem, and component) meet performance requirements.
2. To verify that particular hardware samples meet performance requirements.
3. To eliminate defects in material or workmanship.
4. To discover unexpected interactions between subassemblies, particularly when the system is exposed to environmental stress.
5. To verify that ground-support and data-processing equipment are compatible with the spacecraft.
6. To train spacecraft operations and data processing personnel.

The various types of tests utilized to attain these goals are identified and discussed below.

Functional Testing

Systems. GSFC has endorsed the full-systems test approach, in which the entire system is tested under conditions which are as realistic as possible. Systems tests generally fall into one of two categories:

1. Functional tests are intended to establish that each subsystem is doing its designated job in conjunction with the other spacecraft subsystems. This category includes calibrations which consist of actually calibrating any spacecraft sensors or instrumentation, as well as establishing a spacecraft performance baseline which can be used for comparison throughout the environmental test phase.

2. Compatibility tests, considered here in their broadest sense, are an extension of the functional tests. This category includes electromagnetic capatibility (EMC) assessments, radio-frequency interference (RFI) evaluations, and magnetic moment determinations. Matchmates may be advisable to determine the mechanical compatibility of the spacecraft with the launch vehicle and the aerodynamic fairing. Finally, the flight spacecraft should be tested with the ground-support and data processing equipment, personnel, and procedures that will be used to operate the spacecraft in orbit.

Subsystems. The full-systems test approach does not eliminate the desirability, or in some cases the necessity, of black-box and subsystem qualification and acceptance testing. Several inherent limitations of systems testing are best overcome by testing at the subsystem or black-box level. This matter will be treated in some detail in a later discussion.

## Environmental Testing

These tests are conducted to determine the ability of components, subsystems, and the entire spacecraft to withstand environmental rigors that may be experienced before and during launch, and during orbital operation. The tests may be conducted at any or all levels of assembly, but are required at the spacecraft level. The severity and duration of the tests are related to the purpose of the tests as indicated below. At times the distinction between functional testing and environmental testing is rather difficult to make particularly when a specific environment, e.g., high vacuum, is required for the conduct of a functional test.

Flight Acceptance Testing. All flight hardware is ground-tested in the expected flight environment. The hardware is exposed to actual simulated loading conditions produced by the effects of temperature, pressure (vacuum) and vibration during the launch phase, as well as the orbital phase, of the flight. Functional tests appropriate to the conditions simulated are conducted. In concept, test levels are chosen such that there is one chance in twenty of their being exceeded in flight.

Qualification Testing. A prototype model (i.e., the actual flight configuration) is exposed to an environment intended to produce loading conditions and stresses in excess of those expected in flight. The purpose of prototype testing is to demonstrate a margin in the design to provide for uncertainties in areas such as analyses, materials, and workmanship. For qualification, test levels are chosen such that, again in concept, there is one chance in one hundred of their being exceeded in flight.

Proto-Flight Testing. As programs mature there arise cases of spacecraft which are based primarily on previously flown hardware but at the same time are somewhat different in design and performance. In these cases a prototype spacecraft may not be required. In such an event, flight acceptance and qualification testing is performed on the flight spacecraft. The amount of qualification testing required is made consistent with the magnitude of the hardware changes.

While the establishment of test levels is defined above in terms of overall risk, the paucity of applicable data makes this task one which does not yield exact numerical values.

## GSFC Test Practice

In its application of the test philosophy discussed above, GSFC recognizes that there are differences between in-house projects and those which are conducted entirely by contract. Both follow the same philosophical approach but the amount of formal

testing below the system level is significantly higher for contractor conducted projects. The major considerations in test planning will be discussed separately for the two cases.

## In-House Projects

GSFC has placed heavy reliance on testing at the system level. There are several consequences of taking this approach. One hopes, of course, to reduce the total amount of testing and its associated cost. On the other hand, the risk of entering systems test with a significant design flaw is increased. Program delay for a "fix" at this stage is very costly, but may or may not be more than complete subsystem test program. The key element in maintaining cost-effectiveness is test planning, together with some knowledge of hardware quality.

Testing implies discovering whether or not something performs as intended. Environmental testing requires that tests be conducted under the kinds of stress to which the device may be expected to be exposed in use. The emphasis here is on investigating the functional performance of the hardware when exposed to the stress of the environment. It is clear therefore that systems test planning must begin by assuring that all the intended functions of the system are checked.

With a general outline of functional testing established, one may proceed to a consideration of the manner in which environmental stresses are to be applied. It is immediately apparent that functional checkout and the application of realistic environmental stress may not be compatible in a systems test. For

example, stimulation of a startracker with a collimator interferes with the tracker's view of "cold space". As each of these situations arises, one must consider carefully the implications. In most cases, it will be found that a subsystems test can be run which makes up for the deficiency in the systems test.

There are other situations which suggest the desirability of subsystems testing to supplement the systems test. For example:

1. Systems test seldom run long enough to detect wear-out problems. Wherever a fatigue or wear-out potential exists (chiefly in electromechanical devices), the "life" characteristics should be investigated on a component or subassembly basis. Great care must be taken to assure that the sample is truly representative of the flight hardware.

2. After a spacecraft is completely assembled, devices may no longer be realistically operable during systems testing. This occurs, in some cases, because the spacecraft cannot be operated in all possible orbital modes on the ground.

3. If only systems tests are run, marginal conditions existing at critical subsystem or black-box interfaces may remain undetected. Only information on the input/output characteristics of the individual black boxes can establish the presence of adequate margins.

In systems testing, the launch environment and the temperature and vacuum aspects of the space environment are emphasized. The effects of space plasma, magnetic fields, and energetic particles must also be considered. In many cases, however, facilities do not exist which permit realistic exposure of the complete system to these stresses. Subsystem tests might seem appropriate, but for these factors one can often avoid an extensive test program by a consideration of the damage mechanism which could possibly be operative. For example, large classes of semi-conductor devices are inherently insensitive to energetic particle fluxes at the levels encountered in space. Testing these would be a waste of money. On the other hand it is important to know the expected degradation of solar cells for the orbit in question. This problem can be attacked with a limited number of samples at the piece part level. Careful "design review" will show that there are many similar situations in which piece part testing for a particular environmental stress is the appropriate technique.

Test planning is then approached by first attempting to run all practical functional tests under environmental stress at the systems level. When either a function or resistance to a particular stress cannot be checked at this level, a special subsystem or piece-part test is devised to make up for this deficiency.

Out-of-House-Projects

In attempting to apply a "systems test philosophy" to a project in which the spacecraft is supplied (and probably tested) by a prime contractor, a considerable degree of flexibility is lost. This arises because the contractual vehicle requires definition of the

manner in which successful performance will be judged at a time when it is not possible to establish in detail the probable deficiencies of the systems test program. One is, therefore, likely to specify a greater depth of subsystem testing than is used for in-house programs. This effect tends to pyramid in that most prime contractors buy "black boxes" from vendors and thus themselves face the problem of establishing success criteria too early in the program to account for those aspects of performance which will be thoroughly checked out later in the project.

In this pattern of tiered vendors, it is easily possible to reach extremes in providing "assurance". All designs are based on worst case analysis and mandatory derating, piece parts are purchased to rigorous specifications, each part is screened and burned-in, each board or module is functionally tested under environmental stress, every "black box" is "qualified", and so on until the complete system has reached the point that there is nothing more one can do but fly it. While this approach would tend to make a well ordered and comfortable world, GSFC makes a conscious attempt to limit environmental testing at lower levels of assembly on out-of-house programs (although not as drastically as for in-house programs).

There are two basic reasons for this tactic: 1) there is no purpose in retesting the same attribute of a device again and again (unless the cost of a "fix" at a later stage is extreme), and 2) there is a very large class of problems which cannot be found by testing below the systems level and which will cause delays at that point in the program in any case. The first reason is based on intuitive judgment which must be modified by the situation at hand. The second reason is fully supported by the data presented by Smith and Waltz in Reference 1.

As part of this study a comparison of results of the two categories (in-house and out-of-house) of test practice will be made at the flight spacecraft system test level, and also in the space performance.

## Systems Test Data

A study of the results of using the GSFC test philosophy has been made using a sample of 24 spacecraft. Fifteen were in-house projects and nine were out-of-house projects. The systems test data on flight spacecraft has been analyzed for information and guidance on the following subjects:

a. Comparison of in-house and out-of-house programs. The latter programs usually have more complex and larger spacecraft, have more rigid quality control features, and have more comprehensive and required testing prior to the systems tests on the flight spacecraft.

b. Distribution of malfunctions by test environments, and by spacecraft subsystems.

c. Learning curves on multispacecraft programs.

d. Time required for an adequate thermal-vacuum test of flight model spacecraft.

## Comparison of In-House and Out-of-House Programs

Figure I shows the performance of flight spacecraft in systems tests. The malfunctions per spacecraft were 17 and 41 respectively for the in-house and out-of-house programs. In each case about 40% of the malfunctions were serious enough to be classified

as failures. No significance should be attached to the relative numbers because there has been no normalizing to account for differing complexity between the spacecraft and programs. The data does show conclusively that 1) the systems tests were needed for both classes of program, and 2) despite rigid quality assurance requirements and required testing for all subsystems, complex spacecraft exhibited a surprisingly high incidence of malfunctions in the systems tests.

The experiments which are flown are often state-of-the-art hardware, and invariably are minimized in weight and size. As such, they might be expected to be more vulnerable to environmental stresses than other parts of a spacecraft. Figure II shows that approximately 50% of the malfunctions and failures have been caused by experiments in both classes of programs. In other words, systems tests would have been needed even if there had been no experiment hardware.

Distribution of Malfunctions

In a separate study directed to the learning curves associated with multiple launch programs, all the malfunctions were categorized by environment and spacecraft subsystems from 22 spacecraft.

By environment All the malfunctions were categorized into one of the following environments: 1) Functional. The malfunctions occurred during operation of the system after it had entered the test phase of the program. No environmental stress was applied. 2) Structural. Included all mechanical type of tests, such as, vibration, acceleration, and deployment. 3) Space simulation. Includes all thermal and thermal-vacuum tests.

TABLE I

Malfunctions Observed in Flight Spacecraft
During Systems Tests of 22 Spacecraft

(Distribution of Test Environment)

| Test Conditions | Excluding Experiments | | Including Experiments | |
|---|---|---|---|---|
| | Number of Malfunctions | % | Number of Malfunctions | % |
| Functional | 210 | 56 | 347 | 46 |
| Structural | 50 | 13 | 95 | 12 |
| Space Simulation | 115 | 31 | 317 | 42 |
| TOTAL | 375 | 100 | 759 | 100 |

Table I shows the distribution of 759 malfunctions found during the systems tests of 22 spacecraft. The high incidence of functional type failures is indicative of schedule pressure, some dependence on the test phase to clean up operational details, and the limitations of the subsystem test program in eliminating problems. The small percentage of structural type problems is a tribute to the design, fabrication control, and test program for this part of a spacecraft program. The space simulation environment uncovers a surprising number of problems considering the rather benign stresses involved. The data in Table I indicate that this environment is especially effective with experiments. If only experiment malfunctions are considered, over 50% are detected during the space simulation tests.

By Spacecraft Subsystems Table II shows the same data as Table I but the malfunctions are distributed among the spacecraft subsystems. Two distributions are given. The one which includes experiments shows that 51% of the total problems are attributed to experiments. While this figure does show the need for improvement in performance to eliminate problems at the flight system level, it must be pointed out that most spacecraft have carried 6 to 10 experiments, and some spacecraft have had 20 experiments. When experiments are excluded, the distribution in Table I shows that 41% of the problems were attributed to the Command and Data Handling hardware.

TABLE II

Malfunctions Observed in Flight Spacecraft During Systems Tests of 22 Spacecraft

(Distribution by Spacecraft Subsystem)

| | Excluding Experiments | | Including Experiments | |
|---|---|---|---|---|
| Spacecraft Subsystem | Number of Malfunctions | % | Number of Malfunctions | % |
| Experiments | - | 0 | 384 | 51 |
| Structure | 56 | 15 | 56 | 7 |
| Thermal | 18 | 5 | 18 | 2 |
| Power | 72 | 19 | 72 | 10 |
| Stabilization and Control | 76 | 20 | 76 | 10 |
| Command and Data Handling | 153 | 41 | 153 | 20 |
| TOTAL | 375 | 100 | 759 | 100 |

## Learning Curves

In a program with five spacecraft launches, the fifth spacecraft could reasonably be expected to have fewer problems at the systems test level than the first spacecraft. Five multilaunch programs were reviewed with the results shown in Figure 3. These programs, involving 22 spacecraft, were the source of the data in Tables I and II. Program 2 was an in-house program, and shows a desirable learning curve for the first three spacecraft. The fourth, identified as a protoflight spacecraft, was sufficiently different in design that it ordinarily would have had a prototype. Instead it was a guinea-pig, in that the flight unit was tested to prototype levels - hence the name protoflight spacecraft. While this could explain the increased number of malfunctions for the fourth spacecraft, no satisfactory explanation is available for the performance of the fifth spacecraft.

Programs 1, 3, 4 and 5 shown in Figure 3 were out-of-house programs. Aside from program four, no learning curve is in evidence, whether or not experiments are excluded from the data.

## Simulated Space Test Time

The literature is replete with discussions of the "bathtub curve" (Figure 4), and the need to get past the infant mortality region of the curve to minimize space malfunctions. Supposedly, if sufficient testing is done in the environmental test phase, the random failure rate region of the "bathtub curve" can be reached and space performance would be maximized. The problem is: "How long do we need to test to eliminate the infant mortality malfunctions?"

A special study was made of theperformance of eleven flight spacecraft in thermal-vacuum tests. Six of the spacecraft were of the in-house type and five were of the out-of-house type. If time under vacuum was the only controlling variable, the number of failures should decrease with time. These data are shown in Figure 5. The premise, if valid, indicates a minimum test time of some 16 days to reach a plateau. However, the influence of thermal stress has not been included. When the same data were segregated by thermal stresses and time, the total time was about the same, but the curve had a significantly different shape as shown in Figure 6. These data indicate an infant mortality curve may be associated with each different kind of stress.

The performance of the two classes (in-house and out-of-house) of programs is given in Figure 7. The data show that the time required for both classes in each thermal environment is about the same. The time (minimum) required for an adequate thermal-vacuum test of flight model spacecraft is about 13 days. The distribution of time is 1, 4, 4, and 4 days respectively, for the ambient, transient, hot, and cold environments. Reference 2 has full details on the study of thermal-vacuum test time.

## Space Performance

### Spacecraft Life in Orbit

The GSFC test philosophy can be evaluated by analyzing the flight experience records. Of the 24 spacecraft discussed previously, 23 have produced useful data in orbit. One spacecraft failed to return information, thus a success record of 96% was

achieved for this sample. Table III is a tabulation of the life of each spacecraft after launch with a comparison made with the intended life. The successful spacecraft have life ranges from 26 to 1500 days with the latter still operating as of January 1969. The intended lifetime depends on the program requirements. The majority of the spacecraft have exceeded their life requirements with many spacecraft having operational performance for an additional year or more. Long life of OGO spacecraft have been discussed in Reference 4. Indications are that the parts selection and control with various test conducted at parts and subsystem level have contributed to the long life of spacecraft.

Test Philosophy vs Orbital Results. The "bathtub" type curve depicting frequency of failure vs time[3] Figure 4, has been a guide for setting up the GSFC test philosophy. The curve suggests that if a spacecraft is completely assembled and checked out, it would experience many failures early in life (usually referred to as infant mortality). GSFC's system environmental tests are designed to locate and correct the failures during this period so that flight operation would be conducted during the "chance failure" part of the curve. A comparison of the failures per spacecraft during test phase and the first 30 days of flight for both in-house and out-of-house tested spacecraft is shown in Figure 8. The large decrease of failures after launch appears to indicate the infant mortality failures have been greatly reduced by the systems test.

A further analysis showing the flight experiences vs time is presented in Figure 9. The ordinate represents the failure per spacecraft per 90 days

TABLE III

Spacecraft Life vs Intended Life
as of January 1969

| SPACECRAFT | | | DAYS | | RATIO= INTENDED |
|---|---|---|---|---|---|
| Explorer No. | Name | Launch Date | Intended Life | Useful Life | Useful Life |
| Tested In-House | | | | | |
| XV | S-3b | 10-27-62 | 60 | 95 | 1.59 |
| XVII | S-6 | 4-2-63 | 90 | 100 | 1.11 |
| XII | S-3 | 8-15-61 | 365 | 112 | .31 |
| XXXVIII | RAE-A | 7-4-68 | 365 | 180* | - |
| XXI | IMP-B | 10-4-64 | 365 | 182 | .50 |
| ARIEL II | S-52 | 3-27-64 | 365 | 194 | .53 |
| XVIII | IMP-A | 11-26-63 | 365 | 300 | .82 |
| XXXII | AE-B | 5-25-66 | 180 | 301 | 1.67 |
| XIV | S-3a | 10-2-62 | 365 | 310 | .85 |
| ARIEL I | S-51 | 4-26-62 | 365 | 320 | .88 |
| XXXV | IMP-E | 7-19-67 | 365 | 469* | - |
| XXXIV | IMP-F | 5-24-67 | 365 | 585* | - |
| XXVIII | IMP-C | 5-29-65 | 365 | 702* | - |
| XXVI | EPE-D | 12-21-64 | 365 | 886 | 2.43 |
| XXXIII | AIMP-D | 7-1-66 | 180 | 913* | - |
| Tested Out-of-House | | | | | |
| | OAO-1 | 4-8-66 | 365 | 0 | 0 |
| | NIMBUS-1 | 8-28-64 | 180 | 26 | .14 |
| | OGO-5 | 3-4-68 | 365 | 302* | - |
| | ATS-3 | 11-5-67 | 1095 | 421* | - |
| | OGO-4 | 7-28-67 | 365 | 541* | - |
| | OGO-2 | 10-14-65 | 365 | 719 | 1.97 |
| | ATS-1 | 12-6-66 | 1095 | 755* | - |
| | NIMBUS-2 | 5-12-66 | 180 | 976 | 5.42 |
| | OGO-1 | 9-4-64 | 365 | 1578* | - |

* Spacecraft continues to operate as of January 69.

and the abscissa is 90 day time periods after launch. Failure is defined as: major if a severe degradation of spacecraft performance is noted; and minor if an experiment is degraded or a spacecraft subsystem is lost but backed up by redundancy. The two curves, one for major failures only and the other for major and minor failures, show a very sharp drop in failure rate just after launch. Thus the "infant mortality" failures have not been completely eliminated during pre-launch tests. This initial peak suggests that the spacecraft experience stresses not adequately simulated in test. First day failures do account for a good portion of the early failures. However, if these were eliminated, there would still remain a sufficient number of early failures to indicate that the infant mortality phase still existed. Indications are that "chance failure" period is not reached until approximately a year in orbit.

The plot also indicates that wear-out will not necessarily be related to any specific time period after launch. Although wear-out of some spacecraft subsystems are known to occur, the failure mode of a complete spacecraft does not appear to increase in numbers at a particular time in the life cycle. Thus a specific "bathtub" curve is not considered applicable in the wear-out period to spacecraft in general.

## Analysis of the Initial Post Launch Failures

The flight failure data have been summarized in Table IV. A total of 109 major and minor failures for 24 spacecraft indicate the following:

First Day

46% of the spacecraft experience 1st day failures.

20% of all failures noted during the life of the spacecraft up to January 1969, occurred in the 1st day.

14% are related to experiments and 6% are related to spacecraft subsystems.

First 30 Days

63% of the spacecraft had failures
35% of all failures occurred in the initial 30-day flight period.

TABLE IV

Summary of Spacecraft Failure Data

| | Spacecraft Tested | | | |
|---|---|---|---|---|
| | In-House | Out-of-House | Total | |
| Number of Spacecraft | 15 | 9 | 24 | |
| Major + Minor Flight Failures | 57 | 52 | 109 | |
| 1st Day: | | | | |
| Experiment | 10 | 5 | 15 | (14%) |
| Subsystem | 2 | 5 | 7 | (6%) |
| | | | 22 | (20%) |
| 1-30 Day: | | | | |
| Experiment | 13 | 11 | 24 | (22%) |
| Subsystem | 5 | 9 | 14 | (13%) |
| | | | 38 | (35%) |
| Number of Spacecraft Having 1st Day Failures (Major + Minor) | 8 | 3 | 11 | (46%) |
| Number of Spacecraft Having 1-30 Day Failures (Major & Minor) | 9 | 6 | 15 | (63%) |

## Success Record

The operational performance for 73 spacecraft tested in accordance with GSFC test philosophy has been appraised relative to the production of useful data and the mission objective. Figure 10 shows a plot of the successful spacecraft in % vs launch year. The record has been very good for the initial ten years of this new technology. In fact GSFC has been able to maintain 100% success for all spacecraft tested at the center.

Experiments are the major operational requirement of all scientific and applications spacecraft. The success of the GSFC program is measured by the useful data obtained. Figure 11 shows a summary of experiment record to date. Useful data have been obtained in some cases where vehicle did not meet orbital requirements and even when the mission was declared a failure as in OGO-1. The experimenter also obtained data in some cases with degraded equipment. The overall data collection record appears to be very good for this type of space program.

## Conclusions

The spacecraft utilized in the GSFC scientific program have, in general, had limited life objectives ranging from 60 days to one year. Two ATS experimental applications spacecraft included in this study have had three year objectives. Experience with these programs shows that the goals have been largely achieved.

- Twelve spacecraft have exceeded design life.
- Four spacecraft are still working toward achieving design life.
- Four spacecraft have failed to reach design life through mechanisms external to the spacecraft.

Four spacecraft have had internal failures before achieving design life.

This represents a good but far from perfect record. Furthermore, while returning very useful data, many of the successful spacecraft are operating in a degraded mode when the goal is achieved.

The authors conclude that:

1. The need for systems level environmental testing has been demonstrated for both small and large spacecraft.

2. Early failures suggest the need for careful tailoring of tests for specific spacecraft.

3. Spacecraft subsystems and experiments share the blame for degraded performance in approximately equal proportions.

4. Multi-launch programs do not eliminate the need for environmental testing of later flight systems.

5. Spacecraft life in excess of one year is readily but not easily achievable.

If the scientific spacecraft program in earth orbit is to be extended to long duration planetary investigations, some changes in approach will be required to satisfactorily attain the long lifetimes required. Again in the author's opinion, this revised emphasis should be in the direction of more conservative design approaches (particularly on the part of experimenters) and a more rigorous discipline in parts selection.

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2. Timmins, A. R. "Time Required for an Adequate Thermal-Vacuum Test of Flight Model Spacecraft," NASA Technical Note D-4908, December 1968.

3. Lloyd, D. K. and Lipow, M., "Reliability Management, Methods, and Mathematics," Prentiss-Hall, Inc., Englewood Cliffs, New Jersey, 1962.

4. Leventhal, A., Bloomquist, C. E., Joseph, J. A.; "Spacecraft Failure Rates--Where are We?" Annual Symposium on Reliability, Chicago, Illinois, January 1969.

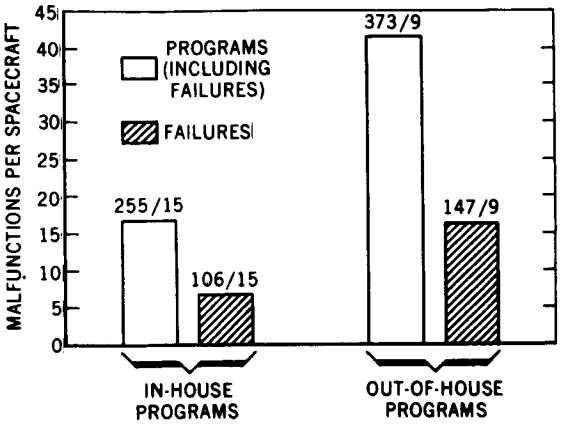


FIG. 1 PERFORMANCE OF FLIGHT SPACECRAFT IN SYSTEMS TESTS

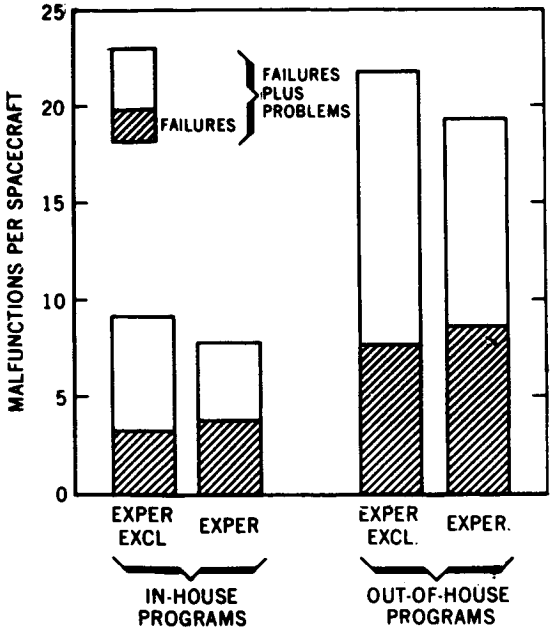


FIG. 2 PERFORMANCE OF FLIGHT SPACECRAFT IN SYSTEMS TESTS

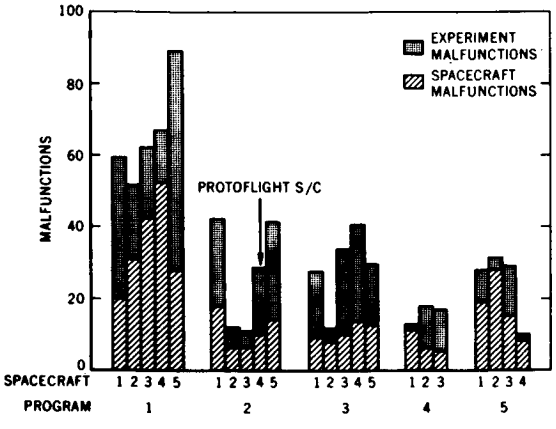


FIG. 3 MALFUNCTION "LEARNING CURVES" ON FLIGHT SPACECRAFT

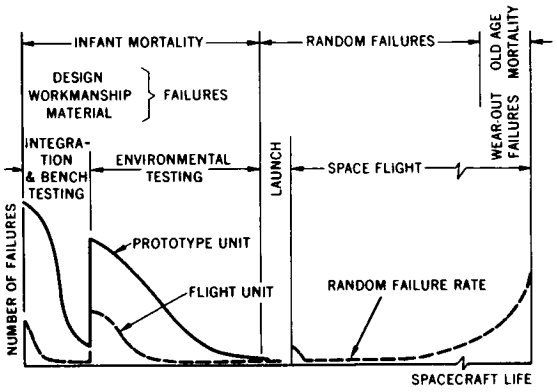


FIG. 4 THEORETICAL FAILURE PATTERN

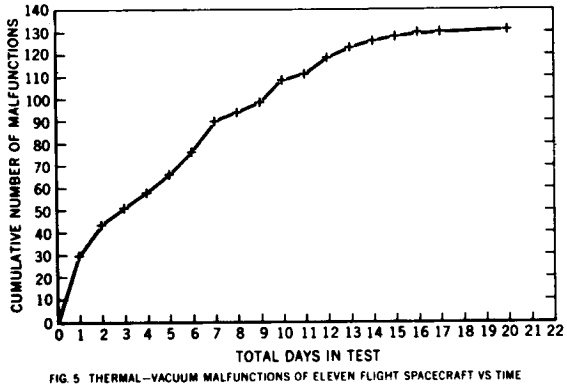


FIG. 5 THERMAL–VACUUM MALFUNCTIONS OF ELEVEN FLIGHT SPACECRAFT VS TIME

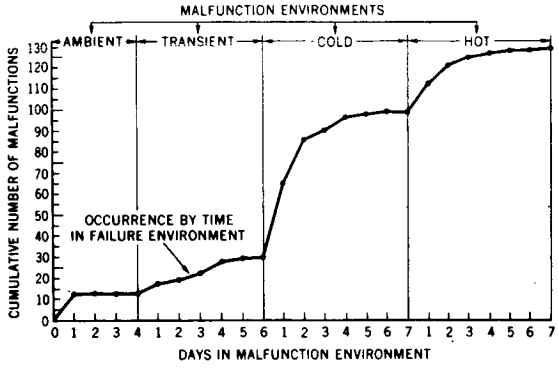


FIG. 6 THERMAL–VACUUM MALFUNCTIONS OF ELEVEN FLIGHT SPACECRAFT VS TIME & ENVIRONMENT

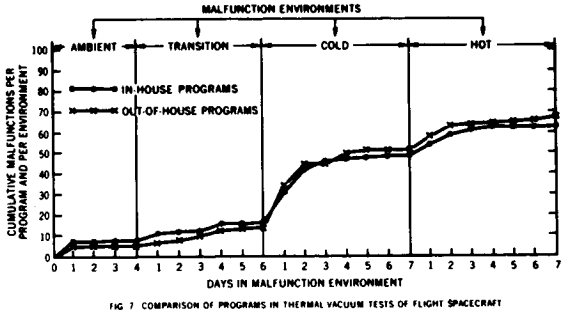


FIG. 7 COMPARISON OF PROGRAMS IN THERMAL VACUUM TESTS OF FLIGHT SPACECRAFT

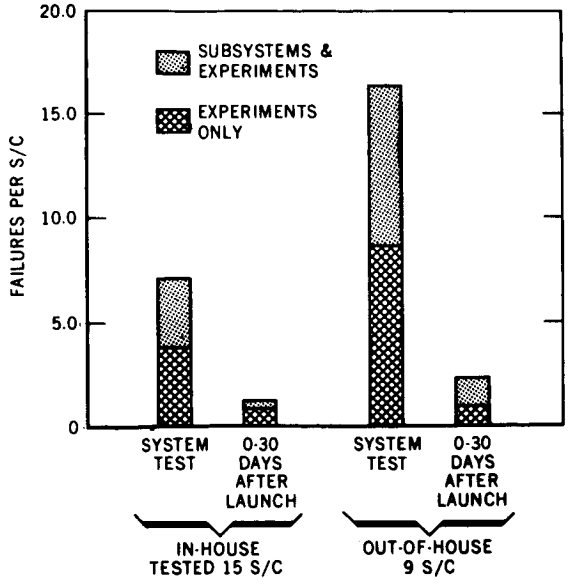


FIG. 8 SPACECRAFT FAILURES TEST VS FLIGHT[1]

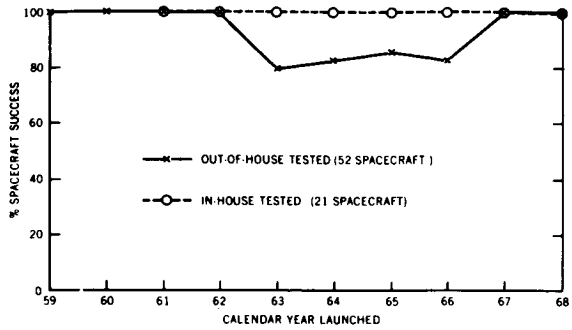


FIG 11 SPACECRAFT FLIGHT SUCCESS RECORD 1959–1968

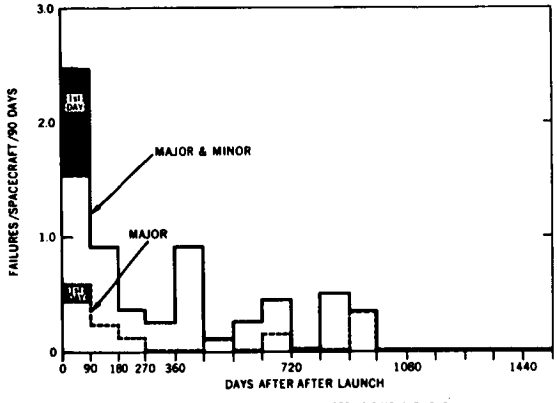


FIG 9 MAJOR & MINOR SPACECRAFT FAILURES VS TIME IN FLIGHT

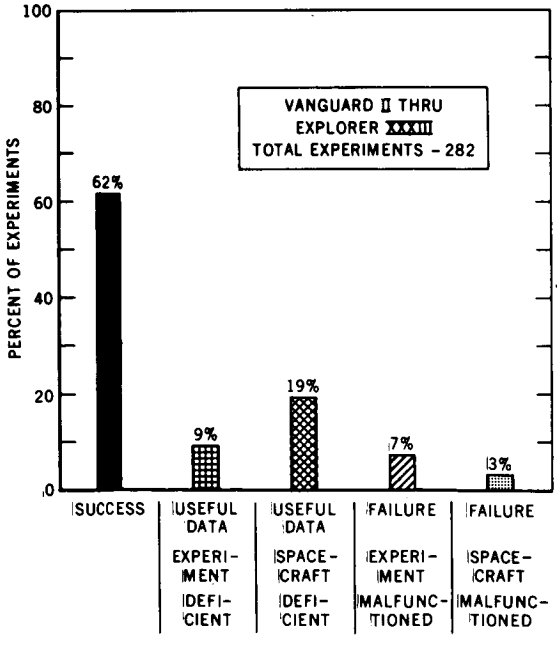


FIG. 10 EXPERIMENT RECORD 1959 – 1966

# TESTING FOR SPACECRAFT RELIABILITY - A MANAGEMENT OVERVIEW

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JULY 16, 1968

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SYMPOSIUM ON LONG LIFE
HARDWARE FOR SPACE,
MARSHALL SPACE FLIGHT
CENTER, ALABAMA, MARCH 18, 1969

## INTRODUCTION

In the aerospace industry, the cost of spacecraft testing is a uniquely important factor to management. Some 50% or more of the allocated program dollars are typically spent in the area of test-parts through system. Thus, large sums of money are involved in test when we consider the individual Program costs involved in our National space effort. Typical of these Program costs are:

- Biosatellite - 135 million
- Orbiting Astronomical Observatory - 280 million
- Surveyor - 483 million

Reference 1 contains a more complete listing of these Program costs.

A fundamental question in the mind of a Program Manager is the attainment of an optimum balance between test dollars spent and results achieved - results in terms of spacecraft reliability and performance per customer requirements and expectations. How, what and when to test, then, becomes a central consideration in sound test planning and program management.

## A CORE ISSUE - TESTING FOR RELIABILITY

Many philosophies and approaches currently exist to guide the formulation of a spacecraft test program. These emanate primarily from past experience and judgement. One might conclude that, today, test planning is more of an art than a science. The dilemma starts right at the piece-part level (e.g., how much burn-in and screening should be performed) and continues on through various levels of spacecraft assembly to the launch site itself (e.g., should extensive prelaunch checks be conducted or is a "hands-off" policy in the field the way to go).

Undoubtedly, there are strong arguments, pro and con, for almost any decision that is made, and again one might conclude that there is more than one 'optimal' test approach. Our current state-of-the-art certainly suggests that this is true. However, there is some evidence to support the case which states that certain aspects of spacecraft testing may be quite basic in nature, and to neglect them is to invite major voids in a test program.

At the very highest test decision level, there are two basic approaches that may be taken:

1) Test to pass - i.e., show that the equipment operates to the intended function.

2) Test to find defects - i.e., prove that no defects are in the equipment to prevent the performance of the required function.

The distinction here is subtle, but quite sharp - especially with respect to high reliability achievement. For management, this distinction may be quickly summarized by invoking some 'good-ole Charlie Brown' thinking:

- Happiness is no ground test failures, thus my cost/schedule commitments are not in jeopardy ( test to pass).

vs

- Happiness is discovered and corrected defects, thus my performance commitments are in less jeopardy ( test to find defects).

This paper adopts the theme that "Testing to find defects" is the prime concern. The achievement of excellent spacecraft performance rests on several factors - not the least of which is the test that finds the defects and provides the opportunity for corrective action before operational use. It is felt that this approach is a significant factor in achieving the optimal test results for the dollars spent - and from a broader point of view, to likewise achieve cost effectiveness in the total program sense.

Test program planning, based on past experience, provides management with the opportunity to effectively utilize lessons learned - lessons that usually deal with the generic problems of defect escapes into the operational/use phase of a spacecraft system. "Testing to find defects" thus demands a continuous vigilance to seek out, understand and eliminate these escape paths. This paper directly addresses this question of lessons learned and through analyses performed on five complex spacecraft systems, examines some trends in the following areas:

- Test vs Level of Assembly
- Test Screen Adequacy
- Defect Cause
- Environmental Test
- Defect Generation

Observations are drawn which are suggested as useful guidelines in developing test policies that impact on the "quality" of spacecraft systems. The information presented is by no means an exhaustive study, but hopefully it does present some additional perspective on the important subject of test planning and management.

## DATA ANALYSIS PROCEDURE

Records of ground and flight test failures (defects) from five spacecraft programs form the basis for the analyses presented in this paper. In each program, all defect records were examined initially, and those which were of a catastrophic or major degrading nature to the mission (or potentially so), or of sufficient severity to have warranted corrective action measures in the program, were selected for use in the analyses. This provided a sample of some 1500 defects, collectively, from the five programs.

Defect records so selected were then individually investigated in detail by a team of 3 to 5 engineers; and in over 90% of the defect cases so studied, the team included a system engineer from the specific program under study. Periodic inputs from subsystem specialists were also utilized when the team requested such assist. As each defect was investigated, several characteristic. f the defect were categorized and from these dat. the analyses described in subsequent paragraphs were derived. The authors were, themselves, members of the defect analysis team on three of the five programs reported, and have also performed directly a large percentage of the system analyses and correlations accomplished with the data (of which this paper is representative).

The five spacecraft programs all involved complex spacecraft systems - i.e., the spacecraft all had active, autonomous, 3-axis stabilization (no passive or spin stabilization) and a significant content of electro-mechanical equipment (as opposed to predominantly electronic equipments). The Programs are identified only as Programs A, B, C, D and E since in some cases it was not possible to obtain all of the appropriate approvals to reveal the source of the data used in the analysis.

The results presented here have been supported on a Company sponsored Long Life Space Systems Program ($L^2S^2$) whose overall objective is the solution of problems related to reliability and long life achievement in Spacecraft. This Program includes several Research and Development activities, both of an analysis and hardware experimentation nature. The specific data described in this paper represent but a small portion of the total data accumulated and analyzed with respect to the lessons learned activity that is currently in progress.

## DATA ANALYSIS AND RESULTS

### *Test vs Level of Assembly*

During the build-up and assembly of a spacecraft, it is fairly common practice to test, to one extent or another, each progressive level of assembly. Generally, the lower the level of assembly at which the test is conducted, the less expensive the test becomes. Defects are more readily discovered and corrected at the board/module level of test than by exercising an entire spacecraft. On the other hand, certain defects will not exhibit themselves until a higher level of assembly has been reached. This is true of many harness, plumbing, thermal and electromagnetic compatibility problems. This raises the question of the importance that should be attached to testing at various levels of assembly, and particularly emphasizes the question of the importance of subsystem and system level testing where test costs increase sharply.

Some perspective on this problem can be seen in the data of Figure 1. These data represent the flight defects experienced on Program A - hence, when the defect occurred, the spacecraft was physically in a system level configuration. Each defect was individually investigated to ascertain the specific level of assembly that was necessary in order for the defect to manifest itself. Each defect was categorized according to one of the four categories shown on Figure 1. As one might intuitively suspect, many of the defects could readily manifest themselves without the necessity for an entire spacecraft assembly around them. Others did require interaction effects before they would reveal themselves.

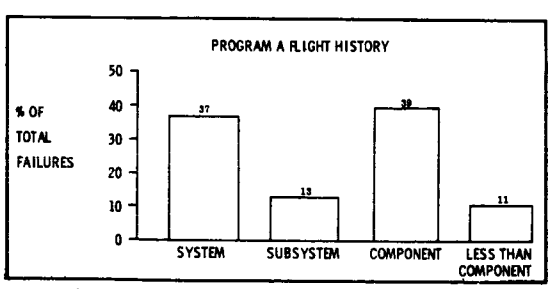


**FIGURE 1. DEFECT OCCURRENCE VS LEVEL OF ASSEMBLY**

The surprising picture represented by these data is that 50 % of the defects required a subsystem or system level assembly before they would occur. In other words, even "perfect" component (black box) level testing would have only reduced the flight defect history by one-half. While the mere occurrence of flight defects is sufficient evidence to note that tests need improvement (something we all know), these data seem to say that tests at the higher levels of assembly are just as important as those at the lower levels of assembly (something that perhaps we all do not fully recognize). Testing at the subsystem and system levels can indeed become expensive, but decisions to "lightly" treat these tests at the higher levels of assembly could well become penny-wise and pound-foolish decisions in the ultimate outcome.

### *Test Screen Adequacy*

As noted in the prior discussion, tests are not perfect. They generally do not represent a 100% screening capability for all of the defects that are present and potentially detectable at a given level of assembly. Exceptions have been noted where 100% screening effectiveness has been achieved - but that is the trouble, they are exceptions. Here, we will examine two cases to illustrate the problem associated with test screen adequacy.

Figure 2 presents data from the five spacecraft programs. Each bar represents the average number of defects in each flight spacecraft when it was delivered to system test. One might consider these values to be a measure of the quality level of the spacecraft as they entered system acceptance test. As the data indicate, there is a fairly wide variation in this quality level from program to program (which in itself may be an interesting facet to explore). The question here is 'how many of the defects present at entry into system acceptance test should have been removed by the prior component acceptance test screen'. The cross-hatched area on each bar shows that from 36% to 65% of all such defects are present because of inadequacies in component acceptance test - "leakage" through the component test screen, so to speak. The open area on each bar, in turn, represents those defects requiring a system level test for their detection - as discussed in the prior section. Other studies not presented here, have shown that component acceptance tests are actually fairly effective in terms of the defects which they do find versus those which escape to system test. However, the "leakage" that does occur places a sizeable burden on system test, and a relatively small improvement in component acceptance tests could well produce a significant beneficial effect on system test from a cost and schedule point of view.

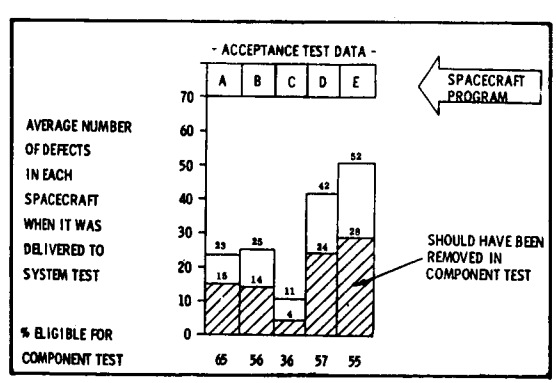


**FIGURE 2. COMPONENT VS SYSTEM LEVEL TESTING**

Figure 3 starts with exactly the same data base as that of Figure 2. The question now is 'how many of the defects present at entry into System acceptance test should have been removed by the prior qualification test program'. The cross-hatched area on each bar shows that from 17% to 68% of all such defects are present because of inadequacies in the qualification test program. There is evidence in other studies

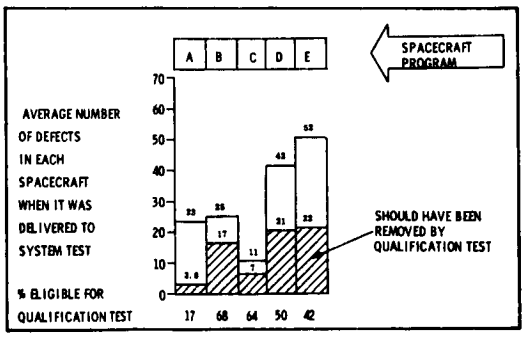


**FIGURE 3. QUALIFICATION VS ACCEPTANCE TESTING**

currently in progress which show that qualification testing (unlike component acceptance testing) is most inadequate. Qualification testing, then, is most certainly an area in need of significant improvement.

### *Defect Cause*

A better understanding of the cause of defects will assist program management in two ways. First, it will help to identify those development and/or production activities in which the greatest improvements may be made and second, it will provide some insight into the means by which test screen adequacy may be improved.

This analysis divides defect cause into three categories:

- Design - a defect which is the result of improper or inadequate design (logic, application, producibility).
- Manufacturing and Test Procedure - a defect which is the result of an error in fabrication, assembly or test of the equipment (the "workmanship" problem).
- Parts/Materials and System Requirements - a defect which is the result of a "random" part or material error or an error in the original system specification.

Data from all five programs are shown on Figure 4, categorized by these definitions. The point of reference is again the average number of defects present in each spacecraft when entering system acceptance test. On Figure 4, by averaging across the five programs, it is noted that the number of design caused defects are about equal to the number of defects attributed to manufacturing/test procedure errors. This is somewhat surprising since the popular opinion seems to be that the workmanship problem (if any) predominates at this stage in a program, and that the Qualification Test Program has already proven that the basic design is satisfactory. Apparently, nothing could be further from the truth! (As previously noted, Qualification testing appears to be grossly inadequate). However, the small number of defect causes rooted in parts and materials or system requirements does follow the pattern generally believed by popular opinion to be the case.

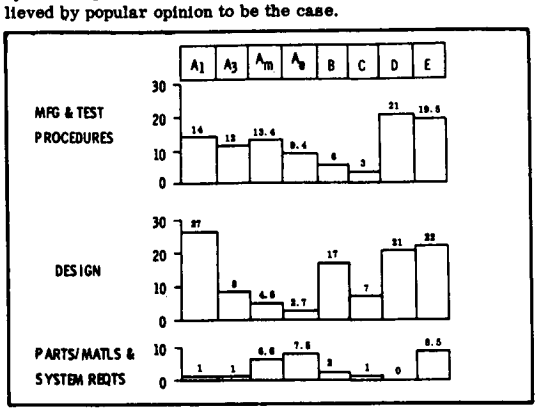


**FIGURE 4. DISTRIBUTION OF DEFECT CAUSES PRIOR TO SYSTEM TEST**

Notice that Program A is represented by four bars. $A_1$ represents the first spacecraft in the Program, $A_3$ the third spacecraft. $A_m$ represents a group of vehicles at mid-program, and $A_e$ a group of vehicles at the end of the program. With respect to the "workmanship" caused defect, there was no discernible decrease in those defects across Program A until the $A_e$ group of vehicles. This could mean that the learning curve so often discussed and applied in spacecraft programs does not take place as rapidly as one would expect. On the other hand, the number of design caused defects entering system acceptance test falls off rapidly as the program progresses (as each one is discovered and corrected, it does not reappear like the workmanship defect). Nonetheless, even the $A_e$ group still had design caused defect in it which means that these same design defects were slipping through test after test, flight after flight, without detection.

Figure 5 presents the flight defects for Program A, by cause. The design caused defect now accounts for 51% of the total flight defect population - twice as much as either of the other categories. This, too, runs contrary to the popular opinion that "workmanship" problems are the predominant cause of failure in space. This information together with that in Figure 4 leads to the conclusion that our ability to detect and eliminate the workmanship defect during ground test far exceeds our capability to do likewise for the design caused defect.

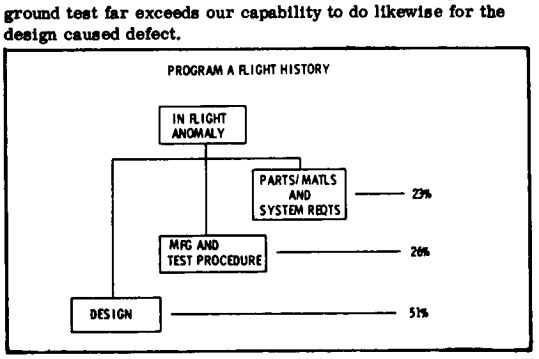


**FIGURE 5. DISTRIBUTION OF DEFECT CAUSES IN FLIGHT TEST**

Figure 6 now takes the design caused defects from the flight data on Figure 5 and examines some additional characteristics about them. The data indicate some interesting trends associated with the design caused defect. Of particular note is that over three-quarters of these defects required the subsystem or system level of assembly for the defect to manifest itself. This is in contrast to the data of Figure 1 where, for all defect causes, only 50% of the defects required the higher levels of assembly in order to manifest themselves. Apparently, the design type problems predominate at the higher levels of assembly, while the "workmanship", part and material, and system requirement type problems predominate chiefly at the lower levels of assembly.

PROGRAM A

- 18% INVOLVED AN EMC PROBLEM.
- 39% WERE ENVIRONMENTALLY SENSITIVE.
- 79% REQUIRED A S/S OR SYSTEM LEVEL OF ASSEMBLY FOR DEFECT TO MANIFEST ITSELF.
- 64% OF THE COMPONENTS INVOLVED WERE NON-ELECTRONIC.
- 52% OF THE COMPONENTS INVOLVED WERE CONSIDERED SIMPLE (VS MODERATELY COMPLEX OR COMPLEX).

**FIGURE 6. SOME ADDITIONAL FACTS ON THE DESIGN CAUSED FLIGHT ANOMALIES**

The potential significance of the environmentally sensitive defect problem is illustrated on Figure 7 with Program A data. When the total defect history subsequent to component test is considered (both ground and flight test), it would appear that only a relatively small percentage of that defect population is environmentally sensitive - i.e., 7%. But when the flight test defects alone are considered, the environmentally sensitive defects jump to 40% of the total. This would indicate that while the environmentally sensitive defects appear to be a small part of the problem, if not properly considered and eliminated in ground test, they can easily become a large part of the defect experience in flight (where the ultimate in an environmental test is achieved).

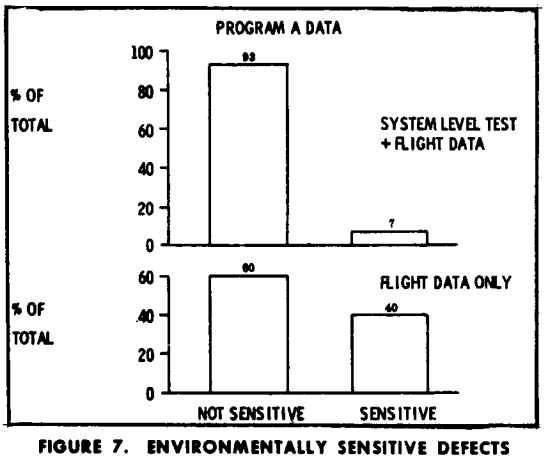


**FIGURE 7. ENVIRONMENTALLY SENSITIVE DEFECTS**

### *Environmental Test*

One of the more difficult decisions facing a Program Manager is the question of environmental test - i.e., how much and at what level of assembly. Cost is a significant element in the decision because environmental facilities and the operation of these facilities do not come cheaply. As vehicle size increases and the question of system level environmental test comes to the fore, technical factors also enter the decision process with respect to the capability that may be present in the facility to supply the necessary energy input to the vehicle (be it mechanical, acoustical, electro-magnetic or thermal). These cost and technical considerations must be judged on their own merits for each individual program, and the appropriate decision made. However, where cost and technical factors warrant the consideration of environmental test, it does become important to consider just what might be gained (or lost) on the decision to use environmental test. In particular, the following analysis addresses the question of environmental test at the system (or spacecraft) level of assembly.

Each defect in the analysis was examined to determine if it was environmentally sensitive; i.e., would the probability of detecting this defect be significantly increased by the presence of an environment other than ambient. For this analysis, only thermal vacuum and vibration environments are considered.

A more concrete illustration of the potential effectiveness that can be achieved with spacecraft level environmental tests is shown for Program A on Figure 8. Program A, a multi-vehicle program, experienced various patterns of system environmental test. System vibration was performed on groups of vehicles at the beginning, mid-point, and end phases of the program with groups of 'non-vibration' vehicles in between. System thermal vacuum was performed primarily on vehicles in the latter phase of the program. As Figure 8 illustrates, the total number (ground plus flight) of vibration or thermal vacuum sensitive failures was essentially a constant, irrespective of the test history. But where systems vibration or thermal vacuum testing was performed, a significant reversal in the ground vs flight defect pattern is observed. That is, more environmentally sensitive failures occurred in ground test than in flight. Statistically, flight failure reductions of 72% and 50% were realized for the vibration and thermal vacuum sensitive defects respectively when ground environmental test was employed on the spacecraft.

The pattern thus described for Program A does not appear to be unique. Returning to an analysis involving all five spacecraft programs, their effectiveness in detecting environmentally sensitive defects in system acceptance test is shown on Figure 9. Effectiveness here is defined to be:

$$E = \frac{\text{Defects Detected (Environmentally Sensitive)}}{\text{Total Defects Present (Environmentally Sensitive)}}$$

PROGRAM A

| TYPE OF TEST | PERCENT OF SPACECRAFT INVOLVED | NUMBER OF ENVIRONMENTALLY SENSITIVE DEFECTS DETECTED PER SPACECRAFT | | |
|---|---|---|---|---|
| | | GROUND TESTS | FLIGHT | |
| VIBRATION | | | | |
| ALL TESTS WITHOUT VIBRATION | 33% | 0.17 | 0.50 | 72% REDUCTION |
| ALL TESTS WITH VIBRATION | 67% | 0.54 | 0.14 | |
| THERMAL VACUUM | | | | |
| ALL TESTS WITHOUT THERMAL VACUUM | 56% | 0.05 | 1.50 | 50% REDUCTION |
| ALL TESTS WITH THERMAL VACUUM | 44% | 0.75 | 0.75 | |

**FIGURE 8. BENEFITS DERIVED FROM SPACECRAFT ENVIRONMENTAL TEST**

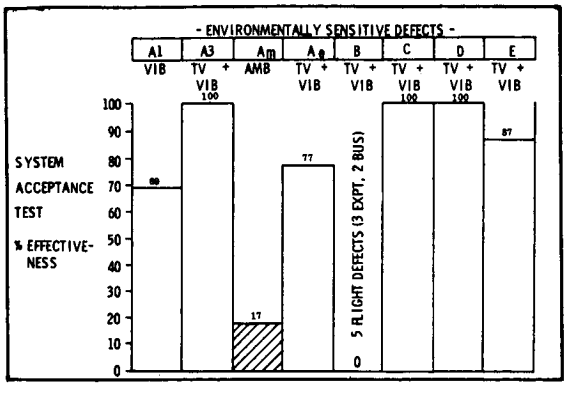


**FIGURE 9. SIGNIFICANCE OF ENVIRONMENTAL TEST**

In general, those programs which included system environmental acceptance test were effective in removing the environmentally sensitive defect with effectiveness values, E, ranging from 69% to 100%. The one exception is Program B for which no explanation has been found. One portion of Program A (cross-hatched bar) did not receive system environmental acceptance test, and its effectiveness value was only 17%.

### *Defect Generation*

Not only are test screens imperfect but quite often, we also find that tests are, in themselves, a generator of defects. Errors in test and installation procedures or faults with test equipment are the underlying cause of this problem.

The defect generation problem is schematically illustrated in Figure 10, using Program A data. A common group of vehicles is depicted in their flow through System Acceptance Test (SAT) at the factory and Launch Acceptance Test (LAT) at the launch site. Defects that were present in the vehicles and eligible for detection in SAT are shown entering the block for SAT. Within the SAT block, we see defects generated during the course of the test. Flowing out of the SAT block are those defects that were detected and diverted, and those which escaped downstream to the launch site. Defects entering the LAT block are the sum of the defect escapes from SAT and an additional group of defects that completely by-passed SAT for reasons of ineligibility (e.g., certain equipments containing defects were not installed until the launch site) or decision (e.g., defects detectable by a test that could have been, but was not run in SAT). Defects generated, diverted or escaping LAT are also shown. Notice that defects escaping LAT are those that go into space, later to be detected there.

There are three points of interest in these data. First, defects are indeed generated in test - 14% of the total defects flowing through SAT were generated there, and 52% of the defects flowing through LAT were generated there. Second, the data would seem to indicate that the defect generation problem is higher in tests conducted remote from the factory site. The number of LAT defects generated was 50% higher than those defects generated in SAT. This latter observation is not to be construed as an indictment of the skills or motivation of the field test personnel, but rather seems to reflect the fact that field conditions (schedule pressures, logistics problems, retrofit actions, drop-ship installations, unavailability of special test equipments and skills that exist at the factory, etc) are in themselves a precipitator of conditions conducive to defect generation. (Compare, for example, the environement in a factory laboratory test to that at 2:00 AM on the fifth level of a launch gantry!) Third, virtually all of the defects generated (~ 98%) are immediately detected and diverted. However, the very small number of generated defects that do escape (especially in LAT) seem to often be subtle and deadly. Typically, these kinds of defects occur when a final check is made on an item of equipment, and if an error results there is no further downstream check that will catch it. (For example, a final leak check is made on a plumbing system, and the technician forgets to replace the cap on the test port - in space, the gas system is pressurized and all the gas dumps overboard through the test port.) This analysis simply points out the extreme care that must be taken in the development of the test procedures, and the proper training of the technicians who must implement those procedures.

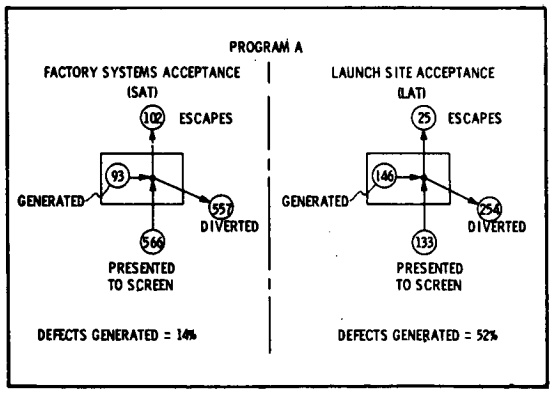


**FIGURE 10. DEFECT GENERATION IN TEST SCREENS**

## SUMMARY OF OBSERVATIONS

The analyses presented above suggest several points which may be useful guidelines in the development of a test program. Several of these are presented below.

1) Equipment defects, and their detection in test, do not seem to be concentrated at any one level of assembly. In fact, the subsystem and system levels of assembly may contain a significant portion of the total defect population. Design caused defects observed in flight, in particular, seem to concentrate at the higher levels of assembly.

2) Test screens are not perfect. As a case in point, system acceptance tests (in addition to searching for the subsystem and system level defects) are "burdened" with the responsibility of searching for defects which escaped prior component acceptance tests or the qualification test program.

3) The design caused defects are subtle in nature, and they are discovered and eliminated chiefly through the "serialized" testing that takes place via repetitive ground (and in some cases, flight) test screens.

4) While design and workmanship type defects are about equal in magnitude in vehicles entering system acceptance test, tests are much more effective at finding the workmanship defect. As a result, the primary cause of equipment failures in flight seems to consistently be the design caused defect.

5) The learning curve associated with workmanship quality does not seem to experience in practice the growth that is so often attributed to it in theory.

6) There are certain defects that are environmentally sensitive. Subsequent to component test, they seem to represent less than 10% of the defect population that may exist in a spacecraft program. However, if not properly treated, they can become a significant portion of the defect population that escapes into flight.

7) Proper treatment of environmentally sensitive defects signifies the use of environmental tests. As a case in point, the effectiveness of system acceptance tests in detecting this type of defect is improved significantly when system level environmental tests are employed.

8) Tests and related test activities do, in themselves, generate defects. This generation problem seems to be aggravated at test sites remote from the factory. Attention to detail in test and installation procedures and test personnel training is necessary to minimize this problem.

## CONCLUDING REMARKS

Spacecraft testing is expensive - and in the immediate future, there does not appear to be any significant breakthrough that will alter that picture, and still result in continued and consistent program success. But the efficient use of test dollars (i.e., optimizing results achieved for the dollars spent) does appear to be a subject in which a great deal of constructive progress can be made - and now.

It is suggested that one such aspect of that progress is the deliberate attempt to structure test programs to find defects. To this end, it is felt that lessons learned from past spacecraft experience can provide a source of invaluable information.

A concerted effort by management to implement that philosophy does, however, require what some may consider to be difficult decisions:

- It requires that much of our current thinking about testing be re-adjusted. For example, passing a qualification test does not really qualify the hardware, and even perfect component testing does not solve the defect elimination problem because a large percentage of such defects do not even appear until the higher levels of assembly are reached.

- It requires that management devote considerably more attention to the reasons for defect escape through a test, and commit the necessary funds for the development of technologies that will permit us to close that gap.

- Most importantly, it certainly follows that any deliberate attempt to find defects will indeed increase the number of defects found. This will have both a cost and schedule impact on the program if there have not been adequate plans and provisions to handle it. This, in turn, requires that management recognize the need for an expanded failure analysis and corrective action system.

But a wise and prudent management will clearly recognize the necessity for such actions, and will clearly understand the longer range benefits to be derived on the basis of the total life cycle of the Program.

A rather well known adage summarizes quite nicely the message that is central to this issue. "An ounce of prevention (cost of effective ground tests) is worth a pound of cure (cost of mission loss due to flight failure)".

## REFERENCES

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# AN ONBOARD MALFUNCTION DETECTION/ANALYSIS SYSTEM

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## Introduction

Air Force direction for the C-5 proposal required a maintenance system capable of providing maximum reliability, accuracy, speed, flexibility, and simplicity. Further, the system had, as a design goal, the requirement for testing down to the lowest line replaceable unit (LRU) and a subsequent announcement to the crew as well as a record-keeping entry. Previous experience with onboard malfunction detection systems indicated that a manned maintenance system offered more potential capability than a completely automatic and un-manned system. Since the C-5 crew complement did not include a person whose prime purpose was maintenance, Lockheed investigated the incorporation of the maintenance function into the duties of a crew member. The logical choice was the flight engineer and the next step was to estimate the time that could be spent performing the maintenance role. A human engineering task analysis indicated that, during normal cruise conditions, approximately 50 per cent of his time could be allocated for maintenance actions.

The synopsis of this direction and experience was to provide a system capable of either identifying a failed LRU immediately upon failure, or predicting the occurrence of a failure. The results of such a system would, therefore, improve turn around time, reduce spares requirements, eliminate specific aerospace ground equipment (AGE) items, and decrease maintenance training requirements. To better define the requirements for an onboard malfunction detection system, Lockheed investigated the broad spectrum of maintenance techniques presently employed. In general, these techniques are as shown in Table 1 in the order of their complexity.

Table 1 - Malfunction Detection Techniques

| | Technique | Announcement |
|---|---|---|
| 1. | Functional manipulation, use of existing instruments | Operator decision |
| 2. | Examination of LRU test signals | |
| | a. Ground data reduction | Recording medium processed to determine faulty LRU |
| | b. On-board data reduction | Signals conditioned, logic levels derived/ processed, LRU equations solved |
| 3. | Built-In-Test Equipment (BITE) | Operator initiated, functional loop tests |
| 4. | Automatic self test | |
| | a. With summary logic | Monitored system provides logic levels for decoding |
| | b. With internal decision | Monitored system provides failed LRU identification |

The first technique uses the normal aircraft controls and indications to detect malfunctioning line replaceable units (LRU's). An example of this technique would be the detection of a faulty fuel pump by re-routing manifold flow paths and observing pressure indications. The efficiency of this scheme, however, is directly proportional to the ability of the operator and is determined by training level, experience, and motivation.

The remaining techniques require on-board hardware in varying degrees of complexity. For ground data reduction, the flight hardware can be relatively simple whereas the on-board reduction scheme requires computation ability. BITE techniques require operator initiation and temporarily interrupt the normal operation of the monitored system in most cases. Automatic self-test is performed on a non-interference basis and does not alter the monitored system performance.

The announcement of detected failures is usually dependent upon the complexity of the system. Large systems could contain decision-making circuitry whereas simpler systems may output sufficient logic signals to permit decisions by auxiliary equipment.

After reviewing the directions, goals, and techniques available, the specific design criteria shown below were established to permit system conception.

Table 2 - Design Criteria

1. Malfunction detection to LRU level
2. Continuous monitoring, non-interference with normal system operation
3. Announcement of failed LRU identity to operator
4. Minimum operator training
5. Flexibility for addition/deletion of monitored test points
6. Recording of data for trend analysis
7. No addition of stimuli generation
8. Visual memory of subsystem status by annunciator panel

The resulting system took the concept shown in Figure 1 and exploits the advantages of the malfunction techniques previously mentioned, permitting each to contribute in the area in which it performs in the most efficient manner. Lockheed identified the system as the C-5A Malfunction Detection, Analysis and Recording System. The acronym "MADAR" was quickly adopted and will be used henceforth in this paper.

As indicated in Figure 1, the system consists essentially of two subsystems - automatic and manual - which share the printout unit and maintenance data recorder for record keeping. The automatic portion operates whenever the switch is "on". The manual portion, however, is used at the discretion of the operator.

The automatic mode monitors LRU's which have signal data outputs amenable to digital techniques. Data sampling, value comparison, limit detection, and malfunction printout are performed automatically according to a preprogrammed scheme.

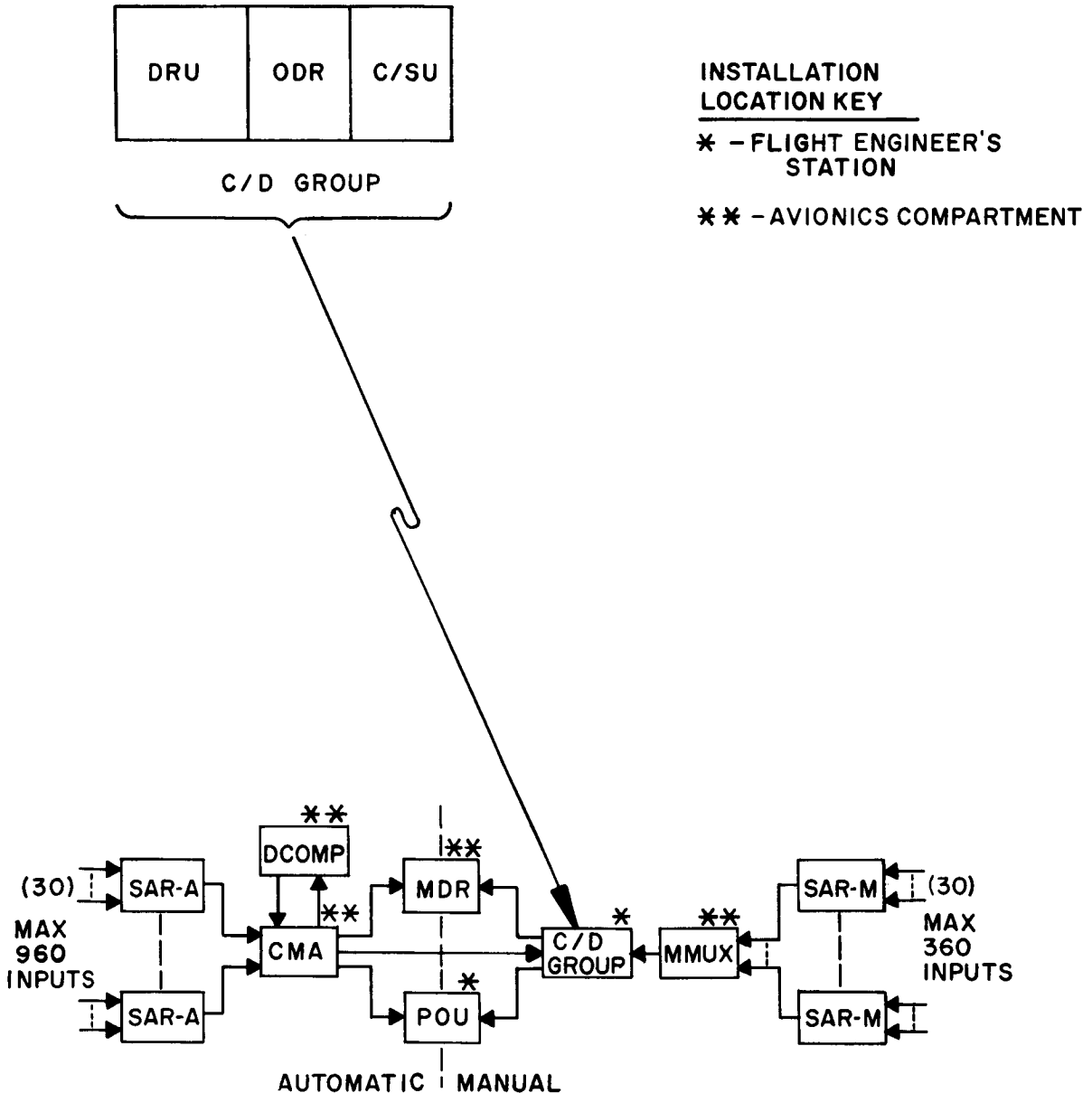


FIGURE 1 - MADAR BLOCK DIAGRAM

During manual mode operation, the automatic mode is uninterrupted. The operator uses the manual (or diagnostic) mode whenever he has any reason to suspect a malfunctioning monitored system. For example, a problem may be indicated in the heading loop. The operator will depress the Flight Control Subsystem entry button and film frame will be projected onto the view screen which will give instructions for initiating an analysis of the heading loop. Specific tests points will be called up by a film program for waveform display and comparison. As the live waveforms are projected, an analysis slide is simultaneously projected showing typical waveform patterns which apply specifically to the selected data point in the loop with instructions for the next test point. The operator can progress sequentially through the entire heading loop examining the loop for faulty signal data flow and locating any faulty LRU that may appear. The control panel also permits call-up of diagnostic procedures as well as waveforms and test point information.

## System Technical Description

The block diagram (Figure 1) indicates the components utilized individually and jointly by the automatic and manual modes. The Central Multiplexer Adapter (CMA) serves as the principal interface component.

Data flow into the system is as follows: The Signal Acquisition Remote (SAR) units monitor thirty test points each and are located throughout the aircraft at optimum locations for minimum wire lengths.

The analog signals entering the SAR are normalized and transmitted to either the CMA or the Manual Multiplexer (MMUX) depending upon signal type. Those signals entering the CMA are digitized and sent to the Digital Computer (DCOMP) for either malfunction analysis or trend analysis. Out-of-limit signals provided to the DCOMP cause the computer to transmit failed LRU information to the printout unit and maintenance data recorder. The MMUX signals are routed to the Control/Display (C/D) Group for display on the oscilloscope (ODR). Operator decisions that indicate malfunctions cause the Control/Display Group to transmit failed LRU information to the Printout Unit (POU) and Maintenance Data Recorder (MDR).

### Automatic Mode

The system can sequentially monitor up to 960 data points in the automatic mode. If a failure is detected, the LRU number, time of day, and date are automatically printed out in alphanumeric code, and also, recorded on the MDR. The subsystem name is illuminated on the Control/Display Subsystem status panel.

The heart of the automatic mode is the Digital Computer. The computer program selects the test point to be addressed. The SAR, upon receipt of test point address from the computer via the CMA, selects the proper data gate and routes the data signal from the conditioning circuitry through the selected gate to the data amplifier for transmission to the CMA. The signal is then converted to a digital word. For automatic mode signals called-up for display by the manual mode, the signal is obtained in the CMA prior to analog-to-digital conversion.

The digital word in the CMA is considered by the computer to be either trend data or LRU data depending upon the program. If the word is trend data, the computer tests the value to determine if it has changed more than its preprogrammed, allowable deviation since its last recorded value. If the value is within the allowable deviation, the data is discarded, or if desired, used for LRU information. If the value exceeds the allowable deviation, however, the test point number and a time reference is supplied to the MDR via the CMA.

LRU data values are tested to determine if they are within a programmed tolerance range. If they are within range, the test data is in a "Go" condition. A value outside this range results in a "No Go" condition. Upon obtaining this condition, all test data points related to the monitored LRU are examined and assigned a "Go" or "No Go" condition. Then a logic equation is formed with the "Go" and "No Go" determinations. This equation is then compared with a known equation of the LRU in a known condition. If the formed LRU equation compares, then the LRU is good. If the formed equation does not compare, then the LRU has failed. If the status of an LRU changes from the previous status, an output is required. An LRU status change would be defined as an LRU changing from normal operation to a failed condition or from a failed condition to normal operation. The information output goes to both the MDR and the POU units.

The arithmetic and branching capabilities of the computer make possible other means of examining data. A transfer function, for example, may be formed from two test points and used in both LRU status determination and trend analysis. The computer can also address the SAR calibrate signals when SAR calibration is required.

Manual Mode

In the manual mode, the MADAR system provides the operator with a visual waveform display for use in analyzing analog parameters which are not amenable to automatic techniques. Manual operation of the MADAR subsystem is also referred to as the diagnostic mode. Critical analog parameters are analyzed by comparison of simultaneously projected live waveforms and stored film patterns on a cathode ray oscilloscope and a view screen, respectively.

The manual mode uses the manual SAR's, a Manual Multiplexer Unit, a Control/Display Group, a human operator, and the two output devices - POU and MDR. Each manual SAR functions as two separate remote multiplexers having 15 channels each.

The manual mode operation is completely controlled by the Control/Display Group. The operator of the C/D Group is the comparator and the decision-making "component" of the manual mode. The operator is also the instigator of the sequence of operation.

LRU status determination by the manual mode is similar to the method used for the automatic mode. The operator selects the desired subsystem and begins the film preprogrammed interrogation of the subsystem test points. As each live waveform appears on the oscilloscope, a corresponding film frame is displayed onto the projection screen of the Data Retrieval Unit.

This film frame provides either information or waveforms to the operator in a manner that permits him a "choice" to continued operation. As choices are made, the operator progresses through the programmed diagnosis of the selected subsystem. When an operator's choice reveals an improper waveform, and subsequent faulty LRU, the actuation of the choice switch produces an output to the POU and MDR. The output will include LRU number, subsystem identification, and time.

Random access to any test point is obtained by entering the test point address through the C/D keyboard. The test data signal is thus displayed without regard to any diagnostic sub-routine. This operation would probably be performed more often by an experienced technician doing detailed trouble shooting.

There will be approximately 10,000 individual 16 mm film frames. An example of a typical film frame format is shown on Figure 2. The digital data appears on the left and the information to be displayed on the screen is shown on the right. A single light source is used for projection. The digital data, however, is split off and projected onto a photovoltaic cell bank for decoding. The various digital codes represent choices, addresses, and control settings for the oscilloscope and digital readout unit. The complete programming of the oscilloscope and digital readout unit by film frame data permits its utilization by persons not generally familiar with test equipment. The visual information data provides typical waveforms for operator comparison. Other film frames will provide diagnostic routines, detailed LRU information, and information pertinent to aircraft maintenance.

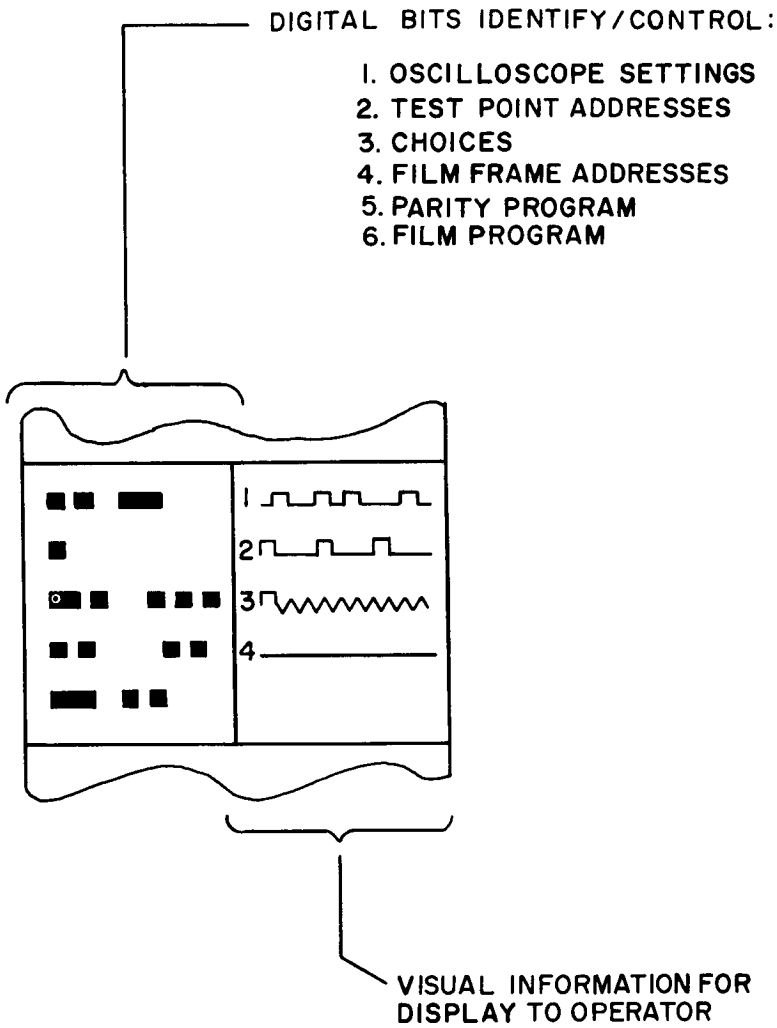


FIGURE 2-FILM FRAME FORMAT

## Component Descriptions

### Control/Display (C/D) Group

The C/D Group is located at the flight engineer's station and provides centralized control of the MADAR system. It is comprised of the Oscilloscope and the Digital Readout Unit (ODRU), the Data Retrieval Unit (DRU), and the Control and Sequencer Unit (C/SU). Their functions are shown below:

1. Control and Sequencer Unit

   - o contains the digital logic necessary to control the operation of the ODRU, DRU, and the MMUX.
   - o controls the routing of incoming and outgoing data to the C/D
   - o provides choice switches to operator
   - o permits immediate subsystem selection and status determination
   - o provides random addressing of test points

2. Oscilloscope and Digital Readout Unit

   - o is completely and automatically programmed from information contained on the film
   - o contains a dual trace, storage, CRT (daylight viewing)
   - o incorporates an AC-DC digital voltmeter with a bandwidth from DC to 2 Mc and capability for quadrature voltage measurement
   - o incorporates an oscilloscope which can be AC or DC coupled and has a bandwidth from DC to 10 Mc.
   - o provides trace storage capability

3. Data Retrieval Unit

   - o is capable of retrieving, in a random manner, any one of 10,000 frames of information
   - o contains a film strip memory with a storage capacity of approximately 1.5 million bits in addition to the 10,000 frames of data used for visual presentation.
   - o has an average access time of 4 seconds with a worst case access time of 14 seconds
   - o provides film updating by magazine replacement

Central Multiplexer Adapter (CMA)

The CMA is located in the avionics compartment and serves as the principal interface point for the automatic SAR's, digital computer, maintenance data recorder, printout unit, manual multiplexer, and C/D Group. In addition to the MADAR internal interfaces listed above, the CMA interfaces with the crash data position indicator recorder and the auxiliary navigation computer. The following are CMA functions:

- o route channel addresses to automatic SAR's
- o select proper SAR output
- o perform analog-to-digital conversion
- o provide power for automatic SAR's
- o provide automatic test point information to the manual multiplexer.

Signal Acquisition Remote Unit - Automatic (SAR-A)

The SAR-A units are optimally located throughout the aircraft to monitor, condition, and transmit signals for selected test data points to the CMA. Each SAR-A has the following characteristics:

- o 30 input channels
- o 2 internal calibrate channels
- o narrow band output
- o single amplifier
- o can provide nominal power for active signal conditioning
- o addressed by 5 bits.

Signal Acquisition Remote Unit - Manual (SAR-M)

The SAR-M units are also located throughout the aircraft to monitor, condition, and transmit test data signals to the manual multiplexer. Each SAR-M has the following characteristics:

- o 30 input channels (two 15-channel groups)
- o 2 calibrate channels
- o can provide 2 broad band simultaneous outputs
- o dual amplifiers
- o addressed by 8 bits (4 bits for each group of 15 channels)
- o can provide nominal power for active circuit signal conditioning.

Digital Computer (DCOMP)

The computer is a general purpose 8000 word memory device which accomplishes the following tasks:

- o performs data compression
- o addresses automatic test points as programmed
- o determines LRU/subsystem status
- o performs calibration
- o maintains real time clock.

Printout Unit (POU)

The POU is an output device that provides a hardcopy record of failed LRU numbers. POU characteristics are as follows:

- o provides alphanumeric printout of LRU number, aircraft identification, date, time, subsystem identification.

Maintenance Data Recorder (MDR)

The MDR is located in the avionics compartment and records digital information regarding trend and LRU failure data. The following are MDR characteristics:

- o incremental magnetic tape recorder
- o 2400 feet of tape (20 hours of recording at maximum step rate)
- o recording format compatible with ground processing equipment
- o tape loaded into readily-removable cassette
- o 556 characters per inch packing density.

Manual Multiplexer (MMUX)

The MMUX is located in the avionics compartment and serves as the interface between the manual SAR's and the C/D Group. Its functions are:

- o receives channel addresses from C/D and routes channel addresses to manual SAR's
- o selects desired SAR-M output(s)
- o provides power for manual SAR's

## Transducers

Existing transducers or sensors were used to extract LRU isolation signals wherever possible. Approximately fifty transducers, however, were required for specific interface with the MADAR system. To insure compatibility and compliance with the accuracy requirements, Lockheed prepared the procurement specifications and obtained these particular transducers. As discussed elsewhere, monitored system vendors were required to provide the necessary mounting provisions.

It was determined that five type transducers would be required. These transducers were basically off-the-shelf units which had previously been used for missile applications. Minor modifications were required for airframe usage and the resulting designs are substantially superior to ransducers normally found in aircraft. The majority of these are used for engine monitoring and must operate in an extremely high temperature environment. The types and characteristics are shown below:

1. Vibration - range: 0 to $\pm$25 g's;
   weight: 1.5 ounces
2. Position - ranges: 0 to 3" and 0 to 110°;
   weight: 9 ounces
3. Temperature - ranges: -100°F to 250°F and +100°F to +700°F;
   weight: 4 ounces
4. Pressure - range: 0 to 40 inch Hg;
   weight: 6 ounces
5. Differential Pressure - range: 0 to $\pm$60 psi;
   weight: 11 ounces

## System Software Requirements

Throughout the foregoing hardware description, there have been numerous references to computer routines, waveform patterns, and film programs. These references imply the necessity for a software development program that is equal to, or greater than, the hardware development program. Concurrent with the hardware development, an extensive effort was initiated to select test points and satisfy these software requirements.

## LRU Definition

To prepare a list of LRU's to be monitored and, subsequently, to select test points, it became necessary to investigate the aircraft systems in regard to anticipated troubleshooting hours. The Lockheed

Maintainability Department furnished LRU troubleshooting hours accumulated during 120,000 hours of C-130 aircraft operation. The units that rated high in troubleshooting hours were compared to LRU's on the C-5A. Since many C-5 units do not have a comparable unit in the C-130, there was not a one-to-one correspondence between the LRU's. A basis for LRU selection was established, however, and provided a general direction for LRU monitoring on the C-5.

Representatives from Maintainability, Reliability, Safety, and subsystem design organizations jointly prepared LRU stacking criteria. Each factor was weighed and a mathematical model was prepared to assist in LRU determination. These criteria items are as follows:

- mean elapsed time required for troubleshooting
- AGE elimination
- mean time between LRU corrective maintenance actions
- LRU or subsystem redundancy - degree of redundancy considering mission probability of success based on MTBF
- warning interval - interval between notice of impending failure and actual failure
- inherent LRU or subsystem testability
- failure effect - determined how LRU failure affects remainder of subsystem
- cost of monitoring

With the criteria established, each subsystem design group was charged with the final responsibility of selecting LRU's for monitoring.

LRU Summary

The resulting list of LRU's was assembled and yielded 3324 LRU's considered as "candidates" for monitoring. After an exhaustive investigation, those LRU's having relatively low failure rates or obvious failure indications were excluded from monitoring. The result was that 1804 or 56% of the candidates were selected to be monitored. These LRU's are monitored by the automatic mode, manual mode or a combination and represent the optimum selection from both an economical and technological viewpoint.

Each of the selected LRU's was assigned to one of ten major categories or subsystems depending upon its function. Correspondingly, each subsystem was associated with one of the status annunciator pushbuttons on the C/SU. By limiting the initial categorization to ten subsystems, two of the twelve status annunciator pushbuttons are retained as spares to provide growth potential. Table 3 identifies the ten major categories, the subsystems therein, the number of LRU's per subsystem, and provides a breakdown of monitoring arrangement.

Table 3 - LRU Summary

| Number | | Total LRU's | Not Monitored | Monitored Auto | Monitored Man. | Monitored Both |
|---|---|---|---|---|---|---|
| 01 | Propulsion (APU, Engines, Fuel Engine Inst.) | 883 | 395 | 106 | 376 | 6 |
| 02 | Environmental | 539 | 135 | 31 | 140 | 233 |
| 03 | Secondary Power (Hyd., Electrical) | 419 | 233 | 0 | 169 | 17 |
| 04 | Mechanical (LDG Gear, Doors & Ramps) | 581 | 324 | 5 | 251 | 1 |
| 05 | Flight Controls (Auto, Primary, Secondary) | 505 | 308 | 13 | 164 | 20 |
| 06 | Instrumentation | 49 | 0 | 0 | 49 | 0 |
| 07 | MADAR | 37 | 0 | 22 | 9 | 6 |
| 09 | Radar (Multi-Mode, Radar Altimeter) | 33 | 2 | 5 | 8 | 18 |
| 10 | Communications | 102 | 2 | 8 | 92 | 0 |
| 11 | Navigation (IDNE, SKE, LORAN) | 59 | 4 | 12 | 37 | 6 |
| | | 3207 | 1403 | 202 | 1295 | 307 |
| | | | | TOTAL 1804 | | |

(08 and 12 are spare)

## Failure Effects Analyses

Designers conducted thorough investigations of their respective subsystems to insure that adequate LRU isolation techniques were provided. The result of these investigations was the definition of the following:

- o test point selection and assignment for automatic or manual mode monitoring
- o logic diagrams for film program (manual mode)
- o computer routines (automatic mode)
- o determination of signal conditioning requirements
- o test procedures for "non-wired" LRU's

Test Point Selection The test point monitoring capability was initially sized at 1408 total inputs with an automatic-to-manual test point ratio of 2.77:1. This configuration was derived from the C-130 LRU study, previous experience with malfunction detection systems, and engineering judgement. Although the total number of points selected is 803 at the present time, the ratio of automatic-to-manual test points is 2.3:1.

It is anticipated that both the total number and ratio will change as the flight test program progresses.

Logic Diagrams Every investigation produced a logic diagram necessary for preparation of the film program. The diagram provided test point selection priorities for specific diagnostic routines, instrument indications, and step-by-step procedures for systematically isolating a fault. These diagrams are used principally by the manual mode but are also applicable to the automatic mode.

Computer Routines The test points selected for automatic mode monitoring require a computer program for selection sequence, frequency of sampling, and determination of Go-No Go limits.

Determination of Signal Conditioning Approximately 50% of the monitored test points do not require signal conditioning. The remaining signals, however, require circuits to be designed and verified. The large percent of preconditioned signals resulted from a MADAR interface requirement that was inserted in all procurement specifications for new hardware. The requirement insured access to test points suitable for LRU isolation and transducer-mounting provisions where a transducer was required.

Test Procedures For "Non-Wired" LRU's As mentioned earlier under malfunction detection techniques, many faulty LRU's can be isolated by clever manipulation of existing controls. The subsystem designers prepared straight-forward troubleshooting routines which followed logical sequences. The procedures were placed on the film and, therefore, became standard procedures which can be accomplished by persons having a wide variation of training/experience. Approximately 1800 LRU's are monitored and 50% of these do not electrically interface with the MADAR system. Operator detection of a failed unit by choice selection, however, does produce a record on both the POU and MDR.

## Design Verification

The design of the hardware and software would be for naught unless an extensive verification program was instituted to evaluate the performance of the entire MADAR system from signal conditioning input to film and computer programming. To accomplish this verification, it was decided to utilize the engine test stand and the various simulators being prepared for evaluation of the other C-5 systems. The following test stand and simulators are each scheduled to accomplish a portion of the over-all verification program:

- Avionics simulator
- Electrical simulator *
- Environmental simulator *
- Flight control, hydraulic, and landing gear simulator *
- Engine test stand

It was apparent that a permanent MADAR installation at each location would be idle a considerable amount of time. To alleviate this situation, a mobile MADAR subsystem (minus SAR's) was built to service the simulators marked by an asterisk. SAR units are permanently installed at each location since signal conditioning is unique for each channel. The MADAR subsystem itself will be evaluated on the avionics simulator.

### Individual Subsystem Verification

The verification of the test points originating within each subsystem will follow a five-phase program. These phases can be described as follows:

(1) Subsystem Normal, Without MADAR Interface This phase represents the normal functional testing that is performed on a subsystem when it is first received from a vendor. The signals that interface with MADAR are carefully investigated to confirm voltage levels, waveforms, etc. Although no actual interconnection is made with MADAR, this represents a significant portion of development since signal conditioning design can be initiated.

(2) Subsystem Normal, Single Channel Investigation When tentative conditioning circuitry has been fabricated, signals will be individually investigated while the monitored subsystem is operated in all normal modes and with induced malfunctions. The selection of malfunction to be induced is another fall-out from the subsystem fault analysis. This phase will also be used to obtain the waveform patterns for development of the film library.

(3) Subsystem Integrated With MADAR, Normal Operation This third phase represents the complete integration of the MADAR and monitored subsystem. The monitored subsystem will be operated in all normal modes. Both the automatic and manual modes of MADAR will be exercised to insure compatibility and observe normal operation.

(4) Subsystem Integrated With MADAR, Induced Subsystem Malfunctions The monitored subsystem will be operated with systematically controlled malfunctions induced (or simulated) to verify the failure isolation capabilities of MADAR. Waveform patterns will be confirmed and test routines (film and computer) will be firmly defined. Completion of this phase will permit finalization of production signal conditioning design. Modification of design resulting from flight testing is expected to be minor.

(5) Subsystem Integrated With MADAR (Long Term, Trend Data Collection) This final phase will be accomplished over an extended period of time to approximate actual operation. It is anticipated that this phase will be conducted on a time-available basis after the urgency of Phases 1-4 has been satisfied.

## Flight Tests

The ground verification program culminates with the preparation of the MADAR signal conditioning circuitry, film program, and computer routine for flight testing. The overall flight test plan for the C-5 is accelerated and flight testing of the aircraft systems is distributed over the first five aircraft. Testing of the basic MADAR system, as well as most avionics systems, is to be done on aircraft No. 4. As tests are conducted on each system, the MADAR interface and its operational capability will be evaluated. Early flight testing of the MADAR

system started in April 1968 on a C-141 flying test bed. MADAR monitoring was limited to the C-5 guidance system as installed in the test bed aircraft. The first C-5 flight test of MADAR was accomplished in January 1969. In addition to determining the LRU isolation capability of MADAR, the flight tests are expected to finalize the film program, computer routine and provide initial evaluation of operator usage and effectiveness.

## Typical Utilization

To illustrate some of the previously described design features, a hypothetical POU record is shown in Table 4 to acquaint the reader with the system capabilities. The numbers down the right margin refer to the interpretations shown in Table 5. It should be noted that a significant portion of the computer memory is accessible by keyboard entry to permit limited program alteration and to provide a means to input data for storage. Operator instructions are detailed on the various film frames of the subsystem being investigated. These film frames are accessible to the operator by depressing the appropriate status annunciator pushbutton and selecting the desired operation from the index appearing on the DRU screen.

Table 4 - Hypothetical Mission Printout Unit Output

| Message | Table 5 Reference No. |
|---|---|
| MADAR FLIGHT PROGRAM | 1 |
| 0930 A/C68222 30/09/70 STBY | 2 |
| 1234567890#':>√=/STUVWXYZ≥ ,("\% | 3 |
| -JKLMNOPQ! *] :≤ =ABCDEFGHI?.)[< | 4 |
| 0933 9007029 CSU INPUT | 5 |
| MADAR FLIGHT PROGRAM | 6 |
| 0934 A/C68222 30/09/70 STBY | 7 |
| 0935 9207029 CSU INPUT | 8 |
| 0938 5011070 CSU INPUT | 9 |
| 0938 A/C68222 01/10/70 STBY | 10 |
| 0939 6102700 CSU INPUT | 11 |
| 0939 1027 CLOCK UPDATE | 12 |
| 1030 1000100 CSU INPUT | 13 |
| 3452 3250 3109 3204 TEMP TIME | 14 |
| 0642 0748 0796 0748 HOURS | 15 |
| 0089 0053 0056 0057 SERIAL | 16 |
| 0638 AIRFRAME HOURS | 17 |

Table 4 - Hypothetical Mission Printout Unit Output (Cont)

| Message | Table 5 Reference No. |
|---|---|
| 1033 1200644 CSU INPUT | 18 |
| 1035 1010100 CSU INPUT | 19 |
| 1035 CSU ERROR | 20 |
| 1035 1001100 CSU INPUT | 21 |
| 1035 A/C68222 01/10/70 OPER | 22 |
| 1035 F07034 S00002 | 23 |
| 1036 7007034 CSU INPUT | 24 |
| 1037 R07034 S00000 | 25 |
| 1040 A/C68222 01/10/70 OPER | 26 |
| 1045 A/C68222 01/10/70 OPER | 27 |
| 1047 TAKEOFF | 28 |
| 1050 A/C68222 01/10/70 OPER | 29 |
| 1055 A/C68222 01/10/70 OPER | 30 |
| 1100 A/C68222 01/10/70 OPER | 31 |
| 1105 A/C68222 01/10/70 OPER | 32 |
| 1110 A/C68222 01/10/70 OPER | 33 |
| 1115 A/C68222 01/10/70 OPER | 34 |
| FRAME ADDRESS 131 | 35 |
| 1118 200001 CSU INPUT | 36 |
| 1118 1 ENGINE HEALTH | 37 |
| N2 BASED ERRORS | 38 |
| +.00350 = DN1 | 39 |
| +.00125 = DTIT | 40 |
| -.00008 = DFF | 41 |
| +.00467 = DN2 | 42 |
| IN LIMITS | 43 |
| 1119 2000002 CSU INPUT | 44 |
| 1119 2 ENGINE HEALTH | 45 |
| N2 BASED ERRORS | 46 |
| -.00400 = DN1 | 47 |
| +.00333 = DTIT | 48 |
| +.00242 = DFF | 49 |
| -.00471 = DN2 | 50 |
| IN LIMITS | 51 |
| 1120 A/C68222 01/10/70 OPER | 52 |

Table 4 - Hypothetical Mission Printout Unit Output (Cont)

| Message | Table 5 Reference No. |
|---|---|
| 1120 2000003 CSU INPUT | 53 |
| 1120 3 ENGINE HEALTH | 54 |
| N2 BASED ERRORS | 55 |
| +.00087 = DN1 | 56 |
| +.00456 = DTIT | 57 |
| -.00316 = DFF | 58 |
| -.00058 = DN2 | 59 |
| IN LIMITS | 60 |
| 1121 2000004 CSU INPUT | 61 |
| 1121 4 ENGINE HEALTH | 62 |
| N2 BASED ERRORS | 63 |
| -.00141 = DN1 | 64 |
| -.00286 = DTIT | 65 |
| -.00314 = DFF | 66 |
| +.00066 = DN2 | 67 |
| IN LIMITS | 68 |
| 1122 F09047 S00041 | 69 |
| 1123 7009047 CSU INPUT | 70 |
| 1124 9009039 CSU INPUT | 71 |
| 1124 7009039 CSU INPUT | 72 |
| 1125 A/C68222 01/10/70 OPER | 73 |
| MADAR FLIGHT PROGRAM | 74 |
| 1128 A/C68222 01/10/70 OPER | 75 |
| 1129 9003008 CSU INPUT | 76 |
| 1130 A/C68222 01/10/70 OPER | 77 |
| 1130 7003008 CSU INPUT | 78 |
| 1132 F05006 S00001 | 79 |
| 1133 7005006 CSU INPUT | 80 |
| 1135 A/C68222 01/10/70 OPER | 81 |
| 1140 A/C68222 01/10/70 OPER | 82 |
| 1145 A/C68222 01/10/70 OPER | 83 |
| 1147 F10075 S00003 | 84 |
| 1147 R10075 S00000 | 85 |
| 1147 F10075 S00003 | 86 |
| 1147 R10075 S00000 | 87 |
| 1147 F10075 S00003 | 88 |

Table 4 - Hypothetical Mission Printout Unit Output (Cont)

| Message | Table 5 Reference No. |
|---|---|
| 1147 R10075 S00000 | 89 |
| 1147 F10075 S00003 | 90 |
| 1147 9410075 CSU INPUT | 91 |
| 1149 7010075 CSU INPUT | 92 |
| 1150 A/C68222 01/10/70 OPER | 93 |
| 1155 A/C68222 01/10/70 OPER | 94 |
| 1157 LANDING | 95 |
| 1157 TAKEOFF | 96 |
| 1157 LANDING | 97 |
| 1159 4000000 CSU INPUT | 98 |
| 1159 LRU STATUS | 99 |
| 05006 | 100 |
| 09047 | 101 |
| 10075 | 102 |
| 09039 | 103 |
| 03008 | 104 |
| ZZZ | 105 |
| 1200 A/C68222 01/10/70 OPER | 106 |
| 1200 1000010 CSU INPUT | 107 |
| 1200 LRUS SUPPRESSED | 108 |
| 10075 | 109 |
| ZZZ | 110 |
| 1201 1000001 CSU INPUT | 111 |
| 1201 END FILE | 112 |
| 1202 1010000 CSU INPUT | 113 |
| 1202 A/C68222 01/10/70 STBY | 114 |

Table 5 - Interpretation of POU Record

| Table 4 Line(s) | Message Interpretation |
|---|---|
| 1-2 | This header message is automatically printed when the MADAR subsystem is energized. The time, aircraft identification number, and date reflect the last values stored in the DCOMP. The "STBY" indicates the automatic mode is in stand-by condition. |

Table 5 - Interpretation of POU Record (Cont)

| Table 4 Line(s) | Message Interpretation |
|---|---|
| 3-4 | As part of the routine MADAR preflight check, the operator depresses the POU BITE pushbutton and receives a printout of all characters. |
| 5 | This simulates a fault found within the DRU. The operator enters (via the keyboard) the faulty LRU number for record and de-energizes the MADAR system to permit replacement. The number 9007029 is composed of: "90" - tag numbers to report an LRU failure, "07" - MADAR system identification number (See Table 3), and "029" - identification number of faulty LRU. |
| 6-7 | MADAR is again energized (See 1-2). |
| 8 | The replacement message for DRU is entered by keyset. The number "92" is used to report LRU replacement. |
| 9-10 | Line 9 is a keyset request to correct the date. The number "5" is a command to initialize date whereas "011070" is the desired date. Line 10 is the header reprinted as an acknowledgement. |
| 11-12 | Line 11 is a keyset request to correct the time. The number "6" is a command to initialize time whereas "1027" is the desired time. Line 12 is command acknowledgement. |
| 13-17 | Line 13 is a keyset request for printout of the DCOMP-stored MDR header message. The number "1000100" is the command entry code for this output. Lines 14-17 represent this header message. Lines 14-16 pertain to engine information and line 17 provides airframe information. |
| 18 | This line represents a change to the airframe hours as requested by the operator. The number "12" is the airframe hours update command whereas the "00644" is the desired correction. |
| 19-20 | This is a command request to place the automatic mode in "operate" by choice selection but is transmitted incorrectly. Line 20 indicates error. |
| 21-22 | The mode change request is made again and acknowledged by a new header. |
| 23 | An MDR failure is detected automatically. The letter "F" indicates a failure, "07" refers to the subsystem (MADAR) number, and "034" is the LRU number. The series "S00002" is the status code. The status code provides supplemental |

Table 5 - Interpretation of POU Record (Cont)

| Table 4 Line(s) | Message Interpretation |
|---|---|
| 23 (Cont) | data concerning the mode of LRU failure for most automatically monitored units. The field consists of five octal numbers, showing relationship of test point signal to test limit for each test made on a LRU. Up to fifteen such tests may be reported for a given LRU. When no test is made, or the signal value is below test limit, a 'zero' is placed at the right of the field; a 'one' is placed at the right for a signal value above test limit. For the next individual test, the binary field is shifted left, and a 'zero' or 'one' entry is made for the next test. At the end of the LRU test, the complete status code is available for printout in octal format (00000 to 77777) and, with supplementary information as to significance of each test, may be used in ground maintenance for troubleshooting a removed unit. |
| 24 | The operator looks up the LRU number in film and acknowledges. The number "70" is the command entry for acknowledgement of LRU failure. |
| 25 | Indicated failure was corrected by closing the MDR cover and a recovery message is printed. The letter "R" signifies recovery. |
| 26-27 | Periodic (every five minutes) time mark-header messages printout automatically. This periodic printout insures that the tape (regardless of length) can be related to a specific aircraft in the event that several aircraft tapes are returned to a central maintenance facility. |
| 28 | Takeoff event message. |
| 29-34 | Periodic time mark-header messages indicate no operator or test events on MADAR. (See 26-27) |
| 35 | Thirty minutes after takeoff, a reminder to test engine health is printed, referring to a specific film pac frame address for further instructions. |
| 36-43 | The flight engineer shuts off bleed air on the affected engine, transmits a request to test by CHOICE, and reads a diagnosis. The number "20000" is the command request and the "1" selects engine number 1. |
| 44-51 | Similar operations and responses are obtained for engine #2. |
| 52 | Periodic time mark - header message. (See 26-27) |

Table 5 - Interpretation of POU Record (Cont)

| Table 4 Line(s) | Message Interpretation |
|---|---|
| 53-60 | Similar operations and responses are obtained for engine #3, with one time mark header occurring in the operation. |
| 61-68 | Similar operations and responses are obtained for engine #4, concluding the engine health test. |
| 69 | A multimode radar failure report is obtained from the automatic subsystem. The number "09" is the multimode subsystem number (See Table 3). |
| 70-72 | Using the DRU, the failure report is acknowledged, the specific LRU fault is diagnosed by film pac and transmitted by CHOICE, and the specific fault is acknowledged by another CHOICE message. |
| 73-75 | A periodic time mark-header message (See 26-27) is followed by a power transient, resulting in a new header. |
| 76 | An electrical subsystem failure is detected by film pac aided diagnosis. |
| 77 | Periodic time mark-header message. |
| 78 | Acknowledgement for electrical subsystem failure is obtained from film frame choice. The number "03" is the secondary power subsystem number (See Table 3) which includes the electrical subsystem. |
| 79-80 | An automatic flight control failure is detected by the automatic subsystem, looked up in DRU film pac, and acknowledged. |
| 81-83 | Periodic time mark-header messages indicate no operator or test inputs on MADAR. |
| 84-90 | A series of failures and recoveries is detected automatically in the communications subsystem. |
| 91-92 | The LRU that failed and recovered in lines 84-90 apparently has an intermittent problem. To prevent this from cluttering the POU record for the duration of the flight, the flight engineer suppresses additional nusiance printouts by the "94" entry command followed by the subsystem number (10) and the LRU number (075). |
| 93-94 | Periodic time mark-header messages. |
| 95-97 | A rough landing with one big bounce is recorded. |

Table 5 - Interpretation of POU Record (Cont)

| Table 4 Line(s) | Message Interpretation |
|---|---|
| 98 | After landing, the flight engineer requests a summary of the LRU's that have failed during flight. This is accomplished by depressing the "LRU STATUS CHECK" pushbutton on the C/SU. At this time, the flight engineer would probably depress the MADAR status annunciator pushbutton and call-up the MADAR shut-down procedure. |
| 99-105 | The numbers of the failed LRU's are printed. The letters "ZZZ" indicate end of summary. |
| 106 | Periodic time mark-header message. |
| 107 | The flight engineer requests a listing of those LRU's that have been suppressed (See 91-92). The command entry "1000010" is keyset into the C/SU to accomplish this task. Instructions appear in the shut-down procedure. |
| 108-110 | The suppressed LRU number is printed together with "ZZZ" to indicate end of list. |
| 111-112 | The flight engineer requests that a tape mark be placed on the magnetic tape record. This is accomplished by a choice selection and is acknowledged by the printout "END FILE". |
| 113-114 | The flight engineer requests by choice selection that the mode be changed from operate condition to standby condition in preparation for next flight (See 1-2). This request is acknowledged by the header message appearing on line 114. |

## Conclusion

This paper has described the hardware design of the Lockheed airborne manual/automatic malfunction detection system called MADAR and the adaptation program now in process to apply it to our C-5 aircraft. The laboratory development tests are progressing well and nearly 200 flight test hours were accomplished while monitoring selected C-5 avionics systems on a C-141 test bed aircraft. Approximately fifty hours of tests on the C-5 have been completed.

MADAR is functional on the engine test stand and has detected a failure 30 minutes before it became apparent to the test crew.

The results to date indicate that the MADAR hardware design is sound and its application to the C-5 is reasonable. There is every expectation that MADAR will fully perform its intended function and substantially contribute to a new standard of maintainability and operational availability of the C-5 aircraft.

# TECHNOLOGY PROBLEMS ASSOCIATED WITH LARGE SOLAR ARRAYS FOR LONG DURATION SPACE MISSIONS

by

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## Abstract

The use of solar arrays as an electric power source for Earth orbital applications and planetary exploration has become routine. Previously considered only for power levels up to 1 $KW_e$, spacecraft are currently being developed and future missions are being planned which would utilize solar array power systems up to 50 $KW_e$ and beyond. In addition, as these missions are identified, the program objectives, modes of operation, and scientific payloads constantly approach a much higher level of sophistication. Inevitably this leads to an assessment of the cost effectiveness of the approach. Invariably, it is found that long duration operation is desired. In interplanetary travel it is axiomatic that so-called minimum energy trajectories take longer. Similarly, in-Earth-orbit missions will ultimately result in more complex activities and payloads, necessitating extended periods of operation. Finally, if the benefits to be derived from the initial Apollo lunar landings are to be economically exploited, it will eventually become mandatory to develop lunar bases capable of long duration operation.

Despite the comparatively advanced state-of-development of solar arrays, many technological problems remain to be solved if mission durations of five years and greater are to be achieved. This paper identifies several significant problem areas related to the design of cellstacks for long duration operation. These include solar cell selection criteria, improved solar cell performance, and reduced degradation from charged particle radiation. This paper discusses the various approaches being taken by both Government and industry. These include improved materials, manufacturing techniques, and thermal control techniques. The emphasis in all these areas is on reduced weight, cost, and stowage volume, as well as increased reliability and extended operating life.

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## Introduction

In assessing the various factors affecting the use of large solar arrays for various long duration space applications one is initially confronted with an overwhelming set of parameters. This evaluation is further compounded by the various technical disciplines involved. Finally, the various performance, design, and operational criteria are so intimately interdependent, that a detailed systems engineering approach is required to arrive at the most system effective solution. Therefore, perhaps the major technological problem confronting both Government and industry today, is to develop an all encompassing, standardized, methodology for solar array design. The need for the generation of such an analytical model is long overdue.

Therefore, this paper, in addressing itself to the technological problems associated with long duration solar array operations in space, reflects only the limited perspectives existing today. As such, its main emphasis is upon identifying the challenges confronting the solar array designer rather than providing specific solutions. However, the current technical approaches being taken are outlined and their effectiveness in enhancing the technology readiness of solar array systems is discussed.

## Design Constraints

The major factors affecting solar array performance and design are the environmental criteria (reference 1) and the operational modes. For long duration space operations, the impact of these conditions is to reduce solar array end-of-life (EOL) performance, impose severe material selection requirements, and introduce increased system complexity. If the solar array power level is high, i.e. up to 50 $KW_e$, additional problems associated with stowage, deployment, and spacecraft interactions also can occur. It is important, therefore to recognize the unique mission dependency of each solar array design. However, if one limits oneself to the basic elements comprising a solar array system, namely; the cellstack, substrate, and power distribution system, then many regions of technological commonality can be identified.

The discussion in this paper is constrained to a review of only those technology problems related to the performance and design of the cellstack, substrate, and related power distribution elements.

## Solar Array Performance Considerations

There are various factors which affect solar array performance. These can generally be classified into two groups. These are:

- time independent factors
- time dependent factors

The first group are independent of mission duration. They are primarily a function of the basic cellstack electrical characteristics, array design and manufacturing techniques, and operational criteria. They are initially determined by experimental evaluation and an analytical assessment of the impact of the spacecraft operating constraints. The latter requires a detailed thermal analysis to establish actual solar cell conversion efficiencies, an assessment of array orientation accuracy, and shadowing losses. The resulting array performance determined from applying these factors, is usually referred to as the beginning-of-life (BOL) power output.

The second group are intimately related to the space environment and the elapsed mission time. In order to assess the impact of these factors on solar cell performance degradation, a priori knowledge of this space environment is required.

All these factors must also be evaluated. Taken together with the time-independent factors, they establish the achieveable EOL power-to-area ratio. The relationship used to determine the power-to-area ratio from these factors can be simply expressed as:

$$\frac{P_o}{A} = \left\{\eta_p \ \eta_c \ \eta_m \ \eta_o \ \eta_{REG}\right\} \left\{\eta_{tc} \ \eta_{RAD} \ \eta_{uv} \ \eta_{mm}\right\} \left|\eta_{CONV.}\right|_{f(T)} \left|I_{SOL}\right|_{f(AU)} \quad (1)$$

These factors are defined in Figure 1. The solar intensity, $I_{SOL}$ will vary inversely as the square of the distance from the sun. At 1 AU and AMO, its value is estimated at 139.6 mw/cm$^2$. Typical electrical performance characteristics for some commonly used N+ on P silicon solar cells at 28°C is shown in Figure 2. It should be noted that higher voltages and maximum optimum power are obtained with the lower base resistivity (2 ohm-cm.) cells. However, for both the 2 ohm-cm. and 10 ohm-cm. cells, as thickness of the cell is increased, the output of the cells increase. Thus for a given base resistivity cell, the greater the thickness, the higher the cell conversion efficiency ($\eta_{CONV}$). This can be seen from the data plotted in Figure 3. However, since it is a high power-to-weight ratio that is desired, rather than high conversion efficiency per se, the thinner silicon cell is to be preferred. This is also shown in figure 3, where for a constant cover glass thickness of 6 MILS, a 2 ohm-cm., 10 MIL silicon cell with an efficiency of 11.1% has a power-to-weight ratio of 200 W/Kg (∿75 W/lb). A 6 MIL cell with similar characteristics and an efficiency of 10.1%, however, has a power-to-weight ratio of 275 W/Kg (∿100 W/lb). Furthermore, if CdS thin films are considered, even higher specific powers (up to 375 W/Kg) are projected for conversion efficiencies as low as 3.5%. In both instances, however, technology problems may preclude the use of these higher power-to-weight ratio cells.

For example, the cost of the 2 ohm-cm, 4 MIL silicon cells is considerably higher than that for 10 ohm-cm, 8 MIL cells. If a conservative price differential of $4/cell is assumed and recognizing that approximately 15000 to 20,000 cells/KW are required depending upon the cell efficiency, a cost penalty of $60,000 to $80,000 per kilowatt results. Decreased handling durability and lower availability further mitigate against use of the thinner cells at this time.

The current status of cadmium sulfide (CdS) thin film cells is reported in Reference 2. Despite intense development efforts over the last few years, the CdS thin film cell is still plagued by many development problems. These include low performance due to inability to control reproducibility techniques, instabilities due to thermal cycling, and degradation in performance due to humidity effects, and thermal vacuum storage at 100°C. Until these foregoing technology problems have been resolved, the CdS thin film cells cannot be recommended for use at this time. However, the long range potential for reduced cost and high power density justify sustained development effort in this area.

The impact of temperature on cell electrical performance, is another major factor affecting the solar array power-to-area ratio. As the cell equilibrium temperature increases from a nominal 28°C to values approaching 60°C for earth orbiting applications, (Figure 4) and 95°C for lunar surface applications (Figure 5), a marked decrease in cell conversion efficiency ($\eta_{CONV}$), occurs (Figure 6). The net effect is that the array must be sized based upon the reduced power-to-area ratio resulting from the peak array equilibrium temperature. In some instances, time oriented load profile matching can be utilized to minimize the impact of this performance degradation factor on the array design. In addition, because of albedo effects, solar arrays for low altitude earth orbital missions and lunar surface applications result in higher equilibrium temperatures than for synchronous orbit and away from the sun (greater than 1AU) interplanetary flights.

For long duration operation the deliterious effects of the space radiation environment presents an even greater technology problem than that associated with temperature. Despite the fact that array performance degradation at BOL is significantly affected by array equilibrium temperature, the magnitude of this effect can be fairly accurately predicted. In addition, experimental confirmation of this effect can be fairly easily obtained in the laboratory. The space radiation environment, on the other hand, represents a region of high uncertainty. The radiation environment encountered in space consists of charged particle radiation and the solar illumination intensity. The charged particle radiation is composed of galatic cosmic rays, solar-protons and electrons. While the galactic radiation maintains a relatively constant level with time, the charged particle radiation rises occasionaly by several orders of magnitude during solar flare disturbances. Normally, this solar flare acti vity is directly related to sunspot quantity. Figure 7 depicts the number of sunspots observed or predicted for three 11-year solar cycles. It should be noted that for missions occuring in the 1971 to 1976 time period

a minimum of solar flare activity is anticipated. Missions conducted during the 1976 to 1981 time period may encounter maximum activity. To assess the impact of charged particle degradation on solar array performance the integrated flux must be determined on a statistical basis from space environmental data of the type available in Reference 1.

The solar spectrum also varies and is considered to follow that of Johnson for AMO and 1.0AU (Reference 1). During periods of high solar activity, the x-ray flux may increase by 1 or 2 orders of magnitude. Generally, its value is in the $10^{-8}$ to $10^{-11}$ w/cm$^2$ range in the region from 100Å to 2Å respectively. This together with the ultra-violet portion of the solar spectrum will contribute to solar array performance degradation.

The meteroid flux and the resultant impact on solar array performance is even more difficult to predict than charged particle radiation effects. The flux can vary widely and erratically as a function of time due to sporadic showers, and its particle sizes, distribution, and velocities can only be evaluated on a statistical basis at best. In addition, secondary ejecta caused by micrometeoroid impacts can result in additional array performance degradation. Also, the actual effect of a strike on a cell or a string of cells is subject to wide variations. Fortunately, experience to date with orbital, interplanetary, and lunar surface spacecraft has not substantiated the degradation values predicted by analytical estimates. However, as solar array areas increase for higher power requirements and mission durations are extended up to ten years, the probability of greater potential damage will be ever present.

Despite the uncertainties associated with the evaluation of the various degradation factors outlined above, the feasibility of using large solar arrays for long duration missions is not in question. Conservative assumptions and judiciously selected component trade-offs can assure a high probability of meeting solar array EOL requirements. Rather, the challenge lies in identifying problems and providing solutions to those technology areas which would result in major reductions in array weight and cost, and in increased reliability.

## Solar Array Cellstack Considerations

The theory of the photovoltaic energy conversion process has received increasing attention during the last two decades. The highly successful application of solar arrays for spacecraft power systems has been one of the major factors in accelerating this interest. A fairly recent review of the various theories devised to analyze this process and a discussion of the experimental efforts engaged in to confirm these analyses is given in Reference 3. A short summary of the state-of-the-art of various solar cells is given in References 2, 4, and 5.

The solar array cellstack consists of various components and materials needed to generate the electric power required and to assure

satisfactory operation in the hostile space environment. Typical silicon cell and CdS thin film solar cellstacks are shown in Figure 8 and 9 respectively. In selecting a particular cellstack for a given mission environment and life, the following key elements must be considered:

- Solar cell type, thickness, and base resistivity
- Cover glass material and thickness
- Cover glass to cell adhesive
- Cell interconnect design, material, and thickness
- Solar cell to substrate adhesive
- Main power cabling configuration and material

The critical selection criteria and tradeoff rationale for specifying the requirements for these cellstack elements is fairly well established. However, as array power levels are increased and mission durations are extended, optimization of these elements from a weight, cost, life and reliability standpoint becomes more complex. A detailed discussion of this optimization process is beyond the scope of this paper. However, as a few of the more significant technology problems confronting solar array designers are discussed, their relationship to the tradeoff process will be outlined.

Solar Cell Performance Related Criteria

For large area arrays, the cadmium sulfide (CdS) solar cell (Figure 9) has the potential for reducing array weight and cost. It also shows promise for increasing array stowage efficiency and reducing electrical degradation caused by charged particle radiation (Figure 10). However, the relatively low cell conversion efficiency (3 to 5%), present operational stability problems in the space environment, and other factors outlined in Reference 2, preclude the selection of these cells for near-term missions (up to 1975).

The 2 cm. x 2 cm., N-on-P, single crystal silicon cell is currently the preferred type. Production quantities are also available in larger sizes (3 x 3 and 2 x 6 cm.). Cost per unit area generally decreases with cell area. Limited operational experience, possible increased handling costs due to breakage, and thermal differential expansion difficulties due to increased length have limited the use of these larger cells. However, as solar array areas approach 5000 ft.$^2$ and larger, the use of the 2 x 6 cm. cell will become increasingly more desirable. Therefore, it behooves both Government and industry to resolve the current technological problems limiting their use at this time. This includes reduced production and handling costs, and improved adhesives, interconnects, and substrate materials to minimize temperature induced design problems and to increase array reliability.

Silicon cell base-resistivity can usually be optimized by selecting for a specific mission, the value that results in the higher EOL power. The standard 10 ohm-cm. cell has a somewhat higher charged particle degradation resistance than do 2 ohm-cm. cells. The latter, however, provide the higher output power initially. For long duration missions, depending upon the total integrated flux, the initial higher output of the 2 ohm-cm. cells may be outweighed by the higher radiation resistance of the 10 ohm-cm. cell. However, on missions where the total equivalent 1 Mev e/cm$^2$ fluence levels are estimated to be low ($<10^{15}$ e/cm$^2$) the 2 ohm-cm. cell may still be preferred (Figure 11).

There are other parameters which must also be taken into account before a final solar cell selection is made with respect to charged particle radiation degradation. These include cell thickness and cover slide thickness. For example, at BOL and to fluence levels of $10^{14}$ e/cm$^2$ (1 MEV equivalent), cell output power increases as cell thickness is increased (Figures 11, 12, and 13), up to about 0.016 inch. This is due to a characteristic of the photovoltaic energy conversion process. In the thicker material, the electron-hole pairs generated in the bulk material away from the junction, have a greater diffusion length to allow the minority carrier to reach the junction. This results in a higher power output. As the level of the charged particle irradiation is increased, the effective diffusion length decreases. The net result is that electron-hole pairs generated far from the junction will generally recombine before the minority carrier reaches the junction. Consequently, thin cells which initially have a lower output than thicker ones, eventually have the same output as can be seen on Figure 14.

The impact of increased cover glass thickness is to reduce the effective radiation level on the cell. This is shown in Figure 15 for both solar flare proton and trapped electron irradiation. Data presented in Reference 6 on the ATS-1 at AMO seems to indicate that beyond a thickness of 0.006 inch, the benefits achieved in reduced solar cell radiation degradation may be offset by other performance degrading factors (Figures 16 and 17) such as reduced illumination. This has not been borne out by previous experiments (Reference 7 and 8) conducted at AMI as shown on Figure 18. Thus, at this time, it cannot be stated with certainty that EOL power output from each cell will increase with increasing cover glass thickness.

From the foregoing, it is evident that the selection of the optimum cell and cover glass thicknesses is not straight forward. They are highly mission dependent and quite sensitive to the dictates of the criteria used to arrive at the figure of merit for the overall solar array. Such factors as power density, cost, weight, volume, and reliability also enter heavily into the decision making process.

## Solar Cell Material Considerations

In addition to establishing the criteria for the desired solar cell performance characteristics, it is equally important to select the proper materials for the elements comprising the cellstack. Not only must they be compatible with the imposed environments, but they must also demonstrate long life and high reliability. Some of the more significant considerations are as follows.

### Cell Contacts

Recent investigations (Reference 9) have found that solderless cells having titanium-silver (Ti Ag) contacts shown severe degradation when subjected to a combination of high temperature and humidity. They have also indicated that completely solder covered cells exhibit insignificant degradation as a result of such exposure. Conventional solder covered cells are obtained by dipping the cells into solder. This results in unnecessarily heavy cells with a solder thickness of 39μm(0.0015 inch). The so-called "dip and sling" method has been developed to overcome this problem. Increased cell wetting (up to 96%) and reduced solder thicknesses results. Another approach, developed by Helitek employs a tin coating (95% Sn/5% Ag) of approximately 2μm thickness on the back surface and 8μm on the front contact areas. The weight penalty in this case was only 4% of that for conventional solder dipped cells but the temperature/humidity degradation resistance was equivalent. These techniques and other newly developed proprietary processes have eliminated this problem without incurring the earlier weight penalties.

### Cover Slides

Cover slides perform three functions on solar array cellstacks:

- Shield cell from excessive charged particle degradation
- Protect cell from micrometeorite damage
- Reduce cell maximum operating temperature

The basic material selection criteria for cover slides are:

- Must not darken substantially when subjected to the space radiation environment
- Should possess a high infared emmittance
- Reasonable cost

There are two materials which will meet these requirements. They are Dow Corning Microsheet No. 0211 and Fused Silica No. 7940. The advantage of the former over the latter is its lower cost. The disadvantage is its greater transmittance loss under x-ray exposure. Therefore, fused silica, due to its long term stability in the space environment is recommended for long duration missions. There still is a

potential technology problem associated with their use. Cover slides are normally coated with interference filters. The outboard side may contain an anti-reflective filter of $MgF_2$. Laboratory tests have shown that $MgF_2$ will deteriorate under low energy proton bombardments causing substantial transmittance losses. The mechanism causing the deterioration is presently not fully understood. However, data from spacecraft in orbits with relatively high low-energy proton fluences have not yet confirmed the laboratory results. Hence, this phenomenon should be investigated further to determine its possible impact on future long duration missions.

The reflective filter is normally located on the cell side of the cover slide. It provides protection for the cell-to-glass adhesive. Two basic types of ultraviolet reflective filters are available; blue and blue-red. Blue-red filters can reduce the array temperature by an estimated 5°C. However, due to a relatively lower transmission efficiency, an overall reduction in power output of approximately 3% results. Due to this factor and their higher cost, the blue filter is normally utilized. Their main function is to prevent severe ultraviolet degradation of the cell-to-glass adhesive. A filter cutoff wavelength of 0.410 micron results in a 3% greater power output than one with a 0.435 micron cutoff. Since the effect of the lower cutoff filter on adhesive degradation is considered negligible, a blue reflective filter with a cutoff wavelength of 0.410 micron is usually recommended. Continued research in this area is recommended, however, since considerable performance gains may be achieved by improvements with this component. Also, additional experimental data is desirable to assure satisfactory operation for longer durations (up to 10 years).

## Glass-to-Cell Adhesives

Adhesives for bonding cover slides to solar cells are predominately organic, high polymeric materials. The basic polymer resins are modified by the control of the molecular weight, degree of cross-linking, and the incorporation of additives to improve specific properties. Changes in the chemical balance of the adhesive can lead to significant modifications to its physical properties. In the case of optical adhesives these can result in adverse effects to the adhesive refractive index or spectral transmittance, thus lowering the solar cell performance.

Adhesives of principal interest are epoxies and silicones. They have demonstrated superior ability to withstand ionizing radiation for long periods without any significant change to structural properties or optical transmittance. The most commonly used are the silicones, with GE/RTV602 and Dow Corning Sylgard 182 or XRG-3489 preferred.

The photochemical decomposition of these adhesives result in degradation of their optical properties, such as adhesive darkening and increased solar absorptance. Because the recommended silicones are

capable of distributing absorbed ultraviolet photon energy along its polymer chain, the degradation of optical properties are minimized. Laboratory tests have shown that Dow Corning's XRG-3489 (a highly purified version of Sylgard 182), has the least coloration due to ultraviolet and charged particle irradiation. Data from satellite experiments have confirmed these results for time-integrated fluxes corresponding to 3 to 5 years at synchronous orbits. Thus, DC/XRG-3489 is currently considered the best selection for long duration missions.

## Cell-to-Cell Interconnect Design

Minimum cost consistent with reliable performance dictates the choice of solar cell material. Candidate materials include copper, silver, kovar, and molybdenum. The materials are listed in order of increasing cost (material plus fabrication). Typical properties of some of these materials are shown in Table 1. Thermal expansion matching between the interconnect material and the silicon cell is an important consideration. However, the major technological problem confronting solar array designers for long duration missions, is that created by thermal cycling. This environmental phenomenon causes solder fatigue at the joint between the interconnect and the silicon cell. The solder having a higher coefficient of expansion than silicon is subjected to repeated tensile stresses during low temperature excursions. The results in fatigue induced microcracks in the solder which ultimately propagate into the silicon cell. Ultimately, separation between the materials occurs. This can cause a reduction in array power when the failures per cell exceed the built in redundancy. Since for many long duration missions a large number of temperature cycles may occur, it is most important that intercell connections be capable of surviving this effect.

It is recognized that the initiation of cracks can be used as an indice of eventual failure. Thus, the number of cracks observed can serve as a useful failure rate indicator. Tests at TRW Systems have shown that there is a general relationship between lower temperature limit, number of cycles, and the relative amount of cracks in the joint (Figure 19). It can be seen that a relatively small increase in the lower temperature limit will permit a large increase in the number of cycles for a given level of cracks. Thus, improved thermal control of solar arrays to increase the lower temperature limit can greatly increase array operating life.

The number of failed joints also varied greatly with the interconnect material used (85% for copper, 7% for Kovar, and no failures for molybdenum after 100 cycles to -175°C). There was also a significant correlation between failure rate and temperature differential. Another observation during this test evaluation showed that it is very important to have oxide free surfaces which are to be joined by soldering to assure good joints. It was also established that proper interconnect plating is important. However, temperature shock rate was not found to be critical for the designs considered.

Specimens tested included those in a freely suspended state as well as those bonded to metallic substrates with silicone adhesives. The bonded specimens using 0.003 inch thick material resulted in the highest failure rate. The differential contraction between the modules and the substrate was the main factor, but material thickness was also critical. Figure 20 shows that the percentage of failures, after 300 thermal cycles, rapidly decreases with reduced interconnector material thickness. This was attributed to the probability that the stress level in the 40-60, lead/tin solder was considerable reduced during low temperature excursions by the greater flexibility of the thinner materials. It was estimated that a stress reduction of 5% can reduce the number of failures by a factor of 5. Variations in the solder composition could also have a major effect on fatigue life and further investigations in this area is recommended.

The tests confirmed that a U-shaped, Kovar interconnector, 0.001 inch thick can withstand 300 temperature cycles from 60°C to -162°C with only 0.7% of the joints separating. A 70% reduction of the stress in the solder joint can be achieved if the -162°C lower temperature limit could be raised to -115°C. For an equal number of temperature cycles (i.e. 300), the number of failures would then be reduced by several orders of magnitude. This can be seen in Figure 21 which shows the percent of cracks as a function of the average stress-to-failure, for a range of standard deviations of fatigue strengths for solder. The range of standard deviations shown brackets the range normally experienced in practice. If, on the other hand one assumes the percentage of open joints to be constant, several orders of magnitude increase in the number of allowable temperature cycles should result.

## Solar Cell to Substrate Adhesive

The selection of the solar cell to substrate adhesive should be made when the actual substrate structure has been defined. There are a wide variety of suitable adhesives available. For bonding to aluminum face sheets, GE/RTV-511 or 577 is typical. To satisfactorily bond to Kapton, G.E./SMBD 745 flexible epoxy adhesive is recommended. Adhesive thickness and bonding pattern should be established by pull tests conducted for the particular environmental levels associated with the particular mission.

## Main Power Cabling

The selection of main power cabling design is intimately related to solar array power and voltage levels, as well as required stowage and deployment techniques. Material selection should be based upon optimization with respect to the following parameters:

- Electrical Conductivity
- Weight
- Solderability
- Environment
- Thermal Conductivity
- Manufacturing Complexity
- Tensile Strength
- Cold Flow Characteristics and Flexibility

Table 2 summarizes several of these parameters for several candidate bus materials. A column indicating a relative weight multiplier for the various materials as compared to copper is included. While aluminum looks attractive, its high coefficient of thermal expansion is not too compatible with large thermal cycling excursions. Copper still remains the most satisfactory alternative.

Designing the main power conductor for constant current-density power transfer results in the lightest weight regardless of allowable power loss or operating voltage (Reference 10). Figure 22 is a schematic representation of one wing of a multi-winged spacecraft. Each wing consists of several identical electrical sections interconnected in parallel to deliver a specified array power and voltage. For a constant current density cable:

$$\text{Total Wing Power Loss} = K_1 \quad N(N+1) \tag{2}$$

$$\text{Total Wing Conductor Mass} = K_w \quad N^2 \ (N+1)^2 \tag{3}$$

Where,

$$K_1 = \frac{\rho L (\frac{P}{V})^2}{A_1} \tag{4}$$

$$K_2 = \frac{\rho D L^2 (\frac{P}{V})^2}{\text{Total wing power loss in watts}} \tag{5}$$

And,

$P$ = Individual section power (constant)
$V$ = Array operating voltage
$\Delta V$ = Voltage drop between each section & the spacecraft
$N$ = Number of sections
$L$ = Conductor length between sections (constant)
$A$ = Conductor cross-section, ($in^2$)
$\rho$ = Material resistivity ($\mu$ohm - cm)

Conductor geometry (i.e., conventional wire or flat strap) selection is based upon the specific solar array structural design and deployment concept being used. The flat strap type is preferred for large area solar arrays since it is compatible with both rigid and flexible designs. It also has the potential for optimization with respect to weight, stowage efficiency, launch survivability, reliability, and cost.

## Solar Cell Design Considerations

There are many solar cell design techniques that can markedly effect array performance, life, and reliability. A few significant technology areas are discussed below.

## Magnetic Fields

A major consideration during the period of design when the arrangement of series strings on the substrate is being established is that of minimizing the magnetic field caused by uncompensated current paths on the array. Both the arrangement of series strings and the location of the main power cabling must make maximum use of current counterflow loops. In this way, magnetic interference from the array circuit can be reduced to negligible levels (approximately 1 pico tesla).

## Array Output Prediction

Solar array output calculations are normally made for the nominal output. This means that the median value of each degradation factor is used, thus resulting in a 50% probability that the predicted output will actually be achieved. This is done to avoid using an unnecessarily conservative EOL performance factor-of-safety with a correspondingly heavy design. Reference 11 describes a technique by which the power output at any desired probability may be determined. This technique utilizes the Monte Carlo method of selecting the magnitude of the degradation factors. The performance is then calculated at the particular operating voltage being considered. Figure 23 shows a typical set of curves (based upon a specific set of degradation factors and where $K=V_{op}/V_{oc}$). Data of this type can be used to determine the array output at any desired probability, thus aiding materially in refining array sizing operations. TRW Systems has determined that the power output expected with 50% probability is approximately 7% higher than that anticipated at 95% probability. Thus, an increase in array size by about 7% above the nominal design, will assure a high probability that the EOL required power can be achieved.

## Low Energy Proton Protection

As additional operating experience is obtained for long duration satellite systems, new technology problems occasionally arise. One such case is the excessive damage caused to uncovered portions of solar cells by low energy protons. Recent data from the ATS-1 Satellite, which has subsequently been verified by laboratory tests, have shown this to be the case. The damage mechanism is hypothesized as having a shorting effect on the solar cells. This has resulted in a power output degradation of 23% in one year. This was far in excess of the exposed cell area of 0.7% which being uncovered, would be the only region where rapid radiation damage would be anticipated. In the ATS-1 application, the cells used were 1 cm. x 2 cm., N-on-P type, with solder covered contacts. The negative contact was along the 2 cm. side. These cells were covered with 0.030 inch thick fused silica cover slides. Because of the standard tolerances on the cover slide and the assembly techniques used, a 0.005 inch wide strip along the 2 cm. length remained uncovered. Preflight degradation estimates for this condition were estimated at 2% per year.

The unexpected damage was attributed to low energy (E< 5 MEV) protons which damaged the exposed cell area. It was found that at fluence levels of $10^{13}$ p/cm$^2$, a shift in the characteristic current-voltage curve knee occurred, resulting in a 25% decrease in the maximum power output. This large drop in performance was experienced despite the fact that the exposed area of each cell was only 0.4% (0.003 in. x 0.788 in.÷ 0.589 in.$^2$ x 100). It was determined that satisfactory protection from this degradation mechanism can be achieved by the application of a thin layer of protective material over the exposed area.

There are several methods that can be used to eliminate the exposed cell area. One low cost technique is to apply a thin layer of the cover slide adhesive to the exposed areas when using cover slides with conventional dimensions and tolerances. This operation can be performed to the solar array panel after final assembly. This adhesive is acceptable since it will not decrease transmissibility at BOL. The extent of ultraviolet darking of the exposed adhesive at EOL has yet to be determined. However, even if transmissibility was reduced to zero (100% darkening) less than a 1% reduction in output power would be experienced. Further testing is recommended to assure that the low energy proton protective capability of this adhesive is not significantly diminished as a result of longer duration operation.

Another approach would be to use oversize cover slides to completely eliminate all exposed areas. This method, although very effective is considerably more costly than the first approach. This results from the tighter tolerances required and the higher reject rate to be expected because of this.

## Lithium - Diffused Radiation Resistant Cells

As was previously discussed, charged particle radiation damage (Reference 12) remains one of the major factors in solar array performance degradation. Recent research in solar cell technology has led to the development of the lithium-diffused radiation resistant cell. These cells have been shown to have a definite ability to recover from radiation damage (Reference 13). There is, however, a considerable lack of understanding regarding the nature of the radiation-induced defects in lithium doped silicon and the annealing process. Figure 24 shows a comparison between changes in the diffusion length for a lithium-doped cell and a control cell under 1-MEV electronic bombardment. Both cells were made from the same silicon wafer. Irradiation was stopped periodically and the cells were allowed to anneal at room temperature for the time periods indicated. It is theorized that the damage site involves a pairing between a defect and a lithium ion. The recovery mechanism then involves the diffusion on another lithium ion to the damage site and its pairing with the lithium-defect complex. It has been established that the speed and extent of recovery are strongly dependent on the amount of free or

or unpaired lithium. There is some evidence that a cell structure which incorporates a diffused phosphorus layer next to the junction (i.e. between lithium - diffused region and P-region) to give a three-layered p/n/n structure, results in a cell with improved radiation resistance without sacrificing the ability to recover. While high efficiency (13 to 14%) lithuim doped cells have been produced, on the average they appear to have lower efficiencies than comparable cells without lithium doping. Effects of elevated temperatures and long term exposure to hard vacuum require further investigation before these cells can be given serious consideration.

## Shadowing Effects on Solar Cells

For various missions, frequently portions of the solar array are shaded by structural elements such as antennas, booms, or portions of the spacecraft. Such shadows are usually time varying and of complicated geometry. Accurate knowledge of the output losses is required for a precise determination of the actual array power output, as well as for assessment of bus voltage variations, ripple, and RF noise.

Because of the electrical characteristics of solar cells, the losses are not proportional to the projected shaded areas, but are greater. This is due to two factors. The shadowed cells which are in series with illuminated cells block the current flow in the entire series string. Shadowed cells in parallel with illuminated cells shunt part of the generated current. Current flow blocking may be minimized by installing so-called shunt-diodes across cell groups. Shunting can be reduced by dividing large parallel groups into small ones and connecting each group through a so-called blocking diode to the bus. This diode isolates shadowed groups of cells from illuminated ones and thereby prevents shunt current losses.

In estimating performance and designing the cellstack circuitry for the solar array, these shadowing effects must be taken into account. The use of mathematical models of the type outlined in Reference 14, can greatly simplify this complex analysis and still provide the desired accuracy for the array output power estimate.

## Solar Array System Considerations

Because the power level requirements of future spacecraft are continually increasing, considerable emphasis is being placed on the structures and mechanisms technologies needed and voltage level selection, for large area, high power, light weight solar cell arrays. Typical of these are some of the development activities outlined below.

### Solar Array Substrate Design Criteria

The trend in photovoltaic solar arrays has been toward developing light weight, lower cost substrate structures. Emphasis has been placed upon increasing the power-to-weight ratio (w/lb). The folding modular and roll-up systems are essentially two basic solar array configurations which

appear most likely to achieve these objectives. The necessity to reduce the substrate specific weight can best be seen on Figure 25. At higher array power densities, the maximum allowable structure-mechanism weight is reduced. Thus, for a selected cellstack design, an upper limit exists for array structural weight, if a desired power density is to be achieved.

Currently several major development efforts are being carried out. The most notable is the folding panel configuration being developed by The Boeing Company. An array power density of 20-24 w/lb. for a nominal 50 $KW_e$, 5000 $ft^2$ area system has been specified. This array uses silicon cells (8.6%, AMO, 55°C, 1 ohm-cm, 8 mil thick, 3 mil cover slides) mounted on a rigid substrate. The substrate consists of an edge frame and center spar made from 0.015 in. beryllium double box beams and a 0.003 in. x 0.2 in. woven fibre glass substrate. A specific weight of 21.8 w/lb. has been achieved. The substrate specific weight is approximately 0.16 $lb/ft^2$ which does not include the deployment mechanism.

Other companies actively engaged in light weight solar array development include TRW Systems, Hughes, Ryan, Fairchild-Hiller, General Electric, and Electro-Optical Systems. EOS using electro-forming techniques is endeavoring to develop a 40 w/lb. rigid silicon cell array. To achieve this high specific weight they are using 4 mil thick cells and 1 mil thick cover glass. These appear to be high cost elements and somewhat beyond the practical regime for current SOA silicon cell array fabrication techniques. The other companies (with the exception of TRW Systems) are under contract to JPL to develop a nominal 10 KW, 30 w/lb. silicon cell roll-up array. Results to date are listed in Table 3. Details of their construction and the deployment techniques utilized are given in Reference 16.

TRW Systems has been concentrating on developing CdS thin film multi-kilowatt arrays in conjunction with an in-house program tied to the development of an electrically propelled interplanetary spacecraft. A typical design for a two-wing, 2.5 KW flexible structure solar array has achieved a specific weight of 27 w/lb, which includes the deployment mechanism. In addition, under contract to NASA/MSFC they are developing fold-up modules for lunar surface applications which show promise of achieving specific weights up to 40 w/lb.

## Rotary Joint Power Transfer Mechanism

Large area solar arrays will require rotary power transfer mechanisms to permit roll-out type deployment and/or deployed array articulation. There are several methods by which this may be accomplished. These are:

- rotary transformers
- spiral wound continuous cables
- slip ring assemblies

Because of their specialized design requirements, rotating transformers are not considered practical at this time. Continuous spiral wound cables have limited life and impose an unnecessary limitation on degree of array articulation. They also result in relatively high power losses and overall power system weights. Slip rings appear to be the most attractive rotary joint power transfer mechanism available today. They combine light weight, low power losses, high reliability, and unrestrained articulation capabilities with relatively low cost and proven performance in the space environment.

Slip rings can be either of two types, i.e., pancake or drum, depending upon the design constraints imposed on the pancake dimensions. This solar array component is reliable, based upon over 150 x $10^3$ vacuum operating hours. This data is the summation of a total of 54 space flown and laboratory tested units. Typical failure rate data is 6 failures/ring in $10^9$ hours.

Power transfer from the array to the slip ring is accomplished through terminal strips or connectors. The power loss through the ring assembly depends upon the particular design. Typical losses are in the order of 0.25% up to at least the 10 KW power level. There are no apparent design restrictions which will prohibit slip ring designs for 50 KW or higher. However, there are several technology areas which will require further evaulation, if required by the spacecraft requirements, due to the lack of operational data. These are:

- o use of multiple rotating joints (one for array deployment and another for array articulation)
- o transfer of power voltages higher than 100 VDC
- o operation at temperatures below -10°C

A considerable amount of detailed design information is available from Ball Brothers Research Corp. They have determined that power transfer at voltages above 200 VDC causes creepage between circuits. Corona and/or electrostatic discharge are additional problems associated with high operating voltages. Close consideration must be given to ring spacing, dielectric insulation properties of materials, material degradation effects and the impact of increased power losses and noise. Normally, proper selection of barrier materials and increased circuit spacing are adequate to minimize any deleterious effects. This results in a negligible weight increase. If slip ring assembly envelope dimensions are not too tightly constrained, problems with high operating voltages can usually be eliminated.

Effective lubrication for low temperature operation is required. "Dry", compact composites having self-lubricating properties and good electrical conductivity are typical. These are usually metal based and contain small percentages of polytetrafluorethylene and a dry lubricant such as niobium-diselenide. This lubricating film transfers back and forth to heal any

faults in the film. "Wet" films supplied by resevoirs, such as Ball Brothers "Vac-Kote or Esso's Poly-Scientific Division Univis P-38 oil are other lubricants commonly employed for space applications.

Space proven slip ring designs have operated for over 50,000 revolutions at speeds up to 60 RPM with negligible wear (based on examination of qualification test models). However, the slip ring assembly remains one of the most difficult single components of the solar array to design for longlife and much further development effort is warranted.

Voltage Selection

Solar array weight optimization is influenced by other factors than component mass-density. Selection of the array operating voltage and allowable loss in the main power cable are two such design criteria.

The weight of arrays with shunt diodes continually decreases with increased operating voltage. This is due to two factors. The inherent high reliability achieved by the use of shunt diodes does not impose any appreciable weight penalty. In addition, the required main power cabling weight continually decreases with increased voltage. Omission of shunt diodes causes the system to have a discrete voltage at which the total weight is a minimum. Figure 26 shows the relationship among operating voltage, power level, and the predicted power loss (required over design) for this case for a reliability of 0.995. It can be seen that with a prescribed reliability level, the required over design increases rapidly with increased voltage. If shunt diodes are incorporated across each parallel cell sub-group the required over design is considerably reduced. For example, at a reliability of 0.999999, the required over design is approximately 0.1% and independent of voltage level.

The optimum voltage range without shunt diodes is between 100 to 300 VDC. With shunt diodes the range can be increased to from 500 to 1000 VDC. The actual optimum voltage depends upon the power level and allowable main cable losses. Above 1000 VDC finite but negligible weight reduction results with shunt diodes.

Solar arrays with shunt diodes operating at optimum voltage, are generally on the order of 10% lighter at 0.999999 reliability, than their alternative with a reliability of 0.995 for a 3 year mission. Reducing the operating voltage of a shunt diode protected array to the range for the alternative results in approximate weight parity but at a somewhat higher reliability. It is clear then that array voltage need not exceed 300 VDC for weight optimization. However, at a given power level and mission life, consideration of other factors such as reliability, cost effectiveness, and circuit compatibility become important. For example, a shunt diode protected array results in an increase in array cost. Shunt diode unit costs (∿$2), amplified by additional tooling costs and increased assembly time reduces their desirability. The cost effectiveness of their use must

be weighed with respect to the requirements of each specific mission. In addition, even though total array weight decreases with increased operating voltage levels for the shunted configuration practical considerations sometime preclude their use. Quite often the voltage level selection may be limited by power transfer mechanism, power conditioning, or load constraints.

For many applications there does not appear to be any significant advantage in operating solar arrays at voltages above 300 VDC. However, should the spacecraft load dictate the requirement (i.e.2000 to 16000 VDC for electric thrustors and T.V. broadcast high frequency electron tubes) then techniques must be developed to make this achieveable. In this case, additional problems associated with interactions with the charged space plasma and increased spacecraft torques due to the impact of the magnetic field between the array and the plasma caused by uncompensated current paths, must also be resolved.

## CONCLUSIONS

The proven performance and reliability of solar arrays will make them the ideal electric power system candidate for the many potential missions being considered for the next decade. Because of their potential for high power, long duration operation, with minimum development risk, mission planners and spacecraft designers will turn increasingly towards their use.

The technology problems stemming from the desire to develop large, high power level solar arrays are primarily related to achieving improvements in the time-independent factors of BOL performance. The emphasis here is on increased solar cell conversion and power conditioning efficiency, improved manufacturing techniques and materials, and a refined assessment of spacecraft operating modes and solar array interactions. Development of light weight structures, and reliable deployment, orientation, and power transmission systems, (Reference 15) are equally important.

The technology problems related to the quest for extended mission duration are dominated to a large degree by the necessity of obtaining major improvements in the time dependent degradation factors. The development of increased radiation resistant materials is the key element to success in this area. Improved thermal control and decreased sensitivity to the range and magnitude of thermal cycling are additional factors that could enhance the reliability of long duration operation.

Finally, because of the multi-faceted aspects of solar array systems, increased emphasis must be placed on developing new analytical and experimental tools. These must be oriented towards achievement of major improvements in solar array design optimization techniques. Only in this manner can the basic objectives of reduced weight and cost, and increased reliability be achieved, to meet the future goals of long duration operation of large solar arrays in the hostile space environment.

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$$P_O/A = \left[\eta_P \times \eta_C \times \eta_M \times \eta_O \times \eta_{REG} \times \eta_{CONV}\right] \times \left[\eta_{TC} \times \eta_{RAD} \times \eta_{UV} \times \eta_{MM}\right] \times I_{SOL}$$

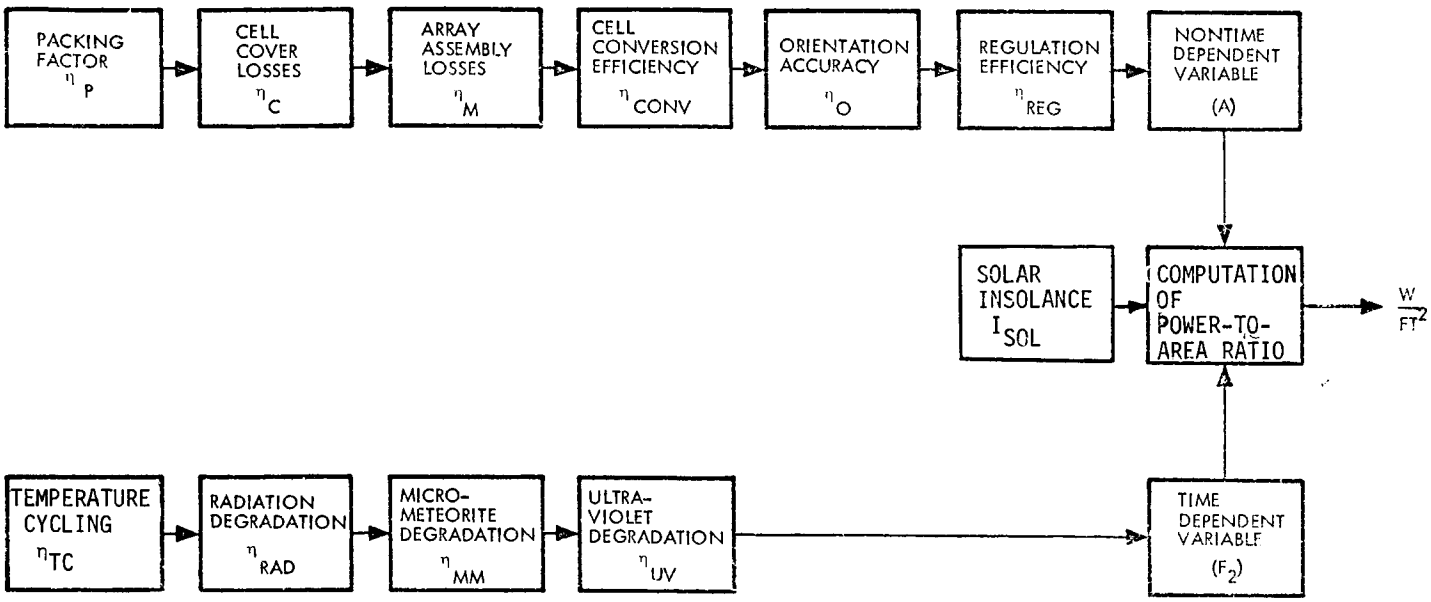


Figure 1. Factors Considered in Power-To-Area Ratio Computation for Lunar-Based Solar Arrays

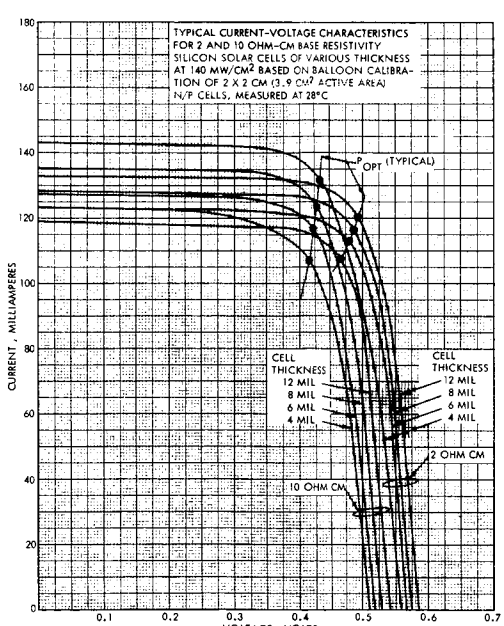


Figure 2. Typical Current-Voltage Characteristics for 2 and 10 ohm-cm Base Resistivity Silicon Solar Cells of Various Thicknesses

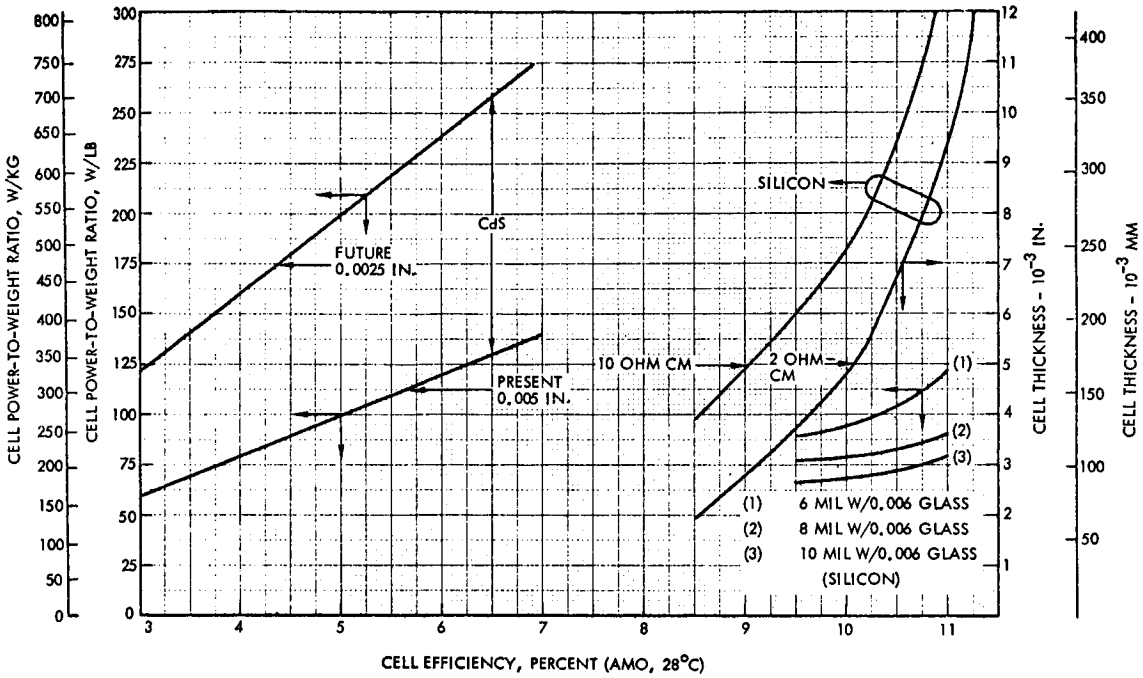


Figure 3. Power-To-Weight Ratio Versus Efficiency for Silicon and Thin Film Solar Cells

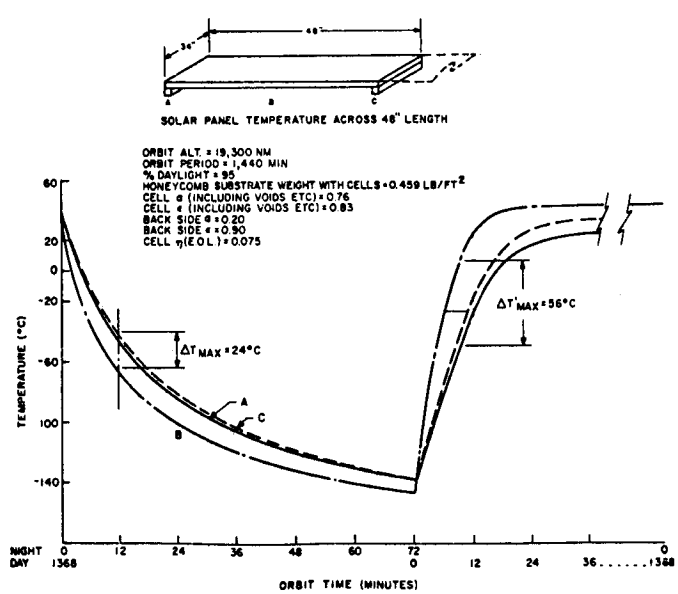


Figure 4. Temperature-Time Profile for Representative Panel

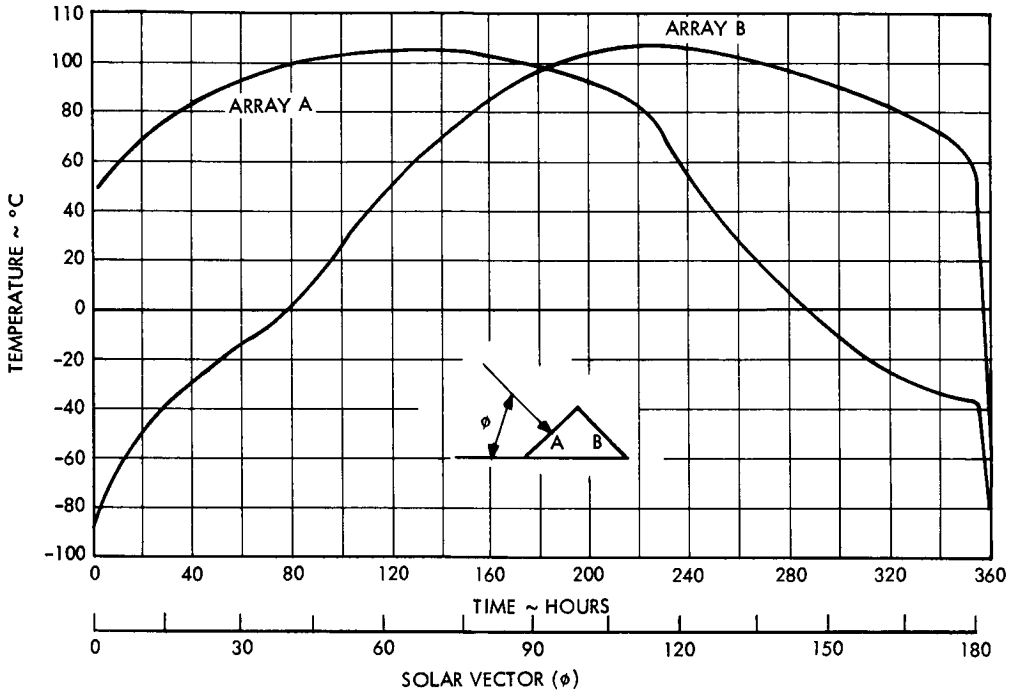


Figure 5. Lunar Based 45° Lean-To-Array Temperatures As a Function of Solar Heating Model I Unmodified

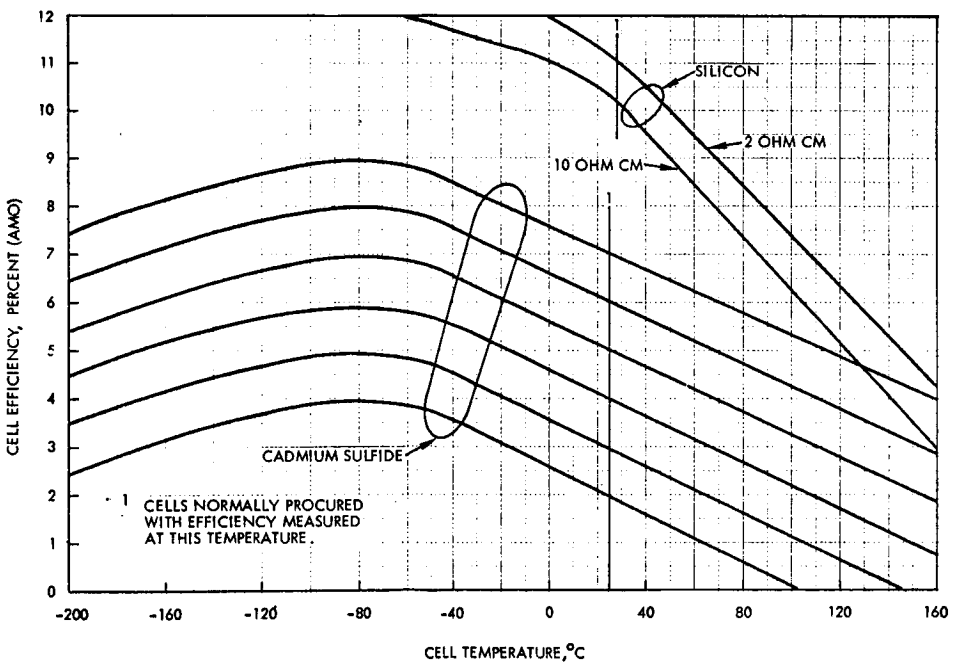


Figure 6. Cell Efficiency Versus Temperature Comparison for Silicon and Thin-Film Solar Cells

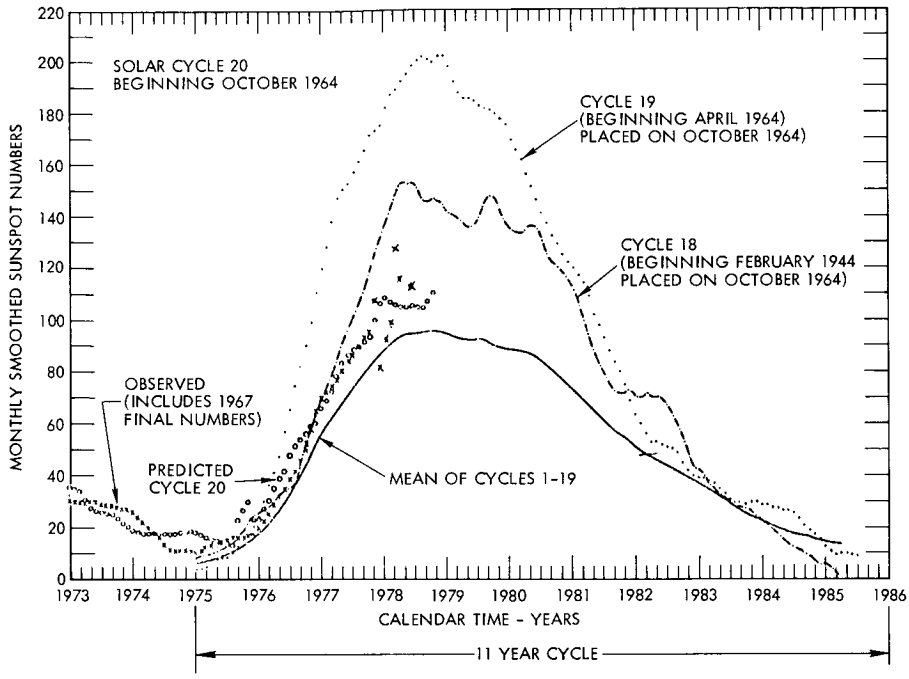


Figure 7. Sun Spot Numbers for Several Solar Cycles

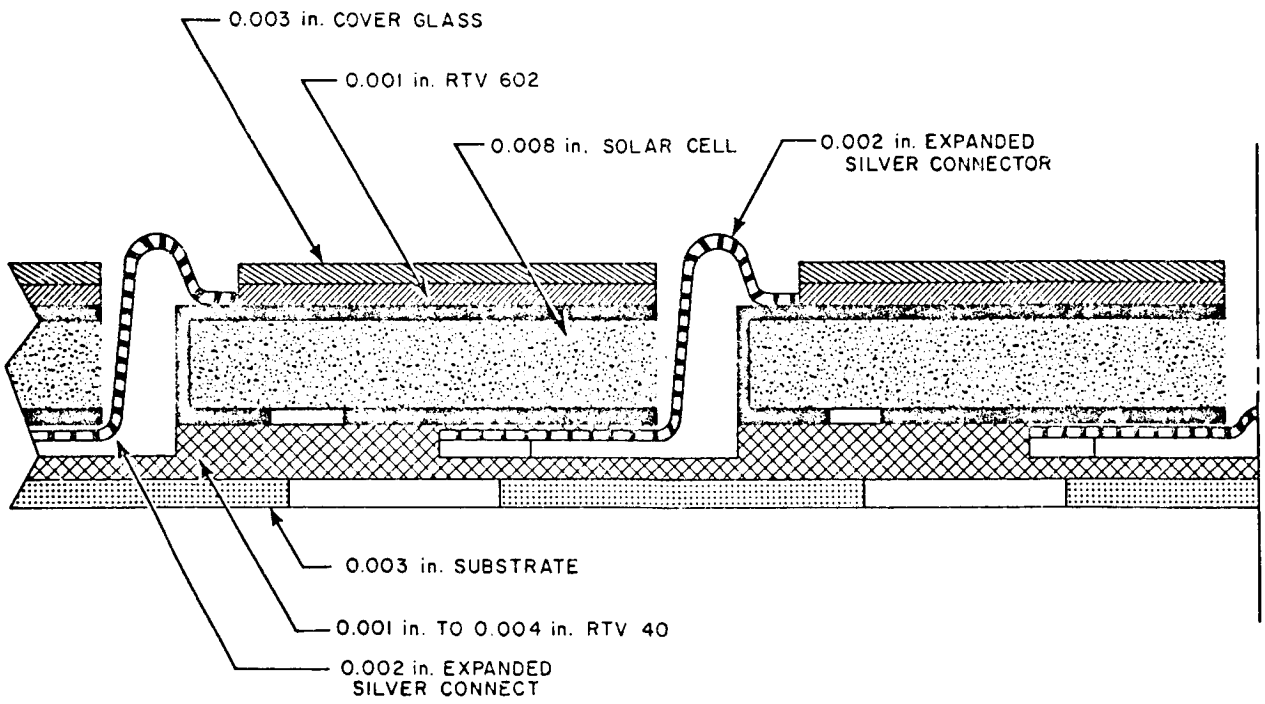


Figure 8. Advanced Solar Cell Stack Concept

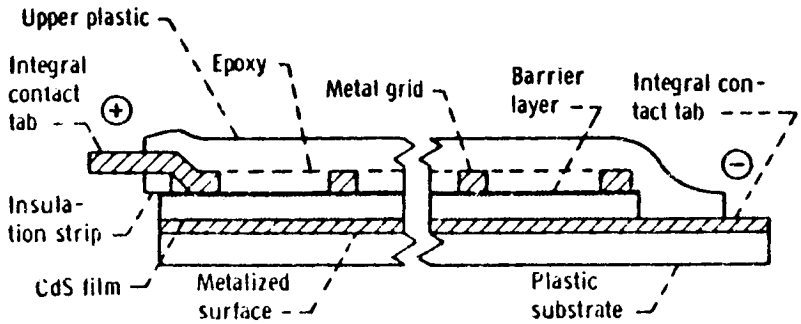


Figure 9. Cross-Section of CdS Thin Film Solar Cell

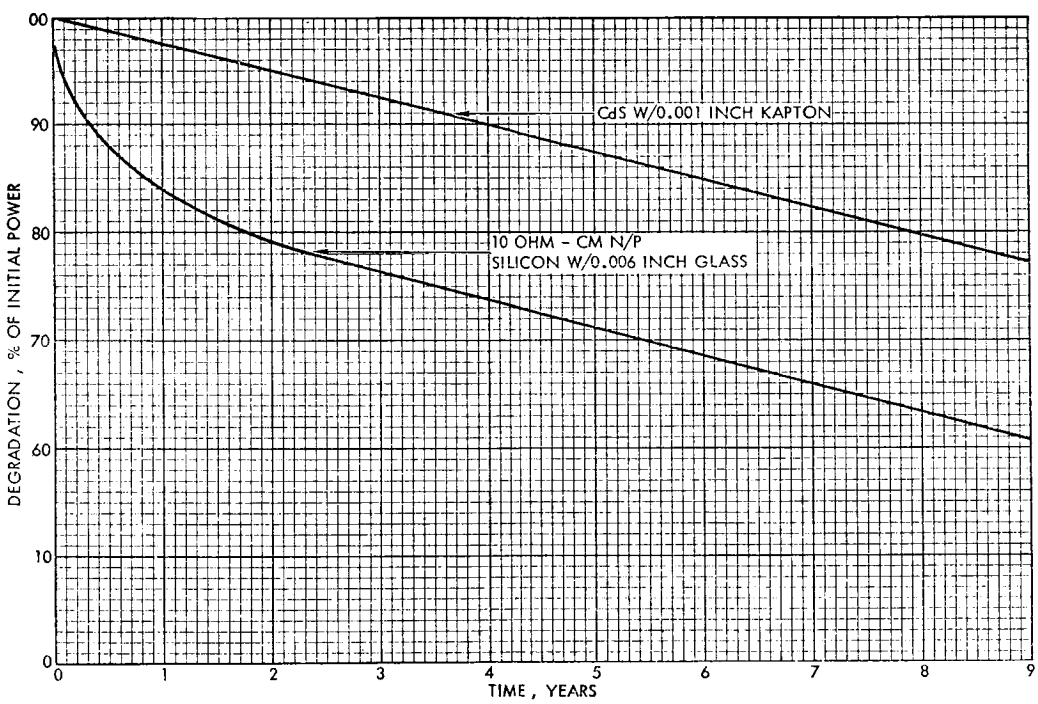


Figure 10. Silicon and CdS Thin Film Solar Cell Degradation due to the Space Environment

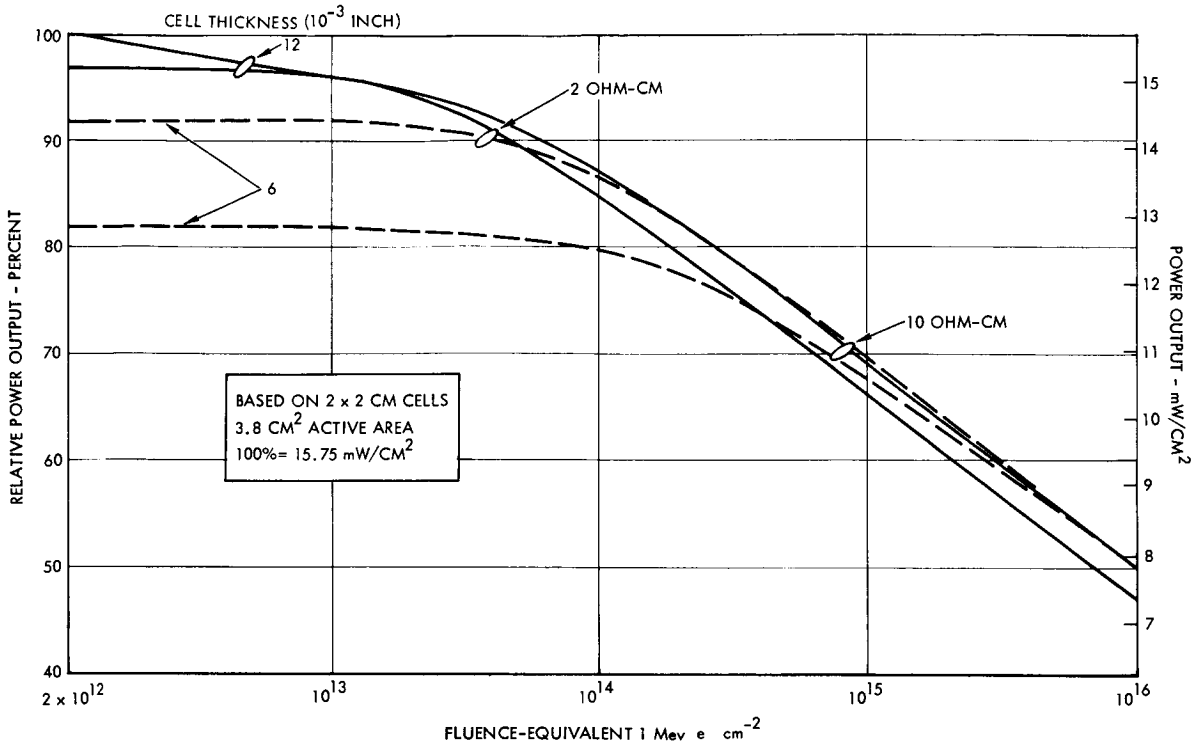


Figure 11. Effect of Fluence on Solar Cell Maximum Power for Various Cell Thicknesses and Base Resistivities.

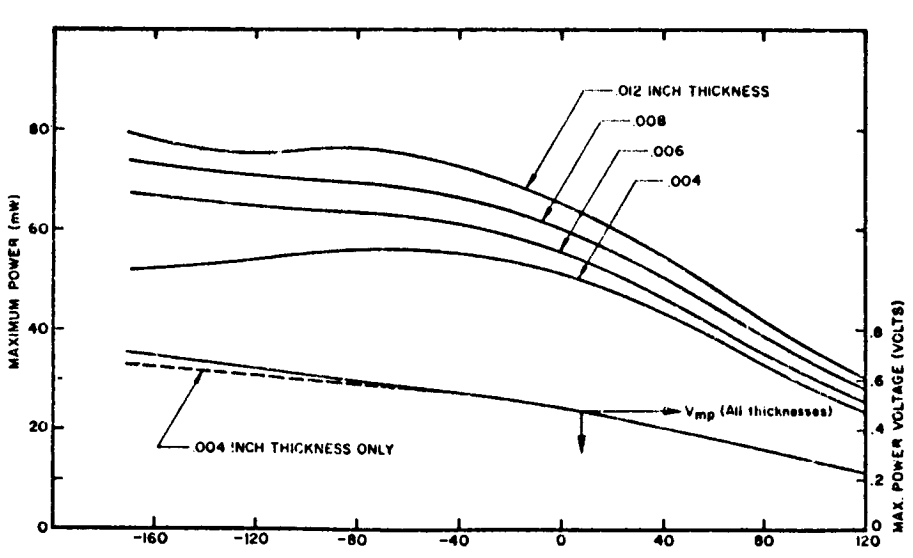


Figure 12. Maximum Power and Voltage vs Cell Temperature for 2 x 2 cm 10 ohm cm N+ on P Solar Cells, Active area 3.9 cm$^2$, Sunlight Simulator 140 mW/cm$^2$.

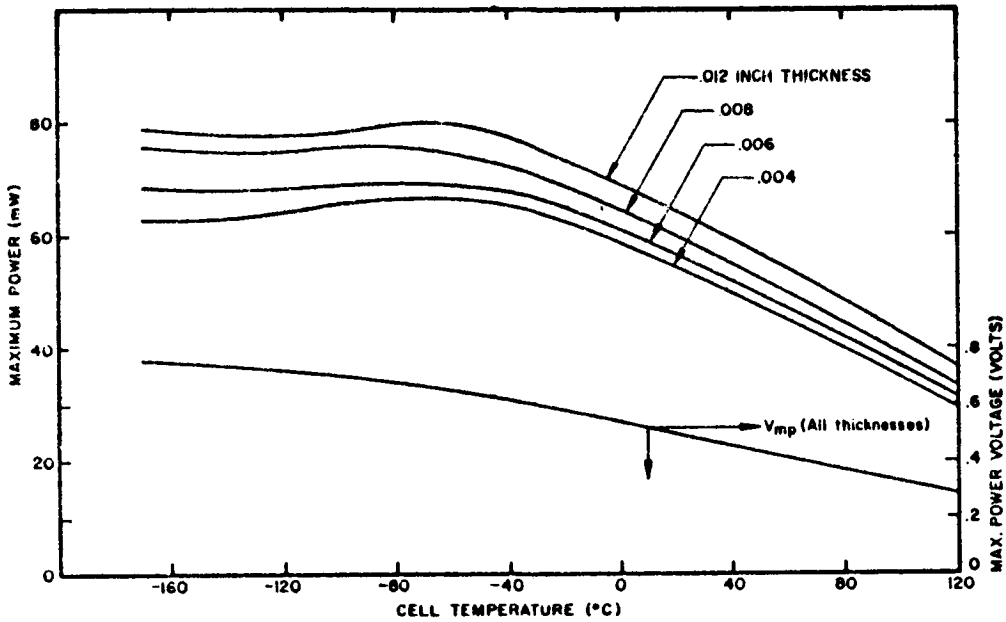


Figure 13. Maximum Power and Voltage vs Cell Temperature for 2 x 2 cm 2 ohm cm N+ on P Solar Cells. Active Area 3.9 $cm^2$, Sunlight Simulator 140/mW/$cm^2$.

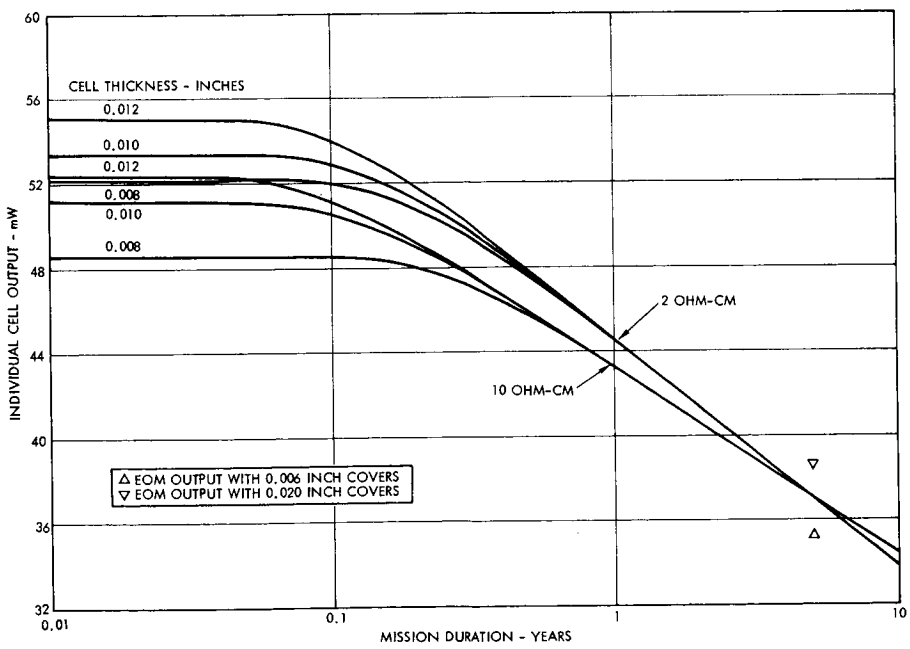


Figure 14. Individual Solar Cell Performance Versus Mission Duration for Various Thicknesses of 2 and 10 ohm-cm Solar Cells at AMO, 28°C for Bodymounted Cells Which are Normal to Solar Irradiation and having 12 Mil Cover Slides. No Spacecraft Geometry Considered.

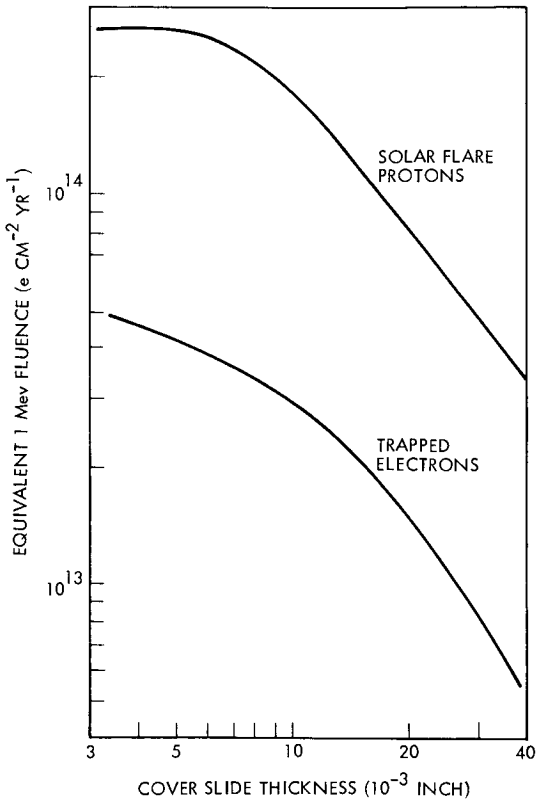


Figure 15. The Effect of Cover Thickness on The Yearly Solar Flare Proton and Trapped Electron Fluence in Terms of Equivalent 1 Mev Electrons, During Peak Solar Flare Period.

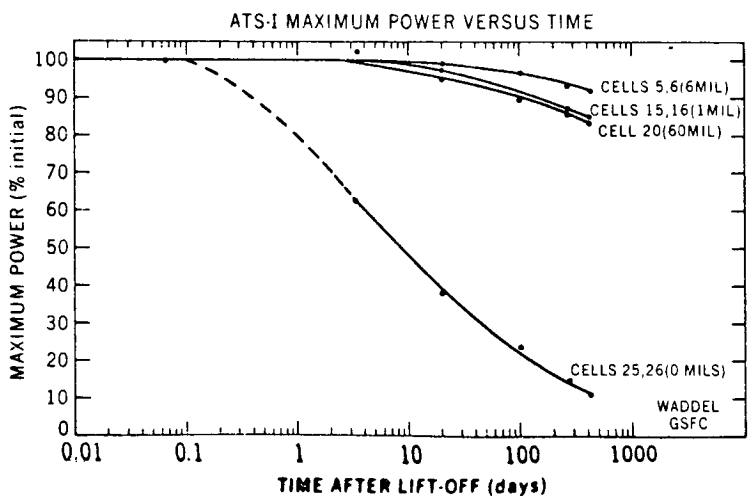


Figure 16. Maximum Power Versus Time for Several Cells with Various Shields.

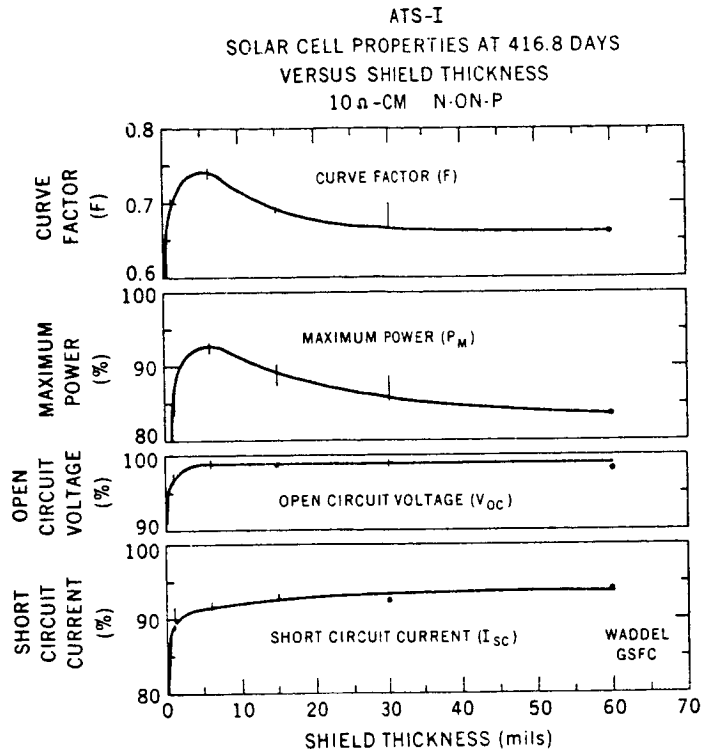


Figure 17. Curve Factor (F), Maximum Power ($P_M$), Open-Circuit Voltage ($V_{OC}$), and Short-Circuit Current ($I_{SC}$) Versus Shield Thickness, at 416 Days After Lift-Off.

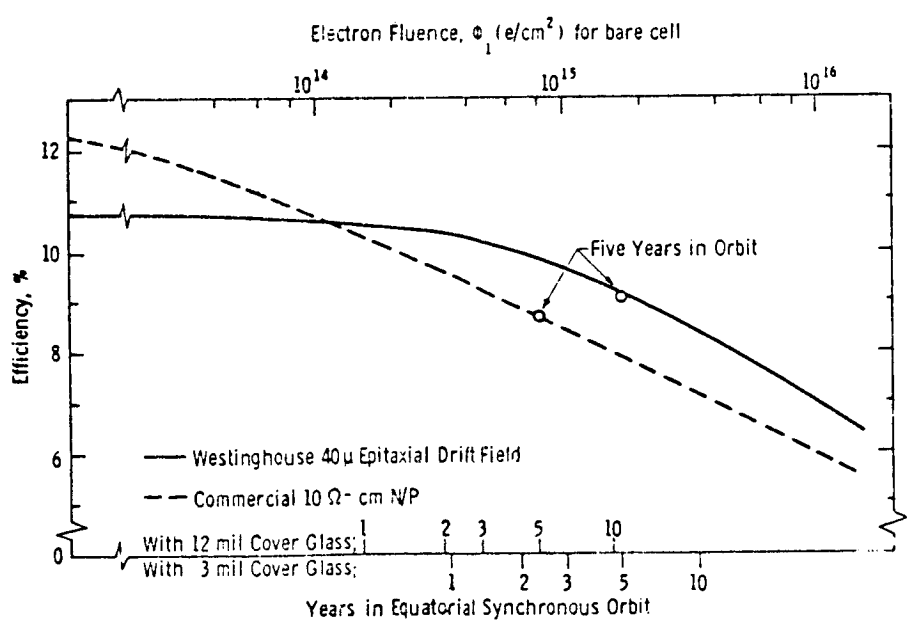


Figure 18. Solar Cell Efficiency vs. 1 MeV Electron Fluence or Years Exposure in Equatorial Synchronous Orbit. Efficiency Measured with 3400°K Filtered Tungsten Light at 100m W/cm$^2$, as Indicated On Standard Solar Cell Calibrated Table Mountain Sunlight.

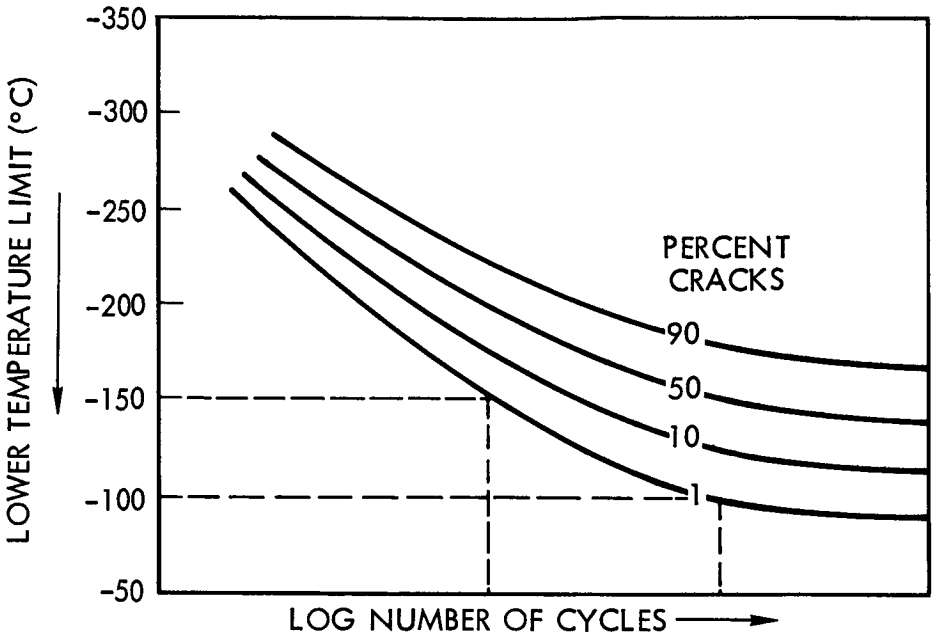


Figure 19. Typical Fatigue Life Curves for Solar Cell Interconnections

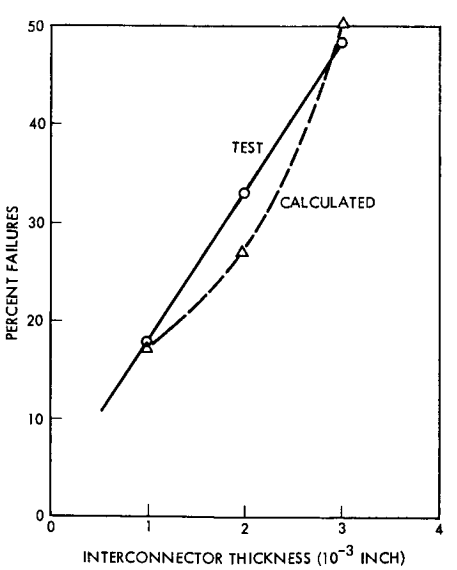


Figure 20. Solar Cell Joint Failure Level After 300 Thermal Cycles Versus Interconnector Thickness as Determined by Tests and by Calculations

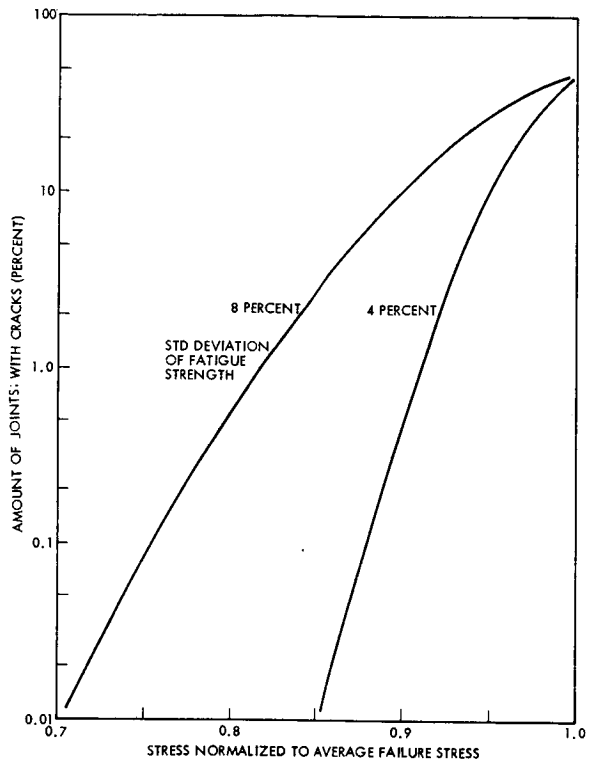


Figure 21. Cell Interconnect Joint Fatigue Cracks Versus Normalized Stress for Typical Range of Solder Fatigue Strength Distribution.

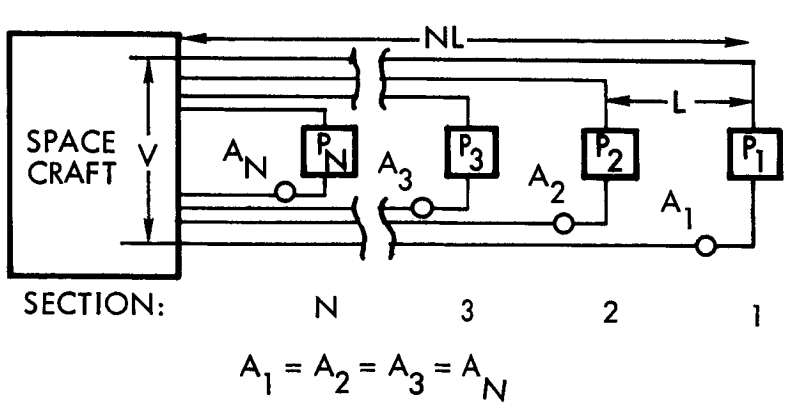


Figure 22. Schematic Representation of Solar Array Wing.

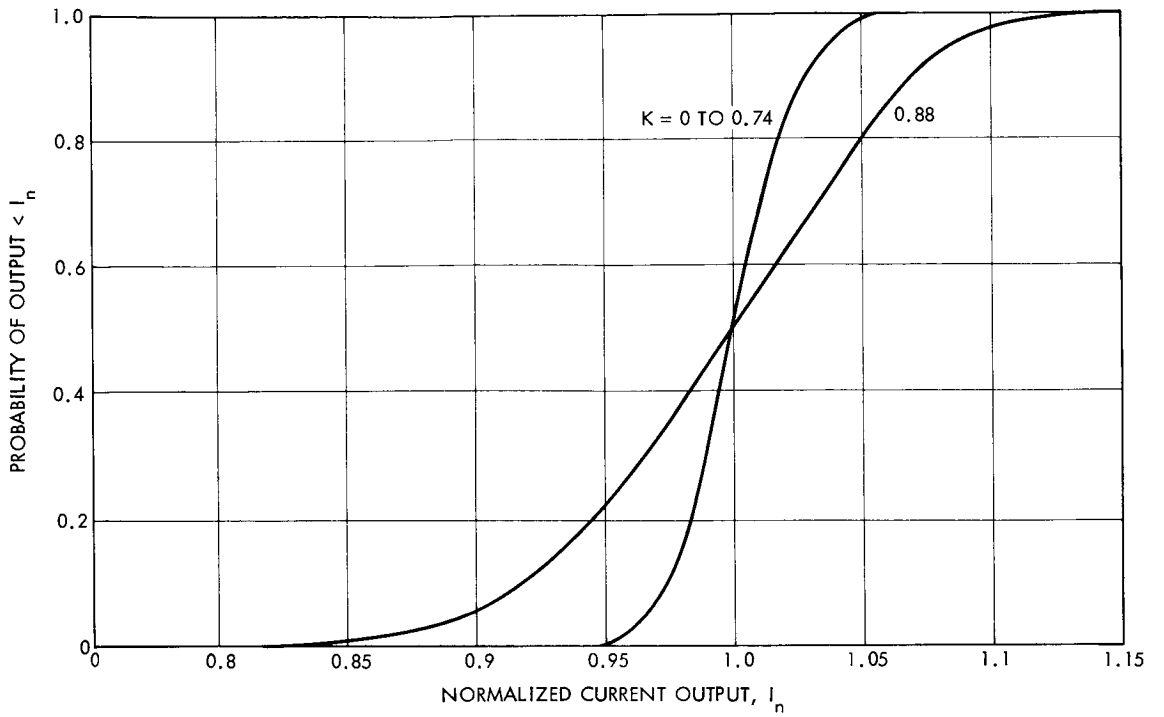


Figure 23. Dimensionless Current Output Distribution Over a Range of Normalized Voltages.

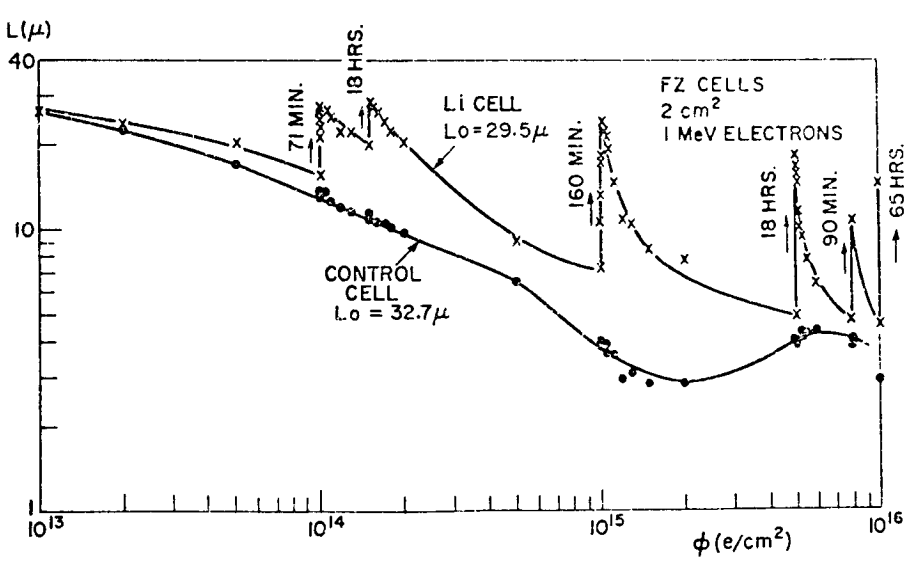


Figure 24. Comparison of the Performance of n/p and Li-doped Silicon Cells under Irradiation

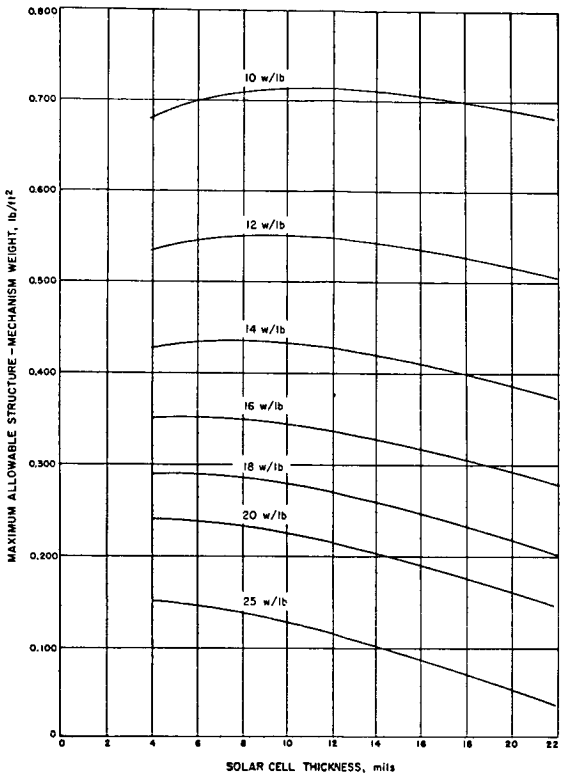


Figure 25 Allowable structures weight($lb/ft^2$) at various levels of Specific Power as a function of Cell Thickness (for) Mariner IV Cell Stack Specific Weight of 0.391 $lb/ft^2$)

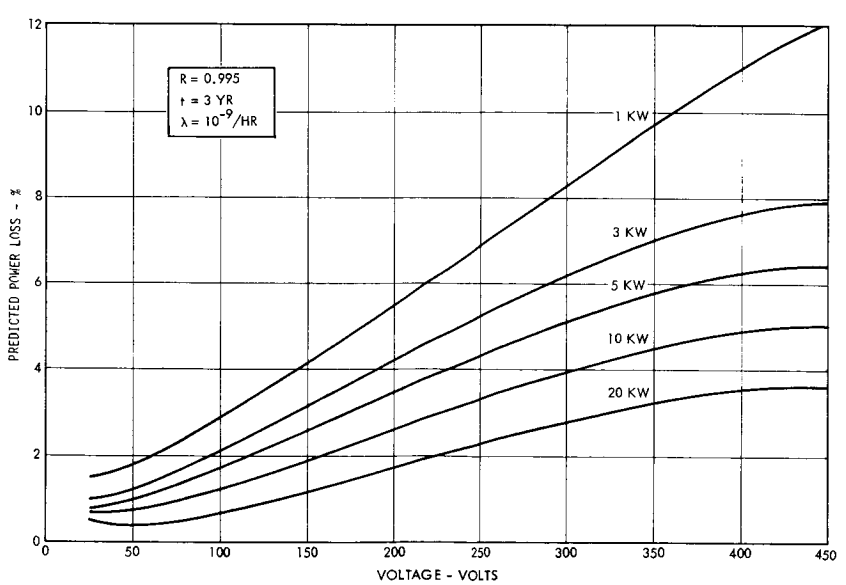


Figure 26 Effect of Array Operating Voltage on Power Loss for Silicon Cells at Various Power Levels (No Shunt Diodes)

Table 1. Comparison of Various Candidate Solar Array Electrical Interconnection Material Densities and Expansion Relative to Silicon.

| Material | Relative Thermal Expansion | Density | |
|---|---|---|---|
| | | $lb/ft^3$ | m/cc |
| Silicon | 1 | 151 | 2.33 |
| Copper | 4.1 | 555 | 8.89 |
| Kovar | 1.18 | 521 | 8.35 |
| Molybdenum | 1.20 | 562.5 | 10.20 |
| Iridium | 1.4 | 1400 | 22.42 |
| Platinum-Iridium | 1.87 | 1335 | 21.4 |
| Tantalum | 1.62 | 1036 | 16.6 |
| Aluminum | 4.6 | 168 | 2.7 |
| Brass | 4.9 | 527 | 8.44 |
| Beryllium | 3.5 | 113.5 | 1.82 |

Table 2. Summary of Candidate Bus Material Physical Characteristics

| Candidate Conductor Material | Resistivity ($\mu$ ohm-cm at 20°C) | Density (gm/cm$^3$) | Tensile Strength (Psi at 20°C) | Thermal Expansion Coefficient ($^oC^{-1} \times 10^{-6}$) | Thermal Conductivity (W/cm°C) | Cold Flow | Solder-ability | Normalized Weight for Equivalent Power Loss | Magnetic |
|---|---|---|---|---|---|---|---|---|---|
| Copper (Pure) | 1.69 | 8.94 | 28,000 | 16.6 | 3.89 | Good | Very Good | 1.0 | No |
| Copper Alloy (Commer-cial Wire) | 1.91 | 8.94 | 60,000 | 17.0 | 3.89 | Good | Good | 1.13 | No |
| Silver | 1.59 | 10.49 | 18,000 | 19.7 | 3.89 | Poor | Very Good | 1.1 | No |
| Beryllium | 4.20 | 1.85 | 50,000 | 12.0 | 1.47 | Very Good | Poor | 0.51 | No |
| Aluminum | 2.66 | 2.70 | 11-15.5K | 23.8 | 2.09 | Poor | Fair | 0.48 | No |

Table 3. Comparison of Solar Array Specific Weights from Industry

| Company | Cell Type | Cell Thick ness, in. | Cover Glass Thick-ness, in. | Cell Performance Characteristics | Array Specific Weight W/Lb |
|---|---|---|---|---|---|
| EOS | N/P | 0.004 | 0.001 | 10.1%, AMO, 55°C | 37 |
| G.E. | N/P | 0.008 | 0.003 | 8.45%, AMO, 55°C | 30 |
| Ryan | N/P | 0.008 | 0.003 | 8.55%, AMO, 55°C | 30 |
| Fairchild-Hiller | N/P | 0.008 | 0.003 | 9.3%, AMO, 55°C | 34 |

# PRACTICAL BENEFITS OF RELIABILITY ANALYSIS FOR LONG LIFE DESIGNS

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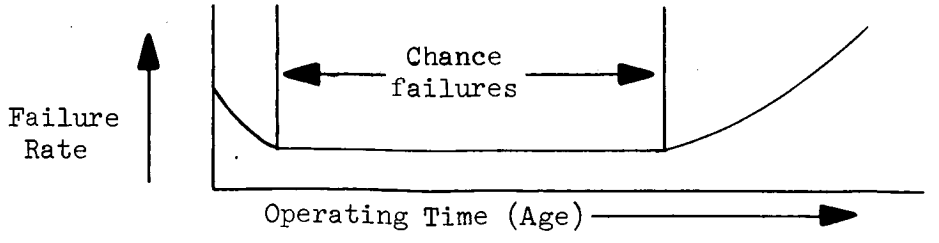
## Introduction

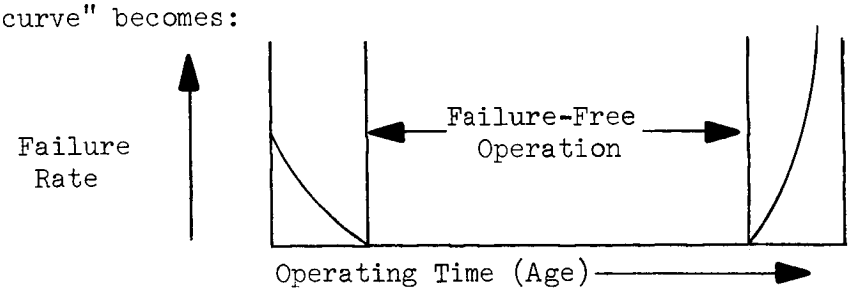
A full understanding of unique project characteristics is the basis for selecting reliability analysis techniques. Long continuous duty cycles, repetitive cycles of use, standby quiescence, and long storage times prior to activation lead the list of Long Life Design requirements unique to long missions. Success in meeting the challenge of long mission durations is evidenced in programs such as Nimbus and Tiros. To reflect this success into large manned missions requires a systematic approach for extending the life capabilities of technologies such as fluid mechanics (life support), high energy devices (propulsion, power generation), and high stress mechanical devices. All these considerations point to a need for failure-free operation in Long Life Designs. This need demands excellent concepts and their thorough application in all the engineering disciplines. Practical benefits must be achieved from all supporting analyses. This paper examines Failure Mode and Effect Analysis (FMEA) for its practical benefits and expresses some thoughts for focusing the analysis on Long Life Design requirements. Other reliability analysis techniques having practical benefits for Long Life Designs are identified.

## Reliability Requirements for Long Life

Perfection is impossible, but exceeding stated performance specifications can be accomplished. Much can be said about the importance of specifications which are within the performance envelopes of the equipment. For failure-free operation, the specification can be expressed in terms of isolating failures to certain portions of an equipment's life cycle. A hint of this specification is contained in the "bathtub curve."



The portion of the curve called "chance failure"[1] also can be referred to as "failures with unknown causes." Experience with electronic circuits has shown that these unknown causes have measurable characteristics which can be used as a basis for screening parts. If all parts with unknown failure causes are successfully and continuously screened, the "bathtub curve" becomes:




These considerations support the school of thought that "all failures have a cause" and suggest that failure-free operation is accomplished by thorough understanding of the causes of failures and the equipment characteristics that produce a signature for these causes. The Failure Mode and Effect Analysis is the design tool that will aid in isolating these equipment characteristics. In passing, it should be noted that this concept of failure-free operation and the ways in which it must be accomplished discard the constant failure rate approach and require the application of Weibull or other multiple degree-of-freedom distribution of failure probability.

## FMEA - A Beneficial Design Aid

A FMEA is cause/effect reasoning and its documentation must be viewed as a means of communicating this reasoning to other people. Long Life Designs require an expansion of the reasoning process which can be done without increasing the documentation.

An examination and expansion of the basic reasoning process are required. FMEA, as the thinking done by the designer to evaluate each design concept, results in a scorecard to isolate those characteristics of the design which breed failures in the use cycle. This continuing evaluation leads to mature design concepts which can accomplish their function simply, directly, and with minimum energy transfer. As mature designs evolve from new concepts, the number of failure-breeding characteristics will continually diminish. The number of these characteristics which exist in flight equipment is related to the amount of time available to mature the design prior to flight. The FMEA is beneficial to the designer as an action item list of failure causes to be eliminated before the design is mature. To eliminate these causes may involve

state-of-the-art advancement, application of new materials, or basic invention. Since the designer is always time-constrained in implementing these approaches, the FMEA thought process must be documented. This documentation of the designers' thinking aids in developing work-arounds to offset failures in the use cycle. Except for this need, the FMEA notes of the designer would require no more documentation than his other notes and sketches.

The failure-causing characteristics and effects of the failure are as unique to an equipment item as its performance parameters and physical dimensions. The effects on higher order and other systems are a function of characteristics external to the equipment being considered. For example, if a valve has a characteristic that leads to binding of the stem or poppet, the effect is a fixed position of the valve regardless of the mission description. Once the binding stem and the associated effect of fixed position have been identified by the designer, no further FMEA of the valve is required even if the mission changes. A very basic principle of documenting and updating FMEA's can be established by this consideration. If complete changes in performance and environmental requirements occur, more extensive revision of the FMEA is required.

## Dichotomy of FMEA - A Better Way

Providing work-arounds for failures not yet eliminated from the designs requires that the designer communicate the basic information concerning the problem to many different people. This information must be evaluated by different persons with regard to their basic responsibilities for the work-around. The systems engineer, test engineer, checkout engineer, mission planner, flight controller, and crew member use the information in a different context and in a specific sequence. Each has a definite concern that his is the latest information and reflects any late improvements or changes. However, it should be noted that only changes in effect on function of the equipment affect the work-around planning. Thus, the designer really needs to communicate design changes that cause a different functional effect or eliminate a functional effect. This over-simplifies consideration of fault isolation and maintenance, but emphasizes that FMEA's need not be revised completely because of configuration and design changes. Failure causes and effects can be treated like performance characteristics and documented accordingly. An FMEA package can be developed that is documented and procured as a part of each hardware procurement. For each case, the scope of the FMEA and the hardware being procured are the same. Successive combinations of the FMEA data from each procurement occurs as the hardware is assembled. This virtually eliminates the problem of finding someone qualified to perform an FMEA from part failure mode to mission effect. This outlined concept of FMEA documentation is considered to be a practical objective for FMEA activities. It is recognized that modifying procurement procedures dictates a progressive implementation of this concept.

## Practical Benefits from FMEA in Design Analysis

The benefits of FMEA in design analysis are related to evaluation and status of the selected design concept and evolution of the design through trade-off studies. Some consider this to be the sole application of the FMEA. Certainly, it is the basic application and requires special emphasis for failure-free operation of equipments. To accomplish this goal, the designer can isolate the characteristics or signatures of known and unknown failure causes. He can thus put more emphasis on failure mode analysis by adding these questions to his basic cause/effect reasoning.

Is the failure cause related to design margin?

Will increased design margin increase the failure-free life of the item?

Is the failure cause related to dimensional tolerances?

Can the manufacturing process be modified to eliminate the tolerance buildup?

Is the failure cause related to human error in any part of the fabrication cycle?

Is the failure cause related to environmental stresses?

Can a practical set of environmental limits preclude failure?

Is the failure cause related to parametric variations across interfaces?

Can more specific interface control preclude the failure?

Is the failure cause related to contamination?

Can the contamination sources be identified and controlled?

Is the failure cause related to material compatibilities?

Does the failure mode warrant special testing of this component?

Other questions of this type can be used to expand the FMEA thought process and to increase the benefits of the analysis. Benefits from reliability analyses of other types also can be increased by this constant awareness of failure-free operation as a way of life. Every failure has a cause and we can possibly recognize the signature before

we can understand the exact cause. Every bit of knowledge gained in this reasoning process adds more items to the designer's action item list. This early identification of problems also simplifies the task of providing interim corrective action.

## Practical Benefits of FMEA in Mission Contingency Planning

If equipment FMEA's are documented and procured along with each element of hardware, the data are combined in such a way that intermediate level data are available to the mission planner for identifying possible failures. The FMEA benefits the planner by relieving him of repeating cause/effect reasoning that the designer has already accomplished. The mission planner can then work from the top down, relating several unique failure causes to their common effect. In most cases, if the planner can isolate subsystem or assembly effects or failure cause signatures, he need not concern himself with individual equipment FMEA's. The designer has little need to repeat his analysis in total if the changes do not add or delete effects to the equipment as a unit. The mission planner must repeat his top-down analysis for each mission concept. He is greatly aided in this analysis if FMEA data exists at several assembly levels. If he must look each time at every individual component, his FMEA thought process becomes very time consuming. Thus, one more benefit of small "packages" of FMEA data is to free the planner from extensive overlapping analysis. If the FMEA's are prepared during the early design phases and maintained throughout the program, they will provide a clear listing of risk elements in the hardware. This enables engineering and management to concentrate on the hardware that may endanger the program. If the danger cannot be completely removed, it still may be possible to develop alternate procedures, warning devices, and escape systems to minimize the failure effects. The management system of Single Failure Point listings requires a separate discussion, but must be considered an important control for failure-free operation.

## A Summary of FMEA Basics

The FMEA is a cause/effect thought process essential to design, system, and functional analysis.

The FMEA data are as unique to equipment as its performance characteristics.

If FMEA's are performed for individual hardware packages, the data are available when the test engineer, mission planner, checkout engineer, and maintenance engineer need it.

The need to revise FMEA's usually is based on changes in the upper level system application.

The effectiveness of the FMEA is measured by the accuracy, completeness of coverage, the timeliness in which it is made available to its users, and the systematic progression of thinkers from designer (equipment level) to mission analysts (system and module level).

The FMEA aids the achievement of failure-free operation. Failure cause signatures can be isolated to burn-in or wear-out phenomena by an extension of the cause/effect reasoning process.

The FMEA is a qualitative analysis. Determination of numerical assessments is also unique to each equipment level. Many of the advantages of making FMEA data a part of procurement packages also hold for assembling quantitative data.

Documentation of FMEA's should be designed to benefit the users and avoid unnecessary revision by the designer. This also protects the designer from predicting failure effects beyond the scope of his visibility.

## Other Reliability Analyses with Practical Benefits

The preceding discussion has dealt primarily with relating the benefits of FMEA activities to the requirements of Long Life Design. Similarly, other reliability analyses must receive the same progressive application. Applying the goal of failure-free operation, and the steps necessary to accomplish this goal, proper relative emphasis should be given to:

Design Specification Review

Parts Selection, Screening, and Application

Environmental Acceptance Testing

Mission Equivalent Testing Under Combined Environments

Maintainability

In-Flight Maintenance[2]

Failure Reporting and Corrective Action

Reliability Training

Special Handling for Critical Components

## Summary

The unique requirements of Long Life Designs require the careful application of all engineering disciplines aimed at failure-free operation. Reliability analysis techniques can be beneficial in supporting these disciplines. On the premise that every failure has a cause and that signatures for the cause can be isolated, the FMEA aids the designer in meeting his goal, as well as aiding in achieving program goals.

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2. Cohen, H., Long Duration Space Missions Through Use of In-Flight Maintenance, Annals of Reliability and Maintainability, #670622, 1967.

# INFLIGHT MAINTENANCE FOR SPACE STATIONS

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## Introduction

It will take more than spectacular breakthroughs in the state of the art to assure a high probability of success for long duration manned space missions. The goal can best be achieved by coupling the combined major technological advances - on all fronts - with the coordinated efforts and ingenuity of spacecraft design engineers and planners. Highly reliable parts applied with conservative stress levels and tested to eliminate incipient failures will certainly augment longevity. Testing will also make it possible to predict failure rates with greater confidence. Judicious use of piece part redundancy may also increase equipment life, but the best available forecasts indicate that these techniques alone will not assure the high probability of success needed for long duration missions. The system's design must permit the crew to correct malfunctions and to restore the system to operation. Inflight maintenance will enable space planners to make effective use of logistic vehicles and to reduce total spare parts requirements through interchangeability.

## Reliability for Long Duration Missions

To determine their effectiveness, let us examine the tools and the extent to which they can be used to assure a high degree of reliability for long duration manned missions.

### Derating

Electronic piece part failure rates can be reduced by operating them at lowered stress levels. However, the penalties associated with these lower levels must be considered in determining the point at which further reductions become ineffective. This implies that designers can quantify both the gains and the penalties of derating. In general, derating extends the Mean Time Between Failure (MTBF) of parts. However, if this characteristic is to be meaningful, one must be able to relate this improvement to some tangible factor. If a reduced failure rate due to derating can eliminate the need for redundancy or minimize the number of spares, the net weight savings of derating becomes a valid measure of effectiveness.

If we consider a typical electronic part, such as a fixed composition resistor, we find that there are derating curves available to show the effects of reduced stress level (wattage) and reduced part ambient temperature. Figure 1 shows such a curve.[1] To analyze the effect of reducing the wattage ratio (operating wattage divided by rated wattage), it will be useful to consider a typical or equivalent black box composed of many resistors. Let us assume that we have a black box consisting of 1000 such resistors.

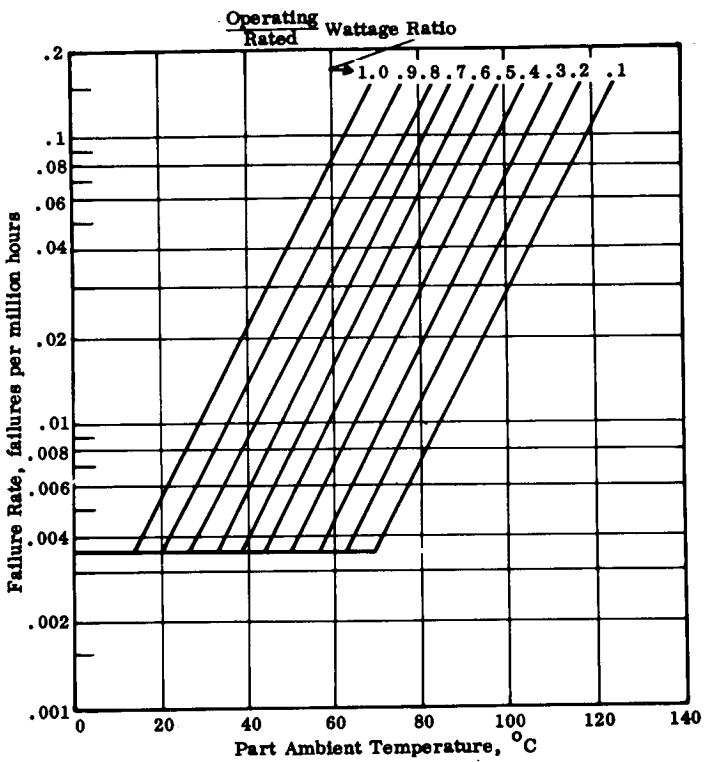



**Figure 1 Failures Rates (in Failures Per $10^6$ Hours) For Fixed Composition Resistors.**

To compute the number of spare black boxes required, we would need information about the mission length and the black boxes' apportioned reliability level. A mission time can be assumed (in this case 10,000 hours), but apportionment is a slightly more complex matter. If we had a mission reliability goal of, say 0.95, we would first have to consider all the elements that made up this goal. As a first cut we could divide it into three major elements:

- Residual reliability - the contribution to mission success of those equipments or items upon which we must rely for inherent reliability (structure, radiators, solar panels, etc.)

- Spares - the probability of not depleting any required spare during the mission
- Maintenance - the probability that all maintenance actions will be performed in the allowable downtime

If, for simplicity, we assign equal values to each of the above, our goal of not exhausting the supply of spares is the third root of 0.95, or 0.983. If the total system has 2000 such black boxes, the goal for one box would be the 2000th root of 0.983, or 0.99999145. That is, the probability that we do not deplete required spares for one black box would be approximately 0.99999, which is quite high. If we had a smaller system, say of only 170 black boxes, our goal for one black box would be 0.9999. The number of spares required are a function of the number of terms in the cumulative Poisson equation:

$$R = e^{-Kn\lambda t} \sum_{j=0}^{m} \frac{(Kn\lambda t)^j}{j!}$$

where
- $e$ = base of the Napierian logrithm, 2.718. . .
- $n$ = number of parts in black box (1000, in our example)
- $\lambda$ = failure rate of piece part, failure per $10^6$ hours
- $t$ = operating hours (10,000)
- $R$ = reliability
- $K$ = environmental factor (10)
- $m$ = number of spares required for reliability goal R

(A convenient technique for computing the required number of spares is shown in the Appendix)

Table 1 shows the number of spares for our black box example for various levels of derating for a constant temperature (60° C), for both the 0.9999 and the 0.99999 reliability goals.

Table I
Spares vs Derating Ratio

| Derating Ratio | Expected Failure | Spares (0.9999) | Spares (0.99999) |
|---|---|---|---|
| 1.0 | 8.0 | 20 | 22 |
| 0.9 | 5.3 | 15 | 17 |
| 0.8 | 3.6 | 12 | 14 |
| 0.7 | 2.3 | 9.5 | 11 |
| 0.6 | 1.5 | 7.5 | 9 |
| 0.5 | 1.0 | 6 | 7 |
| 0.4 | 0.68 | 5 | 6 |
| 0.3 | 0.45 | 4 | 5 |
| 0.2 | 0.35 | 3.5 | 4.5 |
| 0.1 | 0.35 | 3.5 | 4.5 |

In Figure 2, which presents spares versus derating curves, the 0.9999 reliability curve shows that spares are reduced from 20 to 6, when the operating-to-rated wattage ratio is derated by 50 percent (1.0 to 0.5).

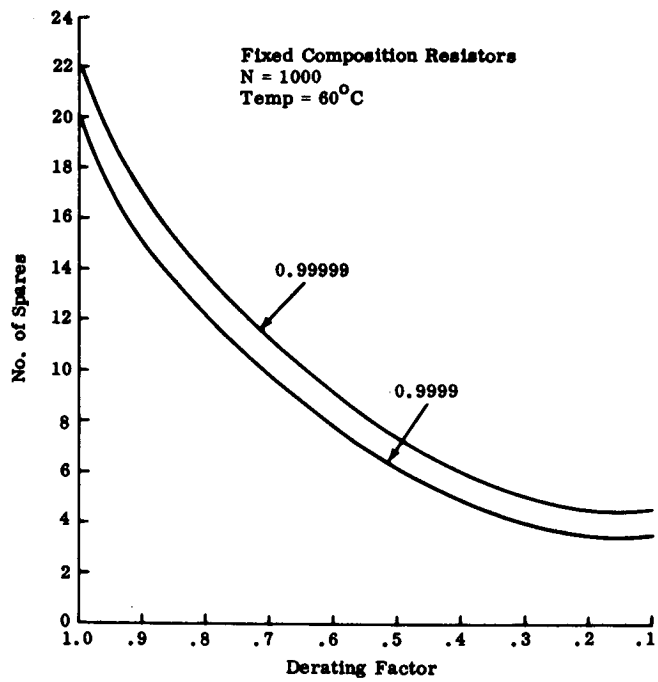



**Figure 2 Spares vs Derating Ratio**

An additional 50-percent derating (0.5 to 0.25) reduces spares from 6 to 3.75. Although Figure 2 is only a plot of Table I it is interesting to note the decreasing slope of the curves at lower derating factors.

If we consider spares and operating equipment, we find that the first 50-percent derating reduces equipment from 21 to 7, a 67-percent decrease. The second derating results in only a 32-percent reduction. The first derating is manifestly much more effective. However, the second can be worthwhile - depending upon the penalty paid for derating. Although derating is usually accomplished by using larger, heavier parts, it would be worthwhile in the 1.0 to 0.5 derating, even if the penalty approached a 200-percent increase in the equipment's original weight. The derating penalty would have to be under 47 percent for the 0.5 to 0.25 derating to be of benefit.

Table II shows how reduced operating temperature for the subject parts affects the quantity of spares required. Figure 3 shows the curves for spares versus part operating temperature. The curves show that the number of spares required decreases as a function of reduced stress (thermal). Parts operating temperature is reduced by improving internal thermal paths and by decreasing the temperature of the equipment cooling loop. Each method carries a weight penalty. In the first it is the increased weight of the black box; in the second, weight is added because larger space radiators are needed to reject heat, due to the lower temperature gradient between coolant and space. The decision as to how far to derate thermally depends on the penalties incurred versus gains.

## Testing

There is an oft repeated cliché that you cannot test reliability into a product: no matter how much testing you perform, it does nothing to improve the inherent reliability of the equipment under test. Reliability is built into a product, not tested into it. However the purpose of testing is to screen out defective parts that may result in failures, and if it is shown that the parts are without defects, then the equipment's reliability has been improved.

Testing also establishes the degree of confidence in reliability predictions and estimates by obtaining actual failure data distributions. These tests do nothing to improve the equipment's reliability, but they do improve confidence in reliability estimates. Both types of testing (screening and confidence) are required in any program where reliability is vital. The question is, "What kind of and how much testing should be incorporated in a program?" To be the most effective, a tradeoff estimate of the potential benefits of large scale test programs versus the costs of such programs must be made.

## Redundancy

Redundancy uses alternative methods to perform a designated function. This can take the form of adding one or more continuously operating identical equipments to the basic system so that the function continues as long as at least one equipment remains operative. This is parallel or operating redundancy. Or it can take the form of standby equipment, ready to be connected into the system should the primary equipment fail. This is standby redundancy.

Another technique often employed is to have a completely different equipment or system available to perform the primary equipment's function. This is functional redundancy. Parallel or standby redundancy is usually applied at the part and equipment level; functional redundancy is usually applied at the system level.

Table II

Spares vs Temperature Derating

| Part Ambient Temp.-Centigrade | Expected Failures | Spares (0.9999) | Spares (0.99999) |
|---|---|---|---|
| 100 | 16 | 31 | 35 |
| 90 | 8 | 19 | 22 |
| 80 | 4 | 12 | 14 |
| 70 | 2 | 8.5 | 10 |
| 60 | 1 | 6 | 7 |
| 50 | .5 | 4.3 | 5.3 |
| 40 | .35 | 3.5 | 4.5 |
| 30 | .35 | 3.5 | 4.5 |
| 20 | .35 | 3.5 | 4.5 |

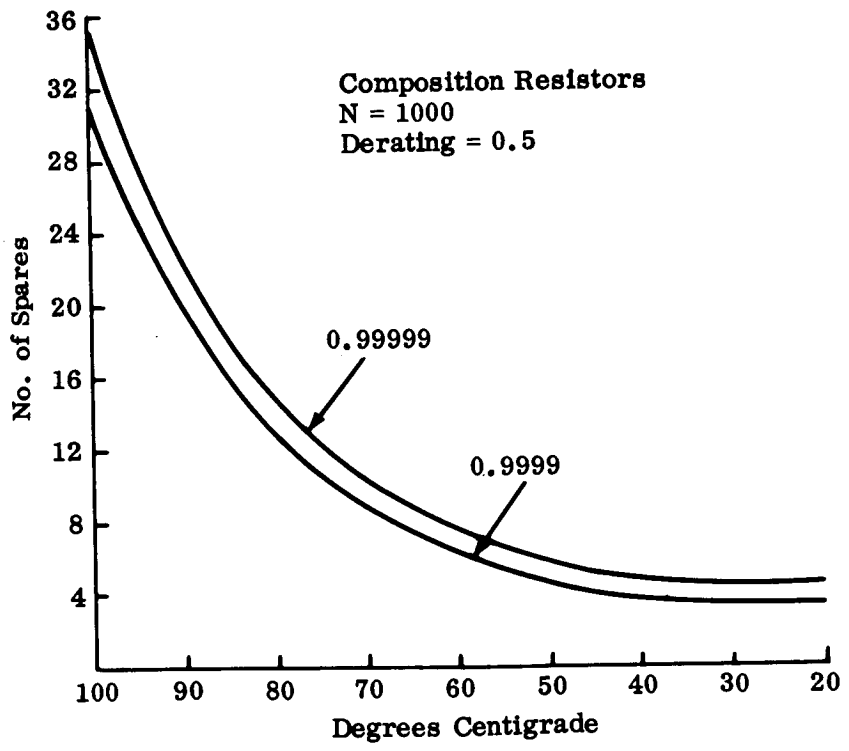



Figure 3 Spares vs Ambient Temperature

Let us examine the relationship between parallel redundancy and standby redundancy in its simplest terms, at the single, redundant unit level. Standby redundancy yields a greater improvement in reliability, as shown in Figure 4, because the standby unit has a lower exposure time to failure. It is not placed on line until a failure has occurred, whereas the parallel redundant unit is continuously operating.

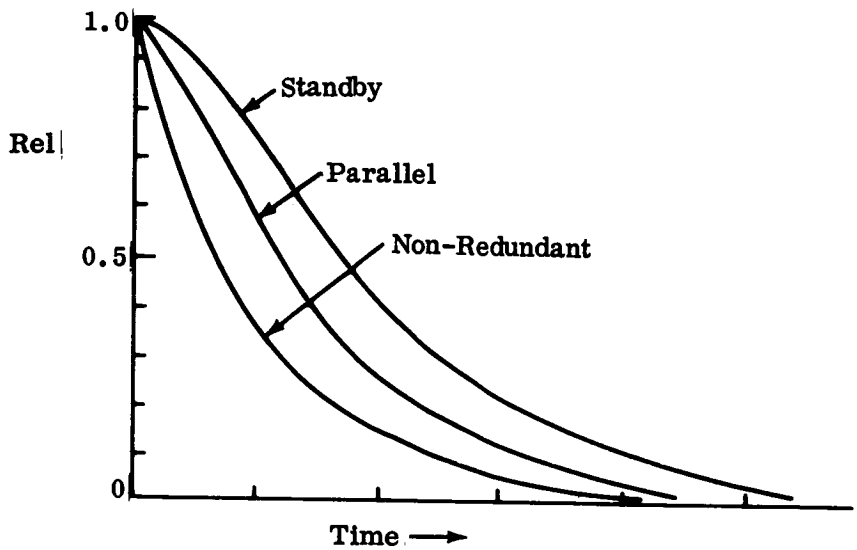



Figure 4 Single Unit Reliability

Another interesting characteristic is that, at some point, parallel redundancy has a weight versus mission time curve with an increasing slope, whereas standby redundancy has a continually decreasing slope, as shown in Figure 5. This implies that after the parallel redundancy curve reaches its point of inflection, further increases in mission time incur progressively larger increases in weight.

The practical application of redundancy merits attention. Parallel redundancy often results in a complex network of series-parallel units, since most parts have multi-failure modes. Also, the addition of series and/or parallel units often change a system's operating characteristics. Figure 6 shows the evolution of a typical parallel operating redundancy configuration. This technique is usually confined to electronic components at the piece part level, due to the relatively low weights involved.

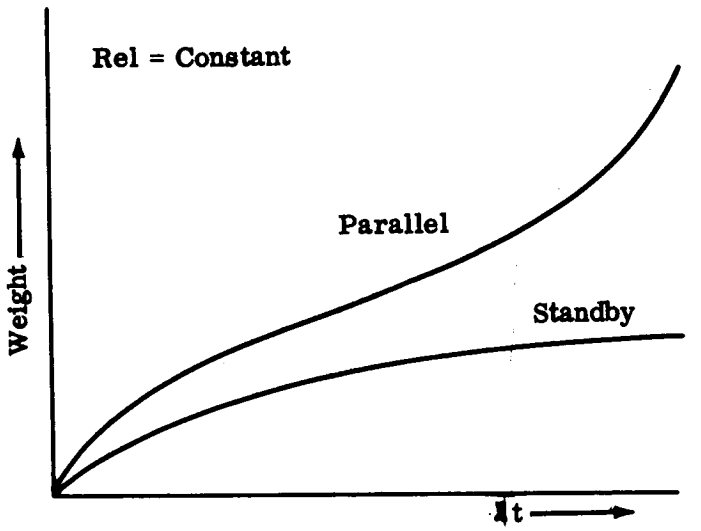
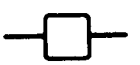
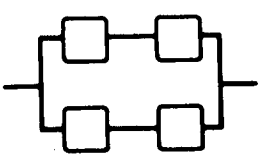


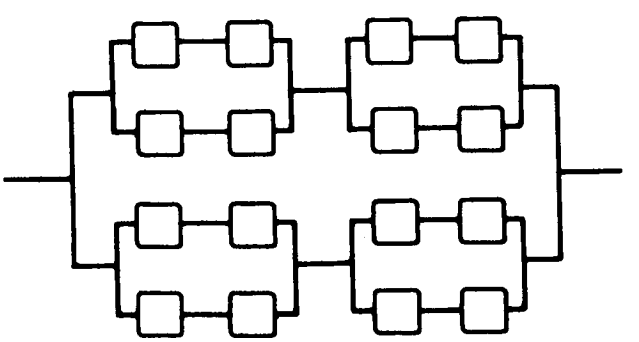

Figure 5 Theoretical Redundancy Curves



A. Basic Unit



B. Quad Redundancy



C. Series Parallel Redundancy

Figure 6 Parallel Redundancy

Standby redundancy requires switching to or changing to a standby unit when the primary unit fails. This may be accomplished by an automatic switch, a manually operated switch, or by manually replacing the defective unit.

Automatic switching can result in difficulties in long duration missions. The switching mechanism's reliability can often be the limiting factor in the total system reliability. A redundancy requirement for the switching mechanism may result as shown in Figure 7.

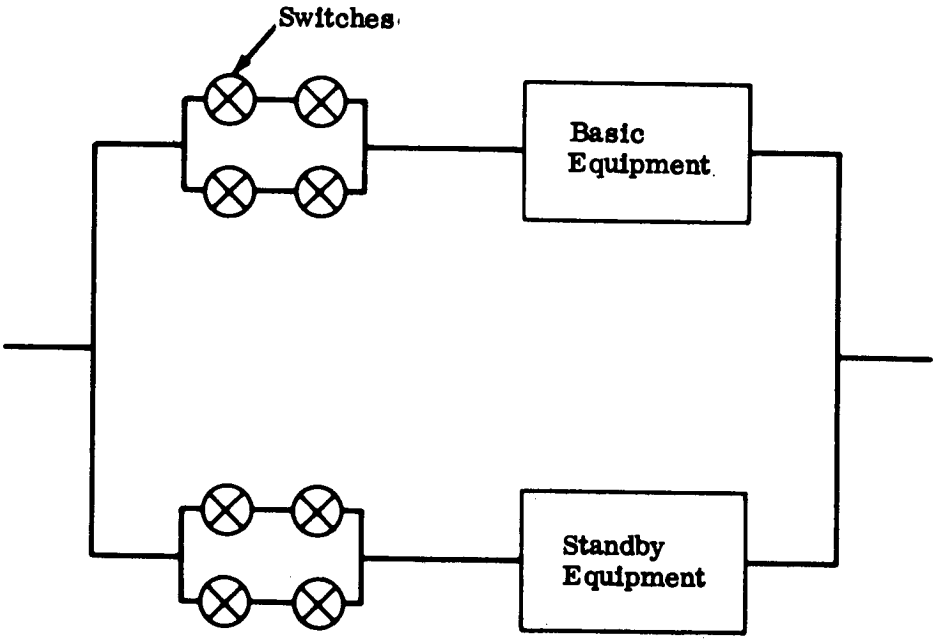



**Figure 7 Automatic Standby**

Manual switching is usually more reliable than automatic switching. But this is contingent upon mans' availability and the designed accessibility of the switch. Again, if man is there, and the unit itself is accessible, the replacement of the defective unit is a practical alternative. Such a system, designed to be maintained by the crew, is relatively simple. It consists of the basic system designed to facilitate replacement.

## Need for Trade-Offs

We have examined some of the techniques which the spacecraft designer can use to achieve reliability for long-duration manned missions. Increasing the inherent reliability of an equipment through derating, screen-type testing, and redundancy will minimize the number

of expected failures of spacecraft equipment. This will, of course, reduce the number of spares required to be carried aboard the spacecraft and hence reduce the astronauts' potential workload. We have seen that these techniques have limitations. Trade-off studies must, therefore, be performed to evaluate the potential value of the techniques in the light of the penalties (weight, complexity, etc) incurred. The reliability improvement techniques should be applied only to the point where benefits exceed costs.

## Maintainable Systems

A recent analysis of maintained versus non-maintained systems showed that a maintained system becomes more weight effective than the non-maintained, if the mission exceeds a few weeks.[2] It also indicated that maintainable systems could benefit from resupply, thereby significantly reducing onboard spares. Such a system could also take advantage of inherent interchangeability of parts and equipments, again significantly reducing the spares required to support the system.

### Logistic Resupply

If the mission profile includes ferry vehicles for crew rotation and the replenishment of expendables, and if the system is maintainable, the number of spares carried on the initial launch can be significantly reduced, thereby effecting a substantial weight and volume savings. Only those spares needed to ensure reliability between resupply intervals need be carried. The greater the resupply frequency, the fewer the spares. In essence, a five-year mission, resupplied every six months, would, for our purposes, be equivalent to 10 six-month missions. Relatively few spares will be needed to replenish those used in the period following the previous resupply. Figure 8 shows a typical curve of initial onboard spares weight versus resupply interval.[3] It also shows the weight of spares to be delivered. If the logistic vehicle were sized for this amount, it would be large enough only half the time. Therefore, Figure 8 also shows logistic vehicle spares weight curves for various levels of confidence. The system must, obviously, be designed for inflight maintenance, in order to take advantage of resupply.

### Interchangeability

Certain components such as some valves, switches, etc., have multiple application in most spacecraft subsystem designs. If the system is to be maintainable, the concept of pooled spares may be applied. This reduces the required spares significantly - enough to make it worthwhile for designers to investigate the potential for induced interchangeability of components. The challenge to the designer is to design an optimum system, with due consideration to the spare implications, rather than merely to design optimum equipment for a particular function. The penalties imposed by interchangeability may possibly be compensated by the advantages conferred through reduced spares requirements.

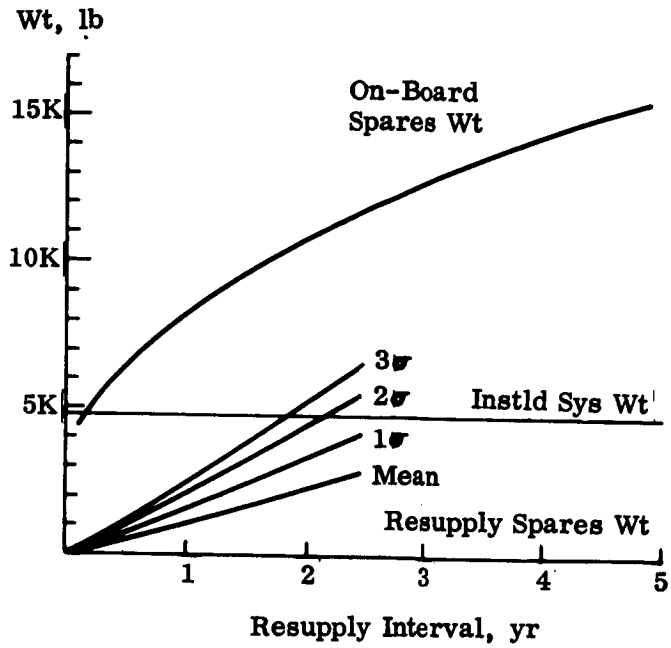


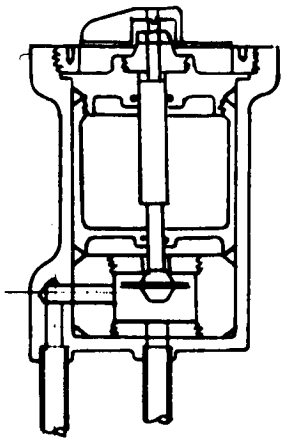

Figure 8 Spares Weight for Typical System

The greatest potential for interchangeability occurs at the piece part level. As an exercise, and using common piece parts, we conceptually designed a family of valves for a variety of functions. Selective combinations of piece parts makes it possible to provide a manually operated shut-off valve, a solenoid shut-off with manual override, a pressure regulator and a pressure relief valve, as shown in Figure 9. Its elements and the total requirements for constructing a set of valves for a specific system are shown in Figure 10.

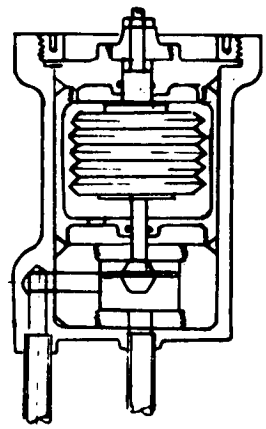
## Level of Maintenance

Once the decision is made to design a maintainable system, the next question is how to implement inflight maintenance. Inflight maintenance will generally consist of replacing malfunctioning equipment. How large can the replaceable element be? To what extent should replacement be made in a single maintenance action and how does the equipment's complexity influence this determination?

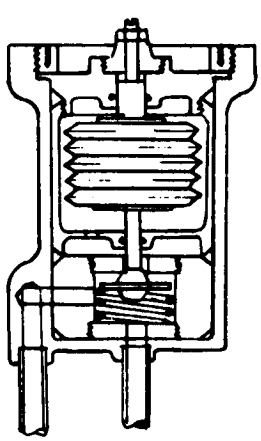
The choice of this level of maintenance is influenced by competing factors. From a spares point of view, the smaller the replaceable element, the lower the total spares required. This is due to two factors: the smaller the replaceable element, the less "good" equipment is thrown out of the system for each failure, and the greater the potential for interchangeability between elements.



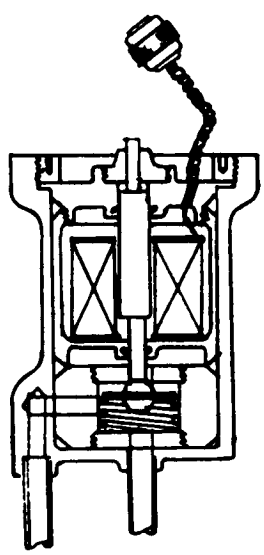
Manual
Shut-Off Valve



Relief Valve



Pressure Regulator



Solenoid Operated
Shut-Off Valve

Figure 9 Multi-Purpose Valve

**Valve Component Utilization**

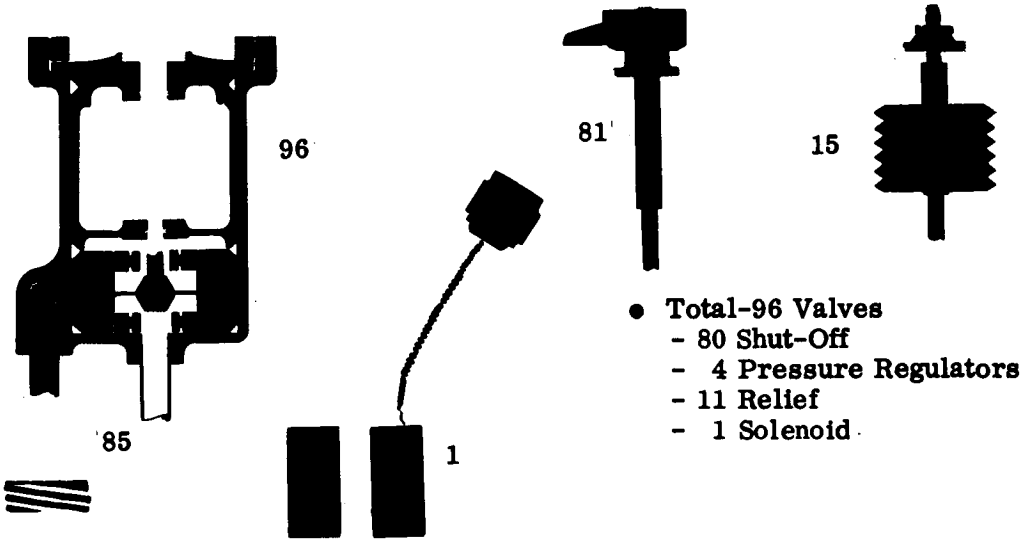



**Figure 10 Multi-Purpose Valve Components**

From a checkout point of view, the checkout system becomes very complex at a low (small) level of maintenance, and is relatively simple at a high level of maintenance. From a manpower point of view, more manhours and higher skill levels are required at the lower levels of maintenance. Figure 11 shows the relationship of these factors for various levels of maintenance.

How then does one select a level of maintenance? The task is to quantify the factors involved and to select the most effective level. The unit of measurement for each factor (manhours, pounds, etc.) must be weighted (normalized) to provide a common basis. One solution is to attempt to convert each factor to equivalent units expressed in dollars. This implies the ability to place a value on one pound and to assign a value for one manhour. There are also practical implications to be considered in choosing a level of maintenance. Good engineering judgement and design practice play important roles in these determinations. Sometimes it is possible to design an equipment that is maintainable at various levels. In this case, the decision as to which level is to be maintained is not constrained by the equipment design and can be made on the basis of other factors under consideration.[4] Figure 12 shows a modular valve designed for a $CO_2$ reduction unit. The total module consists of seven valves and some sensors and indicators. The module can be replaced by disengaging several quick disconnects and by loosening several fasteners. This would be considered assembly level of maintenance. If desired, the individual valves can be replaced; they are designed as plug-in modules of the assembly. This would be considered subassembly level of maintenance. Finally, the valves themselves could be repaired by replacing the O-ring seals. This would be piece part level of maintenance.

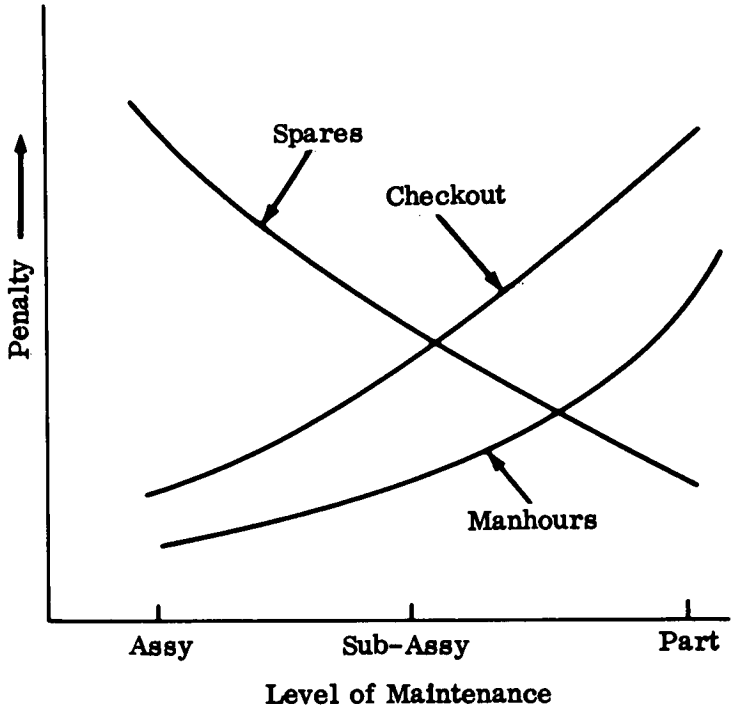



Figure 11 Level of Maintenance Factors

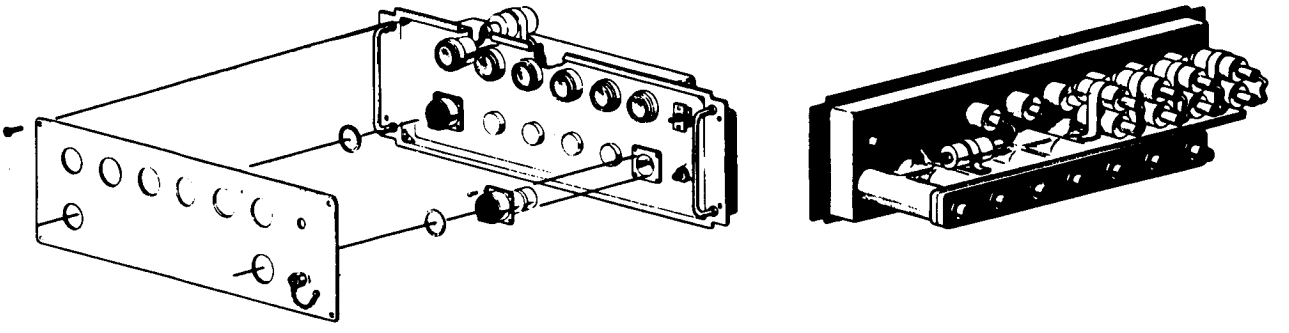


Figure 12 Maintainable Valve Module

## Conclusion

The path to reliable, long duration, manned space missions is clearly defined. We must take advantage of gains in all technological areas to enhance inherent reliability of equipment, but we must also design for inflight maintenance to provide the capability to maintain continuing operations. The immediate task is to develop techniques for implementing maintainability. The expected modes and frequency of failure of each equipment must be predicted, to determine candidate levels of modularization and interchangeability. It is necessary, too, that our design provide for simple and "Murphy-proof" maintenance tasks, to minimize the possiblity of man-induced failures. New concepts for hardware design must be developed, and we must test, evaluate and demonstrate mans' ability to restore malfunctioning systems. We can gain confidence in our ability to maintain systems through an extensive program using mockups, simulations, and early space flight experiments.

## References


1. "Reliability Stress and Failure Rate Data for Electronic Equipment", MIL-HDBK-217A

2. Hodge, P. and Frumkin, B.; "Redundancy vs. Maintainability; Some Conclusions on the Crossover for Manned Missions", Second Space Maintenance and Extravehicular Activities Conference, August, 1968, Las Vegas, Nevada

3. Frumkin, B.; "System Effectiveness Through Interchangeability", AIAA 4th Annual Meeting and Technical Display, October 1967, Anaheim, California

4. Feindler, K., and Brudnicki, M.; "The Impact of Maintainability on EC/LS Design", Second National Conference on Space Maintenance and Extravehicular Activities, August 1968, Las Vegas, Nevada

## APPENDIX

### Calculation of Spares

Calculation of spares is best done by a computer programmed to determine the best mix of spare parts to attain a given reliability at the least weight. Such a computer program uses an iterative technique to select each spare that offers the highest reliability gain per pound of spare. However, such a technique is somewhat cumbersome, if the purpose is to perform tradeoffs of alternatives rather than to schedule spares. This Appendix describes a simple technique for quickly estimating the effects of various configuration alternatives on spares requirements. The technique is intended to facilitate tradeoff studies.

Calculation of spare requires three basic elements:

- A reliability apportionment for the part to be spared.
- A calculation of the expected number of failures of the part to be spared during the mission or the resupply interval.
- A determination of the number of Poisson terms required to reach the apportioned reliability level, when the expected number of failures is given.

The first approximation to a reliability apportionment may be obtained by taking successive roots of the system reliability goal. Given a system goal for not depleting spares, the first calculation would be to find the goal for an individual resupply period. For example, for a five-year mission, resupplied at six-month intervals, we would take the 10th root of the system goal. The next step would be to apportion this quantity down to a particular subsystem by taking the 8th root, in the case of eight subsystems. And, finally, to apportion down to an equipment level by taking a root equivalent to the number of different equipments in the subsystem. Figure 13 shows a graphical means for obtaining roots. Starting with a goal or apportionment at the left hand scale, follow the diagonal line until it intersects the vertical line representing the root shown on the lower horizontal scale. Read horizontally to the left to obtain the answer. This process may be repeated until an apportionment is obtained for the part under study.

The expected number of failures is merely the product of the failure rate for an individual part, $\lambda$, the number of similar parts in the subsystem, N, and the operating hours in the resupply interval, t, or, $N \lambda t$.

The number of spares required, which is the number of Poisson terms, less one, may be determined from the graph as shown in Figure 14. The intersection of a horizontal line extending from the expected failures at the left-hand margin, and the vertical line extending from the

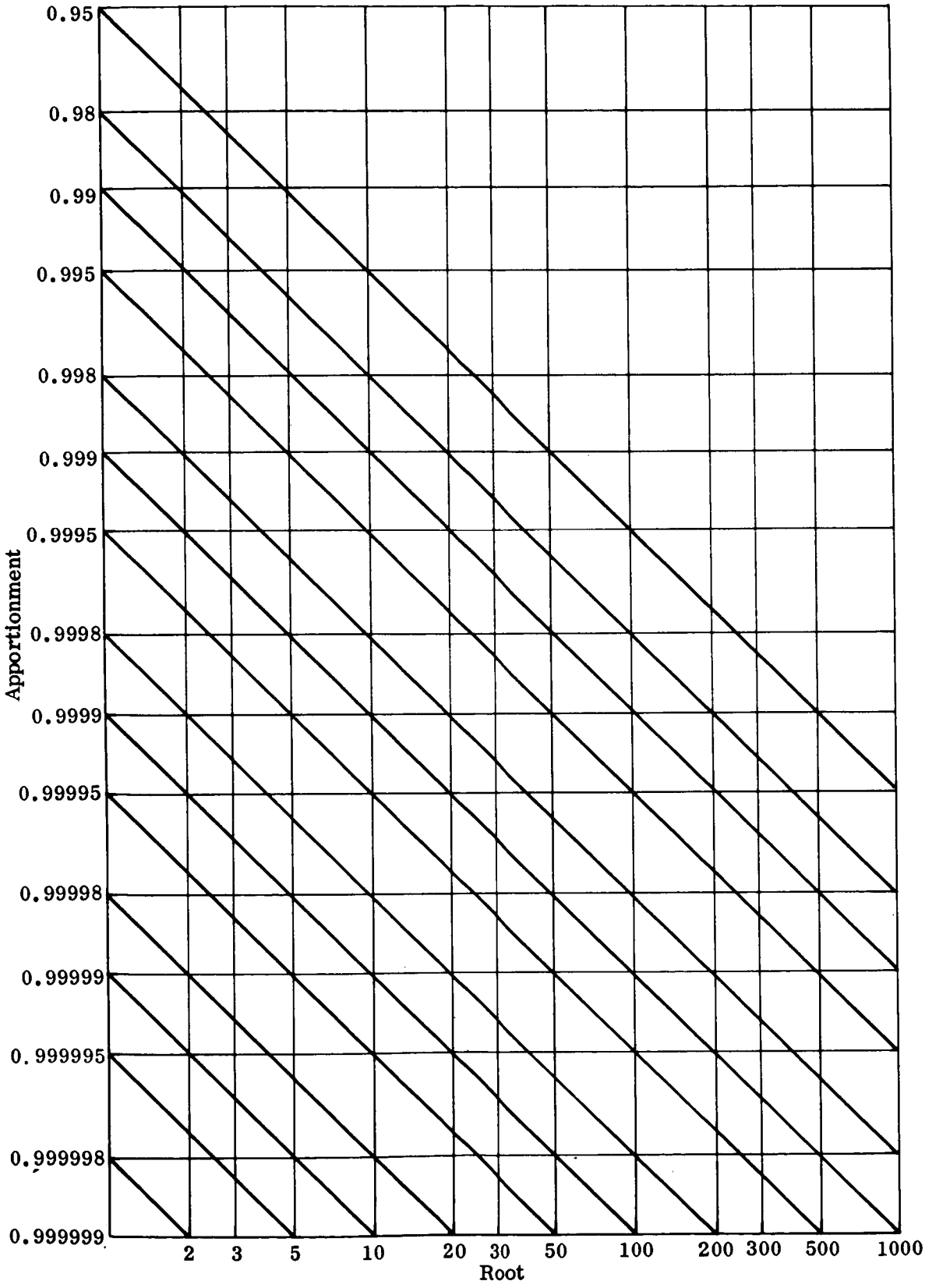



Figure 13 Roots of Fractional Numbers

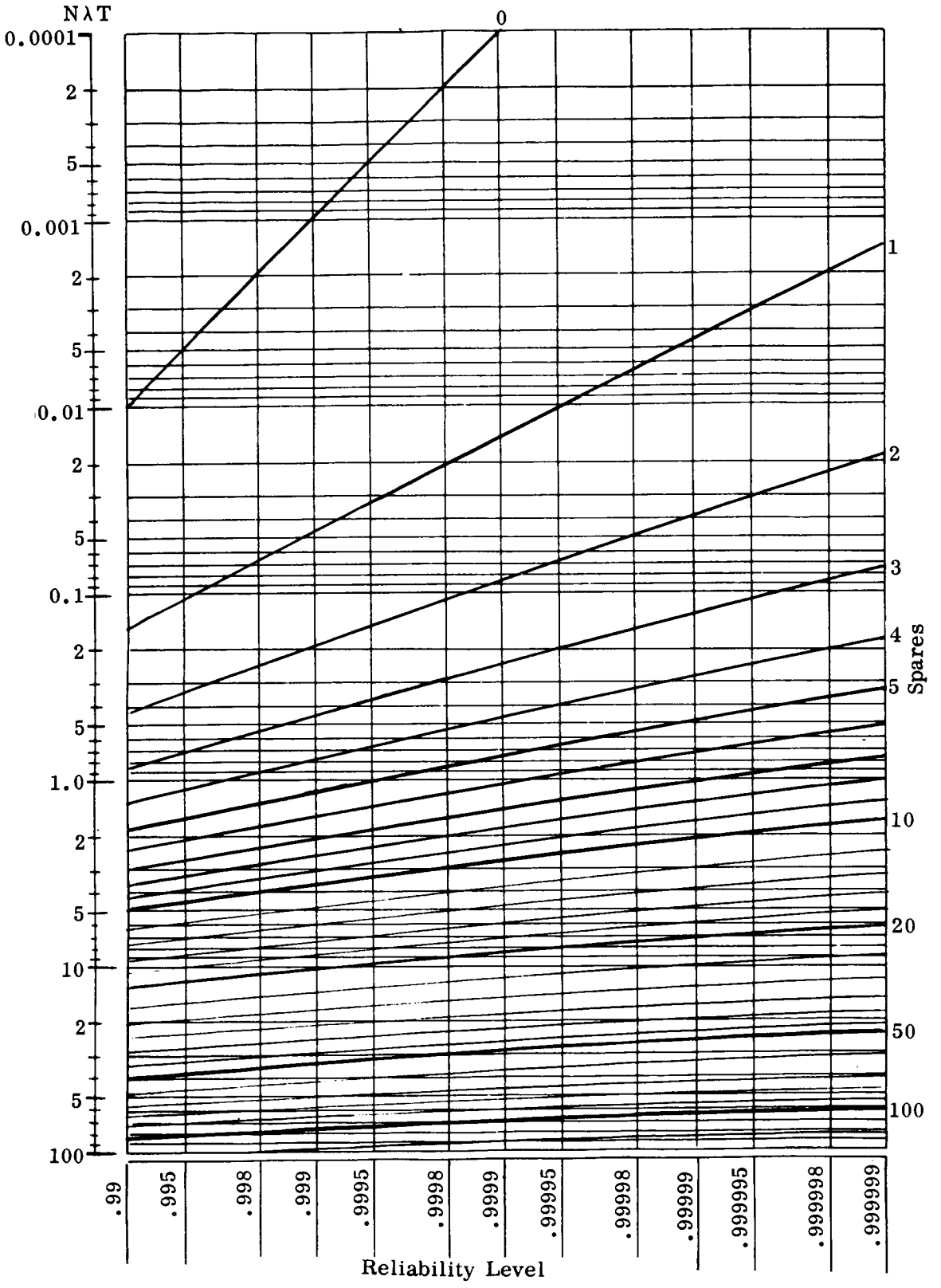



Figure 14 Number of Spares Chart

horizontal scale indicating the apportioned reliability goal, indicates the number of spares required. Of course, when the intersecting point falls between two lines, it will be necessary to read the next higher number. Figure 15 shows these elements combined into a special purpose nomograph designed to calculate spares required for a five-year space station mission.

The example shown in the Nomograph assumes the following situation:

- Resupply interval: 6 months (4380 hr)
- System goal: 0.95
- No. of subsystems: 8
- No. of equipment types in subsystem: 20
- No. of interchangeable equipments: 8
- Failure rate of equipment: 0.000005 failures per hour (5 failures per million hours; MTBF = 200,000 hours)

To determine the number of spares of a given type to be carried aboard the space station using the nomograph (Figure 15):

1. Determine equipment goal (probability of not running out of spares for this particular type of equipment):

    a. Choose appropriate resupply interval - "Time" scale on Chart "A" (6 months).

    b. Follow line to top of Chart "B".

    c. Enter Chart "B" at System Goal - Probability of not running out of any required spares for total five year mission (0.95).

    d. Follow diagonal on Chart "B" until it intersects with resupply interval vertical line. Continue horizontally to Chart "C".

    e. Follow diagonal on Chart "C" until it intersects the "Number of Subsystems" vertical line. (8). Continue horizontally to Chart "D".

    f. Follow diagonal on Chart "D" until it intersects the "Number of Equipment Types in Subsystem" vertical line, (20). Continue horizontally to Chart "E".

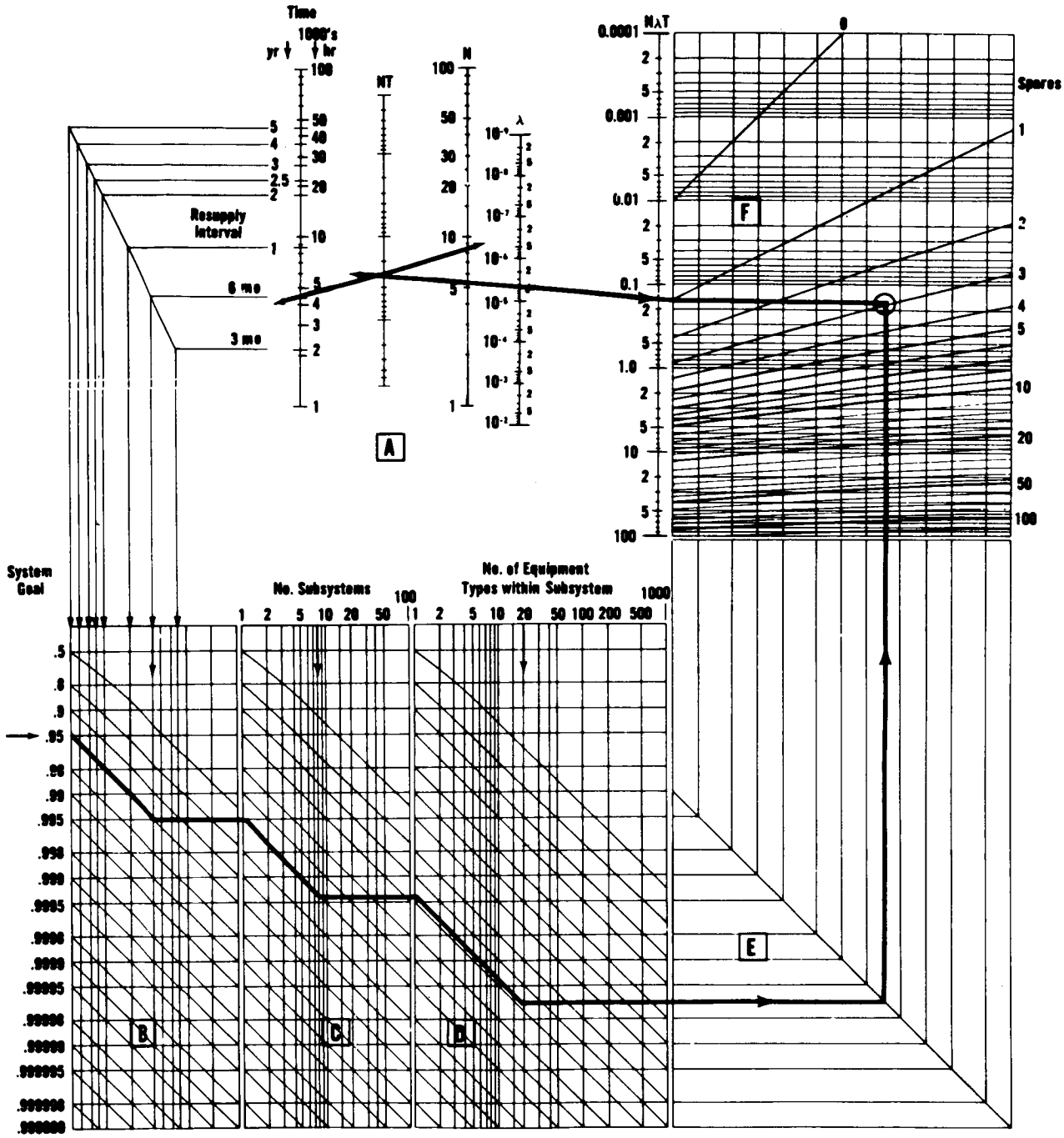



Figure 15 Space Station Spares Selector Nomograph

g. Continue horizontally on Chart "E" until the diagonal line is reached, then continue vertically upward to Chart "F". This is the equipment goal.

2. Determine the expected number of failures of the equipment:

   a. Place a straight edge between the "Time" scale, and the number of similar equipments scale, "N" on Chart "A". (6 months and 8 equipments). Find point of intersect with "NT" scale.

   b. Place straight edge between point of intersect on the "NT" scale and the failure rate on the " $\lambda$ " scale, ($5 \times 10^{-6}$). Project this line to the "N $\lambda$ T" scale. This is the expected number of failures.

3. Determine the number of spares required:

   a. Project expected number of failures (2b) horizontally across Chart "F".

   b. Project equipment goal (1g) vertically upward on Chart "F" until it intersects expected number of failures (3a).

   c. Number of spares is determined by first diagonal line below the point of intersection (3b). (Three spares required).

# UTILIZATION OF PRIME EQUIPMENT FOR IN-FLIGHT TESTING

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## Introduction

Manned space programs, from Gemini to Apollo, have vividly demonstrated that the astronauts can cope with many types of in-flight equipment failures. With the greatly increased mission durations and the added complexity of equipment for the experiments, man's ability will become even more essential for the Apollo Applications Cluster Program. Expanded in-flight testing facilities can significantly enhance the ability of the flight crew to detect and isolate failures so that the required corrective action can be taken.

In recent years, airborne weapon systems have also required increasingly thorough in-flight test capability. These requirements have resulted in built-in-test (BIT) designs that are sufficiently thorough and accurate to justify complete elimination of external test equipment for planeside maintenance. The test design techniques developed for these airborne systems should be equally applicable to the in-flight test design problem for manned spacecraft.

This paper outlines the most important of these techniques and offers several illustrations of their application to actual systems. It is important to recognize the difference between in-flight testing as it is usually implemented and the extremely thorough testing that can be accomplished by use of the techniques described in the following discussion.

Conventional in-flight testing is usually limited to some combination of:

1. Status monitoring (mode indicators, "ready" lights, etc.).
2. Qualitative end-to-end system tests capable of detecting gross failures but poorly suited to detection of subtle failures or to isolation of failures.

3. Monitoring of outputs during normal operation without significant capability for control of test conditions.

Airborne system tests of the type described in this paper have test thoroughness capabilities in excess of 98 percent and can isolate more than 90 percent of the detected failures to a single replaceable assembly. Performance of tests capable of this level of thoroughness requires that the input conditions be carefully controlled and that test results be quantitatively evaluated. These requirements further imply that system functions be capable of individual evaluation.

## Comparison to Ground Testing

From the foregoing, it might be inferred that in-flight equipment approximating the complexity of the ground test equipment is required. This is clearly impossible for a number of reasons. Weight and volume constraints alone would prohibit any attempt to duplicate the ground test functions for in-flight test. In-flight test equipment must be measured in inches and ounces rather than in feet and pounds.

Human factors considerations present equally pressing demands on the in-flight test designer's ingenuity. The in-flight test must be conducted by a flight crew whose primary attention is devoted to non-test tasks such as flight management and performance of experiments. Test times must be short and tests must not be unduly demanding on the crew's attention. Also, the working environment -- restricted space, zero gravity, limited access to equipment, and the need for uninterrupted operation of certain equipment -- must be considered by the test designer. These constraints are in sharp contrast to the conditions under which the ground-test crew works during ground testing.

The flight crew's attitude toward the in-flight test is of utmost importance to the in-flight test designer. Only if the flight crew willingly accepts the in-flight test can the objectives be accomplished. In addition to the need for short test time and minimum distraction from other duties discussed above, experience with airborne in-flight tests has shown that the form of the test readout is a critical factor. Pilots are most enthusiastic about in-flight tests that present results in directly pertinent terms. For example, a radar test readout of relative detection range capability is most acceptable. Once a defect has been established, pilots are interested in the reason for the problem (e.g., low transmitter power or degraded receiver sensitivity).

The in-flight test designer has two powerful tools at his command to help in resolution of these seemingly conflicting demands. Recognition of the differences in the objectives of in-flight testing as compared to those of ground testing can lead to much simpler tests for the in-flight case. The second tool, which is even more useful, is the use of prime equipment as test equipment to test other portions of the system.

## Test Objectives

The in-flight test designer must clearly understand the objectives of his test. For ground testing, there are numerous objectives that must be satisfied. The design engineer's tests are primarily intended to assist him in evaluating and optimizing his design. At least in the early stages, it is not safe to assume that any function is being performed as intended. Every facet of the design must be studied and optimized.

As the design matures and the system integration progresses, the objectives of the test activities shift to the problems of subsystem interface evaluation. Again, the primary purpose is to evaluate designs and to insure that all elements work properly together.

Manufacturing testing of flight hardware, environmental testing, and other test activities are devoted to verifying that the design already established is being recreated in accordance with the intent of the designers and that the resulting equipment still performs its function under adverse conditions.

Compare the diverse objectives described above with the much more limited objectives of in-flight testing. In the in-flight case, the equipment has already been carefully tested to verify the design and manufacturing processes. It is only required to detect failures that have subsequently occurred and to isolate them to a level suitable for corrective action by the flight crew. It is reasonable to assume that failures occur singly or, at worst, in cause-and-effect sets of single primary failures accompanied by associated secondary failures.

An in-flight test designed to satisfy (and limited to) the objectives as described above will be significantly simpler than the corresponding ground tests that must satisfy a much more extensive set of objectives. This simplicity can result in reduction of both test hardware requirements and test procedure complexity.

To illustrate the simplification that can result from this concept, consider a signal processor consisting of a series of bandpass and band-elimination crystal filters, interconnected by amplifiers and frequency translation mixers.

Design and manufacturing tests for this processor require stage-by-stage evaluation of characteristics such as passband insertion loss, skirt shape, input and output impedances, and spurious responses in the elimination bands. Many of these tests are extremely difficult and time-consuming and require the use of sophisticated test equipment and test methods.

However, if it is presumed that the processor was once thoroughly tested but that a failure has occurred, a greatly simplified test can be used. In fact, any such failure would result in degradation of the response in the passbands of the overall processor. Therefore, the in-flight test consists simply of measuring the processor's response to a leveled signal swept through its passband. The signals are generated and measured at reasonable amplitudes and no interstage tests are required. Such a test would be inadequate for either the design test or the manufacturing test, but it provides 100-percent test thoroughness for the purposes of the in-flight test.

## Self-Test Theory

The concept of self-test is one of the most useful techniques available to the in-flight test designer. In its simplest terms, self-test means the use of prime equipment hardware and functions to test other portions of the prime equipment.

A total system can be thought of as an inventory of test functions. Figure 1 illustrates a few of the test functions that are found in typical prime equipment. When a particular function is needed for a test, it can frequently be found elsewhere in the system.

In most practical situations, a combination of self-test and additional built-in-test equipment (BITE) is needed. The hardware relationships that are found in a typical system utilizing the self-test concept are illustrated in Figure 2. Some means is needed to establish the extent to which self-test is used to reduce requirements for BITE. A figure of merit called the self-test-ratio can be used to evaluate the relative benefits of self-test.

Suppose that the in-flight test penalties of particular interest are weight, cost, and power consumption. The cost ($C_B$), weight ($W_B$), and power ($P_B$) of the hardware included in the system solely because of the BITE requirement are estimated. Then, it is assumed that exactly the same test capability is provided entirely in the form of additional BITE hardware without the benefit of self-test. An estimate is made of the cost ($C_X$), weight ($W_X$), and power ($P_X$) that would result from this course of action. Finally, the relative importance of each of the penalties is determined. These weighting factors are denoted $R_C$, $R_W$, and $R_P$ for cost, weight, and power, respectively, and are chosen in such a way that

$$R_C + R_W + R_P = 1.$$

Then the self-test ratio (S) for the design being considered is

$$S = R_C \frac{C_X - C_B}{C_X} + R_W \frac{W_X - W_B}{W_X} + R_P \frac{P_X - P_B}{P_X} .$$

It is important to note that the penalties attributable to the self-test hardware are not used in this model. The reason is that a function needed for in-flight test may exist in the prime hardware in a form that is more expensive (etc.) than would be required if only the test requirement were applicable. Suppose, for example, that a signal of 30-MHz $\pm$ 5-MHz is needed. This could easily be provided by a simple low-cost LC oscillator. If an oven-controlled 31-MHz crystal oscillator, available in the prime system, is used for self-test, only the cost, weight, and volume of the LC oscillator is of significance in computing the self-test ratio.

An example will illustrate the use of the self-test ratio. Assume that cost, weight, and power of the hardware included in the system only for in-flight test are \$6,000, 2 pounds, and 3 watts, respectively. Also, assume that the penalties that would be attributed to in-flight test if self-test were not used would be \$42,000, 10 pounds, and 24 watts. Finally, the relative importance of cost, weight, and power in this system are estimated to be 0.6, 0.3, and 0.1, respectively. Then, the self-test ratio is

$$\begin{aligned} S &= (0.6) \frac{42{,}000 - 6{,}000}{42{,}000} + (0.3) \frac{10 - 2}{10} + (0.1) \frac{24 - 3}{24} \\ &= (0.6)\ (0.86) + (0.3)\ (0.80) + (0.1)\ (0.88) \\ &= 0.52 + 0.24 + 0.09 \\ &= 0.85. \end{aligned}$$

This calculation shows that self-test resulted in a cost savings of 86 percent ($36,000/system), a weight savings of 80 percent (8 pounds), and a power savings of 88 percent (21 watts), for a total penalty reduction of 85 percent.

Cost, weight, and power were selected simply for the purpose of illustration. In practice, the self-test ratio must be constructed to satisfy the needs of each system to which it is applied. Other penalties that might be included are failure rate, component count, volume, and cooling fluid flow. Generally, it is best to select only the few most critical factors. In the above example, since the relative importance of power consumption was judged to be only 0.1, it could have been eliminated without significant effect on the model.

Considerable judgement is needed to construct and apply this model. Some penalties are not obvious or are difficult to estimate accurately. The close integration of in-flight test into the prime hardware makes it particularly hard to assign values to items such as structure weight and cooling requirements. It is more important to apply consistent objective methods to the evaluation of a few key elements than to construct a perfect model that is difficult to use.

It is often possible to replace a test having a low self-test ratio with another less obvious test that makes more effective use of prime equipment functions for test purposes. Although this may result in a BIT test that is conceptually more complex, the net in-flight test penalty will be reduced if the self-test-ratio can be raised by this technique.

Another test technique has been of particular value in simplifying fault-isolation tests. This concept is described in the Naval Air System Command Document AR-8 as the diagnostic response test. AR-8 requires the use of "logical deduction and diagnostic analysis ---- to achieve minimization/optimization of test access ---". Application of this concept is even more essential for in-flight test. While the idea of diagnostic response testing is far from new, the extent to which it should be applied to in-flight test is much greater than is normally suitable for more conventional test procedures.

Diagnostic testing permits isolation of a fault to a function without testing that function individually. As an oversimplified example, suppose that functions A and B in Figure 3 can be tested as a pair and that the test shows that either A or B has failed. If a second test, involving A and C passes, it can be reasonably concluded that the fault lies in B. In practice, there are usually many more possibilities and combinations to be analyzed. The diagnostic test method is particularly suitable for self-testing. In Figure 3, it is obvious that separate tests of the two prime functions will establish the status of the self-test function without need for a separate test of that function alone.

## Self-Test Examples

Satisfaction of the requirement for thorough in-flight testing without need for excessively heavy or voluminous built-in-test equipment by use of the self-test concept can be illustrated by some typical examples of self-test as applied to actual systems.

### Self-Test with Computers

If the prime system includes a digital computer (which is not uncommon in this era of microelectronics and digital communications and data processing), powerful self-test capability is available. A computer, when used for in-flight test, can perform countless functions as suggested in Figure 4. With a special program, it can initiate test steps, measure test results, control signal simulation functions, and evaluate data. The memory and computational capability offer valuable opportunities for diagnostic response testing that might otherwise be too complex.

With the addition of an analog-to-digital converter, which may also be available in the prime system, the computer becomes a digital voltmeter. Direct-current or low-frequency ac voltages can be measured to any desired degree of precision within the limits of the digital resolution. The frequency range can be extended by the addition of an appropriate detector at the A-to-D converter input. An AGC voltage is an excellent self-test accessory to the computer when the RF or IF signal to be measured can be applied to the input of the AGC amplifier. Addition of a narrowband filter, a mixer, and a swept frequency source converts the computer into an automatic spectrum analyzer. The computer can control the swept frequency source by addition of a D-to-A converter. A discriminator provides some capability for phase and frequency measurement. In the system from which these examples were taken, this peripheral signal processing equipment was available in the prime system.

The computational capability and random-access memory available in a digital computer provides significant advantages over conventional step-by-step predetermined test procedures. To continue the example mentioned in an earlier section, consider the test of a search radar. While it is possible to separately evaluate transmitter power and receiver sensitivity, neither of these parameters is the primary concern of a pilot. The detection range capability of the system is a more directly applicable parameter. Thus, in evaluating mode availability, a computation of relative detection range using power and sensitivity test results will be more valuable to the flight crew.

Computational capability is also useful when absolute test limits are difficult to establish. Sometimes, the change of a measured value is of more significance than the absolute value. This is especially true when the performance of one function in a series of functions must be evaluated. As an example, consider a receiver sensitivity test. Suppose that the receiver includes a parametric amplifier than can be switched in or out of the system to adapt to changing conditions. In a computer-controlled test, the receiver noise figure can be measured with the paramp in and measured again with the paramp out. The computer, by using its memory and computational capability, can evaluate paramp performance from these two measurements. A test that could accomplish the same task by comparing each measurement with a predetermined limit would be difficult to implement because the variability of the basic receiver performance could mask paramp noise figure variations. The extremely low signal levels preclude any thought of direct evaluation of the paramp alone. This example is a variation of the diagnostic response test concept described earlier.

The existence of a computer in the prime system implies numerous digital interfaces between the computer and the balance of the system. Each of these interfaces (both input and output) is an opportunity for a test control or measurement function that has a self-test-ratio of 100 percent. These digital interfaces can often be used differently for test than for normal operation simply by provision of special test programs.

## Self-Test with Controls and Displays

The in-flight test designer must solve two problems relating to controls and displays. A highly automated test is not well suited for equipment that has numerous switches, controls, and readouts. Manual participation inevitably slows the test process, offers opportunity for error, and involves human judgement which is highly variable. When diagnostic response testing is performed with internal programmed logic, it is difficult to incorporate data resulting from manual observations.

The second problem is an even more severe constraint on the test designer's activities. There is seldom enough cockpit panel space available for even the most critical operational functions. In such an environment, in-flight test must take second place, and is usually assigned a few square inches suitable for only the most rudimentary control and display functions. In the face of such limited facilities, the designer must find ways to display identification of failed hardware, mode availability status, and other test results. Also, when manual test participation is involved, cues and instructions to the operator should also be displayed.

Thus, this two-fold problem involves a difficult-to-test subsystem and a limitation on needed in-flight test hardware. Such a combination is typical of situations to which self-test can be successfully applied. A study of the operational controls and displays will usually reveal a method for presenting test data to the operator without need for special readout equipment. The test readout problem is one of presenting large amounts of data, and most system displays are well suited to high-data-rate applications.

Two examples will illustrate the flexibility that can be obtained in this manner. Figure 5 shows an electro-optical situation display. In normal operation, this display projects a map selected by the computer from a library of slides stored in the system on the display screen. Superimposed on the map display are projections of computer-generated and positioned symbols representing aircraft locations, navigation points, and other points of interest.

To the in-flight test designer, such a device offers countless opportunities for display of test instructions and results. In place of the map, a special slide showing permanent information and establishing the data format can be displayed. The computer-generated symbols can be positioned on the display to convey the variable data. Go/no-go results can be indicated by positioning a symbol over appropriately labeled points on the optical display. Analog information, using a symbol positioned along a numbered scale similar to a meter face can be easily presented. The operator's attention can be directed to a printed instruction by appropriate symbol location. With this type of display, the possibilities for machine-to-operator communication are unlimited.

To demonstrate the need for adapting in-flight test designs to maximize use of prime hardware, consider another type of display. In this case, also illustrated in Figure 5, a cathode-ray tube displays symbols consisting of straight-line segments selected and positioned by the computer. As in the previous example, this display is normally used to display tactical information to the flight crew. For in-flight test, however, alpha-numeric symbols constructed from straight lines in the manner frequently used for scoreboards can be displayed.

Thus, identification numbers of tests and defective hardware can be displayed at will. Also, mode names can be engraved around the perimeter of the faceplate and used for mode availability identification. In this example, "X" denotes a faulty mode, "O" is used for a mode with acceptable performance, and "V" indicates that the test has not yet progressed to the point where a final decision can be made but that no failure affecting that mode has been detected as yet. Other techniques, such as blinking symbols, can perform specialized functions like calling the operator's attention to a manual action that must be performed. Again, countless additional possibilities can be found for use of this display to communicate test information to the operator.

A secondary benefit resulting from use of operational displays for in-flight test is the verification of the display performance. This is a special form of diagnostic testing. Other controls and displays can be evaluated by involving them in the tests, provided that sufficient time can be allowed for manual participation.

If the controls and displays are used in a manner resembling their use in normal operation, the operator is checked out together with the equipment. Since man is an important element in most avionics systems, this type of test can often be highly advantageous. This is particularly true when seldom used, but critical, modes are tested. The review provided by each test run will help to keep the operator familiar with operational procedures.

Microwave Frequency Generation by Self-Test

Stable low-noise microwave test frequencies are often required. The penalties that result from including a microwave frequency generator such as a crystal oscillator and a varactor multiplier are so large that some alternate low-penalty method must be found to perform this function.

If the system includes a precise microwave source of a different frequency than is needed for test, at least two methods for using it to generate the needed signal can be considered. Figure 6 illustrates these methods in block diagram form. If the existing frequency is $f_o$ and the required test frequency is $f_1$, both methods require an oscillator with output frequency $f_x = f_o - f_1$. It may be possible to find an oscillator meeting this requirement elsewhere in the prime system. Otherwise, a special BITE oscillator can be provided.

The first method involves translation of $f_o$ to $f_1$ with a mixer or a modulator. The mixer output will contain an image frequency, the carrier frequency, and higher order products, in addition to the desired test frequency. The undesired signals can be attenuated by filtering. If a double-sideband or single-sideband balanced modulator is used to obtain the translation, the undesired signals can be further reduced at the expense of additional complexity.

The second method provides a signal of greater purity, but requires considerably more complex hardware. In this case, a phase-lock loop is used to tune a voltage-controlled oscillator (e.g., a backward-wave-oscillator or IMPATT-diode oscillator) to the desired frequency, using the existing frequency as a stable reference. Here, the only output frequency is $f_1$.

These techniques are not limited to microwave frequencies, but at lower frequencies the more straightforward method of generating the required signal directly is less expensive so the incentive to use an existing signal source is less. Of course, if the prime equipment contains two precision signals whose sum (or difference) is the desired frequency, the suggested method will probably result in the lowest penalty at even relatively low frequencies. In this event, a crystal oscillator and a VCXO controlled by the computer offer considerable latitude in frequency generation.

## Example of a Complete Test

As a final example, a test that uses all of the in-flight test design techniques described in this paper will be described. It is required to verify three* closely spaced pulse-repetition-frequencies (PRF's). In studying this test, note that the principle of diagnostic analysis for fault isolation is used, that the self-test-ratio is high, and that the conceptual test design is entirely different and more complex than the equivalent laboratory test. Despite the complexity of the test, the high self-test-ratio indicates that the actual hardware-related penalties are lower than could be obtained using conventional procedures. For comparison, the laboratory test for PRF frequency requires the use of a conventional frequency counter.

The accuracy of the PRF's exceeds that of any other frequency or frequency measuring function in the system. Therefore, it is decided that they must be verified by cross-comparison with each other. This method, while not yielding values for absolute frequencies, will establish that the differences are correct. Since all three PRF's are independently generated, this method will have a high probability of detecting an erroneous frequency. Also, if a fault does occur, diagnostic analysis of the failure pattern will isolate the fault to one of the three PRF generators.

Let the required frequencies be $f_1$, $f_2$, and $f_3$. Then the test will measure the difference frequencies

$$f_{12} = f_1 - f_2 \quad ,$$
$$f_{13} = f_1 - f_3 \quad ,$$
$$f_{23} = f_2 - f_3 \quad .$$

Since there is no means for determining which is the higher frequency, each difference frequency must be treated as absolute.

---

* This test, with obvious modification, would be appropriate for any number of PRF's greater than two. For two PRF's, the entire test, with the exception of the diagnostic fault-isolation routine, is also valid.

Now, suppose that $f_3$ drifts $f_\Delta$ Hz high. The test results will then be

$$f_{12} = f_1 - f_2 \quad ,$$
$$f_{13} = f_1 - (f_3 + f_\Delta) = f_1 - f_3 \pm f_\Delta \quad ,$$
$$f_{23} = f_2 - (f_3 + f_\Delta) = f_2 - f_2 \pm f_\Delta \quad .$$

By diagnosis, it is obvious that the fault is in the $f_3$ generator.

Unfortunately, the PRF frequency differences are too small for convenient measurement with the required accuracy. Therefore, it is concluded that the differences can be increased to measurable values if the PRF's can be multiplied to higher frequencies. If a conventional multiplier is not available in the prime system, some other means must be found to accomplish the multiplication.

Recall that the pulsed output of a radar transmitter consists of a sin x/x spectrum of discrete frequencies centered at the carrier frequency ($f_o$) and spaced by the PRF ($f_x$). Thus, as shown in Figure 7, the $n^{th}$ spectral line is $nf_x$ Hz above (and below) $f_o$. This provides the mechanism for multiplication of the PRF's to obtain the increased difference frequencies needed for measurement.

Using prime equipment functions, somewhat reconfigured to satisfy the requirements of this test, as shown in Figure 7, the $n^{th}$ spectral line can be located by observing the detected output of a narrow-band filter as the spectrum is swept across the filter by the VCXO. Figure 8 is a crt display of an actual test of this type. The computer stores the VCXO control voltage required to place the $n^{th}$ line in the filter passband for each of the three PRF's. Then these voltages are multiplied by a factor corresponding to the VCXO transfer function (Hz/volt) and the results are analyzed to determine the difference frequencies as described above. Of course, the differences are also multiplied by n so that they can be determined with the required accuracy.

Note that this test does not depend on precise knowledge of the absolute VCXO transfer function. It is only required to provide a transfer function whose slope is accurately controlled in the narrow region used for this test.

## Conclusion

The in-flight test design techniques described here have been used to provide missile control system test capabilities that weigh less than 3 percent of the total system weight and that are sufficiently thorough and accurate to permit complete elimination of planeside test equipment. The application of these techniques to manned space systems can result in equally impressive capabilities for in-flight testing.

A willingness to adopt unconventional approaches to the test design problem, together with recognition of the needs of the crew and the value of the self-test concept, are the principle ingredients of successful in-flight test design.

FIGURE 1

## TYPICAL TEST EQUIPMENT FUNCTIONS FOUND IN A LARGE SYSTEM

- R-F SIGNAL GENERATION
- DIGITAL SIGNAL GENERATION
- SPECIAL WAVEFORM GENERATION
- SWEPT-FREQUENCY SIGNAL GENERATION
- AMPLIFICATION
- AMPLITUDE DETECTION
- PHASE DETECTION
- FREQUENCY MEASUREMENT
- VOLTAGE MEASUREMENT
- POWER MEASUREMENT
- FILTERING
- FREQUENCY TRANSLATION

FIGURE 2. HARDWARE RELATIONSHIPS

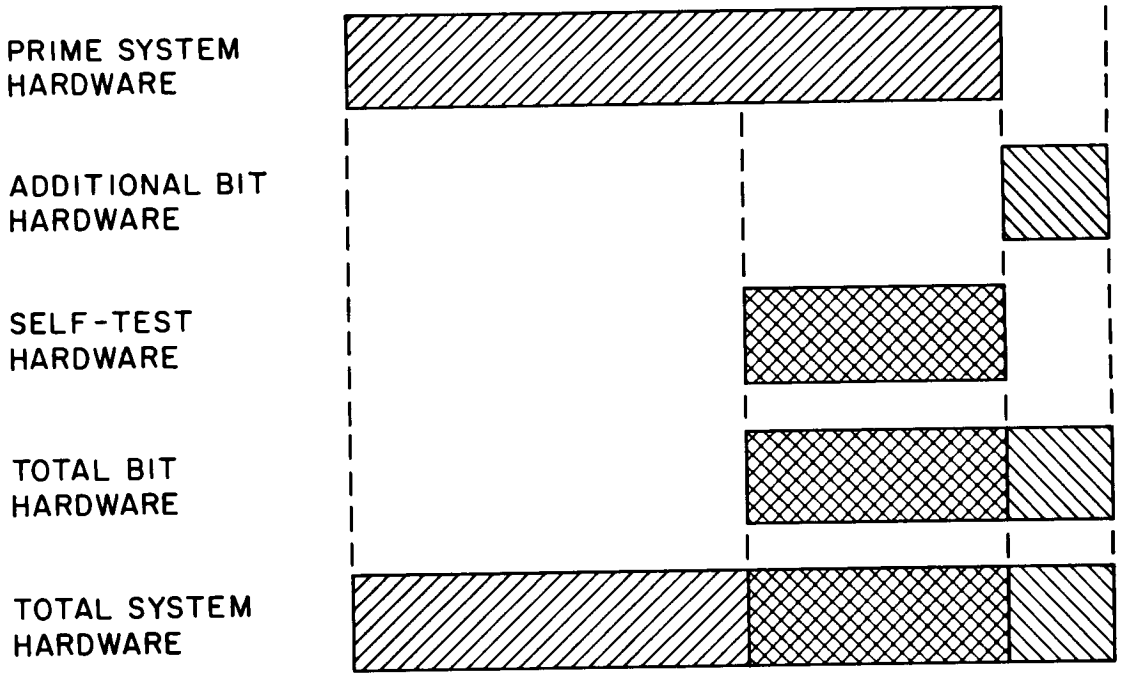
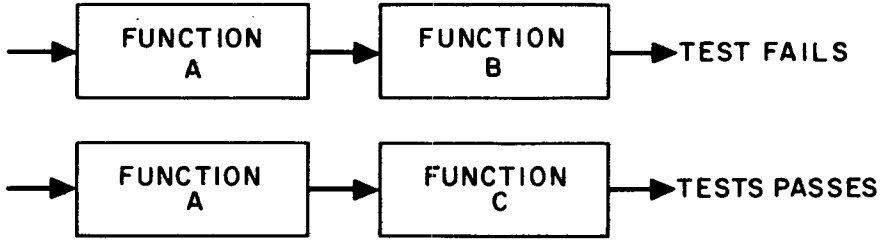


FIGURE 3

FAULTS CAN BE ISOLATED BY LOGICAL ANALYSIS OF RESULTS OF A SERIES OF INDIRECT TESTS.




CONCLUSION: FUNCTION B IS DEFECTIVE

DIAGNOSIS IS APPROPRIATE FOR SELF-TEST

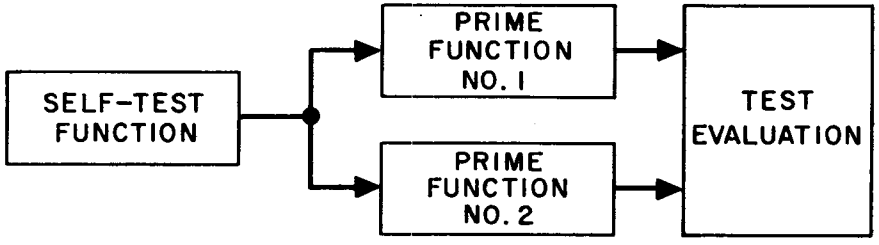
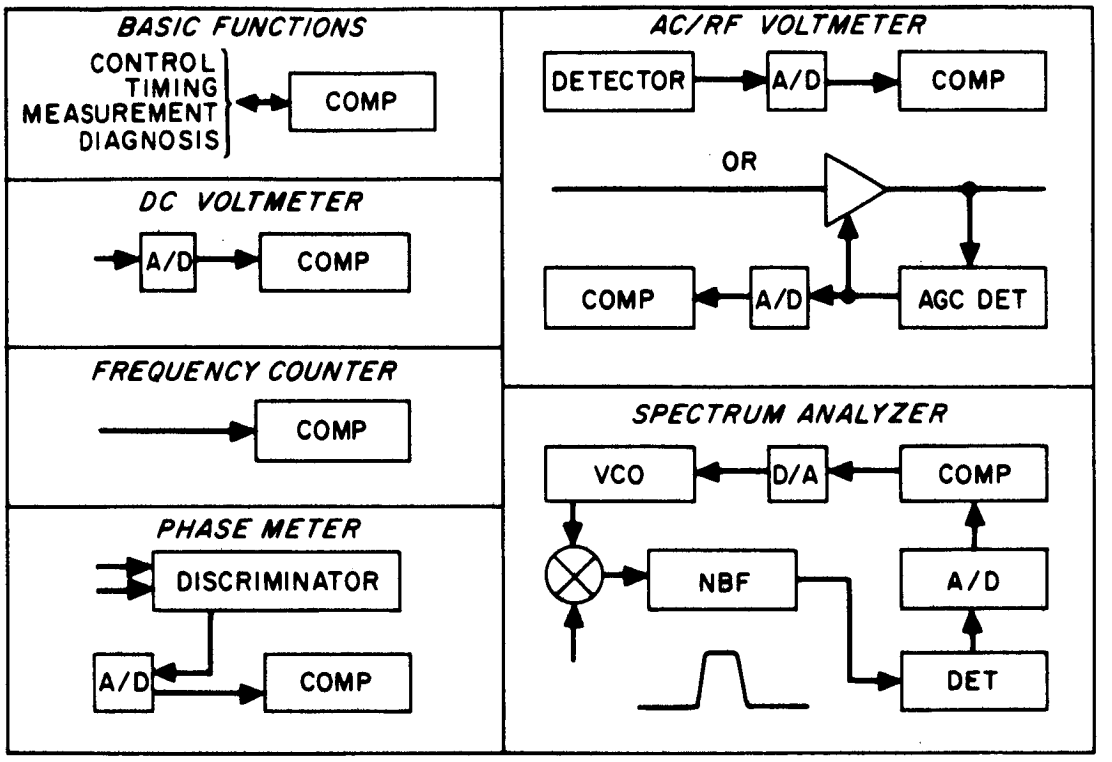


FIGURE 4

SELF-TEST WITH A COMPUTER



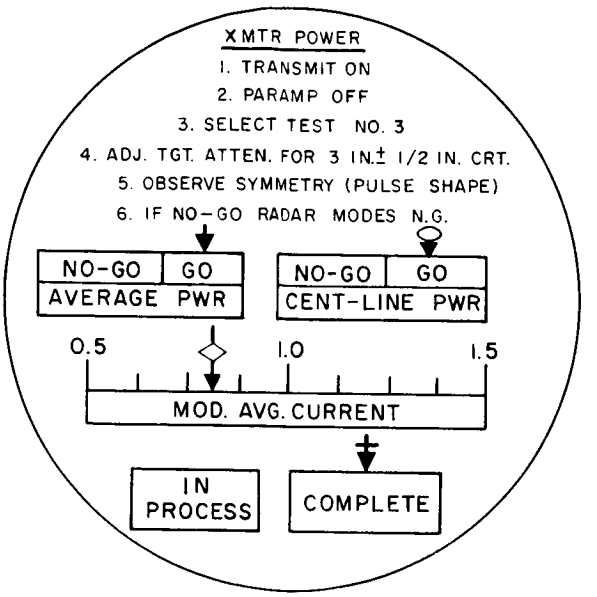



FIGURE 5a. ELECTRO-OPTICAL DISPLAY

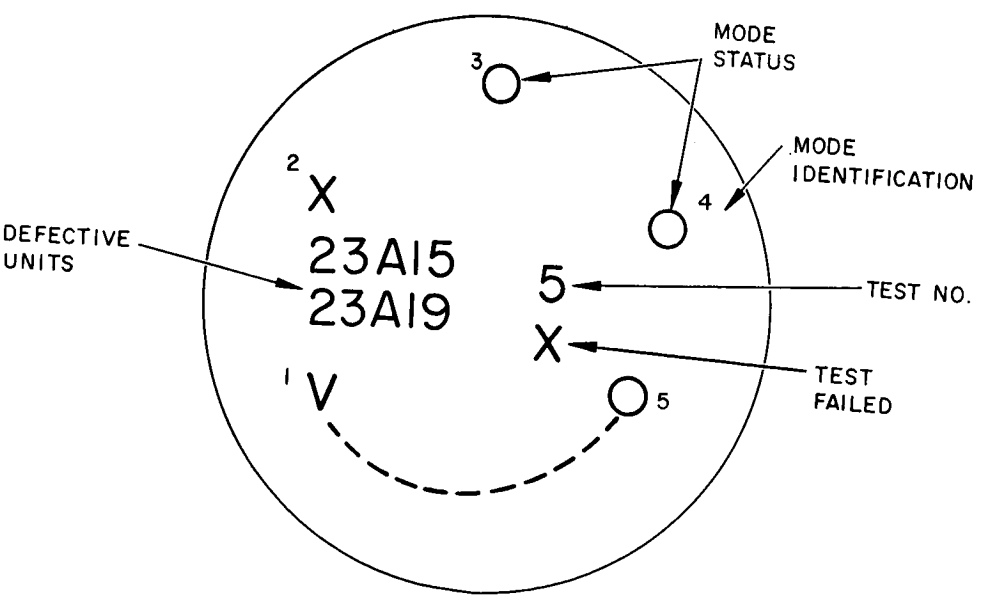



FIGURE 5b. CRT DISPLAY

FIGURE 6

MICROWAVE SIGNAL GENERATION

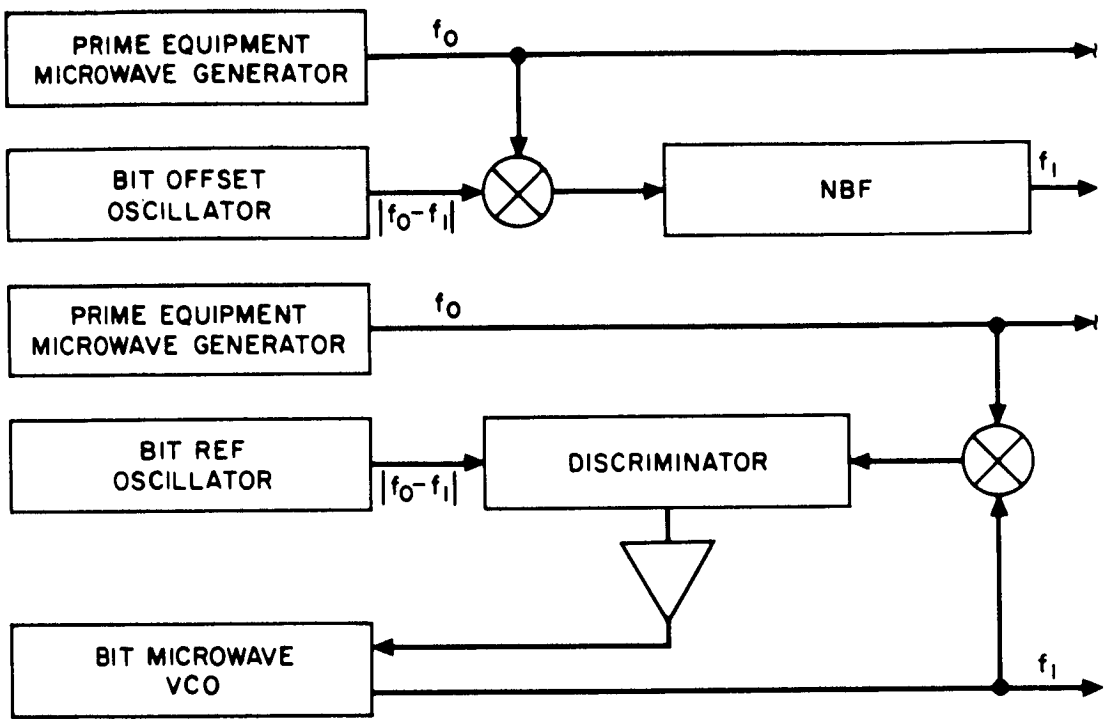


FIGURE 7

PRF MEASUREMENT

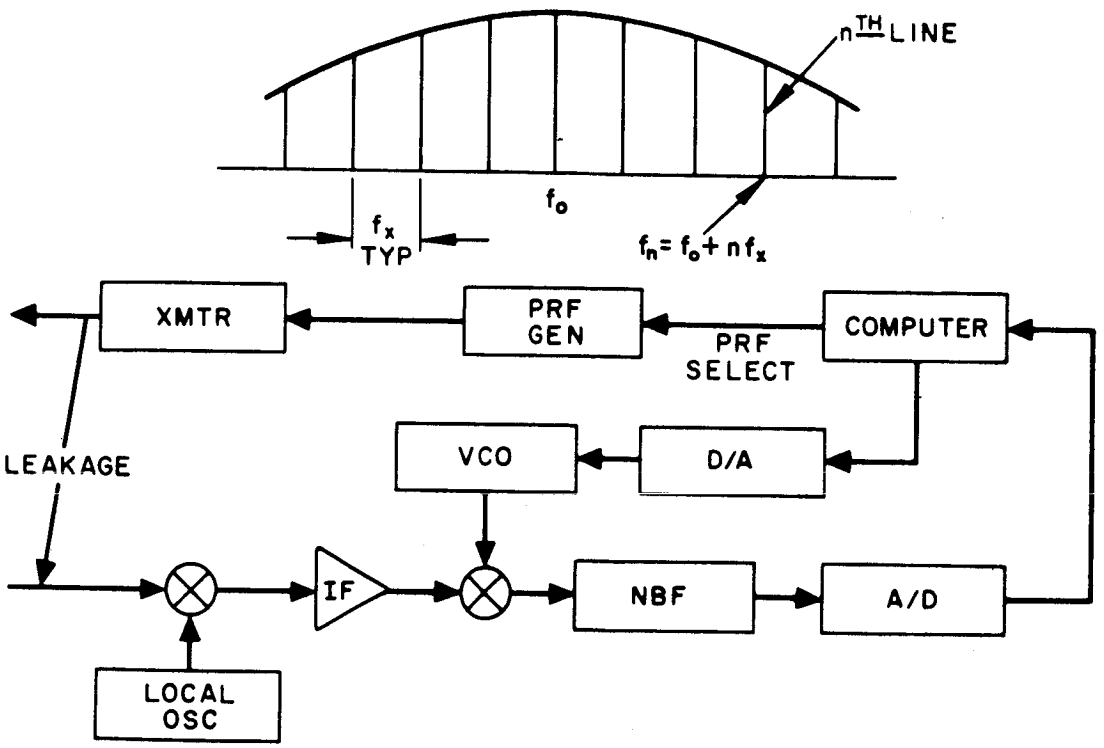
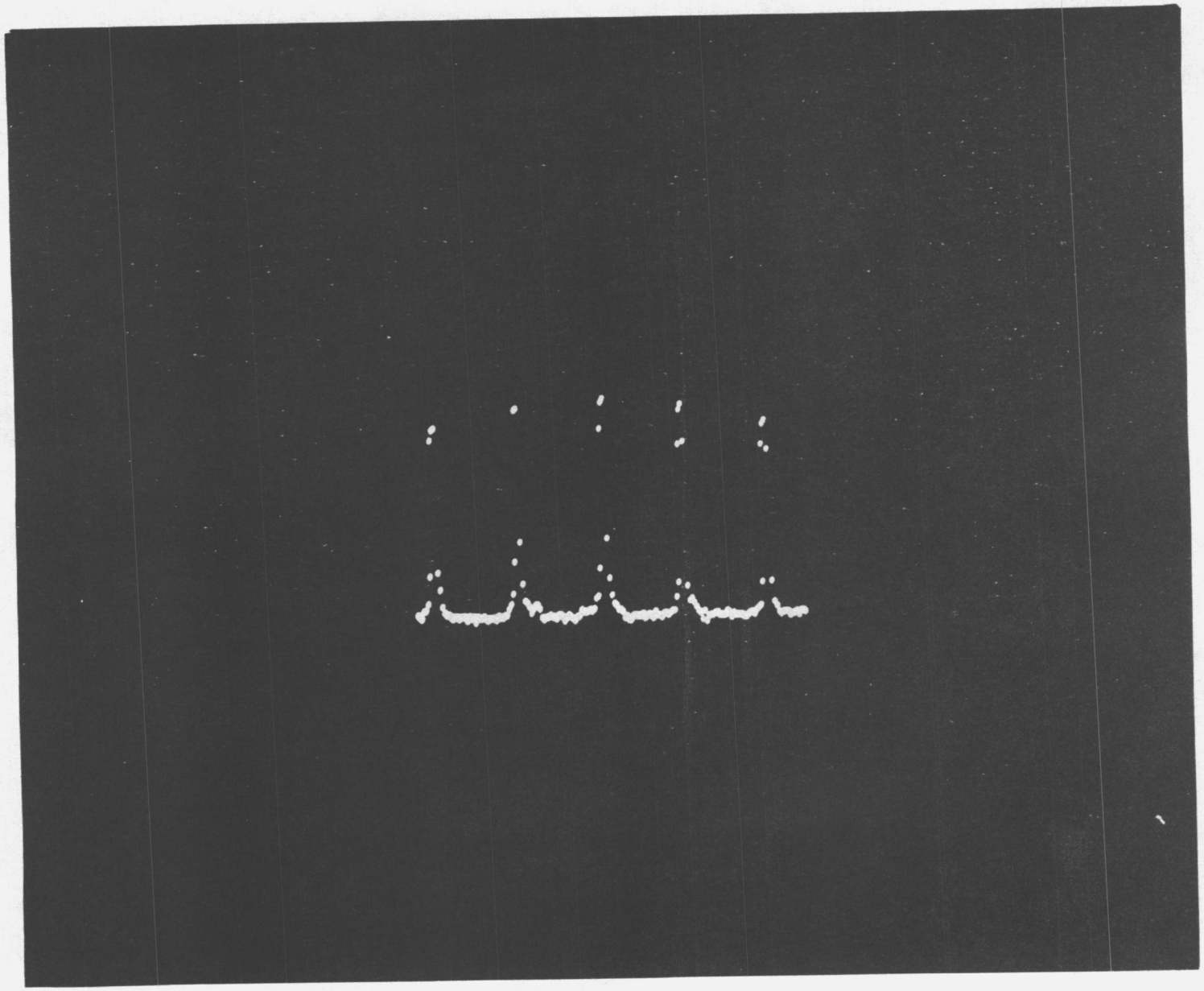


FIGURE 8. CRT DISPLAY OF PRF TEST



# AN AUTOMATIC TEST SYSTEM BASED UPON PATTERN RECOGNITION DISPLAYS

David M. Goodman

ABSTRACT

Employing a large number of optical and electrical input signals, an automatic test system is proposed which rapidly samples a plurality of test data from an operating ensemble and which then displays the data block format so that interpretation is made on the basis of pattern recognition. The display is via a cathode ray tube which is operated in a TV mode so that the displayed test data can be continuously and rapidly up-dated. By means of rear ports in the cathode ray tube it is shown how failure patterns can be recorded on film and preserved to make the test system self-programming, thereby overcoming one of the major obstacles in presently conceived automatic test systems.

## I. INTRODUCTION

The techniques of automation have been employed by the electronics industry at an ever increasing rate, beginning in the early 1950's. These techniques have been used with varying degrees of success to solve the information handling problem in command and control, missile launch, surveillance, and fire control systems to name a few. These techniques have also been applied in an attempt to provide adequate maintenance for these types of systems. The nature of these prior efforts is well documented*.

A new test technique described in Volume II of these lecture notes, makes use of optical fibres for transmitting test data in optical form; a vidicon for scanning the optical test data; and a cathode ray tube for displaying in real-time the results of test. Presented herein are proposed test systems which take advantage of and are in accordance with this advanced technique. The first arrangement pertains to an electronic assembly containing 96 test points. The data from these test points is displayed for analysis in such a manner that a fault anywhere in the assembly is detected and pinpointed at a single glance. The test system then is expanded so that a larger collection of test data can be displayed, stored, and interpreted.

Accordingly, it is an object of this test system

(1) to provide real-time matrix type displays which present a large quantity of test data in multi-color easy-to-interpret patterns;

*"Automation In Electronic Equipment", N.Y.U. Press, New York, New York 10003

(2) to provide in conjunction with the matrix display a film strip record-keeping device which operates to retain test data which is generated when the system under test develops a malfunction or out-of-tolerance condition; and

(3) to provide in photographic and handbook form an information storage facility which contains a library of pre-recorded failure patterns representative of specific malfunctions that may occur in the system under test.

In order to set the stage for the detailed explanation of how these objectives are achieved, some of the terms which will be used in the description are now defined. These definitions should be reviewed by the reader for they set in perspective the logistic packages which typically are encountered in maintenance activities, and it is towards a maintenance type display that this test system is oriented.

## II. DEFINITIONS

Piece-part. A separately identifiable component such as a resistor, or vacuum tube. In micro-electronics a chip, or flat pack.

Major Piece-part. A component of substantial size and cost such as a high power transmitting tube.

Sub-assembly. A combination of piece-part physically located on a common chassis or carrier board. In micro-electronics, a replaceable card containing a plurality of flat packs or the like.

Assembly. A combination of piece-parts, major piece-parts, and sub-assemblies arranged electrically and mechanically in such a manner that the entire unit functions as a whole. The assembly generally is a replaceable unit.

Equipment. A combination of assemblies designed to perform a given function. An example is a radar, sonar, or communications transmitter or receiver.

System. A combination of equipments arranged so that a plurality of functions are grouped together to achieve an end result. An example is a radar fire control system; or an air traffic control system.

Major System. A combination of systems arranged as in Sage; or in Dew Line; or Nike-Zeus; or Apollo; etc.

Module. Any sub-assembly, assembly, or equipment which is packaged so as to be removed and tested as an entity.

Unit Under Test (UUT). A generic expression applying to any of the foregoing definitions.

(PUT) - Piece-part under test
(MUT) - Module under test
(EUT) - Equipment under test
(SUT) - System under test

Test System (TS). Combinations of any of the foregoing UUT, PUT, MUT, EUT, and SUT with special circuits for achieving the test functions. The test system may be external or built-in. In this paper, they also generate the displays for pattern recognition.

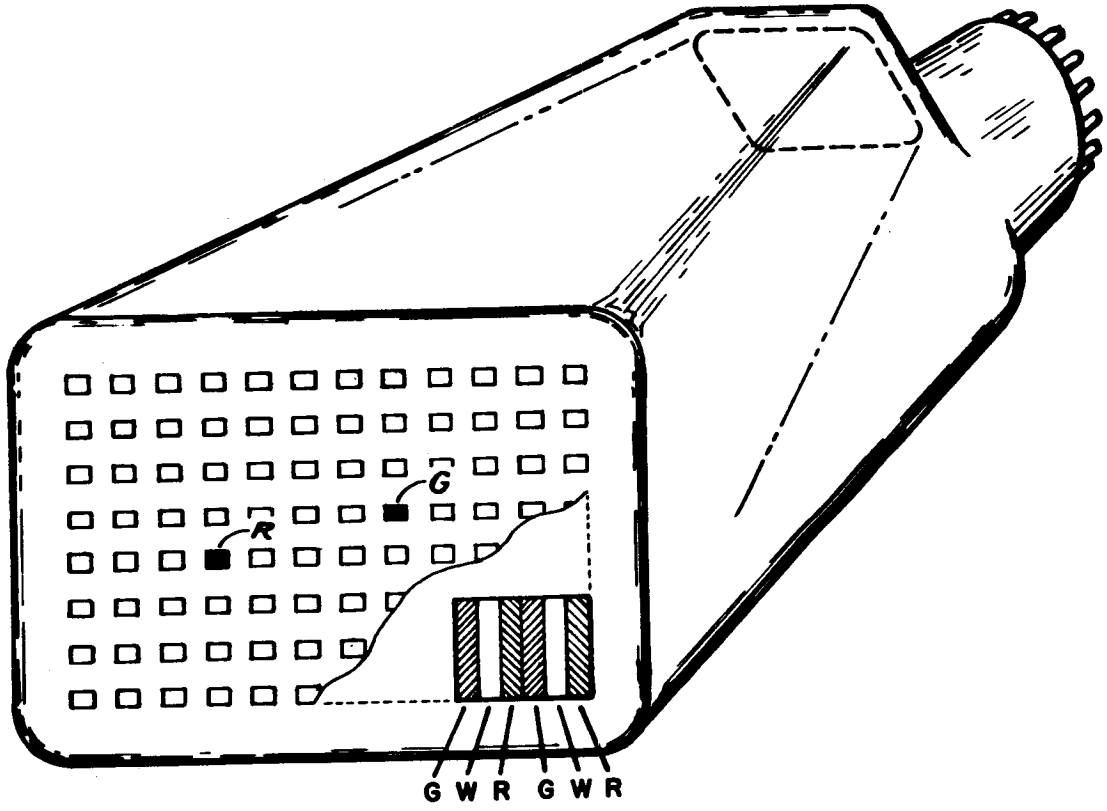



FIGURE 1

Fig. 1. A matrix of 8 X 12 or 96 data points is presented in real-time on a color cathode ray tube. One of the data points registers a high value and therefore appears in red; another data point registers a low value and appears in green. The other 94 points are within tolerance and therefore appear white. The sketch in the corner shows in greater detail the vertical line structure of the target screen on which the colors are produced for the different data points. An optical port is shown at the rear of the CRT envelope where a camera may be positioned for taking color photographs of the rear of the target screen.

## III. THE MATRIX DISPLAY

To describe one form the Test System may take, an illustrative example is chosen which has 96 electronic components in an assembly, each of which can fail without causing the failure of any of the others. The assembly is designed with one test point available for each component, and when the circuit is energized a voltage is obtained from each test point. These voltages are attenuated or amplified so that they all measure 1 volt dc for a normal functional assembly. The 96 voltages are sampled sequentially by a scanning device which feeds into circuitry to compare these 96 samples to their nominal value of 1 volt. A display is then generated in which the test data derived from the 96 test points is presented as luminous spots in the form of an 8 x 12 matrix of white dots. As long as the assembly is energized, and there is no failure, the appearance of the 8 x 12 matrix of white dots remains uniform and unchanged.

The circuits which measure the sampled data operate to route the 96 test voltages into five different channels according to the magnitude of the test voltage. The channels are divided into voltage levels of 0-0.7, 0.7-0.9, 0.9-1.1, 1.1-1.3 and over 1.3. By definition, a normal test voltage is in the range 0.9-1.1 volts. A low-marginal voltage is in the range 0.7-0.9 volts; a high-marginal voltage is in the range 1.1-1.3 volts. One failure voltage (low) is less than 0.7 volts; and the other failure voltage (high) is greater than 1.3 volts.

The five channels feed into selection networks which control the color of the dots generated on the display. Typically, a test voltage in channel 1 generates a red dot; channel 2, an orange dot; channel 3, a white dot; channel 4, a yellow dot; and channel 5, a green dot. These color dots are clustered together for each of the 96 test points in the 8 x 12 matrix. What this display means to an observer thus becomes evident. Suppose test point 37 has changed color; it is because the test voltage is not normal and it takes but a brief inspection to determine from the actual color of the test point if the drift is marginal, or if the component has failed. The color also reveals whether the drift is high or low, or if the failure is open circuit or short circuit. These relationships are tabulated here for convenience.

| Channel | Voltage Range | Color | Significance |
|---|---|---|---|
| 1 | 0.-0.7 | Red | "Short" Circuit |
| 2 | 0.7-0.9 | Orange | Marginal-low |
| 3 | 0.9-1.1 | White | Normal |
| 4 | 1.1-1.3 | Yellow | Marginal-high |
| 5 | >1.3 | Green | "Open" Circuit |

## IV. THE TEST SYSTEM

A closed circuit television chain is used to generate the display. A rectangular raster is scanned at a 60 cycle rate in a standard 525 line format so that each of the 96 test points is measured in sequence and the results displayed 60 times each second. In other words, the voltage from test point 37 is examined every 16.6 milliseconds; and the dot on the matrix corresponding to this test point is energized once every 16.6 milliseconds. If the voltage is normal the dot remains white. If the voltage is not normal it will be routed into the proper one of the other four channels and the dot will appear as red, orange, yellow, or green. If the voltage drifts through more than one range, then the color of the displayed test point will drift accordingly. If the voltage varies rapidly, as in an intermittent condition, then the dot will flicker in color.

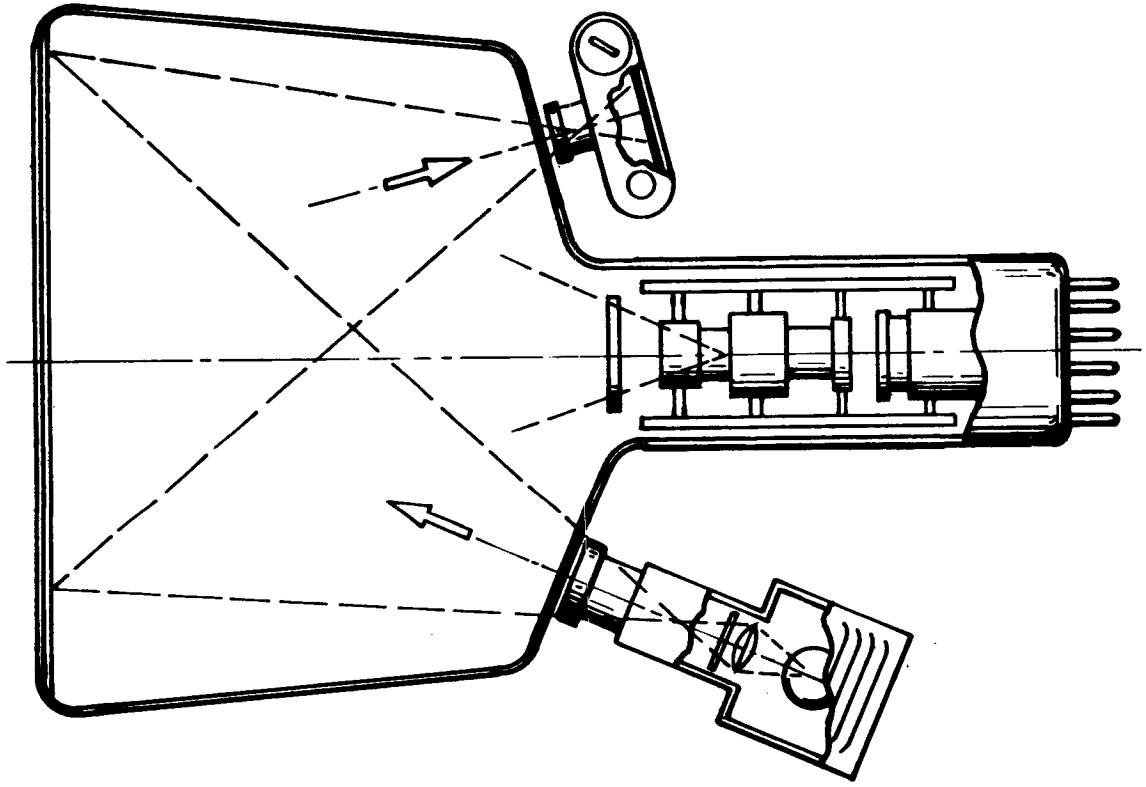


FIGURE 2

Fig. 2. A sectional view of a cathode ray tube with two separate optical ports provided at the rear of the tube envelope. Adjacent the upper port is a camera for taking pictures of the rear of the target screen. Adjacent the lower port is an optical projector for casting an image on the target screen. The projected image is viewed from the front of the CRT. And for this purpose, the target screen is made translucent.

Thus, once attracted by an off-normal color the observer can determine after a few seconds of examination if this is an intermittent condition; if it is a drift; or if it is a failure.

## A. FAILURE DIAGNOSIS

When a fault appears the observer refers to a previously prepared handbook of test point abnormalities, looks at the entry for test point 37, and finds the diagnosis for the color condition he just observed. This diagnosis can be made in less than a minute, the time it takes to find the entry, if the handbook is complete. To achieve these rapid results the handbook of test point abnormalities obviously was prepared in advance. To do this for 96 individual test points is not very difficult. But suppose, as is often the case, that failures of components do not occur singly but in combinations. Now failure patterns have to be derived for the handbook that take into account the various combinations and permutations of the 96 voltages derived from the test points. Even for only 96 test voltages this can become an excessive burden, especially since each voltage can take on five different effective values. A more practical method of preparing the handbook is to predict, from the design, the most probable failures and to generate from these (either by calculation or experimentation) a series of different failure patterns. These patterns, representing multiple malfunctions, also become part of the handbook. When a complex failure occurs, the observer has to match these patterns with those actually displayed. This matching process will take a little more time in the case of multiple malfunction than it did in the case of a single abnormality. But after a history of learning on a display of this sort it can safely be assumed that the observer's skill will increase to the point where he will recognize certain patterns for the often repeated failures; he will not have to refer to the handbook; and so diagnosis again becomes almost instantaneous.

Using the foregoing arrangement it is conceivable, even likely, that on occasion a failure pattern will be generated for which there is no equivalent in the handbook. It is necessary in this situation to trouble shoot the test assembly using any technique then available under the circumstances that prevail. After the failure is thus tracked down, the failure pattern and this new-found diagnosis is entered into the handbook. Except for this last step of entering the information in the handbook, this procedure is equivalent to ordinary present day maintenance procedures. And even with respect to this last step, it should be noted that when a technician traces down a fault new to him he stores the information in his mind so that if the failure is repeated he need not go through all the steps of re-tracing the fault.

## B. SELF PROGRAMMING

To supplement the handbook and to achieve results similar to that inherent in a thought process, a camera unit is positioned to take photographs of the cathode ray tube as illustrated in Figure 2.

In a properly functioning assembly, the 96 test points create an 8 x 12 matrix of white dots which are uniform in disposition as has been stated. For this normal condition there is no need to take any pictures. The information is repetitive and redundant. But when a test point varies from normal, thereby generating a different color, it becomes desirable to record the event and the time of occurrence. Accordingly, a photo-cell detector responsive to the drift or failure colors is positioned to face the display screen. The detector is energized when a non-normal color is displayed thereby to open the shutter of the camera. Preferably the shutter stays open for one full raster scanning

cycle, in this case 1/60 second. In so doing the camera records

(1) the last cycle of information (due to the persistance of the phosphors in the CRT) and

(2) the data displayed in the 1/60 of a second following the opening of the shutter and

(3) either clock time or running time, or both.

The shutter is then closed and the film is advanced. If the non-normal condition persists, the camera is again activated, etc. It is thus that the camera takes a series of pictures in color which yields a permanent history of performance of the 96 component assembly.

By repeating this process of photographing the display, a handbook is soon compiled which contains the pattern for each possible drift and failure mode. Each new diagnosis which is entered into the handbook should be complete and accurate since the electronic post-mortem examinations can be carried out carefully and with precision.

## C. FAILURE REPORTING

It is a further advantage of the camera arrangement just described that a filmed record has been made showing the status of events for the scanning cycle that preceded the actual failure. This record often will suffice to determine the sequence of stages that the assembly went through as it reached the failure mode. This type of failure identification is best used so that re-design or retrofit of the assembly under test can be better accomplished. Not only will the actual failure be recorded but the original stresses which brought about this condition will be part of the record. In other words, the instant arrangement can provide a good failure reporting system for it identifies the failed component as well as the cause of failure, which may originate with some other component. Additionally, if a given failure is not catastrophic but is due to gradual deterioration process then the observer of the display will see the onset of drift, he can anticipate the failure, and he can take corrective action in advance of the failure itself. For this purpose, the handbook also is prepared to contain instructions specifying corrective action that may be taken under these circumstances.

## D. INFORMATION-RETRIEVAL AND SELF-TEST

The next step in the development of this Test System is to provide the observer with mechanical assistance in sorting through the different failure patterns in his handbook. To aid in this information-retrieval process, each of the 96 points in the 8 x 12 matrix is provided with a circuit controlling relay. Thus, test point 37 has its associated relay 37. This relay is energized only when one of the four non-normal voltages appear at the test point. This relay functions to select the failure cards which are punched with a hole at position 37. These cards may be of the type conventionally used in machine accounting systems, or they may be designed specifically for the Test System. In either case, the details of punching, collating, and sorting are believed sufficiently well known so that this brief reference thereto suffices for the purposes now at hand. Position 37 is punched on four cards, with one card and one hole corresponding to each of the four non-normal voltages. When relay 37 is energized all the cards with a hole at position 37 are sorted from the deck. The cards are then sorted as a consequence of the test voltage being in either of channels 1, 2, 4, or 5. Likewise if test points other than 37 were off normal, their cards too

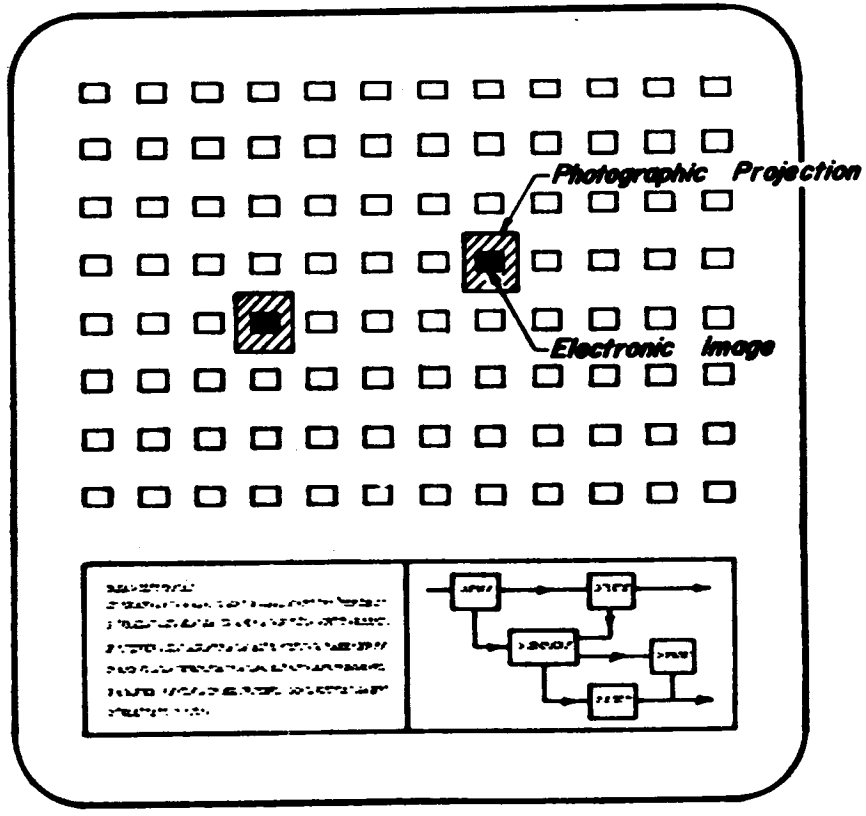



FIGURE 3

Fig. 3. Another matrix of 96 data points in which the real-time data is compared by an operator to data previously stored on film chips. Two of the data points are out-of-tolerance. The automatic information retrieval system has selected the proper mal-function chip, which fact is confirmed by the photographic projection properly overlaying the electronic real-time image. In the lower half of the CRT there is projected from the same film chip some text and schematic drawings which contain appropriate instructions for the operator.

would be removed from the deck; and there would be a further selection which takes into account the passage of the test signal through channels 1, 2, 4, or 5. Written, typed or printed on each card is a description of the component which caused the drift (or malfunction) together with a description of the adjustment (or repair) which is to be made. Additionally, each card carries a first photographic film transparency with this same descriptive information; and a second photographic transparency which contains the multi-color drift or failure pattern.

In response to the operation of the photocell detector, a transport mechanism inserts the appropriate failure card 37 (due to operation of relay 37) into the optical projector which is positioned towards the rear of the display CRT as illustrated in Figure 2. The information on the photographic transparency thus is projected onto the rear face of the CRT. Therefore, the operator who is observing the real-time test data also sees the descriptive information together with the failure pattern stored on the two film transparencies. On one section of the CRT set aside for the projection of the first transparency, there appears the English language instruction or description of malfunction. An aural or visible alarm or message may accompany this projection of the data to make sure that the observer's attention has been drawn to the display. Simultaneously, the failure pattern recorded on the second transparency is projected for viewing by the observer. In this case, however, the failure pattern recorded on the film is not projected on a special portion of the CRT but is projected to be super-imposed over the 8 x 12 matrix of test points as illustrated in Figure 3.

This is done so that the observer can match the failure pattern generated by the test system with that stored on the film, thereby to verify that the proper failure card has been selected. This visual comparison is made on the basis of (1) dot positions in the matrix and (2) the color of the dot at each position. This is a form of self-test of the Test System which can be 'eyeballed' by the observer with relative ease.

When a complex fault occurs involving more than a single component a number of relays will be energized. To retrieve the proper failure card a series of card sorting operations commence. The first pass selects all possible faults associated with one of the non-normal test points. The second pass takes into consideration all possible faults collected from the first pass, and selects only those cards which have a fault associated with the second non-normal test point. This selection process continues until only the failure card remains whose pattern matches that generated by the assembly under test. The two photographic transparencies on that card then are projected for viewing by the observer. This is illustrated in Figure 4 where the concept of pattern recognition is clearly revealed.

## V. SUMMARY OF THE TEST SYSTEM

The observer so far has been given a test system in which any drift, intermittent, or failure in the assembly under test shows up as a "color flag" in a matrix of white dots which are displayed on a television-like receiver. To determine the significance of this "flag" the observer may rely on his memory, may refer to his handbook, or he may read the information from the display screen itself. Perhaps most important, he knows that the matrix-like failure pattern in his handbook is also projected on the face of the CRT to overlay that generated by the electron beam in real-time. This gives him the opportunity to confirm immediately that the failure card that was selected by the automatic information-retrieval sub-system does in fact yield the failure pattern generated by the faulty assembly. If the two patterns do not match, as illustrated in Figure 5, the observer, is put on notice that either or two conditions prevail.

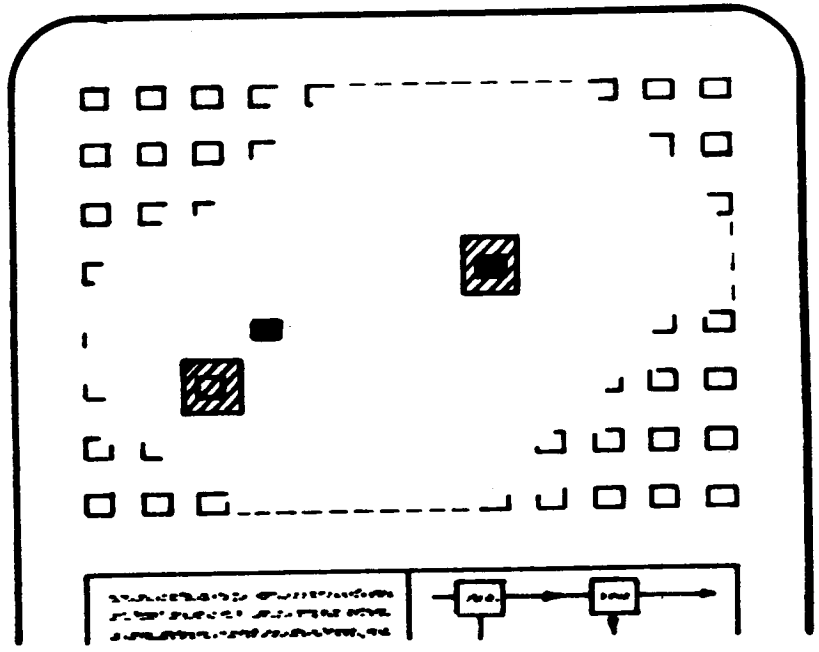


FIGURE 4

Fig. 4. An example of the failure of the information retrieval system to select the proper malfunction chip. Note that one of the out-of-tolerance data points and the photographic overlay are not in proper register.

First: the retrieval sub-system or some other unit of the Test System may not be functioning properly. And, this may be verified by exercising a special subroutine in the information-retrieval unit. Second: the Test System may be functioning properly but there may not be a suitable failure card in the library. This is equivalent to a "program-stop", type of condition which probably is inherent in any automatic physical system which is designed to make logic decisions. Fortunately, this mis-match of the test patterns should be the exception rather than the rule in a well instrumented system and the possibility of its happening should not detract from the already described substantial gainful results.

## VI. VARIATIONS IN THE TEST SYSTEM

### A. STRAIGHTFORWARD MODIFICATIONS

A number of extensions of the test system are worthy of mention at this point.

First: the test point data, which was generated in dc analogue format, was converted or quantized into five different voltage ranges. These ranges can be reduced to three (HI-GO-LO) or increased to almost any desired extent. Each range or step is presented as a different color in the visible matrix. For a limited number of steps, discrete color producing phosphors are used on the face plate of the CRT. For a substantial number of steps, the plurality of colors are generated more suitably by varying the excitation relative to each other of three different primary color producing phosphors. Thus, instead of using five discrete colors in the display (white being one of the colors) the three primary colors may be used in varying proportions to generate the red, orange, white, yellow, and green colors. This method is increasingly advantageous as the number of quantized levels is increased.

Second: instead of using a rear port for optical projection of the data stored on the film transparency, a second electron gun can be incorporated in the CRT to be energized by a flying spot scanner which has as its input the same film strip.

Third: instead of using punched cards, the failure selection slots and instructional information may be recorded magnetically on cards; or on tape, drums, discs, or cores.

Fourth: the scan rate may be altered, or the test system may be time-shared to better suit the needs of testing. Thus, if the assembly under test has a high inertial mechanical or thermal component there may be no need to sample the data 60 times each second.

Fifth: devices other than CRT's can be used for display purposes especially if the rate of data presentation does not require high resolution and high scanning speed.

### B. COLOR SYNCHRONISM

Suppose the twelve different test voltages to be displayed on the first line of the 8 x 12 matrix are not steady in value, but that they vary at a rate slow in comparison to the sampling time of 16.6 milliseconds. Suppose further that all voltages vary in magnitude at the same rate. Then, the twelve points in the line will change color in synchronism to yield a pleasing visual sensation.

This often may be desirable from the observer's point of view, but as the Test System is now arranged it will trigger the camera unit. Since the system is functioning properly the camera unit should not be energized, and therefore this line of data will have to be masked from the color-sensitive photo-cell detector. This can be done by providing suitable gating circuits synchronized with the raster scanning, or it can be done by mechanical masking means. Alternatively, a properly phased reference voltage varying at the same rate as the data can be supplied to the comparison network for these twelve points of test data so that the uniform white matrix of visual data is maintained.

## C. THE BLINK TECHNIQUE

Suppose now that a digital signal, rather than a sampled analogue voltage, is made to control the first line of the matrix. The color of the twelve points in the line will take on a given set of values which are representative of the twelve coded numbers. If the digital signal is repetitive the pattern of the color dots is stationary. This condition is apparent from visual inspection and is meaningful. On the other hand, if the digital data is rapidly varying, in non-cyclic order, then visual inspection generally will yield little information. To better interpret patterns of this type, a scanning raster is generated which may be considered akin to the blink microscope. As is well known from the field of astronomy, the repetitive flashing of two patterns in the field of view of an observer creates an image which enables the observer to detect quite small differences between the two patterns. In this manner, the planet Pluto was discovered after many years of comparing thousands of photographs of the sky.

This principle of the blink technique for pattern comparison is applied to compare two groups of test signals to each other. Thus, by way of example, an assembly under test is arranged to process a digital signal which activates all the components in the assembly. This digital test signal may be the test message "See the quick brown fox jump over the lazy dog's back" often used in teletype systems. The response of the assembly is picked off at some suitable point and is displayed bit by bit across the matrix pattern. When the test signal is made repetitive, and is timed to be in synchronism with the scanning of the raster which generates the matrix pattern, and with a stable response from the assembly under test, a fixed pattern is generated on the display device which is easily recognized. Furthermore, a typical overall response from such an assembly will have the input signals and the output signals identical in character. Under these circumstances, the display is made to generate on the first scanning cycle a pattern representative of the input signal. On the second scanning cycle, the display is made to generate a pattern representative of the output signal. On the third cycle the input signal again controls the pattern, etc. Therefore the observer sees the alternate presentations of the input and output digital data in matrix format. If the input and output signals are identical, signifying proper operation of the equipment under test, the matrix of data reappears with each scan in a smooth and uniform fashion. On the other hand, if there is a lack of correspondence between input and output signals the observer will see a blink or color alternation, at the point in the matrix where the discrepancy exists.

## D. COMPUTER TESTER

A digital computer serves as another example for demonstrating the usefulness of this color blink phenomena. In this case, special test programs are used to exercise the computer. For each programmed exercise a response is obtained from the computer which is compared to a predetermined response. Here again, the two responses are compared on a bit-by-bit basis across and down the matrix

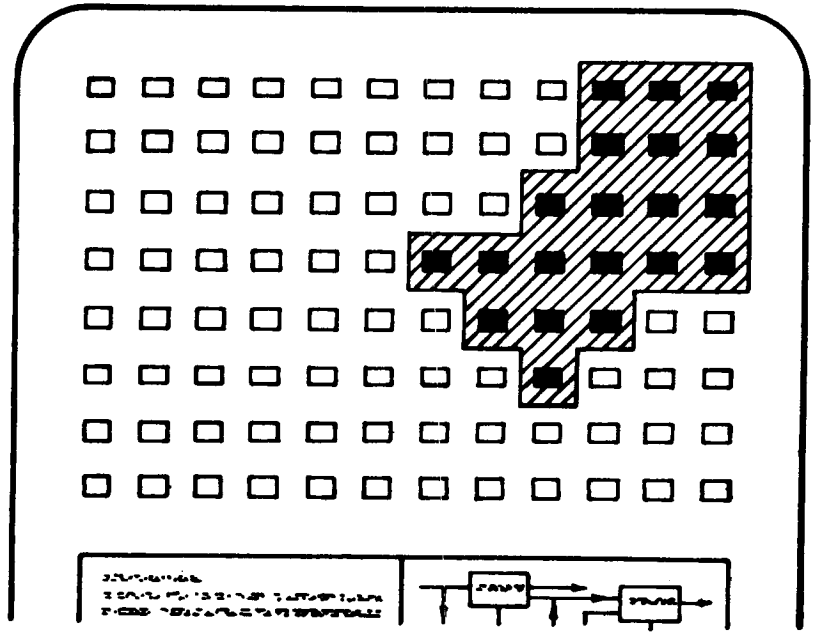


FIGURE 5

Fig. 5. An example of a multiple failure, or out-of-tolerance condition, in a complex system. A proper diagnosis has been made of the fault inasmuch as the photographic display is in proper re-register with electronic display. After a few exposures to this fault condition, the operator will recognize the meaning without having to read the text which is projected for viewing below the matrix.

on the CRT display. Both the program and the correct response may be stored in the memory section of the computer, be it on magnetic drum, disc, core, or delay line. The test sequence and the response thereto is programmed from this memory section in synchronism with the CRT raster scanning cycle to provide a stationary pattern. On the first scanning cycle, the measured response is displayed; on the next cycle the "calculated" response is displayed; and so forth. Therefore, a discrepancy between the measured response and the predetermined response shows up as a color blink on the face of the CRT at each point in the matrix where a discrepancy exists. Since this condition reflects a failure mode, there is advantage in recording this failure pattern. Again, it is convenient to photograph the display whenever a blink appears. But since the matrix ordinarily is filled with a plurality of test data in different colors, a photo-cell detector responsive to these colors cannot be used to trigger the camera; and masking the CRT as earlier proposed will be ineffective. Instead, a parity check is made of the data from one scanning cycle to the next. This parity check consists of first counting the bits displayed for a given scanning cycle to establish a reference count. Then, if the bit-count on the next cycle is the same, no photograph will be taken. If the bit-count is different, a photograph will record the matrix pattern. This is a much desired mode of recording the failure data for the camera is not triggered to take repetitive pictures of the same failure pattern.

To cite a simple example, let the digital data be in such format that a "yes" or "1" bit is displayed on a CRT as a green dot; and that a "no" or "0" bit is displayed as a red dot. These dots result from the excitation by the electron beam of a green phosphor and a red phosphor, respectively. For convenience in visual comparison, all color dots are produced by phosphors having decay times which are approximately equal. The first line of dots in the matrix is controlled by a first sequence of bits and therefore has a series of green and red color dots. The next row has a second sequence of green and red dots, etc. This display is used in the block diagram of Figure 6 where an arrangement is shown for checking the memory section of a digital computer. The scanning of the electron beam on the face of the CRT is synchronized with the scanning of the storage elements on a magnetic drum, a core matrix, or the like.

The readout of each bit of data in the storage medium is delayed in time so that the CRT displays a matrix of points which corresponds to the position at which the data is stored. This "electronic development" of the surface of the magnetic drum is repeated fast enough so that a steady image is seen by the operator. In between each scan of the data in the memory, there is alternately presented an image of the comparison data previously stored on a film chip. Towards this end the raster generated by the flying spot scanner is also synchronized with the rotation of the memroy drum. When the pattern of the magnetic drum and that of the film chip is properly matched the display is continuous or even. When there is a discrepancy in the patterns, a blink occurs at the position where the discrepancy exists. This blink may be observed in position, in brightness, and in color variation.

MAGNETIC DRUM

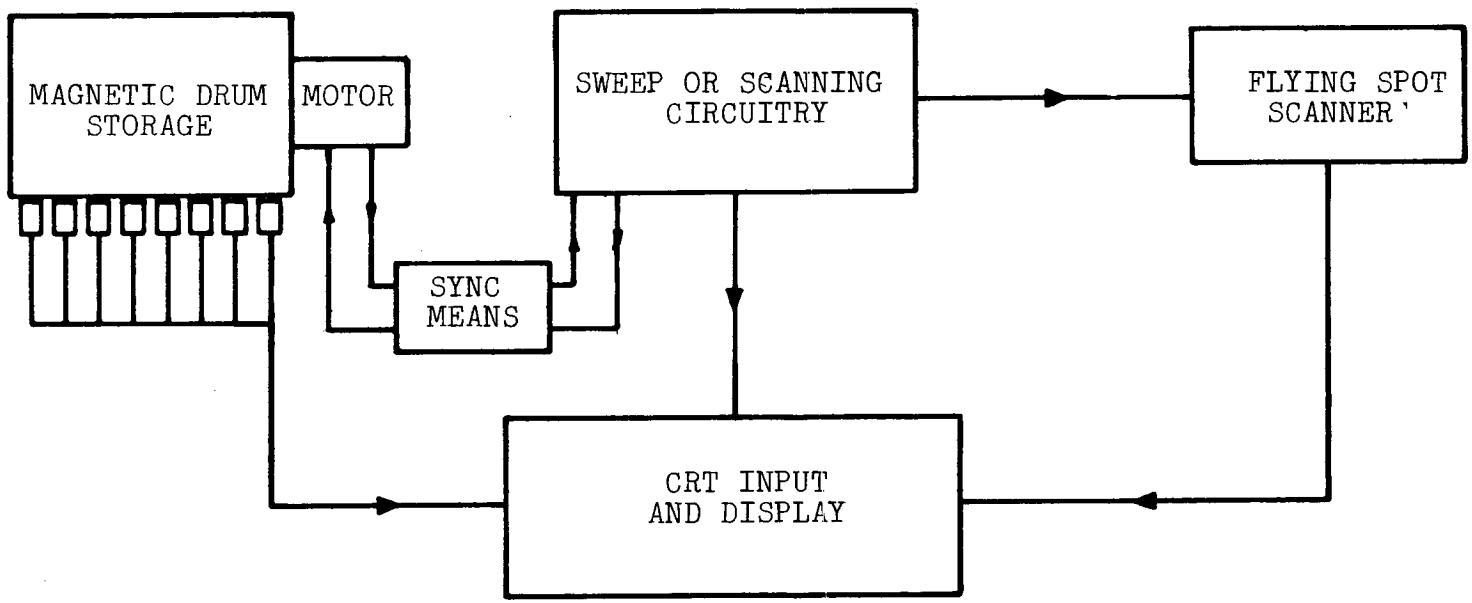



FIGURE 6

Fig. 6. The Blink Technique of Pattern Recognition as applied to the checkout of the memory portion of a digital computer system. The scanning of the electron beam on the face of the CRT is synchronized with the scanning of the storage elements on a magnetic drum, a core matrix, or the like. The read-out of each bit of data in the storage medium is delayed in time so that the CRT displays a matrix of points which corresponds to the position at which the data is stored. This "electronic development" of the surface of the magnetic drum is repeated fast enough so that a steady image is seen by the operator. In between each scan of the data in the memory, there is alternately presented an image of the comparison data previously stored on a film chip. Towards this end the raster generated by the flying spot scanner is also synchronized with the rotation of the memory drum. When the pattern of the magnetic drum and that of the film chip is properly matched the display is continuous, or even. When there is a discrepancy in the patterns, a blink occurs at the position where the discrepancy exists. This blink may be observed in position, in brightness, and in color variation.

## VII. SUMMARY

A review of the preceding descriptions will show that the major objectives set forth earlier have been achieved. A multi-color easy-to-interpret pattern of test data has been displayed on a cathode ray tube. The test data which is displayed during a malfunction is retained on photographs taken with color film, or is recorded in some other way. These photographs preferably are taken during the design stages of the prime equipment and are used in the preparation of a handbook, or library, of predicted drift and failure patterns. Experimentation and experience augment this library so that with time it becomes full and complete, even in the presence of design modifications. The information thus stored in the handbook is provided for manual use by a human observer; and that stored in the library is in machine form for use with electro-mechanical or electronic information-retrieval systems. An example of a punched card information-retrieval system was described which makes searches of the failure patterns, selects failure cards corresponding to faults in the unit under test, and projects the failure patterns upon the CRT so that a comparison may be made between the test data furnished by the unit under test and the pre-stored data.

A television type CRT display, employing NTSC color signals, generates a rectangular raster pattern of 525 horizontal lines at a 16kc rate. The display can be designed for 400 active horizontal lines with ease. A 20 inch CRT (measured in the horizontal direction) with vertical color strips that are each 40 mils wide provides a vertical resolution of 500 lines. If 450 of these lines are active, and if a three color arrangement is used for the display of the test data, it is clear that $400 \times \frac{450}{3}$ or 60,000 test points can be presented for display. Even with a five color display of the data, each color having its own vertical color strip, there would be room for $400 \times \frac{450}{3}$ or 36,000 test points. This amount of data is substantial and obviously would be derived from a complex system comprised of many equipments and assemblies. The display of the data is arranged so that different "blocks" of the matrix correspond to the different equipments, assemblies, or major components making up the system under test. A color change, or a blink, at a single point in this matrix of 60,000 test points will show up as a distinct disturbance in an otherwise regular pattern. Many techniques can be used for investigating this disturbance. The technique described entails the projection of a pre-recorded pattern upon the section of the test matrix where the disturbance is located. Another technique might use a second CRT where a blown up view is presented of the disturbed area. In an extremely large or major system where 600,000 points of test data are being generated, this greater amount of data can be displayed by improving the resolution of the CRT, by using a larger display, and by going to projection type CRT's. Alternatively, the test patterns may be controlled so that only those that are defective are transmitted for display. This transmission may be to a central monitoring station where it is desired to observe and evaluate the test results of the overall major system.

## VIII. CONCLUSIONS

It should be evident from the foregoing that it is proposed, via these pattern recognition displays, to enhance the role played by operating and maintenance personnel in the day-to-day use of complex electronic systems by providing a more attractive man-machine interface. This might be done in the aerospace environment by time-sharing some of the lesser used displays already contained in the vehicle. For example, a display used for airborne rendezvous or for a blind landing system can be used for maintenance purposes for extensive periods of time without compromising its original intended application. There are undoubtedly other displays which can be used. If the original equipment display is small in size; then the data it is called upon to display is small; perhaps so that it only tests itself. This display of the data in block format makes it possible for judgements to be rendered which presently are not possible. The pattern recognition type of display lessens the need for the numerous dials and digits that are characteristic of present maintenance efforts. The conventional voltmeter, the digital voltmeter, the oscilloscope, and the mechanical printer with its reams of machine-oriented data have been replaced with a real-time multi-color display. The instant arrangement yields a simultaneous display of significant data from tens, hundreds, or thousands of test points. The data presented is up-dated 60 times each second, the data is quantized so that crisp changes in the display are made whenever test signals change their quantum levels. This new test system thus can provide detailed status displays needed for maintenance purposes. At the same time, it may provide a central display for making the operational decisions that depend upon equipment availability and equipment performance. These decisions be they maintenance or operational are made by humans; and the information on which these decisions are made is displayed in the foregoing arrangements in a format well suited to human comprehension.

March 17, 1969

# OUTGASSING PHENOMENA OF MATERIALS IN SPACE

Paper Presented at the
Symposium on
Long Life Hardware for Space

by

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# OUTGASSING PHENOMENA OF MATERIALS IN SPACE

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## ABSTRACT

The outgassing phenomena of materials subjected to extended environments of space are discussed. Also discussed are the stability of organic materials, desorption of absorbed surface gases, and the molecular reaction that occurs changing their properties. Prior knowledge and understanding of the outgassing characteristics of materials used in long-life hardware for space vehicles are essential. Changes in material properties can occur and result in equipment malfunction, and optical surfaces can become contaminated to the point that transmission and reflectance characteristics are impared at high energy infrared and low energy ultraviolet wavelengths. Experiments, tests equipment, and procedures that have been performed to date by Astrionics Laboratory are briefly discussed.

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## SECTION I. INTRODUCTION

Presented in this paper is a brief description of the outgassing phenomena of materials subjected to extended environments of space. The materials are used on the Apollo Telescope Mount (ATM) and other Apollo Applications Program (AAP) hardware.

Contamination from materials is very important because the optical surfaces could be coated, affecting the success of the mission. Outgassing of materials at ultra-high vacuum levels is considered a potential hazard because of the particularly detrimental effect to optical equipment, such as telescope lenses, solar cells, and photographic lenses which are used on the ATM. For this reason, outgassing tests of polymeric materials at a vacuum to $10^{-8}$ torr have been conducted on many of the dielectric materials qualified under Marshall Space Flight Center (MSFC) electronic packaging specifications.

When dielectric materials are used in space vehicles, it is essential that prior knowledge of their characteristics be available because outgassed products will contribute to optical problems or electrical malfunctions. Electrical malfunctions may be caused by the formation of an insulating film on contacts, affecting electrical continuity.

Dielectric and other materials considered for use in the ATM were tested for percent weight loss, steady state weight loss, and rate of weight loss during temperature increase. The materials considered for use were subjected to vacuum levels to $10^{-8}$ torr and temperatures to +100°C. Gas analyses were conducted on materials using a Diatron or an Aero Vac mass spectrometer.

An infrared spectrum has been made on some of the materials tested, and a structural formula is given for a typical material. The structural formula for materials can be determined, based on the infrared spectrum.

Residual gas analyses have been conducted on several materials which have previously met outgassing requirements. Information received from these analyses is presented as relative intensity in percentage versus mass weight. The infrared spectrum, the structural formula, and the gas analysis mass weights were used to identify the higher mass weight outgas products. The lower mass weight gases, such as oxygen, nitrogen, and water vapor, are readily identifiable when using this technique.

Charts, figures, photographs of test equipment and operational schematics, as well as test data, are presented in this paper.

## SECTION II. CHEMISTRY OF ORGANIC ELECTRONIC PACKAGING MATERIALS

### A. POLYMER CHAIN SCHEMATIC

A long chain unbranched ploymer, a branched polymer, and a cross-linked polymer are shown in figure 1.

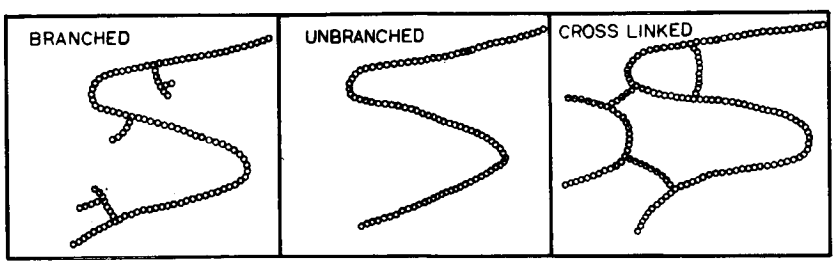



Figure 1. Polymer Chain Schematic

### B. LADDER POLYMERS

Electrical insulation has many formulations and many different physical properties. Many electrical properties can be determined by extrapolation with other materials that have similar chemical structures and the knowledge of the differences in their chemical structures. For example, polyvinylchloride, polyethylene, polystyrene, and polypropylene all have the same basic carbon-carbon chain structure, with differences being in the chemical groups attached to or branching off from the basic chain.

The ladder polymers (figure 2) are a chain of polyhedra, often hexagons. Perfluorocarbon, silicone, and polyimide materials have become available. The perfluorocarbons have the same basic carbon-carbon skeleton. The fluorine groups that are attached shield them from attack. The silicones have a different skeleton structure. More energy is required to rupture the silicone-oxygen bond than the carbon-carbon bond. The monomers or reactants for these polymers are different from other polymeric precursors because of the following:

(1) They are all ring rather than chain structures.

(2) Instead of having reactive heads and tails that can link with other heads and tails to form the conventional polymer, they have in effect at least two heads and tails so that the ladder structure can grow without any weak links.

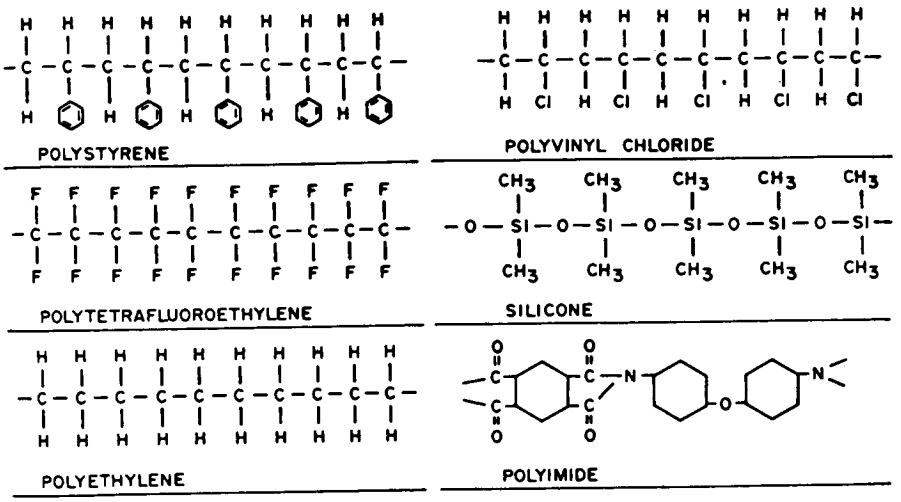



Figure 2. Ladder Polymers

An advantage of a ring structure is that when it is hit with an overload of energy (heat or radiation), the energy is distributed all around the ring and absorbed by the ring rather than by the individual atom that is hit. The same amount of energy hitting a long chain structure will usually cause rupture or scission, while a ring structure often absorbs the energy without degradation. This property accounts for the exceptional radiation resistance and thermal stability of ladder polymers and explains their flexibility at cryogenic temperatures and lack of a melting point at elevated temperatures.

## SECTION III. MATERIAL PROPERTY CHANGES VERSUS ENVIRONMENT

Many physical properties of materials undergo changes due to the loss of fragments through outgassing. The molecular structure of materials as a result of heat and vacuum sometimes undergo changes which result in property changes. Examples of changes in a polyimide material (one of the most heat resistant organics known) due to heat are shown in figures 3 through 7. As materials are exposed to high temperature, they lose many of their physical properties and, when returned to lower temperature, they sometimes regain their original properties; however, if molecular fragments have broken off the structure due to heat and vacuum, as typically happens in outgassing, the physical properties are never regained since molecular rearrangement has taken place.

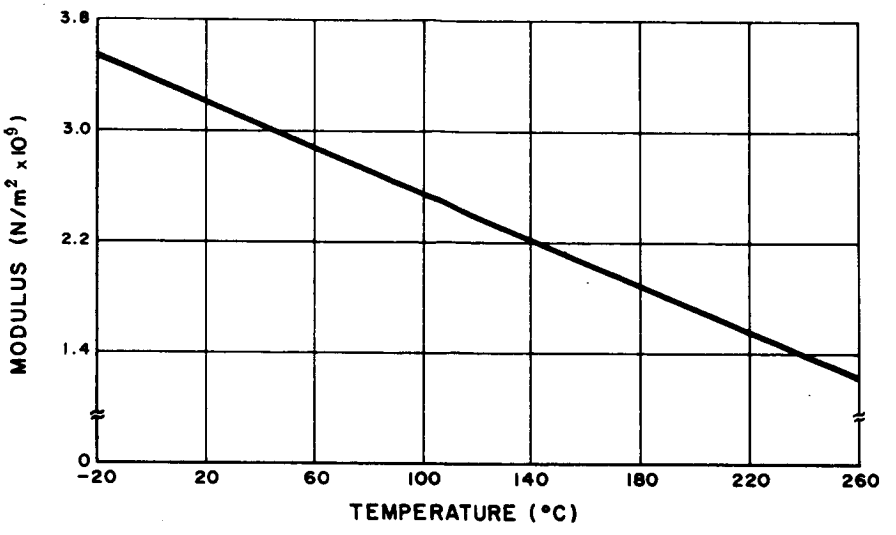



Figure 3. Polyimide Modulus versus Temperature (Typical)

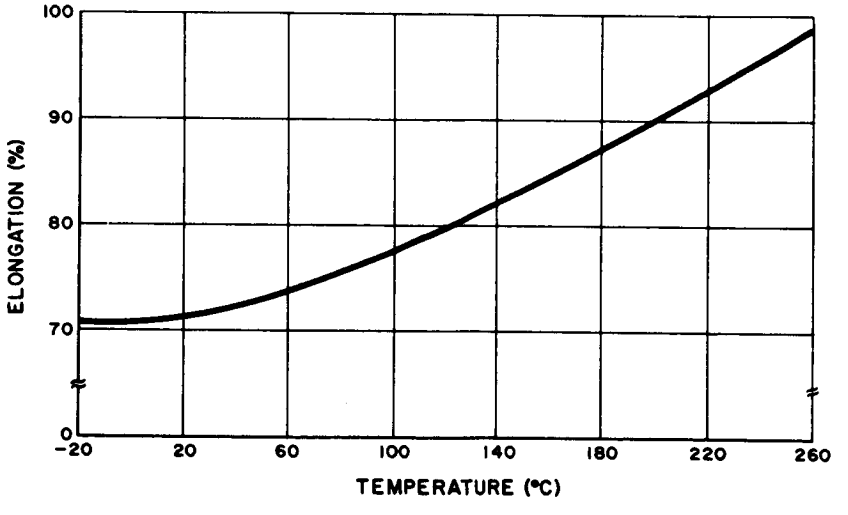



Figure 4. Polyimide Elongation versus Temperature

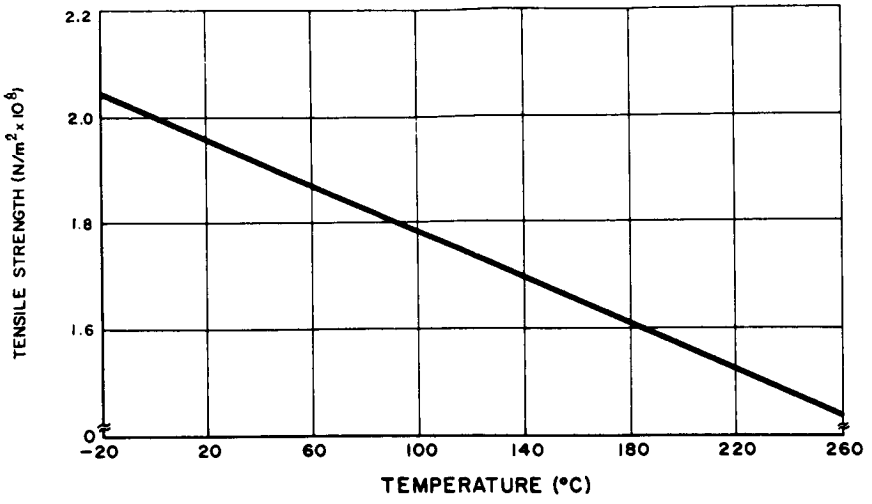



Figure 5. Polyimide Tensile Strength versus Temperature (Typical)

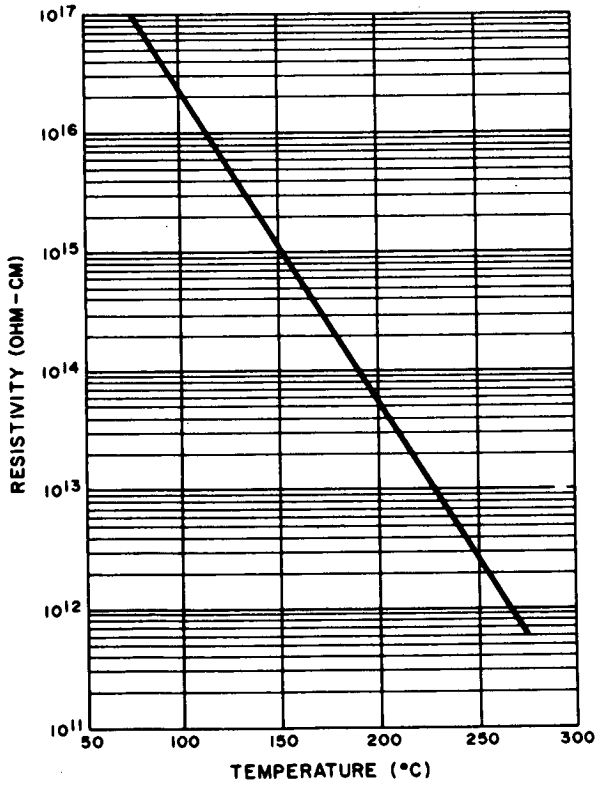



Figure 6. Polyimide Resistivity versus Temperature (125 VDC)

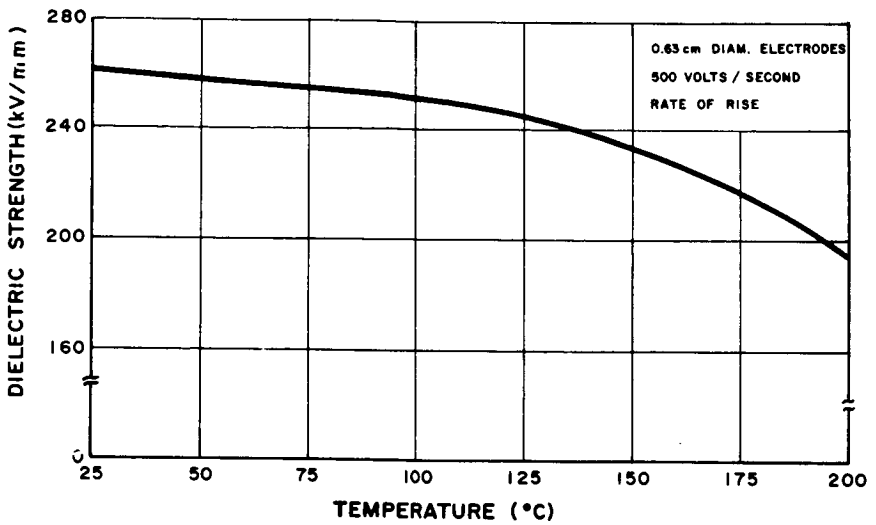



Figure 7. Polyimide Dielectric Strength versus Temperature (0.0254 mm)

The electrical properties and thermal stability of materials at ambient temperature are comparable for such dielectrics as Kapton and Teflon. At 400°C, it was found that Teflon loses weight at a greater rate than the polyimides (figure 8), and its total decomposition occurs before the polyimides.

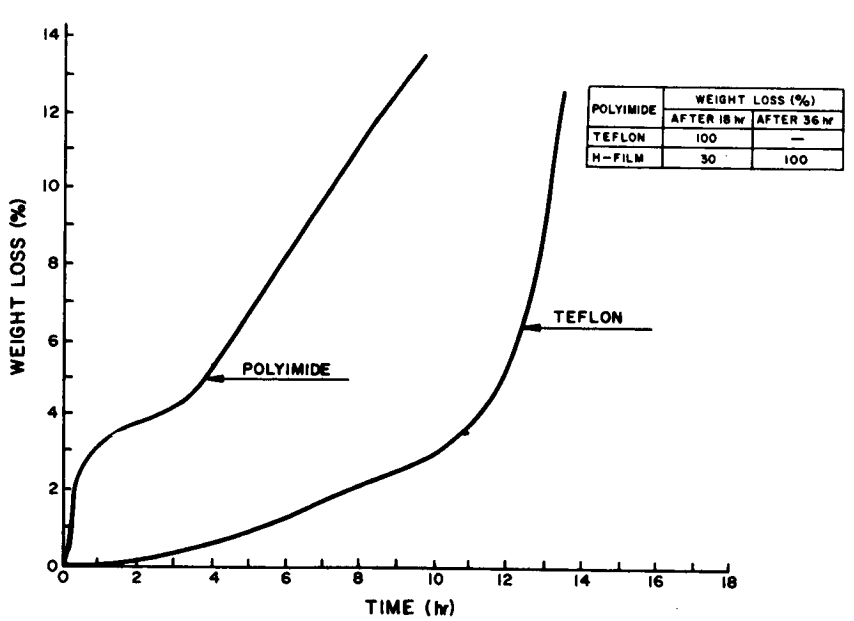



Figure 8. Thermal Stability Comparison at 400°C

## SECTION IV. WHY ORGANICS OUTGAS IN SPACE

Organic dielectrics will outgas in space because of vacuum, heat, and radiation. There are four basic reasons that cause an organic material to lose weight in space and these are given in the following listing:

(1) Chain scission
(2) Cross linking
(3) Absorption
(4) Improper catalyst-to-resin ratio

Chain scission is breaking the molecular bond that holds the chain of the polymers together. When chain scission occurs, the material is usually cheesy, crumbly, and has very little strength. Further cross linking can take place and is the molecular bonding that occurs between the chains of the polymer. This usually results in a polymer becoming very hard and brittle. Absorption occurs in polymers and is the result of the polymer absorbing either atmospheric gases or organic solvents. These absorbed materials usually are the first materials to leave a polymer in a thermo-vacuum environment. A polymer can outgas due to improper weighing or mixing of the catalyst and the resin, which will upset the proper catalyst-to resin balance. A polymer, such as the epoxy shown in figure 9, will usually outgas because of a broken molecular bond caused by heat and vacuum.

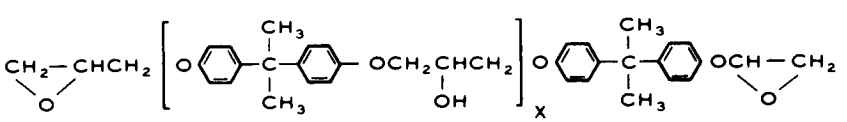


Figure 9. Epoxy Structural Formula

## SECTION V. CHANGES IN MOLECULAR STRUCTURE OF ORGANICS DUE TO OUTGASSING RESULTING IN MOLECULAR FRAGMENT OUTGASSING IDENTIFICATION

With over one thousand different materials anticipated for use in ATM equipment, it is possible to synthesize new polymers and copolymers from outgas products in space environment. This phenomenon can occur when gaseous products are mixed and deposited on a surface in the presence of ultraviolet radiation.

As an initial investigation of this possibility, typical epoxy, silicone, and polyurethane were chosen from the materials subjected to both weight loss test and mass spectrometer analysis. The chemical structural formula for each material was obtained as accurately as possible. Infrared spectroscope analysis techniques were used to identify the materials by comparison with infrared spectra of known materials.

## A. IDENTIFICATION OF EPOXY OUTGASSING FRAGMENTS

An epoxy was identified by comparing its infrared spectrum with published data. It was found to be Diglycidal Ether of Bispheral A 2, 2-bis (P-2, 3 epoxypropoxy) phenyl-propane shown by the structural formula in figure 10.

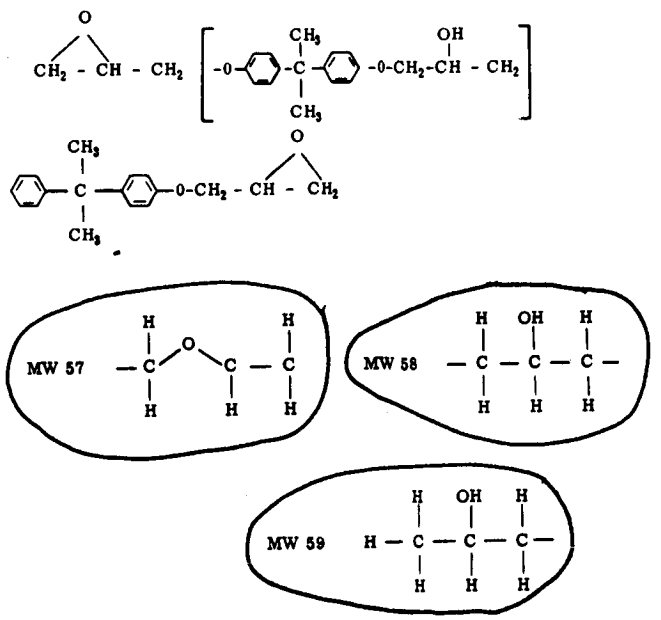



Figure 10. Epoxy Outgassing Fragments

Mass weights 57, 58, and 59 were detected in the outgassing of an epoxy using the mass spectrometer. Based on the chemical structure of the epoxy, the molecule is believed to have broken at the ether bond which resulted in the outgas products of mass weights 57, 58, and 59. These ions are believed to have the structural formula shown in figure 10.

## B. IDENTIFICATION OF SILICONE OUTGASSING FRAGMENTS

The structural formula for silicone RTV was identified, by its infrared specturm, as a methyl phenyl silicone having a general formula as shown in figure 11.

Mass weights of 73 were detected in the outgas of this material by the mass spectrometer. Based on the chemistry of a methyl phenyl silicate, this outgas product is believed to have the chemical structure shown in figure 11.

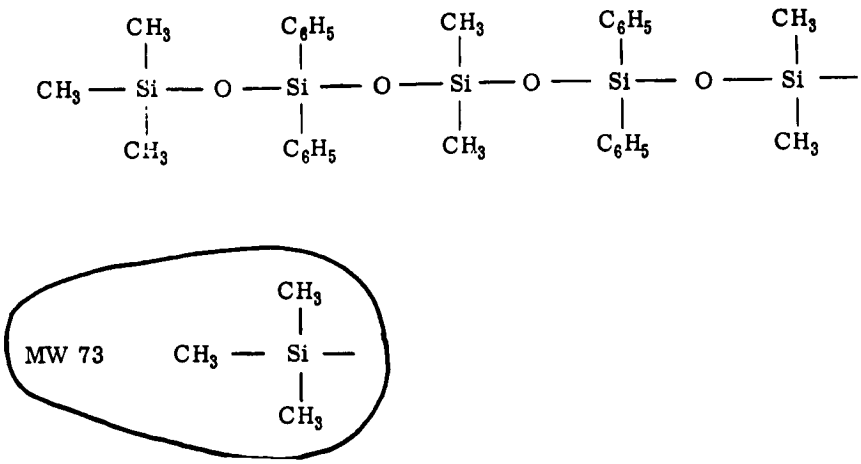



Figure 11. Silicone Outgassing Fragments

## C. IDENTIFICATION OF POLYURETHANE OUTGASSING FRAGMENTS

The structural formula for specific polyurethanes was identified by their infrared spectrum. The formula shown in figure 12 for a given polyurethane is based on the infrared spectrum. Mass weight analysis of the outgassing fragments was made using a Residual Gas Analyzer.

The urea groups formed by a diamine cure exhibit strong interchain attraction. This attraction functions as a cross linking that reinforces the polymer and aids in developing high strength.

When heat is applied in the outgassing process of polyurethane, the reactions shown in figure 12 take place.

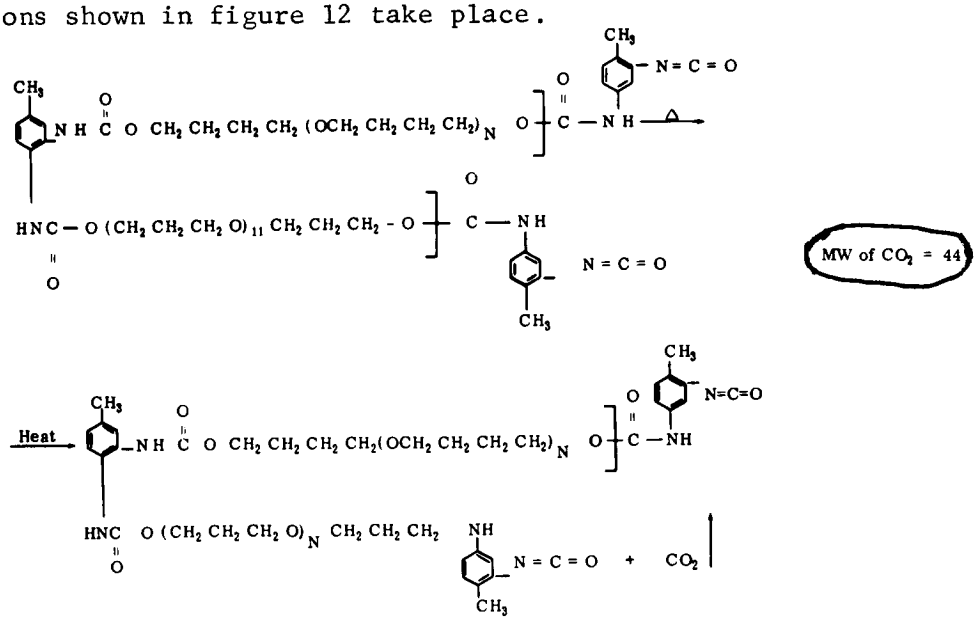



Figure 12. Polyurethane Outgassing Fragments

Mass weight above 44 was not detected in the outgassing of the polyurethane by the mass spectrometer. Based on the chemical structure of the polyurethane molecule, it is believed to have broken at the urea bond, giving off $CO_2$ which has a mass weight of 44.

## SECTION VI. TEST EQUIPMENT AND RESULTS

The test equipment and results of the contamination program for the ATM program are given within this section.

### A. WEIGHT LOSS TESTS

Since many dielectric materials experience weight loss when exposed to low pressure and heat, the vacuum environment of the ATM limits the use of these materials. Organic polymers subjected to these pressures may lose weight in two ways as follows:

(1) Because of increased evaporation of volatile materials and absorbed gases at low pressures. (An unimportant consideration relative to long chain polymeric materials, but an important factor for some of the additives which are combined with polymers.)

(2) Because of decomposition of the basic polymer by breaking down the chain into low molecular weight fragments which are subsequently lost through evaporation.

The tests were performed to qualify materials from the standpoint of outgassing in a vacuum at temperatures up to 100°C. The materials used in the ATM shall have a steady state rate of weight loss at 100°C of not more than 0.04 percent per $cm^2$ per hour. The rate of weight loss during temperature increase shall not exceed 0.2 percent per $cm^2$ per hour. Steady state is defined as that point where the rate of weight loss has been constant for eight hours.

Results obtained from the outgassing program are illustrated in weight loss versus time and vacuum curves. (Refer to paragraph C.) In the initial portion of the weight loss versus time curve for polymers exposed to vacuum conditions at elevated temperature, a substantial weight loss is shown. This initial high outgassing rate is caused by loss of absorbed water and other atmospheric contaminants, as well as some degradation of the base polymer. After these products are released, the outgassing rate will decrease. In the latter part of the curve, outgassing is shown at a minimum which may be caused by degradation of the base polymer.

The sample temperature (100°C) is maintained by the use of a Thermac temperature and power control center. The output of a 250-watt infrared bulb (controlled by the Thermac) is reflected into the electronic balance and onto the sample and a thermocouple which is connected to the

Thermac. The thermocouple and the sample are at the same temperature, with the thermocouple acting as the temperature sensor for the Thermac. The temperature of the sample is raised at a rate of 2°C per minute and maintained at 100°C for the duration of the thermovacuum.

B. TURBO-MOLECULAR VACUUM PUMP

The turbo-molecular pump used in conducting the outgassing tests is equipped with all-metal seals, which are highly desirable over elastomer seals because of outgassing and the tendency of the elastomer to absorb solvents and moisture.

This pump is the most suitable vacuum pump for studying contamination. Stator bearings are cooled with a refrigeration system, eliminating the need for excessive lubricating oil in the bearings which could possibly cause outgassing problems.

The 16,000 rpm drive speed of the pump is obtained in approximately five minutes from the time the pump is energized, and the pump is capable of evacuation to $1.0 \times 10^{-9}$ torr; however, this vacuum level was not obtained within the 24-hour test period. Some samples were subjected to vacuum for 72 hours.

A diffusion pump is not recommended for outgas studies because oil in the pump may become overheated, thereby permitting back-streaming to the sample area and contamination of the sample and balance equipment. An ion pump is also not recommended because the electrodes may become coated with outgas vapors or with back-streaming oils, causing vacuum to be lost.

C. ELECTROBALANCE

The Cahn, model RG 200, electrobalance was used for measuring weight loss occuring under vacuum during the tests. The instrument is sensitive to a 0.1 microgram weight differential. Although the electrobalance is capable of accommodating a maximum sample weight of 2.5 grams, the samples used during the outgassing tests weighed 0.2500 gram.

D. WEIGHT LOSS DATA

Weight losses occuring during the tests were recorded on a Leeds and Northrup Speedomax recorder. An extrapolation of the graphs obtained, which provide a graphic portrayal of weight loss versus time and vacuum, is shown in figures 13 and 14.

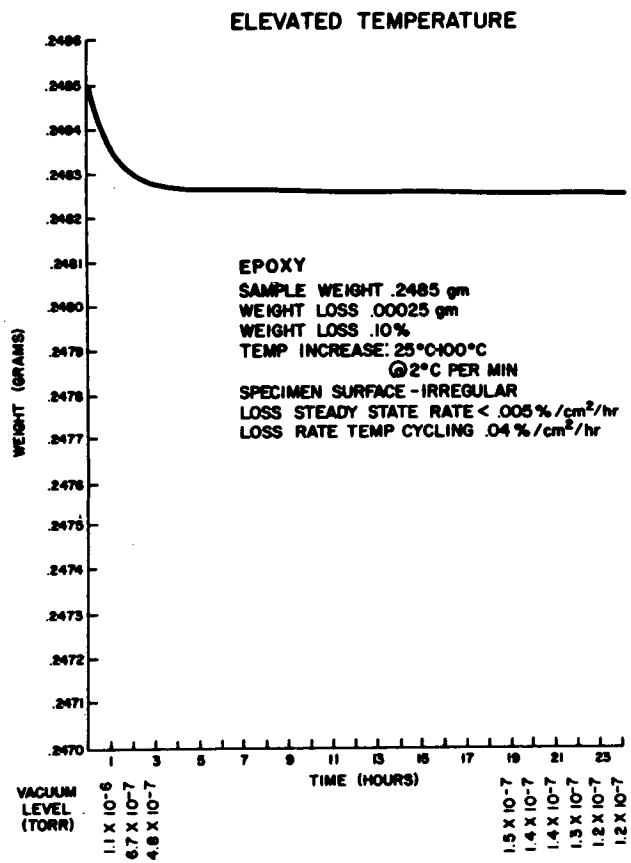



Figure 13. Weight Loss and Outgassing Curve for Epoxy

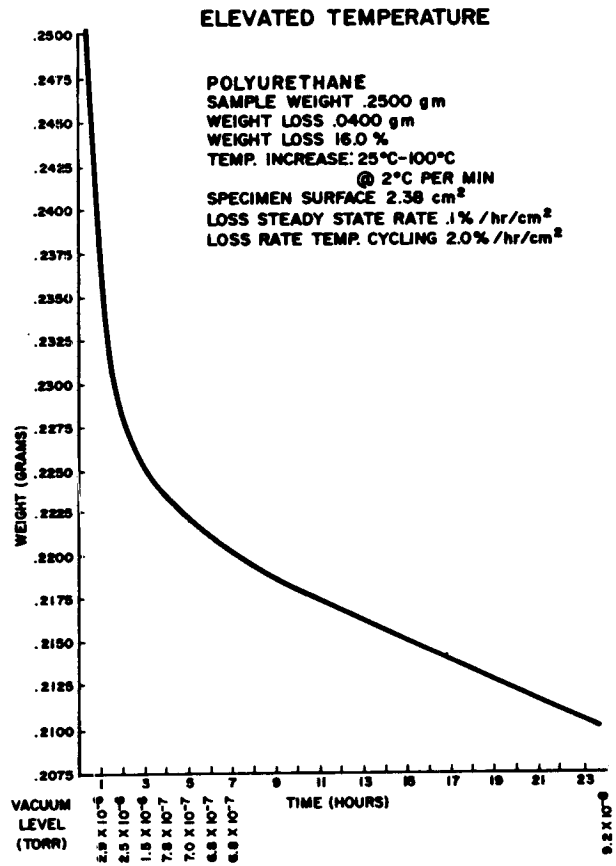



Figure 14. Weight Loss and Outgassing Curve for Polyurethane

## E. GAS ANALYSIS EQUIPMENT

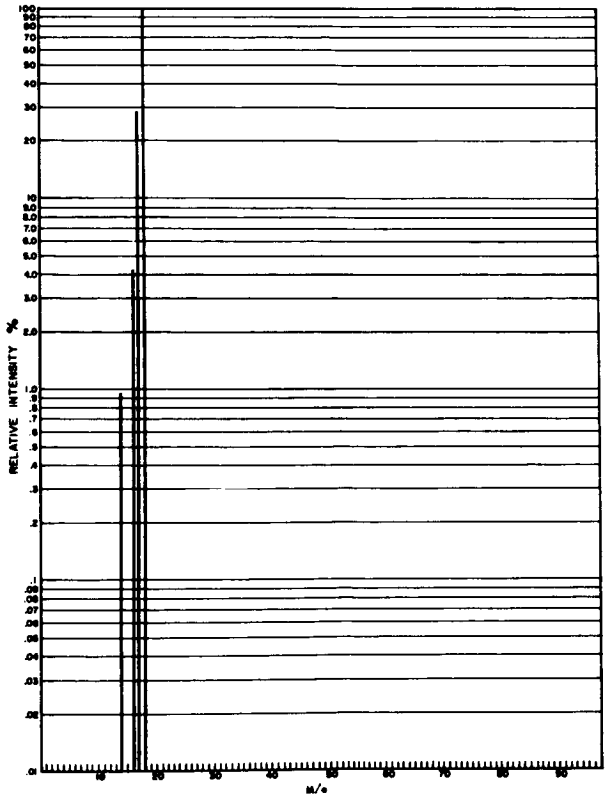
Diatron. Gas analysis of the outgassing fragments are accomplished by using a Diatron residual gas analyzer, which is essentially a mass spectrometer without the vacuum system. The instrument is designed for continuous analysis of gas in evacuated systems. When installed on the system to be monitored, the Diatron is capable of measuring minute quantities of gas, gas mixtures, and vapors up to a mass range of 250 atomic mass units (AMU).

Since an atom infrequently loses more than one electron (e), the value of e is normally one. By admitting a very small amount of the gaseous material into the evacuated analyzing section, the analysis of a sample is accomplished. Within the Diatron, a stream of electrons is emitted from an incandescent tungsten filament and flows to an adjacent anode with considerable energy. The gaseous sample is emitted to this environment and intensely subjected to destructive bombardment of the electron beam. When electrons strike the molecules of the sample gas, they exert a force on the atoms in the gas and break the bonds which hold the atoms together as a molecule. The molecules lose one or more electrons, usually one, thus becoming positively charged or ionized. Ions are then identified by ratio of the mass of the unionized particles to the number of electrons they lose (m/e). Mass is designated numerically by its atomic or molecular weight, e.g., the mass of hydrogen is 1.008 since the atomic weight of hydrogen is 1.008, or for all practical purposes, one. There are only certain points at which molecules will break and become ions; therefore, the number and types of ions which can be formed from the small gaseous sample are limited. The mixture of ions must be separated into groups, with each group consisting of only one specific mass. This separation or group collection is accomplished by applying electric or magnetic fields. The magnetic field segregates the ions into beams, with each beam containing a different m/e. This phenomenon exists because the heavier ions travel in circles of larger radii than the lighter ions. The ions are forced into a collector where the particles release their positive charge by admitting electrons. The resultant current flow is passed through a high resistance, developing a voltage which is then amplified and recorded as a graph.

## F. GAS ANALYSIS SYSTEM AND DATA

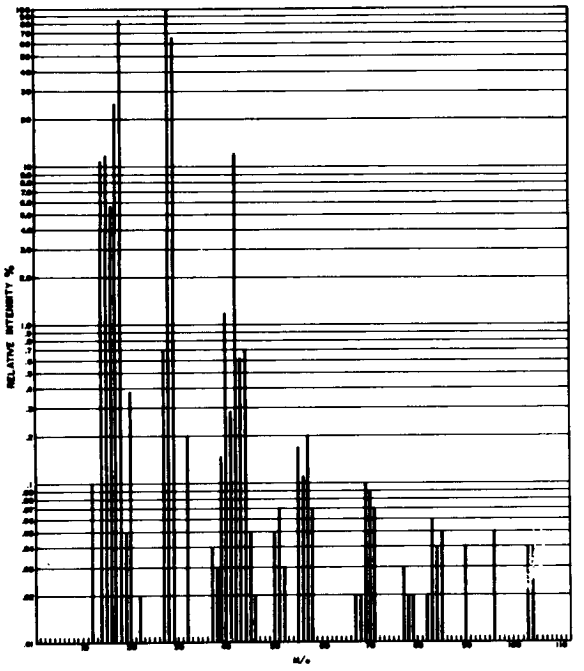
Residual gas analyses are also conducted using a Aero Vac RGA. The RGA system consists of a Welch turbo-molecular vacuum pump, stainless steel chamber with a thermostatically-controlled radiant heater, and an Aero Vac, model 610-611, mass spectrometer. The readout is to a strip chart recorder (L&N, model W) with a semilog scale. Chart readout data have been calculated and replotted as shown in figures 15 through 18.

The heart of the RGA system is the mass spectrometer which analyzes the gas mixture that has escaped from the specimen and moves in the direction of the vacuum pump by ionizing some of the molecules and separating the ions according to their charge-to-mass ratio. The theory of operation of the Aero Vac, model 610-611, mass spectrometer is essentially the same as the Diatron.



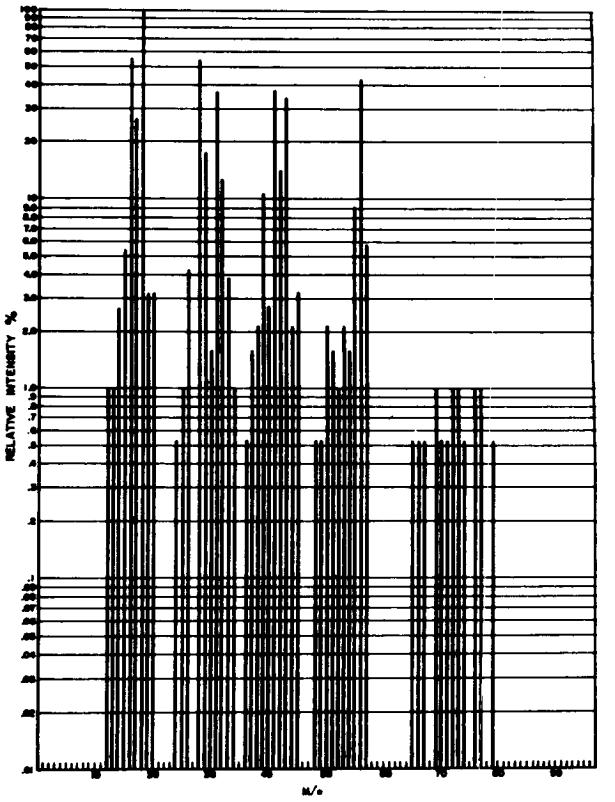

EPOXY

Figure 15. Gas Analysis Peak Heights of Epoxy



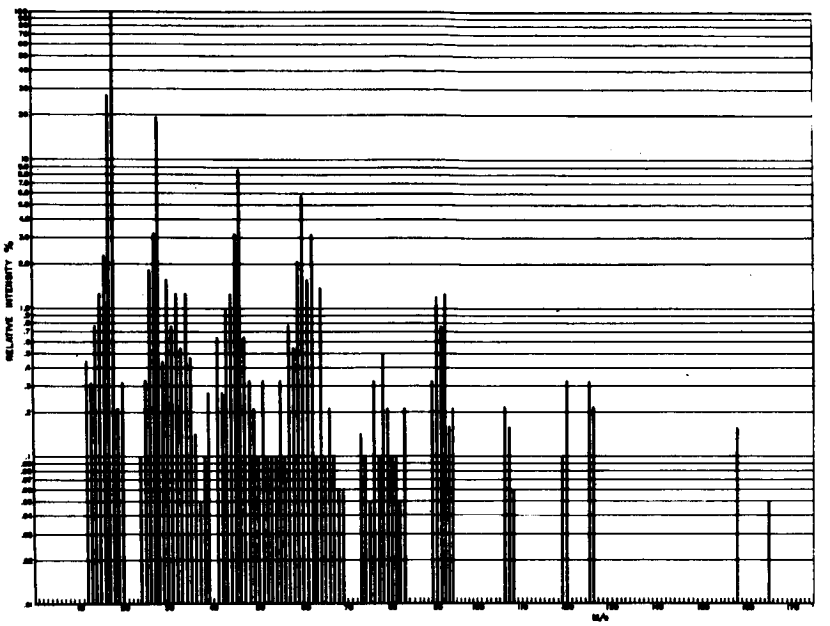

POLYURETHANE FOAM

Figure 16. Gas Analysis Peak Heights of Polyurethane Foam




EPOXY

Figure 17. Gas Analysis Peak Heights of Epoxy



POLYSULFIDE RUBBER

Figure 18. Gas Analysis Peak Heights of Polysulfide Rubber

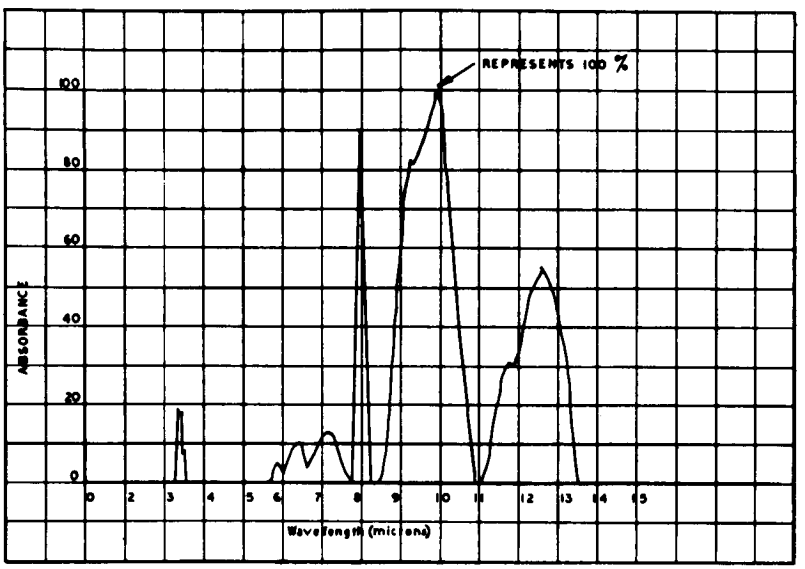
G. OPTICAL COATING

The light transmittancy of an optical coating will change as the wavelength changes from infrared to visible to ultraviolet. Equipment aboard the ATM will operate in the ultraviolet and X-ray portions of the electromagnetic spectrum. Therefore, measurements must be made at all of the various operating wavelengths of the telescopes, sun sensors, and other optical experimental equipment contained on the ATM. The light transmittance for all of the various wavelengths is not the same as the transmittance for the visible light range of 0.4 to 0.6 microns.

An infrared spectrophotometer operates at 2.5 to 25 microns wavelength. Since this is true, a problem exists in measuring the light transmission in the infrared range since the materials will absorb a great deal of the light energy. Quartz is opaque from $0.5 \times 10^{-2}$ to $0.5 \times 10^{-3}$ centimeters wavelength in the infrared region and $1.6 \times 10^{-5}$ to $0.5 \times 10^{-8}$ centimeters wavelength in ultraviolet and X-ray regions.

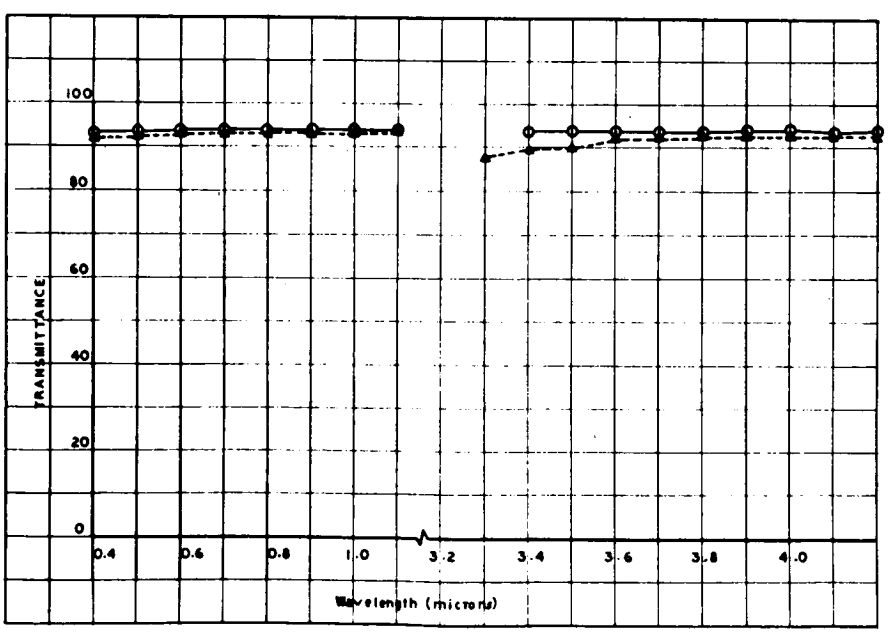
## H. LIGHT ABSORPTION AND TRANSMISSION DATA

The infrared absorption and the ultraviolet and infrared transmission effects of RTV silicon contamination deposits on optical lenses are shown in figures 19 and 20.




RTV SILICONE

Figure 19. Infrared Absorption of RTV Silicone




RTV SILICONE

Figure 20. Ultraviolet and Infrared Transmission of RTV Silicones

## I. SIMULATED SPACE ENVIRONMENT CHAMBER

A space environment chamber has been designed and fabricated to simulate ultraviolet, thermal, and ultrahigh vacuum space conditions on dielectric materials.

The ultraviolet source consists of a Xenon lamp with input power of 5500 to 6000 watts and chilled water cooled in a quartz enclosure. The lamp produces high-intensity ultraviolet radiation in the wavelength range of 2800 to 4700 Angstroms. With the quartz enclosure, the radiation is very close to natural sunlight rays at noon on a June day except for higher intensity.

The specimen temperature is thermostatically controlled and programmed for the desired rise by use of a radiant heater inside the vacuum chamber, which has capabilities to 600°C. Opposite the heater and near the exhaust port of the vacuum chamber, a mirror or glass slide is placed in contact with a chill plate of $LN_2$ temperature. Liquid nitrogen is circulated through coils at the back of the chill plate.

The ultrahigh vacuum chamber is of stainless steel construction and is completely welded except for two eight-inch ports. One port is provided for admission of ultraviolet rays into the chamber; the other port is for viewing and accessibility. The vacuum is provided by two cryosorption pumps connected to the chamber by a six-inch stainless steel tube from the main pump and a one-and-a-half inch tube from roughing pump. The cryosorption pumps are constructed of stainless steel and contain baffles coated with molecular seive materials with cryogenic coils attached for cooling of the seives. With no mechanical or electrical power used, the pumps are operated with liquid nitrogen and liquid helium which reduces the temperature of the molecular seives and causes them to absorb and retain the molecules within the chamber, which produces a vacuum.

The simulated space environmental chamber contains environments of ultraviolet radiation, elevated temperature, low temperature, and vacuum combined.

When a test specimen is placed in the chamber and the test begins, the ultraviolet rays pass through the quartz window and into the chamber where the specimen outgas products are evolving and moving to the chill plate and vacuum port, which is caused by heating of the specimen in vacuum. As outgas products move through the ultraviolet rays in gas form, a change in chemical structure can take place, and new materials can be formed, as a result of the intermolecular reactions, and collected on a glass slide in contact with the chill plate. The plate is located in the path of gases flowing to the vacuum port. The analysis is conducted for light transmittancy and reflectance through light pipes or by the removal of glass slide for microscopic examination and infrared analysis.

This investigation of the behavior of polymeric materials in simulated space environments shows the reaction of the outgas products when they sublime on spacecraft windows, mirrors, electrical contacts, etc. It is important that these facts are known and necessary steps taken to prevent contamination of critical areas outside as well as inside the spacecraft.

## J. GAS ANALYSIS BY GAS CHROMATOGRAPH AND INFRARED SPECTROPHOTOMETER

Outgassing fragments will also be analyzed using a gas chromatograph and infrared spectrophotometer. The organic sample is placed inside the vacuum chamber and heated to 100°C by a thermocouple heating element. The vacuum is obtained from the turbo-vacuum pump to $10^{-7}$ torr. As the sample outgasses, the main vacuum valve is closed and a bypass valve is opened. The outgassing contamination is pumped into a holding tank. The vacuum valves are closed and a valve leading to the gas chromatograph is opened. The helium valve is then opened and the holding tank is purged with helium gas at 30 psi. The contamination is carried into the injectors of the chromatograph by the helium and then into the gas column inside a programmed oven where the contaminations are separated by heat in the vapor state at different temperatures. The separated gas then moves into the manifold where the hot wire detection is mounted. A recorder connected to the gas chromatograph indicates when a sample of the contamination gas is passing through and a peak is recorded. The sample then passes to a gas sampling valve and then into a gas cell mounted in the grating infrared spectrophotometer. The sample is exposed to the infrared and an absorption spectrum is recorded on the infrared spectrophotometer chart. This chart is then compared to a Standard infrared chart, and the identification of the outgas fragment is accomplished.

## K. LIGHT REFLECTANCE MEASUREMENT OF OUTGAS DEPOSITS

This test will be performed in a distribution box. The test configuration for the distribution box technique consists of a setup as shown in figure 21. The sample is placed inside a cylindrical box with only three holes (one directly over each of the condensing surfaces). The three condensing surfaces used are an aluminized mirror (for specular reflectance measurements in the near ultraviolet and visible region), a KBr or NaCl disc (for infrared transmission), and Chance microsheet (to determine the effect of the condensate on optical resolution). The reflectance off the aluminum mirror is monitored continuously in situ during the 102-hour condensability test with a 1P28 phototube and a 0.4 to 0.6 micron bandpass filter. Spectral reflectance measurements of the mirrors are made between 0.35 and 0.75 micron before and after the condensability test. Infrared transmission is conducted from 2.5 to 15 microns on the infrared analysis discs. The Chance microsheet from the test and a control piece of microsheet are put over an optical resolution chart (equally spaced black and white lines) to determine any change in effective resolution in lines per milimeter due to the condensate. If any change is noted, it will be photographed for future references and reporting purposes.

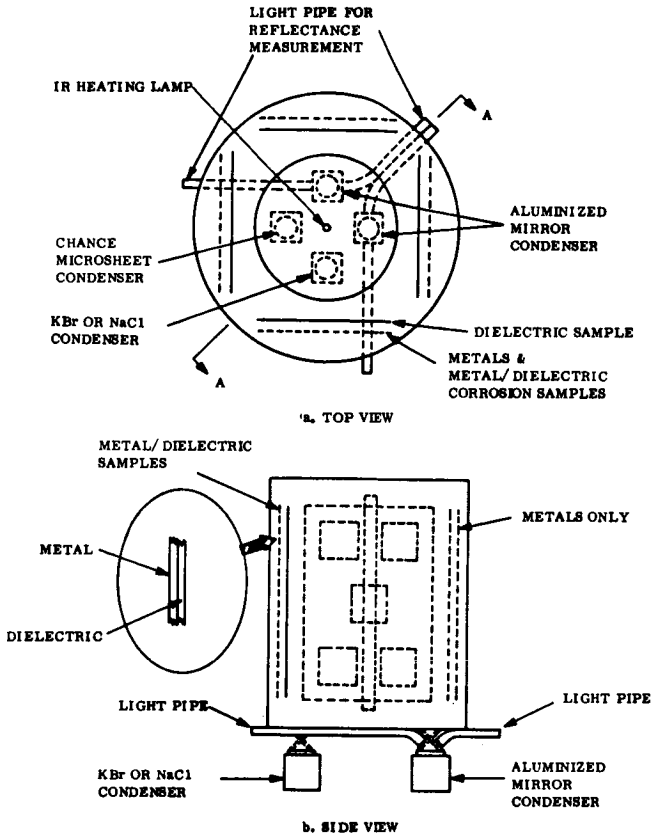



Figure 21. Light Reflectance Apparatus

The sample is heated by radiation from the box, which is heated with resistance heaters, and by radiation from a quartz-jacketed infrared lamp centered in the box. The temperature of the sample is regulated with an imbedded thermocouple so the sample is always less than or equal to the minimum box temperature, which is obtained from numerous thermocouples mounted on the box. The outgassed material is directed onto the four cooled condensers through the holes cut into the bottom of the box. The condensers are maintained at temperatures typical of spacecraft optics.

The principle by which the distribution box increases the amount of the condensate is to enable large sample surfaces to contain a large effective view factor to a small condenser surface. This is obtained by forcing the outgassed material from a large sample surface to exit the distribution box only through a relatively small hole which has a large view factor to a small condenser. Problems associated with outgassed material from the sample condensing on the inside walls of the

distribution box are eliminated by maintaing the walls of the box at a temperature greater than or equal to the sample temperature. The distribution box will be constructed so that easy and economical replacement is possible, if necessary. Past experience has shown that if a material is a heavy outgasser of condensables, the interior surface of the distribution box should be discarded, rather than cleaned, to eliminate the possibility of contamination in future tests. All other sources of condensables will be eliminated by using only metals, unimpregnated fiberglass, glazed ceramics, and Teflon in the test systems.

The mirrors used will be fabricated from a ferrotype plate substrate (used in photographic processes for making glossy prints) with an evaporated aluminum mirror and an overcoating of magnesium fluoride. A metal plate is suggested as the mirror substrate to allow positive attachment of the mirror (with screws) to the temperature control platen, thereby eliminating uncertainties in the surface temperature of the mirror due to variations in contact resistances between different mirrors and the control platen. The light reflectance and transmittance measurements are being performed by General Electric under NASA contract; however, these experiments will also be performed at MSFC.

## L. LIGHT TRANSMITTANCE OF OUTGAS DEPOSITS

This test will be performed in a distribution box. The test configuration for the distribution box technique consists of a setup as shown in figure 22. The sample is placed inside a cylindrical box with two holes (one directly over each of the condensing surfaces). The two condensing surfaces used will be quartz plates whose transmission will be monitored in situ during the 102-hour condensability test with a 1P28 phototube and a 0.4 to 0.6 micron bandpass filter. Spectral transmission measurements of these plates will also be made between 0.35 and 0.75 micron before and after the condensability test. These tests shall show the changes in light transmission versus time, pressure, and temperature on plotted curves.

The light transmittance measurement apparatus will be made of the same type materials and associated problems as outlined in paragraph K, Light Reflectance Measurement of Outgas Deposits.

## M. EFFECT OF ENVIRONMENT ON MATERIALS

Typical examples of contamination and material property changes due to outgassing in heat and vacuum are shown in figure 23.

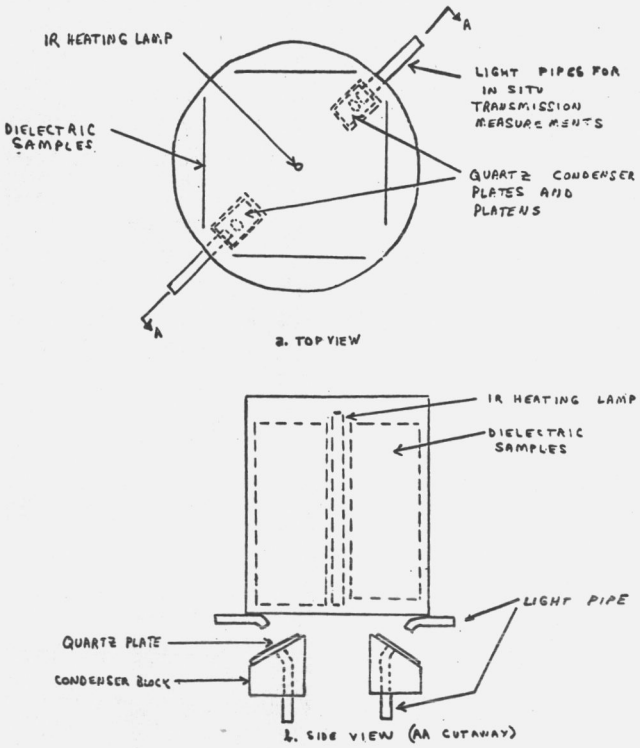



Figure 22. Light Transmittance Apparatus

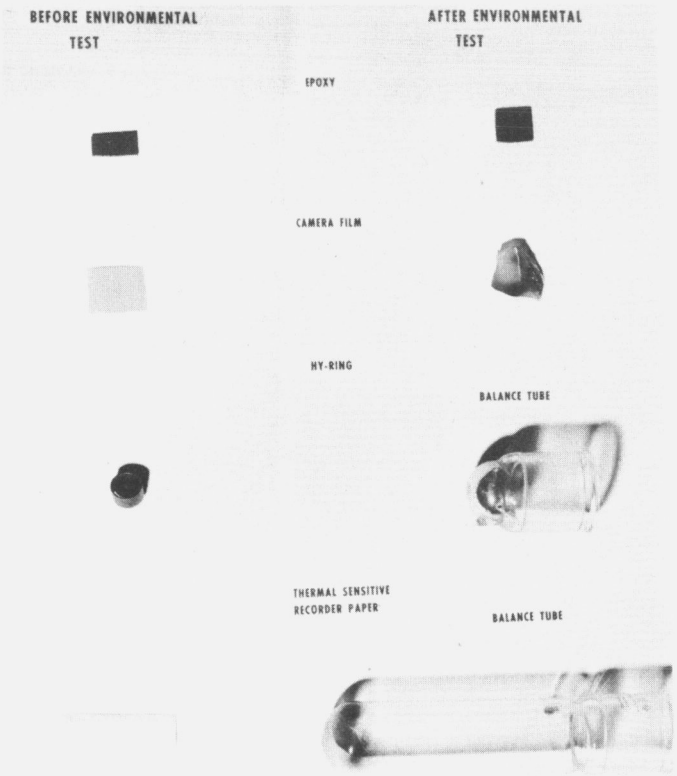



Figure 23. Effect of Environment on Materials

## SECTION VII. MISCELLANEOUS - OUTGASSING COLLECTOR

An outgassing filter mounted on an electronic box is shown in figure 24. A filter has been designed to collect and hold outgassing molecular fragments that are generated from the heat and vacuum of non-metallic materials used in space vehicles. The filter can be mounted on any black box that is sealed. The filter will vent the box to the space vacuum. Materials with extremely high outgassing rates were tested using the residual gas mass spectrometer. Relativity mass peak highs were recorded regularly up to 250 AMU. The materials were retested in simulated black box condition with the filter in place. Results from this test proved that the filter absorbed all of the outgassing fragments, and no peak was recorded.

OUT GASSING FILTER
MOUNTED ON
ELECTRONIC BOX

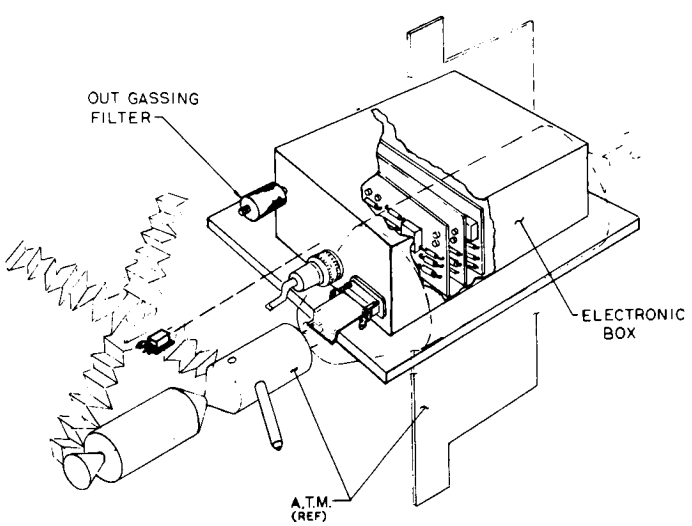



Figure 24. Outgassing Filter Mounted on Electronic Box

## SECTION VIII. CONCLUSIONS

The amount of outgassing of a material is a function of the total volume, whereas the rate of outgassing is a function of the surface area, with the rate increasing with increased surface area. Moisture, in most cases, is the first substance to outgas from a polymer under vacuum; hydrogen, oxygen, and traces of hydrocarbon compounds have been detected and, in some cases, have been structurally identified. Outgassing under vacuum may be a problem for the ATM because of gaseous materials that may contaminate optical lenses and distort light transmittance and reflection during photographic operations. Outgassing may also contaminate solar cells and other equipment.

Ultraviolet radiation can cause cross linking or chain scission to occur and may produce a different material with completely different properties; it may also cause material properties to change. When various gaseous monomers are cross linked because of exposure to ultraviolet radiation, new plastics may be produced.

The combining of fragments to produce new and unknown materials has not been discussed. There is no simple way to predetermine what the composition of outgas products from materials will be or how many materials will be produced. Damage to the ATM because of contamination cannot be predetermined; however, precaution is taken by eliminating as many of the materials that have high outgas rates as possible using the present test procedures.