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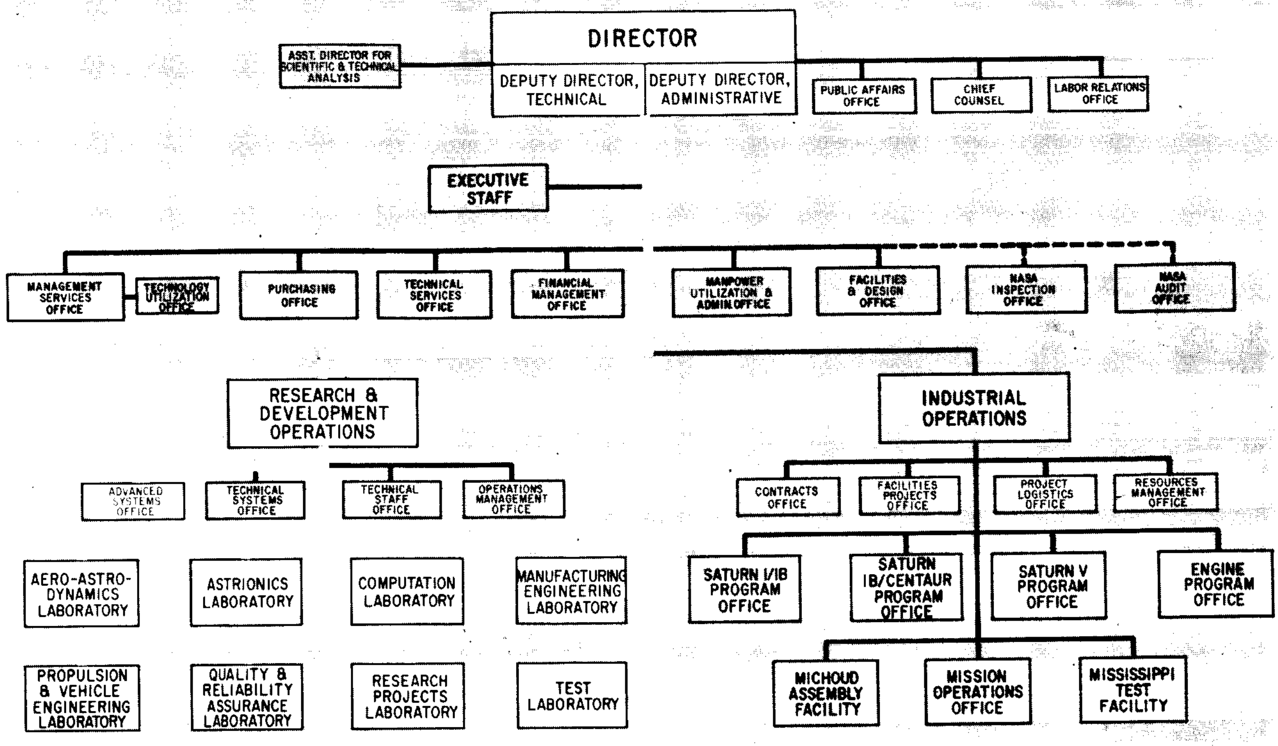
RESEARCH ACHIEVEMENTS REVIEW SERIES NO. 13

RESEARCH AND DEVELOPMENT OPERATIONS
GEORGE C. MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA

GEORGE C. MARSHALL SPACE FLIGHT CENTER



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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON, D. C.

INSTRUMENTATION RESEARCH AT MSFC

RESEARCH ACHIEVEMENTS REVIEW SERIES NO. 13

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PREFACE

In 1955, the team which has become the Marshall Space Flight Center (MSFC) began to organize a research program within its various laboratories and offices. The purpose of the program was twofold: first, to support existing development projects by research studies and second, to prepare future development projects by advancing the state of the art of rockets and space flight. Funding for this program came from the Army, Air Force, and Advanced Research Projects Agency. The effort during the first year was modest and involved relatively few tasks. The communication of results was therefore comparatively easy.

Today, ten years later, the dual purpose of MSFC's research program is still the same. Funding for the program now comes from NASA Program Offices. The present yearly effort represents major amounts of money and hundreds of tasks. Most of the money goes to industry and universities for research contracts. However, a substantial research effort is conducted in-house at the Marshall Center by all of the Laboratories. The communication of the results from this impressive research program has become a serious problem by virtue of its very voluminous technical and scientific content.

The Research Projects Laboratory, which is the group responsible for management of the consolidated research program for the Center, initiated a plan to give better visibility to the achievements of research at Marshall in a form that would be more readily usable by specialists, by systems engineers, and by NASA Program Offices for management purposes.

To initiate the plan, monthly Research Achievements Reviews have been established, repetitive over a yearly cycle, with each review covering one or two fields of research. These verbal reviews are documented in the Research Achievements Review Series.

Ernst Stuhlinger
Director, Research Projects Laboratory

This paper presented September 30, 1965

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INTRODUCTION AND SUMMARY TO INSTRUMENTATION RESEARCH AT MSFC

By

C. T. Paludan

Presented at Research Achievements Review
September 30, 1965

The development of launch vehicles requires considerable flight instrumentation during research and development flight phases. In many cases, the needs of the vehicle designers can be met by conventional, off-the-shelf instrumentation items previously used on earlier projects such as Jupiter or Air Force vehicles. However, a few requirements always exist that cannot be met with available hardware. Sometimes these requirements can be anticipated, and a research and development program can be initiated to meet the new requirement. Very often instrumentation must be defined far in advance of the delivery dates, not only because of the necessity for timely implementation into the documentation and procurement programs, but also because of restrictions and requirements from non-MSFC sources. An example of this is the UHF telemetry program, details of which will be given later.

As shown in the initial summary, MSFC is conducting instrumentation research in a number of areas. Table I summarizes some of the items being studied in other laboratories. These are generally for non-flight instrumentation. Many of the items shown have been presented at previous Research Achievements Reviews, or will be in the future. Table II summarizes some of the items being studied in Astrionics Laboratory strictly for flight instrumentation. Several specific tasks are discussed in detail. These are relatively typical of the type of effort that is required to fulfill the unusual requirements of instrumentation for launch vehicles and payloads. The goal of such tasks is to finally make available flyable hardware for implementation of existing or anticipated measuring requirements.

TABLE I
TYPICAL INSTRUMENTATION RESEARCH OTHER THAN BY ASTRIONICS

	I. Propulsion and Vehicle Engineering Laboratory
NAS 8-5491	Technique Development to Measure Vehicle Engine (Pulse Rocket) Performance
NAS 8-11311	Determination of Propellant Mass in a Large Storage Tank
NAS 8-5323	Theoretical Studies to Establish Design Parameters for Accurate Calorimeters
	II. Quality Assurance Laboratory
NAS 8-11705	Integration of Automatic Calibration System for Stage Instrumentation
NAS 8-11715	Development of Improved Sensing Methods and Devices for Stage Checkout
	III. Test Laboratory
In-House	Liquid Level and Quantity Instrumentation
NAS 8-11080	Point Density Sensor for Cryogenic Liquids

TABLE I (Cont'd)

NAS 8-11666	Development and Evaluation of High Capacity Load Cells
NAS 8-11072	High Thrust Measuring System
NAS 8-5186	Development of Damped Piezoelectric Accelerometers and Related Calibration Equipment
NAS 8-11534	Automatic Low Temperature Calibration System
NAS 8-11623	Prototype Cryogenic Temperature Measuring System
NAS 8-11629	Development of Low Range Absolute Pressure Calibration System
NAS 8-2673	Development of a Pressure Transducer Utilizing Friction Free Potentiometer
In-House	Cryogenic Environmental Effects on Transducers
NAS 8-5439	Development of Digital Reporting System
NAS 8-11076	Design and Development of an Improved Digital Measuring System
NAS 8-11088	Development of a Mass Computer
	IV. Aero-Astrodynamics Laboratory
NAS 8-11258	Local Optical Measurements of Turbulent Flow Properties on Ground Tests
-H- 71500	Research Study of Gas Density by Radiation Scattering
NAS 8-11220	Design and Development of a Breadboard Model of an Ultraviolet Air Density Gage
NAS 8-11046	Pressure Probe Characteristics in Transitional Knudsen Number Range
NAS 8-5350	Development of a Pressure and Force Transducer Calibration Procedure for the Hypersonic Shock Tunnel

TABLE I (Concluded)

	V. Manufacturing Engineering Laboratory
In-House	Development of Advanced Flight Strain Measuring Techniques
NAS 8-11115	Development of Instrumentation to Control Cleaning Procedures of Vehicle Components
In-House	Development of Continuously Monitoring X-Ray Examination of Weldments by Television Viewing
	VI. Research Projects Laboratory
-H- 71459	Thermal Radiation Measurement Techniques
-H- 71460	Thermal Testing Techniques
NAS 8-11202	Seismic Signals Resulting from Large Rocket Firings
In-House	Electric Field Meter Investigation
NAS 8-5336	Meteoroid Penetration Distributed Transducer

TABLE II
TYPICAL FLIGHT INSTRUMENTATION RESEARCH BY ASTRIONICS LABORATORY

- Cryogenic Densitometer Using Nucleonic Technique
- Advanced Pressure Transducers
- RF Approach for Measuring Liquid Level
- Explosion and Explosion Hazard Detection and Evaluation
- Techniques for Measuring Ambient Air Density from Space Vehicles at Orbital Altitudes
- Inflight Hydrogen Detection by Mass Spectrometer
- Special Thermocouple Gauges
- Fire Detection System for Cryogenic Fuel

TABLE II (Cont'd)

- Contoured Germanium Solid State Radiation Detector
- Advanced Heating Rate Transducers
- Thermally Isolated Sensor for Space Application
- Infrared Sensing System for Lunar Temperature Studies
- Cryogenic Temperature Sensor
- Special Calorimeter
- Quality Meter
- Strain Gage Accelerometer Using Piezoelectric Technique
- Vibration Spectrum Analyzer for Space Vehicles
- TV on Film Recorder for Flight Use
- Liquid/Vapor Sensor
- Transmitters
  - 220 MHz - 20W. T.W.T. (VHF)
  - 2200 MHz - 5W. S.S. (VHF)
  - 215 - 260 MHz 20W. S.S. (VHF)
- Airborne Tape Recorders
  - Analog
  - Digital

TABLE II (Concluded)

  - Combination Analog and Digital
- Time Division Multiplexers
  - Remote Programable Hi - Lo 810 Channel Multiplexer
- SS/FM Systems
  - Improved SS/FM Airborne Hardware
  - Improved SS/FM Demodulators (GSE)
- PCM Systems
  - Improved Analog to Digital Converter
  - Improved Remote Digital Submultiplexers
  - Airborne Computer Interface
- FM/FM Systems
  - Constant Bandwidth FM System
  - Improved IRIG FM/FM Hardware
- GSE for Automatic Telemetry Checkout
- Addressable Remote TM Multiplexers (Weight Reduction)
- Adaptive TM System (Bandwidth, Power and Weight Reduction)
- Advanced SS/FM Systems (Improved Accuracy, Phase Correlation and Response)
- Onboard Data Storage (For Use in Remote Orbital Operations)

INFLIGHT HYDROGEN DETECTION BY MASS SPECTROMETER

By

James C. Derington and Alexander Hafner, III

ABSTRACT

Methods of detection of hydrogen, hazardous mixtures, and explosions are discussed. The mass spectrometer developed for detection of hydrogen and explosive mixtures in flight is described in detail.

I. INTRODUCTION

The advent of liquid-hydrogen-powered space vehicles increased the possibility of fires or explosions which could result in loss of a mission and loss of the crew. A need was therefore evident for inflight instrumentation that could lessen this danger. This paper summarizes Astrionics Laboratory's efforts in this area of instrumentation research and development and then concentrates on one particular phase as being typical, the inflight detection of hydrogen by mass spectrometer.

II. APPROACH AND APPLICATION

We have approached the problem from three aspects: (1) hydrogen detection; (2) hazard, or explosive mixture, detection; and (3) explosion detection. In a fully instrumented vehicle, several detectors could be located in the interstage areas or anywhere hydrogen could accumulate. The equipment could possibly be made a part of the emergency detection system, thereby providing a warning to the crew and ground monitors that a dangerous concentration of hydrogen was present, that an explosive mixture of hydrogen and oxygen was present, or that an explosion had occurred - in particular, the minor explosion which usually precedes a major hydrogen explosion. Appropriate action, such as abortion of the flight or possibly suppression of the explosion, could then be initiated. Another obvious use for such instrumentation would be to provide information, through telemetry, that would be a valuable analysis tool in case of hydrogen leakage or in case of a catastrophic failure.

III. DETECTION METHODS

When Astrionics Laboratory began investigating the problem, there were no detection instruments suitable for flight use and only a few non-flight instruments, none of which were completely satisfactory. A research and development program was therefore begun and various techniques were investigated. Some of the more promising are described below.

A. HYDROGEN DETECTION METHODS

The hydrogen detection methods currently under development or proposed for development are:

1. Kryptonate

2. Acoustic

3. Polarographic

4. Fuel cell

5. Mass spectrometer

The kryptonate method of hydrogen detection is currently under development by Astrionics Laboratory. The principle involved is the release of radioactive krypton 85 gas from a kryptonated base metal when exposed to hydrogen gas. The amount of radiation is a measure of the amount of hydrogen present. This technique offers promise of providing a good inflight hydrogen detector.

The acoustic method of hydrogen detection is currently under development by Propulsion and Vehicle Engineering Laboratory in cooperation with Astrionics. This technique uses the principle that the velocity of sound in a gas is a function of the molecular weight of the gas. A device using this principle can be designed to be specific for hydrogen and therefore has potential as an inflight hydrogen detector.

The polarographic technique uses a palladium membrane through which only hydrogen will diffuse. The hydrogen comes into contact with an electrolyte between two electrodes, resulting in a change in the current produced, which is a measure of the amount of hydrogen present. This technique has several potential problems, such as poisoning of the membrane and freezing of the electrolyte, which must be solved before it can be used as an inflight detector.

The fuel cell method uses the normal fuel cell technique on a small scale to generate an electric current when both hydrogen and oxygen are present. If oxygen is contained in a small reservoir in the device, the device becomes a hydrogen detector, generating a current proportional to the amount of hydrogen present. This technique shows promise and will be developed in the near future, depending on the availability of funds.

B. HAZARD DETECTION METHODS

Hazard detection implies the detection of an explosive mixture of hydrogen and oxygen, rather than of hydrogen alone. Methods under investigation include the catalytic method, using a catalyst such as platinum to stimulate a reaction between any hydrogen and oxygen that are present. The resulting heat can be measured to provide an indication of the concentration of hydrogen and oxygen present and therefore of the explosive hazard. A mass spectrometer can be used for hazard detection by measuring the concentration of both hydrogen and oxygen.

C. EXPLOSION DETECTION METHODS

Explosion detection methods include a pressure rate-of-rise sensor, infrared rate-of-rise surveillance detectors, and ultraviolet detectors. Research and development work is underway on these detectors, and we hope to have flight hardware available in 1966.

IV. MASS SPECTROMETER

The mass spectrometer offered a solution to both hydrogen detection and hazard detection and appeared to be one of the most promising and most quickly attainable devices. Development was therefore begun on a spectrometer capable of performing this dual function as a flight instrument.

There are several types of mass spectrometers, including the magnetic deflection, time-of-flight, radio frequency, omegatron, and quadrupole mass spectrometers. Each has its advantages and disadvantages. Some have been used in satellite and other applications for qualitative and rough quantitative analysis, but none had all the desired features of high sensitivity, wide pressure range, good accuracy, and ruggedness for space vehicle environment. The quadrupole was chosen as the most promising for our application, and Consolidated Systems Corporation was selected to perform the research and development needed to produce a device to meet our requirements.

The quadrupole, like all mass spectrometers, has the general block diagram shown in Figure 1.

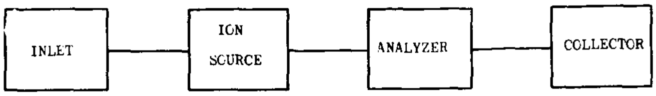


FIGURE 1. MASS SPECTROMETER GENERAL BLOCK DIAGRAM

The inner part of the spectrometer must operate at a pressure of 10^{-4} mm Hg or less, so the inlet can be a serious problem when the ambient pressure covers a wide range. In this application we will use a molecular leak to give flow rate in proportion to the partial pressures of the gases being analyzed. The molecular leak will take one of two forms which are being investigated. The first consists of a series of small holes, of the order of a few tenths of a micron, drilled in a thin gold foil by electron bombardment. The total flow rate will depend on how many such holes are formed. The second consists of a spherical valve seated in a circular knife-edge seat. The flow rate of this inlet aperture can be reduced by increasing the seating pressure. Tests indicate that this technique gives very good molecular flow without as great a clogging problem as has been experienced with the other construction.

The ion source is straightforward. A filament supplies ionizing electrons which are forced into a semicircular path by electrostatic focusing. The ions, formed by collision of the electrons with the gas molecules, are focused into the analyzer section by accelerating and focusing grids.

The analyzer section is the heart of the spectrometer. Here the ionized gases are separated according to mass. The different types of mass spectrometers differ mainly in the way the analyzer operates. The quadrupole analyzer, as shown on the simplified schematic view of Figure 2, consists of four rods, with opposing rods electrically connected and with a dc and an ac voltage applied to the rod pairs.

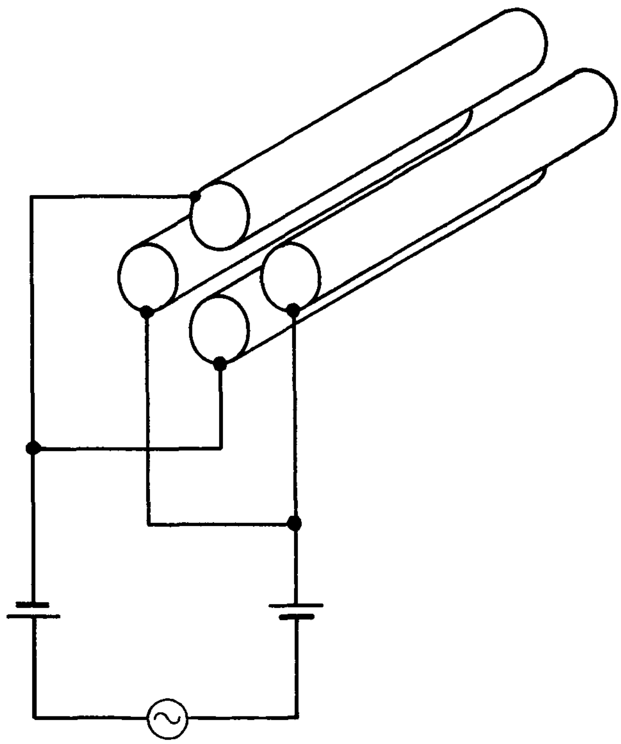


FIGURE 2. SIMPLIFIED SCHEMATIC OF QUADRUPOLE ANALYZER ROD ASSEMBLY

The ion source in front of the rod assembly projects a stream of ions from the sample gas longitudinally down the instrument between the rods. The combined dc and ac fields are such that only one mass will be resonant and will be focused toward the other end, the particular mass being selected by the voltage and frequency of the fields. All other masses collide with the rods and are collected by them. For this reason, the quadrupole is often referred to as a mass filter, analogous to an electrical bandpass filter, instead of a spectrometer, since it does not separate all ions present into a spectrum according to mass. In this case, two different excitation frequencies are selected and switched in so that both hydrogen and oxygen can be detected. The quadrupole can then serve as both a hydrogen detector and as an explosive mixture detector. A third oscillator frequency, tuned to resonate for iodine, is switched in periodically as a calibration reference. A small sealed quantity of iodine with a separate inlet system is supplied for this purpose.

The ions emerging from the other end of the analyzer strike the first dynode of an electron multiplier, which amplifies the resulting secondary emission electrons, thereby producing an output proportional to the amount of the particular gas present.

Figure 3, a complete block diagram of the quadrupole spectrometer, shows the four main parts plus all the auxiliary circuits such as the vacuum pumping system, reference gas supply, temperature controllers, control circuitry, etc.

The quadrupole spectrometer is comparatively simple and rugged mechanically and requires no magnet. Size, weight, and power are important, but satisfactory performance is the primary consideration. The spectrometer must be sensitive, since the lower explosive limit of hydrogen in air is about a four percent concentration. Response time must be short if it is to be useful as a warning device. Accuracy must be good enough that the measurement can be relied upon. Considering these factors, the specifications in Table I were derived.

TABLE I
SPECIFICATIONS FOR QUADRUPOLE MASS SPECTROMETER

1. Ambient pressure:	760 to 10^{-4} mm Hg
2. Temperature:	+75°C to -50°C
3. Vibration:	35 G random
4. Sensitivity:	7.6 x 10^{-3} mm Hg (partial pressure)
5. Range:	7.6 x 10^{-3} to 200 mm Hg (partial pressure)
6. Accuracy:	5% of reading
7. Output:	0 to 5 V in three ranges
8. Time Constant:	30 to 100 ms
9. Power:	40 watts
10. Operating Modes:	H_2 Only O_2 Only H_2 and O_2 alternately

The quadrupole rod assembly is shown in Figure 4; the ion source is at the left. Figure 5 shows the quadrupole analyzer mechanism packaged, with the inlet at the right, then the ion source, the analyzer section, the electron multiplier at the rear, and a small ion pump to maintain the internal vacuum during operation. The complete quadrupole mass spectrometer gas detector, with the case removed to show the electronic circuit cards, analyzer, reference source, pump, etc., is shown in Figure 6. Figure 7 shows the device in its case as it would appear for flight. The complete prototype package, including the electronic circuitry, is about 25 x 35 x 51 cm and about 23 kg in mass. Production versions could be considerably smaller, especially if only hydrogen detection was required. Delivery of the two prototype units is expected early in 1966. These could be flight tested on an early vehicle, although they are not presently scheduled to be flown. Production units could be available sometime next year.

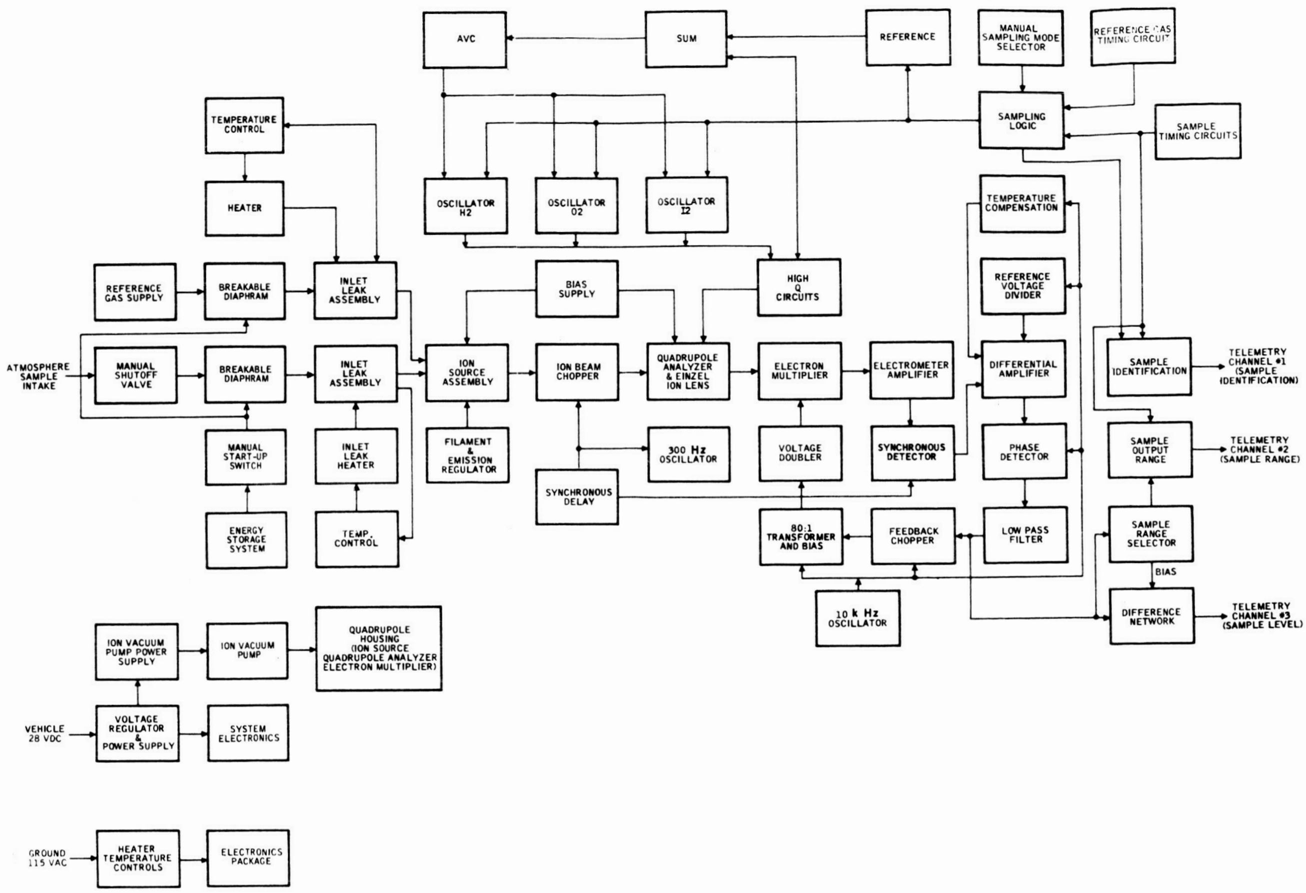


FIGURE 3. QUADRUPOLE MASS SPECTROMETER BLOCK DIAGRAM

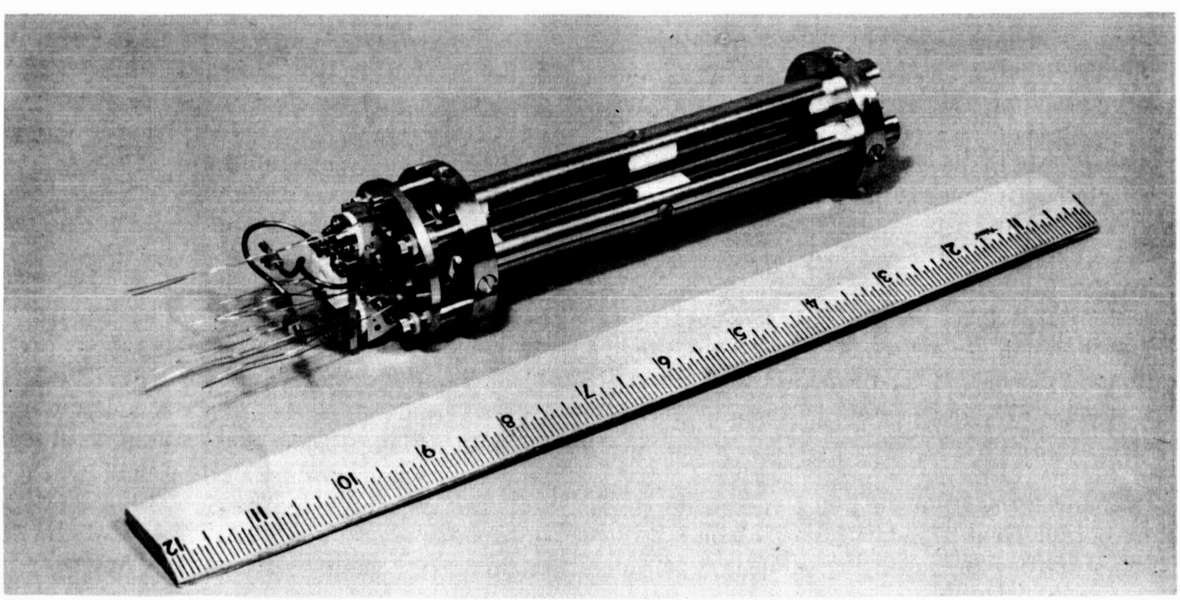


FIGURE 4. QUADRUPOLE ROD ASSEMBLY

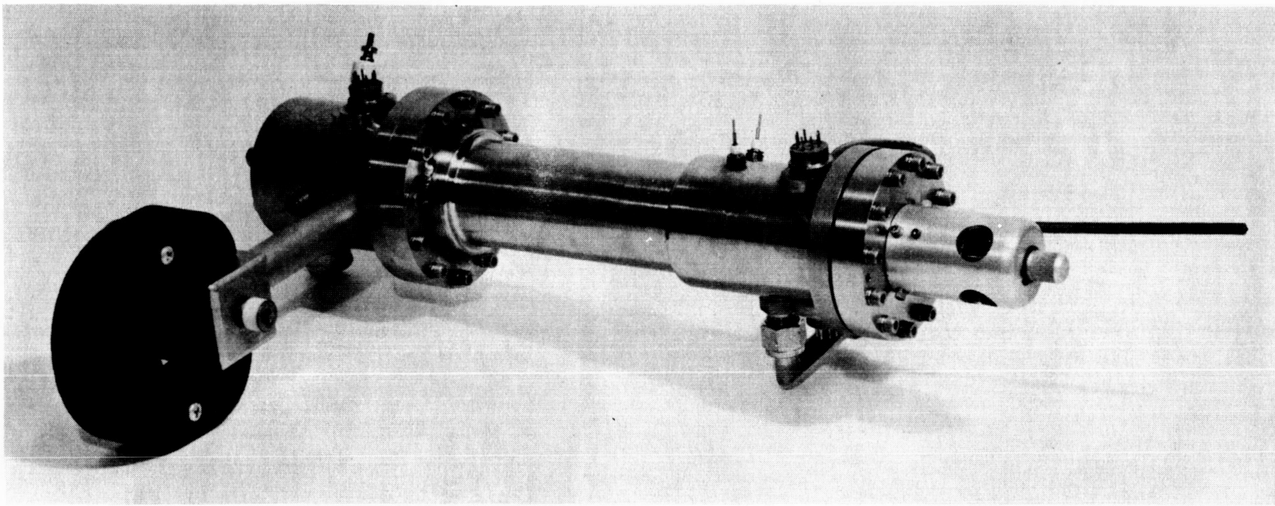


FIGURE 5. ASSEMBLED QUADRUPOLE ANALYZER MECHANISM

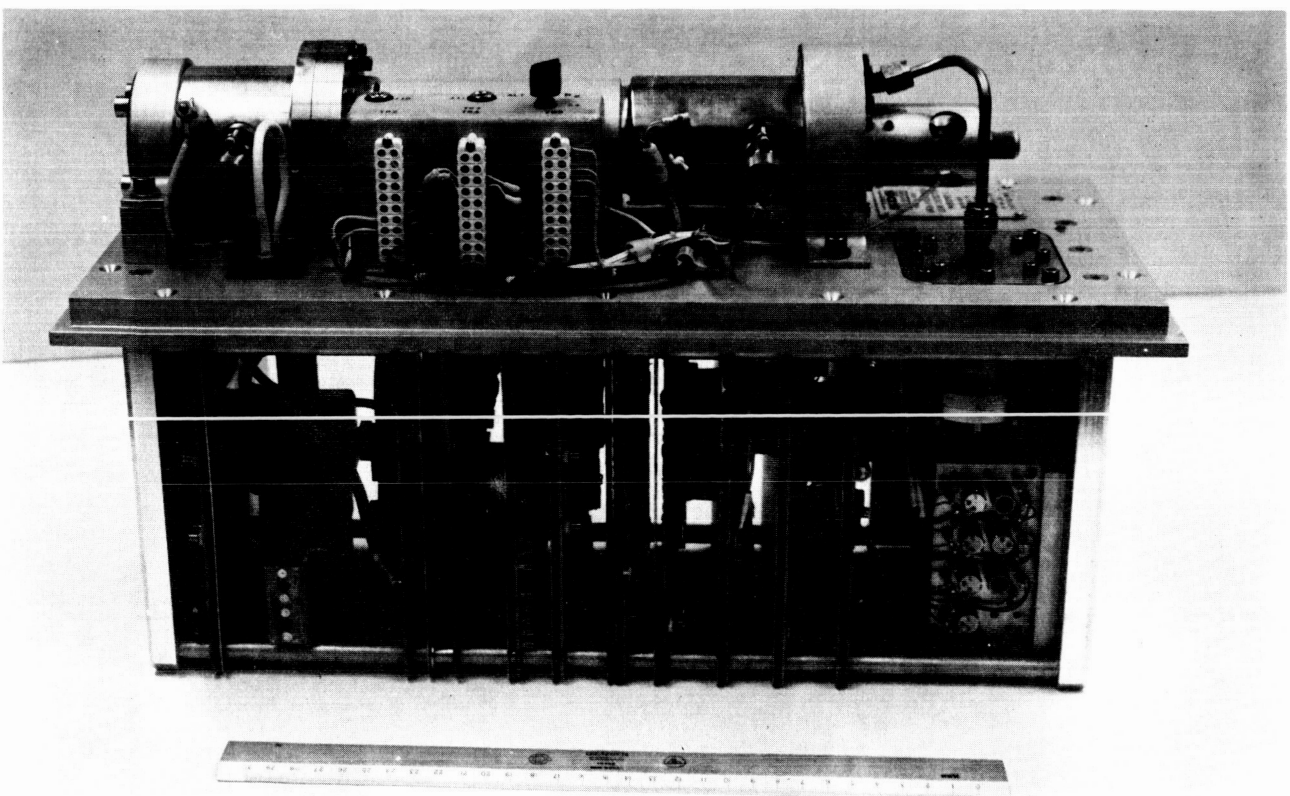


FIGURE 6. COMPLETE MASS SPECTROMETER, CASE REMOVED

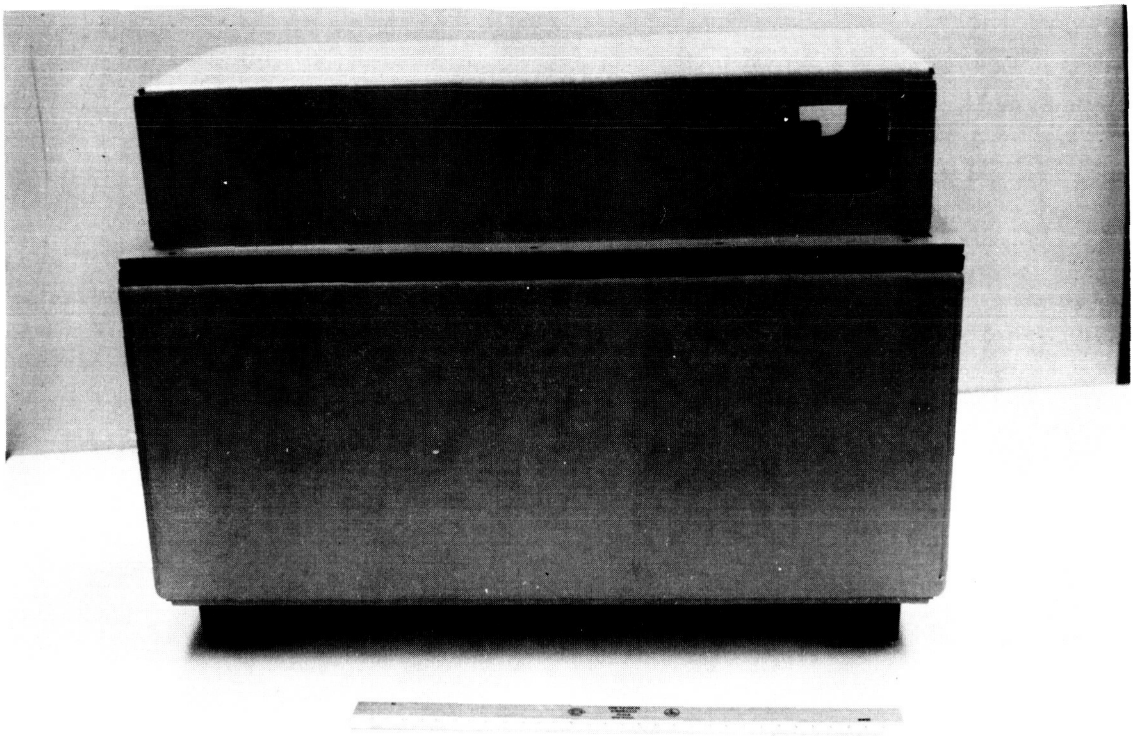


FIGURE 7. COMPLETE MASS SPECTROMETER WITH CASE

V. CONCLUSION

Considerable effort is being made to develop instrumentation suitable for inflight detection of hydrogen, explosive hazards, and explosions. The mass spectrometer is one example. The ultimate goal is to reduce the danger to the vehicles and to the crew.

THERMALLY ISOLATED TEMPERATURE SENSOR FOR SPACE APPLICATION

By

Harlan Burke

ABSTRACT

The requirements for inflight thermal sensors are discussed, as are the design guidelines for such instruments. The four-element thermal sensor experiment used on the Pegasus satellite is described and some of the flight measurements are presented.

GLOSSARY

C_1 = Conductance

R_1 = Radiance

C_T = Total thermal conductance

$C_2 = K_1A_1/l_2$

$C_3 = K_1A_1/l_3 + K_1A_1/\ l_1^2 + l_3^2$

$C_4 = K_1A_1/l_4$

K_1 = Conductivity of Ti (6AL-4V)

A_1 = Cross section of Ti support rods

$l_{1,2,3,4}$ = Lengths of Ti support rods

K_2 = Conductivity of constantan

A_2 = Cross section of electrical leads

l_5 = Length of electrical leads

A_3 = Surface area of sensing disc

E = Emissivity of gold coated underside

σ = Stefan-Boltzmann constant

P = Density of Al 6064

V = Volume of sensing disc

S = Specific heat of Al

I. INTRODUCTION

Inflight measurements originate when the need for confirmation of a vehicle system design or a control parameter is recognized. Many factors influence the selection of the measuring techniques and systems to be used in obtaining the desired data. Frequently a measuring problem arises that cannot be accomplished with either existing or modified components, and it becomes necessary to develop new methods and instruments. The thermally isolated sensor was selected to illustrate the development of an instrument to satisfy a unique measuring requirement.

The design of temperature control systems for space vehicles depends primarily on the application of the thermal radiation characteristics of the vehicle surfaces. It is essential that the thermal radiation properties of these surfaces remain stable when exposed to the space environment. The requirement that each space vehicle be successful in its mission has limited the use of new coatings whose properties are untested in space. As part of an investigation to study the long term stability of a number of new temperature-control coatings, an experiment was devised by Research Projects and Astrionics Laboratories to measure the radiation properties of surfaces in the space environment. This experiment has been included in the Explorer XI, Saturn SA-4, and the more recent Pegasus satellite flights.

Although the primary purpose of the investigation is to study the emissivity stability of surfaces, this discussion will describe the technique for measuring these characteristics during flight in space. Since each vehicle requires a different configuration, the Pegasus experiment package will be used as a typical example of the application of these techniques.

The method consists of measuring the temperature history of a number of thermally isolated test surfaces. Thermal isolation of the test surface from the satellite structure minimizes the extraneous heat losses of conduction and radiation from the sensor. A transient thermal analysis is used to determine the radiation characteristics of the surfaces from the temperature response curves.

Four different surfaces have been tested so far in this experiment. A fifth surface, which was designed to remain unchanged in space, was used as a reference. Comparisons of the temperatures of the test surfaces with that of the reference surface provide a basis for evaluating changes in the thermal characteristics of the test coatings. Additional coatings are scheduled for testing utilizing this technique on future space vehicles assigned to MSFC.

II. DESIGN GUIDELINES

The following design guidelines were set forth at the beginning of the program to assure overall accuracy and reliability in the technique for determining the radiation characteristics of the test surfaces:

1. Thermally isolated test surfaces to minimize corrections for heat exchanges caused by conduction and radiation.

2. Thermal mass of the test sensor to be such that the sensors would respond rapidly to temperature changes, yet have enough thermal lag to permit measurement of the rate of temperature decay as the sensor moves from the sunlight into the shadow.

3. Structural rigidity to withstand the space environment.

4. At least one stable reference surface so that any changes in the test surfaces caused by the space environment would be detected.

5. Telemetry and temperature measuring systems of sufficient sensitivity and accuracy.

III. RADIATION SENSOR DESIGN

In accordance with these requirements, the sensors were designed to minimize extraneous heat losses from the test surfaces. Careful attention was given to the selection of materials for the sensor assembly and to the preparation of the mounting configuration. Figure 1 is the completed sensor assembly in a cutaway view of the mounting cylinder.

This assembly differs from earlier models in that the aluminum sensor disc was supported by a single Kel-F support rod. When flight data indicated that the sensor temperature was approaching the point at which the Kel-F began to deteriorate, it became necessary to replace this material with one of higher temperature resistance.

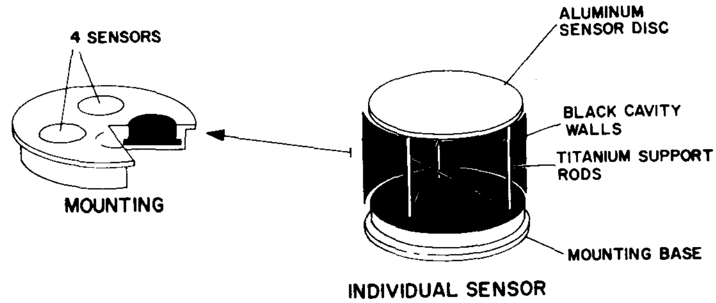


FIGURE 1. INSTALLATION DRAWING OF INDIVIDUAL SENSOR

In the present design, the sensor disc is supported by three 6AL4V titanium rods. The diameter of these rods was selected so that the conductive heat losses from the sensor disc through the rods did not exceed the heat losses through the Kel-F support rods of the previous sensors. The small size of these rods increased the flexibility of the unit, and cross braces were necessary to increase the mechanical strength to withstand the vibration of the vehicle.

The front surface of the sensor disc was coated with the material to be tested. An emissivity of $\cong 0.05$ was obtained for the back surface of the disc by vapor depositing gold on the surface after installation of the temperature sensor. Internal surfaces of the mounting base and the walls of the cylinder were black anodized to provide an emissivity of $\cong 0.70$. Thus, the low emittance surface of the sensor disc was installed in a cavity of comparatively high emittance characteristics.

The thermodynamic design properties of the sensor were determined by Research Projects Laboratory and are shown in Table I.

After finalization of the design, a contract was given for fabrication of the flight units. The contractor was unable to deliver acceptable units because of titanium-to-aluminum welding problems, poor emissivity of the gold coating on the back surface of the disc, and the close tolerances on the assembly necessary for mounting.

Astrionics Laboratory developed an electron beam welding technique to fuse the aluminum and titanium and a method of vapor depositing the gold over the back surface of the disc to provide the desired emissivity. All of the flight units were fabricated and calibrated by this laboratory.

TABLE I
THERMODYNAMIC DESIGN PROPERTIES OF THE MMC THERMALLY ISOLATED SENSOR EXPERIMENT (R-RP-T-WP-6-64)

PROPERTY	EQUATION	VALUE
DIAGRAM	DISC — C_1 / R_1 — CASE	
CONDUCTANCE THROUGH THE SUPPORTS	$\frac{1}{C_T} = \frac{1}{C_2} + \frac{1}{C_3} + \frac{1}{C_4}$	$C_T = 2.5 \times 10^{-4}$ WATTS / °K
CONDUCTANCE OF THE ELECTRICAL LEADS	$C_5 = KA/\ell$	$C_5 = 0.05 \times 10^{-4}$ WATTS / °K
CONDUCTANCE FROM THE DISC TO THE CASE	$C_1 = C_T + C_5$	$C_1 = 2.6 \times 10^{-4}$ WATTS / °K
RADIANCE OF THE DISC TO THE CASE	$R_1 = AE\sigma$	$R_1 = 2.2 \times 10^{-4}$ WATTS / °K
HEAT CAPACITY OF THE UNCOATED DISC	$H_1 = PVS$	$H_1 = 1.03$ JOULES /°K

The radiation sensors were arranged in a cluster of four, as shown in Figure 2, for mounting on the Pegasus satellite. The overall weight of the sensor assembly was 563 grams.

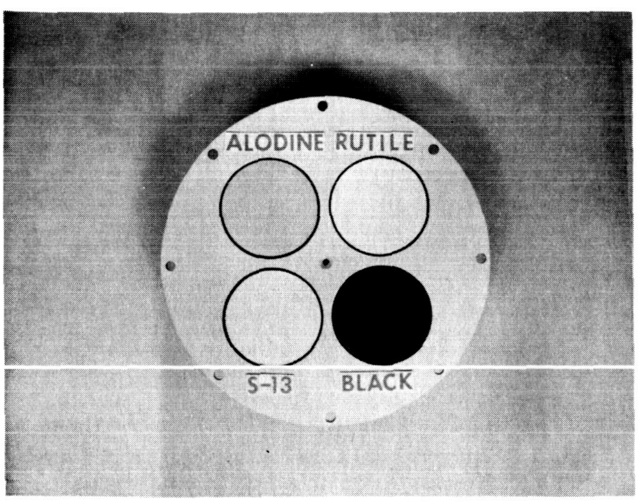


FIGURE 2. PEGASUS EXPERIMENT PACKAGE

Each sensor was coated with a different material by the Propulsion and Vehicle Engineering Laboratory. Table II is a list of the materials used in this experiment.

A 200 ohm nickel-element resistance thermometer encased in a thin sheet of bakelite was selected

TABLE II
COATINGS FOR MMC THERMALLY ISOLATED SENSOR EXPERIMENT

COATING	EMITTANCE	VEHICLE	PIGMENT	COMMENTS
BLACK PAINT	0.91	DOW CORNING SILICONE 808	CARBON BLACK ROYAL SPECTRUM MOCKS	DETERIORATES WHEN SUBJECTED TO A TEMPERATURE OF 370° C FOR TWO OR MORE HOURS. USED ON MMC STRUCTURE NEAR ELECTRONICS CANISTER. CONSIDERED AS A REFERENCE.
S-13 WHITE PAINT	0.89	POTASSIUM SILICATE	ZINC OXIDE	WILL BE THE EXTERNAL COATING ON THE S-IV, IU, AND APOLLO ADAPTER.
ALODINE	0.60 (VARIES FROM ONE PROCESSED GROUP TO ANOTHER)	CHEMICAL CONVERSION OF ALUMINUM-FORMS: ALUMINUM PHOSPHATE, CHROMIUM PHOSPHATE, WATER, FLOURIDES		USED ON THE MMC STRUCTURE AND DETECTOR PANELS.
RUTILE	0.78	SILICONE	TITANIUM DIOXIDE	CHOSEN FOR COMPARISON WITH TEST DATA FROM OTHER AGENCIES

as the temperature sensor because of its size, resistance to radiation, repeatability, resistance-versus-temperature curve over the desired range, and time response. Four of these resistance thermometers were used on the sensor discs. A fifth unit of higher resistance was mounted on the base plate to measure the temperature of the cavity walls. Since the base plate and the cavity housing were in good thermal contact, they were considered to be at the same temperature. This additional measurement permitted correction for heat exchanges between the test surfaces and the sensor mounting assembly. Figure 3 is a block diagram of the temperature measuring system for this vehicle.

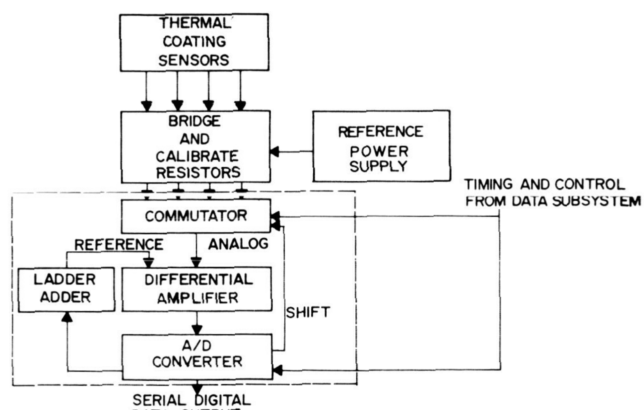


FIGURE 3. MMC TEMPERATURE SUBSYSTEM

The radiative constants were determined by total normal emittance measurements and were used to calculate a theoretical response curve for each sensor. The sensors were then placed in a test fixture in a vacuum and alternately heated and cooled, as shown in Figure 4.

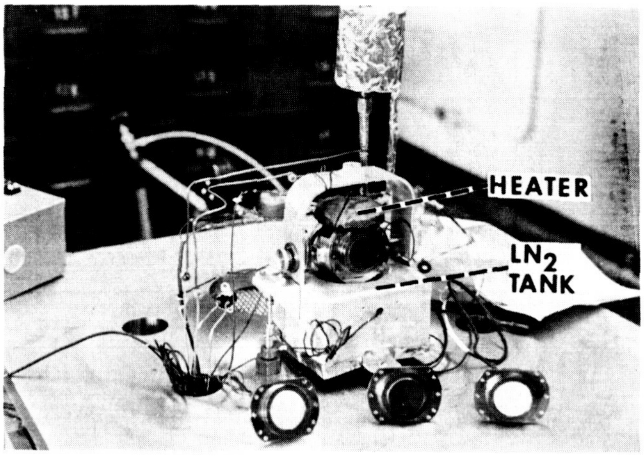


FIGURE 4. CALIBRATION FACILITY

The disc and case temperatures were monitored and compared with the calculated response curves. Figure 5 is typical of the data obtained from these tests.

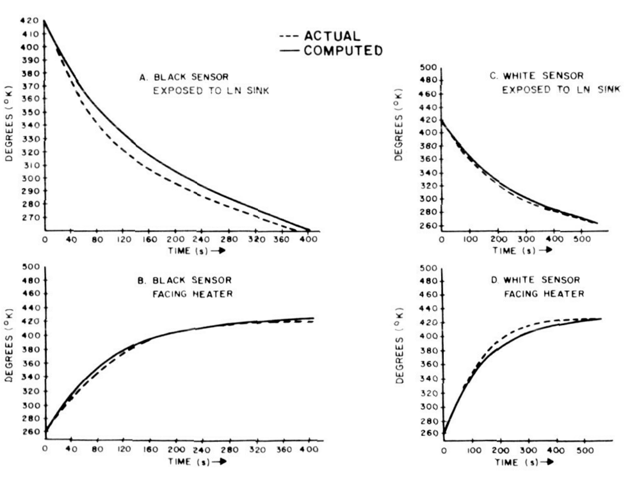
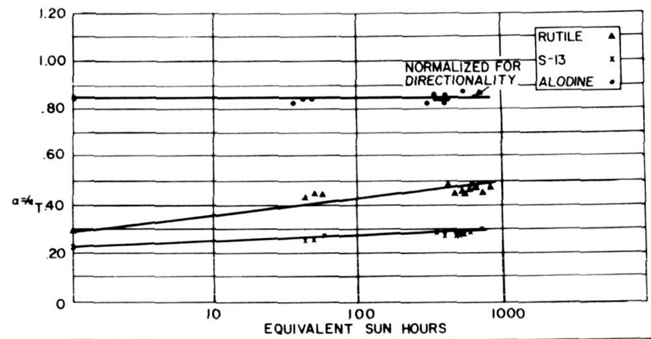


FIGURE 5. ISOLATED SENSOR: THEORETICAL AND MEASURED TEMPERATURE RESPONSE CURVES

IV. RESULTS OF FLIGHT MEASUREMENTS

The results of the flight measurements cannot be presented without considering the heating sources for the test surfaces in the space environment. In space, the surfaces receive heat primarily from direct solar radiation, reflected solar radiation from the earth, and direct radiation from the earth. In addition, a small amount of energy is transferred from the underside of the disc to the mounting cylinder. Energy from direct solar radiation is constant during the time that the sensor is in the sun. Heating from reflected sunlight varies with the sensor position and aspect. A computer program, written by Research Projects Laboratory, incorporates all the factors necessary in the data analysis of this experiment.

Reduction of the Pegasus A thermally isolated sensor data is continuing. The α_S/ϵ_T ratio of solar absorptance to infrared emittance as a function of sun time is shown in Figure 6. The data were taken from orbits in which the sensor was normal to the sun at points where the earth was not visible to the sensors. This condition permits the ratio of the solar absorptance to infrared emittance to be obtained from the steady state disc temperature, which is primarily a function of solar input. Details of data reduction can be obtained from R-RP-T of Research Projects Laboratory.



$$\alpha_S/\epsilon_T = \frac{\sigma T^4}{S\cos\theta} - \frac{Q_l}{AS\epsilon\cos\theta}$$

T = DISC TEMPERATURE
σ = STEFAN-BOLTZMANN CONSTANT
S = SOLAR CONSTANT
θ = SUN ASPECT ANGLE
A = AREA OF DISC
Q_l = SMALL EXTRANEOUS HEAT LOSS TO SENSOR CASE

FIGURE 6. ABSORPTANCE-INFRARED EMITTANCE OF PEGASUS "A" REFERENCE SENSORS VS EQUIVALENT SUN TIME

V. CONCLUSION

The thermally isolated sensor is a useful tool for the materials research engineer in determining the properties of coatings and surfaces in the space environment. It also provides information in other areas of interest. For example, the Apollo Office observed an apparent change in the α/ϵ characteristics of the coating used on the service module adapter on the SA-10 flight. This change was not observed in the protected thermal sensor package and was considered to be the result of some external condition. MSFC has been asked to place instrumentation upstream of the retrorocket exhaust to determine if this is the source of the problem. Thermally isolated sensors will be used in the testing program for the solution of this problem.

UHF TELEMETRY DEVELOPMENT AT MSFC

By

Donald G. Davis

ABSTRACT

The background of recent decisions relating to frequency bands used for telemetry and changes planned to be effective by 1970 are discussed. A telemetry program and details of UHF transmitter developments are presented.

I. INTRODUCTION

The radio frequency spectrum is considered a natural resource and, as such, its use is regulated by the Federal Government. The worldwide use of frequencies is covered by international treaties.

By means of the Communications Act of 1934, Congress designated that regulation of frequency utilization by government agencies be placed under the President and regulation of non-government radio and wire communications be placed under the Federal Communications Commission (Fig. 1).

The President has delegated the responsibility for regulation of government usage of radio frequencies to the Director of Telecommunications Management (DTM), who is an assistant director of the Office of Emergency Planning. The DTM is assisted in this by the Interdepartmental Radio Advisory Council (IRAC).

For many years the frequency band between 225 and 260 MHz has been designated by IRAC for interim use by the telemetering services. A target date of January 1, 1970, was established for completing the move of telemetering to two other frequency bands: 1435 to 1535 MHz and 2200 to 2300 MHz. Based on an agreement between NASA and DOD in 1963, MSFC has based its telemetry planning on almost exclusive use of the older 225 to 260 MHz band. However, in February 1965, DOD issued a firm directive to the three military services requiring that they completely vacate the 225 to 260 MHz band by January 1970. In March 1965, DOD requested that NASA also vacate this band so that it might be used for tactical and operational military communications. This matter is now under study by NASA and a decision is expected to be made within a few months (Table I).

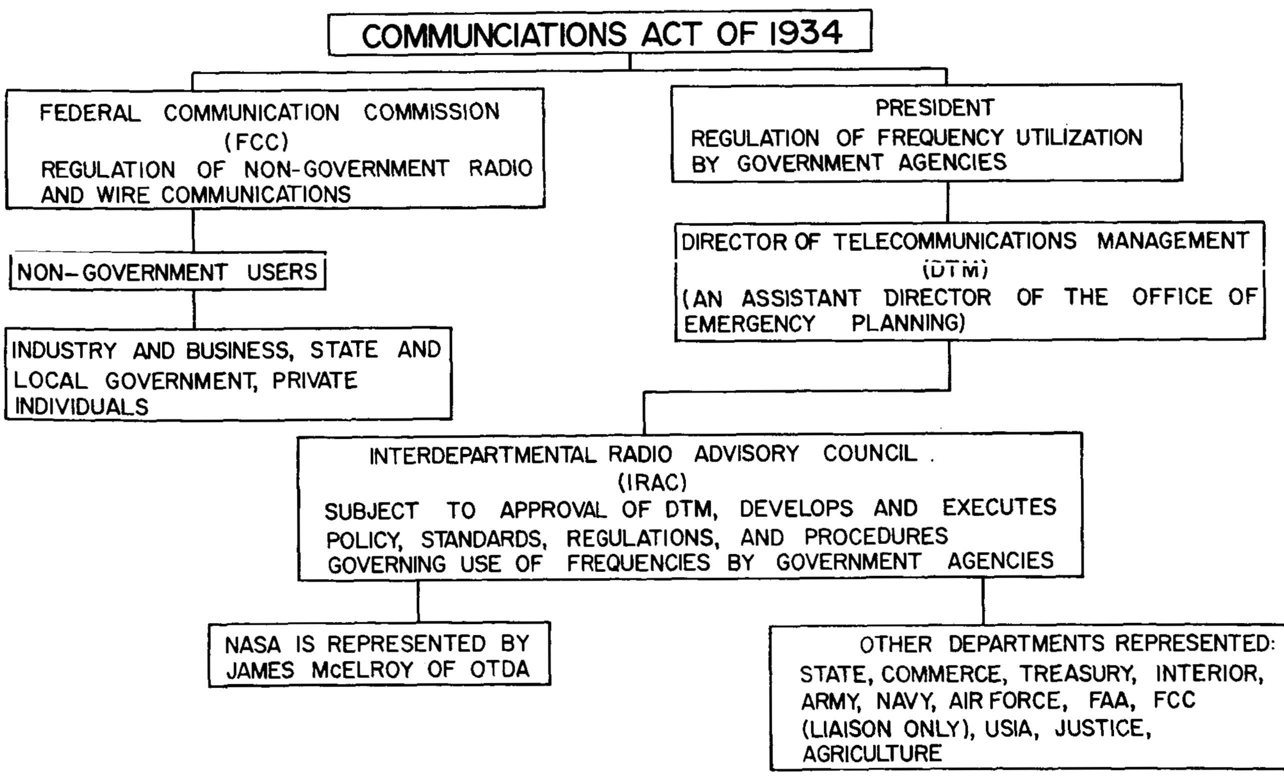


FIGURE 1. COMMUNICATIONS ACT OF 1934

TABLE I
TELEMETRY FREQUENCY ALLOCATION ACTIVITIES

1953-1956	- The 216 to 225 MHz frequency band was widely used for telemetering.
1956	- The military began use of the 216 to 225 MHz band for air defense radar. Telemetry users objected.
1958	- IRAC made available 44 channels in the 225 to 260 MHz band for interim telemetry use. A target date of January 1, 1970, was established for telemetering to vacate this band.
1958-1962	- Some telemetering development in the 2200 to 2300 MHz band by DOD agencies and NASA. Very little non-government sponsored industry effort.
1962	- DOD began to circulate reports encouraging industry to design 1400 MHz and 2200 MHz telemetry equipment.
1963	- A joint DOD/NASA memorandum to IRAC noted that "there is a need for continuation of telemetering operation in the 225 to 260 MHz band beyond 1970."
Feb. 1965	- MCEB issued a firm directive to the three military departments and notified IRAC that DOD telemetry operations would completely be removed from the 225 to 260 MHz band by 1970.
Mar. 1965	- Dr. Harold Brown, (then) Director of Defense R& D, wrote a letter to Dr. Seamens requesting that NASA vacate the 225 to 260 MHz band by 1970.
Apr. 1965	- NASA requested that IRAC delay its recommendation to the DTM pending completion of NASA study on future needs for telemetering in the 225 to 260 MHz.
June 1965	- VHF telemetry study group established with representatives from affected centers and NASA Headquarters.
Sept. 1, 1965	- Target date for completion of study.

In general, the 225 to 260 MHz band (generally referred to as the VHF telemetry band) is more desirable for telemetering space vehicles during launch and orbital operations than the higher frequency (or UHF) bands. This is due primarily to the relative difficulty of ground antenna acquisition of signals, the relative state of the art in development of equipment in the two bands, and the greater complexity of equipment for operation at the higher frequencies. After a space vehicle leaves earth orbit (for example, the translunar trajectory), the use of UHF frequencies becomes more advantageous because directional vehicle antennas are more easily implemented at the higher frequencies.

One of the major problem areas associated with telemetry operations at the UHF frequencies is the availability of RF transmitters which meet the necessary specifications. Since 1963 MSFC has carried on an active program in the design and development of UHF transmitters. Much of this work has been accomplished under the five contracts shown in Table II.

II. FIVE-WATT UHF TRANSMITTER DEVELOPMENT

In June 1963, a contract was awarded to Hallicrafters, Pacific Division, for a five-watt solid-state S-band transmitter. A block diagram of the proposed transmitter is shown in Figure 2.

TABLE II
MSFC UHF TRANSMITTER CONTRACTS

August 1, 1965

Contract Number	Date Awarded	Recipient	Objective	Status
NAS8-5497	June 1963	Hallicrafters, Inc. (Pacific Division)	Development of a 5-watt solid-state S-band transmitter	Contract cancelled in October 1964 for convenience of the government with no payment
NAS8-5494	June 1963	Radiation at Stanford (Later this group became Energy Systems, Inc.)	Development of a 20-watt hybrid S-band transmitter	Evaluation of first unit began in September 1965
NAS8-11771	June 1964	Energy Systems, Inc.	Development of a 5-watt solid-state transmitter	Transmitter components being assembled at contractor facilities
NAS8-11822	Jan. 1965	Electro-Mechanical Research, Inc.	Off-the-shelf purchase of 20-watt L-band transmitter	Delivery expected January 1966
NAS8-20505	June 1965	Energy Systems, Inc.	Development of 20-watt solid-state S-band transmitter	Various design approaches are being evaluated

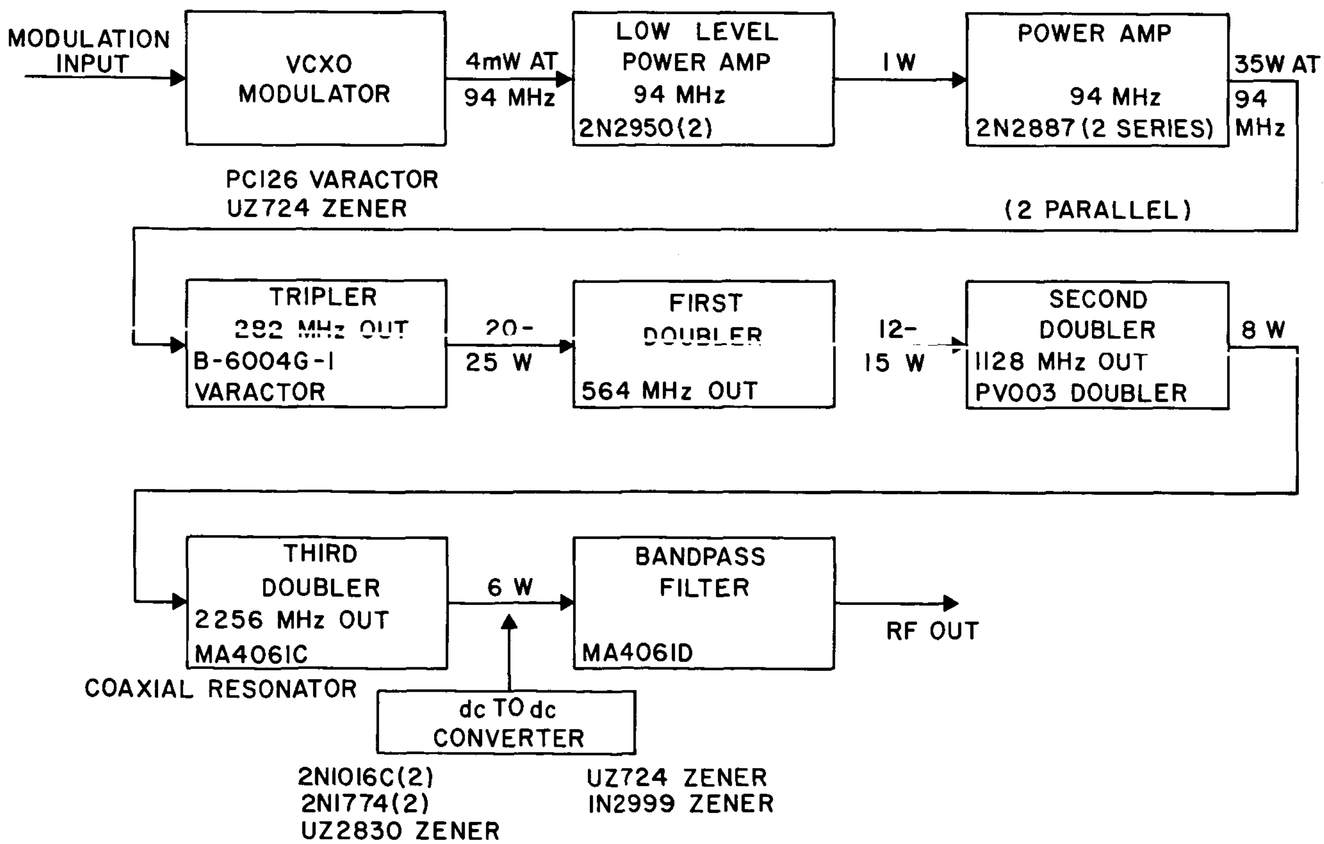


FIGURE 2. BLOCK DIAGRAM OF INITIAL 5-W TRANSMITTER DEVELOPMENT EFFORT

A voltage controlled crystal oscillator (VCXO) modulator was employed with output at 94 MHz. The 94 MHz signal was amplified to over 35 W in a power amplifier and then coupled to a multiplier chain of one varactor tripler and three varactor doublers. Power output at 2200 to 2300 MHz was to be 5 watts.

The required frequency was obtained by using an ideally cut quartz crystal with a temperature stability of about ±0.002 percent from -20°C to +90°C. By using negative-temperature-coefficient capacitors in series with the crystal, it was found possible to stabilize the drift to within ±0.001 percent of the frequency at 25°C while varying the temperature from -20°C to +85°C.

The low-level power amplifier stage produced 1 W output for 1 mW input for approximately a 30 db gain. Since the modulator output was about 4 mW, the low-level power amplifier supplied more than 1 W to the power amplifiers. The main power amplifier consisted of three parallel stages, each using four RF power transistors. With 1 W input, the measured output power was 35 W at 94 MHz.

The output of the power amplifier stage is applied to a tripler which uses a single ended varactor. The efficiency of the tripler was measured at about 60 percent, producing an output between 20 and 25 W at 282 MHz. Both the first and second doublers had an efficiency of about 60 percent, which resulted in an output of 9 to 10 W at 1128 MHz. The third doubler produces an output of approximately 6 watts at 2256 MHz.

In January 1964, a unit was demonstrated at MSFC. Tests run at that time demonstrated that the unit did not meet the specified requirements. In April 1964, Hallicrafters was visited by an MSFC technical representative who found that no progress had been made on correcting the problems that existed. In May 1964, the work was transferred to Hallicrafters in Chicago for completion.

In October 1964, nine months after scheduled delivery, the contract was cancelled for the convenience of the Government with no payment to the contractor.

III. TWENTY-WATT HYBRID TRANSMITTER

In June 1963, MSFC awarded contract NAS8-5494 to Energy Systems, Inc., (at that time it was called Radiation at Stanford) for the development of a 20-W S-band transmitter. A block diagram of the proposed transmitter is shown in Figure 3. Basically, it consists of circuitry generating the S-band frequency at a low power level to drive a traveling wave tube (TWT), which produces the required output power. A phase-locked loop, using a crystal oscillator as reference, is used to minimize the effects of incidental frequency modulation.

The circuitry is entirely solid state except for the final TWT. Excitation for the TWT is generated by a 190-MHz voltage-controlled oscillator followed by three stages of amplification, which provides an output of about 300 mW. This signal is applied to a varactor quadrupler and then to a varactor tripler. A bandpass filter is then used to eliminate undesired spurious outputs from the multipliers, resulting in about 25 mW at the input to the TWT.

A crystal oscillator operating at approximately 81 MHz is the basic frequency stabilizing element for the transmitter. This frequency is multiplied by a transistor tripler followed by a varactor times-9 multiplier and is then mixed in a hot carrier diode mixer with a sample taken from the TWT output via a directional coupler. This mixer provides a difference frequency of 89 MHz, which is amplified and fed to one side of an 89-MHz phase detector. An 89-MHz voltage-controlled oscillator, whose exact frequency is determined by the modulation input, is amplified and fed to the other side of the phase detector.

If these two frequencies are identical with a 90-degree phase difference between them, the output of the phase detector is zero. The output of this detector is fed through a loop stabilizing filter, and thence to the 190-MHz voltage controlled oscillator (VCO). If for any reason the 190 MHz VCO should drift toward a higher frequency, the phasor would start to advance relative to the phasor being generated by the VCO. This generates an error signal, which decreases the 190-MHz VCO frequency sufficiently that it stays locked to the 89-MHz VCO. However, if a modulation signal changes the frequency of the 89-MHz VCO in the upward direction, the phasor presented to the phase detector moves in such a direction as to cause the 190-MHz VCO to follow. From the diagram, observe that the 190-MHz VCO changes frequency only one-twelfth as much as the 89-MHz VCO because of the intervening multipliers and that frequency drift in the 89-MHz VCO is simply added to the output frequency of the transmitter rather than multiplied by 12.

One reason for choosing a phase-locked loop approach was as follows. There was some concern,

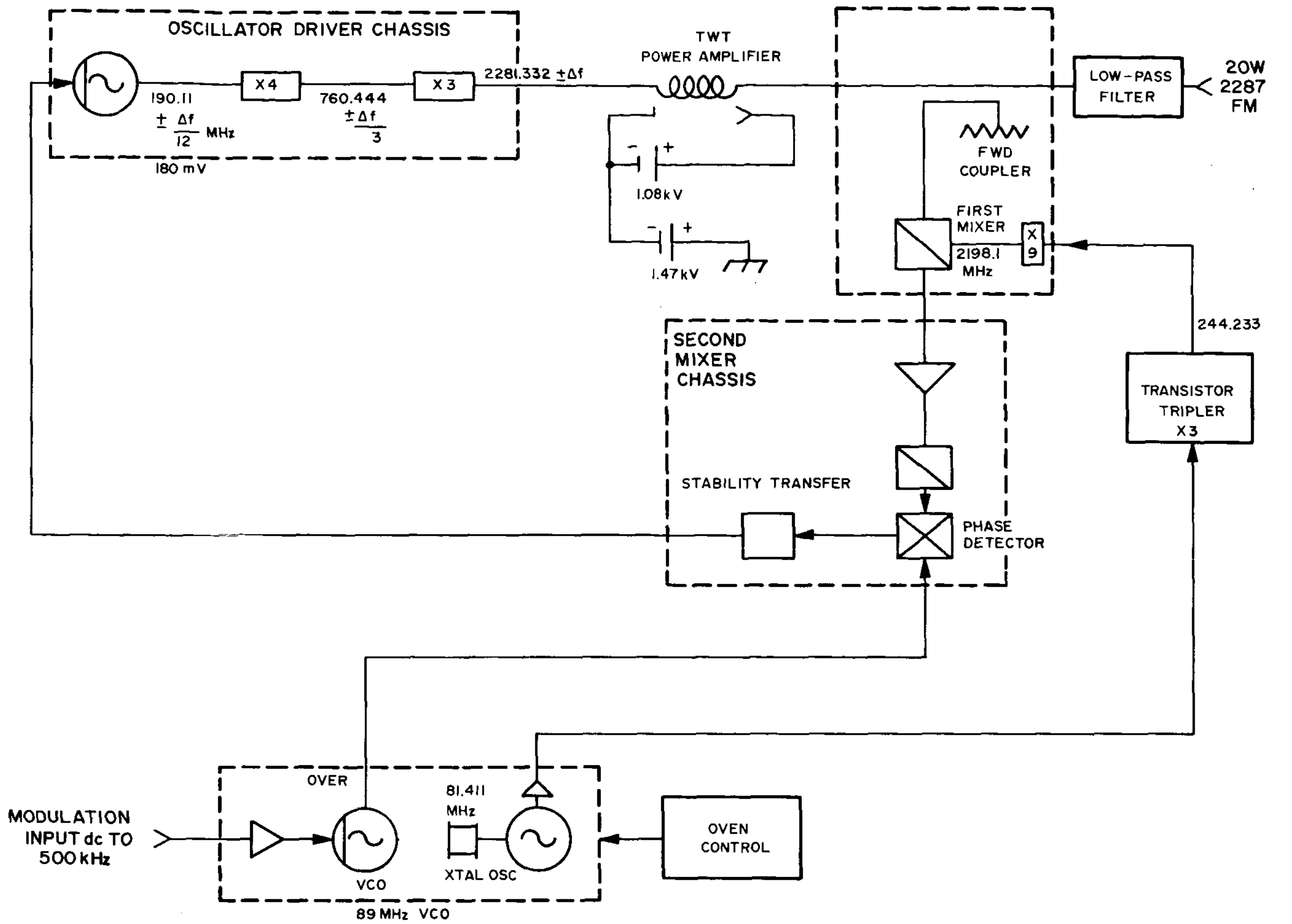


FIGURE 3. BLOCK DIAGRAM OF 20-W HYBRID TRANSMITTER

because of limited experience with TWT's under high vibration and low incidental FM requirements, that the vibration of the relatively long helix might cause a phase deviation which would have the appearance of incidental FM. The phase-locked loop connected in the feedback arrangement described tends to remove any phase modulation generated within the TWT. There were some data at the time the design approach was finalized indicating that the particular method of supporting the helix used by Hughes limited the peak phase deviation to only 2 or 3 degrees, with vibration levels exceeding those in the specification. However, it was deemed desirable to incorporate this scheme as a backup in case this turned out to be an optimistic estimate. However, the development contractor recently reported that there is very little difference in the incidental FM of the transmitter under vibration with the loop open or closed, indicating that the tube does not generate a significant amount of incidental FM when subjected to vibration.

The prototype 20-W transmitter was first tested at MSFC in December 1964. Several deficiencies in the design were discovered and the unit was returned to the contractor for additional development work. Subsequently, a unit which met most of the specifications was delivered in September 1965. At this time the contractor stated that he felt the existing design was not reproducible on a production basis. Redesign is now underway to alleviate this deficiency.

IV. FIVE-WATT SOLID-STATE S-BAND TRANSMITTER

In June 1963, MSFC awarded contract NAS8-11771 to Energy Systems, Inc., for the development of a 5-W S-band solid-state transmitter. The technical approach for this transmitter design is shown in

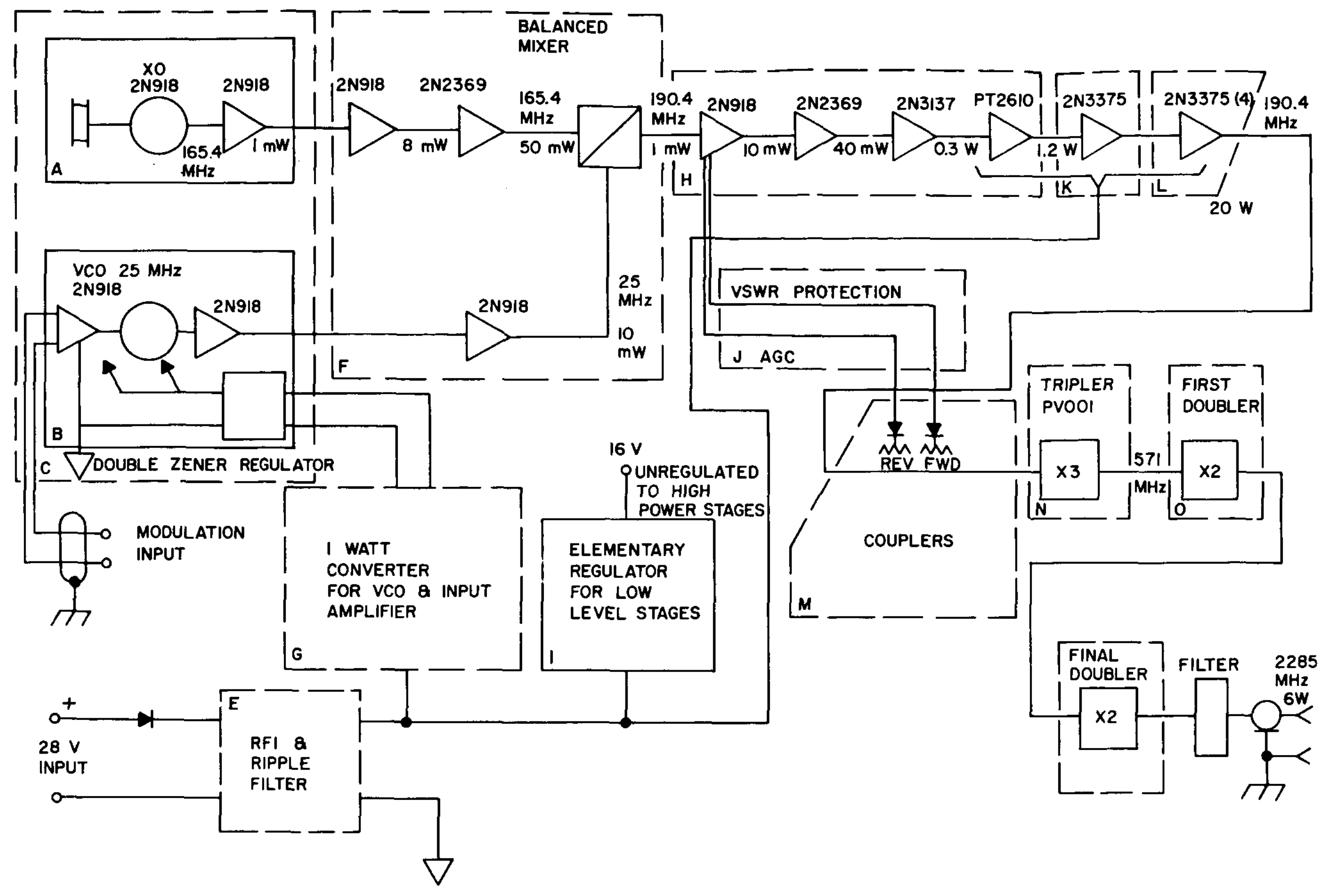


FIGURE 4. BLOCK DIAGRAM OF 5-W SOLID STATE S-BAND TRANSMITTER

Figure 4. A transistor that became available about the beginning of the development effort, the 2N3375 by RCA (development transistor RCA TA2307), enhanced the feasibility of the approach. It uses a type of construction called an "overlay" which employs an emitter electrode design whereby more periphery-to-emitter-area ratio has been achieved. This transistor is capable of operating directly from a 28-volt supply making it practical to operate high power VHF stages without voltage converters. The transistor is electrically insulated from the case and is connected through a high-thermal-conductivity ceramic insulator to its stud.

The fundamental frequency is generated by a stable crystal oscillator essentially identical with the one used in the 20-W TWT transmitter. The same package contains a one-stage amplifier to boost the power of the oscillator output to 1 mW. The milliwatt power level is increased by two amplifiers in series to obtain approximately 50 mW into the balanced mixer circuit. The other input to the mixer is generated by a voltage-controlled oscillator and its buffer amplifier. The VCO has a center frequency of 25 MHz and is approximately 10 times less stable in frequency than the crystal oscillator signal. However, since the VCO frequency is $7\frac{1}{2}$ times less than that of the crystal oscillator, the effect of VCO instability on the output frequency is correspondingly decreased. The VCO is driven from a stabilized dc amplifier which provides an input impedance of 10,000 ohms at the modulation input. The 25-MHz VCO is essentially the same design as the 89-MHz VCO used in the 20-W TWT transmitter. The stability is expected to be on the order of one part in 10^5 for the VCO and one part in 10^6 for the crystal oscillator. Therefore, the transmitter frequency stability should be approximately three parts in 10^6.

Based on the experience gained from the hybrid UHF transmitter, the balanced mixer uses hot carrier diodes. A balanced mixer was selected instead of a single-ended one to obtain the maximum possible elimination of the carrier frequency power with the intrinsic circuit characteristic without relying totally on filtering. The power is then introduced to a string of five amplifier stages. These amplifiers, because of their staggered tuning, suppress unwanted frequency components of the mixer circuit and prevent these spurious signals from being applied to the varactor multipliers.

A parallel arrangement of transistors in the final power amplifier is used because it is simpler than a push-pull arrangement. This simplicity is realized only if separate emitter resistors are used, tending to make the stages self-adjusting so that their outputs will be more nearly identical. These four transistors are normally operated in a class C mode to obtain the best possible efficiency, and the collector current operating point is selected to obtain the closest possible approach to optimum large signal gain bandwidth.

The 20 W of VHF developed power is delivered through a pair of directional couplers to the varactor tripler, which operates at an efficiency of about 65 percent with an output power of about 13 W. Traps are placed at the input and output to prevent feedback of the second and third harmonics to the amplifier and the first and second harmonics to the quadrupler. The last two stages are doublers which provide an output of about 6 W at 2285 MHz.

The reverse power output from the directional coupler is used for voltage standing wave ratio (VSWR) protection and the forward power output is used for automatic gain control (AGC). The same control point is used for both VSWR and AGC controls and the signals are decoupled by diodes. High VSWR's at the quadrupler output of a multiplier chain of two stages are reflected to the final amplifier stage, but the transistors are protected by the VSWR detection circuit which reduces the power level of the final power amplifier proportional to the VSWR magnitude.

Breadboards of all sections of the 5 W UHF solid-state transmitter have been completed and tested both individually and as a system. All parameters that could be tested met the requirements with an output power greater than 6 W. The transmitter is now being packaged for final testing.

V. TWENTY-WATT SOLID-STATE TRANSMITTER DEVELOPMENT

In mid-1965 it seemed practical to undertake the development of a 20-W solid-state transmitter although it had not appeared feasible a year earlier. This decision was based on the rate at which the solid-state component companies had been completing designs for new S-band devices. The rate of progress in development of these devices makes it probable that changes in a design will be made before the production unit is released. Therefore, this task must follow the state of the art of solid-state components which are suitable for applications in S-band, high-power transmitters for airborne applications.

In June 1965, MSFC awarded contract NAS8-20505 to Energy Systems, Inc., for development of a 20-W solid-state UHF transmitter. The proposed approach to the design of this transmitter is shown in block diagram form in Figure 5. Note that the approach is similar to that followed in the 5-W S-band transmitter. The major challenge lies in the greatly increased RF power level. None of the low-level circuitry of the 5-W S-band transmitter, including the crystal oscillators, VCO, mixer, etc., requires significant changes to raise the power level by 400 percent. However, some minor differences exist because of the development of transistors which operate satisfactorily at higher frequencies and powers since the inception of the 5-W transmitter design.

The modulation input is applied through the VCO circuit to a balanced mixer where it is mixed with the reference signal from a crystal-controlled oscillator. The balanced mixer is essentially identical with the one used in the 5-W transmitter with a frequency scale-up. The mixer uses high performance silicon diodes, hot carrier diodes, and varactors. The bandpass filter is used to further enhance the spurious response of the circuit over what the intrinsic operation of the balanced mixer will provide.

The power is stepped up 10 db in each of the next two stages, increased to 1 W by the next stage, and finally to 5 W by the stage preceding the main amplifier.

There is now in development by RCA for NASA/GSFC a new transistor, the TA2675, which is expected to provide 20 watts of RF power at 430 MHz. The collector efficiency is expected to be about 50 percent and the gain about 6 dB at 430 MHz. The potential advantage in using this transistor is that the intermediate amplifier would require only one transistor and the final power amplifier would require only four paralleled transistors. These amplifiers must use two and eight 2N3733 transistors in parallel, respectively, if the TA2675 or an alternate does not become available.

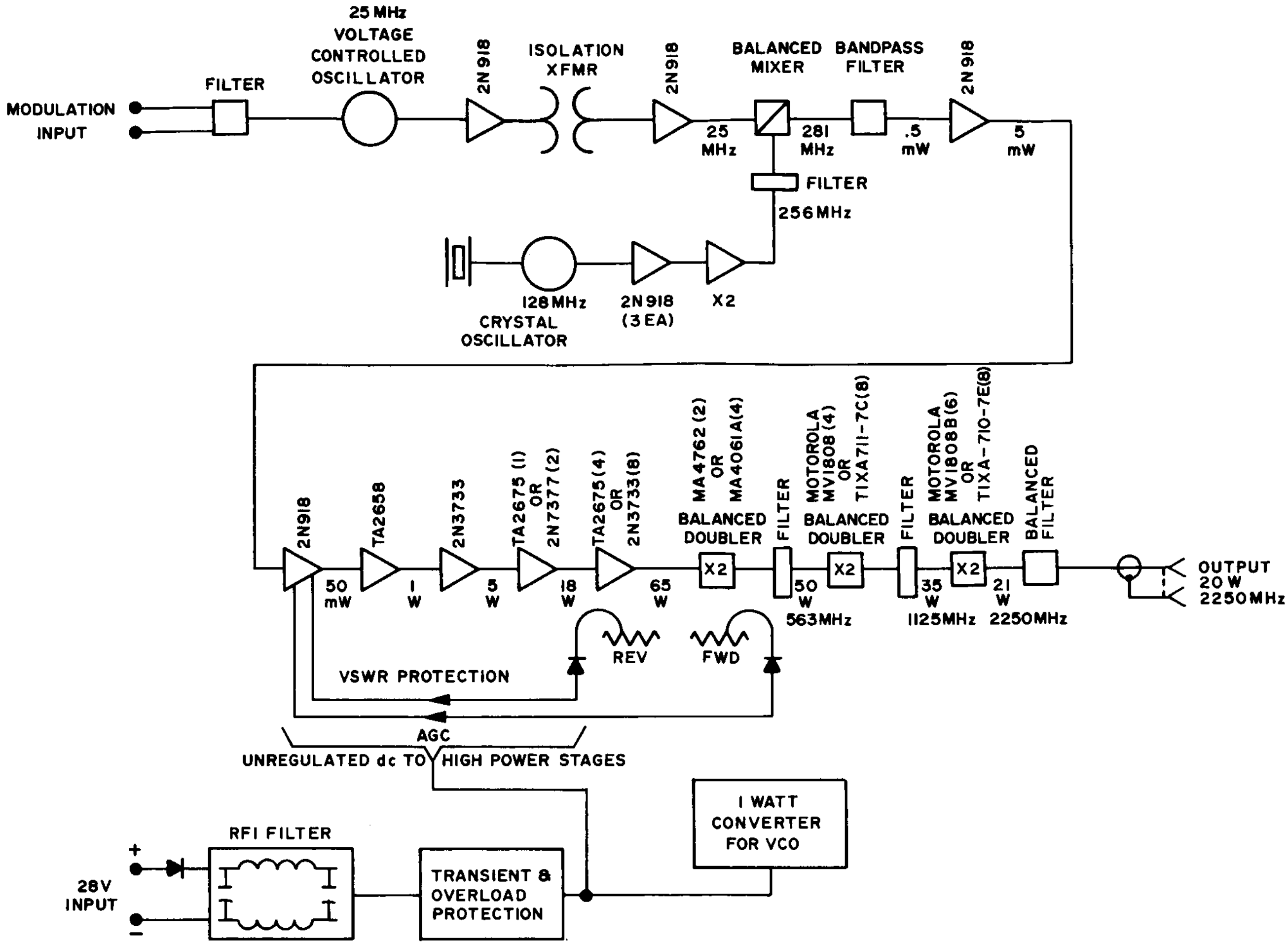


FIGURE 5. BLOCK DIAGRAM OF 20-W SOLID-STATE UHF TRANSMITTER

The present approach to paralleling the transistors is to lay them out in a symmetrical fashion in a circle around a center feed point to assure that the phase of the drive to each transistor is identical. Figure 6 is an example showing how the input circuitry to the paralleled transistors is expected to be arranged. (This shows the worst case of using eight transistors but the principle will remain the same if only four are used.) The 5-W source is delivered to two capacitors which provide a step up of impedance at point A. Transformers are symmetrically spaced in a radial pattern about the capacitor C_1 to resonate each of the input circuits. A step-down transformer, consisting of only one or two turns, is then used to link couple the input of each transistor. This provides the necessary physical spacing between the units so that they may be practically arranged on a common plate and also provides the lower impedance necessary to drive the large transistors.

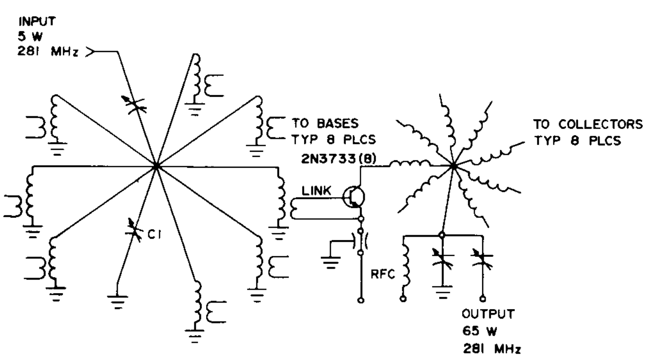


FIGURE 6. PARALLEL VHF POWER TRANSISTOR AMPLIFIER

The output of the main amplifiers is 65 W at 281 MHz. This output is applied to the multiplier change consisting of three balanced doublers. The output of the first doubler is 50 W at 563 MHz; the second is 35 W at 1100 MHz; and the third is 20 W at 2200 MHz. The development of the 20-W solid-state UHF transmitter is in an early phase with major emphasis on determining the most appropriate solid-state devices to use.

VI. CONCLUSION

In February 1965, the Department of Defense directed the three services to completely remove telemetry operations from the 225 to 260 MHz band by 1970. Until recently NASA planning was based on the use of the 225 to 260 MHz band for an indefinite period beyond 1970. Shortly after the firm directive to the three services, DOD requested that NASA also vacate the VHF telemetry band so that these frequencies might be used for other high priority applications.

In June 1965, NASA convened an intercenter study group to investigate and define NASA requirements for telemetry frequencies beyond 1970 and to evaluate the impact of the move on program costs and schedules. The results of this study had not been released at the date of this report.

If NASA initiates the move in telemetry operations as requested by DOD, one of the major problem areas is likely to be availability of RF transmitters with the required performance and environmental capabilities. The purpose of the telemetry development program described in this report is to meet this requirement.

ADDRESSABLE TIME DIVISION DATA SYSTEM

By

Roy Williams

ABSTRACT

The advantages and design considerations of a central programer, a common address, and a data return bus are compared to the telemetry system of the Saturn IB.

I. INTRODUCTION

Present telemetry systems route individual transducer signals to a central telemetry package, adding the bulk and mass of numerous wires to the vehicle. The system described herein will eliminate most of this wiring by using a central programer, a common address, and a data return bus.

II. PRESENT DESIGN

With the increase in size and mission requirements of space launch vehicles has come an accompanying demand for measurements; this requires a significant increase in cables and connectors. The current study on an addressable time division data system seeks to eliminate most of these cables and connectors.

As an example of what may be done in this area, consider an application from the 200-series vehicles. The Saturn IB space launch vehicle shown in Figure 1 gives an overall view of the distance that some conductors are run. The telemetry package used as an example is in unit 13. Sixty percent of the transducers monitored are located in unit 9 next to the engines; 30 percent are located in unit 12.

Figure 2 shows that to reach from the instrument compartment in unit 13 to the measuring distributor in unit 9, 34 meters of conductors are required for each measurement; to unit 12, it takes 7.62 meters. The wire presently being used for these runs weighs 2.0 kilograms per 305 meters, which gives us approximately 0.22 kilogram of wire to unit 13 for each measurement and 0.05 kilogram to unit 12.

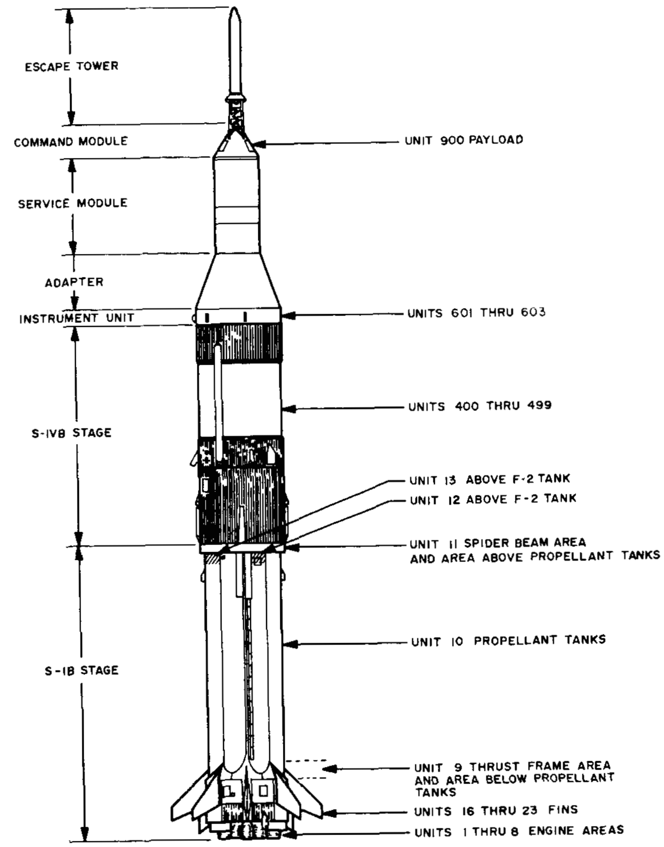


FIGURE 1. SATURN IB STAGE AND ELECTRICAL UNIT DESIGNATIONS

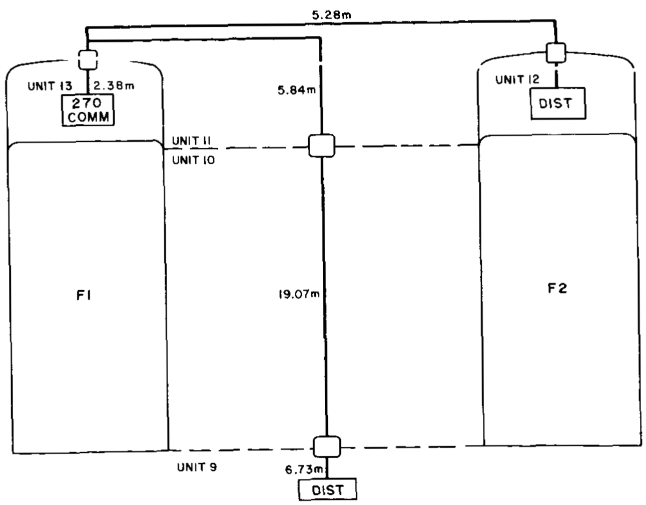


FIGURE 2. S-IB STAGE

III. PROPOSED DESIGN

If one wire could be used to carry the information and one wire to select or address the desired information, it would be possible to save approximately 41 kilograms minus the amount of weight added to the instrumentation. This is the type of system presently being evaluated.

Figure 3 shows the major elements of a system of this type. MSFC has a contract with Martin-Denver to furnish a study and prototype of the cables and measuring sources. The master programer may be designed in house if the concept proves feasible.

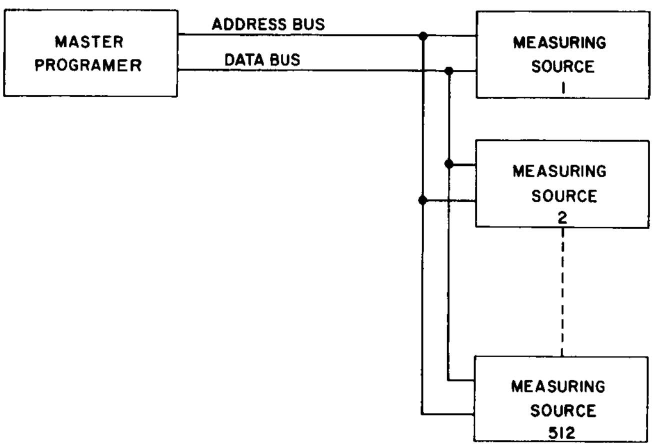


FIGURE 3. ADDRESSABLE TIME DIVISION DATA SYSTEM

The master programer consists of an interrogator and an analog-to-digital converter. The data will be transmitted in analog form from the measuring source to the master programer. There it will be converted to pulse code modulation. This signal then will be used to frequency modulate a transmitter. The measuring source consists of an address decoder and an address decoder and data switch. One measuring source might handle several transducers.

The interrogation signal consists of a serial digital waveform, containing the address of the measuring source to be interrogated, and timing or synchronization information. This signal uses one of the two system cables. The present plan is to use a return-to-zero signal with double amplitude pulses for synchronization as shown in Figure 4 for a 10-bit address.

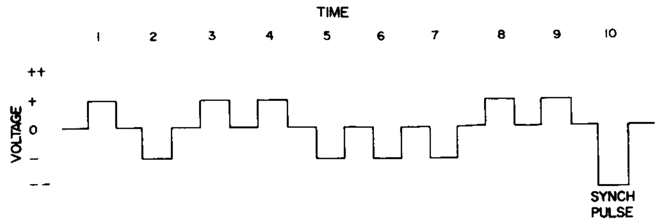


FIGURE 4. RETURN TO ZERO ADDRESS WAVEFORM

IV. ADVANTAGES OF PROPOSED DESIGN

The advantages of using a serial digital interrogation signal are that circuitry on both the transmitting and receiving ends is simple (no filters or oscillators are necessary) and integrated circuitry can be used to increase reliability and to reduce size, weight, and power consumption. To allow for expansion of the system, each address can be made one or two bits longer than the immediate requirements, thus adding two extra bits. For example, from 8 to 10 would expand the capability from approximately 500 to 1000 addresses. The address words will be easily changed, making it possible to interchange measuring sources and relocate them easily. The measurement source (Fig. 3) must be able to perform two functions; first, it must decode the address to the data system and, second, upon being addressed it must switch its data to the data bus.

The address decoder's function is to identify the address information from the address bus and convert it to a switching signal for interrogation of a specific data source. The decoder also must derive a clock or synchronization pulse from the address or contain a clock synchronizer to the address; in our case a synchronization pulse is transmitted with the address. It is important that decoder circuitry be highly reliable. A failure of a decoder can mean the loss of data from one or several sources; therefore, reliability will have to be a major consideration.

The measurement source switch can be one of two types; analog or discrete. The discrete data will be quite simple to handle; however, the analog, which can be low level or high level, will be much more difficult because of switching noise and switching voltage offsets. The data can be returned by three methods; pulse amplitude modulation, pulse position modulation, and pulse code modulation. It is planned that pulse amplitude modulation will be used. Pulse code modulation, from the standpoint of signal only, is the most desirable; however, this would require an A/D converter for each measuring source. A/D converters are relatively large and expensive and would add a large amount of mass to the system. Because we operate in a closed system over a relatively short distance, pulse amplitude modulation can be used with little deterioration of the signal.

V. CONCLUSION

It can be seen that mass can be reduced by eliminating cables and reducing the size of the connectors. However, there are some problems in the area of reliability that must be overcome. As soon as it is possible to predict how much instrumentation is involved, an investigation of mass tradeoff will be initiated to determine the extent of the program.

NASA TM X-53394

APPROVAL

INTRODUCTION AND SUMMARY TO
INSTRUMENTATION RESEARCH AT MSFC

The information in this report has been reviewed for security classification. Review of any information concerning Department of Defense or Atomic Energy Commission programs has been made by the MSFC Security Classification Officer. This report, in its entirety, has been determined to be unclassified.

This report has also been reviewed and approved for technical accuracy.

W. Haeussermann

W. HAEUSSERMANN
Director, Astrionics Laboratory

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MSFC INTERNAL (Cont')

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In a prepared statement presented on August 5, 1965, to the U. S. House of Representatives Science and Astronautics Committee (chaired by George P. Miller of California), the position of the National Aeronautics and Space Administration on Units of Measure was stated by Dr. Alfred J. Eggers, Deputy Associate Administrator, Office of Advanced Research and Technology:

"In January of this year NASA directed that the international system of units should be considered the preferred system of units, and should be employed by the research centers as the primary system in all reports and publications of a technical nature, except where such use would reduce the usefulness of the report to the primary recipients. During the conversion period the use of customary units in parentheses following the SI units is permissible, but the parenthetical usage of conventional units will be discontinued as soon as it is judged that the normal users of the reports would not be particularly inconvenienced by the exclusive use of SI units."

The International System of Units (SI Units) has been adopted by the U. S. National Bureau of Standards (see NBS Technical News Bulletin, Vol. 48, No. 4, April 1964).

The International System of Units is defined in NASA SP-7012, "The International System of Units, Physical Constants, and Conversion Factors," which is available from the U. S. Government Printing Office, Washington, D. C. 20402.

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