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A SUMMARY OF CONFIGURATION DESIGN STUDIES FOR A RANGE INSTRUMENTATION CALIBRATION SATELLITE

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DDC AVAILABILITY NOTICES

FOREWARD

This Technical Documentary Report (TDR) is published to summarize the results of five separate investigations which examined the technical problems associated with the development of a range instrumentation calibration satellite. The study effort summarized herein was undertaken in FY 1963 under ESD Task 5930.03, Calibration of Range Instrumentation, in Project 5930, Range Trajectory and Orbital Measurements, The work in this task is now being continued under Project 7818, Calibration of Range Instrumentation.

The technical studies summarized in this report were conducted by the following contractors:

Contractor	Subject	Contract No.
General Electric Co.	MISTRAM Transponder	AF 19(628)-2992
General Dynamics/ Astronautics	GLOTRAC Transponder	AF 19(628)-3274
Motorola, Inc.	C-Band Transponder	AF 19(628)-3235
E. G. and G.	Optical Beacon	AF 19(628)-2979
Radio Corp. of America	Satellite Integration	AF 19(628)-3260

The Geodesy and Gravity Branch, Terrestrial Sciences Laboratory, Air Force Cambridge Research Laboratories, served as technical monitoring agency. The Directorate of Aerospace Instrumentation, Headquarters ESD, exercised overall management supervision. The author is the ESD project engineer in charge of this effort.

Technical Documentary Report ESD-TDR-64-188, Summary of Investigations Into Instrumentation Calibration, dated 31 January 1964, has been previously published under this task. That TDR summarized the in-house and contractor theoretical feasibility and trade-off studies which served as the basis for the satellite-oriented work described herein.

ABSTRACT

This report summarizes the results of five separate but related analytical investigations into the engineering design of a calibration satellite. The suitability and adaptability of the MISTRAM, GLOTRAC, and AN/DPN-66 C-Band radar transponders, and xenon flash optical beacon, for use in a proposed calibration satellite is discussed. The overall results of a separate satellite integration study are likewise presented. Design specifications have been developed which allow initiation of transponder development. The electronic transponders required in a calibration satellite can be developed essentially from existing missile-rated designs. Some significant changes must be made however to meet the more stringent satellite requirements, such as extended operational life (specified at six months), high vacuum, space radiation, and extended temperature ranges. The detailed preliminary satellite design study has established that an adaptation of the RCA TIROS satellite will fulfill calibration satellite requirements. Design areas where advanced techniques are required include the attitude stabilization and the command/control subsystems. A recommended program for development of an austere calibration satellite "package" is also presented, in lieu of a full-scale satellite program.

PUBLICATION REVIEW AND APPROVAL

This technical documentary report has been reviewed and is approved.

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SECTION 1 - INTRODUCTION

1.1 Statement of Calibration Problem

The problem of calibration of precision electronic trajectory measurement systems involves, basically, the separation and determination of the systematic errors present in the tracking system. Having some detailed knowledge of the form, magnitude, and behavior of these systematic components permits compensation for the errors, which results in increased system accuracy. Adequate calibration of tracking systems is becoming increasingly important as more and more stringent metric measurement requirements are being laid on ranges and global tracking networks. Extremely accurate position and velocity information is required for such needs as spacecraft command and control and ballistic missile guidance system evaluation. Unless tracking system systematic errors can be suppressed to, or below, the level of inherant random error, program objectives may not be met.

In any discussion of instrumentation accuracy, distinction must be made between evaluation and calibration. The metric evaluation of a tracking system consists of the determination of the accuracy of each type of measurement made by the system.

Evaluation usually extends no further than the compilation of a "catalog" of total errors in each channel of data, and a comparison of these with a set of specifications. The central objective of evaluation is the determination of whether or not a set of specifications or criteria are being met. The breakdown and analysis of errors proceeds no further than the separation of random and systematic components of the total system error.

Calibration begins where evaluation leaves off and has as its central objective the explanation and ultimate compensation for systematic errors defined in the process of evaluation. Thus the goal of calibration is the improvement of instrumentation system accuracy through the application of appropriate corrections to each channel of observations. The process of evaluation thus provides the "raw material" required for the calibration operation. Often a tracking system is able to meet its design specifications only after calibration. Furthermore, systems which do meet their specifications without calibration can often be significantly improved by means of a thorough calibration.

There are essentially three main steps involved in the calibration function. The first step is the determination of the type or form of the individual systematic errors which make up

the error budget. The mathematical representation of the individual systematic errors is usually expressed as an "error model" for the particular data channel. Having determined a suitable error model, methods must be found to determine the magnitude of the various terms in the model. The third major steps involves an examination of the short and long term stability of the error term coefficients. This nearly always demands a capability for repeatability of the experiment, to account for such non-constant effects as atmospheric refraction. It may also be found in this phase of the experiment that certain important systematic error effects were neglected in the original error model or, conversely, some terms may be found to be related or negligible.

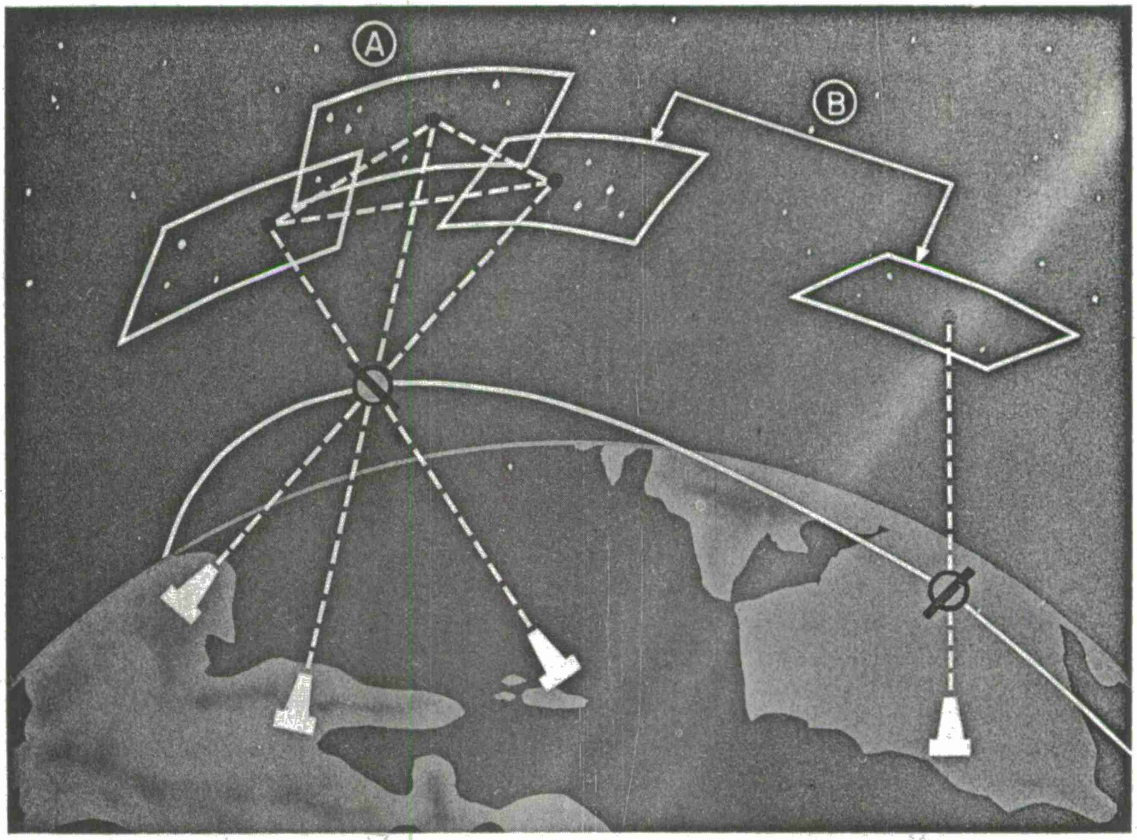
1.2 Basis of Investigations

The calibration function described in general terms in the preceeding section is generally implemented by comparing observed electronic tracking data with comparable data derived from a system of higher accuracy. The residuals so obtained are then analyzed through a regression analysis using the error expressions previously derived. Historically, the higher order data against which the electronic data are compared have been obtained by using an aircraft-borne optical beacon which could be photographed against

a star background. The aircraft method has a drawback in that it does not simulate the dynamic ranges and velocities encountered during normal tracker operations. Thus, significant error terms may not be exercised during the test. Also, since the aircraft is within the reasonable atmosphere, refractive effects are not the same as those experienced in missile tracking.

Extensive feasibility and trade-off studies (reference 1) have been performed which examined in some detail the general calibration problem and alternative methods of solution. The techniques usually advanced as potential solutions to the problem have involved applications of specially-designed satellites and/or rockets. The studies showed that both techniques are theoretically feasible but the satellite method was shown to be the most accurate and least costly when comparable programs were considered. Further, the satellite technique is the only method whereby a continuous check could be maintained on the critically-important long and short-term stability of the various terms making up the error model. The satellite also serves as a valuable geodetic tool for accurate survey of ground stations (figure 1).

The feasibility studies, which showed the overall advantage of the satellite technique, have therefore served as a logical basis



Mode A = "Intervisible" technique. Mode B = "Orbital analysis" technique . A third mode, not shown, called "Short Arc" technique, involves multi-observer triangulation of two non-intervisible flash sequences.

Figure. 1 Optical Beacon-Carrying Satellite operating as a geodetic tool for highly accurate positioning of both ground stations and satellite vehicle.

for the more detailed engineering-type analytical studies summarized herein. Since any first calibration attempt would certainly be expected to deal with existing range systems requiring calibration, the MISTRAM, GLOTRAC, and C-band radar systems were considered. The specific studies which were undertaken, and which are summarized in this TDR, are outlined in the following section.

1.3 Analytical Studies Undertaken

As mentioned, it was clear from the beginning of the calibration study that the initial calibration effort would involve precision CW interferometer and radar tracking systems on the Atlantic Missile Range (AMR). Analytical design and feasibility studies were therefore undertaken which were designed to show what detailed problems would arise in adapting missile-rated electronic transponders and an optical beacon for use in a range instrumentation calibration satellite. These studies were performed under in-house Air Force technical direction and management by the contractors who had developed the basic transponder design. Of prime consideration of course were the anticipated difficulties which could be expected due to high vacuum, space radiation, long required operating life, high required reliability, and extended temperature

ranges. The studies dealt with the transponders used with the MISTRAM and GLOTRAC interferometer systems, the AN/DPN-66 C-band radar beacon, and the xenon flash optical beacon. Additionally, a complete satellite integration study was performed which examined the problems anticipated in utilizing the above components, and other required associated equipment, in a full-scale calibration satellite. The procedures, results, and conclusions of all these studies are summarized in the following five sections of this TDR.

SECTION 2 - MISTRAM TRANSPONDER STUDY

2.1 Summary of Study

The study program summarized in this section was conducted by the General Electric Company and was directed toward the generation of an equipment specification for a MISTRAM transponder capable of operating in an orbiting satellite. It considered the unusual environmental conditions that will exist in the proposed calibration satellite and the operating requirements that must be met for successful mission accomplishment. It was also concerned with related hardware and other activities, such as satellite antennas and field test equipment and programs. The following specific areas of study were considered:

a. Environmental Considerations

   Thermal
   Vibration
   Altitude
   Irradiation
   Radio Frequency Interference
   Magnetic Field

b. Operational Considerations

   Antenna Requirements
   Reliability
   Primary Power
   Telemetry
   Magnetron Testing

c. Testing

Test Equipment
Test Philosophy

d. Equipment Specification

2.2 Results and Conclusions

a. Thermal

During the life of an electronics package, several different thermal environments will be encountered, namely, ground storage and shipping, bench or depot checkout, pre-flight checkout, boost flight phase, and orbital operation. Each of these thermal environments was considered for any possible adverse affects on the Satellite Rated Transponder package.

The ground storage and shipping thermal environment is, in nearly every case, a non-operating environment, and therefore no component or part of the package can exceed the temperature of the environment. Even if the package were stored in an unsheltered area, its temperature would not be expected to exceed about 71°C including the effect of solar heating. Since all components are rated to operate at temperatures of 100°C or above, the storage and shipping conditions should not cause any failure or degradation of components.

The bench checkout and missile checkout environments were considered together, since in both environments, the package will be operated electrically at sea level pressure. An analysis was performed to determine the average temperature while operating with the package dissipated heat removed by radiation and natural convection from its outer surfaces. The results of this analysis showed that the voltage tunable magnetron temperature should not exceed 97°C, which is within limits. Further, an analysis of the boost and orbital phases showed that the missile checkout environment is the most severe that the transponder will encounter.

b. Vibration

The dynamics effort was applied in three main areas: review of the proposed vibration specification, tests of critical components or sub-assemblies and review of design from the vibration viewpoint.

Without more detailed information about the satellite vehicle it is difficult to propose a specification. However, based on available information a very conservative or severe specification would be:

"The spectrum shall be flat from 20 to 320 cps at 0.165 g^2/cps. From 320 cps it shall rise at 6.6 db/oct. to a

magnitude of $0.3g^2/cps$ at 420 cps. It shall be flat from 420 cps to 1000 cps at $0.3g^2/cps$. From 1000 to 2000 cps the roll-off will be 9 db/oct. and above 2000 the roll-off shall be greater than 36 db/oct." The overall vibration level of this spectrum is 19g rms.

c. Altitude

One of the primary elements of the space environment is that of high vacuum, generally at some pressure level less than 10^{-4} millimeters of mercury. This high vacuum of space may degrade the life or operation of a piece of electronics equipment and was, therefore, considered with regard to the satellite rated MISTRAM transponder during the study program. Some of the effects of the high vacuum are reduction of conduction heat transfer at mechanical joint interfaces, sublimation or evaporation of materials, outgassing of materials, increased friction in sliding surfaces, and reduction of useable strength and fatigue resistance of structural materials. Although pressures less than 10^{-4} millimeters of mercury will increase somewhat the sublimation and outgassing rates of materials, no substantial package degradation is expected at lower pressures. Each of these effects was examined in detail. The results indicated that magnesium must be eliminated as a structural material and Teflon should probably not be used due to susceptability to radiation degradation.

d. Radio Frequency Interference

Because electronic equipment is found in close proximity, radio frequency interference has proven itself to be a key factor in systems performance. Military specifications on radio frequency interference (RFI) compatibility require that enough tests be made on an equipment to ensure that it is compatible.

The investigation and evaluation of test results indicate that there are no compatibility problems existing or contemplated for the satellite rated transponder passing the MIL-I-26600 specification which has been proposed for this program.

e. Magnetic Field

During the course of the study program the vehicle contractor indicated considerable concern that the magnetic field associated with the MISTRAM transponder would have an adverse effect on the stabilization system of the satellite. A dummy chassis structure of the proposed transponder design, with two magnetrons mounted in place, was made available to the vehicle contractor who took readings on the surrounding magnetic field. It was concluded that the magnetrons were the dominant factor and that core materials to be used in the power supply would not significantly change the results.

If it is deemed desirable, the final design of the Satellite Rated Transponder will have one magnetron with reversed polarity. If the resultant field still proves to be a problem for satellite stabilization, it is assumed shielding or other compensating methods will be provided by the vehicle contractor. However, these must be considered very carefully for any affect on the MISTRAM transponder. It is quite probable that large amounts of magnetic shielding material close to the unit would distort the field of the magnetrons enough to cause a significant shift in transmitted frequencies.

f. Antenna Requirements

One of the main purposes of this study program was to examine and determine what specifications are required for the MISTRAM transponder to accomplish a successful satellite mission.

One of the many contributors for errors in predicting the MISTRAM turn around time accurately is the antenna system. The major sources for error are VSWR, quality of antenna patterns and signal leakages. These sources can be severe enough to introduce additional phase errors in the system. It is therefore important that since the airborne antennas are a direct link in the systems performance that adequate consideration for the antenna design be indicated. The satelliteborne antenna

system shall accommodate and be compatible with the operational and performance requirements of the ground and satelliteborne portions of the MISTRAM system. Static and dynamic characteristics of the antenna system and its associated components will contribute to the overall system accuracy and performance.

g. Reliability

When a reliability requirement is placed upon a contractor in the design of equipments, there are two areas of cost involved. First, the cost of component part testing, research, and circuit reviewing during the equipment design phase. The second area of cost is concerned with the demonstration of compliance with the equipment specification regarding reliability. The high number of operating hours required to even demonstrate a reliability of 0.900 with 80% confidence becomes expensive when one considers the small number of transponders initially planned for this program.

To avoid the expense of verifying system MTBF in this program it is recommended that the following reliability requirement be placed in the equipment specification:

"The predicted reliability of the transponder shall be 0.900 for an operating time of 100 hours. This figure yields a required predicted MTBF of 950 hours. The predicted MTBF shall be derived from the failure rates of component parts used in the transponder."

h. Telemetry

For most space mission probes, it becomes necessary that some data be sent from the vehicle to the ground periodically. This data provides information which pertains to the general condition of the total complex operation of the satellite. Such items are necessary for correlating test data and predicting the overall systems performance.

It is anticipated for the MISTRAM unit, that there will exist twelve telemetry functions that will be brought to the MISTRAM transponder's output connector and telemetry channels should be available for them. It is obvious that the more test functions made available, the easier one my evaluate the performance of a system. The advantage is, should a failure occur, the point of failure may immediately be localized. Knowledge of points of failure will enable design correction of subsequent units to reduce future failures.

Each of the outputs from the transponder shall not vary beyond the limits of zero (0) to + 2.4 volts D. C. The

maximum source impedance of each of the functions is 2000 ohms. The sample rate of ten/second considered for these telemetry functions will be adequate. However, it would be desirable to use a slightly higher rate of thirty/second for the two major loop phase detectors. A measured accuracy of five percent should be specified for these readings.

i. Testing

The operating mode and mission requirements proposed for the satellite rated MISTRAM transponder will require quantitative capability for field testing. The suggested equipment to be used for this quantitative testing is a presently-designed Transponder Test Set, (TTS), modified where necessary to operate with the proposed transponder. Cost of design and manufacture of a new test set for this program would be high, so for economic reasons the present TTS with modifications is suggested.

The suggested parameters to be checked during field testing are:

1. The RF output characteristics (power, frequency and proper lock) of both the range and calibrate channel.

2. The receiver characteristics, sensitivity, tracking capability for dynamic range and frequency shifts and acquisition time.

3. Phase delay.

4. The power supply regulation (checking for proper operation across a range of input dc values).

5. Transponder current drain.

6. Measurement of voltage proportionals.

7. Gain margin.

j. Equipment Specifications

One of the major purposes of the MISTRAM study was to assist in the generation of equipment specifications for a satellite-rated transponder. The results and conclusions drawn from the 14 major areas of study summarized in the preceeding few pages resulted in such a specification. The complete details are not listed here but are available in the final contractor report published for this study (reference 2). From these proposed specifications, it has been possible to derive a complete Technical Exhibit for a satellite-rated MISTRAM transponder.

SECTION 3 - GLOTRAC TRANSPONDER STUDY

3.1 Summary of Study

The analytical investigations conducted by General Dynamics/Astronautics involved both theoretical study and a test program. The theoretical investigations were designed to provide recommendations for improving the reliability of the GLOTRAC Type G transponder when used in a proposed calibration satellite. The recommendations were to cover necessary redesign of the transponder as well as testing necessary to ascertain with a high level of confidence the transponder suitability. The test program was intended to provide information on the reliability of the transponder when operating under simulated satellite environmental conditions. The tests were designed to provide data which could be used to verify and complement the results of the theoretical investigation.

The theoretical investigations were subdivided into three major studies as follows:

a. Heat dissipation and packaging study

b. Component and circuit study

c. Radiation study

Each of these areas was examined with respect to the requirements peculiar to the calibration satellite mission.

The primary objectives of the heat dissipation and packaging study were as follows:

a. Find a method of sealing the transponder case reliably for the duration of the satellite mission.

b. Investigate problems occurring from unpressurized transponder operation in a high vacuum.

c. Determine what transponder modifications are required on account of temperature considerations.

As part of the component and circuit study, some of the original analyses conducted during the design phase of the basic GLOTRAC transponder were repeated and expanded. The reason can be found in the fact that the transponder was designed for a short-life mission at a relatively low altitude. The first task was an investigation of all components used in the transponder. All component parts lists were reviewed for proper selection, usage, and qualification to meet the operational environment and the specific circuit designs. With the exceptions of semiconductors, specification-controlled parts, and trimpots, all components meet military specifications and have a long history of usage under many types of environments. During the initial design phase of the

transponder, waivers were obtained from the Air Force for those parts not manufactured in accordance with military specifications. These parts were selected to meet the design requirements of the transponder and non-conformance to military specifications was unavoidable. Part manufacturer questionnaires were prepared to obtain additional information and data on their electronic components. Inquiries were made relative to part failure rates, parameter drift characteristics, effects of electrical and environmental stresses on the parts, failure modes and causes, failure rate statistical distributions, test data, and manufacturer's reliability surveillance programs. About one-half of the part manufactures responded to the inquiries and from these, only partial information was received. Other sources, such as preferred parts lists, were consulted to provide some verification that the parts used were suitable for their application and environment.

The objectives of the <u>radiation study</u> were to perform an analytical analysis to determine which materials and components are expected to fail when exposed to the satellite radiation environment. Furthermore, the study was to determine whether a radiation test is considered necessary to substantiate the analytical

calculations, to recommend the type of test and to suggest suitable test facilities. During its normal mission the GLOTRAC transponder remains in orbit for only a few hours. Radiation effects do not constitute a serious problem for this short interval. Although some care was taken during the original design phase to use radiation resistant parts, no concerted effort was made to build a transponder which could operate reliably in a radiation environment.

3.2 Results

For the normal GLOTRAC mission the transponder is designed to operate with an internal pressure of one atmosphere. Air is sealed into the transponder case by an 0-ring made of Buna-N. The seal is designed to hold in a vacuum up to 4×10^{-2} mm Hg. When the transponder is operated in orbit the vacuum will be approximately 10^{-9} mm Hg. The 0-ring is not expected to hold in such an environment for the required length of time.

Sealing of the transponder was suggested to overcome such problems as electric arcing and evaporation of materials. Arcing is not expected to be a serious problem in the GLOTRAC satellite transponder since the highest operating voltage utilized is only 56 VDC and since operation normally does not occur during the decompression phase. Evaporation will be appreciably accelerated

in a vacuum environment, especially when materials are exposed to radiation at the same time. For this reason it would be highly desirable to operate the circuits under normal pressure. However, appropriate choice of transponder materials can effectively eliminate the pressuriation requirement.

Wire-wrap terminals were considered for use in the transponder. However, tests performed on other associated programs indicate that these terminals do not lend them selves to usage in severe shock and vibration conditions. Also, the solid wire used with wire-wrap terminals is subject to breakage from shock and vibration. No further consideration was given to this method of interconnection. No problems are anticipated with the existing type of interconnection which utilizes solder type terminals on the modules and stranded wire between the terminals with interconnections soldered at the junctures. Wiring is cemented to the chassis at closely spaced points.

The following results were obtained in the study to improve the transponder <u>heat dissipation</u> characteristics:

a. Use of an all-copper enclosure on the power supply module will lower resistance to heat flow.

b. Thickening of the base portion of the enclosure will allow heat to be spread more evenly over the surface mating with the housing.

c. Controlling the surface finish and flatness of the mating surfaces will provide more effective area of contact.

d. Additional mounting studs will increase clamping force between the power supply enclosure and the housing and thus reduce heat flow resistance.

e. Addition of an interposing heat transfer material between the base of the enclosure and the housing will provide an improved heat flow path.

In addition to the results listed above, a polyurethane coating could be added in accordance with MIL-C-27227, to the outside surface of the top cover. This does not affect greatly the heat removal characteristics of the transponder, but will add to the reliability aspects due to increased heat radiation from a relatively large surface.

The results of the component and circuit study can be summarized as follows:

a. All components used in the GLOTRAC transponder have been properly chosen for their respective function in the circuits. The component investigations pursued during this study revealed no obviously better choice of parts with the exclusion of Hi-Rel Minuteman parts.

b. In general, the application of existing transponder components in the circuits is consistent with the requirements.

c. Investigation of Hi-Rel Minuteman components revealed that some passive components could be substituted on a one for one basis. For some active elements, there are similar part specification characteristics. However, present applied semiconductor technology in the transponder precludes the replacement of these parts without a certain amount of re-design work including breadboarding and testing.

d. Within the circuit function constraints, the initial part tolerance selections are considered adequate.

e. The d-c operating point stability of all transistor stages is considered adequate for the intended functions with the exception of one stage in the sweep network.

f. The estimated reliability of 0.97 is based upon a catastrophic component failure rate and is considered adequate for the calibration satellite requirement.

Finally, the study into radiation effects on the GLOTRAC transponder yielded the following results:

a. The electron component incident flux in a 400 n. mi. 40° inclination orbit is the most pronounced deleterious type radiation.

b. The electron flux incident upon a transponder component may be largely eliminated through the use of 4 g/cm^2 polyethylene shielding material. The attenuation of the electron flux by shielding properties of polyethylene, by the satellite framework, and by internal satellite components will increase the transponder component life.

c. Material Utilization Factors developed indicate that Teflon-100 will fall short of the operational six month mission life. Components and materials such as transistors and KEL-F have MU factors between 10 and 100. and are considered borderline.

d. Data are insufficient to permit a definite statement as to the radiation resistance of the transponder electronic circuitry based on analytical methods alone. Irradiation testing is necessary to select the best transponder configuration.

e. Calculations indicate that 3.12×10^9 electrons/ (cm^2-day) of 8.22 MEV electrons will be needed to simulate the environmental equivalent damage to the transponder components.

3.3 Conclusions

The GLOTRAC transponder was designed with a specific mission in mind. One of the major goals was to fabricate a small, light-weight unit. Quite often, the requirements of small size and

light weight do not coincide with the requirement for high reliability. In such cases, realistic trade-offs had to be made. This study has revealed that in a number of areas, these trade-offs, although adequate for the GLOTRAC mission, will require some reconsideration for the satellite mission. Furthermore, the satellite mission has added design considerations which were not required previously. Space radiation is an example.

This section summarizes the conclusions arrived at during this study. Study results have shown that modifications are necessary to assure transponder reliability. In a number of areas, design changes may be desirable only if they can be made simply. In these cases, some further investigation is recommended to remove all doubt. The specific conclusions, as related to the three major study areas, are as follows:

<u>Heat Dissipation and Packaging Study</u>

a. Remove the 0-ring and operate the transponder unsealed.

b. Modify the electronic modules so that they can be completely evacuated. Any trapped air may become ionized and interfere with the proper operation of the module.

c. Cadmium, zinc and magnesium display high sublimation rates. Replace these metals with aluminum or stainless steel.

Change the material of the transponder top cover from magnesium to aluminum.

d. Conduct a vacuum test at 10^{-9} mm Hg. on the modified transponder. This corresponds more closely to actual space environment than the 10^{-6} mm Hg test specified.

e. Perform a test designed to detect electrical arcing during the decompression cycle.

f. Conduct a vacuum test to determine the suitability of polyurethane foam for the satellite environment.

g. Use a thermal conducting putty on the mounting surface of the transponder to improve the flow of heat.

h. Use a copper enclosure for the power supply module to improve the heat flow.

Component and Circuits Study

a. One transistor in the amplifier multiplier module can, under certin conditons, reach a junction temperature of 181°C. It is recommended that a heat sink be provided for the transistor.

b. One transistor in the power supply can, under certain conditions, reach a junction temperature of 190°C. This transistor is already heat-sinked and it is recommended that this stage be redesigned to lower the junction temperature.

c. Transistors in both the VCO and discriminator modules have rated collector-to-emitter breakdown voltages of 15 volts. In these circuits, the working collector-to-emitter voltages are 14.5 volts and 14.0 volts respectively. It is recommended that these stages be redesigned to reduce the collector-to-emitter voltage or that transistors with a higher V_{CE} ratings be employed.

d. Consistent with the adopted derating guidelines, certain components in several circuits should be derated further. This derating can be accomplished by using parts with higher ratings, or, in other instances, paralleling of parts with the same ratings.

e. Under certain conditons, a transistor in the sweep network can have its d-c operating point near transistor saturation. With the a-c signal level applied, clipping of the waveform could occur, which might result in a reduction of transponder lock-on bandwidth. It is recommended that this stage be redesigned to make allowance for drift of part parameters due to temperature.

f. The harmonic generator represents an advance in the state-of-the-art. Reliability information on some of its special components, such as the high frequency varactor diodes, is not available. A more detailed study of the harmonic generator

reliability should be performed.

Radiation Study

a. It is recommended that the transponder sensitivity to radiation be reduced by additional shielding, such as a coat of polyethelene.

b. Replace teflon with polyethelene.

c. Perform a radiation test on the modified transponder.

It should be noted in conclusion that none of the above conclusions and recommendations were altered as a result of extensive testing carried out to verify the theoretical results. The complete details of the entire GLOTRAC transponder feasibility study are available in the final contractor report published under this contract (reference 3).

SECTION 4 - C-BAND TRANSPONDER STUDY

4.1 Summary of Study

The third major study effort to be summarized in this TDR was conducted by Motorola, Inc., and was designed to show the suitability of the AN/DPN-66 C-band transponder for operation in a range calibration satellite. The contractor performed, in essence, the following major tasks:

a. Determine the location and extent of any possible problem areas, determine solutions for these problems, apply these solutions to a test unit, conduct tests to verify the validity of the solutions, and document the study and test results for use as the basis for transponder design changes.

b. An accelerated life test at high altitude, with frequent on-off cycling of the test unit, was performed on the AN/DPN-66 in order to ascertain the effects of this type of environment on the performance of the transponder.

c. The existing test procedures, and the test equipment required, were evaluated with respect to the satellite end use, expected transponder modifications, and the best and most economical methods for tests, with the results of this evaluation to form a recommendation.

d. The sources, spectra, and effects on transponder performance of probable radio frequency interference, as well

as the required corrective isolation and filtering, were determined.

e. Investigate and define the transponder antenna requirements for satellite use, including the effects of satellite stabilization, receive-transmit isolation requirements, and antenna gain and coverage requirements.

4.2 Results and Conclusions

All aspects of the study program show very favorable results, with only minor problems encountered. RFI tests were satisfactorily completed. The occurrence of destructive corona and/or arcing within the unit for a range of pressure corresponding to altitudes between 60,000 and 200,000 feet were expected according to Paschen's Law. Tests showed that destructive arcing does occur and operation of the transponder within the above pressure range must be prevented. Several techniques for achieving this solution are proposed.

The AN/DPN-66 performance characteristics during the life - altitude test indicate that the major problem areas that would occur during the proposed transponder use in a space environment have been defined and apparently solved. While the test unit met or exceeded the performance goals set for this initial test, it is felt that the relatively short duration of the test and the small size of the test sample mean that the data

obtained must be viewed as qualitative, not quantitative. Until further testing substantiates the observed performance, thereby increasing the level of confidence in the ability of this transponder to successfully complete the required mission, Motorola must recommend use of this transponder in a redundant configuration.

The final electromagnetic compatibility tests showed that the modified AN/DPN-66 fully complied with MIL-I-26600. Half of the semiconductor types used in the transponder could be procured to MIL-I-19500. While tentative MIL-I-19500 substitutes have been found for most of the remaining devices, Motorola feels that the circuit redesign required to incorporate the MIL semi-conductors would result in a lower reliability due to the loss of production history. The reliability analysis indicated that the addition of telemetry circuitry had negligible effect on the probability of mission success. The use of a redundant transponder configuration, as recommended by Motorola, will increase the probability of success from 87 per cent to 98.3 per cent. The paper design of a diplexer is complete. A proven antenna design is proposed, with alternate designs suggested should the satellite stabilization technique be changed. Corona and arc-over within the unit can be avoided

by reduction of internal pressure to orbital ambient. Several methods for relieving the pressure are suggested. The life-altitude test results were favorable, with the primary problem area remaining being transmitter frequency drift with age. The magnetron vendor claims that this problem has been solved by tuner redesign.

4.3 Recommendations

The results of this study can be divided into two classifications: areas wherein the unit has met the requirements; and areas wherein further investigation is recommended in order to ascertain or ensure the ability of the transponder to perform successfully the requirements of its proposed mission. The first item has been summarized in section 4.2 above. In order to accomplish the second item, the following recommendations should be considered:

a. The addition of five telemetry circuits to monitor and predict transponder orbital operation.

b. Test of a transponder to the expected satellite vibration levels and correction of any problems.

c. Use of a redundant transponder to meet the desired reliability.

d. Writing Brief Operating and Installation Instructions as a supplement to the AN/DPN-66 Technical Manual.

e. Providing technical liaison during transponder installation and final system check out.

f. Testing of a positive type of pressure relief if the pre-launch environment so demands.

Prior to the performance of any of the above tasks a more complete definition of the Satellite power supply, expected vibration levels, and pre-launch environment is required. Additional RFI testing, although not recommended, can be done if system tests indicate the necessity. As with the other transponder studies summarized in this TDR, the complete details are available in the final study report (reference 4).

SECTION 5 - OPTICAL BEACON STUDY

5.1 Summary of Study

The following tasks were undertaken by Edgerton, Germeshausen, and Grier, Inc., under a contract to evolve a preliminary design plus a set of definitive preliminary specifications for a xenon flash system which could be carried aboard the calibration satellite:

a. Space System Hardware - Each major component is to be studied individually and bread-board systems will be fabricated and tested to simulate functionally the final anticipated beacon system.

1. Converter - The converter will be redesigned to accept a new transistor to replace the one presently used in ANNA. This will necessarily dictate the new design of the transformer parameters. The regulator in the converter system will be changed to sense both the positive and negative capacitor voltages, or the total voltage summation rather than just the positive as it does in the ANNA-1B satellite.

2. Sequence Controller - New type shift registers will be incorporated as well as new transistors and general components.

3. Capacitor Bank - A re-evaluation of the capacitors will be made to determine whether or not to continue with the electrolytic type or to use a new metalized paper type. A review of hermetic sealing techniques will be made for sealing the capacitor bank assembly.

4. Trigger Circuit and Lamp Assemblies - An attempt will be made to provide a means of disconnecting the lamps from the trigger assembly. New lamps and lamp-reflector assemblies will be investigated to give a greater luminous efficiency for a given beam angle. This investigation will include fabrication of lamp reflector assemblies and the measurements of beam patterns.

5. Power Relay - A search will be made for a newer type relay or other components as a replacement in order to enhance reliability.

6. Packaging - On all the above items, an investigation into new packaging techniques will be made to allow for a more compact construction, which in turn will reduce size and weight. New potting compounds and three-dimensional packaging will be considered.

b. Battery Specifications - A review of the specifications and requirements for the optical beacon battery will be made. These specifications and requirements will cover such things as storage capacity, voltage, internal resistance and input and output currents expected.

c. Memory - A study of the electrical interface problems with respect to the memory or other programming equipment for the optical beacon will be performed.

d. Emergency Override System - A study in use of an emergency override system and what functions it should be capable of overriding will be made.

e. Reliability - As "state-of-the-art" components are selected, a reliability program will be instituted to determine their fitness for such a program. This will include component applications with respect to stress levels and combined operating environments. The mean time between failure, with applicable confidence levels, will be determined.

f. Additions or Changes to the Optical Beacon Requirements

Studies will be performed on the ANNA 1B data to determine correct light intensity, optimum time interval between flashes and adequate telemetry data output.

g. Payload Integration - Services will be provided as necessary relative to the overall payload integration with respect to the optical beacon.

h. Light Versus Film Emulsion Compatibility Study - A study shall be conducted to determine what film emulsions are most compatible with the light characteristics generated by the beacon.

i. Light Output Versus Image Size - A study shall be conducted to determine the relationship of light output to image size and the optimization of light output.

5.2 Results and Conclusions

Edgerton, Germeshausen & Grier, Inc., has evolved a preliminary design plus a set of definitive preliminary specifications for a xenon flash system which can be carried aboard and meet all the optical needs of the proposed Range Calibration Satellite. The resulting optical beacon should both fulfill the demanding light output requirements of the stellar cameras to be used for range calibrations and at the same time come within the severe constraints imposed by the satellite itself as to limited available electrical energy, low permissible weight, high required reliability and rigorous launch and in-orbit environments. In an attempt to keep this TDR reasonably brief, the detailed preliminary design is not included here. The interested reader is referred to the final study report by E. G. & G. (reference 5).

Back of the preliminary design results lies a series of thorough studies made by EG&G to evaluate the major system components. The specifications that result reflect an analysis and experimental verification of functional performance, reliability and general suitability of each major functional system component.

As further back-up for the proposed beacon design, supporting studies have been made of ANNA-1B telemetry data, photographic plates have been analyzed, and an emulsion selection study has been performed. The results therefrom have aided in proper selection of system parameters.

The total design effort leans heavily on the ANNA experience -- the first successful satellite with a xenon flash optical beacon aboard. Significant electrical and optical improvements have, however, been proposed, evaluated, and finally specified only where fully justified. The resulting design specifications should, therefore, reflect a balance between proven experience and realistically achievable improvements. The block diagram of the proposed xenon flash system and a typical satellite installation are shown in figures 2 and 3 respectively.

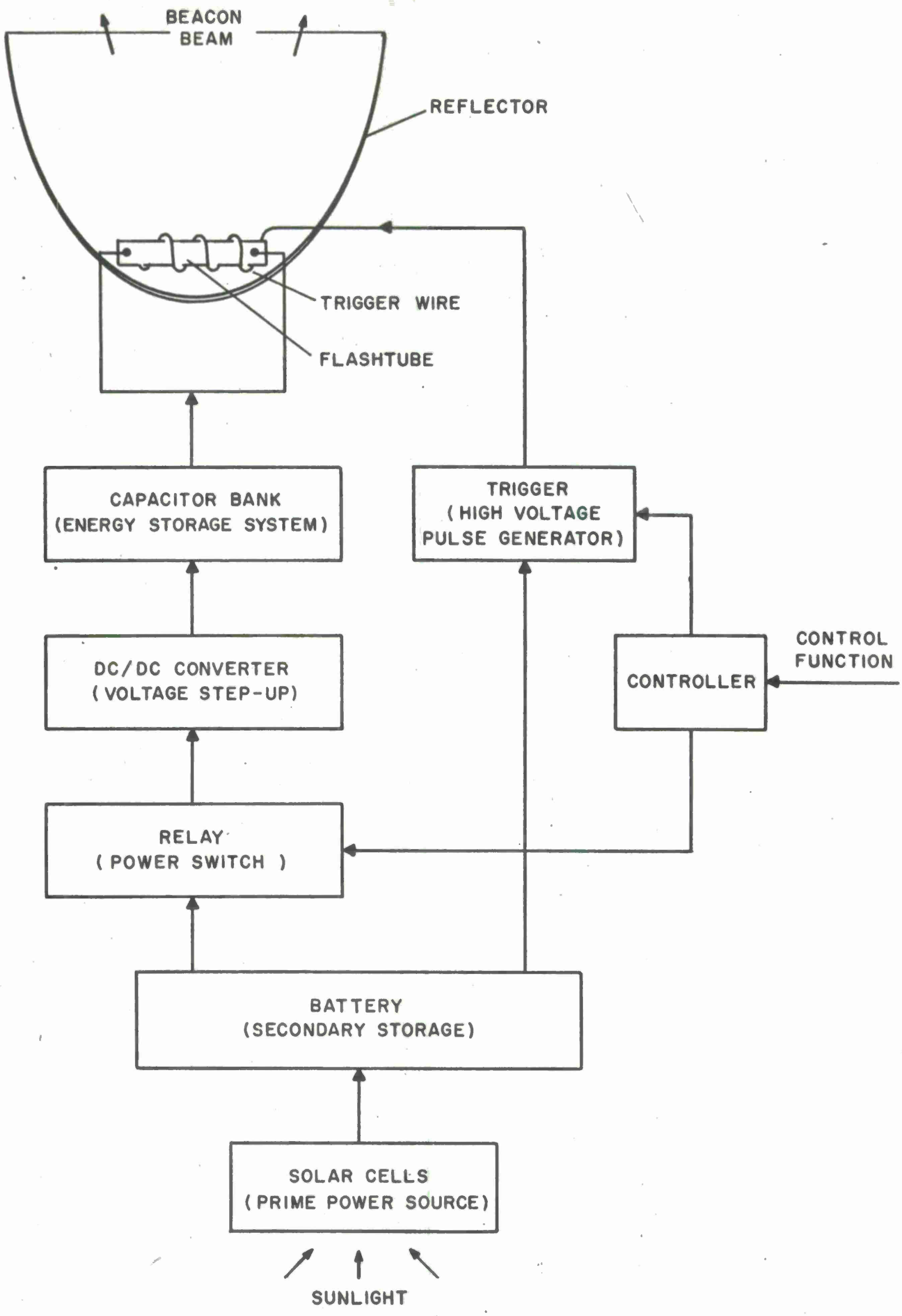


Figure 2 Principal Elements of a Satellite-Borne Xenon Flash System

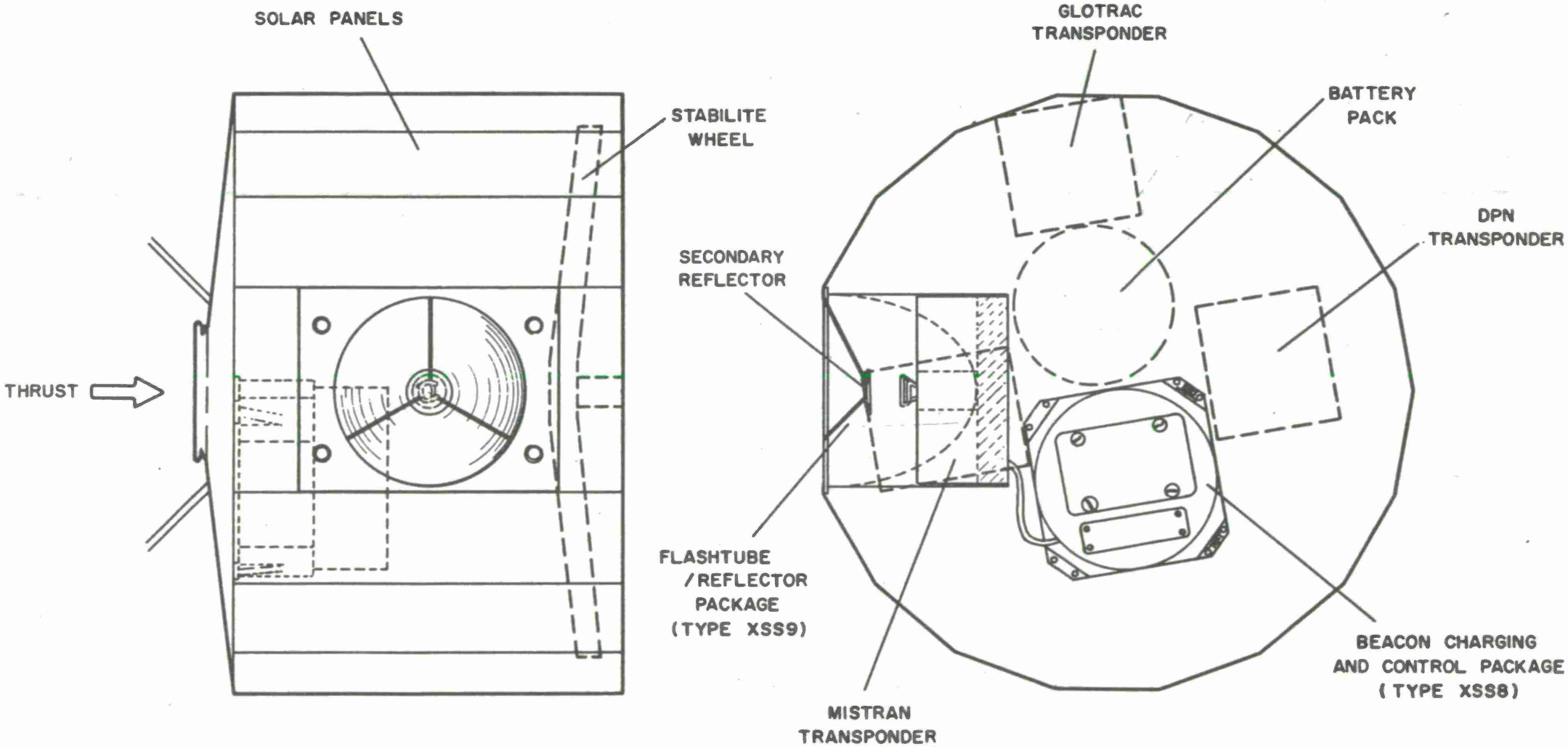


Figure 3 Cal-Sat Beacon System Components in Place in Satellite
EG&G Model XLS-51

SECTION 6 - SATELLITE INTEGRATION STUDY

6.1 Summary of Study

In this section, results are summarized for a preliminary design and integration study of a complete Range Calibration Satellite (RCS). The study was conducted by Radio Corporation of America (RCA) and full details are available in the final contractor study report (reference 6).

All significant aspects of the RCS design and utilization were considered in the study and are summarized in the following two sections of this TDR. The RCA study furnishes complete details on the following associated topics:

a. Spacecraft Configuration

b. Thermal Control

c. Attitude Stabilization Subsystem

d. Power Supply Subsystem

e. Command and Control Subsystem

f. Beacon and Telemetry Subsystem

g. Transponder Antenna Design

h. Passive - Tracking Aid

i. System Integration and Test

j. Environmental Test Program

6.2 Summary of Satellite Design

a. General

The design requirement was for a spacecraft which would provide a minimum 6-month operational period. Since the TIROS design appeared to be compatible with the RCS payload constraints, it was used as a basic pattern in the development of the RCS configuration. The external physical appearance and structural design of the Range Calibration Satellite are similar to the TIROS space; modifications having been made only where necessary to satisfy the mission requirements.

The spacecraft attitude stabilization system is an extension of the TIROS magnetic attitude control technique for spin-stabilized spacecraft. Since the range calibration mission requires that broad, smooth transponder antennae patterns be directed continuously toward the earth with low residual angular rate during actual calibration, a three-axis stabilization system is required. This stabilization is necessary to satisfy the stringent microwave phase stability tolerances on the MISTRAM coherent radar signal paths.

Command of the RCS spacecraft is accomplished by serial digital code transmission on the VHF command link. Such commands provide direct actuation of spacecraft control functions.

Optical beacon and radar transponder programmed operation timing is also received via this command link. This data is inserted into a spacecraft magnetic core memory and time-updated regularly by a high-precision digital clock to yield "turn-on" timing accuracies on the order of 1 millisecond over a 4-day time span. This high precision event-timing accuracy is a particular requirement for operation of the spacecraft optical beacon in conjunction with the ground PC 1000 ballistic camera, in order to relate the optical flash accurately to the star field background.

The overall Range Calibration Satellite system block diagram is shown in Figure 4. The RCS system is composed of the structure and the payload (transponders and optical beacon) and the following subsystems: attitude stabilization; command and control; beacon and telemetry; power supply; and transponder antennae.

b. Attitude Stabilization Subsystem

The design approach to RCS spacecraft stabilization involves a high-speed flywheel mounted within the confines of the spacecraft. This flywheel gives the RCS the properties of spin stabilization about the figure or symmetry axis without the necessity of spinning the entire craft. The axis of the wheel is maintained in alignment with the orbit normal by a space-proven

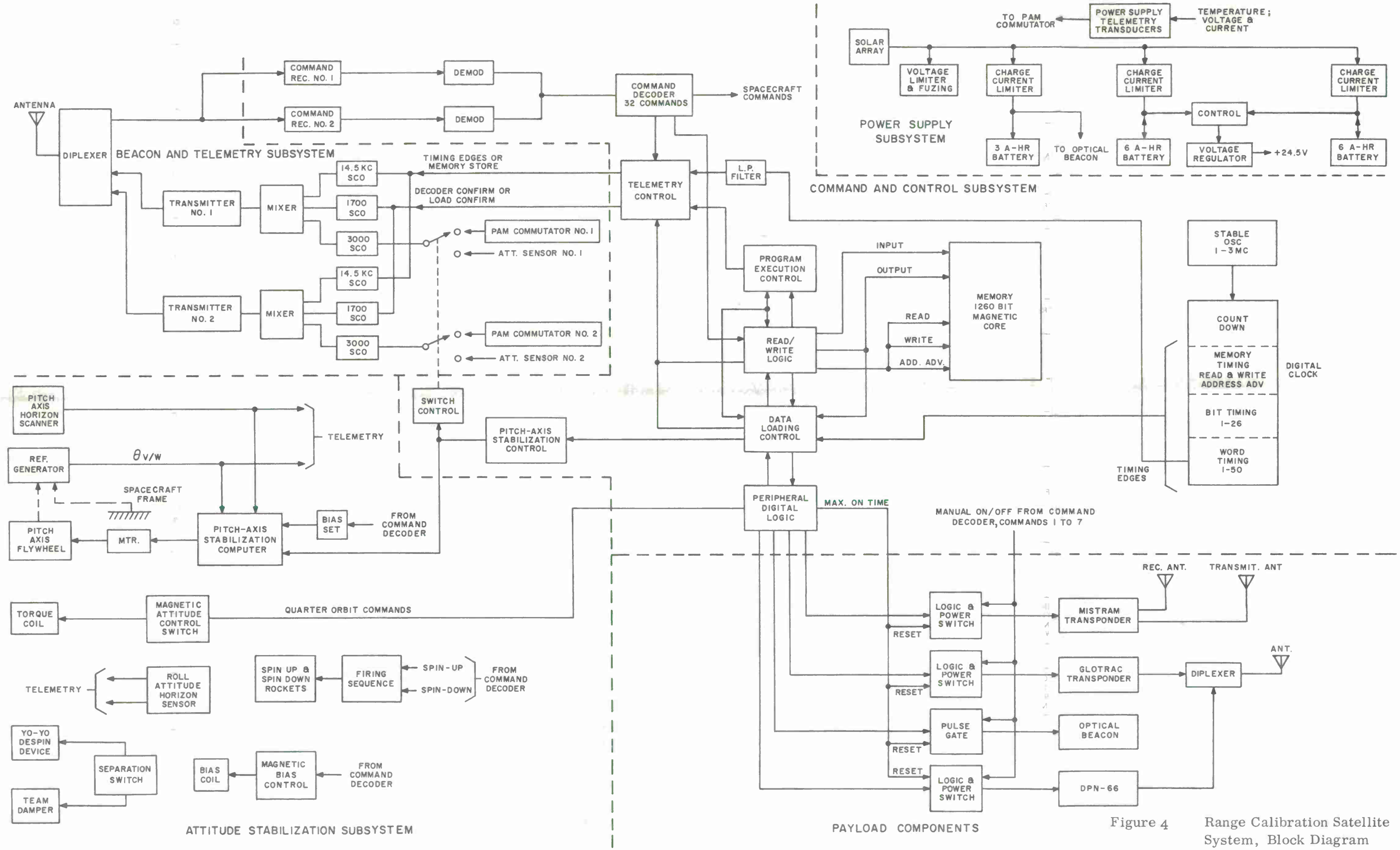


Figure 4 Range Calibration Satellite System, Block Diagram

magnetic (programmed coil current) attitude control system. The transponder antenna axes are normal to the flywheel axis and therefore "look" through one of the eighteen side panels of the spacecraft. Despin and referencing of the transponder system to local vertical is accomplished by servo controlled torque reaction to flywheel rotation. Despin and earth orientation is provided by memory command during the actual range-calibration phases of the mission only. At other times, the total structure is permitted to rotate at low velocity, thereby providing a cooling effect to the solar cells and a consequent advantage to the thermal and power supply designs.

c. Spacecraft Structure

The RCS spacecraft structure is similar to the TIROS structure with the following modifications:

1. One of the 18 side panels contains transponder antenna instead of solar cells.

2. The solar-cell "hat"(spacecraft height) has been increased by 4 inches to improve mission power capability.

3. A pitch-stabilization flywheel system has been added to the spacecraft. It is mounted in the top rib structure of the spacecraft solar-cell "hat".

d. Spacecraft Thermal Control

The spacecraft must accept a wide range of solar input and spacecraft internal power conditions while maintaining

the temperatures of the electronics, batteries, and solar cells within acceptable operating limits. An added complication is that during range calibration when the spacecraft is not spinning the temperature of the solar-cell panels which face the sun tends to rise.

A passive thermal design has been developed for the Range Calibration Satellite. This design, which specifies mylar thermal shields on the baseplate and special surface finishes, will enable the range calibration mission to be performed with ample thermal safety margins.

e. Power Supply Subsystem

The basic power supply components are the solar array, battery packs, charge-current limiters, under- and over-voltage protective devices, and a series voltage regulator.

The solar-cell array consists of 2344 shingles. Each shingle unit is composed of five 1- by 2-centimeter N-on-P type solar cells with a 0.006-inch fused-silica protective cover. Cell efficiency will be 9 percent at 25°C and air mass zero.

The power supply has three battery packs, each pack composed of 23 rectangular nickel-cadmium batteries. Two of the battery packs have a 6-ampere-hour capacity and are used in a redundant manner to insure power capability for the spacecraft

electronics and the radar transponders. The optical beacon is powered by a separate 3-ampere-hour battery pack to minimize conducted interference.

The payload voltage is +28 volts with a tolerance of +15, -10 percent. The voltage is supplied directly from the battery bus. The series regulator maintains the output to the spacecraft electronics subsystems at 24.5 $\pm$ 0.5 volt.

Even under the worst conditions of least-favorable sun angle and minimum sun-time occurring simultaneously, the RCS power supply will support a daily calibration mission in addition to normal spacecraft housekeeping, as indicated by the following summary. Under least favorable solar input conditions:

1. The MISTRAM transponder will operate 4 times per day for a total of 45 minutes per day.

2. The GLOTRAC transponder will operate 4 times per day for a total of 90 minutes.

3. The optical beacon will provide 30 sequences of seven flashes each per day.

4. The DPN-66 transponder will operate 6 times per day for a total of 180 minutes.

Other features of the design include individual protective devices against short circuit in any of the payload components and special

techniques to minimize conducted interference.

f. <u>Command and Control Subsystem</u>

The command and control subsystem consists of redundant command receivers and demodulators, a command decoder, the magnetic core memory with its peripheral control circuitry, and the precision oscillator with a digital timing chain (digital clock).

After reception and demodulation, 5 bits of the command word, which consists of 5 data bits plus an "execute" bit, are injected into a shift register in the command decoder. Simultaneous with injection into the shift register, the digital data is re-transmitted bit by bit to the ground station for verification. After verification, the sixth bit is transmitted from the ground for the execute command. The command system provides 32 commands for spacecraft control and manual over-ride command of the transponder systems. In addition to the basic command-bit-pulse format, wider pulse signals may be sent through the receiver to actuate special functions.

The spacecraft digital clock, which performs the memory updating, is monitored from the ground via the telemetry link so that accurate knowledge of digital clock time to the order of 0.1 millisecond always exists at the ground station.

g. Beacon and Telemetry Subsystem

The Range Calibration Satellite will have a redundant telemetry system; each of the two sections will consist of three subcarrier oscillators multiplexed on its own transmitter. This FM/FM telemetry system will provide command and storage confirm signals, timing edges, a PAM commutated train of spacecraft and payload performance measurement data and attitude sensor data. The subsystem will be developed using space-proven components.

h. Transponder Antenna Subsystem

The transponder antennas have been designed so that increased gain compensates for greater range at low elevation angles. In this manner, power input to the transponder receivers is relatively independent of range. The polarization, VSWR, and isolation that are required by the MISTRAM, GLOTRAC, and DPN-66 transponders have been included in the design.

A breakdown of the Range Calibration Satellite weight is shown in Table I.

6.3 System Utilization

a. Ground Coverage and Orbit Geometry

Potential users of the Range Calibration Satellite includes FPS-16 and FPQ-6 radars and their mobile versions,

TABLE I BREAKDOWN OF RCS WEIGHT

Group	Weight (pounds)	Group	Weight (pounds)
Structure		Power Supply	
Baseplate with Separating Ring	23.00	Battery Pack	12.40
Hat Structure	43.50	Battery Pack	18.30
Subtotal (22.7 percent of gross)	66.50	Battery Pack	18.30
Stability & Control		Voltage Regulator	
Stabilite and Flywheel	12.60	Telemetry Conditioner	
Despin Unit	1.30	Charge Limiter	10.00
Despin Unit	1.30	Under Voltage Protector	
Spin-up Rocket	1.00	Overload cutout	
Rocket Switch	0.43	Cable and Harness	10.00
Damper	3.00	Solar Cells	39.50
Attitude Coil	2.50	Subtotal (37.4 percent of gross)	108.50
Attitude Coil Switch	0.68	Mission Equipment	
Preamplifier	1.00	Optical	
Horizon Scanner (V-unit)	1.00	Optical Parabolic Source	5.00
Subtotal (8.5 percent of gross)	24.81	Converter and Capacitor Bank	27.00
Communication & Telemetry		Trigger Control	0.70
Transmitter	0.50	Power Relay	0.65
SCO & Mixer Amplifier	1.20	Sequence Control Unit	4.00
Transmitter	0.50	Radar	
SCO & Mixer Amplifier	1.20	MISTRAM Transponder	12.00
Command Receiver		MISTRAM Antenna Trans.	0.40
Demodulator		MISTRAM Antenna Recvr.	0.40
Command Receiver	3.00	GLOTRAC Transponder	5.70
Demodulator		DPN 66 Transponder	10.80
Diplexer Plate	2.10	GLOTRAC & DPN 66 Antenna	0.60
Telemetry Antenna Inst.	1.60	Subtotal (23.0 percent gross)	67.25
Memory Control and Decoder	7.50	GROSS WEIGHT	291.56
Memory	4.00		
Commutator	1.10		
Commutator	1.10		
Stable Oscillator	0.70		
Subtotal (8.4 percent of gross)	24.50		

MPS-25 and TPQ-18 radars; MISTRAM and GLOTRAC radars; and ballistic cameras of the Atlantic Missile Range (AMR) and various other agencies.

The nominal orbit is circular, 400-nm altitude, and inclined at an angle of 40 degrees to the equator. This orbit provides the following advantages:

1. It is high enough to make atmospheric drag negligible over short arc lengths, thus minimizing uncertainties in orbital prediction.

2. It is low enough to provide good trilateration and triangulation geometry for the ground instrumentation.

3. It is low enough to reduce transponder, flasher, and communications power requirements.

4. The 40-degree inclination provides almost optimum coverage of most of the important range instrumentation.

5. The circular orbit minimizes booster requirements while keeping variations in range to the satellite as low as possible.

b. Satellite Utilization

An orbit has been selected that provides excellent observation time for the ground tracking sites that will utilize the RCS. It was necessary to determine the practical mission utilization consistent with spacecraft power constraints. The

main energy users aboard the satellite are the three radar transponders (GLOTRAC, MISTRAM, and DPN-66) and the optical flasher. The criteria for power usage are as follows:

1. Energy balance (batteries fully charged) is to exist at least once per day for the worst case of minimum suntime, maximum usage, and poorest sun angle.

2. Maximum DPN-66 usage is to be 30 minutes per orbit.

3. MISTRAM and GLOTRAC transponders may be exercised when observation conditions permit.

4. Sun angle (angle between the spin axis and the rays of the sun) will be kept between 26 and 80 degrees. In this range 80 degrees is the worst sun angle and 30 degrees the best.

5. Maximum flasher usage is to be 30 sequences of 7 flashes each per day, or 210 flashes in all.

c. Operational Modes

Two separate operational modes of the RCS are envisioned. These are the intervisible mode and the short-arc mode. In intervisible operation, the satellite is observed simultaneously by the calibration standard (1000-millimeter-focal-length ballistic cameras) and uncalibrated range instrumentation (GLOTRAC, MISTRAM, and the DPN-66 transponder radars).

The position of the satellite as determined by the uncalibrated system is compared to that obtained by the ballistic cameras, and the difference is used to remove bias errors consistent with statistical knowledge of random and systematic error magnitudes.

The short-arc mode is used to calibrate electronic instrumentation on the basis of short-arc orbit estimates of satellite position. As the satellite passes over the instruments to be calibrated, a best estimate is made of the orbital elements. Measurements by the uncalibrated instrumentation are compared to the predicted position and calibration is deduced for the electronic systems using the optical systems as a standard. The intervisible mode is used to perform the range surveys to eliminate the geodetic errors prior to the calibration (see again Figure 1).

d. Orbit Perturbations

Assuming as a model spheroidal Earth surrounded by a complete vacuum out to infinity in all directions, the significant perturbations of a satellite in orbit around the real Earth are as follows:

1. Higher harmonics of the field of the Earth; namely, the tesseral harmonics and the zonal harmonics of order greater than two.

2. Atmospheric drag.

3. Ionic and magnetic drag.

4. Lunar gravitation.

5. Solar gravitation.

6. Solar radiation pressure.

It is known that for a 400-nm orbit, atmospheric drag can be neglected when making orbit predictions on arc lengths with time periods of up to 16 minutes. As a consequence, the only perturbations which must be considered are the higher harmonics of the gravitational field of the Earth.

SECTION 7 - OVERALL CONCLUSIONS AND RECOMMENDATIONS

7.1 Conclusions

As mentioned in Section 1.2, extensive feasibility and trade-off study conducted under the ESD calibration task (reference 1) has shown that a calibration of precision trajectory measurement systems can be accomplished. It has been further established that the most accurate technique for achieving this goal is through the use of an appropriately-designed calibration satellite. The results of the five detailed design specification studies summarized in this TDR allow one to conclude that the design, development, and utilization of such a calibration satellite is well within the present state-of-the-art. Although some problems have been pinpointed for additional study, no really significant difficulties should arise in developing a calibration satellite.

The results of the MISTRAM, GLOTRAC, and C-band transponder studies summarized in Sections 2, 3, and 4, respectively, of this TDR have served to provide the basis for complete Technical Exhibits for these transponders (reference 7). On the basis of this study effort, and with the Technical Exhibits on hand, development

of satellite-rated transponders can be initiated. Further, the RCA satellite integration study summarized in Section 6 provides the necessary background for implementation of a complete system development.

7.2 Recommendations

It is recognized that initiation of a full-scale calibration satellite program, which would result in a complete calibration system (reference 8), may not be appropriate or possible at this time. In lieu of such a full-scale, multi-satellite, program, a smaller-scale program has been proposed (reference 9). A very austere calibration satellite "package" could be easily designed for piggyback or orbital pod (General Dynamics/Astronautics SATAR) launch. It could be designed such that nearly any required transponder could be integrated into the basic package with minimum development time and money. The package would carry the appropriate electronic transponder (or transponders), an optical beacon for use as a standard with a ballistic camera network, and all necessary support equipment such as telemetry, timing, stabilization, etc.

A prototype calibration satellite package as proposed above could probably be developed for a million dollars or less, depending

upon the level of required transponder development. It could be available in approximately 18 months from program initiation, again depending on long lead time transponder developments. Such a satellite package is the logical first step which should be taken. It would provide the capability for achieving greater accuracy in trajectory measurement systems.

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