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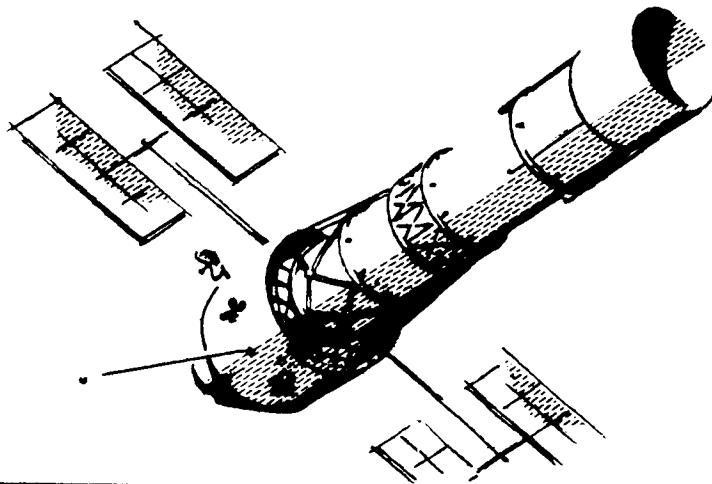
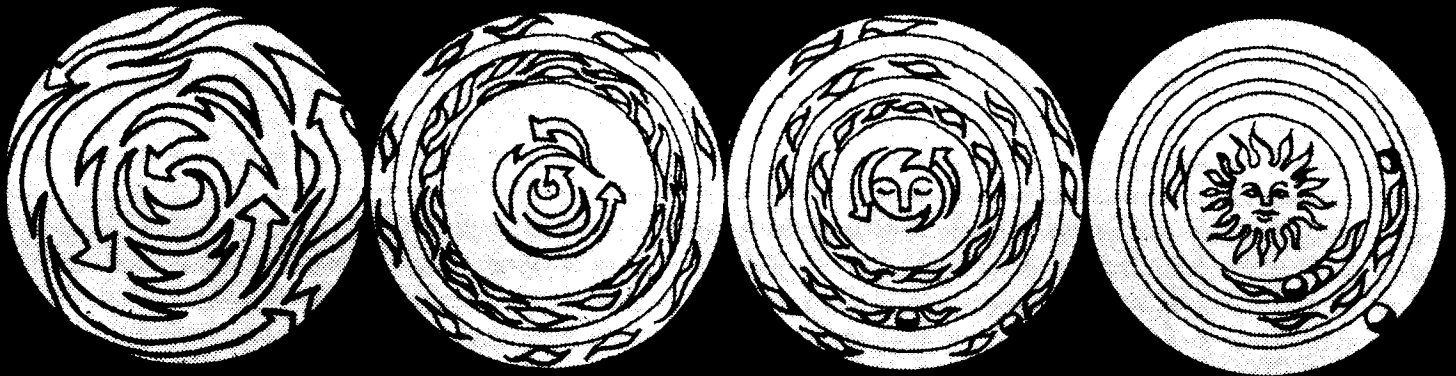
OPTICAL TECHNOLOGY
APOLLO EXTENSION SYSTEM
PHASE A

VOL III

FINAL TECHNICAL REPORT
NAS 8-20256

EXPERIMENTS

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PREFACE

The final report of the Optical Technology Apollo Extension System study prepared for NASA/Marshall Space Flight Center under Contract Number NAS8-20256 is presented in six volumes. The study was a team effort by Chrysler Corporation Space Division (CCSD) (prime contractor), Kollsman Instrument Corporation (KIC) and Sylvania Electronic Systems (SES).

Volume 1, containing the Program Results (1.0, 2.0, 3.0 and 4.0) and the Optical Technology Development Plan (5.0, 6.0, 7.0 and 8.0) was the responsibility of CCSD.

Volumes 2 and 3 contain the Recommended Experiments (9.0 through 11.0), Integrated Experiment Requirements (12.0) and Other Experiments (13.0).

The SES experiments are:

- 9.6.1 Optical Heterodyne Detection on Earth
- 9.6.2 Optical Heterodyne Detection in Space
- 9.6.3 Direct Detection on Earth
- 9.6.4 Megahertz Optical Communication
- 9.6.8 10 Micron Phase and Amplitude Correlation
- 9.6.9 Pulse Distortion Measurements

The KIC experiments are:

- 9.6.5 Precision Tracking of a Ground Beacon
- 9.6.6 Point Ahead and Space-to-Ground-to-Space Loop Closure
- 9.6.7 Precision Tracking from One Ground Station to Another
- 10.2.1 Fine Guidance
- 11.4.1 Thin Mirror Nesting Principle
- 11.4.2 Primary Mirror Figure Test and Correction

The CCSD experiment is:

- 10.2.2 Comparison of Isolation Techniques

Integrated Experiment Requirements (12.0) and Other Experiments (13.0) were the responsibility of CCSD except for Residual Atmosphere Scattering Experiments (13.2.4) which was the responsibility of SES and Astronomical Uses of the OTAES Fine Guidance Telescope (13.3.2) which was the responsibility of KIC.

Volume 4 contains Systems Integration. CCSD prepared Candidate Missions (14.0), Manned Operations (15.0), Mission Analyses (16.0), Baseline Space Environment (17.0), Reliability (18.0) and System Requirements (20.0). SES prepared Ground Stations (19.0).

Volume 5 containing Subsystem Design was the responsibility of CCSD except for the Data Management Subsection which was prepared by SES and the Fine Guidance Thermal Analysis (27.5 and 27.6) which was prepared by KIC. This volume includes Design Integration (21.0), Structural and Mechanical Subsystem Design (22.0), Guidance, Navigation and Control (23.0), Propulsion and Reaction Control (24.0), Electrical Power (25.0), Environmental Control (26.0), Thermal Control (27.0), Data Management (28.0), Weight and Mass Properties (29.0) and Crew Equipment (30.0).

Volume 6 containing the Resource Analysis (31.0, 32.0 and 33.0) was the responsibility of CCSD.

MASTER

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10.0 STELLAR-ORIENTED EXPERIMENTS

The purpose of the stellar-oriented experiments is to develop telescope stabilization technology in support of future high resolution astronomical applications. To fully realize the benefits of space astronomy, the telescopes must be precisely stabilized over long exposure periods. Problems with achieving this long term stability for a manned spacecraft are logically divided into two major groups:

- a. Disturbances, originating in the manned vehicle and transferred to the telescope, which would prevent the required stability being achieved with any practical stabilization hardware.
- b. Inherent limitations in the stabilization hardware that would prevent the required stability being achieved even with a disturbance-free telescope.

The first experiment in this group, the Fine Guidance experiment, is concerned specifically with hardware limitations. Its explicit objective is to develop new stabilization hardware which, inherently, will allow the 0.01 arc second stability to be achieved. The other experiment in this group, the Isolation Systems Comparison experiment, is concerned specifically with isolating the telescope from the spacecraft, and its specific objective is to develop the simplest isolation technique which will remove spacecraft-produced disturbances as an obstacle to achieving 0.01 arc second stability.

10.1 JUSTIFICATION

10.1.1 Contribution and Need

The two experiments in this group, the Fine Guidance experiment and the Isolation Systems Comparison experiment, taken together are designed to advance the orbital telescope pointing technology at least one order of magnitude beyond presently known capability. This advancement in pointing technology is essential if the requirements of the future large, diffraction-limited orbital telescope are to be met.⁽¹⁾⁽²⁾⁽³⁾ The projected technology advance is illustrated in figure 10.1.1-1, where the existing pointing capability for a +2 magnitude star is given as 0.1 arc second (fine guidance specification for the Princeton Experiment Package for the Orbiting Astronomical

-
- (1) Space Research - Directions for the Future, Part Two, Space Science Board, National Academy of Sciences, National Research Council, Washington, D.C. January, 1966.
 - (2) Feasibility Study of a 120-inch Orbiting Astronomical Telescope, J. W. Fecker Division, American Optical Company, Pittsburgh, Pa. Under Contract NAS 1-1305-18, 1963 to NASA, Langley.
 - (3) A System Study of a Manned Orbital Telescope - The Boeing Company, Seattle, Wash., Under Contract NAS 1-3968, prepared for NASA-Langley Research Center, Oct. 1965.

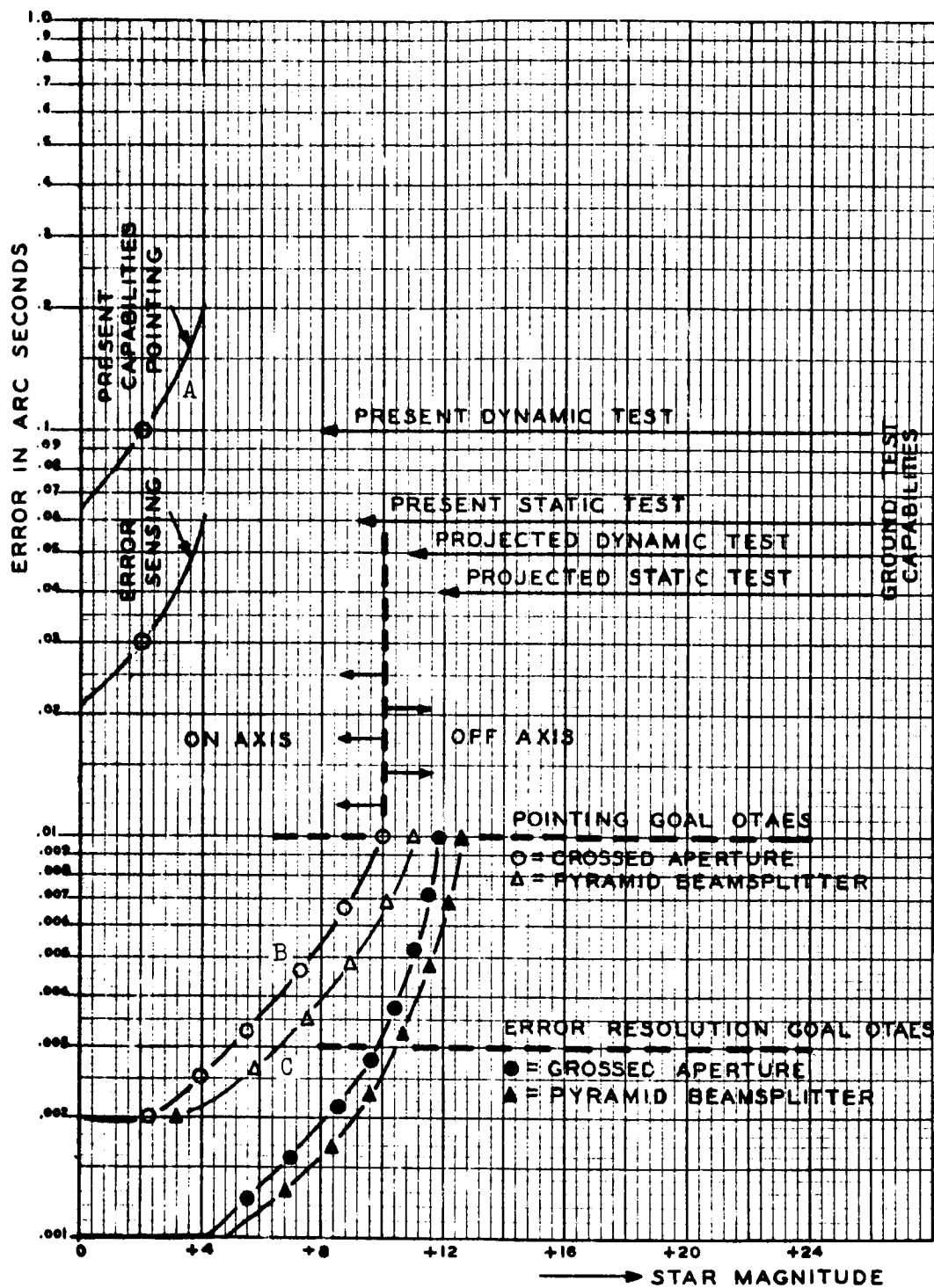


Figure 10.1.1-1 Fine Guidance Error vs Star Magnitude

Observatory). Even this pointing stability has not been certified in ground based tests. It has been demonstrated in the laboratory, however, that 1 arc second stability can be achieved (Goddard Experiment Package for OAO-B built by Kollsman. An improved version designed for 0.02 arc second noise equivalent error will be tested soon.)

In contrast to these existing systems, the OTAES Fine Guidance experiment proposed here is designed for 0.002 arc second stability for a +2 magnitude star and 0.01 arc second for a +10 magnitude star, both for on-axis pointing as shown in figure 10.1.1-1, curves B and C for two sensor types.

The contribution of the Isolation Systems Comparison experiment to achieving this goal of 0.01 arc second stability is illustrated in figure 10.1.1-2, where the dependence of image stability on disturbance torques and the undamped natural frequency of the isolation system is shown. The torques are RMS values derived for three different levels of astronaut activity. The figure clearly illustrates the need for isolation and the dependence of image stability on the natural frequency of the isolation system.

10.1.2 Need for Space Testing

To achieve the factor of five or greater improvement in pointing technology, as proposed for OTAES and illustrated in figure 10.1.1-1, clearly requires a large experimental development effort. Much of that development will be done on the ground. However, it is maintained here that certain vital design information can be obtained only by experiments conducted in space under actual operating conditions. This conclusion is based on two major points. First, the required pointing stability of 0.01 arc second cannot be achieved in any ground based facility presently existing, or any reasonable facility projected for the future. Therefore, either pointing experiments must be conducted in space, or the future orbital telescope, designed for diffraction-limited performance, must be designed and deployed without the performance of a key subsystem, the stabilization system, having been certified. Second, the isolation system, which allows the precision stabilization to be achieved in space, cannot be satisfactorily tested on the ground because this system will not operate in a one-g environment. Thus, it is concluded that both the isolation and stabilization system should be tested in space. It should be pointed out, however, that the two experiments proposed here involve more than simply testing a particular design to verify that the required stabilization can be achieved. The experiments are configured and planned so as to provide the maximum design information.

To this end, several alternate design and combinations of designs will be tested under a wide variety of disturbance conditions. It is expected that for certain of the designs tested, and conditions imposed, the required pointing stability will not be achieved. Indeed, it may not be achieved at all; but failure in this sense will not mean failure of the experiments. The detailed technical reasons for the failure will be known, and this information will provide for the design of future telescope stabilization and isolation systems that will not fail.

The conclusion that 0.01 arc second pointing stability cannot be achieved in a ground-based test facility is based on a survey of such facilities. The facility requirements would include both the capability to conduct the stability experiments in a simulated space environment and a standard reference system to independently evaluate the experimental results. The reference requirements, then, would be the most stringent, and for an 0.01 arc second experiment should be better than 5 milliseconds. The best capabilities of existing facilities, on the other hand, vary around one arc second with the best claimed stability being 0.1 arc second. (4)(5)

To isolate the test platform requires a large seismic mass which is itself isolated through a suspension system for seismic vibrations. One of the largest and most costly of such facilities is at Patrick A.F.B., Florida. Consisting of a seismic block 98 feet long by 7 feet wide supported by 37 air cushion mounts, the installation is designed for extremely low frequency isolation. However, tests show that little isolation is afforded for frequencies below 2 Hertz, and the disturbances are actually amplified below 1 Hertz. (6) These dynamic test facilities will continue to be improved but without some major breakthrough, the required reference stability is not likely to be achieved during the next few years. At any rate, it would seem unwise to base future telescope stabilization technology on such an eventuality; hence, the herein proposed space experiments.

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- (4) Test Report Stabilization and Control Subsystem Orbiting Astronomical Observatory, Flight B, Grumman Aircraft Engineering Corporation, No. AV252R-189.0, Vol. I, May 14, 1965.
 - (5) Private Communications from W. H. Zimmerma, Boeing Aircraft Co., dated 21 October 1966.
 - (6) Vibration Isolation Test at Equipment Test Laboratory Facility, Patrick A.F.B., Florida by K. Tsutsumic and F. S. Merenda, Geonetics Corp., 29 December 1965.

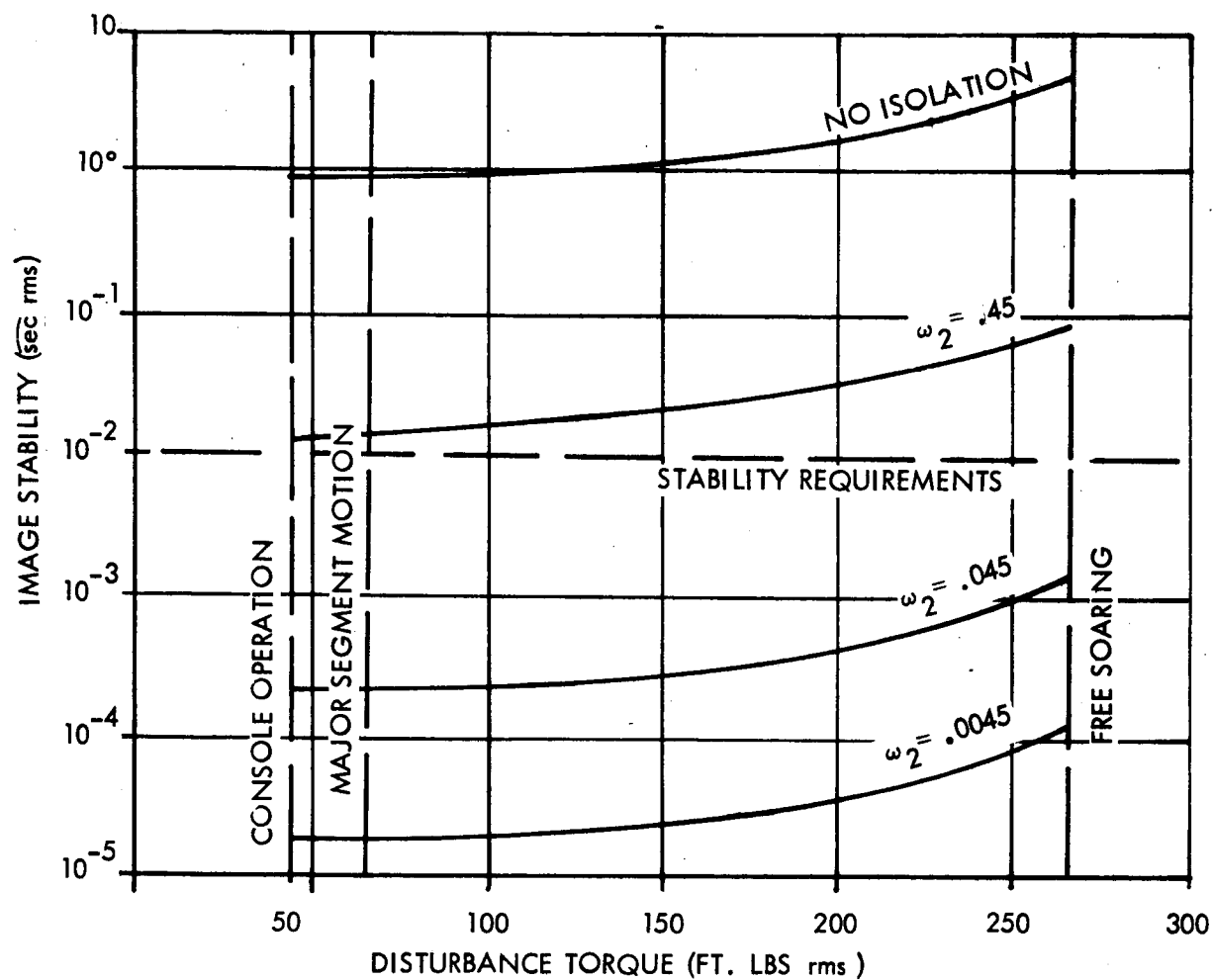


Figure 10.1.1-2. Image Stability Dependency on Disturbance Torque and Isolation Frequency.

10.2 EXPERIMENTS

10.2.1 Fine Guidance Experiment

A prime goal of the OTAES program is to develop the technology required to place a 120" diameter telescope of diffraction limited capability in space for observation of celestial bodies. Such a telescope would have a resolving power of 0.04 arc second in the visible portion of the spectrum and, unlike earthbound telescopes, would not be resolution-limited by atmospheric aberrations, which are generally at best in the order of 2 to 3 arc seconds. But to realize the capability of the telescope it must have a pointing stability which is a fraction of its resolving power. The goal of the OTAES fine guidance experiment has been set at 0.01 arc second pointing stability.

Lengthy time integration of received energy will be necessary to study such sources as Quasars (+16 to +20 mag.) and it will be necessary to hold the image stable for several hours within the pointing stability limit of 0.01 arc second. Calculations based upon present day capabilities of detectors indicate that the target star can be used as a guide star directly, down to the 12th magnitude. For stars fainter than this, offset tracking would be required.

The fine guidance system of the 120" telescope will operate over the field of the telescope but will siphon off a minor fraction of the radiant energy for direct tracking purposes. The optics of the OTAES fine guidance telescope approximately simulates this minor fraction. For offset tracking, however, the full energy gathered by the telescope from the guide star will be available.

The completely self contained attitude control system includes three control loop levels of increasing sensitivity: coarse, intermediate and fine. To acquire the guide star the telescope optical axis is initially pointed by the spacecraft along the line of sight to a selected target star with a ± 1 degree accuracy. Two 2-axis star trackers are programmed to acquire and track two selected bright stars. The errors detected by the two star trackers are transformed into coarse telescope pointing errors in pitch and yaw; and the telescope attitude is controlled until the intermediate error sensor acquires the fine pointing target star and takes over the telescope pointing correction.

The pointing capability of the telescope in the intermediate mode will be ± 0.5 second of arc, which is within the ± 1.0 second of arc field of view of the fine error sensor. Fine pointing will be obtained using either of two beam deflection elements inside the telescope. Based on signal-to-noise and servo analyses, this fine pointing correction is expected to provide pointing stability in the order of 0.01 second of arc.

Two candidate fine error sensors will be compared in performance by simultaneous operation - one a crossed aperture scanner consisting of two apertures driven in quadrature at dissimilar frequencies, and the other a pyramid prism which splits the incoming radiation into four mutually perpendicular components. The performance of two fine beam deflection types is also to be compared. One of these is a diasporameter, an actuator of refractive type comprising two optical wedges rotated relative to each other. The other actuator is of reflective type: a cantilevered mirror driven by magnetic coils surrounding the supporting shaft. The optimum of the four possible combinations of error sensor and fine actuator is to be identified.

Image stabilizing capability is to be studied as a function of image properties. Such properties include the magnitude of the guide star and its color temperature as well as the background noise in the star field. The effect of spacecraft disturbance level on image stability will also be investigated.

10.2.1.1 Experiment Design

The fine guidance experiment has been devised to assess the relative merits of selected error sensors and beam pointing actuators as elements in a long term image stabilizing loop; to provide a means of stabilizing the telescope while evaluating the various suspensions in the comparison of isolation techniques experiment; and to investigate the telescope pointing stability achievable as a function of the following: guide star brightness and color temperature, background luminance level, thermal condition of the telescope, mechanical torque disturbances, and position of bright luminous sources such as the sun, earth, and moon relative to the telescope line of sight.

10.2.1.1.1 Acquisition of Celestial Target

After the necessary checkouts have been accomplished and the spacecraft pointed to the proper heading for the target star line-of-sight to appear in the center of the fine guidance telescope gimbal range, the offset coordinates for the reference stars are inserted into the two astro trackers mounted on the telescope tube. They, in turn, acquire and track the reference stars, and the difference between the actual and commanded astro tracker gimbal angles generate error signals for the coarse acquisition loop. When these error signals are near null the telescope is uncaged and the signals are fed to the control moment gyros (CMG's) mounted on the telescope. The CMG's now null out the error signals by rotation of the telescope. At null the target star is in the field of view of the intermediate error sensor.

At this point the CMG input is transferred from the astro tracker gimbal difference to the intermediate error signal on which it proceeds to null. At the appropriate time one of the soft suspensions is uncaged and the CMG's can continue to null out the intermediate sensor error until the target star appears in the field of the fine error sensor. From this point on the CMG's continue to maintain a null on the intermediate error sensor while the fine error sensor in use drives one of the fine actuators as a vernier to hold the target star in the center of its field.

10.2.1.1.2 Operation of Experiment

All four error sensor-actuator combinations are exercised and compared against one another in order to select the most suitable. The selected combination is used in the balance of the experiment.

After the selection of actuator and sensor has been made, and the suspension selection as well, the main part of the experiment is conducted. This includes making a series of tracking stability measurements on selected stars. The

stars are chosen to provide a controlled range of magnitudes, spectral classes, background conditions and geometrical relationships to the sun, earth, and moon. During the tracking, temperatures at critical points are monitored and a series of programmed disturbances is introduced into the system for evaluation of stability and recovery capability.

10.2.1.2 Experiment Measurement Objectives

By measuring and recording the error signals in pitch and yaw of both fine error sensors simultaneously it is possible to determine their relative merits in sensitivity, error resolution and geometrical fidelity for the purpose of selecting the preferred configuration. As it is not practical to attempt simultaneous operation of the two actuators, because one would cloud the effects of the other, these are measured by sequential tests in which similar conditions are maintained. Recorded error signals from the coarse and intermediate sensors as well as the fine sensors will provide a time history from which can be derived the characteristics of the stabilization loops. Parameters to be measured include star magnitude and color, background noise level, spacecraft and telescope environment, the switching of loops from coarse to intermediate to fine, recovery from perturbations including those involving temporary signal loss, and the effectiveness of the suspensions being evaluated in an associated experiment.

10.2.1.3 Experiment Equipment Design

10.2.1.3.1 Telescope General Design and Primary Optics

The fine guidance telescope is a long focal length Cassegrainian telescope with outboard mounted astrotrackers for acquisition assistance, control moment gyros for telescope pointing, an instrumentation package behind the primary mirror for optical processing and an experimental suspension for isolation from the spacecraft proper. The arrangement is shown in the schematic diagram Figure 10.2.1-1.

Tracking requirements set the focal length at about 72 meters for the fine guidance sensors and 1.2 meters for the intermediate sensor. This is accomplished by means of a Cassegrainian telescope of 12 meters effective focal length followed by image relays of 6 power and 0.1 power magnification respectively. The objective consists of a parabolic beryllium primary mirror of 4 meters focal length and 0.75 meter aperture, and a quartz hyperboloidal secondary mirror with a focal length of -1.575 meters and an aperture of 0.2 meter.

The two mirrors are separated by an inter-element distance of 2.95 meters. The image formed by the Cassegrainian telescope is brought to a focus 200 millimeters behind the primary mirror where it is intercepted by a field lens of 12 millimeter clear aperture and a focal length of 168 millimeters. The features of the telescope are illustrated in detail in Figure 10.2.1-2.

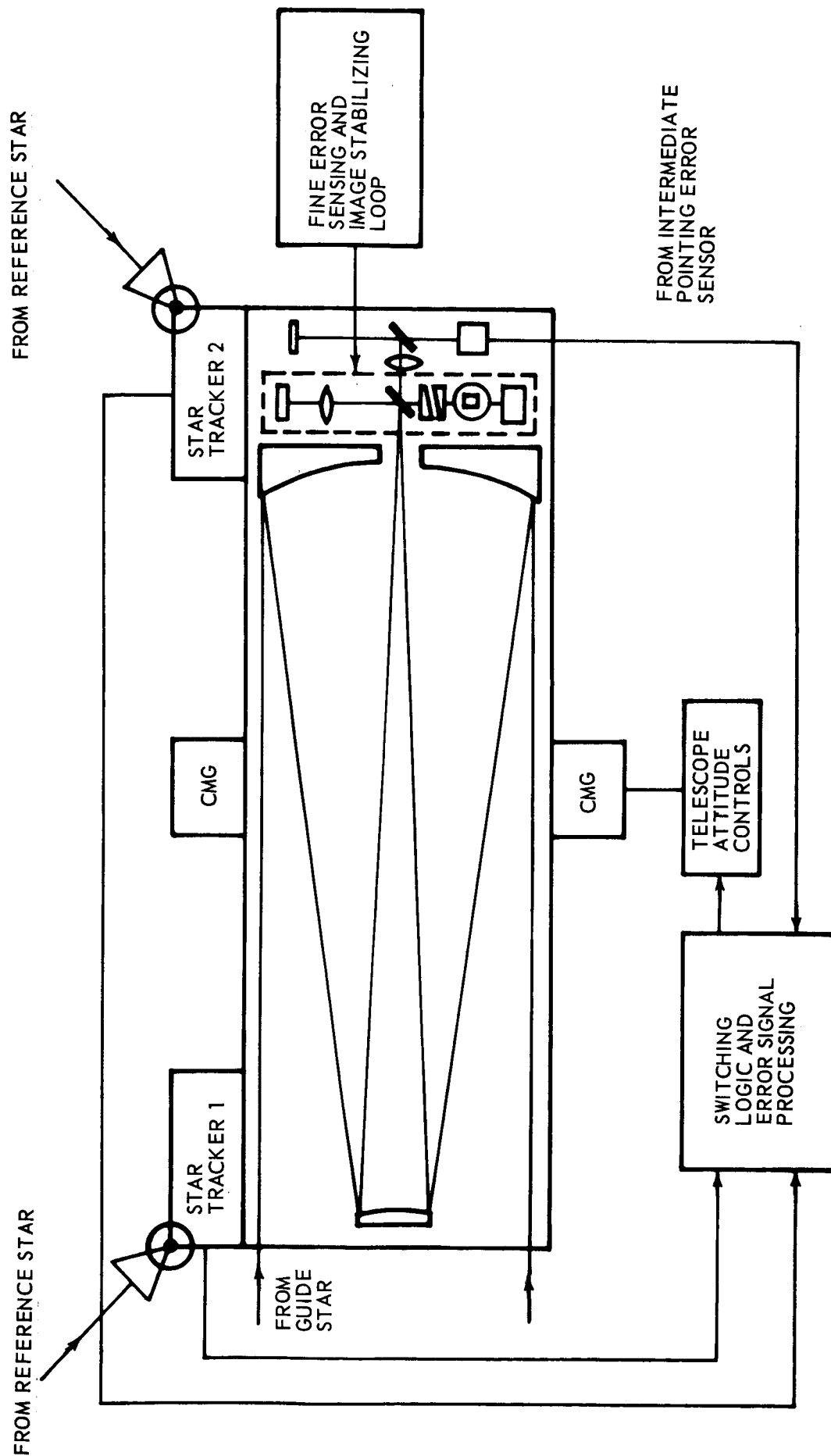
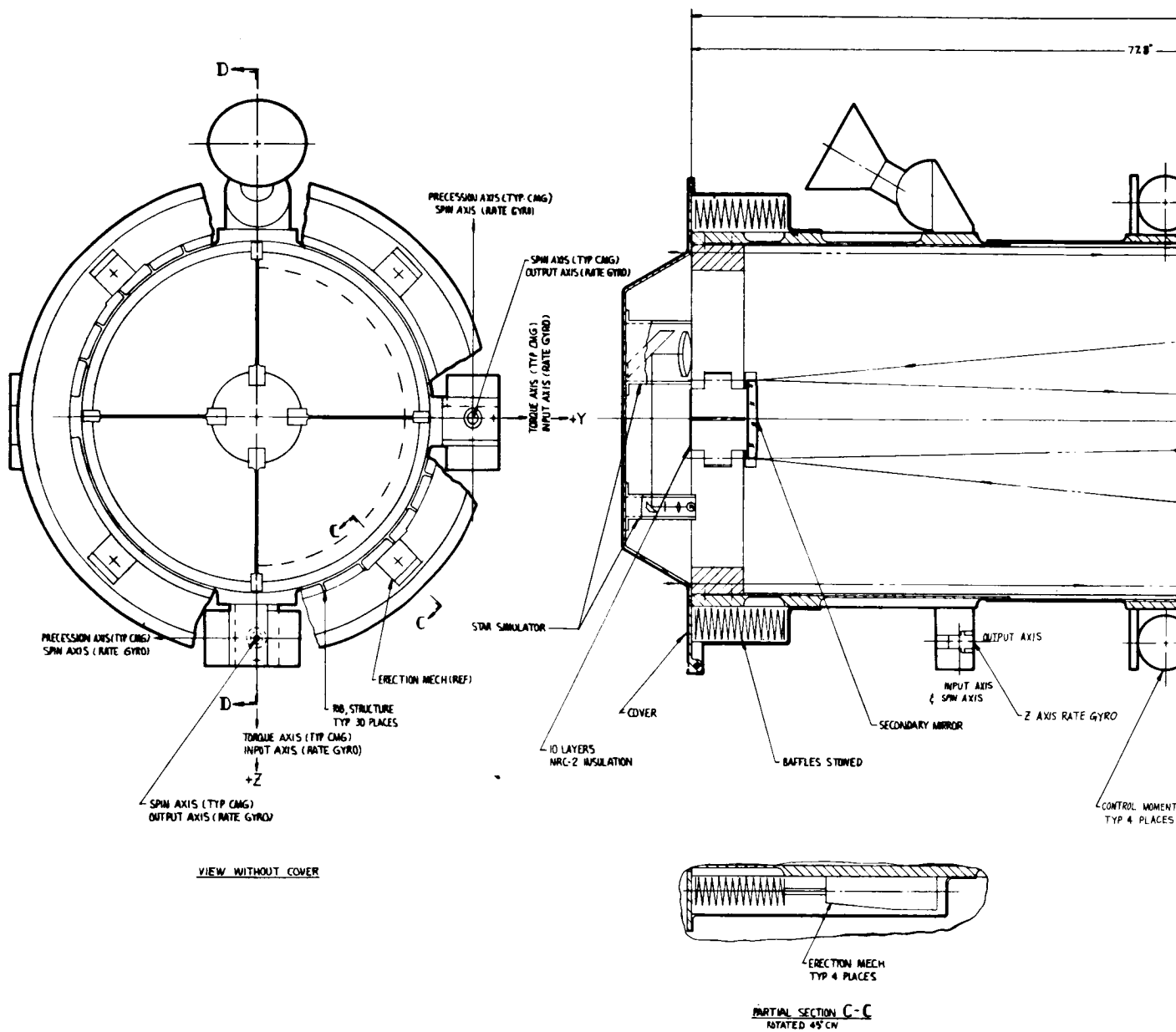


Figure 10.2.1.1-1. Fine Guidance Telescope Functional Schematic



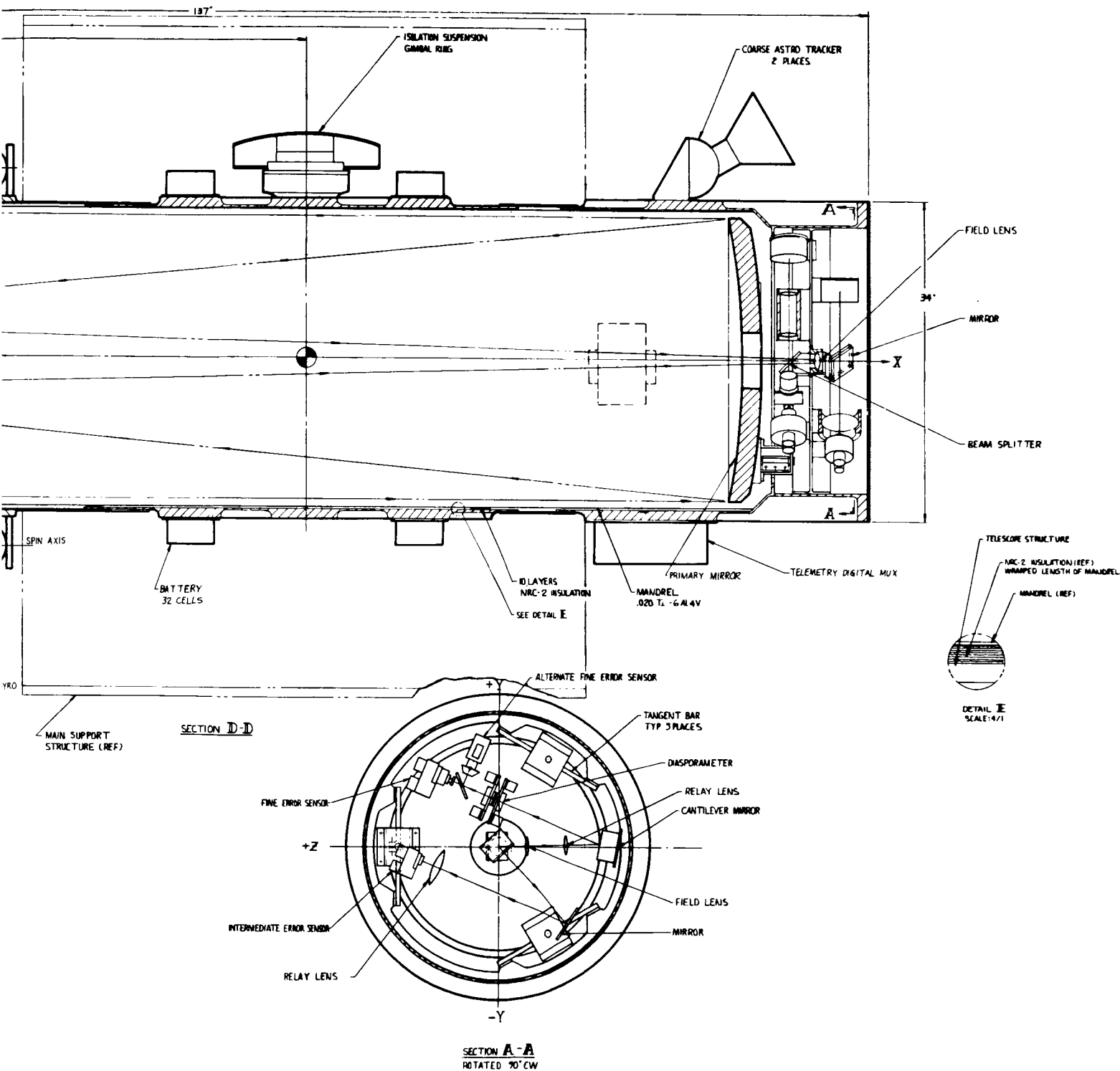


Figure 10.2.1-2. Fine Guidance Telescope.

Common primary optics are used for both the intermediate and fine error sensors, separating them only in the processing optics relays. This feature keeps the lines of sight precisely boresighted in the critical portions where a temperature gradient could bow the telescope. Therefore, the pointing of the telescope need only be accurate to the field of view of the intermediate error sensor, ± 1 min. Taking into account the 3x magnification of the secondary, the maximum allowable bowing of the telescope is in the order of 20 sec. in any direction. The maintenance of boresight to the accuracy of the fine error sensor need only be maintained in the compact pancake section devoted to the processing optics. Even here errors can be calibrated out by use of the diasporameter actuator in conjunction with the star simulator.

The processing optics include two paths, (1) a fine guidance path comprising a beam splitter, a field stop, a relay lens, an electromagnetically driven mirror, a diasporameter, a second beam splitter and two fine error sensor packages, and (2) an intermediate guidance path including two folding mirrors, a relay lens and an error sensing package.

10.2.1.3.2 Telescope Processing Optics

Behind the telescope objective described in the previous section are located the processing optics. These are schematically illustrated in Figure 10.2.1-3, which is a rear sectional view of the telescope.

10.2.1.3.2.1 Intermediate Error Sensor Optics

The first beam splitter intercepts the converging light bundle before it comes to a focus, permits about 20% to pass through to the field lens and reflects the balance to the field stop of the fine guidance line of sight. The intermediate line of sight is reflected off the telescope axis by a flat mirror and folded back at an acute angle by another flat mirror to a minifying relay lens. This lens has a focal length of 163.4 mm and a clear aperture of 12 mm. It provides a magnification of 0.1 for the image it forms in the focal plane of the intermediate error sensor. This is similar to the crossed-aperture sensor described in a following subsection.

10.2.1.3.3 Fine Error Sensor Optics

Returning to the first beam splitter, the reflected 80% of the incoming star signal is directed toward a field stop which consists of a pinhole, 0.5 mm in diameter. This restricts the field to about 4 seconds of arc and serves as a protection against stray radiation. The field transmitted by the stop is relayed, magnified six diameters, to the image planes of the two fine guidance sensors. The relay lens has a focal length of 19.6 mm and an aperture of 2 mm. Between the relay lens and the fine guidance error sensor image planes are the two fine guidance actuators. The first actuator is an electromagnetically driven mirror which serves the dual function of a beam steerer in response to fine error sensor commands and folding mirror to aid in packaging the optics. The second is a diasporameter pair which can

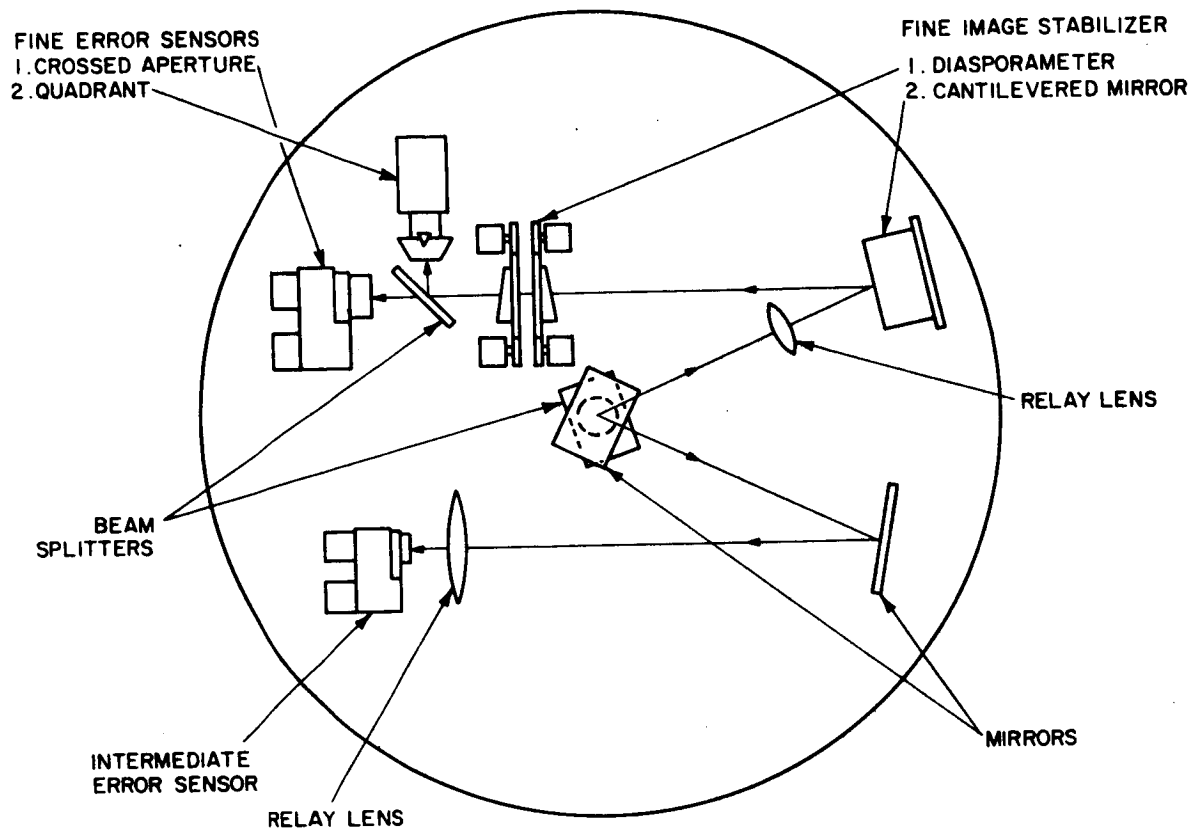


Figure 10.2.1-3. Fine Guidance Telescope Processing Optics.

also deflect the beam in response to error signals. Both these devices are described in subsection 10.2.1.3.8. After passing through the actuators the light forms an image at the fine error sensors. A beam splitter placed before the image plane divides the energy equally between the two sensors. Both sensors are described in detail in subsection 10.2.1.3.7.

10.2.1.3.4 Star Simulator

For inflight checkout and alignment a star simulator is provided. This unit consists of a collimator with a calibrated light source illuminating a pinhole at the collimator focus. The intensity of the light source is adjustable to provide an overload for the purpose of testing the sun protect circuit. The simulator is mounted in the telescope cover so that with the telescope open the simulator is out of the telescope field. The cover is indexed so that when it is closed the simulator is precisely in position within acceptable tolerances. When the simulator is in use the telescope gimbals are caged and the images formed at the sensors can be relayed to the spacecraft via an optical or television link.

Prime function of the star simulator is to align the intermediate and fine error sensors so that the nulls of the fine sensors coincide with the center of the intermediate sensor field. The zero positions of the fine actuators are adjusted for this purpose through the link mentioned above.

Preliminary checkout will also be performed by introducing known offsets into the simulator optics and monitoring resulting error signals from the sensors.

10.2.1.3.5 Telescope Structural Design

The primary structure telescope tube has been configured for light weight and high rigidity. A preliminary dynamic response analysis and subsequent stress analysis has validated the structural design configuration of the FGT (See section 10.4.1.8). The cylindrical telescope structure supports the bulk of the loading at three locations: (1) the secondary mirror and supporting structure, baffle and baffle mechanism, cover mounted star simulator and the forward looking star tracker near the forward end; (2) the CMG's, rate gyros and mating surface (inner gimbal ring) to the gimbal isolation suspension system, in the vicinity of the center of gravity of the system; and (3) the primary mirror and supporting structure, aft looking star tracker, and the fine and intermediate error sensor subsystems near the aft tip of the telescope cylinder.

The areas of the tube which support the external loading are aluminum rib-stiffened tubular castings which are connected by thinner tubular aluminum sections. The forward and centrally located stiffened sections are 0.25 inches thick, reinforced circumferentially by 30 axial 0.75 inch x 0.25 inch ribs, with a maximum O.D. of 34 inches. The aft section of the well is stiffened to support the primary mirror and its structure by 30 axial ribs, 2.75 inches x 0.25 inches. This section configuration gives additional stiffness to the tube supporting the external loading while uniformly distributing the load circumferentially to the tubular structure connecting the stiffened sections.

The connecting structure is 0.063 inch thick aluminum tube 32.6 inches in outside diameter.

Secondary structures connect the primary and secondary mirrors to the telescope tube. The primary mirror is supported by three tangent bars 120° apart, which mount through an interface ring to the telescope well. The tangent bars offer increased stiffness to laterally applied loads (along y, z axes). The secondary mirror is supported by four vanes located 90° apart.

Insulation is provided by a thermal shield outside of the telescope tube, 2 inches from the structural wall. The thermal shield consists of an alzak cover sheet, incorporating 15-20 layers of NRC-2 Super Insulation. The outside of the telescope baffle is coated with a low absorptivity/emissivity ratio coating such as silicon oxide. The inner surfaces of the telescope tube and baffle are optical black (black epoxy paint). Insulation is also provided for the primary and secondary mirrors.

10.2.1.3.6 Attitude Control System

10.2.1.3.6.1 Coarse Loop

The coarse attitude control loop is illustrated schematically in figure 10.2.1-1. The active elements in the loop are two astro-trackers with variable offset which establish the corrections required, and control moment gyros to null out the error signals by applying the appropriate torques to the telescope. In operation the pointing attitude of the telescope is initially known to the accuracy of the spacecraft controls. Based on this information the desired offset for the reference stars is calculated and entered into the gimbals. With a field of view of ±1 degree the astro-trackers acquire the reference stars and drive their respective lines of sight to null the errors. This introduces a difference between the actual and commanded astrotracker gimbal angles. These differences are converted into pitch and yaw error signals and applied to the appropriate control moment gyros to torque the telescope and thus null out the difference between the commanded and actual gimbal axis.

10.2.1.3.6.2 Intermediate Loop

As the gimbal angle difference approaches null the target or guide star image appears in the field of the intermediate error sensor. At this point the switching logic transfers control of the CMG's over to the intermediate error sensor. The intermediate loop operates much the same way as the coarse loop except that the error signal is developed directly at the error sensor rather than from an auxiliary loop through the star tracker gimbals. As the intermediate loop approaches null the image of the guide star appears in the field of both fine error sensors. During the fine loop operation the intermediate loop continues to operate to maintain the telescope axis pointed at the guide star.

10.2.1.3.6.3 Fine Loop

The fine loop is basically different from the coarse and intermediate loops insofar as it makes no effort to steer the telescope, but rather it steers the beam by means of actuators, to keep the guide star centered in the fine guidance field. A schematic block diagram is shown in Figure 10.2.1-4. The main constituents of the loop are the two error sensors and two beam pointing actuators. Both sensors are in simultaneous operation with respect to image scanning and error signal production. The error signals from one are used to close the loop and are recorded, while the signals of the other are recorded only. On the other hand, only one of the beam pointing actuators is used at a time, the other being locked in a neutral position. In operation, the error signals developed by the selected error sensor are applied to the selected fine beam actuator. Any departure from null generates a signal which drives the actuator to restore the null.

10.2.1.3.6.4 Fine Guidance Loop Summary

Table 10.2.1-5 summarizes the parameters of the various stages of spacecraft, telescope and image stabilization, emphasizing the adequacy of the fields of view of the successive stages to exceed the residual errors or cones of uncertainty of the preceding stage.

10.2.1.3.6.5 Roll Stabilization

For coarse and intermediate acquisition when roll is significant the telescope is constrained in roll by the hard mount gimbal of the isolation suspension. In the soft mount modes the target is essentially on axis so that roll is not significant and the roll stability provided by the elasticity of the suspension suffices.

In possible applications not part of the fine guidance experiments, such as star field photography which is discussed in section ,roll stabilization by the suspension would not be satisfactory. Further measures to control roll for this purpose are outlined in the foregoing section reference.

10.2.1.3.7 Fine Error Sensors

In this experiment, two error sensors are to be compared in performance. In appearance, construction and operation they are dissimilar but in function they are identical.

10.2.1.3.7.1 Dual Aperture Sensor

The primary fine error sensor is shown in Figure 10.4.1.4-1 in which section it is discussed in detail. It comprises two rectangular apertures of equal width at right angles to each other, which are driven by two tuning forks at dissimilar frequencies. At rest an instantaneous transparent square aperture is formed, but in operation this aperture covers the total available field in a precise mathematical pattern. By two axis Fourier analysis a linear error gradient for position, and intensity level for recognition or

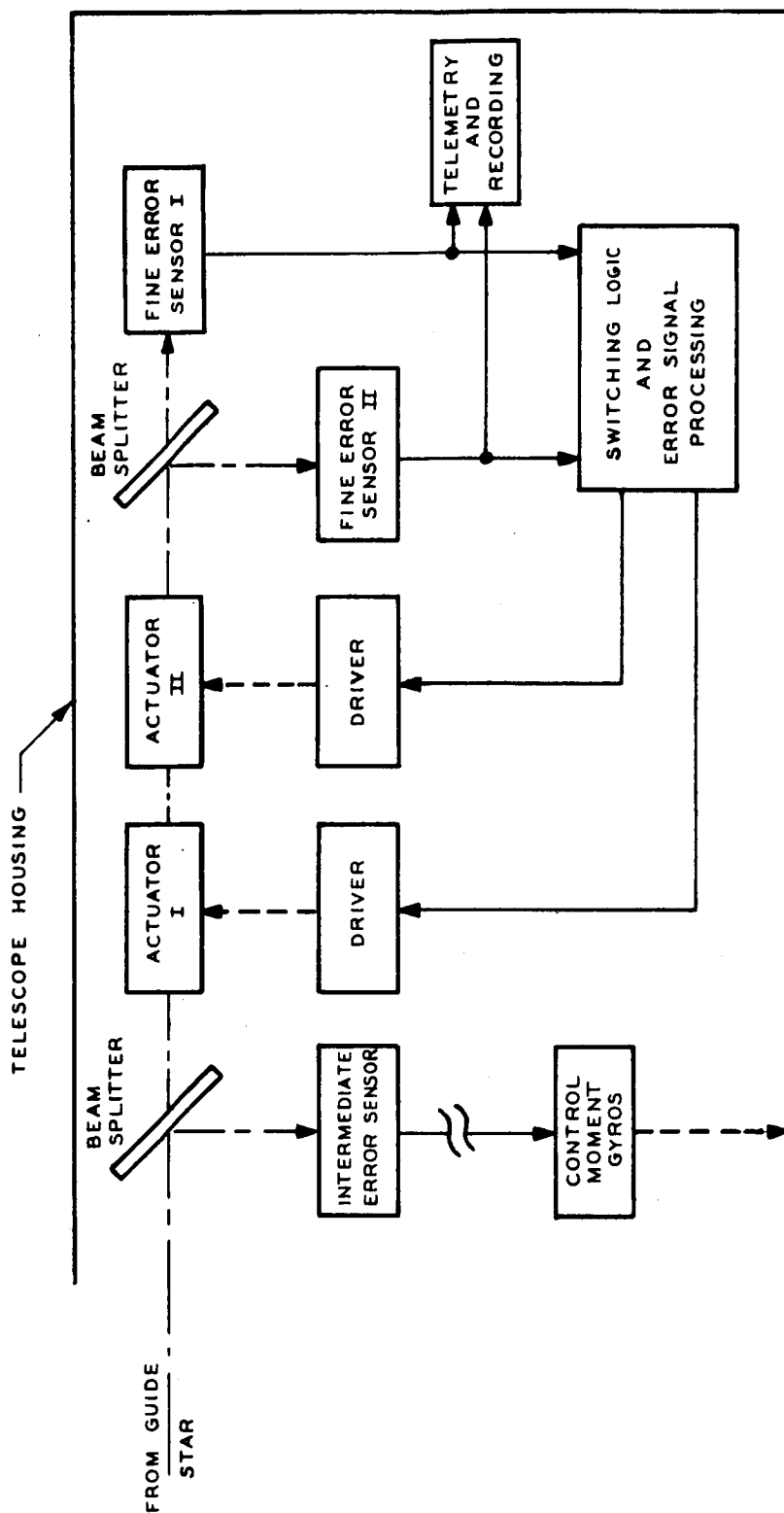


Figure 10.2.1-4 Fine Error and Image Stabilizer Loop, Block Diagram

TABLE 10.2.1-5
SEQUENTIAL SENSING AND POINTING OF THE FINE GUIDANCE EXPERIMENT

SPACECRAFT	
Attitude Sensing: Pitch, Roll and Yaw Inertial System, sun and horizon sensors	Pointing: to $\pm 2^\circ$ Pitch and Yaw CMG's and reaction jets
FINE GUIDANCE TELESCOPE	
Coarse Error Sensing Two Star Trackers (fore and aft). Individual Optics. Two axis. Bright star +1 to +4 mag. req'd. Coordinate transformation into Pitch and Yaw Field of View: $\pm 1^\circ$ Resolution ± 15 sec. of arc. Residual Error ± 30 sec. of arc.	Coarse Pointing: Pitch and Yaw using CMG's $\pm$ 40 sec of arc.
Intermediate Error Sensing Fine Guidance Telescope 10-30% of energy from primary optics (FL = 1.2 m, F/1.6) Pitch and Yaw Field of View: ± 1 min. of arc. Resolution: ± 0.2 sec. of arc.	Intermediate Pointing: Pitch and Yaw using CMG's ± 5 sec. of arc.
FINE IMAGE POSITION	
Sensing: Fine Guidance Telescope EFL 72 m 70-90% of collected energy F/96 Pitch and Yaw Field of View: ± 1 sec. of arc. Resolution: from ± 0.003 sec. of arc.	Fine Pointing: Pitch and Yaw Diasporameter Accuracy from ± 0.01 sec. of arc.

"presence" is obtained from the fundamental and second harmonic components of the error signals. The transducer converting the modulated light to electrical signals is a photomultiplier tube. Solid state detectors for this function are also under review.

This modulator-detector combination can measure image position deviation accurately from a stable and repeatable null. Prime advantage of this sensor are the high stability of the modulator and the a-c characteristics of the generated signals which permit stable amplification and phase discrimination. The electro-mechanical aperture motion is completely analogous to the operation of an image dissector tube with a square aperture, the mathematics for each type of device being identical. The dual aperture modulator is presently preferred over the image dissector because of superior null stability.

10.2.1.3.7.2 Pyramid Prism Sensor

The pyramidal reflector considered for the alternative fine error sensor is shown in Fig. 10.2.1-6. Star radiation impinging upon the apex of the pyramid is split into four equal beams at null, and in the usual configuration four individual photo detectors are arranged at right angles to receive the split beams. The outputs of opposite beams are applied to one differential amplifier which then provides the error polarity and magnitude read-out. The use of an amplifier whose gain is controlled by the intensity of the received energy (an A.G.C. amplifier), permits maintenance of a nearly constant error gradient from the differential measurement in spite of intensity variations.

A system of separate photo detectors has a weakness, however, in the development of drifts or differential sensitivity between opposing detectors. The innovation proposed here is the use of a toroidal reflector to reimage the four beams on a single scanning type detector such as an image dissector. The individual images are then alternately scanned and the pulsed information derived from the single detector permits elimination of drift. Here "null stability" is provided by the fixed pyramidal apex.

10.2.1.3.8 Fine Actuators

As with the error sensors there are two actuators. However, these act in series rather than in parallel, and only one is permitted to operate at a time, the other being held in a neutral position. Each changes the direction of the beam of light which is converging to a focus, the displacement in the image plane being equal to the product of the angular deviation and the lever arm associated with the actuator.

10.2.1.3.8.1 Diasporameter

The operation of the diasporameter, considered the primary fine pointing actuator, is shown in Fig. 10.2.1-7. The opposed optical wedges are shown in dotted lines within the driven housing. Two pairs of wedges are used - each pair comprising a separate diasporameter for the individual

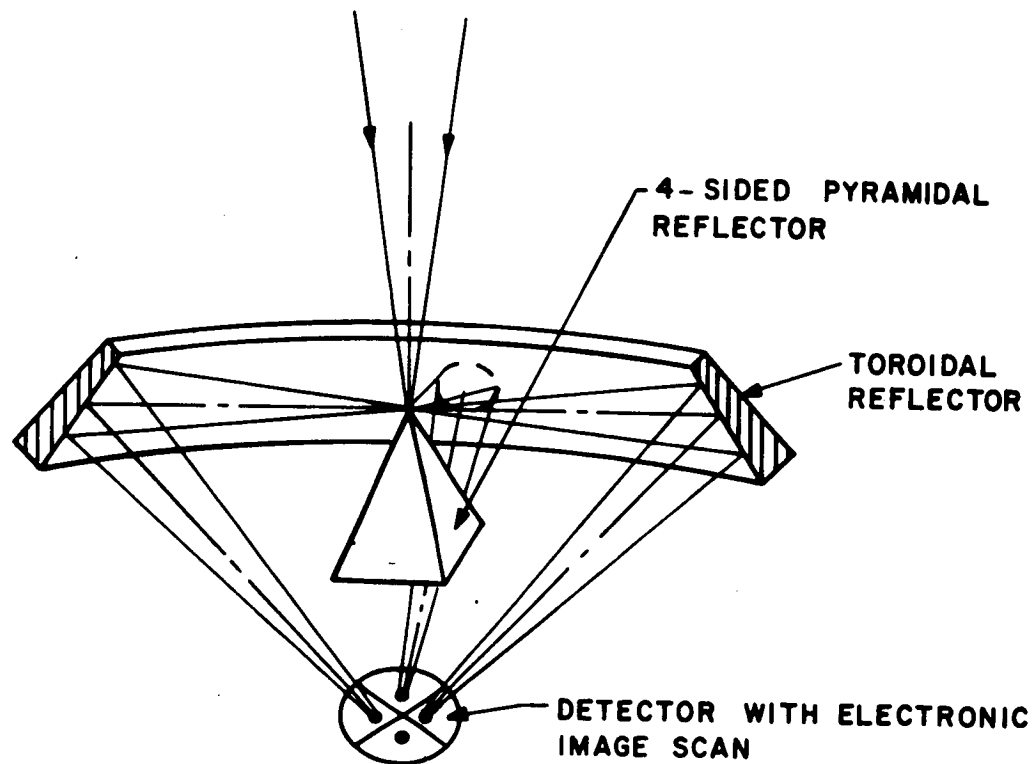


Figure 10.2.1-6 Alternative Fine Error Sensor

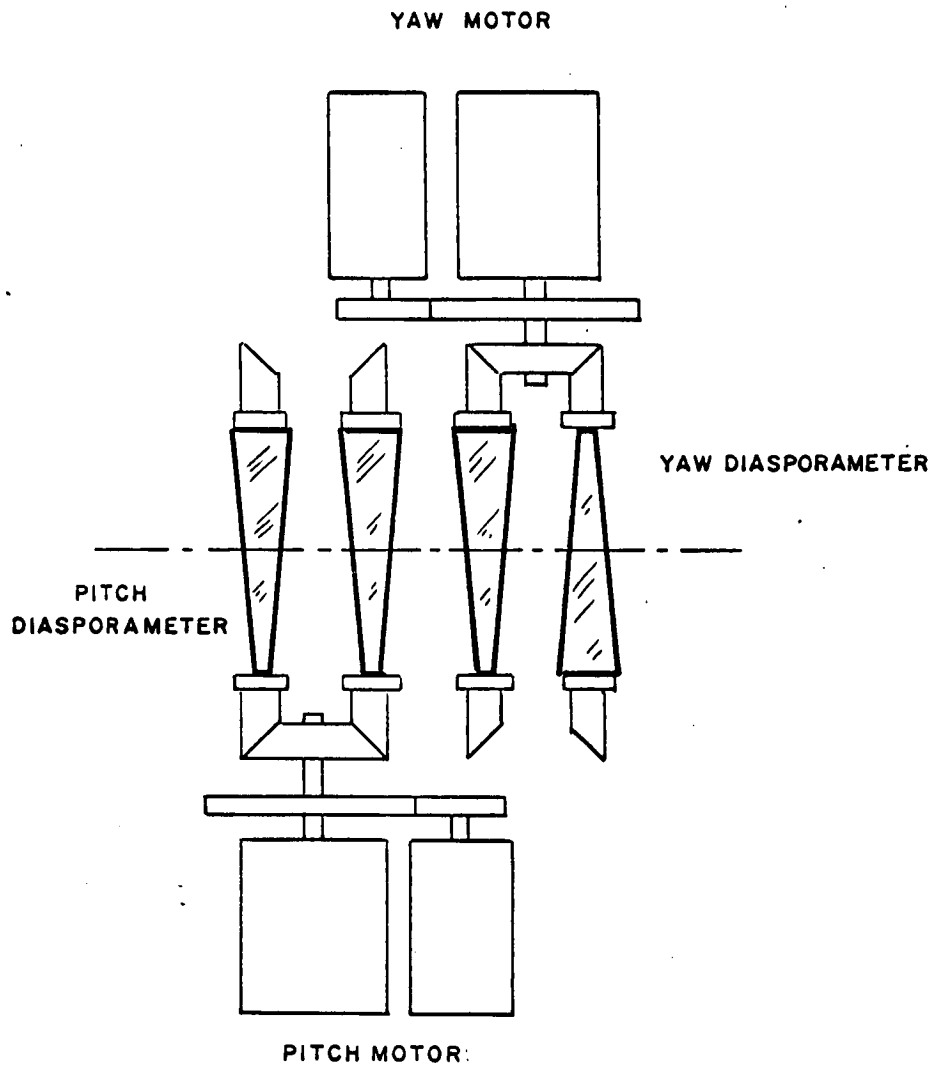


Figure 10.2.1-7 Primary Fine Pointing Actuator

axis. Each pair of wedges is driven by a bevel differential gear train operating through a planetary reduction from a motor. The bevel differential drives each individual wedge in a precise equal and opposite displacement relative to its mate. The diasporameter has very large leverage - that is, 180° relative rotation is equivalent to the deflection range measured in seconds. (See Sec. 9.5.1.1.2.1 for a discussion of the theory of the diasporameter). Thus, the effects of gear backlash in the bevel differential are negligible.

10.2.1.3.8.2 Cantilevered Mirror

The reflective type actuator considered as the alternate, as shown in Fig. 10.2.1-8 has been conceptually designed. The flat mirror is mounted in a mirror support to which is attached a magnetic disc forming an armature. This assembly is cantilevered on a slender rod which is surrounded by an electromagnet and a permanent magnet. The cantilever was designed for a deflection ratio of 200 to 1 between full deflection when used as an actuator and residual deflection due to gravity with no external drive power applied. The cantilevered configuration shown has a natural frequency of about 2000 cycles, and calculations indicate a required power of .32 watts per coil for a gap of 250 microns. Thus the actuator constitutes a low power deflector which does not use rotating components and appears very desirable for space application.

A breadboard version of this actuator built for another project but employing the same principle, is shown in Figure 10.2.1-9. Light focused on the oscillating mirror by the lens, is successfully deflected in a closely controlled manner by variation of current in the coils.

10.2.1.3.8.3 Actuator Stability in Neutral Position

The diasporameter presents no stability problem with respect to introducing changes in pointing when the cantilevered mirror is in use, since the gears and motors supply ample friction torque to keep the wedges from shifting from their neutral position.

In the case of the cantilevered mirror the mechanically isolated system of the fine guidance telescope will not transmit vibrations. Other possible sources of deflection are temperature, weight, and magnetic influence. Temperature effects will be slow and minute (principally from possible lateral thermal gradients across the metal rod of some 1/8 in. dia.). Gravitation is negligible in orbit; g-loads of 1 or more in earth testing and launch will be withstood by mechanical limitation of excursion. Protection from external magnetic fields will be provided by encasing the actuator in a soft iron shield.

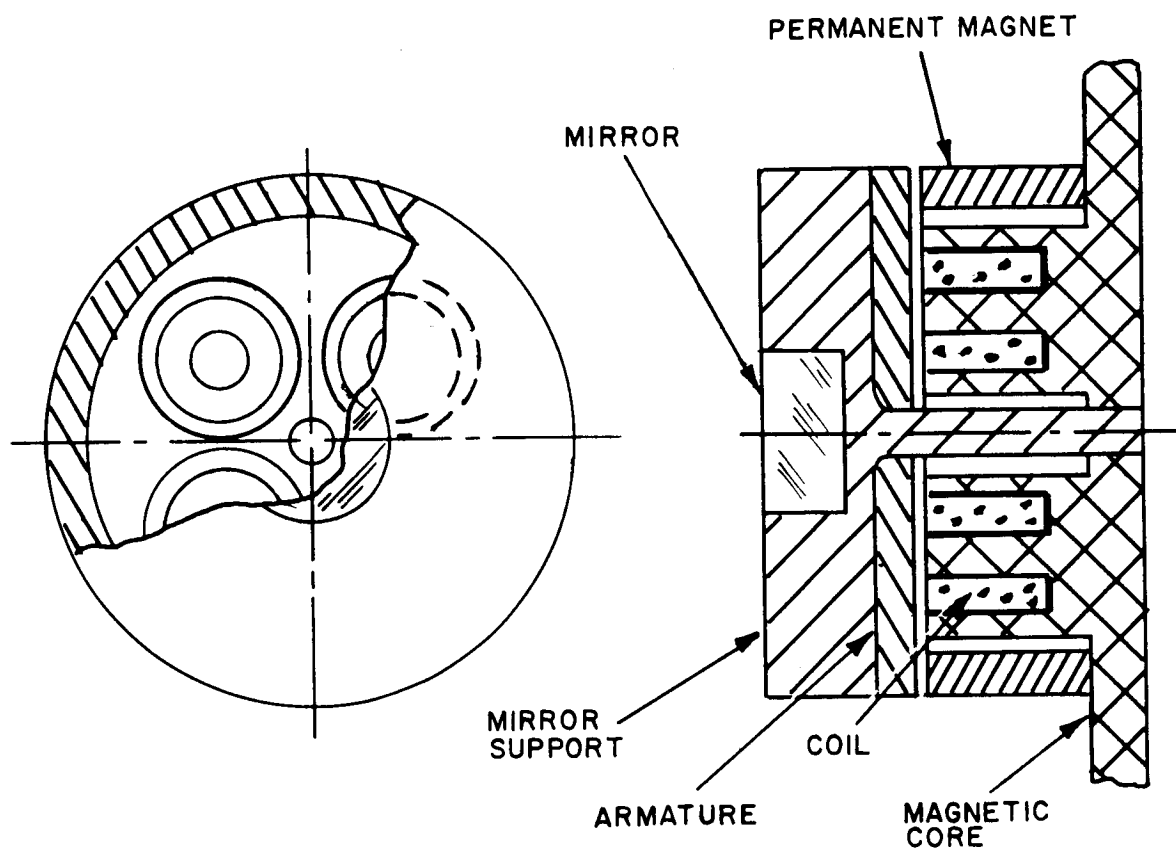


Figure 10.2.1-8 Alternate Fine Pointing Actuator

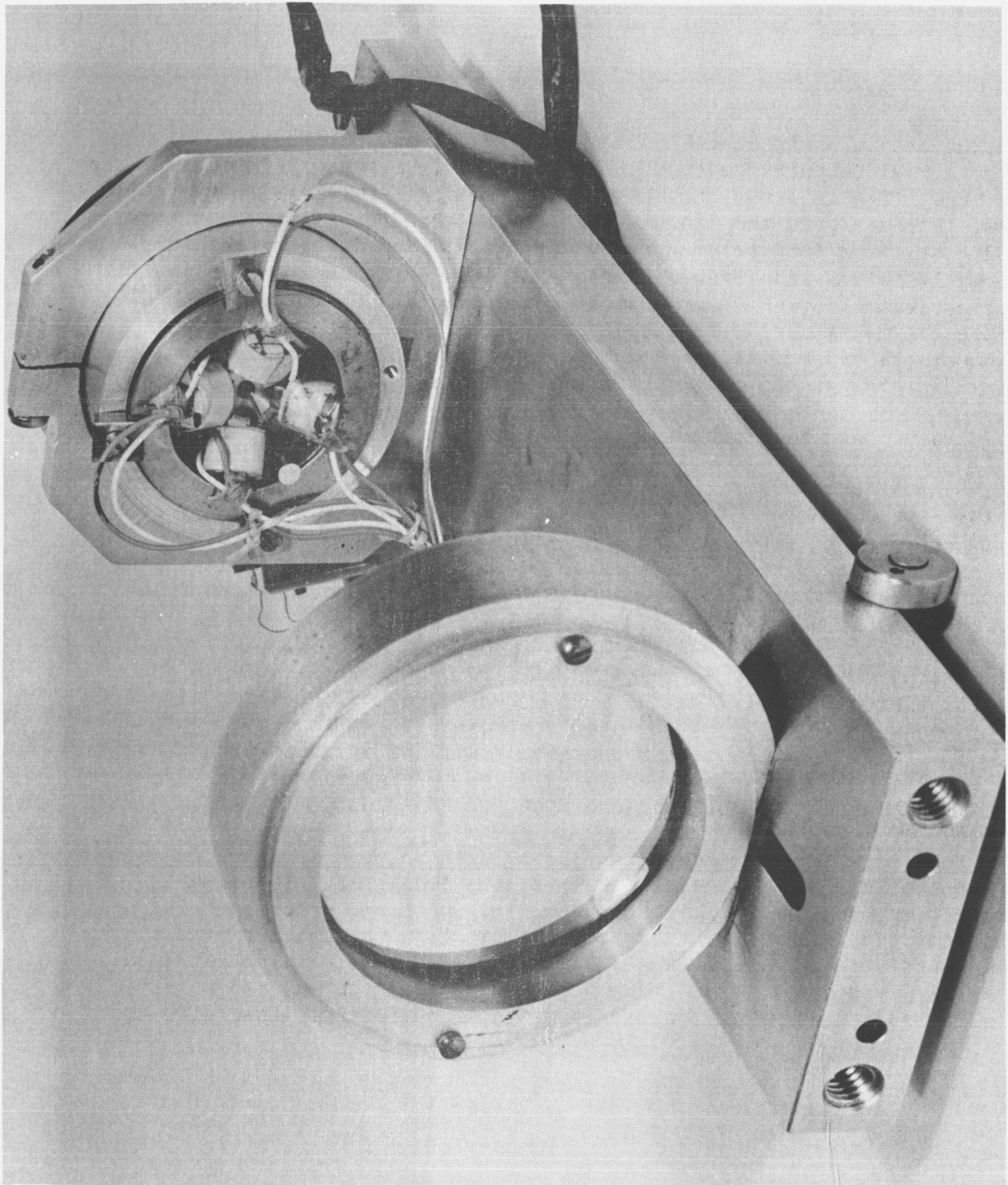


Figure 10.2.1-9. Cantilevered Mirror Breadboard.

10.2.1.4 Data Management

10.2.1.4.1 Data Taking

The fundamental data source in the fine guidance experiment is the error signal output of each of the two fine error sensors in each of the two axes. The information is sampled in analog form and displayed to the crew. For analysis the signals are sampled once per minute, converted to digital form, and telemetered to ground station for storage. Concurrently but at less frequent intervals sampled measurements of sun protect status, tracking mode status, star magnitude in each of the sensors, temperature at strategic points in the telescope and other measurements of significance such as magnetic field intensity can be relayed to the ground station via the telemeter link and recorded.

10.2.1.4.2 Data Reduction

The error signals generated by the fine error sensors are telemetered to the ground station where they will be displayed simultaneously on a strip chart along with appropriate status and environmental information. The simultaneity permits comparative measurements and the isolation of disturbances common to both sensors such as disturbances of the telescope or spacecraft.

In plotting the results an envelope will be drawn representing a departure of ± 0.01 sec. from the mean. Departures of the error signal beyond this limit will be tabulated according to amplitude, duration and frequency. In this manner one sensor will be compared with the other, one actuator with the other, and it will be determined whether any given tracking sequence was acceptable or not.

With the error signal plots, star identification, tracking system status and appropriate environmental data are made available. By comparing these items with the performance record of the error sensors it is possible to determine the satisfactory operation threshold of the system as a function of star magnitude, star spectral class, star background intensity, angular separation from bright bodies, thermal and magnetic conditions, spacecraft disturbances and combinations of these effects.

10.2.2 Isolation Experiment

To stabilize an orbital telescope to a precision of 0.01 arc second as required for future space astronomy missions, disturbance torques, which act to destabilize the pointing, must be reduced to an absolute minimum. Such torques arise from sources both external and internal to the spacecraft and telescope. The external torques, caused by such diverse sources as aerodynamic forces, differential gravity, micrometeoroid impacts and solar radiation are inherently small, and it is believed that with the development of improved stabilization technology, as discussed in the above experiment, the telescope can be stabilized to the required precision against them.

The internal torques, however, are more difficult to handle. Caused primarily by movements of the crew inside the spacecraft, they can be quite large if crew freedom is unrestricted. If the required telescope stability is to be achieved, a means must be developed to attenuate these man-produced disturbance torques to a level comparable to that of the external torques. One way to achieve this attenuation is to mount the telescope in a suspension system, such as springs, so that it is dynamically isolated from the spacecraft. Analyses performed here and elsewhere⁽¹⁾⁽²⁾ indicate that theoretically such a suspension system will provide the required isolation. Also, much design experience with spring isolators for ground-based and airborne use is available.

However, the isolation problem in space is essentially different from that on ground. First, the isolation system must be effective at frequencies comparable to the orbital rates which requires a very soft suspension. Second, due to the very low torques that can be tolerated, the force levels that are transmitted through the suspension system must be extremely small and non-ideal behavior of the suspension becomes important. Third, the peculiar zero-g environment of space essentially nullifies the validity of test data taken in a one-g environment. Finally, in considering the need for testing isolation systems in space, the major function of such systems, which is to support the telescope stabilization system, must be remembered. Before deploying a large manned orbital observatory, designed for diffraction-limited performance and requiring precision pointing, it should be demonstrated, somewhere, that in the projected operating environment the required pointing stability can be achieved. At the present time, even under the best possible conditions on earth, this demonstration cannot be made. For these reasons, this experiment, in which critical experimental data will be obtained in space under actual operating conditions, is proposed.

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- (1) Boeing - A System Study of a Manned Orbital Telescope - The Boeing Co., Seattle, Washington. Under Contract NAS 1-5968 prepared for NASA-Langley Research Center, October 1965.
 - (2) NASA/EIA Briefing on Aerospace Electronics Systems Technology - E. C. Foudriat, Satellite Attitude Sensing and Control, May 3 - 4, 1967.

To obtain the required experimental data, the performance of several types of isolation systems, ranging in complexity from simple gimbals to active magnetic suspensions and including springs, will be evaluated and compared.

The flight data from this experiment, coupled with that from the Fine Guidance experiment, should determine not only the performance of future telescope stabilization systems, but should include that vital experimental information without which these systems cannot be designed.

10.2.2.1 Justification

As discussed above and in subsection 10.1, the justification for performing this experiment is based on the following factors:

- a. The suspension systems studied here, and all suspension systems effective in the low-g environment of space, will not support the weight on earth of the telescope to be suspended. This means that, without elaborate, cumbersome and expensive simulation facilities, no more than two axes of the suspension can be tested at any given time. Facilities for this purpose have been investigated in detail in a Chrysler-sponsored parallel study. Based on the results of this study, it was concluded that the most promising approach to development of isolation technology is to use a simple low-g simulation facility, effective in two axes of translation and one of rotation, to develop the prerequisite technology for flight experiments, then in the flight experiments themselves collect the design information essential to future orbital astronomy applications.
- b. Even if the elaborate facilities for low-g simulation, in which the performance of the suspension systems would be essentially like that in space, the complementary part of this experiment, the Fine Guidance experiment, could not be incorporated. This is an important point. In space, the entire space system, made up of the telescope and its two control systems, the suspension, the spacecraft and its control system and the crew personnel, will comprise an extremely complex dynamic system. One promising suspension system, for example, incorporates eight degrees of freedom. It is not realistic to try to simulate that entire system on the ground. Yet, preliminary dynamic analyses show that, for the disturbance frequencies, control frequencies and elastic and rigid-body modes involved, simplified approximation are not valid.⁽³⁾ This is another reason, then, why the entire system should be tested and evaluated in the space environment in which it finally will be deployed.

(3) Spicer, J. A., The Mechanics of a Telescope Suspension System, Chrysler Corporation Space Division, TB-AE-87-367.

10.2.2.2 Experiment Design

A ground rule established at the beginning of this study specified that, in designing the individual experiments, maximum consideration should be given to alternate approaches and comparison of competing techniques. That is, the objective of the experiment should be to collect the maximum design information on competing techniques, rather than evaluating the performance of a particular design. This objective imposes the following broad requirements on experiment design:

- a. More than one competing technique should be evaluated.
- b. Operation of each suspension and the evaluation of its performance should be independent of the other suspensions.
- c. The experiment should be conducted so as to evaluate the performance of each suspension type, collect design information on that type for future applications and to resolve as many unanswered questions as possible relative to isolation requirements and over-all telescope stabilization problems.
- d. Consideration should be given to the use of the crew in performance of the experiment.

These requirements might be summarized by stating that the space experiment, which is a technology development effort, should be conducted within the inherent constraints like a similar ground-based effort.

In order of complexity, the isolation systems are:

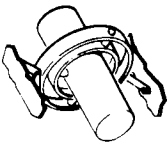
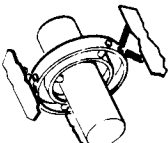
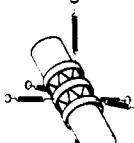
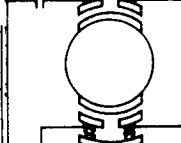
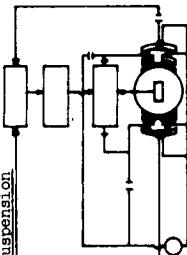
- a. Hardmount. Comprised of mechanical gimbals, this system is designed to decouple rotation of the telescope from that of the spacecraft in two axes.
- b. Spring Suspension. The hardmount is suspended from the spacecraft by springs so as to dynamically decouple telescope torques from spacecraft rotation and translation.
- c. Magnetic Suspension. The telescope is suspended, without gimbals, along three orthogonal axis so as to dynamically isolate it from the spacecraft in six degrees of freedom.

10.2.2.3 Measurement Objective

The three types of suspension systems are organized into five operational modes. Illustrated and summarized in table 10.2.2.3-1, these modes are designed to resolve the major questions relative to manned orbital telescope

TABLE 10.2.2.3-1.

OPERATIONAL MODES

OPERATIONAL MODE	DESIGN PARAMETERS EVALUATED	EVALUATION TECHNIQUE
A. Hardmount Mode 	1. Effect of gimbal friction torque on telescope image stability.	Evaluate pointing stability with and without roller bearings under same operating conditions.
	2. Effect of inertial unbalance torques on telescope image stability.	Operational procedure provides for correlating pointing stability with spacecraft motion over a wide range of disturbance magnitude and frequency.
	3. Effect of spurious lead torques on telescope image stability.	Image stability is evaluated first with flex lead coupling, then without leads by use of batteries and telemetry transmitter on telescope.
	4. Effect of CMG control actuator operation on telescope image stability.	Image stability is evaluated with and without CMG operation by means of permanent magnet torquers on the gimbals.
B. Suspended Hardmount Mode 	1. Effect of coupling between gimbal friction and spring constraints on dynamic stability.	Operational procedure provides for evaluating image stability across a wide spectrum of dynamic excitation.
	2. Effect of power and data lead constraints on spring response.	Dynamic response of suspension and telescope image stability will be measured with and without flex leads, by using the telescope batteries.
	3. Damping Requirements of spring suspensions and effectiveness of several techniques of meeting these requirements.	Provision is made to compare dynamic response of springs under varying damping condition including no damping, spring hysteresis damping, and magnetic hysteresis damping.
C. Three-Axis Spring Suspension 	1. Effect of spring rotational constraints on static tracking accuracy and dynamic image stability.	Evaluation of image stability as a function of commanded spacecraft/telescope relative orientation error under static disturbance conditions.
	2. Effect of power and data leads in bypassing spring isolation.	Comparative evaluation of telescope image stability with and without flex lead coupling, by use of batteries and transmitter on telescope.
	3. Damping requirements for three-axis spring isolation. Performance evaluation of damping techniques, and evaluation of the bypass effect of auxiliary dampers.	Comparative evaluation of isolation performance and image stability with no damping, with spring damping, and with auxiliary magnetic hysteresis damping.
D. A.C. Magnetic Suspension 	1. Effect of magnetic cross-coupling on the isolation performance and telescope image stability.	Evaluation of static tracking accuracy as a function of airgap and commanded telescope/spacecraft error angle.
	2. Evaluation of isolation performance and image stability as a function of suspension compliance.	Provision is made to vary the equivalent spring constant by electrical means.
	3. Evaluation of isolation performance and image stability as a function of damping.	Provision is made to vary the damping coefficient from zero to critical.
	4. Evaluation of isolation performance as a function of magnetic airgap.	Provision is made to vary the magnetic airgap over a small range.
	5. Determination of optimum compliance/damping combination.	Evaluation of image stability and isolation performance with different combination of damping and compliance.
E. D.C. Servo Suspension 	1. Force attenuation of the D.C. suspension as a function of compliance, airgap, and damping and comparison with Mode B.	The accelerometer loop can be disabled, allowing the isolation performance of the D.C. suspension to be evaluated separately and compared to the A.C. suspension.
	2. Effect of accelerometer loop gain on isolation performance and stability of suspension system.	Means are provided for control of the damping and bandwidth of accelerometer loop.
	3. Effect of spacecraft/telescope orientation on performance of both position and acceleration loops.	Provision is made to evaluate suspension performance with a commanded error angle between telescope and spacecraft.

stabilization problems and to collect specific design information on the suspension types. The second column of the table gives the design parameters to be evaluated, or questions to be answered, and the third column gives the specific design feature of the experiment that answers that particular question.

The objective of the first mode, the hardmount mode, is to establish the isolation performance of the hardmount, evaluate the factors which limit this performance and establish a reference for comparing the other techniques. The disturbance torques which will limit the performance of the hardmount include gimbal friction, inertial unbalance torques, spurious lead torques, and vibration of the CMG inside the telescope. By providing both flex and roller bearings, gimbal torquers and inertial controllers, batteries and flex leads, and including both programmed disturbances and imposed quiet periods, all these effects may be separately evaluated.

It is not expected that the hardmount mode will provide for a stability of the rigid telescope to 0.1 arc second. However, if a stability of 1 arc second can be achieved, then the fine loop of the telescope stabilization system might provide the required image stability especially during the imposed quiet periods (see subsection 10.2.1.3.6.9 for a discussion of the fine stabilization loop).

The second operating mode shown in table 10.2.2.3-1 is the suspended hardmount mode. Simplest from a hardware viewpoint of the so-called "soft" modes, but the most complex from a dynamic viewpoint, this mode suspends the gimballed telescope by springs so as to dynamically isolate the gimbals from translation of the suspension support point. It is designed specifically to evaluate inertial unbalance torques, which are caused by crew motion coupled with spacecraft rotational instability and telescope mass unbalance.

From a performance viewpoint, the effectiveness of this mode depends on the effects of gimbal friction and spurious disturbances that might be introduced by flex leads. These effects may be determined by alternating the use of batteries and flex leads and excitation of the suspension with a controlled range of frequencies. It is desirable also to know the damping requirements, whether hysteresis damping is in the springs and translated coupling to the damped telescope stabilization system will suffice or if a separate damper is required. A separate magnetic hysteresis damper, which may be alternately engaged and disengaged, is provided for this purpose.

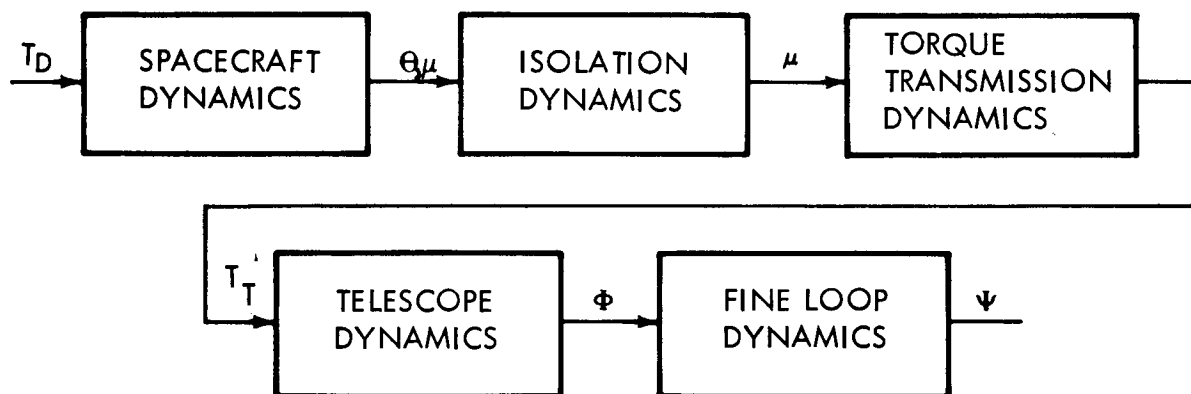
The other three operational modes are designed to evaluate different types of three-axis suspensions. For these modes, the gimbals are locked to simulate attachment directly to the telescope, and the objective is to dynamically isolate the telescope from the spacecraft in all six degrees of freedom. Thus, dynamically, it behaves like an unmanned telescope, but is constrained to fly in formation with the spacecraft. The first of these modes, the three axis spring suspension, has three sets of suspension springs, which

act along three orthogonal axes. The other two modes operate similarly, but the physical springs are replaced with "magnetic springs". Both A.C. and D.C. magnets are evaluated, the latter including one mode in which accelerometer signals are used for "active" control of the disturbances. All these modes are designed to collect performance data on three-axes suspensions as a group, to compare the relative performance of each, and to collect specific design information for each type. For this purpose, operation with and without the auxiliary damper with and without batteries and with variable magnetic suspension design parameters and a wide range of disturbances excitation are provided

10.2.2.4 Equipment Design

10.2.2.4.1 Requirements Analysis

The overall spacecraft system that determine the image stability of a manned orbital telescope consists of the spacecraft and its control system, the telescope and its coarse and fine control systems, and the disturbance environment to which the system is subjected. These various elements are shown schmatically below to illustrate their interrelationships.



The spacecraft is subjected to a disturbance torque environment, assumed here to be produced primarily by crew movements, and undergoes an angular disturbance motion the magnitude of which depends on the dynamic properties of the spacecraft and its control system. This causes the telescope support point, located at some distance from the spacecraft center of rotation, to undergo a linear motion, and some part of this motion is transferred through the isolation system to the telescope. This will produce inertial reaction forces on the telescope, and unless it is perfectly balanced about its support point, the result will be disturbance torques to the telescope. Part of this motion may be compensated by a fine control system in the telescope that stabilizes the image against small angular excursions of the telescope. The object here is to determine the isolation requirements, that is, to determine the extent that spacecraft motion must be attenuated before it reaches the telescope. To determine these requirements, the dynamic properties of all the other elements in the system must be known or assumed.

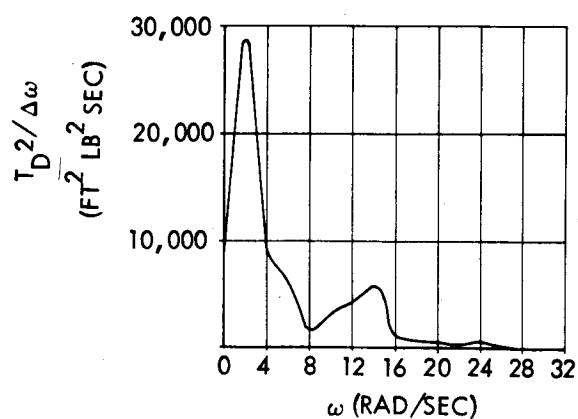
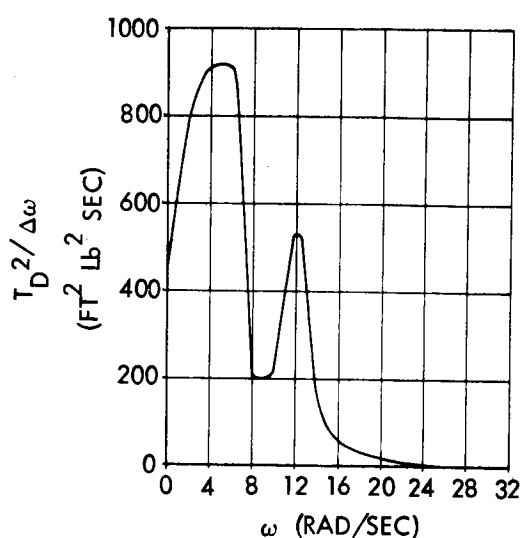
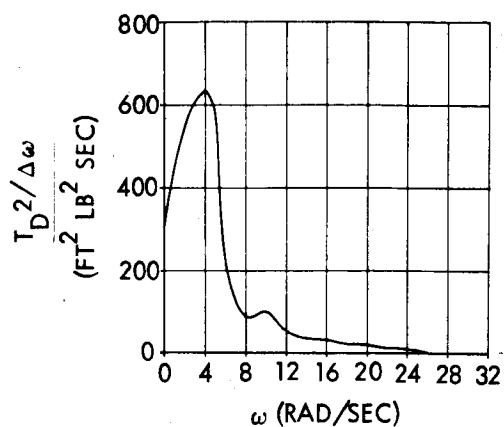


Figure 10.2.2.4.1-1. Power Density Spectrum of Man-Produced Disturbances

Each element in the above sketch is treated as a filter described by a linear transfer function. The disturbance environment is described by a torque amplitude vs frequency characteristics and is applied to the several filters in series. The required output for the filters is known; the telescope image stability must be 0.01 arc second or better for all disturbance amplitudes and frequencies. Therefore, the required overall transfer function is known, and, knowing the transfer functions of the other elements, the required isolation can be determined.

The parametric relationships among these elements are given in figures 10.2.2.4.1-1 through 10.2.2.4.1-7. These results are derived in subsection 10.4.2.1 (Supporting Analysis) where the assumptions and limitations involved are also discussed.

Power-density spectra for the man-produced disturbance environment are given in figure 10.2.2.4.1-1, for three cases of crew freedom of movement. The disturbances, it will be noted, are heavily concentrated around one cps with almost all the power contained in the two-decade range between 0.5 and 50 cps. For the spacecraft control system to be effective in stabilizing the spacecraft against these disturbances, its bandwidth should encompass this range. This is illustrated in figure 10.2.2.4.1-2, where the translation of the telescope support point on the spacecraft is given as a function of the spacecraft control frequency.

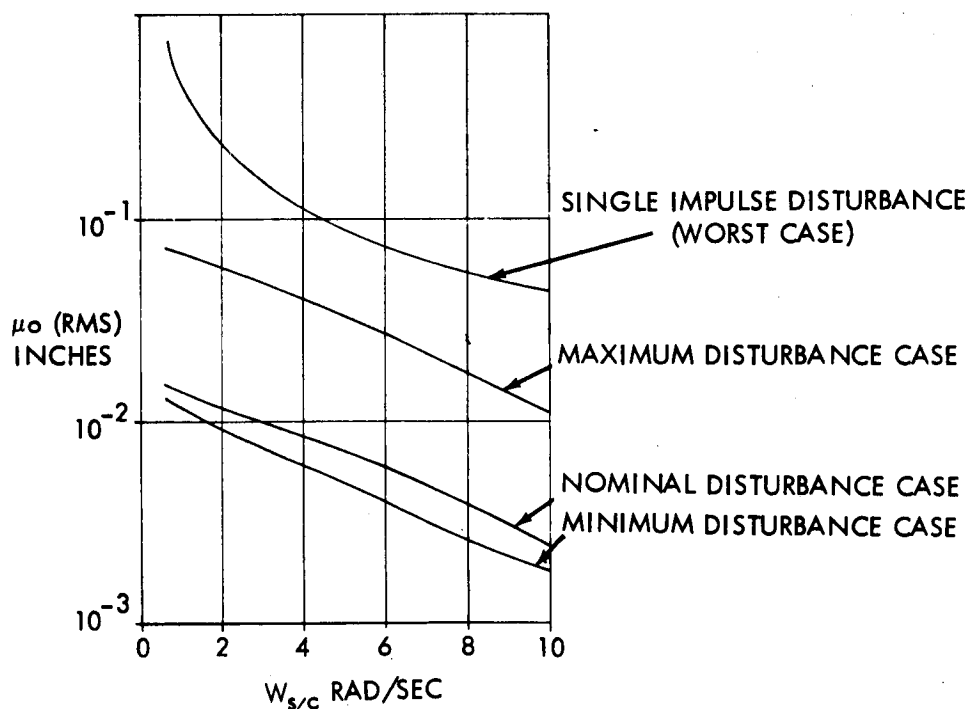


Figure 10.2.2.4.1-2. Translation of the Telescope Support Point

The bottom three curves are for the disturbance spectra given in figure 10.2.2.4.1-1 and represent a steady-state mode of operation in which the motion of the support point (rms) is very small. For a control system bandwidth below 0.6 rad/sec, for example, the response is practically independent of the control system. Even for this case, though, the expected worst-case periodic excursion is less than 0.1 inch. However, in deriving the isolation requirements, both periodic and impulsive disturbance must be considered. Thus, the upper curve of figure 10.2.2.4.1-2 is for an impulse caused by a 175 lb man importing 3 ft/sec velocity to himself by reacting on the spacecraft. It is a plot of the maximum motion of resultant damped-sinusoid response, which is highly sensitive to the spacecraft control system bandwidth, rising to 0.5 inch at a little less than 1 rad/sec. This value of 0.5 inch maximum telescope support point excursion is chosen as a design value for hardware mechanization of the isolation systems.

The next element of the system to be considered is the isolation system. The spacecraft side of the suspension system μ_0 is driven periodically with an rms amplitude as indicated in figure 10.2.2.4.1-2. The response of the telescope side of the suspension μ is determined by the transmissibility μ/μ_0 . All the numerical results in this section are based on a single degree of freedom model. The transmissibility of this model is shown in figure 10.2.2.4.1-3 as a function of the damping ratio z and the undamped natural frequency ω_I , called the isolation frequency.

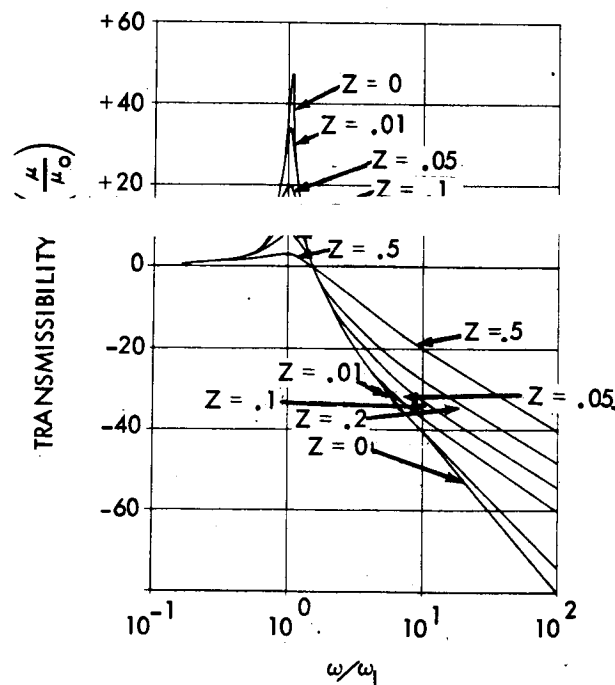


Figure 10.2.2.4.1-3. Transmissibility Curves.

At higher frequencies the slopes of all the curves approach 20 db/decade of frequency but the transmissibility at any frequency is determined by z . The effect of damping on isolation is shown in figure 10.2.2.4.1-4. As can be seen, a small damping ratio provides better isolation, except at frequencies close to ω_I . By choosing ω_I well below the disturbance frequencies, little damping is required which alleviates the hardware mechanization problems. Since the man-produced disturbances are concentrated around 1 cps, this is a condition easy to satisfy. Where applicable to the results of this section, a damping ratio of 1 per cent of critical is assumed.

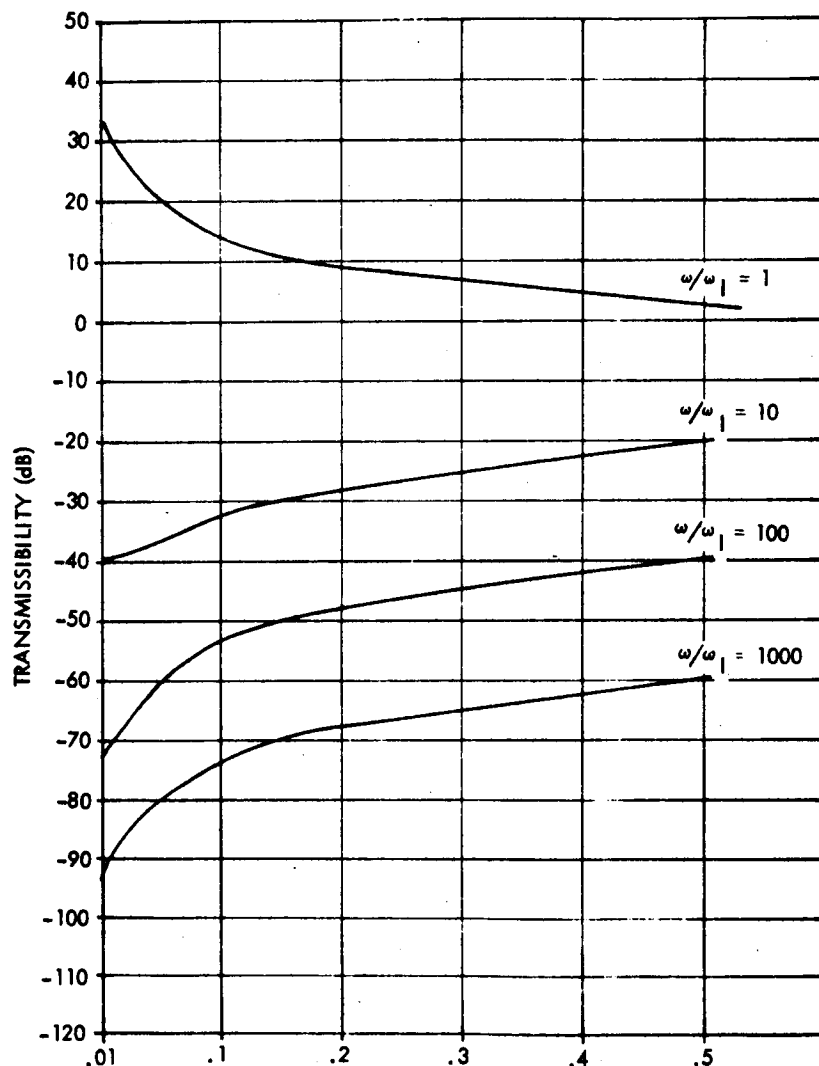
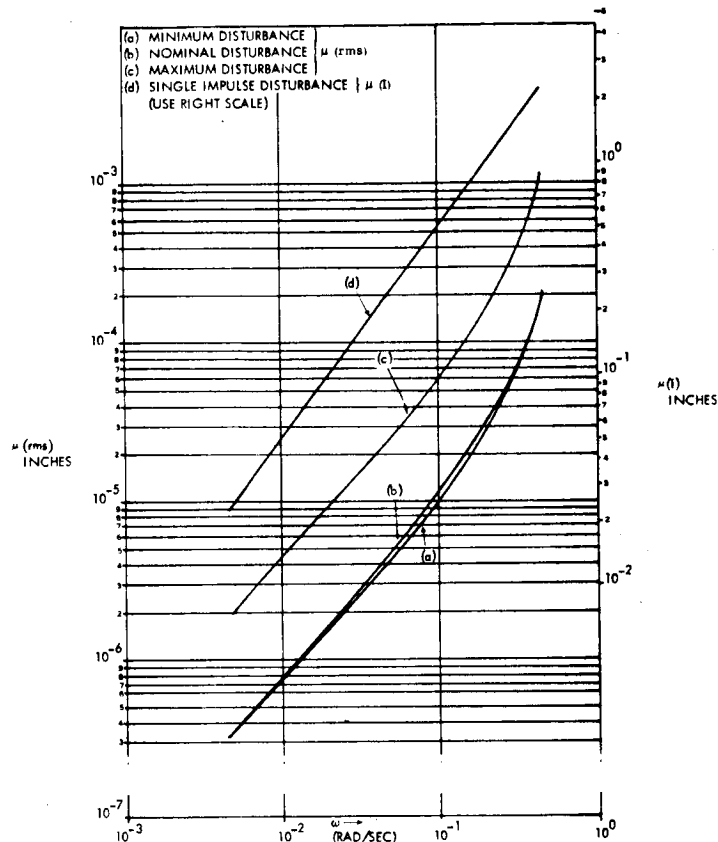


Figure 10.2.2.4.1-4. Effect of Damping on Transmissibility.

Before choosing the isolation frequency ω_I , one must look at the effect of this choice on the response of the telescope and its stabilization system. Figure 10.2.2.4.1-5 shows the effect of ω_I on the translational motion μ of the telescope for the several baseline disturbance cases.



10 2.2.4.1-5 Telescope Response Curves

These curves are for a spacecraft control frequency of 0.6 rad/sec and suspension damping ratio of 0.01. Two different scales are used for the periodic disturbances (left scale) and the single impulse disturbance (right scale). For the periodic disturbances, the response amplitude is extremely small for all isolation frequencies considered. The steepness of the higher-frequency end of these curves reflects the fact that the isolation frequency is approaching the lower end of the disturbance spectrum. The ordinate for these curves is the expected rms amplitude. Curve (d) shows the response to the baseline impulse, and the ordinate of the right-hand scale is the maximum amplitude of the time response, measured in a fixed inertial coordinate system.

For purposes of determining the requirements on the suspension system, it is not the amplitude response shown in figure 10.2.2.4.1-5 that is important, however. Rather it is the maximum relative displacement between telescope and spacecraft and the acceleration of the absolute displacement. Due to the slow response of the suspension system compared to that of the spacecraft, the maximum relative displacement is essentially the displacement of spacecraft shown in figure 10.2.2.4.1-2. The effect of acceleration is considered next.

The disturbance torque acting on the telescope that is caused by acceleration of the telescope is plotted against isolation frequency in figure 10.2.2.4.1-6. Again, the three periodic disturbance conditions and the single impulse are considered and a damping ratio of 0.01 for the suspension system and a spacecraft control frequency of 0.6 rad/sec are used. Generally, it is an objective of the isolation system to keep the low-frequency torques transmitted to the telescope below the level of external disturbances and the high-frequency disturbances to a level consistent with the pointing requirements. Low-frequency torques are those well within the bandwidth of the telescope control system, so that, to that system, they appear as slowly-varying "constant" torques.

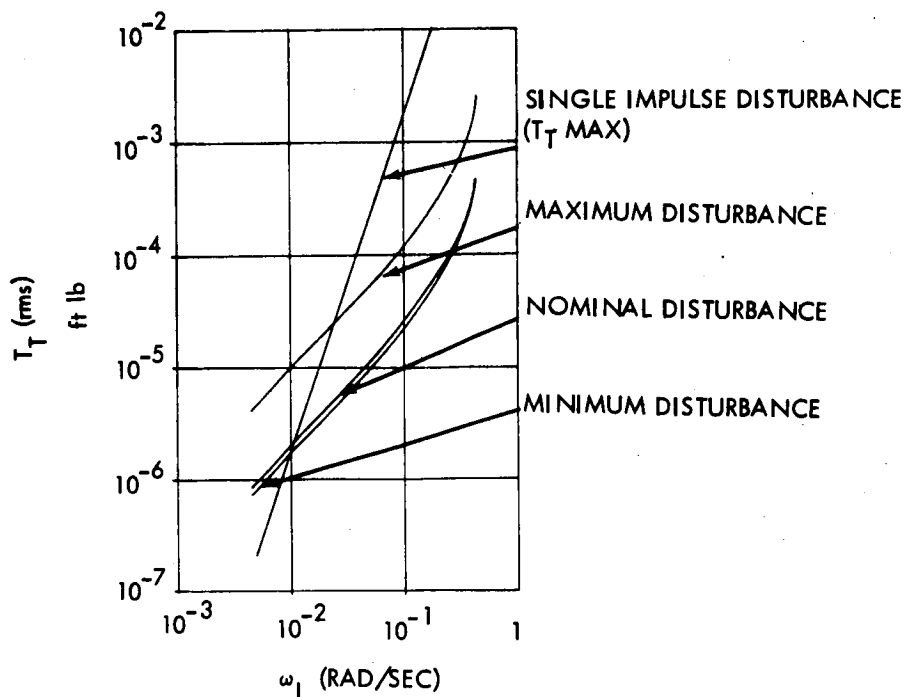


Figure 10.2.2.4.1-6. Telescope Disturbance Torque Curves.

All external torques are low-frequency, and the telescope response is determined by its control system. The response to higher-frequency disturbances, on the other hand, is governed by the telescope inertia independent of the control system. Most of the periodic disturbances considered here and plotted as the lower curves of figure 10.2.2.4.1-2 are the higher-frequency kind. But the disturbance caused by the impulse, curve (d), is in effect low frequency because of the filtering effect of the suspension. The resulting "constant torque" transmitted to the telescope is a cubic function of isolation frequency ω_I . And to keep the magnitude of this disturbance comparable to the magnitude of external disturbances (about 10^{-5} ft lb at synchronous altitude) requires an isolation frequency of about 0.01 rad/sec.

The effects of these disturbances on telescope image stability is shown in figure 10.2.2.4.1-7. The requirements of 0.01 arc sec is marked on the graph. The disturbance curves, spacecraft parameters, suspension damping, etc. are as previously defined. The telescope stabilization system consists of a fine pointing loop internal to the telescope, called the fine guidance loop, and a coarser pointing loop, called the intermediate loop, that acts on the telescope itself. These loops are defined and discussed in subsection 10.2.1. The purpose of the fine loop is to compensate for motion of the telescope itself.

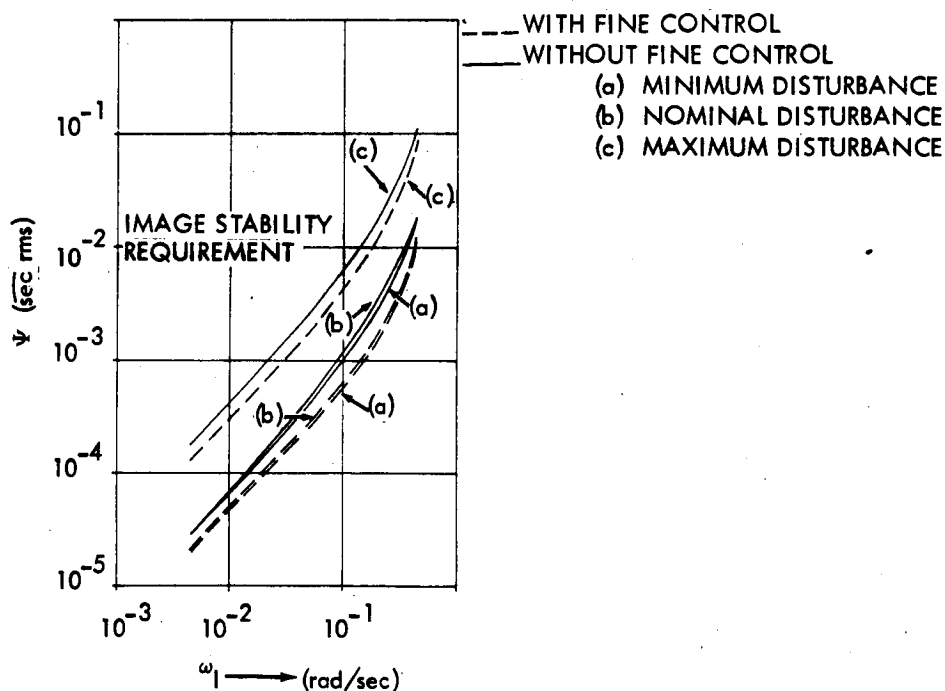


Figure 10.2.2.4.1-7. Image Stability Curves.

Since it contains only low-mass elements, it is possible from a hardware viewpoint to make it a fast responding system. However, for its intended application noise entering with the signal limits the bandwidth to about 5 rad/sec. Thus, for high-frequency disturbances the fine loop contributes little to image stability. This is illustrated in the figure, where for the periodic disturbances, most of which are at higher frequencies, the two sets of curves are close together. For the single impulse, however, only the case without fine control is shown. Fine control, under the assumptions for which these results were derived, would provide perfect stability for the impulsive disturbance. The reason for this is that, after being filtered in the suspension system, the impulse becomes a periodic torque at the isolation frequency, causing the telescope pointing to vary at the same rate but with an amplitude as shown by curve (d) in figure 10.2.2.4.1-7. The fine control loop will follow such a variation with negligible offset.

These results show clearly the advantages of using a low isolation frequency ω_I . What is the lower limit on this parameter? To answer this question, it is necessary to consider the space environment and its effect on the suspension system. If this environment were truly zero-g the telescope could be completely released from the spacecraft during operation, thereby providing perfect isolation and requiring no suspension system as such.

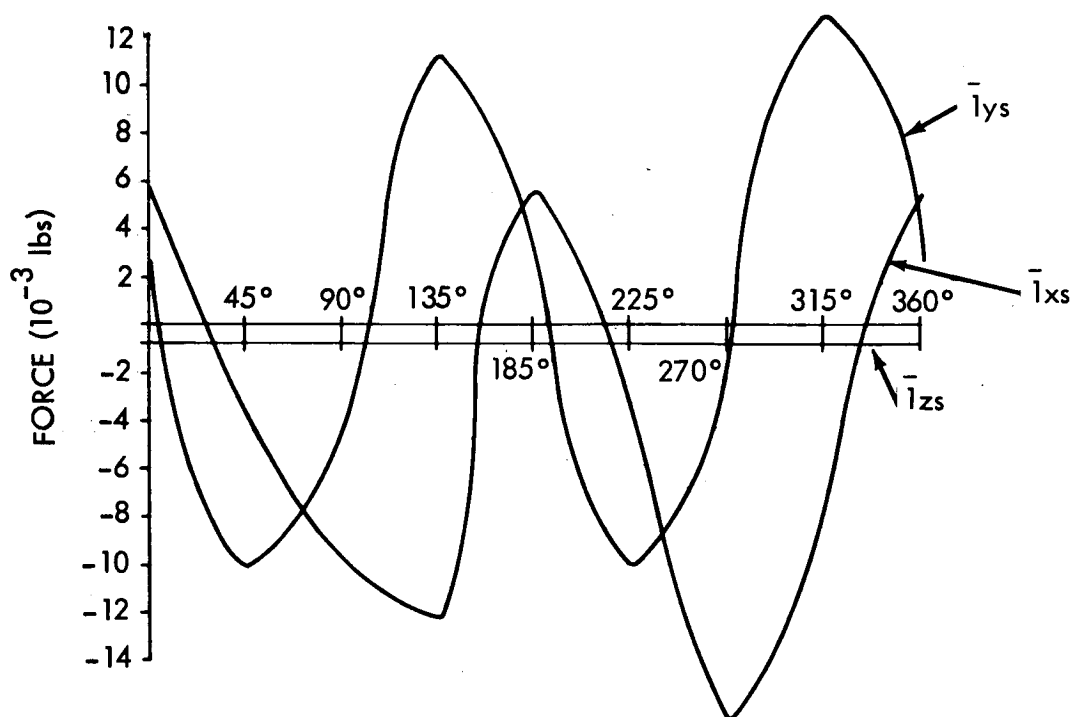


Figure 10.2.2.4.1-8. Orbital Separation Forces - 200 nmi. Orbit.

However, this is not the case; small incremental forces caused by differential gravity, solar radiation pressure, and aerodynamic drag will cause the two bodies to separate over the period of time required to photograph a stellar source.

Typical orbital force curves are shown in figures 10.2.2.4.1-8 and -9. The first figure shows the force components along each of three orthogonal axes, as the telescope-spacecraft configuration is rotated about a particular axis in the spacecraft. The force shown is the net force required to hold the two bodies in their initial relative position. At the initial orientation, which corresponds to the earth-oriented laser experiments operation, the net force is about 6.1×10^{-3} lb. The greatest value, which corresponds approximately to the peak value for the stellar-oriented experiment, is about 21×10^{-3} lb. This is the "static" load which the suspension system must support. The magnitude of this force and its frequency, which includes components at twice the orbital rate, determine the lowest equivalent spring constant that may be used in the suspension system.

The curves of figure 10.2.2.4.1-9 portray the same information for synchronous altitude. The significant feature illustrated there is that the orbital forces are reduced by roughly two orders of magnitude. For the particular application considered here, a synchronous orbit is recommended because of the laser experiments; however, the ultimate application of the suspension technology will probably be at low-earth orbit.

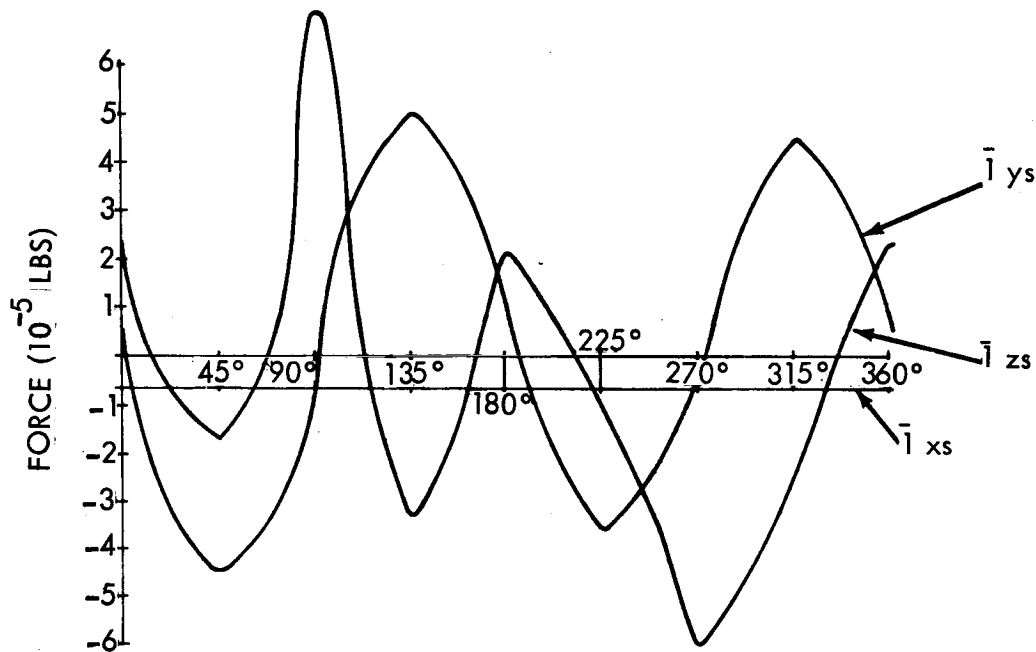


Figure 10.2.2.4.1-9. Orbital Separation Forces - Synchronous Orbit.

10.2.2.4.2 Mechanization

The experiment is divided into four subsystems: the hardmount, the spring suspension, the magnetic suspension, and the damping mechanism. The design of each of these is discussed below.

10.2.2.4.2.1 Hardmount

The telescope is said to be hardmounted to the spacecraft when it has only rotational freedom relative to the spacecraft. To provide the two rotational degrees of freedom required for this experiment, and to provide for suspension of the gimballed telescope, the telescope is mounted in two gimbal rings as shown in figure 10.2.2.4.2.1-1. The telescope is free to rotate inside the inner ring called the gimbal ring, and this ring rotates inside the outer ring, called the suspension ring. Mounted on each of the axes are a torquer, an inductosyn and a data bridge. Because of the small relative angle of rotation (± 2 degrees) between the telescope and spacecraft, and the need to reduce friction, brushless permanent magnet torquers are used. The data bridges couples power and data between the telescope and the spacecraft through flexible conductors designed for minimum restraint torque. The inductosyn measures the relative angles between the telescope and the spacecraft for instrumentation and control purposes.

Also shown in figure 10.2.2.4.2.1-1 are gimbal ring caging devices and a suspension ring caging mechanism. The suspension ring caging mechanism provides for engagement and disengagement of the ring to terminate and initiate, respectively, the soft mode of operation. Consisting of a drogue and cone apparatus, it provides limit stops during the soft mode operation and retracts the telescope from any position.

10.2.2.4.2.2 Spring Suspension

In considering ways to dynamically isolate the telescope from the spacecraft, two competing concepts are immediately apparent. First, the hardmount discussed in the previous subsection may be physically suspended by springs during actual telescope operation. The telescope is free, then, to rotate in the gimbals independent of spacecraft angular motion.

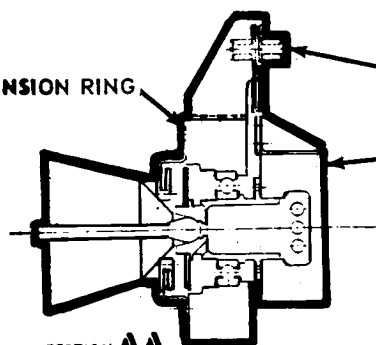
The second method is to suspend the telescope directly with no intervening gimbals. The choice between these two methods, from the viewpoint of performance, depends on the effect of gimbal friction for the first method as opposed to possible spring-caused restraint torques for the second. This choice is not amenable to simple analysis. The first method entails 8 degrees of freedom between the telescope and the spacecraft plus extremely complex control and stabilization systems in both telescope and spacecraft. For the performance levels of concern to this experiment, the usual rigid-body assumptions for the several structural components involved are not valid. The result is that to make a choice between the two methods or to attempt to prove that either of the complex dynamic systems can satisfy the image stability requirements will require the elaborate and expensive simulation scheme planned for the follow-on study phase. Even then, the performance will not have been proven; that will require a flight experiment.

ELECTRO-MAGNETIC
INDUCTION COILS
2 REQD

SUSPENSION RING
REF

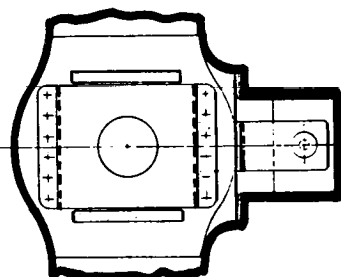
GIMBAL RING TO
SUSPENSION RING
CAGING MECHANISM

GIMBAL RING REF

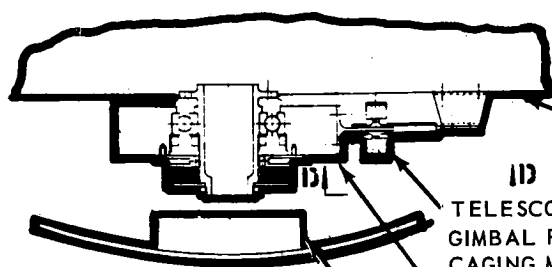
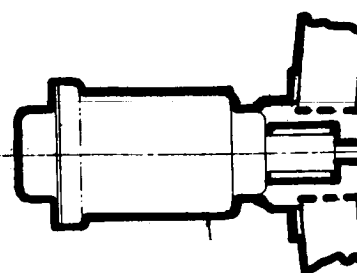


SECTION AA

IRON SPHERICAL SHELL SEGMENT
2 REQD



SECTION CC



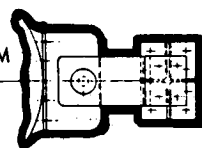
SECTION BB

TELESCOPE REF

TELESCOPE TO
GIMBAL RING
CAGING MECHANISM

GIMBAL RING REF.

SUSPENSION RING REF.



VIEW DD

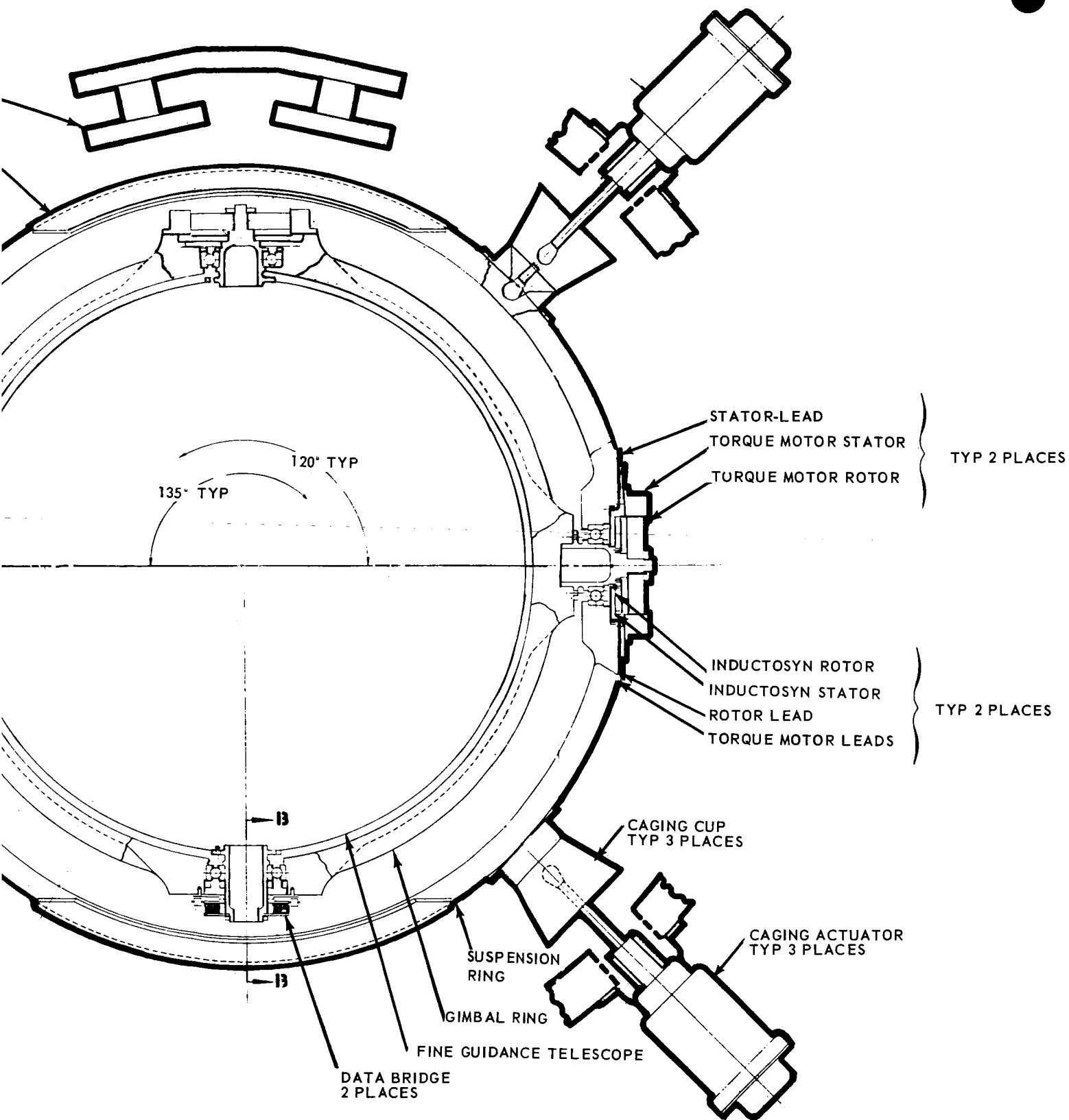
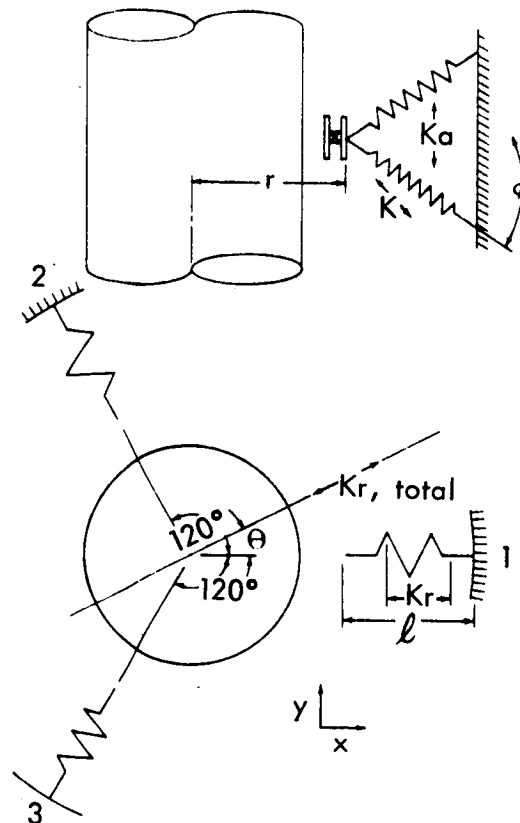


Figure 10.2.2.4.2.1-1. Hardmount Mechanization.

But the simulation is required, of course, to complete the detail design of the experiment.

The analysis here is confined to establishing trade-offs on spring design, looking at non-linear and second-order effects in the suspension and deriving the constraints torques for the second method of suspension. The model of the suspended hardmount is shown below.



Three sets of springs are arranged symmetrically around the suspension ring 120 degrees apart. Each spring set consists of two springs. The angle ϕ is chosen to give equal axial and radial spring constants. As shown in subsection 10.4.2.3, Supporting Analyses, the equivalent spring constant in the radial direction is independent of θ for zero deflections and is $1.5 K_i$ where K_i is for a single spring. The effect of deflection is shown in figure 10.2.2.4.2.2-1, where the ratio of the effective spring constant to that of an individual spring is given as a function of displacement. The increase in spring constant for the maximum required deflection (0.5 inch) is less than 1 per cent. This means that for all the disturbances considered in subsection 10.2.2.4.1 operation of the spring suspension will be essentially independent of the deflection amplitude. That is, for the deflection geometry considered here the compliance of an individual spring pair for pure translation in any radial direction is constant independent of the spacecraft and telescope motions.

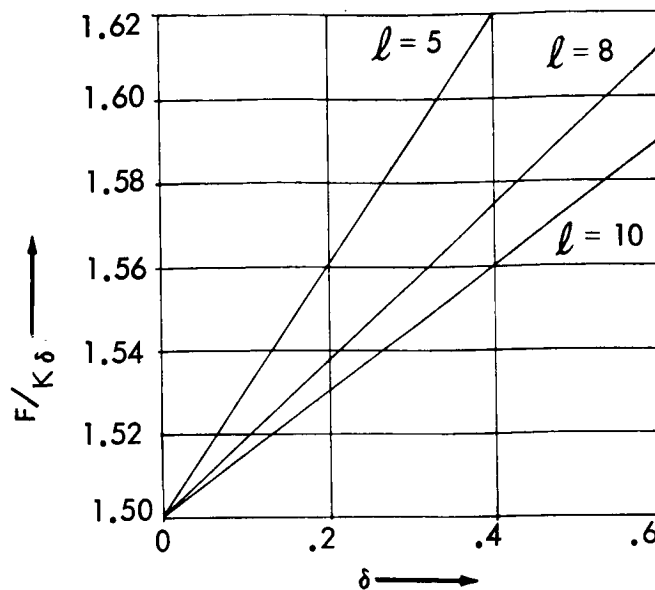


Figure 10.2.2.4.2.2-1 Effect of Deflection on Spring Constant

The rotational constraints introduced by the springs are illustrated in figure 10.2.2.4.2.2-2. Part (a) of the figure is for the case of springs pinned in frictionless pivots so that the only constraints to rotation are imposed by extension of the springs. This is of particular interest for the second method or three axis suspension in which any constraint torque will contribute directly to telescope bias offset.

Part (b) of the figure shows the same parameter for the case of the spring ends being rigidly attached to both the spacecraft and the telescope. The constraint in this case is caused by spring bending, and as can be seen the torque constraints are larger.

More important than the magnitudes, however, is the difference in slope between the the two curves. For the cantilevered springs the torsional spring constant is almost independent of the angle for angles up to a few degrees. But for the pinned end case, the torsional spring constant is highly dependent on the magnitude of deflection, as can be seen from part (a) of figure 10.2.2.4.2-2. Thus, for the small amplitude oscillations discussed in subsection 10.2.2.4.1 and shown in figure 10.2.2.4.1-2, the rotational constraint introduced by the pinned-end case will be entirely negligible, whereas for the cantilevered spring this will not be so. Thus, for the three-axis suspension the springs probably should be mounted in universal pivots to minimize the constraint torques, and for the suspended hardmount case they should be cantilevered to provide a stiffer support.

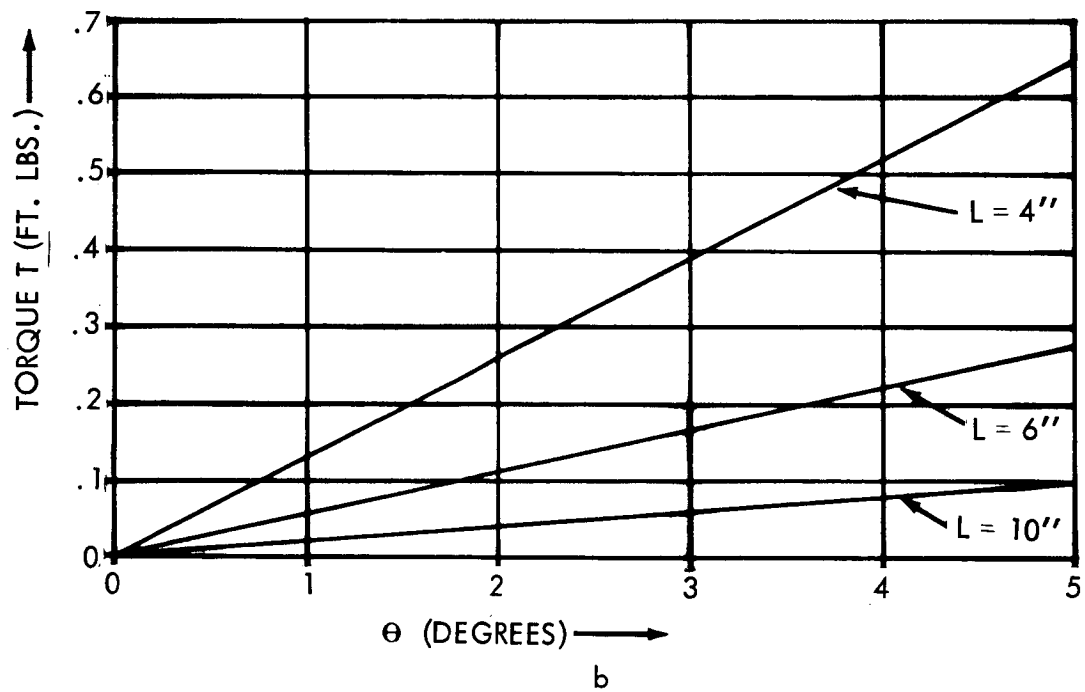
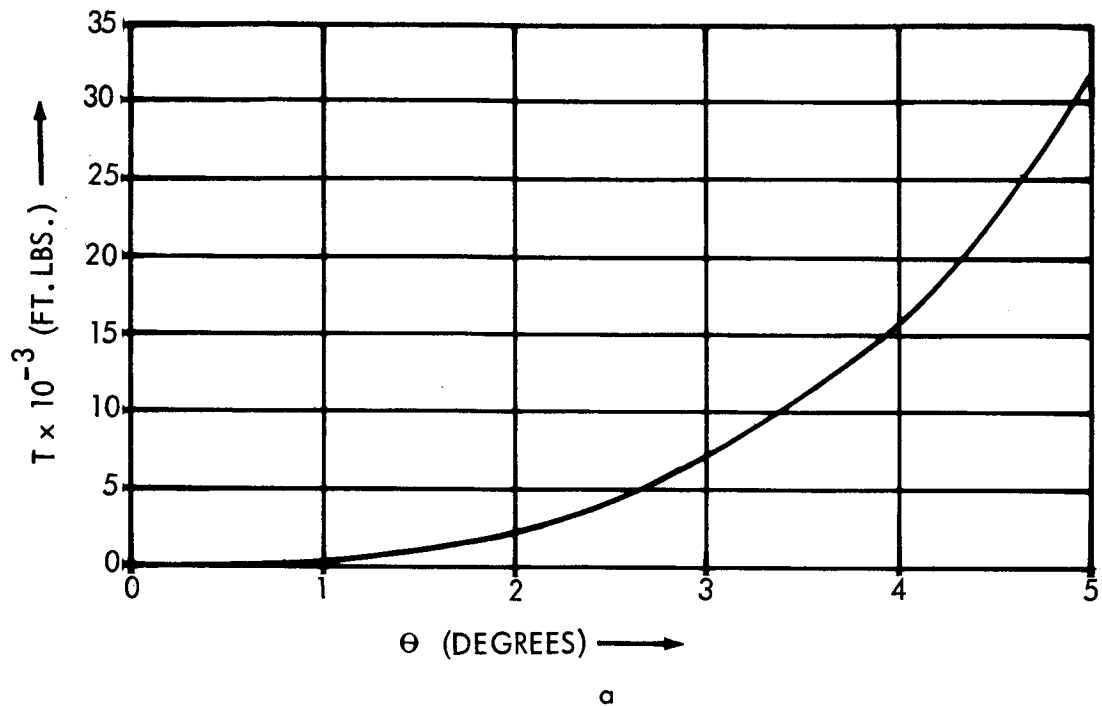


Figure 10.2.2.4.2.2-2 Rotational Constraints of Spring (a) for Pinned End; (b) for Attached End

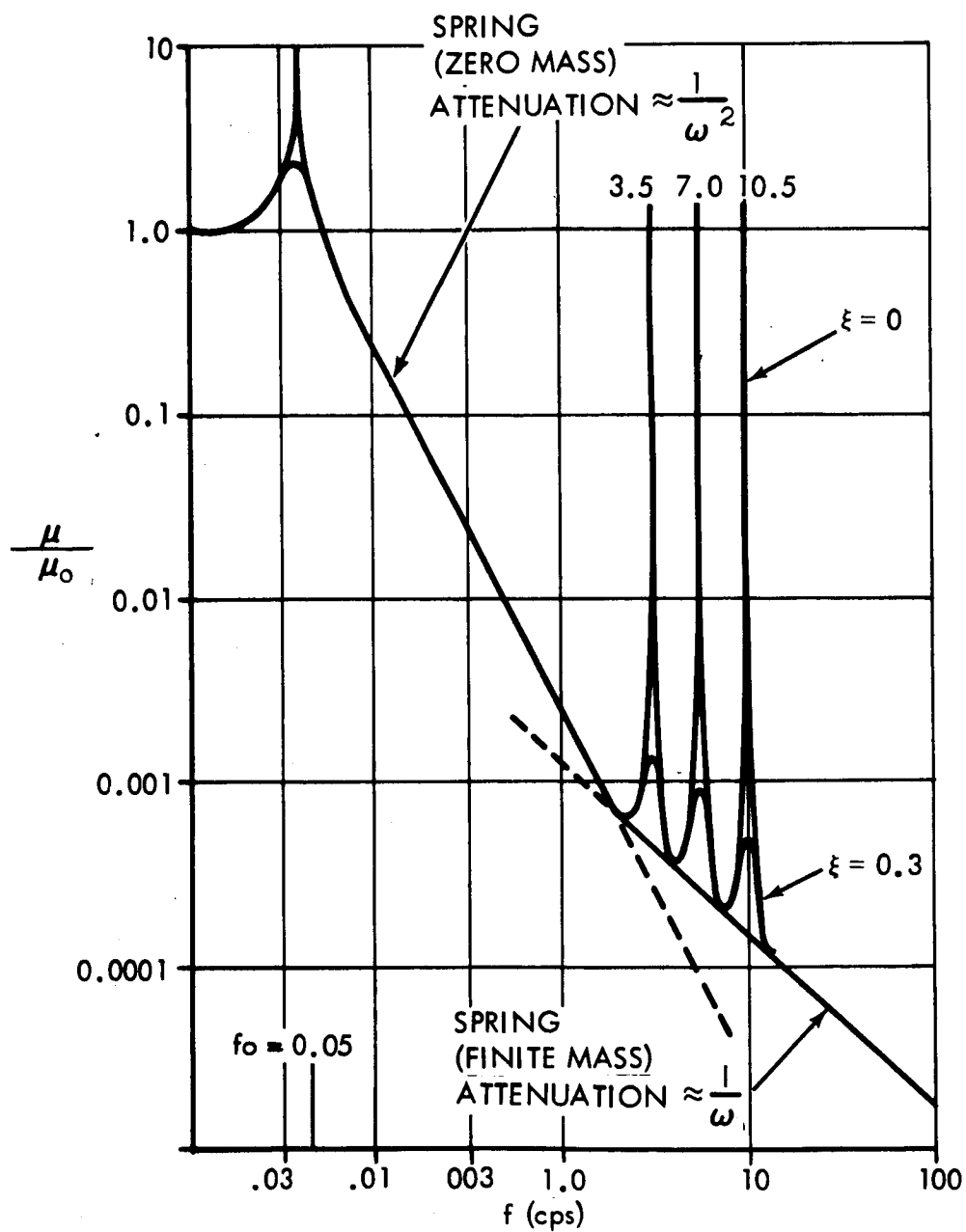


Figure 10.2.2.4.2.2-3. Illustration of Standing Wave Phenomena.

When the forcing frequency becomes large compared to the isolation frequency, the assumption of massless springs is of questionable validity. The effect of a finite mass can cause standing waves to be set up in the springs, and at the standing wave frequencies, and zero damping, no isolation is provided. This is illustrated in figure 10.2.2.4.2.2-3 where the transmissibility of a single degree of freedom system is plotted to show the standing wave phenomena.

Zero damping is assumed and for the case of massless springs the attenuation goes as $1/\omega^2$. The effect of a finite mass is to reduce the slope and introduce the zero-attenuation modes. The effect of damping is to reduce the peaks of these nodes as illustrated in the figure. However, not much damping will be used in this application, and zero-damping case should closely represent the real situation. The standing wave frequencies fall within the spectrum of man-produced excitation. Whether or not they present a real problem must be determined by experiment.

The trade-offs for the design of an individual springs are illustrated in figures 10.2.2.4.2.2-4 and 5. The first figure shows the dependency of the number of turns on the wire diameter for a given coil diameter. These curves are for an isolation frequency of $(1/\sqrt{3}) \times 10^{-2}$ rad/sec. The curves are derived for the suspended hardmount case in which six springs are used. For a coil diameter of 5 inches and 20 turns requires a wire diameter of about 18 mils.

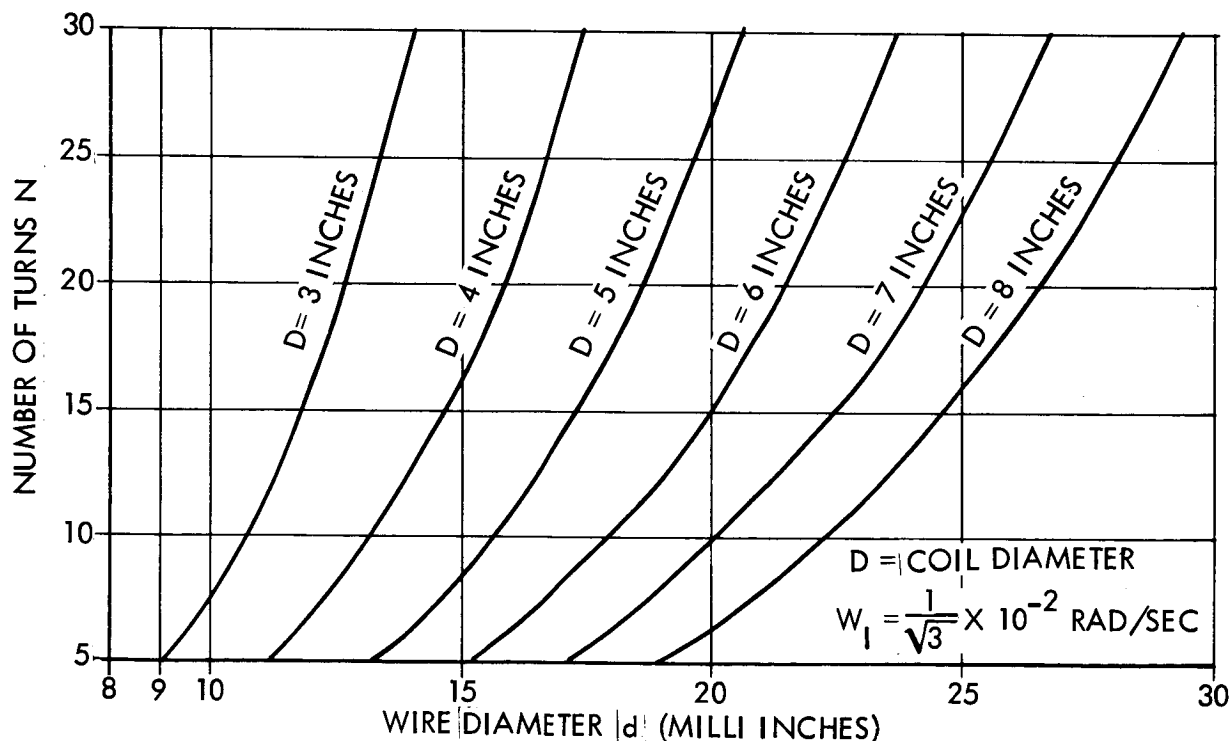


Figure 10.2.2.4.2.2-4. Spring Trade-Off Parameters vs. Coil Diameter.

Figure 10.2.2.4.2.2-5 shows the effect on wire diameter and coil diameter of varying the isolation frequency. For a 20-turn coil of the isolation frequency is increased by a factor of 10 to $(1/\sqrt{3}) \times 10^{-1}$ rad/sec, for example, a 5-inch coil will require about 60 mil wire. The size of the wire is particularly important if power is transmitted through the individual springs, which is under consideration.

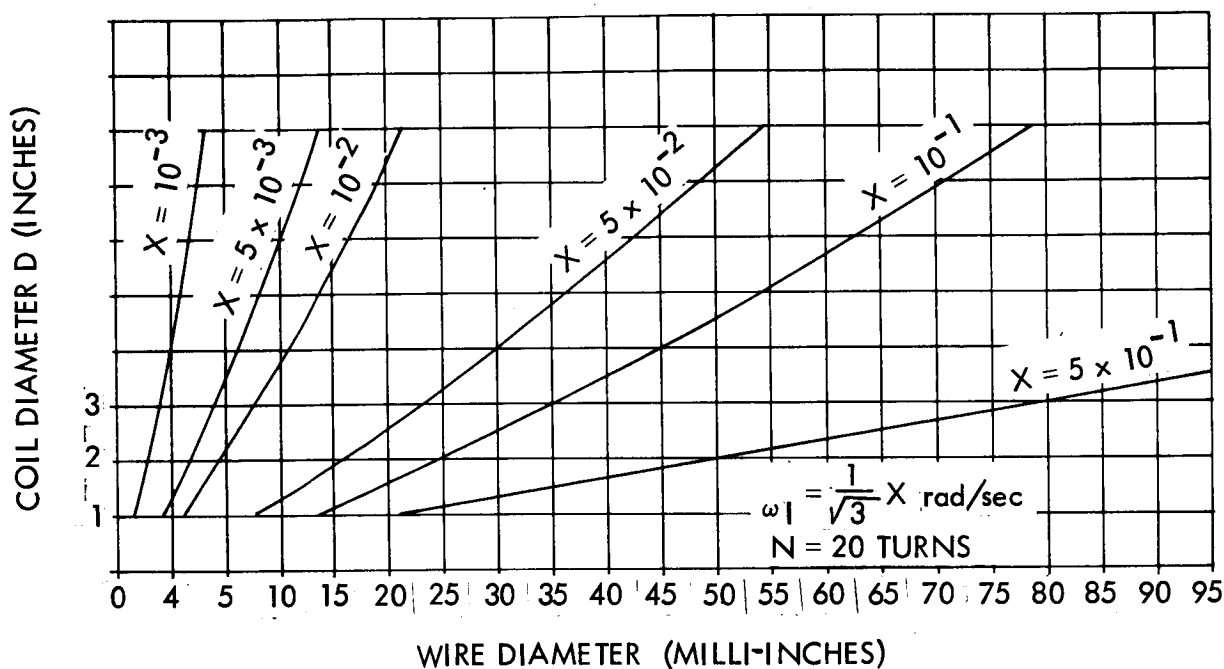


Figure 10.2.2.4.2.2-5. Spring Trade-Off Parameters vs. Isolation Frequency.

The trade-offs for transmitting power through the coils are shown in figure 10.2.2.4.2.2-6. It is assumed that 300 watts power must be transmitted through 6 individual springs. Plotted in the figure is the solid height of the spring as a function of the power loss, with coil diameter as a parameter.

Beryllium copper is used for spring material. For a coil diameter of 5 inches and a power loss of 5 per cent, a solid spring height of 36 inches is required. But by reducing the coil diameter to two inches, a more reasonable figure of 5 inches solid height is obtained. Based on these results it is questionable that power transmission through the springs is feasible for the application requiring very low isolation frequencies.

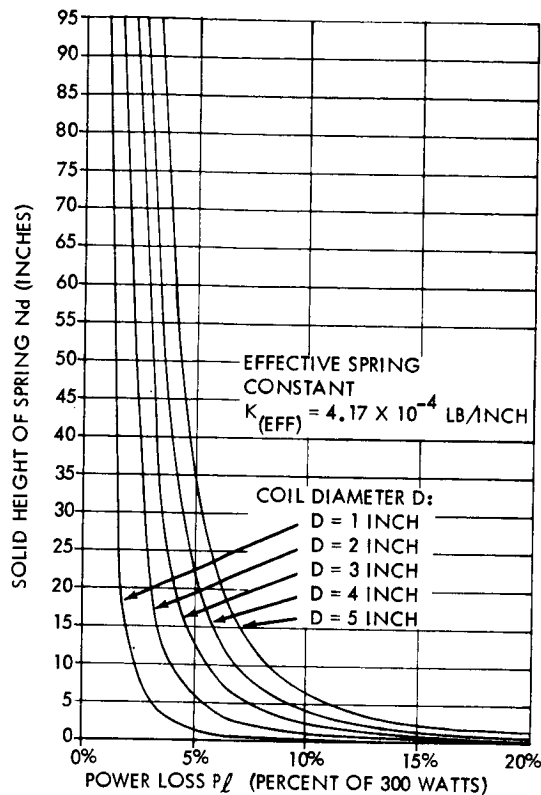


Figure 10.2.2.4.2.2-6. Trade-Off Curves for Power Transmission Through the Springs.

10.2.2.4.2.3 Magnetic Suspension

The magnetic suspension system is a three-axis suspension that utilizes six electromagnets arranged in pairs on three orthogonal axes. Each magnet is analogous to a spring in the three-axis spring suspension. Two types of magnetic suspensions are designed. The first, the A.C magnetic suspension, is the simplest of the active type suspensions, as it makes use of resonance phenomena to produce an inherently stable support without feedback circuitry. The second type of magnetic suspension, the D. C. servo suspension, is considerably more complex, including servo electronics and incorporating two feedback loops for each axis. However, to save weight and power, the two types are closely integrated. Both use the same magnetic cores and the A. C. suspension is used as a position sensor for the D. C. operational mode. Each electromagnet consists of a coil, located on the spacecraft structure, and an armature located on the suspension ring, with the two separated by a nominal air gap.

The A. C. electromagnet coils are supplied from the 400-cycle primary spacecraft power system through capacitors in the line. The values of the capacitors are chosen to give proper tuning in the individual series LCR circuits. Each electromagnet pair on a given suspension axis operates independently of the electromagnets on the other two axes.

The concept of the A. C. magnetic support derived from experimental work done at MIT in 1953 on a suspension system for a floated gyroscope.⁽¹⁾ Principle of operation is as follows: the A. C. circuit, illustrated in figure 10.2.2.4.2.3-1, is tuned for operation at the second half-power point of the resonance curve shown in figure 10.2.2.4.2.3-2.

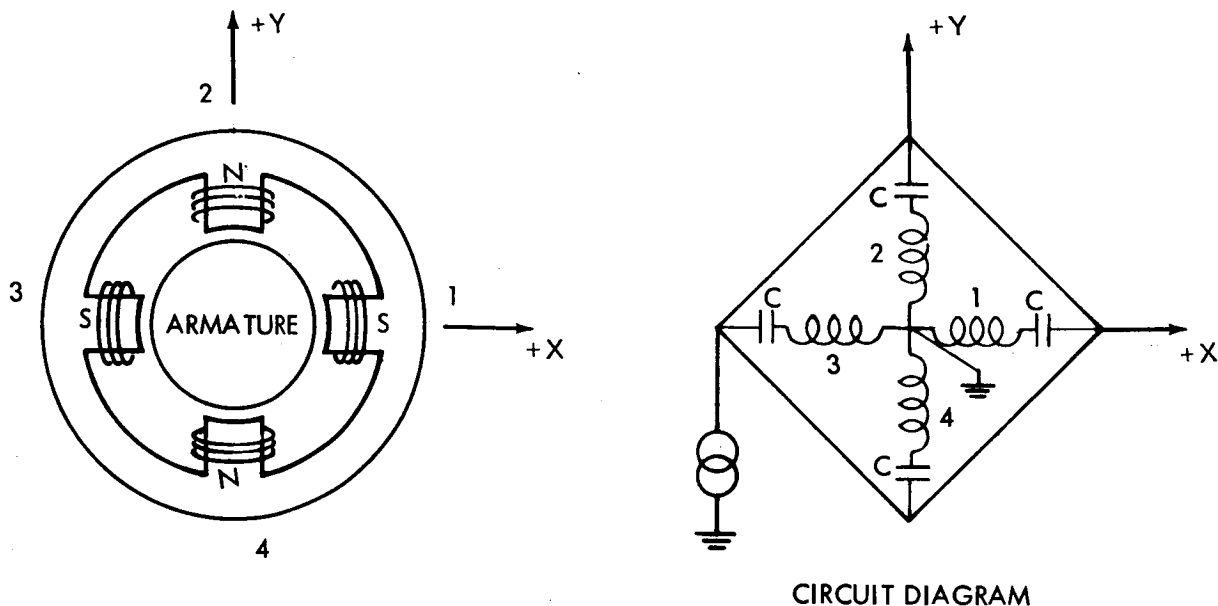


Figure 10.2.2.4.2.3-1. Illustration of AC Resonant Magnetic Support.

A change in telescope position causes the current through the coil to change, because of detuning, and this produces a restoring force. For example, let the rotor be displaced along the positive x-axis. This increases the inductance in coil No. 1 and decreases that in coil No. 3. The current decreases with increasing inductance and increases with decreasing inductance. Thus, the force on the rotor due to coil No. 1 decreases and that due to coil No. 3 increases, producing a net radial restoring force.

(1) Gilinson, P. J. Jr. and Frazier, R. H. "Electromechanical Components". Air Space and Instruments McGraw-Hill, 1963

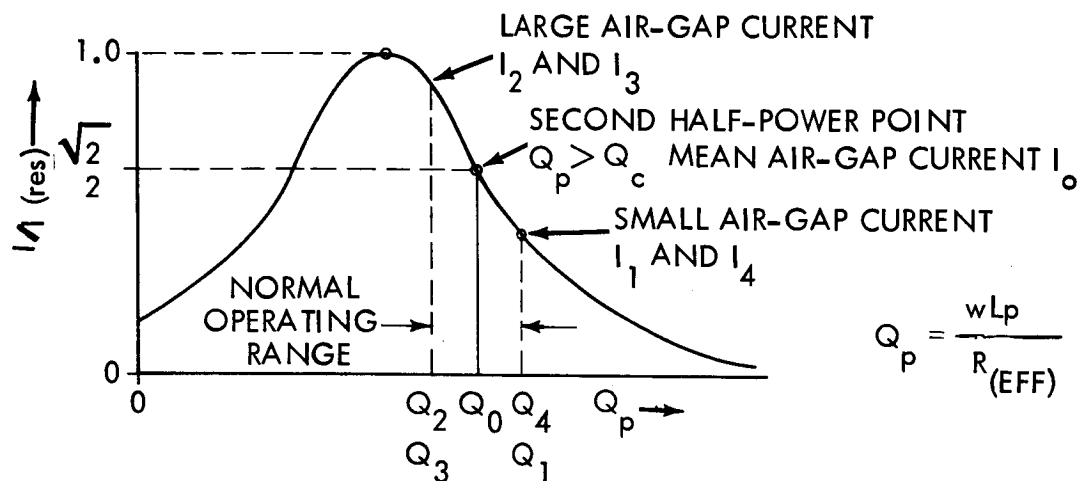


Figure 10.2.2.4.2.3-2. Resonance Curve.

A typical force-displacement curve is shown in figure 10.2.2.4.2.3-3. Normalized restoring force is plotted against normalized displacement and the slope changes sign when the incremental displacement is greater than 20 per cent of the total airgap.

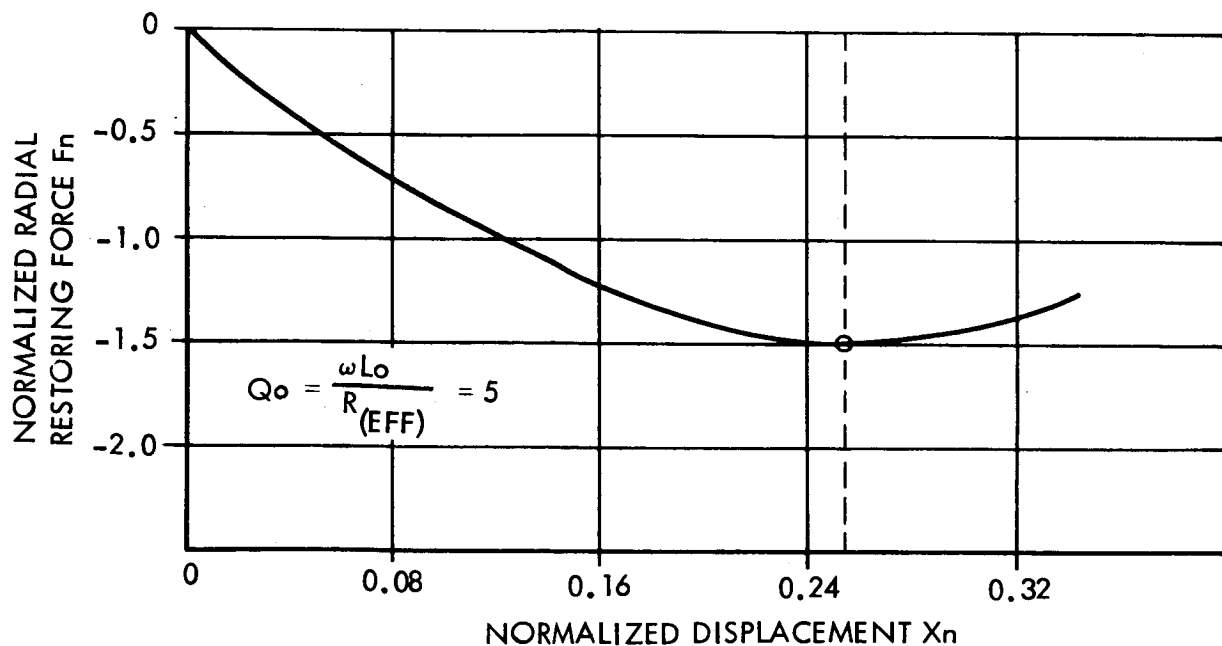


Figure 10.2.2.4.2.3-3. Typical Force Displacement Curve.

Since the worst impulsive moment on the spacecraft can cause a relative translation between telescope and spacecraft of 0.5 inch, as discussed in subsection 10.2.2.4.1, the airgap should be at least 2.5 inches. This is illustrated in figure 10.2.2.4.2.3-4, where the effective incremental spring constant (lbs/inch) is plotted against displacement of the telescope from the nominal position. In practice, the telescope/spacecraft relative translation will be physically limited to less than the value at which dynamic instability occurs. To obtain low weight and low power consumption for a given force-displacement relation, one may set up a relative-proportioned magnet and armature unit that may be scaled up or down to find the optimum size.

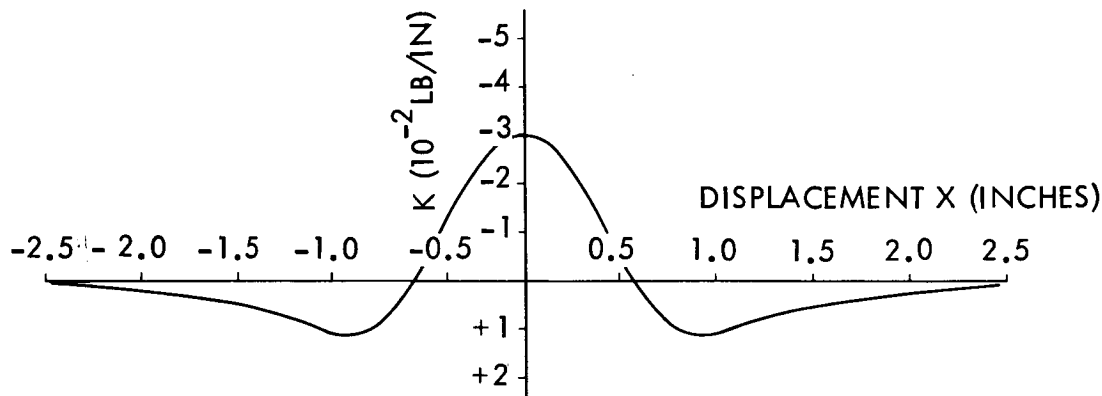


Figure 10.2.2.4.2.3-4. Incremental Spring Constant vs. Displacement.

For a magnet having a yoke thickness, total length, pole face arc length, and height in the ratio of 1:12.5:4:2, the power loss (without copper loss) and the total weight are given as a function of yoke thickness in figures 10.2.2.4.2.3-5 and 6. Curves for both 60 Hertz and 400 Hertz are given.

It is seen that the power losses are at a minimum for $t = 2.83$ inches. This value of t is independent of the frequency; however, the power loss P at any value of t is proportional to the square of the frequency. At $f = 60$ hz, P increases slowly from its minimum value as t is decreased below 0.50 inches, at which point it turns up sharply for a small change in t . The weight W , on the other hand, increases very rapidly as t becomes larger than 0.50 inches. The best trade-off values of P and W are thus found for t close to 0.50 inches.

Figure 10.2.2.4.2.3-7 shows the trade-off curves for the electrical design. Good values for N and R , the number of turns/coil and the resistance coil, respectively, are obtained for high frequency and low voltage. The current per coil, however, has good values for low frequency and higher voltage. Since the power losses of the winding are more strongly affected by the current, and since the weight increase of the windings at the higher voltage is not large, the trade-off here is for the lower frequency (60 hz) and the higher voltage of 115 volts.

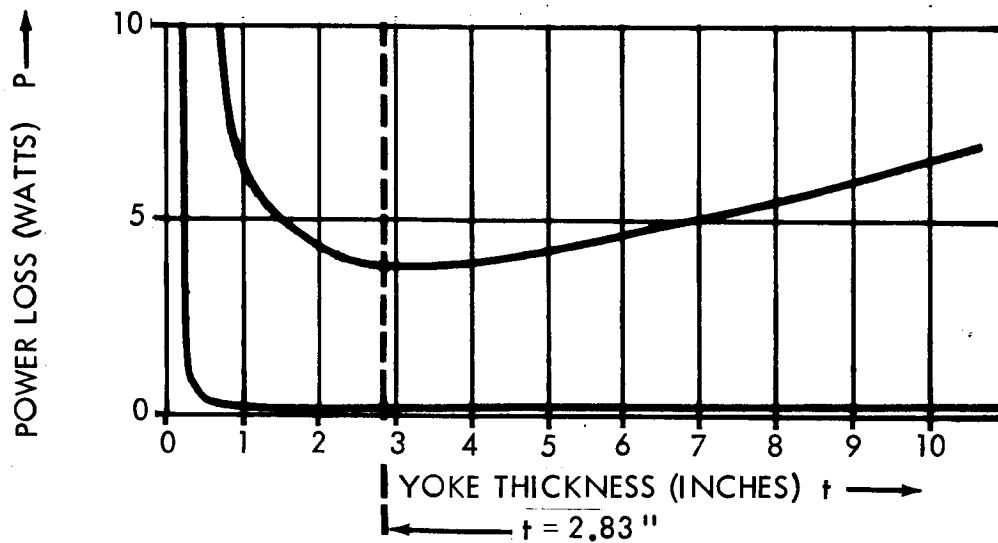


Figure 10.2.2.4.2.3-5. Power Loss vs. Yoke Thickness.

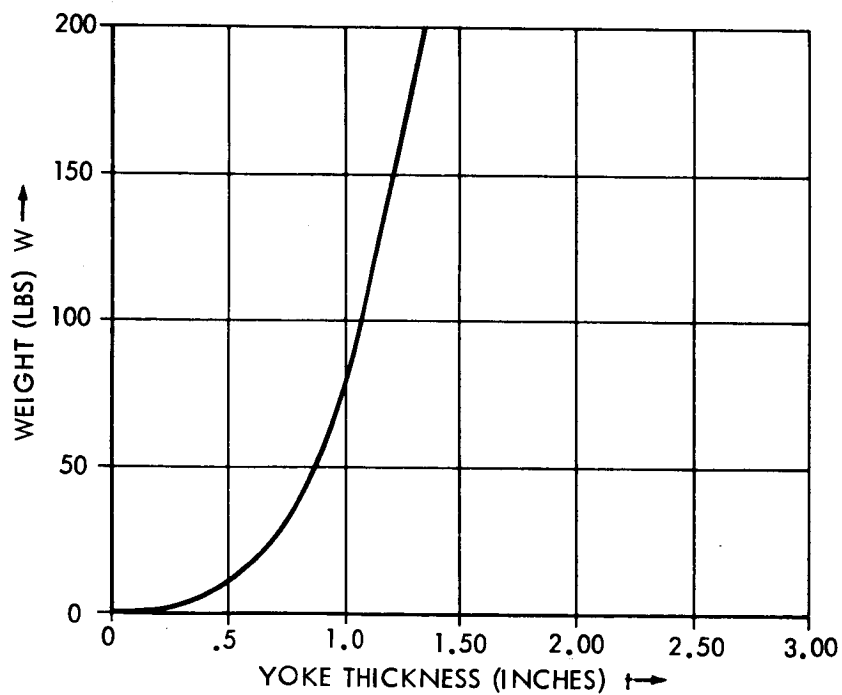


Figure 10.2.2.4.2.3-6. Weight vs. Yoke Thickness.

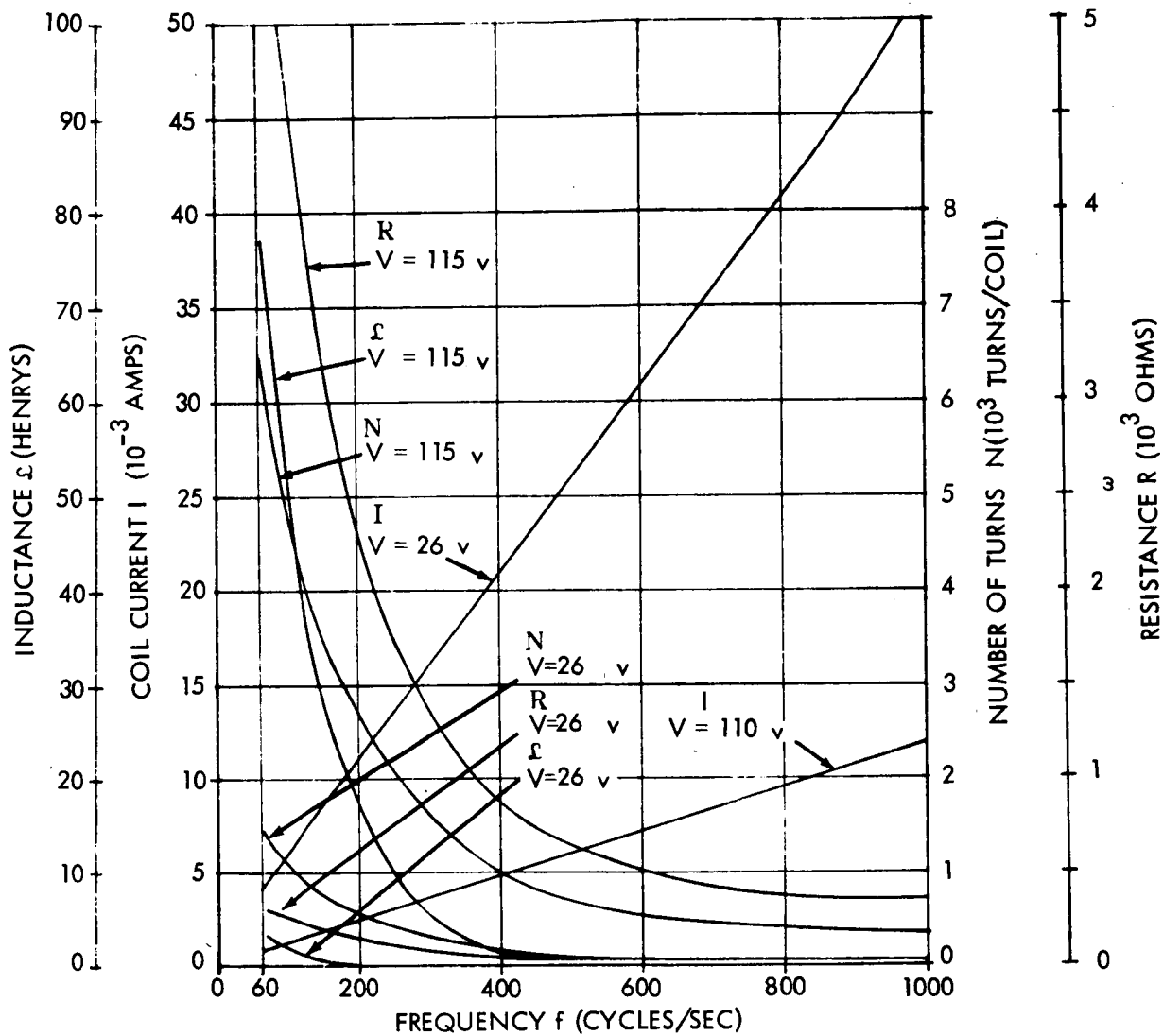


Figure 10.2.2.4.2.3-7. Trade-Off Curves for the Electrical Design.

A sketch of the proposed magnet design is shown in figure 10.2.2.4.2.3-8. The support system consists of six such magnets and armatures on three mutually orthogonal axes. One pair is on the diameter of the telescope and the other two are set on the ends of the telescope so that the telescope is surrounded by six symmetrically placed magnet-armature units. The optical axis is in the same plane and makes an angle of 45° with the axes of the units set on the ends of the telescope.

The magnets have a flat, rectangular yoke and flat, square pole enlargements. The pole cores have a square cross-sectional shape. Since the magnet must also have direct current windings for the D.C. servo support system, weight and D. C. power consumption considerations dictate a small core cross-section. By installing polar enlargements on the ends of the poles, the core cross-section is substantially reduced without seriously affecting the air-gap flux density and the effective pole face - air-gap distance relationship.

The yoke is flat and broad so that the support system will fit in the telescope well and still have a relatively large yoke cross-sectional area for low power losses. The armature beneath each magnet is a thin spherical shell with the sides cut square as seen from the top. The origin of the shells' radii is at the center of mass of the telescope. This shape of the armatures is the result of the trade-off between low weight, power loss requirements, and space limitation versus the problem of allowing limited rotation to the telescope without generating significant restraint torques. This is further discussed below:

For this configuration, the calculated values of the relevant parameters are:

$I = 6.24 \times 10^{-3}$ amps, current in coil

$N = 12,700$ turns/coil, No. 36 A.W.G.

$R = 7290$ ohms/coil

1.44 lb/coil, copper weight

75 lb/unit, iron weight, one magnet and armature

0.284 watts/coil copper power loss

0.35 watts/unit iron power loss

0.92 watts/unit total power loss/unit

The coils are of square-shaped, paper section enameled copper wire construction. The magnet material is 29 gage (14 mil) transformer 72 (M-19) hot rolled silicon steel sheet.

Figure 10.2.2.4.2.3-9 shows the resonance curve with the operating point and range for this particular configuration. The slope of the resulting force curve is shown in figure 10.2.2.4.2.3-4.

EITHER CENTER LINE
MAY BE PARALLEL TO
THE OPTICAL AXIS
WITHOUT DISTURBING
THE MAGNETS OPERATION.

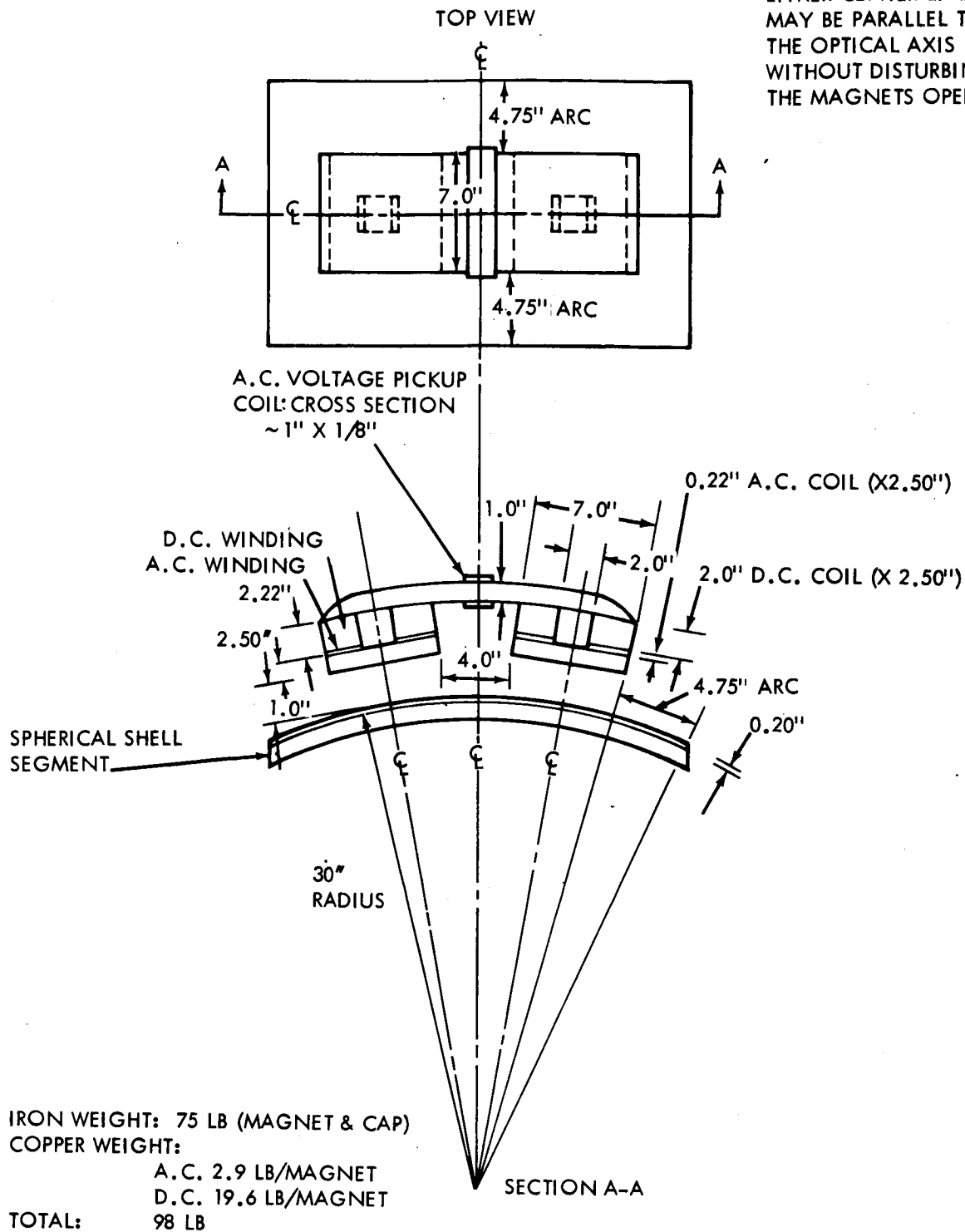


Figure 10.2.2.4.2.3-8. Sketch of the Proposed Magnet Design.

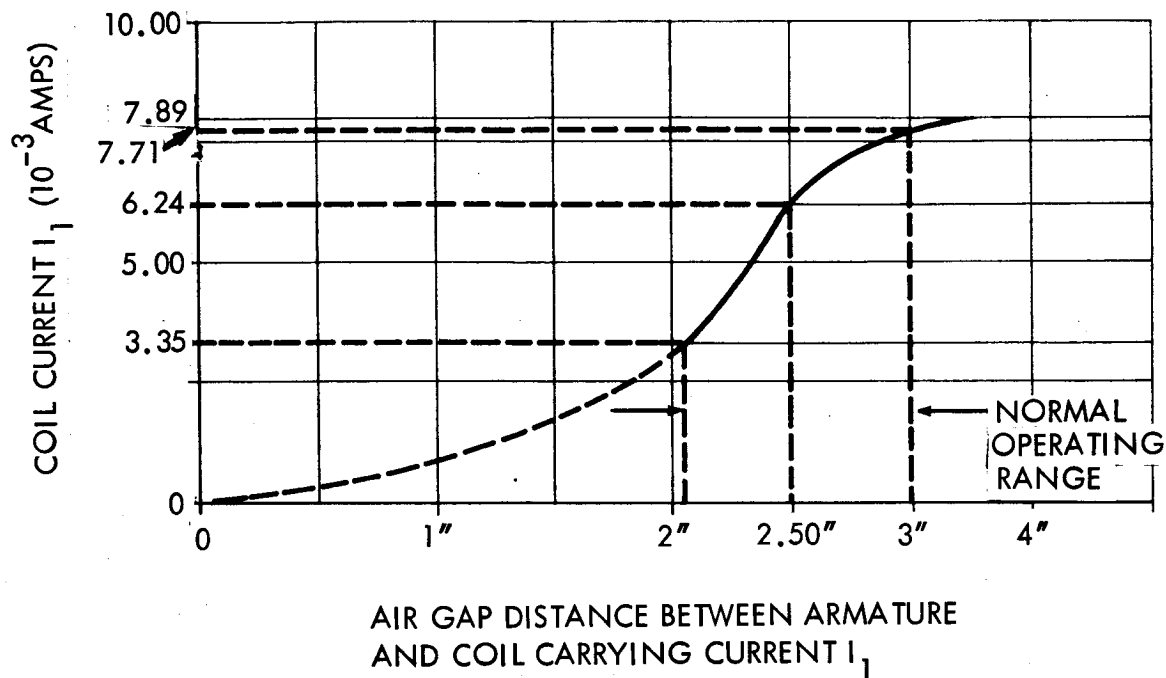


Figure 10.2.2.4.2.3-9. Air Gap Distance Between Armature and Coil Carrying Current.

Due to the large air gap, there is significant fringing of the air gap flux. If the armature is displaced to one side, the symmetry is broken and the fringing flux around the magnet will produce a net torque on the telescope. This torque may be reduced by shaping the armatures as spherical surfaces that subtend large arcs relative to the pole forces. This is expensive, however, in terms of weight penalty. In practice, the armatures should be large enough to meet a specific restraint torque requirement, and no larger.

It is not practical to develop analytical expressions for the restraint torque. However, for a particular configuration rough estimates can be made. Using one technique developed in 10.4.2.4, Supporting Analysis, the estimated restraint torque for the proposed design is on the order of 10^{-3} ft-lb. This exceeds the specified requirement, but in practice good estimates must be obtained experimentally. If laboratory tests show that the restraint torque is too great, then the armature areas will have to be increased accordingly.

The D. C. magnetic suspension is a three-axis suspension that uses the same electromagnets as the A. C. suspension. It differs from the A. C. suspension principally as follows:

- a. For stability, a position sensor must be provided. The A. C. suspension is used for this purpose.
- b. A much greater force capability is provided.
- c. An accelerometer loop is added for purposes of nulling the disturbances.

The concept of the servo suspension is illustrated in figure 10.2.2.4.2.3-10 where a single-axis schematic diagram is presented. Actuators positioned on each of the three independent suspension axes control both the position of the telescope relative to the spacecraft and the acceleration of the telescope. Position is controlled through a low-gain position loop, which acts as a soft suspension, and acceleration controlled through the accelerometer loop, which acts to artificially increase the telescope inertia.

The A. C. resonant circuit is used as a position sensor in the D. C. servo suspension mode. Pick-off coils are provided for this purpose on each electromagnet, as illustrated in figure 10.2.2.4.2.3-10. The sensitivity of the circuit is illustrated in figure 10.2.2.4.2.3-11 where the A. C. error current as a function of displacement is plotted. The amplitude of this current which is plotted in the figure, determines the scale factor of the sensor. The phase of the voltage compared to the reference, determines the sense of the error.

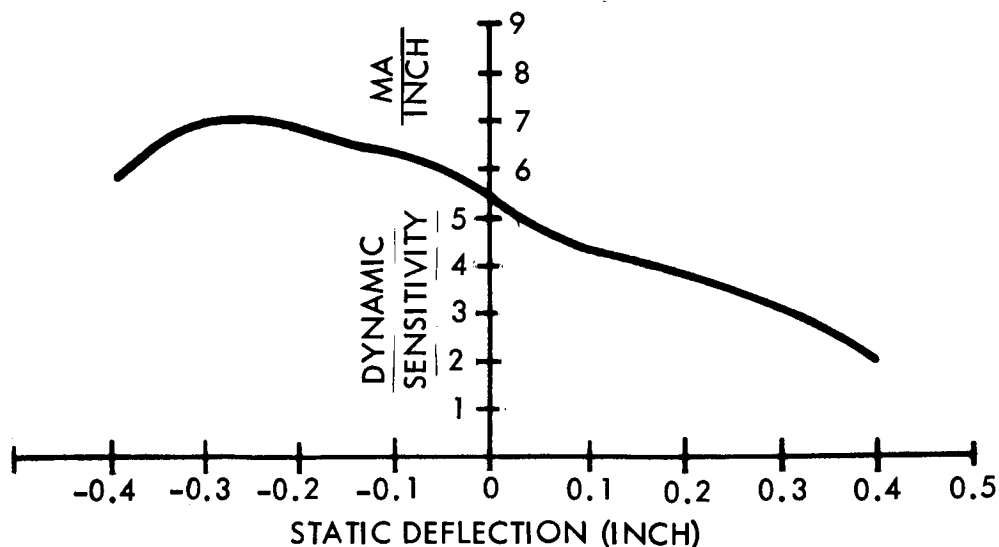


Figure 10.2.2.4.2.3-11. Sensitivity of the Error Sensor.

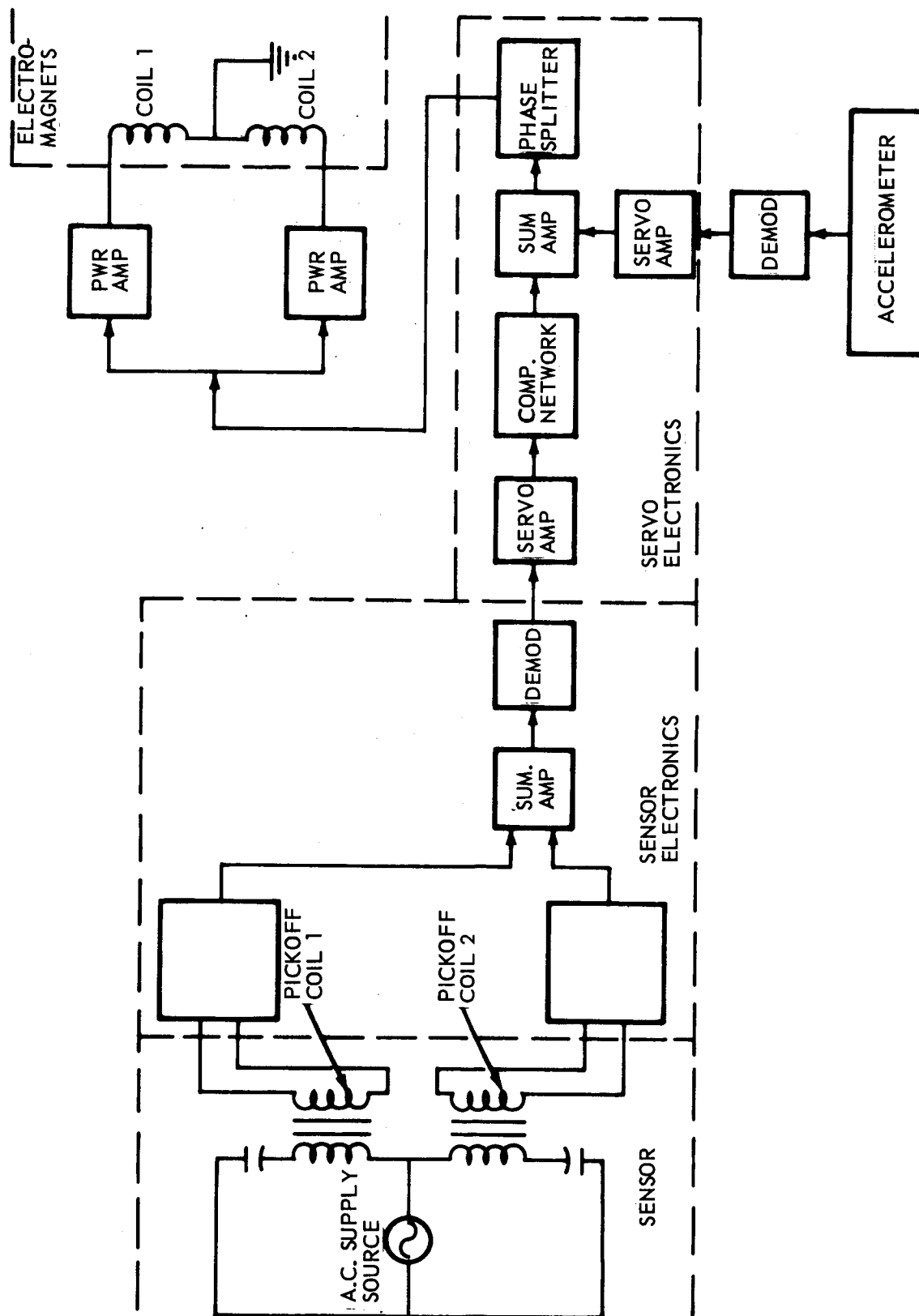


Figure 10.2.2.4.2.3-10. Single-Axis Schematic Diagram of the Servo Suspension System.

Actuation is provided by D. C. windings on the electromagnets. The actuators are designed to produce a specified force across the air gap with minimum weight and power requirements. Figure 10.2.2.4.2.3-12 shows how the power dissipation varies with the weight of the copper in the D. C. windings. Since the D. C. and A. C. windings are placed on the same iron cores, the choice of size and shape of the core was made to minimize the A. C. core losses, leakage and fringing fluxes, as well as the weight of the iron, copper and the power requirements.

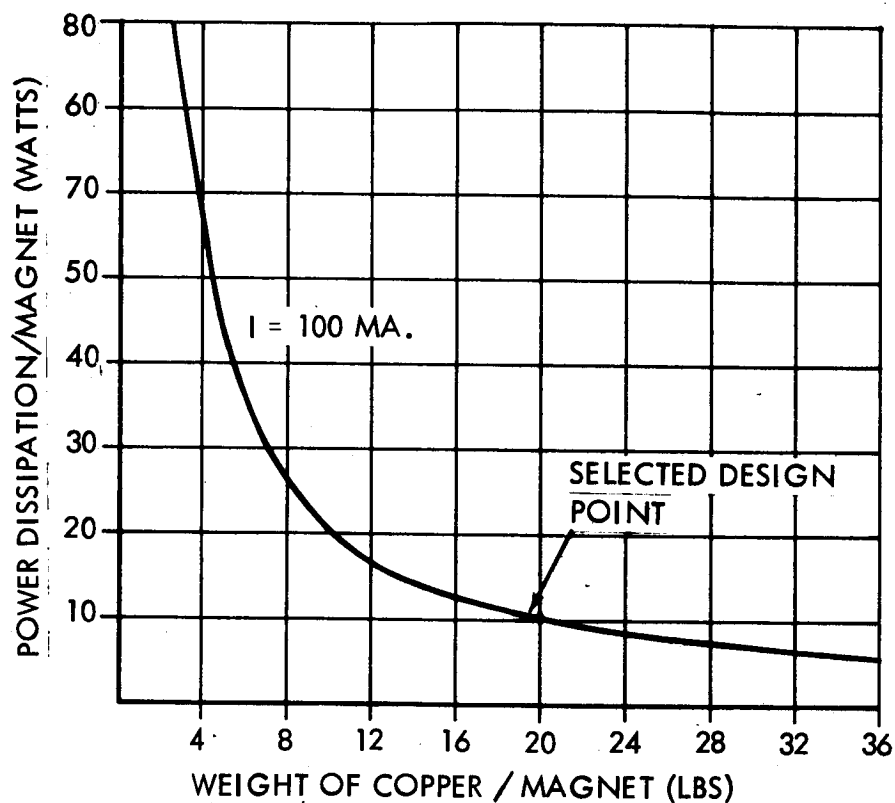


Figure 10.2.2.4.2.3-12. Power vs. Copper Weight for the D.C. Windings.

Table 10.2.2.4.2.3-1 shows the design data and requirements for the D. C. magnets.

TABLE 10.2.2.4.2.3-1
ACTUATOR DESIGN DATA

Requirements

Normal Force/magnet	2 lb
Normal air gap length	2.4 in.

Winding Spec.

	D. C.
Number of turns/coil	9,550
A. W. G No.	26
Resistance/coil	530 ohms
Copper weight/coil	9.8 lb
(Copper) power dissipation/coil	5.3 watt
Gross cross-sectional area of coil	2 x 2.5 in.
Winding space factor	0.538
Normal current/magnet	0.1 amp

A simplified servo block diagram of the suspension system, for a single axis is shown in figure 10.2.2.4.2.3-13. The actuator dynamics are represented by the L/R time constant of the coil which is on the order of 0.1 sec. By closing a high gain accelerometer loop around the actuator and telescope mass dynamics, the effect of the actuator dynamics is rendered negligible and the mass of the telescope is effectively replaced by the inner loop gain. The primary advantage of the servo suspension system is that by use of the accelerometer loop the acceleration response of the telescope to external forces which by pass the suspension can be reduced by the ratio of telescope to accelerometer gain. All the other suspension systems, on the other hand, are ineffective against all but the very slow (orbital forces, for example) disturbances. The problem with using the accelerometer in this way is that noise in the accelerometer, multiplied by the high gain, will appear as a disturbance input to the telescope. Thus, although the static gain of the loop can be made large, the bandwidth must be severely limited. The approach used here, then, is to design the outer loop to give the desired transmissibility to disturbances entering through the suspension system, and design the inner loop to give the desired response to external disturbances, but within the constraints of practical accelerometer noise levels.

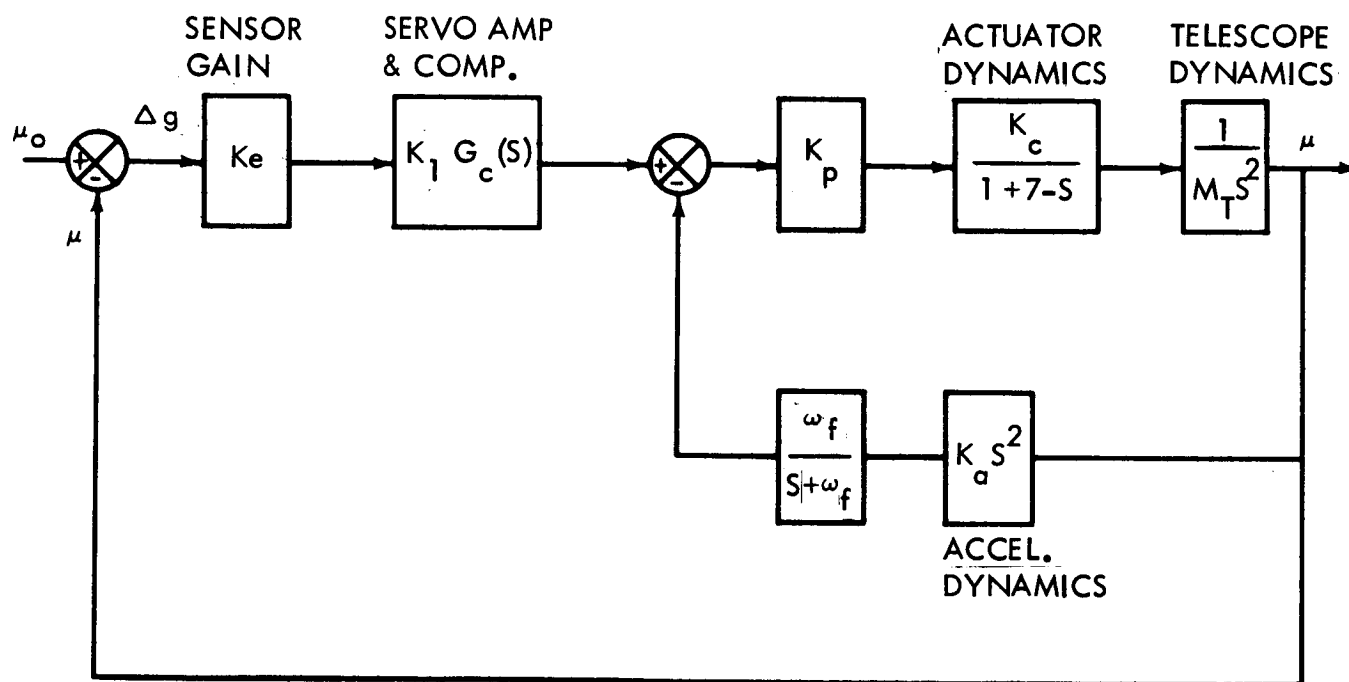


Figure 10.2.2.4.2.3-13. Single-Axis Block Diagram for the Servo Suspension System.

The purpose of the compensation block in the inner loop is to give the desired damping to the transmissibility characteristic. As shown in figure 10.2.2.4.2.3-14a, a simple RC lead network will provide the required damping, which is assumed to be 1 per cent of critical. The system is stable for all values of gain. The nominal operation point is very close to the origin where

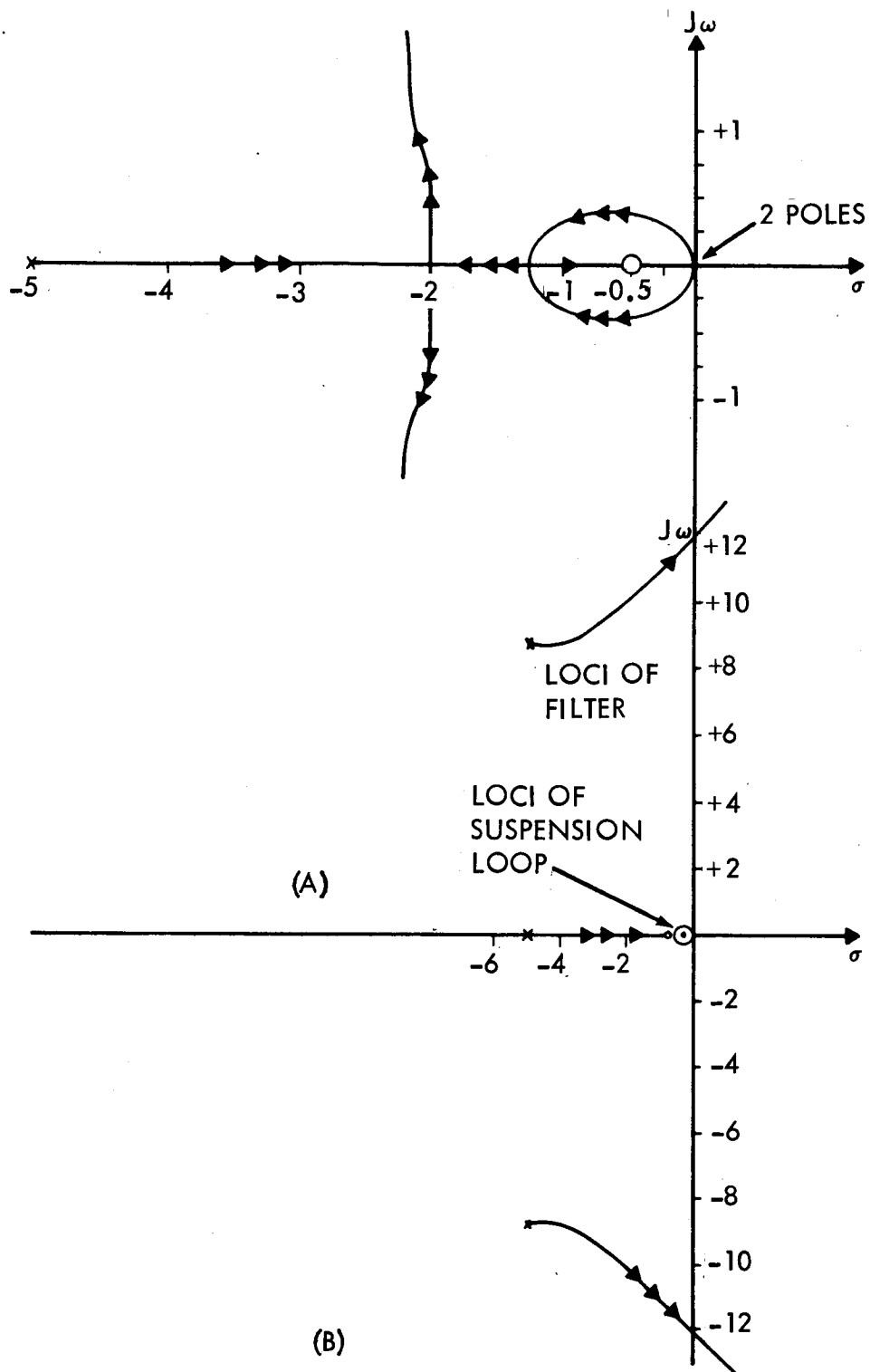


Figure 10.2.2.4.2.3-14. Root-Loci of the Servo Suspension System (a) Without Filter; (b) With Filter.

the gain, the damping and the natural frequency are all small. However, part of the experiment operational procedures (see subsection 10.3) involves varying these parameters over a wide range and evaluating the resultant isolation performance. The effect of the root loci of including the filter in the accelerometer loop is shown in figure 10.2.2.4.2.3-14b. The filter cut off frequency, which is chosen on the basis of a trade-off between the acceleration response to an input force disturbance and the disturbance contributed by accelerometer noise, is set at 0.1 rad/sec. The effect of the filter and actuator dynamics is to contribute a real zero and two complex poles to the open-loop transfer function, but for the levels of gain considered here, the effect on the transmissibility of the suspension system is virtually nil. The gain at which system goes unstable, for example, is very high.

The effect of the filter on performance of the inner loop is to increase the response time of that loop. Without the filter, the acceleration response to a disturbance is reduced immediately (neglecting the dynamics of the actuator and accelerometer itself) by the ratio of accelerometer gain to telescope mass. With a filter time constant of 10 seconds, the time required is about 0.1 second for an accelerometer gain to mass ratio of 100.

10.2.2.4.2.4 Damping

The entire stabilization system, made up of the telescope mass, the telescope stabilization system and the suspension system must be properly damped. Several ways are available to do this. First, the telescope stabilization system will have rotational damping inherent to that system. Due to coupling between telescope rotation and translation, the translational motion of the suspension system will also be damped, and the damping available by this means may be adequate. However, it is desirable to include damping in the suspension system itself rather than depend on rotational coupling which is a function of the telescope and suspension geometry. Therefore, as part of the present study, independent means of damping each of the suspension systems were investigated.

First, alternate approaches to mechanizing a damper were investigated. These included anelastic materials damping, as would be inherent to lossy springs; viscous damping; coulomb friction damping; eddy current damping; and magnetic hysteresis damping. All but the magnetic hysteresis damper were rejected, for one reason or another. The problem with anelastic material damping is that for materials compatible with the space environment, not enough energy can be dissipated per cycle to achieve an effective damping ratio. The best damping reported for metal, for example, was for a cadmium plated wire of 7 mills diameter stressed in torsion to 200 psi with a one-hour period.⁽¹⁾ Ten per cent of the energy was dissipated per cycle, or about 65 per cent in ten cycles. Though this performance is promising, not enough information is available on which to base a design. Therefore, it was decided not to depend on hysteresis damping in the springs, although this would be highly desirable for the spring suspension.

(1) Fischell, R. : Passive Gravity-Gradient Stabilization for Earth Satellites. Torques and Altitude Sensing in Earth Satellites, Academic Press (New York) 1964. pp. 13-30.

Coulomb friction damping was rejected because of the difficulty in controlling its erratic behavior at the low frequencies and force levels involved, and danger of interference with the weak suspension systems. Eddy current damping at low frequencies, requires large volumes of magnets and conductor materials.(2)

Magnetic hysteresis damping, however, does appear feasible for our application. Based on the principle of converting mechanical energy to heat through change in magnetization of magnetically hard material, magnetic hysteresis damping (energy per cycle) is independent of frequency and therefore well suited to the suspension application.

One mechanization of the magnetic hysteresis damper is shown in figure 10.2.2.4.2.4 -1. A hard ferromagnetic disk is suspended by a torsion wire in a permanent magnet field. The housing is attached to the spacecraft through a pulley arrangement. Relative translation between telescope and spacecraft causes the disk to rotate in the magnetic field, which dissipates the driving energy into heat.

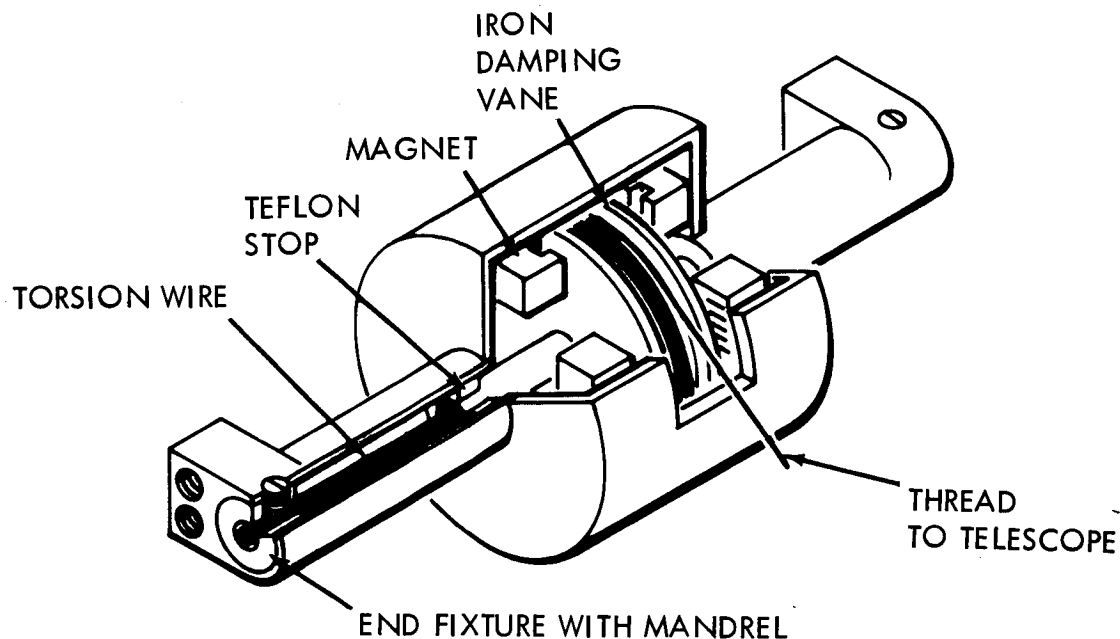


Figure 10.2.2.4.2.4-1. Sketch of the Proposed Damper.

(2) Page 136, Magnetic Hysteresis Damping for Gravity Gradient Stabilization by G. S. Reiter, J. P. O'Neil, and J. R. Alper TRW Systems in "Symposium on Passive Gravity Stabilization". Ames Research Center, Moffett Field, California, May 10-11, 1965.

Stresses in the wire at the lower bound on torsion pivot at about 5×10^{-6} ft-lb/rad. The fact that the torsion spring constant is in series with the suspension system sets an upper bound of about 10^{-3} ft-lb/rad for this application.

The damper shown is being breadboarded in the laboratory as a part of the pre-requisite Technology Program described elsewhere.

The damping system is modeled in figure 10.2.2.4.2.4-2. The torsion wire suspends the damping vane and supplies the rotational restoring force. The telescope of mass M is connected to the dampers by threads which wind up onto the damping vane. Figure 10.2.2.4.2.4-2 also shows the relationship between the functional component dimensions. An example explains the nomograph. The required spring constant is chosen as .1 lb/in. on the K scale, the torsion wire length is chosen as 4.5 inches on the L scale, and the damping vane radius is chosen as 2.5 inches on the R_1 scale. This nomograph then yields the torsion wire radius on the R scale to be 1.42×10^{-2} inches. Notice that the inclined lines are pivoted on the index scale. This pivot point may be moved vertically on the index scale to obtain independent hardware dimensions on the K and R scales or the L and R_1 scales. Therefore a large number of combinations of hardware dimensions can be sorted rapidly to satisfy particular dimensional restraints and to meet the desired spring constant.

The dependence of energy lost during one complete cycle of magnetic field to the maximum field strength shown in figure 10.2.2.4.2.4-3 for pure iron. This information is combined with the design requirement to dissipate 90 per cent of the maximum telescope kinetic energy relative to the spacecraft in ten cycles. This yields the required amount of iron that must be magnetically stressed during the first half cycle of telescope oscillation during its maximum amplitude displacement.

10.2.2.4.2.5 Integration

The experimental operational procedure requires that the several suspension systems be tested sequentially and with different combinations of fine guidance hardware. The overall hardware configuration, which is shown in figure 10.2.2.4.2.5-1, is designed for the purpose. The telescope is mounted in the inner gimbal ring, which contains the torquers, the data bridge and the angle pickoff device. The inner gimbal ring is mounted in the suspension ring structure which consists of three rings connected by means of a truss structure. The springs that are used with the Suspended Hardmount Mode are connected at 120° intervals to this structure in such a way as to provide both radial and axial constraint. The electro-magnets and the three-axis suspension springs are arranged on the front and rear rings so as to provide rotational constraints along three orthogonal axes. The suspension axes are arranged as shown in the figure so as not to interfere with the telescope line of sight. Batteries and associated conversion and regulating equipment are located on the telescope (see subsection 25.3.1 for a discussion of the telescope power system). The use of telemetry equipment is also planned for certain operational modes, and the hardware is called out in the figure.

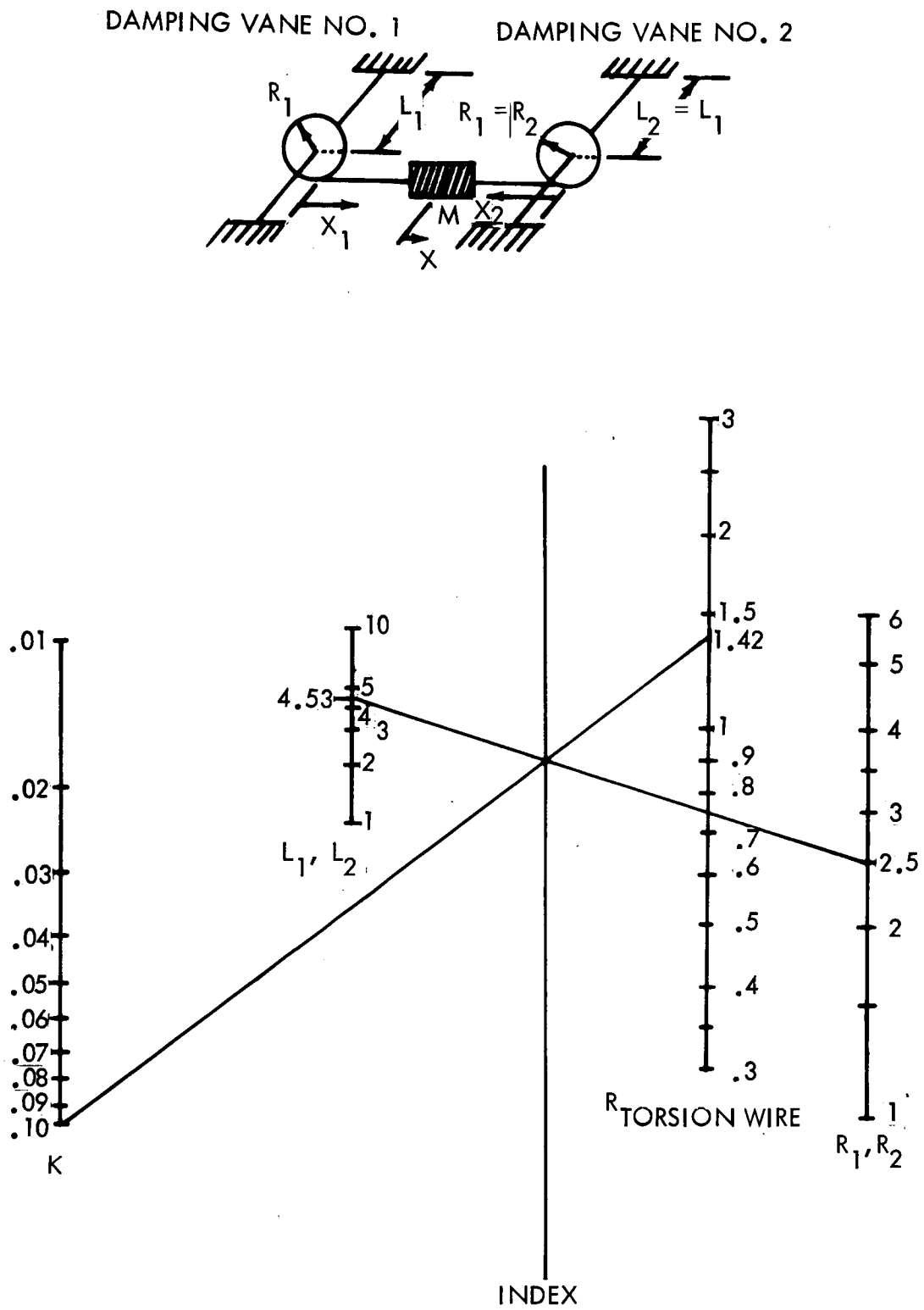


Figure 10.2.2.4.2.4-2. Model and Curves for the Damper.

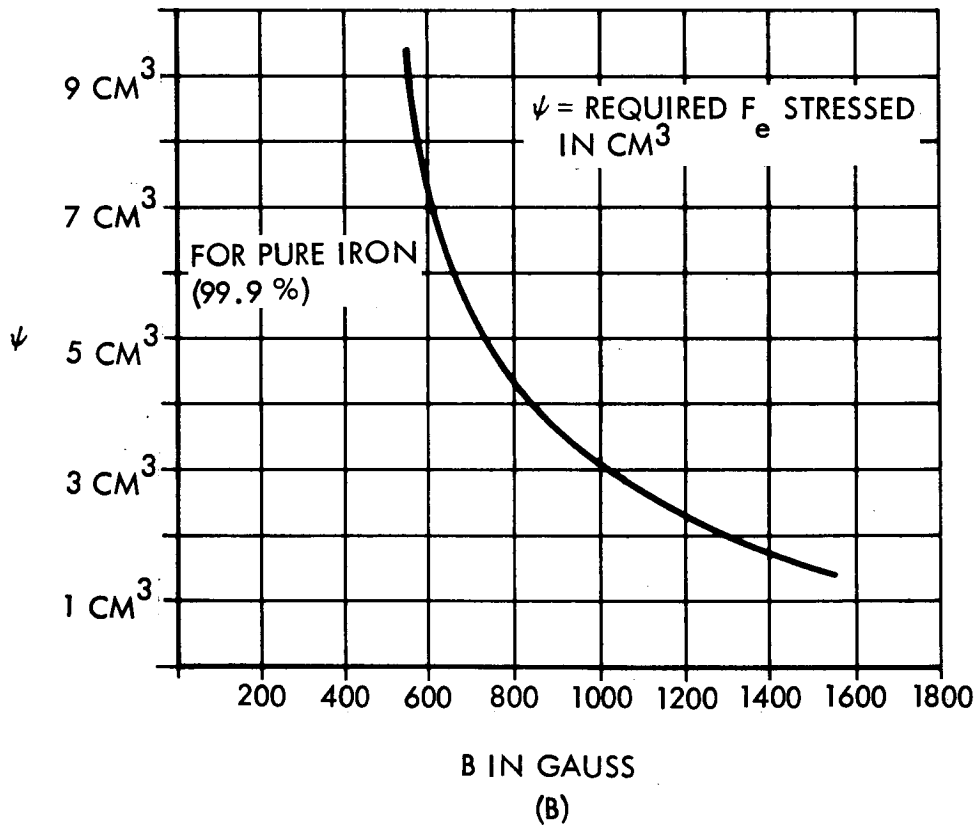
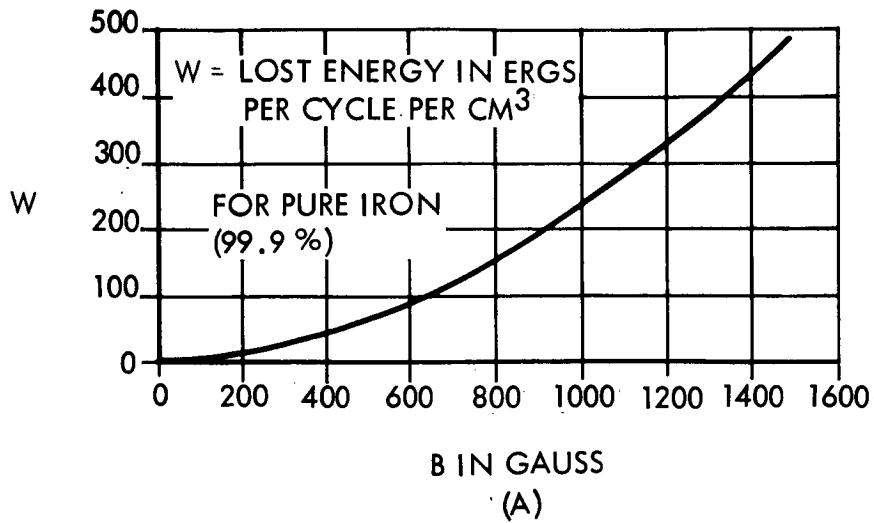
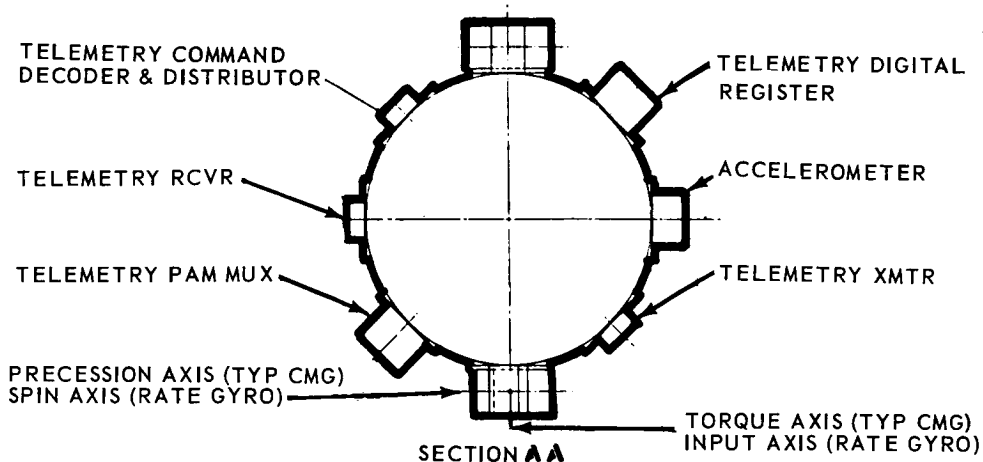
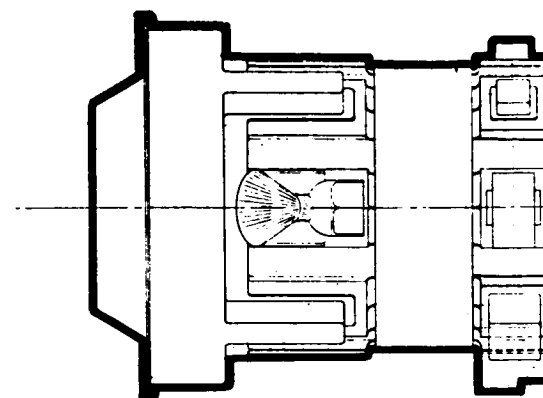
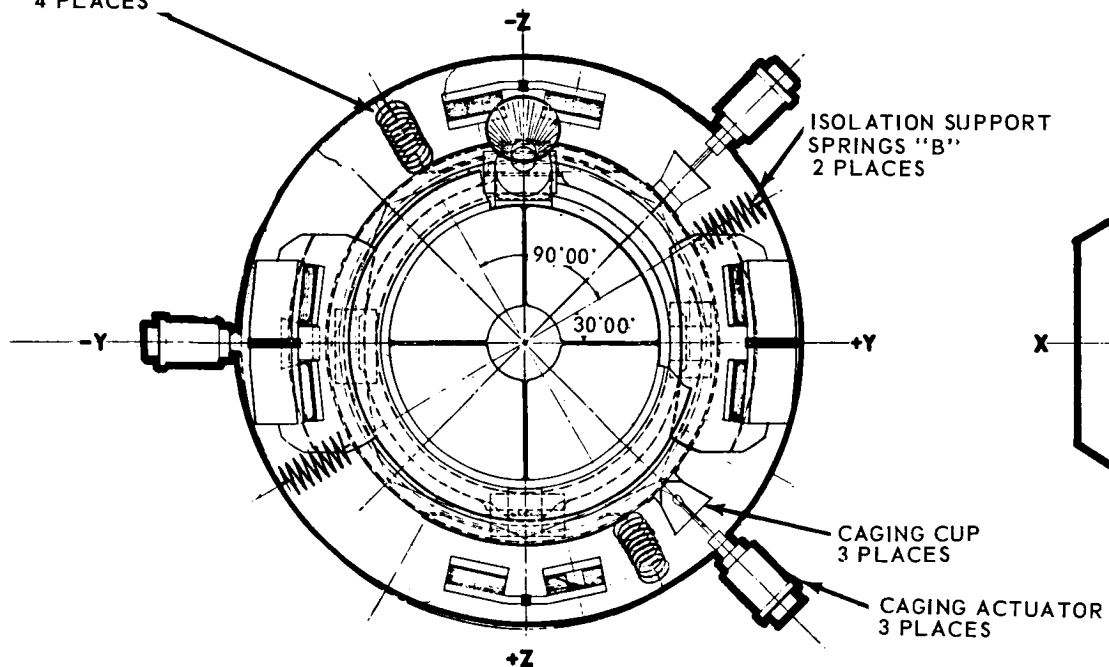


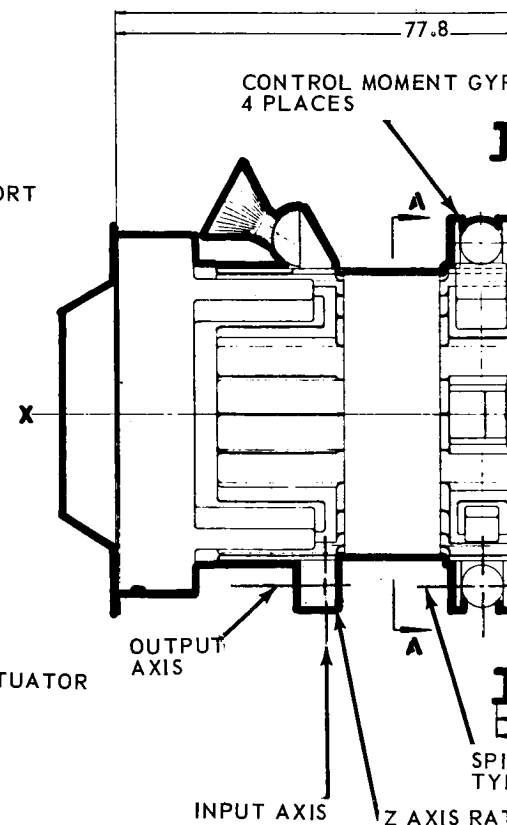
Figure 10.2.2.4.2.4-3. Loss Characteristics (a) Hysteresis Characteristics; (b) Volume Required to Meet a Specific Damping Requirement vs. Magnetic Field.



ISOLATION SUPPORT SPRINGS "B"
4 PLACES



GIMBA



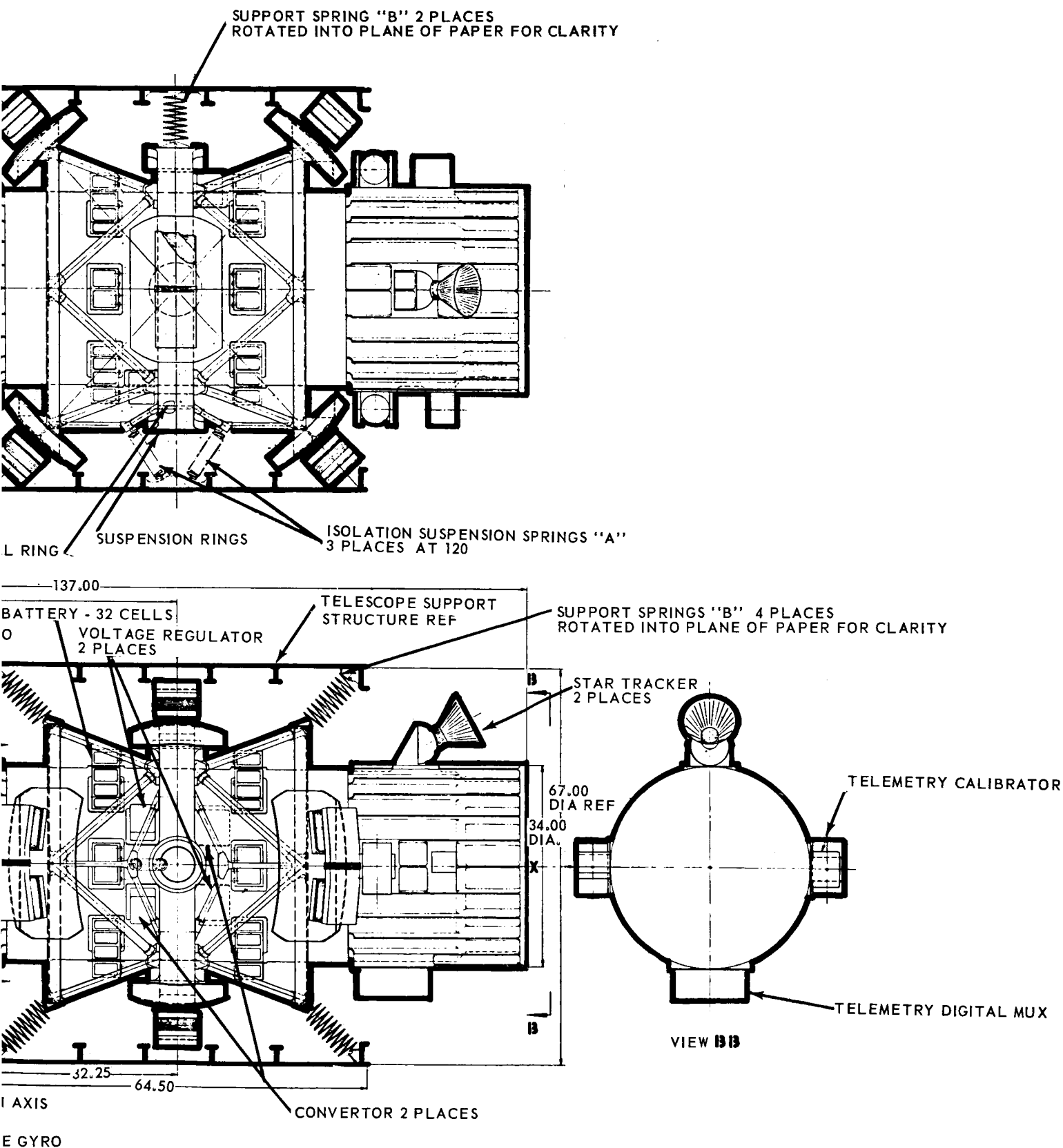


Figure 10.2.2.4.2.5-1. Experiment Mechanization.

Certain components that are required as the system is presently defined are not detailed on the drawing. These include the flex bearings that would be incorporated into the hardmount (see OTAES Part I Final Technical Report, Volume 2, CCSD for details of how these would be incorporated) and the angle pickoff device for the softmount modes (see subsection 23.2.4.2.6 for one possible mechanization of this sensor). Also, the fine guidance telescope and experiment do not indicate roll control capability as being inherent to the telescope and control system. Yet, for the three axis suspension modes some measure of control is necessary. However, the stability requirements for the roll axis are not stringent and are expected to present no hardware mechanization problem.

10.2.2.5 Data Management

The primary measurement of suspension system performance is the measured pointing stability of the telescope as a function of the disturbance environment. The procedure for collecting this data is discussed in subsection 10.3 and the handling requirements in subsection 28.0. Independent means are provided for evaluating the performance of each individual suspension type. The performance of the various suspension is completely defined, at every instant of time, by one or more of the following parameters: acceleration of the point on the spacecraft at which the suspension is attached; rotational motion of the spacecraft well, or extended structure, to which the suspension is attached; acceleration of the telescope center of mass; relative position of the telescope and spacecraft; and constraint torques exerted on the telescope by the suspension. All of these are measured either directly or by inference from other measured data, as discussed below.

- a. Spacecraft acceleration - acceleration of the telescope attach point for all operational modes is measured by a triad of instrumentation accelerometers located in the telescope well.
- b. Rotational motion of the telescope well - This is measured by a triad of rate gyros in the telescope well.
- c. Acceleration of the Telescope Center of Mass - Measured by accelerometers mounted in the telescope. Displayed to the crew.
- d. Relative Spacecraft/Telescope Position - This applies only to the soft modes of operation. For the A. C. and D. C. magnetic suspension, relative position is measured by instrumentation inherent to the A. C. suspension. For the spring suspensions, relative position is inferred from stresses in the springs attach points. This information is displayed to the crew.
- e. Constraint Torques - There is no practical means of directly measuring the constraint torques (as separate from the disturbance torques transmitted through the suspension). However, this information can be deduced from analysis of the measured parameters and correlation between the pointing error and spacecraft rotation. This analysis will be part of the on-ground data reduction.

In addition to the above data, which is concerned solely with performance of the individual suspensions, certain design information is also collected. For example, provision is made to test several levels of suspension compliance by means of the console controls.

Data from both this experiment and the fine guidance experiment that is taken from the telescope-located hardware must be transmitted across the suspension interface. For certain operational modes, a telemetry link will be used. Thus, this data must be sensed, conditioned, multiplexed and transmitted to the spacecraft. Within the spacecraft the information is received and reconstructed and applied to displays, to the space-to-ground telemetry link, to control feedback circuits and to the suspension system servo loop. On/off commands and control functions originating within the spacecraft follow a path in the opposite direction. Both optical and RF transmission links are being contemplated. The optical link would require development but it would not create electromagnetic interference problems and could be made insensitive to RF radiation from other systems. The RF link, on the other hand, could be designed using existing standard circuit techniques; it would require less power, weight, and volume and could be made extremely rugged. The choice of link types will be contingent on the extent of the electromagnetic interference problems in the system.

Exclusive of the choice between RF and optical transmission methods, the remainder of the telemetry subsystem can have various configurations. Choices are to be made between FM/FM subsystems or combinations of both. FM/FM techniques offer full time coverage and suspension frequency response, for those signals that require high frequency monitoring, and accuracy in the order of $\pm 2\%$.

PCM techniques offer an order of magnitude increase in accuracy at a slower sampling rate. The PCM option is favored at this time; however, a final selection can be made only after all factors involved in the servo loops have been considered, and the potential of modulation techniques have been further evaluated. Some factors to be considered are, frequency response requirements, accuracy requirements, phase shift introduced by the telemetry link, calibration techniques, weight, power and volume.

10.3 OPERATIONAL PROCEDURES

10.3.1 Fine Guidance Checkout and Calibration

10.3.1.1 Initial Conditions

The spacecraft may be in any orientation or operational mode consistent with IVA operations. All function switches on the control panel are in the off position.

10.3.1.2 Procedure

- a. Place operating mode switch in standby position. This energizes the fine and intermediate error sensors and all electronics for warmup but leaves the coarse and fine actuators functionally disabled.
- b. Place the star simulator switch in "on" position and set to desired level.
- c. Read the value of the fine and intermediate error signals from panel, compare with reference values and decide whether calibration is required.
- d. Assuming calibration is required, place operating mode switch in calibrate position and try to null error signals with bias control. If the error signals null, calibration is complete. Otherwise go to step (e).
- e. This is an optical alignment procedure requiring IVA and the relaying of optical images across the suspension system interface (using techniques and procedures yet to be worked out).
- f. Increase star simulator intensity to second pre-selected position (sun protect threshold) and note that the sun protect indicators are illuminated. Reduce intensity and note that the lights are extinguished.
- g. Turn star simulator to "off" position, completing the functional checkout procedure for the fine guidance mode.

10.3.2 Isolation Systems Checkout and Calibration

10.3.2.1 Initial Conditions

The spacecraft is in any orientation and its control system is in the "hold" mode (in which rates are nulled about all axes). Selector switches are in the "off" position and the cage/uncage switches are in the cage position. Both servo gains are turned off.

10.3.2.2 Procedure

- a. Place mode switch in standby position. This applies warmup power to all electronics and energizes the indicator systems but all components are functionally disabled.
- b. Check that the cage indicator lights are lighted.
- c. Check battery voltages.
- d. Depress the spring-loaded toggle switch on gimbal stop warning panel, and run through each position of the selector to verify that all the warning lights are operative.
- e. Place mode switch in checkout position and suspension selector switch in Hard Mount position. Verify that the gimbal angles are within tolerance.
- f. Place suspension selector switch in a.c. position. This applies a.c. power to the electromagnets.
- g. Check a.c. winding currents and record readings.
- h. Check the telescope translation indicators for out-of-tolerance readings. If necessary, adjust bias to null indicators.
- i. Recheck the a.c. winding currents and record readings.
- j. Place suspension selector in servo position. This applies d.c. power to the electromagnets and energizes the accelerometers.
- k. Read and record the readings of the telescope translation indicators, the d.c. winding currents, and the a.c. winding currents. Do not null the translation readings.
- l. Turn servo position gain to preselected setting.
- m. While monitoring the d.c. current of axis no. 1, increase the bias of translation indicator no. 1 and note the change in current. Repeat for the other two axes.
- n. Place the servo position gain in "off" position.
- o. Check the accelerometer indicators for out-of-tolerance readings. (Note: the tolerance range depends on the spacecraft rates in the "Hold" position, and may require spacecraft instrumentation for comparison purposes). Correct out-of-tolerance readings with the bias pots.

- p. Initiate a spacecraft roll of preselected value about the telescope optical axis. Note the readings of each accelerometer and compare with known correct value.
- q. Turn the servo acceleration gain of axis no. 1 to preselected value, and note the d. c. winding of axis no. 1. Repeat for the other two axes.
- r. Place the suspension selector and mode selector switches to "off" positions. Return spacecraft to "Hold" mode. This completes isolation experiment checkout, except for the springs which must be checked separately.

10.3.3 Telescope Star Tracker Acquisition

10.3.3.1 Initial Conditions

The spacecraft is in a known inertial orientation. The telescope gimbals and suspension ring are caged.

10.3.3.2 Procedure

- a. Place both the isolation experiment and the fine guidance experiment in standby modes for warmup.
- b. Request orientation data from ground station for proper spacecraft attitude to point the telescope line-of-sight toward target star.
- c. Execute spacecraft maneuvers to achieve required orientation, and let spacecraft stabilize.
- d. Request star tracker gimbal angles from ground station.
- e. Insert star tracker gimbal angles using thumbwheels. Press slew buttons.
- f. Note appearance of lock-on signals.
- g. Verify proper star acquisition by comparing star magnitude readouts with ground-supplied data.

10.3.4 Hard Mount Acquisition

10.3.4.1 Initial Conditions

The conditions prevailing through (g) in subsection 10.3.3.2 above.

10.3.4.2 Procedure

- a. Place the Isolation Systems Comparison suspension selector switch to "hard mount" and the mode switch to "acquire." This applies

power to the telescope gimbal torquers and connects the torquers to gimbal angle pickoff to create a short circuited "electrical cage" loop.

- b. Verify that the gimbal angles are nulled to within tolerance.
- c. Uncage both telescope gimbals.
- d. Place spacecraft control system in the fine guidance mode. This prepares it to accept error signals from these experiments when made available.
- e. Place isolation mode switch to "operate." This automatically removes the electrical cage, makes the telescope gimbal angle pickoff signals available to the spacecraft control system and accepts error signals to the gimbal torquers from the fine guidance experiment. This concludes Hard Mount Acquisition.

10.3.5 Intermediate Sensor Acquisition

10.3.5.1 Initial Conditions

The conditions prevailing through (e) in subsection 10.3.4.2 above.

10.3.5.2 Procedure

- a. Place fine guidance mode switch to "gimbal-null" position. This applies the star tracker gimbal errors to the telescope gimbal torquers, causing these error signals to be nulled, and placing the guide star within the field of view of the intermediate error sensor. A star presence signal is generated which automatically switches the star tracker gimbal angle error signals out and the intermediate error signals into the telescope gimbal torquers.
- b. Note appearance of lock-on signal.
- c. Verify proper star acquisition by comparison of magnitude readout with ground-supplied data.
- d. Check difference between telescope gimbal angles (as nulled by spacecraft control system) with pre-calculated angles as obtained from ground station, and compare with maximum allowable (as set by field of intermediate error sensor).

10.3.6 Soft Mode Acquisition

10.3.6.1 Initial Conditions

The conditions prevailing through (d) in subsection 10.3.5.2 above.

10.3.6.2 Procedure

- a. Place spacecraft control system into the "fine guidance-operation" mode. This switches the gimbal angle pickoffs out of the spacecraft control loop and switches the auxiliary angle measuring system into that loop.
- b. Place the isolation suspension selector to the servo position and the mode switch to "acquire." This automatically closes the position loop into that loop. At the same time the telescope data transmission is switched to the auxiliary telemetry system, and telescope power is switched to batteries.
- c. Turn position gain to high preset value.
- d. Place spacecraft operations into the "quiet mode."
- e. Monitor the telescope acceleration indicators, and when the acceleration levels along all three axes are within tolerance, uncage the suspension ring. The disabled power and data plug will automatically be disconnected.
- f. Verify that the suspension ring has uncaged. The telescope is now mechanically free of the spacecraft, but is still "electrically caged" through the high gain position loop.
- g. Gradually lower the position gain, keeping close monitor over the telescope translation and acceleration indicators.
- h. Turn the position gain off. This completes soft mode acquisition.

10.3.7 Fine Pointing Acquisition

10.3.7.1 Initial Conditions

The conditions prevailing through (g) in subsection 10.3.6.2 above. Intermediate pointing acquisition has been completed. Soft mode acquisition has been completed so that the telescope is "floating free" of the spacecraft.

10.3.7.2 Procedure

- a. Verify that both fine sensor select and fine actuator select switches are in no. 1 position.
- b. Note appearance of star presence signal.
- c. Observe error signals from both sensors.

10.3.8 Spacecraft Disturbance Program

10.3.8.1 Initial Conditions

For the general case, the fine pointing acquisition (subsection 13.3.7) has been completed and the isolation system to be evaluated has been placed in operation. Exceptions for the hard mount mode are noted in that procedure.

10.3.8.2 Procedure

- a. Initiate the reaction-jet disturbance program. In this mode the reaction jets are fired in a preset sequence of increasing frequency and magnitude so as to introduce precisely known disturbance momenta into the spacecraft.
- b. Monitor the fine error, telescope translation, and telescope acceleration as the automatic sequence is carried out.
- c. Terminate the sequence when one of the following occurs:
 - (1) Fine pointing acquisition is lost as indicated on the fine guidance panel
 - (2) The gimbal stop warning indicators show danger of a physical collision between telescope and spacecraft, or
 - (3) The pre-programmed disturbance profile is completed.
- d. Record the maximum acceleration and translation reached and the point at which the program is terminated.
- e. Initiate the planned astronaut disturbance program. This is a program of distinct types of astronaut movements (limbs and body segments) in which an attempt is made to separate the effects of one type of movement from those of another.
- f. Record the type of disturbance (coded) and insert time markers into the recording system for data analysis later.
- g. Terminate under the same conditions as outlined for step (c) above.
- h. Initiate the natural astronaut disturbance program. This is a program in which the astronauts execute complex movements in a "natural manner."
- i. Repeat steps (f) and (g) above.

10.3.9 Hardmount Operation

10.3.9.1 Initial Conditions

The conditions prevailing through(d) in subsection 10.3.5.2. Intermediate acquisition has been completed. Control of the telescope is by the gimbal torque motors driven by the intermediate error sensor. Control of the spacecraft by CMG driven by the gimbal angle pickoffs (pitch and yaw) and star tracker (roll).

10.3.9.2 Procedure

- a. Initiate the reaction jet disturbance.
- b. Monitor the intermediate sensor error, telescope acceleration and telescope translation as the automatic sequence is carried out.
- c. Terminate the sequence when one of the following occurs: intermediate error indicates that loss of intermediate acquisition is imminent or gimbal stop warning indicators show imminent collision.
- d. Record the maximum acceleration and gimbal angle excursion reached and the point in the program at which the sequence was terminated.
- e. Initiate the planned astronaut disturbance program.
- f. Same as subsection 10.3.8.2 (f).
- g. Terminate under the same condition as step (c) above.
- h. Initiate natural astronaut disturbance program and terminate as in step (c).
- i. Decide whether to attempt fine pointing acquisition for this mode. This is a subjective decision requiring astronaut judgement and is based on the intermediate sensor error as monitored on the fine guidance panel.
- j. Assuming a positive decision, initiate the spacecraft "quiet mode," and turn fine pointing acquire switch to "on." This initiates an automatic procedure in which the fine loop will acquire if the intermediate error and rates are sufficiently small.
- k. Verify that fine pointing has acquired as indicated on the panel.
- l. Repeat step (a) above.
- m. Terminate when fine acquisition is lost and mark the time.
- n. Decide whether to proceed to step (h) above. If so, complete that procedure.

- o. Turn mode switch to "standby" and suspension selector to "off."
This completes the formal procedure for the hard mount.

10.3.10 A. C. Suspension Operation

10.3.10.1 Initial Conditions

The conditions prevailing through (h) in subsection 10.3.6.2. The intermediate acquisition and soft mode acquisition have been completed.

10.3.10.2 Procedure

- a. Place the suspension selector switch in A. C. position.
- b. Place the mode switch in "operate" position. This applies A. C. power to the electromagnets.
- c. Perform the fine pointing acquisition procedure detailed in subsection 10.3.7.
- d. Perform the spacecraft disturbance program procedure (subsection 10.3.8).
- e. Turn the mode switch to "standby" and selector switch to "off."

This completes the procedure for evaluation of the A. C. suspension mode.

10.3.11 Servo-Suspension Operation

10.3.11.1 Initial Conditions

The fine pointing acquisition and soft mode acquisition have been completed.

10.3.11.2 Procedure

- a. Turn the selector switch to "servo" and the mode switch to "operate."
This applies a. c. and d. c. power to the electromagnets.
- b. Turn the position gain to preselected value.
- c. Turn the acceleration gain to preselected value.
- d. Perform the spacecraft disturbance program (subsection 10.3.8).
- e. Experiment with the gain control settings to determine optimum gain values. This is a procedure that will require judgement and decision-making on the part of astronaut operator, assisted from data analyzers on the ground. By varying the gain settings, a wide spectrum of suspension characteristics can be obtained and hence invaluable design data for future development.

- f. Turn mode switch to "standby" and selector switch to "off."
This completes the procedure for evaluating the servo mode.

10.3.12 Spring Mount Operation

10.3.12.1 Initial Conditions

The spacecraft may be in any orientation and is inertially stabilized. The gimbals and the suspension ring are caged. Both the fine guidance experiment and the isolation experiment are electrically disabled.

10.3.12.2 Procedure

- a. Proceed to install the springs. This will require EVA activity. The springs, the spring attachment mechanism, the spring attachment tool, and access well must all be designed to accommodate this activity. The design details and the installation procedure are yet to be worked out.
- b. Turn the isolation and fine guidance mode switches to "standby" for warmup.
- c. Perform procedures outlined in subsection 10.3.3 through 10.3.7 (Star Tracker, Hard Mount, Intermediate Sensor, Soft Mode and Fine Pointing Acquisitions).
- d. Turn selector switch to "springs" and mode switch to "operate."
- e. Perform spacecraft disturbance program procedure (subsection 10.3.8).
- f. Turn mode switch to "standby" and selector switch to "off."
This completes this operational mode.

10.3.13 Sensor and Actuator Selection

10.3.13.1 Initial Conditions

The suspension judged best on the basis of the previous tests is selected and soft mount and intermediate acquisitions are completed.

10.3.13.2 Procedure

- a. Repeat subsection 10.3.7, Fine Pointing Acquisition
- b. Repeat subsection 10.3.8, Spacecraft Disturbance Program.
- c. Turn fine sensor selector to position 2.
- d. Repeat (b) above.

- e. Turn fine actuator selector to position 2.
- f. Repeat (b) above.
- g. Turn fine sensor selector to position 1.
- h. Repeat (b) above.
- i. Recage telescope and disable both experiments.

10.4 SUPPORTING ANALYSES

10.4.1.1 Optical Tolerances of Primary Mirror

An imaging astronomical telescope is generally desired to be diffraction limited, that is the angular size of the stellar image is determined by the formula $1.22 \lambda/D$ where λ is the wavelength of interest and D is the diameter of the objective. This is practically achieved when the figure of the mirror departs from geometrical perfection by no more than one quarter wavelength.

Where guidance is concerned rather than image information, however, diffraction limited figure may not be required. Thus, in the fine guidance telescope the performance of the fine sensor loops is best (from tracking gradient and resultant loop dynamics considerations) when the size of the focused spot of light is commensurate with the apertures of the crossed aperture detector, or the apex diameter of the pyramid prism in the quadrant detector.

In the case of the crossed aperture modulator the intensity gradient is maximum with the smallest possible spot and the limit in spot size for best collection efficiency and signal-to-noise ratio is the aperture width. Thus the optimum lies somewhere between a finite point and the aperture width.

The pyramid prism on the other hand requires a spot size larger than the dimensions of the apex of the prism but smaller than the projected area of any one facet. In practice it is a few times larger than the apex dimensions.

In the fine guidance system the desired precision is 0.01 second, and the smallest possible spot size represents 0.16 second or about 16 times the desired precision. In linear dimensions the diffraction limited spot size is about 30 microns in diameter. A lower limit is set however, by the prism apex, about 50 microns in diameter and an upper limit by the crossed apertures, which are 350 microns wide. The spot can be enlarged either by defocusing a diffraction limited spot or by relaxing the tolerances of the telescope. An expansion by a factor of four would provide a spot 120 microns in diameter which is 2.5 times the apex size, one third the aperture width and about 60 times the required tracking accuracy. Experience at Kollsman with the dual aperture tracker indicates the optimum is in the vicinity of one third the aperture width.

To obtain this spot size the figure tolerance can be relaxed from the nominal quarter wavelength for a diffraction limited telescope to one wavelength.

10.4.1.2 Attitude Control System Analysis

10.4.1.2.1 Coarse and Intermediate Attitude Control

This is one of the two loops of the fine guidance attitude control system, and includes a coarse sensor (star trackers) and intermediate sensor (primary mirror), but one actuator (CMG's) responding to either sensor.

DEFINITION OF SYMBOLS

<u>SYMBOL</u>	<u>MEANING</u>
G_T	Star Tracker or Sensor Transfer Function
K_A	Amplifier Gain
K_t	Torque Gradient of C.M.G. Torque Motor
C	Gyro Internal Damping Constant
S	Laplace Operator
τ_G	C.M.G. internal time constant (= $\frac{\text{Gimbal Inertia}}{\text{Damping Constant}}$)
H	Angular Momentum of C.M.G.
J	Moment of Inertia of Telescope
K_{RG}	Rate Gyro Gradient
K_F	Filter Gain Factor
τ_F	Time Constant of Filter
K_I	Integrator Scaling (gain) factor
K_D	Gain of Driver Amplifier
G_a	Transfer Function of Fine Actuator
K_S	Servo Gain Factor (Including effect of rate feedback, if used)
τ_S	Servo Time Constant
ω_n	Undamped Natural Frequency of Loop
ξ	Ratio of Damping to Critical

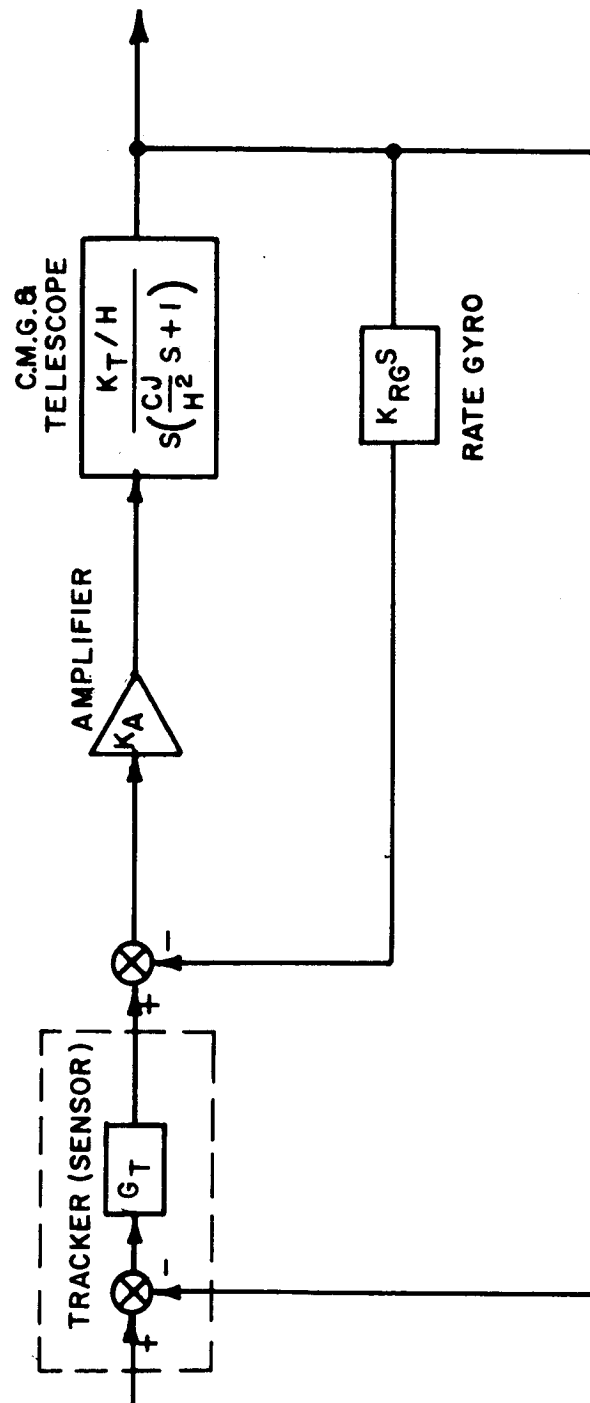


Figure 10.4.1.1.2-2 Equivalent Simplified Attitude Control Loop
(Coarse and Intermediate)

Numerical values used in the coarse and intermediate attitude control loops, for the purpose of a first approximation to performance, are listed above (see section 10.4.1.2.3 for C and H, and sections 10.4.1.2.6.1 and 2 for derivation of other values).

It should be noted that the quantity K_{RG} includes both the output gradient of the rate gyro itself and a separate rate feedback amplifier following it. The approximate gains of this amplifier in the coarse and intermediate modes are shown in the above table as the last item.

10.4.1.2.2 Fine Guidance Control

When the coarse and intermediate attitude control loops have brought the star within the field of view of the fine guidance sensor, the coarse sensor and the intermediate loop remain operative while the fine guidance control loop maintains the desired accuracy of the telescope line-of-sight.

While the intermediate attitude control system holds the telescope in position, the fine guidance system holds the line-of-sight on the star by controlling the position of the line-of-sight within the telescope tube.

Two types of control actuation will be utilized alternately with either of the two fine sensors. The sensor not used for control will be used for monitoring purposes.

One actuator consists of a mirror mounted on a cantilever support, and deflected through the required angle electromagnetically. Figure 10.4.1.2-3 is a mathematical model of the fine guidance control loop utilizing this actuator, where the bandwidth of G_A considerably exceeds that of the filter. The other actuator consists of a servo driven diasporameter in the line-of-sight. Figure 10.4.1.2-4 is a mathematical model of the fine guidance control loop utilizing this actuator.

The two actuators will be used alternately, for comparison purposes.

After the actuators, the light passes through a beam splitter, sending half of it to each of the two sensors. While one sensor is in the control loop, the other will be used as an error transducer to monitor and compare the effectiveness of the two types of actuators. The roles of the two sensors will also be alternated to compare the effectiveness of the sensors.

One sensor will consist of a photomultiplier with crossed vibrating apertures providing modulation of the impinging light. The error signal component of the modulation is a function of the position of the star in the field of view. (See figure 10.4.1.5-1.) The other sensor will be a quadrant type, in which the field of view is divided into quadrants, with effectively separate sensing systems for each quadrant.

Initial estimates of the characteristics of the Fine Guidance control loop produce the following numerical values:

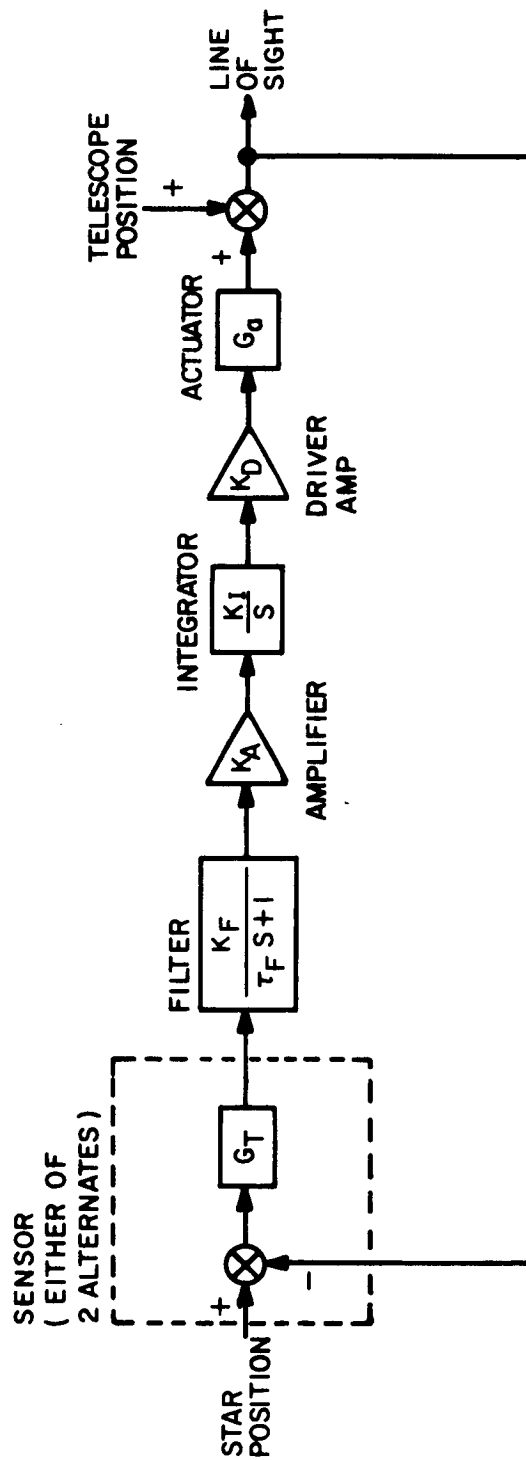


Figure 10.4.1.2-3 Fine Guidance Loop, (Cantilever Mirror Actuator)

- a. For the Cantilever mirror actuator (Figure 10.4.1.2-3)

$$G_A = 6.67 \text{ sec./amp} = 3.24 \times 10^{-5} \text{ radians/amp.}$$

$$G_T K_F = 850 \text{ volts/radian}$$

$$K_A K_I K_D = 116.2 \text{ amps./volt-sec.}$$

$$\tau_F = 0.159 \text{ sec.}$$

- b. For the Diasporameter Actuator (Figure 10.4.1.2-4)

$$G_T K_F = 850 \text{ volts/radian}$$

$$K_A K_S = 0.00377 \text{ radians/volt-sec.}$$

$$\tau_F = 0.159 \text{ sec.}$$

$$\tau_S = 0.05 \text{ sec.}$$

In each case the closed loop bandwidth is estimated to be at least twice that of the intermediate loop. The fine loop would thus be rapid enough to compensate for disturbances in the intermediate loop output, and this bandwidth ratio appears adequate for a first approximation to the system. Computer simulation, with expected input disturbances to the intermediate loop, will enable further refinement to be made.

10.4.1.2.3 Characteristics of C.M.G. Used in Analysis

The C.M.G. used in this discussion is the Kearfott C709547013. This is a unit which has been built in production; and is available. Figure 10.4.1.2-5 is an envelope drawing of this unit, and shows the orientation of the axes.

The gyro has a single wheel mounted in a precession gimbal assembly, which in turn is mounted on bearings within the housing. The precession gimbal is provided with a torque motor and an angle pickoff. The torque motor is used to drive the precession gimbal as an input. The gimbal is viscously damped. The output of the angle pickoff is used to determine when the gimbal angle has reached that at which it will be "unloaded" or "de-saturated".

Two of these units are used on each controlled axis of the telescope tube, oriented as shown on Figure 10.2.1-2. This orientation is designed to minimize cross-coupling effects.

Pertinent specifications are as follows: (values are per gyro, and where they are specified as adjustable, the values used in the analysis are shown):

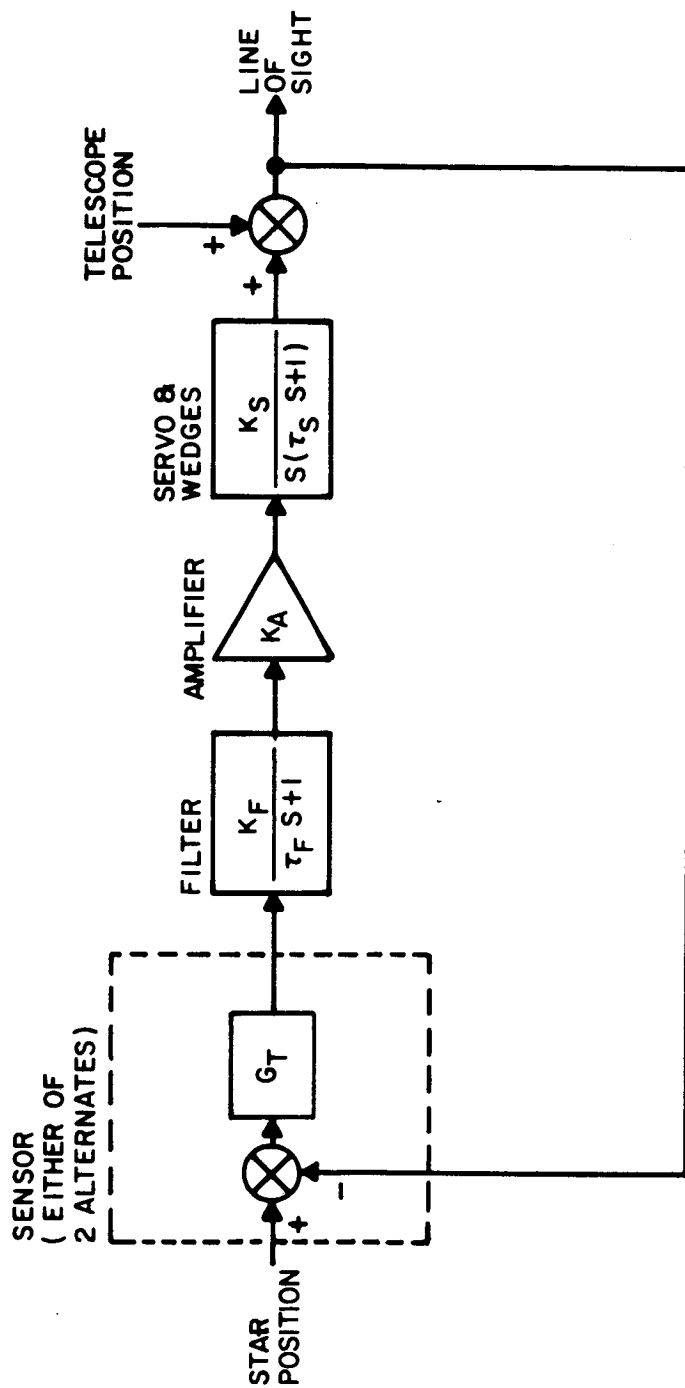
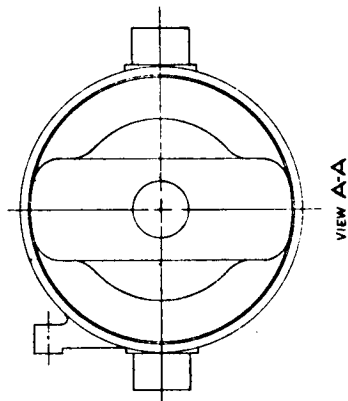
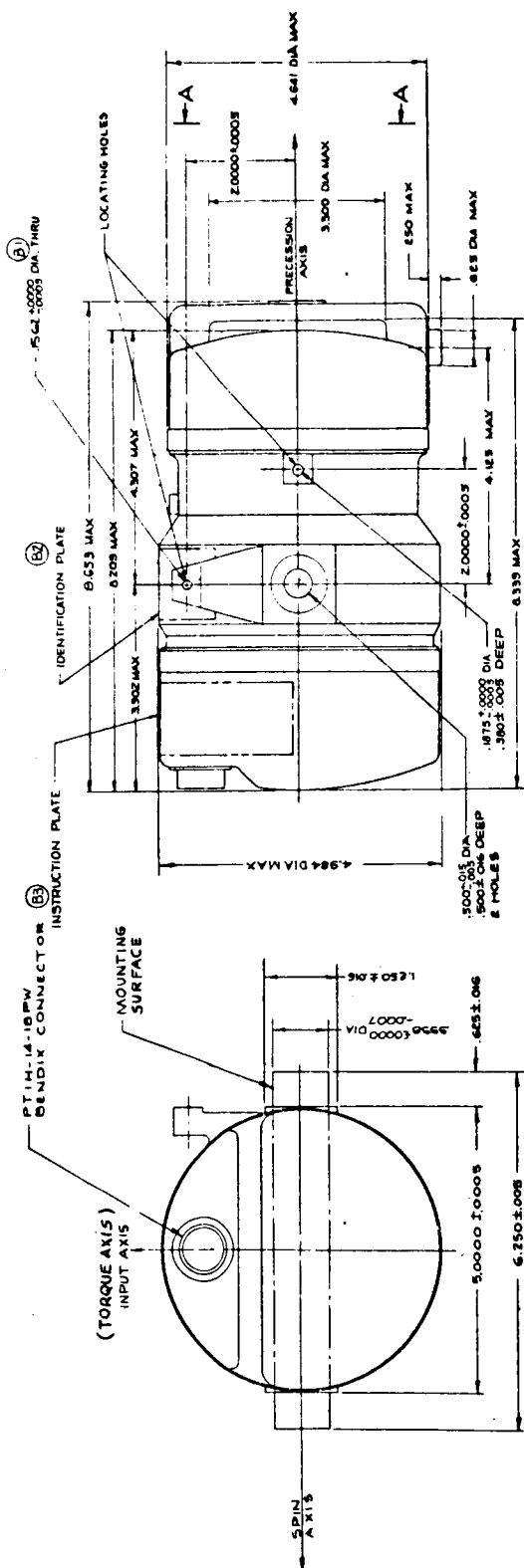


Figure 10.4.1.2-4 Fine Guidance Loop, Diasporameter Actuator)



- NOTES
- 1-FINISH: BLACK WRINKLE
 - 2-PERFORMANCE IS IN ACCORDANCE WITH PERFORMANCE SPEC. C17954701Z (B5)
 - 3-END PRODUCT NO. C70954701Z
 - 4-LOCATING HOLES AND MOUNTING SURFACES ALIGN PRECESSION AXIS (B4)

Figure 10.4.1.2-5 Envelope Drawing Gyroscope Control Moment

Wheel momentum = $H = 45 \times 10^6 \text{ Gm-Cm}^2/\text{sec}$
= 639 oz-in-sec

Torquer sensitivity = $K_t = 28.36 \text{ oz-in/amp}$

Damping coefficient = $C = 325 \text{ oz-in-sec}$

Wheel speed = 11,700 rpm $\pm 100 \text{ rpm}$

Power input - starting - 40 watts max.
running - 6 watts max.

10.4.1.2.4 Field of View and Accuracy of the Star Tracking Sensors

10.4.1.2.4.1 Coarse Star Tracker

Field of view = 1° (for initial acquisition)

After acquisition, in "track" mode, the tracker gimbals will hold the star in the center of the tracker field within the limits of the gimbal motion, provided the line-of-sight remains unobstructed.

Resolution = ± 15 arc seconds

Residual error = ± 30 arc seconds

10.4.1.2.4.2 Intermediate Sensor

Field of view = ± 1 arc minute (= ± 60 arc seconds)

Resolution = ± 0.2 arc second

Residual error = ± 0.5 arc second

10.4.1.2.4.3 Fine Sensor

Field of view = ± 1.0 arc second

Resolution = ± 0.003 arc second

Residual error = ± 0.01 arc second

10.4.1.2.5 Initial Acquisition Requirements - Coarse Mode

In order for the coarse star tracker to acquire the desired star, that star must be within the $\pm 1^\circ$ field of view of the tracker telescope for a period not less than the acquisition and settling time of the star tracker subsystem (typically 2-3 seconds).

Since there are no reference inputs to the Fine Guidance Attitude Control System, the Fine Guidance telescope suspension must be locked at the center of its travel during initial acquisition.

Under this condition, the combined effects of the input (offset) command to the tracker, and the spacecraft attitude control must place the star within the $\pm 1^\circ$ field of view for at least the required time, as mentioned above.

With excursions of as much as $\pm 1.5^\circ$ possible in the station-keeping mode of spacecraft attitude control, it is possible that the above conditions may not be met. It is certainly not possible to state with any confidence that they will be met.

It will therefore be necessary to use a tighter spacecraft attitude control, such as that employed for docking, during the initial acquisition phase.

Once acquisition and tracking by the coarse tracker is established, the tracking mode, utilizing the gimbal capabilities of the star tracker, should allow a return of the spacecraft attitude control to the station-keeping mode if desired. Similarly, in the intermediate mode and during fine guidance, the intermediate sensor combined with $\pm 2^\circ$ of suspension motion will maintain telescope line-of-sight as required, even with the $\pm 1.5^\circ$ spacecraft excursions.

10.4.1.2.6 Detailed Analysis of Fine Guidance Loops (Y-Y and Z-Z Axes Similar)

10.4.1.2.6.1 Coarse Attitude Control

The inertia of the telescope is estimated as $1,594,546 \text{ lb-in}^2 = 66,100 \text{ oz-in-sec}^2$.

The preliminary gyro chosen has the following characteristics (see section 10.4.1.2.3):

$$H = 45 \times 10^6 \frac{\text{Gm-Cm}^2}{\text{sec}} (= 45 \times 10^6 \text{ dyne-cm-sec}) \\ = 639 \text{ oz-in-sec}$$

Torquer Sensitivity, $K_t = 2700 \text{ dyne-cm/ma}$,

maximum, and is adjustable. A representative value of 2000 dyne cm/ma was chosen for these computations.

$$K_t = 2000 \text{ dyne cm/ma} = 28.36 \text{ oz-in/amp.}$$

Maximum Available Torque = 1.913 oz-in . The value for K_t/C is given as $10.0^\circ/\text{hr/ma}$ minimum. The maximum gyro torque gain, H/C is given as 2.0 .

Using the value of $K_t = 28.36$ oz-in/amp, and assuming $K_t/C = 18^\circ/\text{hr/ma}$ yields

$$C \equiv K_t/K_t/C = 325 \text{ oz-in-sec},$$

and

$$H/C = \frac{639}{325} = 1.935 < 2.0; \text{ Therefore, O.K.}$$

The time constant of the gyro = J_{gimbal}/C , has not been determined, but it is estimated to be on the order of 0.05 second or possibly less.

Without the Rate Gyro, the mechanical time constant

$$= \frac{CJ}{H^2} = \frac{(325)(66,100)}{(639)^2} = 52.6 \text{ seconds}$$

Compared to this, τ_G may be assumed negligible. In addition, the star tracker closed loop response of about 1 cps represents a time constant on the order of 0.159 second, which may also be considered negligible when compared to 52.6 seconds.

On this basis, the transfer function of the gyro + load (telescope inertia) may be reduced to:

$$\frac{K_t/H}{S \left(\frac{CJ}{H^2} S + 1 \right)}$$

It should be noted that this simplification still contains the essential variables of the C.M.G., and the telescope inertia. The transfer function of the star tracker, G_T may also be assumed as a gradient (gain factor) in the frequency region of interest (see section 1.3.1).

As a first estimate, it is assumed that the loop gain is set to produce peak (saturated) torque of the gyro torque motor at ± 1.1 degree (± 66 arc minutes). This is to assume (as a first approximation) a control gradient at acquisition sensitive to magnitude of displacement as well as sense, obtained by setting the saturation level just beyond the field of the coarse sensor. Torque input to CMG gimbal per radian of star tracker error is

$$G_T K_A K_t = \frac{1.913}{(1.1)(0.0175)} = 99.5 \text{ oz-in/radian}$$

Putting the closed-loop transfer function (CLTF) into the classical form for a second-order system and without the rate gyro shown in the figure

$$\text{CLTF} = \frac{\omega_n^2}{s^2 + 2\zeta\omega_n s + \omega_n^2}$$

we get

$$\frac{\frac{(G_T K_A K_t) H}{CJ}}{s^2 + \frac{H^2}{CJ} s + \frac{(G_T K_A K_t) H}{CJ}} = \text{CLTF}$$

$$\omega_n^2 = \frac{G_T K_A K_t H}{CJ} = \frac{(99.5)(639)}{(325)(66,100)} = 0.002956 \text{ sec}^{-2}$$

and

$$\omega_n = 0.0544 \text{ rad/sec}$$

$$H^2/CJ = \left(\frac{1}{52.6} \right) = 0.0191 \text{ sec}^{-1} = 2\zeta\omega_n$$

Therefore,

$$\zeta = \frac{(0.0191)}{2(0.0544)} = 0.176 \text{ of critical}$$

This is quite low and may be expected to produce a rather long term transient response. To improve this, the rate gyro is added to improve the loop damping characteristics.

Starting with the same forward gain, and hence, the same natural frequency, $\omega_n = 0.0544 \text{ sec}^{-1}$, but setting $\zeta = 0.7$,

$$2\zeta\omega_n = (1.4)(0.0544) = 0.07616 \text{ sec}^{-1}$$

If the new equivalent transfer function is derived and put into the classical form, we find

$$CLTF = \frac{\frac{G_T K_A K_t H}{CJ}}{s^2 + \frac{H (H + K_A K_2 K_{RG})}{CJ} s + \frac{G_T K_A K + H}{CJ}}$$

Therefore,

$$\tau = \frac{1}{2\zeta\omega_n} = \frac{1}{0.07616} = 13.13 \text{ seconds}$$

and is still large compared to τ_G and to the tracker time constant of approximately 0.159 second.

Also,

$$0.07616 = \frac{(639) (639 + K_A K_t K_{RG})}{(325) (66,100)}$$

or

$$K_A K_t K_{RG} = 1922 \text{ oz-in-sec.}$$

Star tracker scaling is adjustable over a wide range, since it is derived from a digital-to-analog conversion of the difference between the tracker gimbal readout and the computed gimbal offset.

Since $G_T K_A K_t = 99.5 \text{ oz-in/radian}$,

and

$$K_t = 28.36 \text{ oz-in/amp}$$

then, nominally setting $K_A = 30 \text{ amps/volt}$,

$$G_T = \frac{(99.5)}{(30) (28.36)} = 0.117 \text{ volts/radian, which is easily attainable.}$$

Further, rate feedback gyro and amplifier gradient, K_{RG} , then becomes:

$$K_{RG} = \frac{1922}{(28.36) (30)} = 2.26 \text{ volts/rad/sec}$$

A typical rate gyro would have an output gradient of about 0.02 volts/rad/second, and an amplifier gain of approximately

$$\frac{2.26}{0.02} = 113 \text{ volts/volt is required as part of } K_{RG}.$$

10.4.1.2.6.2 Intermediate Mode Attitude Control

In the intermediate mode, the following assumptions are made:

- a. Gyro characteristics and amplifier gain, K_A , are assumed the same as for Coarse.
- b. Tracker gradient is assumed to increase to 8.78 volts/radian (about 41.5 microvolts/arc second), which is a reasonable value.
- c. Rate feedback amplifier gain is adjusted to again achieve damping of approximately 0.7 of critical.

$$G_T K_A K_t = (8.78) (30) (28.4) = 7480 \text{ oz-in/radian}$$

$$\omega_n^2 = \frac{(7480) (639)}{(325) (66,100)} = 0.2225 \text{ sec}^{-2}$$

$$\omega_n = 0.472 \text{ sec}^{-1}$$

$$\frac{1}{\tau} = 2\zeta\omega_n = 2(0.7) (0.472) = 0.661 \text{ sec}^{-1}$$

$$\tau = 1.512 \text{ seconds}$$

At this point, it can be seen that, though τ_G is still insignificant, the assumption that G_T is a simple gradient results in an approximation which is somewhat coarser than the previous case. It is still usable as a first approximation to performance, however.

$$\frac{1}{\tau} = 0.661 = \frac{(639) (639 + K_A K_t K_{RG})}{(325) (66,100)}$$

$$K_A K_t K_{RG} = 21,591 \text{ oz-in sec}$$

and

$$K_{RG} = \frac{21,591}{(30)(28.36)} = 25.3 \text{ volts/rad/sec}$$

(for the same gyro sensitivity, the required amplifier gain increases to $\frac{25.3}{0.02} = 1265 \text{ volts/volt}$, which is also reasonable).

10.4.1.2.6.3 Fine Guidance Control Loop - Initial Estimates

a. Cantilevered Mirror Actuator

The estimated sensitivity of the mirror deflection actuator is 0.15 amp in a 1000 turn coil to produce a full-scale deflection equivalent to 1 arc second referred to the line-of-sight. It is also anticipated that the actuator will have no critical frequencies below 2K Hz, and therefore it is assumed that it is frequency insensitive in the region of operation.

The use of a permanent magnet, together with the proper design of pole pieces and air gaps, permits a nearly linear relationship between deflection and driving current. The deflection sensitivity calculated below is an estimated average value for a representative design.

The actuator transfer function, G_A , may then be considered as

$$G_A = \frac{1}{0.15} = 6.67 \text{ arc sec/amp} = \frac{6.67}{(3600)(57.3)}$$
$$= 3.24 \times 10^{-5} \text{ radians/amp}$$

It is estimated that the sensor sensitivity, through the low-pass filter, is

$$G_T K_F = 850 \text{ volts/radian } (\approx 4 \text{ mv/arc sec})$$

The low-pass filter time constant is assumed set at

$$\frac{1}{2\pi} \text{ sec} = \frac{1}{6.28} \text{ sec.}$$

The closed-loop transfer function (see figure 10.4.1.2-3) may be shown to be

$$\begin{aligned}
 \text{CLTF} &= \frac{G_T K_F K_A K_I K_D G_A}{\tau_F s^2 + s + G_T K_F K_A K_I K_D G_A} \\
 &= \frac{\frac{G_T K_F K_A K_I K_D G_A}{\tau_F}}{s^2 + \frac{1}{\tau_F} s + \frac{G_T K_F K_A K_I K_D G_A}{\tau_F}}
 \end{aligned}$$

if ζ is set at 0.7,

$$\omega_n = \frac{\pi}{0.7} = 4.48 \text{ rad/sec}$$

$$\omega_n^2 = 20.1 \text{ sec}^{-2} = \frac{G_T K_F K_A K_I K_D G_A}{\tau_F}$$

$$K_A K_I K_D = \frac{20.1}{2 (850) (3.24 \times 10^{-5})}$$

$$K_A K_I K_D = 116.2 \text{ amps/volt-sec.}$$

The velocity constant, K_V for the loop is

$$K_V = G_T K_F K_A K_I K_D G_A = \frac{(20.1)}{2 \pi} = 3.20 \text{ sec}^{-1}$$

b. Diasporameter Loop

The Diasporameter loop is designed to give essentially the same performance as the cantilever mirror loop. The additional time constant, however, will result in a somewhat different transient response, with the damping ratio dropping a little at the same gain ($\zeta \approx 0.58$ at $K_V = 3.20 \text{ sec}^{-1}$, taken from a root-locus plot, see figure 10.4.1.2-6).

$$K_V = 3.20 = G_T K_F K_A K_S$$

As before

$$\tau_F = \frac{1}{2 \pi} = \frac{1}{6.28} \text{ second.}$$

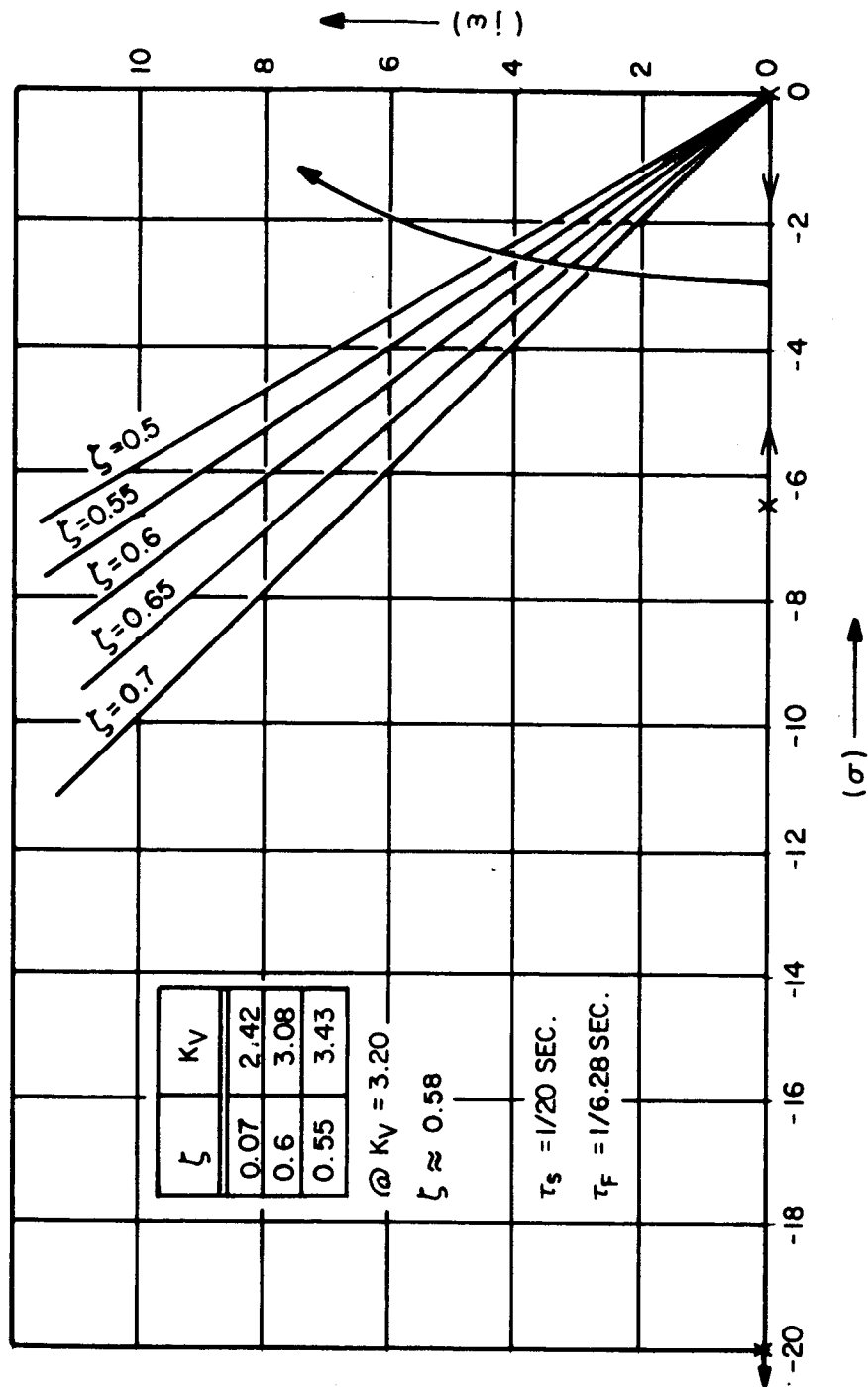


Figure 10.4.1.2-6 Root Locus Plot, Diasporameter Loop

A representative value of τ_S is assumed, so that $\tau_S = \frac{1}{20}$ sec = 0.05 sec.

As before, $G_T K_F$ is estimated at 850 volts/radian

$$K_A K_S = \frac{3.20}{850} = 0.00377 \text{ radian/volt-sec}$$

This value includes (in the value of K_S), the optical ratio between line-of-sight deflection and diasporameter rotation, as well as any gearing, etc., in the diasporameter drive. Increasing the bandwidth of the low-pass filter a little will allow a K_V of 3.20 at a damping ratio of 0.7. The presence of the additional lag, τ_S , will still keep the effective noise bandwidth of the closed loop very nearly the same as that obtained where

$$\tau_S = 0 \text{ and } \tau_F = \frac{1}{2\pi}$$

10.4.1.2.6.4 Adjustable Parameters

Quantities which are, or can be, adjustable and the ranges over which they may be adjusted are as follows:

a. Coarse and Intermediate Loops

K_t Subject to a maximum value of 38.25 oz-in/ampere

C Minimum value for presently available gyro
 $= \frac{H}{2.0} = 319.5 \text{ oz-in-sec, Max} = \frac{H}{0.3} = 2130$
 oz-in-sec

K_t/C Minimum value = 0.0484 rad/amp-sec

CMG Input

Torque Maximum (saturation) level = 1.913 oz-in

K_A Amplifier gain constant - subject only to practical limitations of amplifier design

G_T , Coarse Adjustable over a wide range by adjusting D-A converter scaling and electronic gain or **attenuation**

G_T , Inter- mediate	No significant lower limit. Upper limit determined only by amount of gain available after tracking sensor. For practical purposes it is not unreasonable to assume that values up to at least an order of magnitude greater than those used can be obtained.
K_{RG}	Basic gyro sensitivity can be varied over a range of perhaps 5:1 or more. In addition, the use of an amplifier allows this factor to be adjusted over a wide range, limited only by amplifier design limitations.

b. Fine Guidance Loop - Cantilever Mirror Actuator

G_A	Subject to adjustment if actuator is redesigned. Present design is not finalized, but is used as an estimate of what can reasonably be attained.
G_T	Some adjustment upward (perhaps a factor of ten) may be possible. No practical low limit.
K_F	Electronic gain constant. May be adjusted over a wide range, depending on filter design, and use of amplification or attenuation.
τ_F	Subject only to limitations of art of filter design.
K_I	Function of integrator design and scaling.
K_A , K_D	Electronic amplifier gain, therefore, adjustable over wide range

c. Fine Guidance Loop, Diasporameter Actuator

G_T	Same as in b.
K_F	Same as in b.
τ_F	Same as in b.
K_A	Electronic amplifier gain. Adjustable over a wide range.
K_S	Servo drive electro-mechanical gain constant, typically on the order of $3 \times 10^{-5} \frac{\text{rad/sec}}{\text{volt}}$

10.4.1.2.6.5 Disturbance Torque Effects

Disturbance torque level required to push the intermediate loop to the point where the fine loop is on the verge of losing the star:

The "stiffness" of the intermediate loop is

$$\frac{G_T K_A K_t H}{C} \text{ oz-in/radian} = \frac{(7480) (639)}{(325)} \text{ oz-in/radian}$$

and the torque to produce 1 arc second error is then:

$$\frac{(7480) (639)}{(325) (57.3) (3600)} = 0.0714 \text{ oz-inch}$$

This level of torque must not be exceeded on a d-c or low frequency basis, or else fine guidance will be interrupted.

In the preceding analysis, the magnitudes of such external torques as gravity gradient and solar pressure have been assumed negligible. The estimated maximum gravity gradient torque, for instance, is somewhat less than 1×10^{-4} oz-inch. Solar pressure could produce a higher maximum torque, depending upon design and exposure of the telescope tube. If it is found, in a later design phase, that it is high enough to introduce excessive error, the design of the intermediate loop would be modified to provide adequate stiffness to avoid this problem.

The fine actuators are designed not to saturate within the field of view of the fine sensors. The intermediate actuator has a total gimbal travel of $\pm 60^\circ$ to $\pm 65^\circ$. Assuming the $\pm 60^\circ$ value, the gyros saturate, then at a gimbal

angle of $\frac{60}{57.3}$ radians. Allowing a small safety margin, we assume 1.0 radian.

The output torque for one gyro is $H \frac{d\theta}{dt}$, or for 2 gyros $2H \frac{d\theta}{dt} = T$

$$\text{or } 2H \frac{d\theta}{dt} = T$$

$$\int_0^{1 \text{ radian}} 2H d\theta = \int_0^t T dt$$

$$2H = \int_0^t T dt = 2(639) = 1278 \text{ oz-in-sec}$$

This represents the integrated angular impulse which will saturate the CMG's.

For illustration, a constant unidirectional torque of 0.05 oz-in would take approximately 7.0 hours to accomplish CMG saturation.

10.4.1.2.6.6 C.M.G. Desaturation Technique

The results above indicate little likelihood of saturating the C.M.G.'s during the course of a typical tracking period. It is therefore recommended that the following desaturation procedure be utilized.

Whenever the Fine Guidance Telescope is not in active operation, the suspension should be caged (at the midpoint of its travel). The gimbal torque motor of each C.M.G. should be fed, through an amplifier of adequate gain, from the output of its gimbal angle pickoff, in such a phase as to reduce the gimbal angle to zero. The maximum disturbance torque which this procedure can couple into the spacecraft is

$$\frac{2(1.913)(639)}{325} = 7.6 \text{ oz-in, per axis}$$

In no case would the total angular momentum imparted to the spacecraft exceed 1278 oz-in-sec per axis and in general, it would be appreciably less. A more typical value might be 60 oz-in-sec or less.

10.4.1.3 Computation of Optical Bench Errors Due to Microseismic Activity

10.4.1.3.1 Introduction

In the following evaluation, alignment errors of a large optical bench caused by microseismic activity are computed. The telescope under test is mounted on one end of a 30 meter long seismic block and the star simulating collimating mirror on the other end, with its light source (in the mirror focal plane) located in front of the telescope. The seismic block rests on compacted sand. Both constitute a resonant system.

10.4.1.3.2 Conclusion

It can be shown that even without accounting for magnification at resonant frequency of the foundation-block combination, the desired system accuracy of 0.003 arc second maximum mirror rotation will not be met. Magnification between eight and fifty is possible at resonance.

10.4.1.3.3 Discussion

The above conclusion is based on the numerical evaluation presented here.

From measurements listed in the Survey of Microseismic Activity⁽¹⁾ it can

(1) L.M. Murphy: "A Survey of Microseismic Activity," Earthquake Notes, Vol. XX, No. 3, 1949.

be seen that the period of micro seismic waves varies from 0.03 - 9 seconds with amplitudes varying from 0.5 to 5.0 microns. Using representative figures of 0.05 second for the period and 1 micron (1×10^{-6} meter) for the amplitude, the input frequency $f = \frac{1}{0.05} = 20$ Hz. Since the wave velocity v given in Ref. (1) is $V = 3000$ meter/sec, the wavelength corresponding to 20 Hz is $\lambda = \frac{V}{f} = \frac{3000}{20} = 150$ meter.

Assume that the 30 meter block rests on the crest of the halfwave:

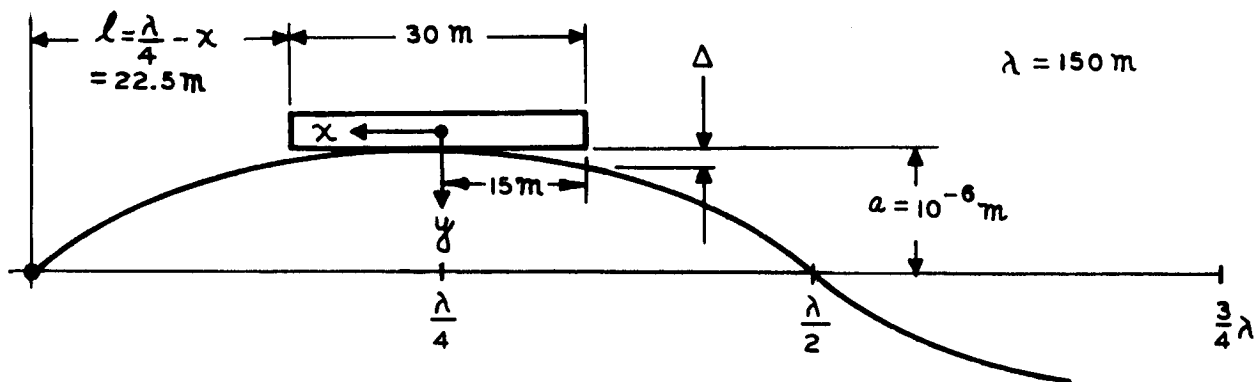


Figure 10.4.1.3-1 Test Bed on Microseismic Wave

The wave amplitude in inches

$$a = (39.37) (10^{-6})$$

Then

$$\frac{22.5}{37.5} = 0.6 = \frac{x^\circ}{90^\circ}$$

$$x^\circ = 54^\circ$$

$$\sin 54^\circ = 0.80902$$

and the deviation of the wave from the crest at the extremities of the block is

$$\Delta = (1 - 0.80902) (39.37) (10^{-6}) = 7.55 \times 10^{-6} \text{ inch}$$

The angular deviation from the horizontal at the same point on the wave is now obtained as follows:

With

$$l = \frac{\lambda}{4} = 37.5 \text{ m}$$

$$y = a \sin \frac{\pi x}{2l}$$

$$y' = a \left(\frac{\pi}{2l} \right) \cos \frac{\pi x}{2l}$$

With

$$\frac{\pi x}{2l} = 54^\circ, \cos 54^\circ = 0.588$$

$$y' = \tan = \frac{(1 \times 10^{-6}) \pi (0.588)}{(2) (37.5)} = 2.48 \times 10^{-8}$$

For small angles, the tangent is the same as the angle, therefore

$$\theta = 2.48 \times 10^{-8} \text{ radians} = 0.0051 \text{ arc second}$$

This angle θ represents the deviation of the mirror from the vertical; thus the light beam is turned through an angle 2θ by this means.

The telescope and light source at the other end of the block also turn through an angle θ , producing a rotation of the beam relative to the telescope of 2θ , if the mirror had remained stationary, and a total relative rotation of 4θ including the mirror rotation. After a half period $t = \frac{T}{2}$ the angle θ reverses and an overall relative rotation of 8θ results, excluding any magnification factor.

$$x = 8\theta = (8) (0.0051) = 0.0408 \text{ arc seconds}$$

It is obvious that x is much larger than the desired error resolution of 0.003 arc second shown in figure 4.12-1.

It remains to be shown that the foundation-block combination follows the motion of the seismic wave. This is done by computing the resonant frequency of the foundation block combination and comparing it with the input frequency of the seismic wave.

According to DEGEBO studies⁽¹⁾ using 6000 lb (3T) with a contact area of 10.7 ft², the measured and computed resonant frequency of non uniform compacted sand

$$f_n = 26.7 \text{ Hz}$$

The foregoing test was based on a bearing pressure

$$P = \frac{6000 \text{ lb}}{10.7 \text{ ft}^2} = \frac{3 \text{ T}}{10.7} = \frac{0.28 \text{ T}}{\text{ft}^2}$$

(1) G.P. Tschebotarioff, Solid Mechanics Foundations and Earth Structures, McGraw Hill, N.Y., 1951

whereas, the proposed block produces a bearing pressure

$$p = \frac{(10)(100)(10)(150)}{(10)(100)} = \frac{1500 \text{ lb}}{\text{ft.}^2} = \frac{0.75 \text{ T}}{\text{ft.}^2}$$

Using the method outlined by Tschebotarioff (pp. 586-587), the equivalent resonant frequency of the foundation - block (fn) is obtained

$$f_n' = \frac{f_{nr}}{\sqrt{P}} = 26.7 = \frac{f_{nr}}{\sqrt{0.28}} = \frac{f_{nr}}{0.53} = 1.88 f_{nr}$$

$$f_{nr} = \frac{26.7}{1.88} = 14.2$$

$$f_n = \frac{f_{nr}}{\sqrt{0.75}} = \frac{14.2}{\sqrt{0.75}} = \frac{14.2}{0.86} = 16.5 \text{ Hz.}$$

Where, f_{nr} denotes the resonant frequency based on a unit bearing pressure

$$\left(1 \frac{\text{T}}{\text{ft}^2}\right)$$

The resonant frequency of the proposed foundation-block combination is 16.5 Hz and is therefore in the same range as the seismic wave input frequency of 20 Hz.

An extended examination of means to minimize disturbances caused by micro-seisms will be performed in phase B of the OTAES study. It will encompass two directions:

- a. Determination of localities with minimum microseismic transmission (mountain areas) and optical bench stability computation for them.
- b. Decoupling of the seismic block from its supporting soil element using supports which will filter out the predominant microseismic frequency. For this configuration the optical bench stability will also be computed, taking into consideration magnification due to resonance.

10.4.1.4 Dual Aperture Light Modulator

10.4.1.4.1 General

The light modulator provides a means of accurate image position deviation measurement from a stable and repeatable null position. The high stability of the modulator and the AC characteristic of the generated signals which permit stable amplification and phase discrimination constitute the main advantages of this fine error sensing method.

In this system, two elongated apertures are sinusoidally oscillating at different frequencies about two orthogonal axes (pitch and yaw) as shown in figure 10.4.1.4-1. The dual aperture light modulator was derived from a dual fork scanner assembly, each fork operating with its electronic circuitry in a self-starting oscillator configuration, compensated for amplitude and frequency change with temperature. The following description and analysis also applies to an image dissector tube wherein the electromagnetic or electrostatic electron beam deflection in two axes across one stationary square aperture substitutes for the two mechanical aperture motions.

When a star image appears in the field of view, the aperture encodes two-axes position information upon the emerging star light. This modulated light signal is then converted by the photomultiplier tube (see figure 10.4.1.4-1) into electrical signals for use in the error detection circuit. The light modulator also generates electrical reference signals, f_y and f_z , for use in the synchronous demodulation of the corresponding errors, pitch (y) and yaw (z), respectively.

The required 1.0 X 1.0 second-of-arc field of view for the fine pointing system will be derived from the following parameters:

- Optics, EFL $\cong$ 2900 inch,
- Light modulator aperture width, $W = 0.014$ inch.
- Scanner double amplitude deflection, $2 A = 0.020$ inch.
- Scanner frequencies, $f_y = 350$ Hz and $f_z = 450$ Hz.

10.4.1.4.2 Analysis

Useful tracking and recognition signals can be derived from the dual scanning technique. This technique uses two superimposed, elongated slits, oscillating about two axes at 90 degrees to each other in the focal plane of the tracker telescope. The two frequencies $f(y)$ and $f(z)$ are chosen to be non-integral multiples of each other so that the sum and difference terms of the signal generated, as shown in the following Fourier analysis (figure 10.4.1.4-3), can be kept outside the pass bands of the tracker narrow band amplifiers. The light flux that passes through one aperture is a function of the time frequency and the image position with respect to the center of oscillation.

The developed waveforms for one axis are shown in figure 10.4.1.4-3 for the star position in three discrete regions.

The displacement of the slot, measured from the center of oscillation, is given by:

$$d = A \sin \omega_y t$$

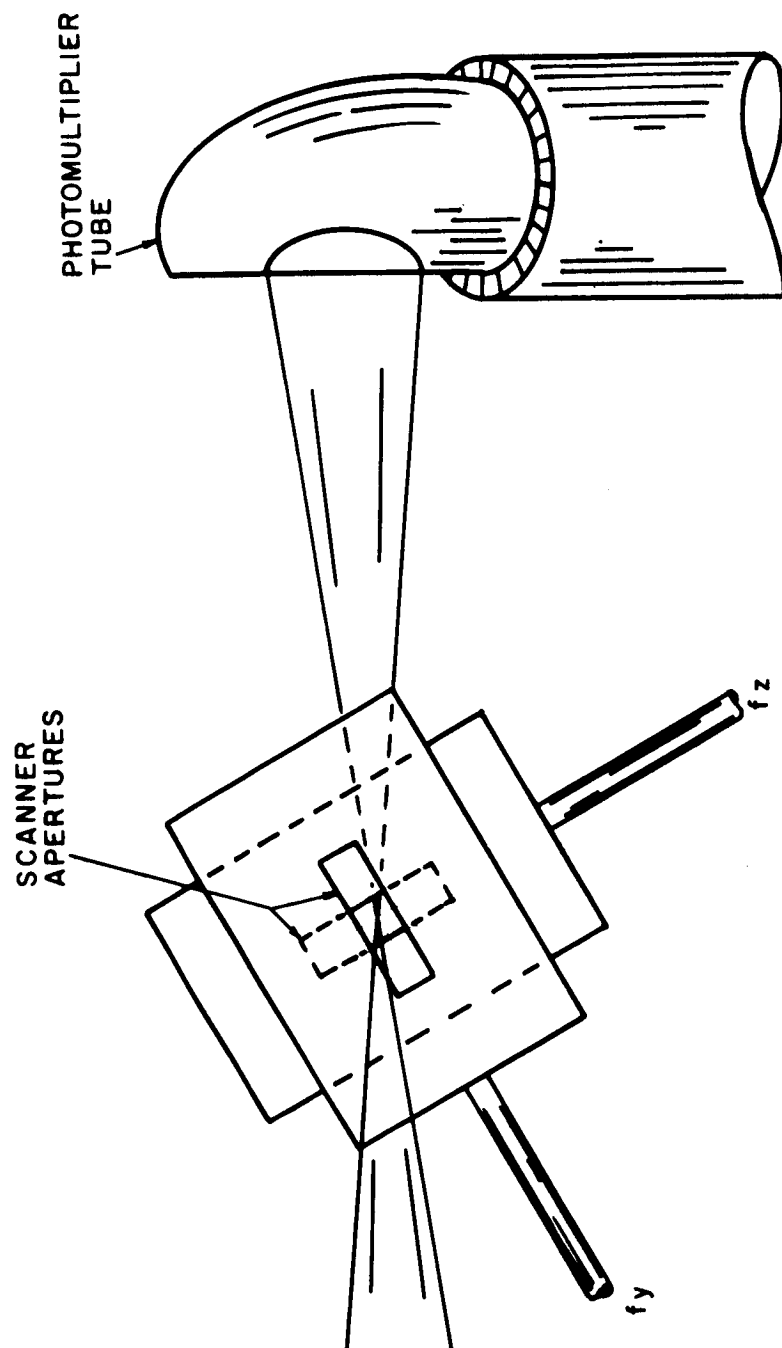


Figure 10.4.1.4-1 Crossed Aperture Light Modulator and Sensor

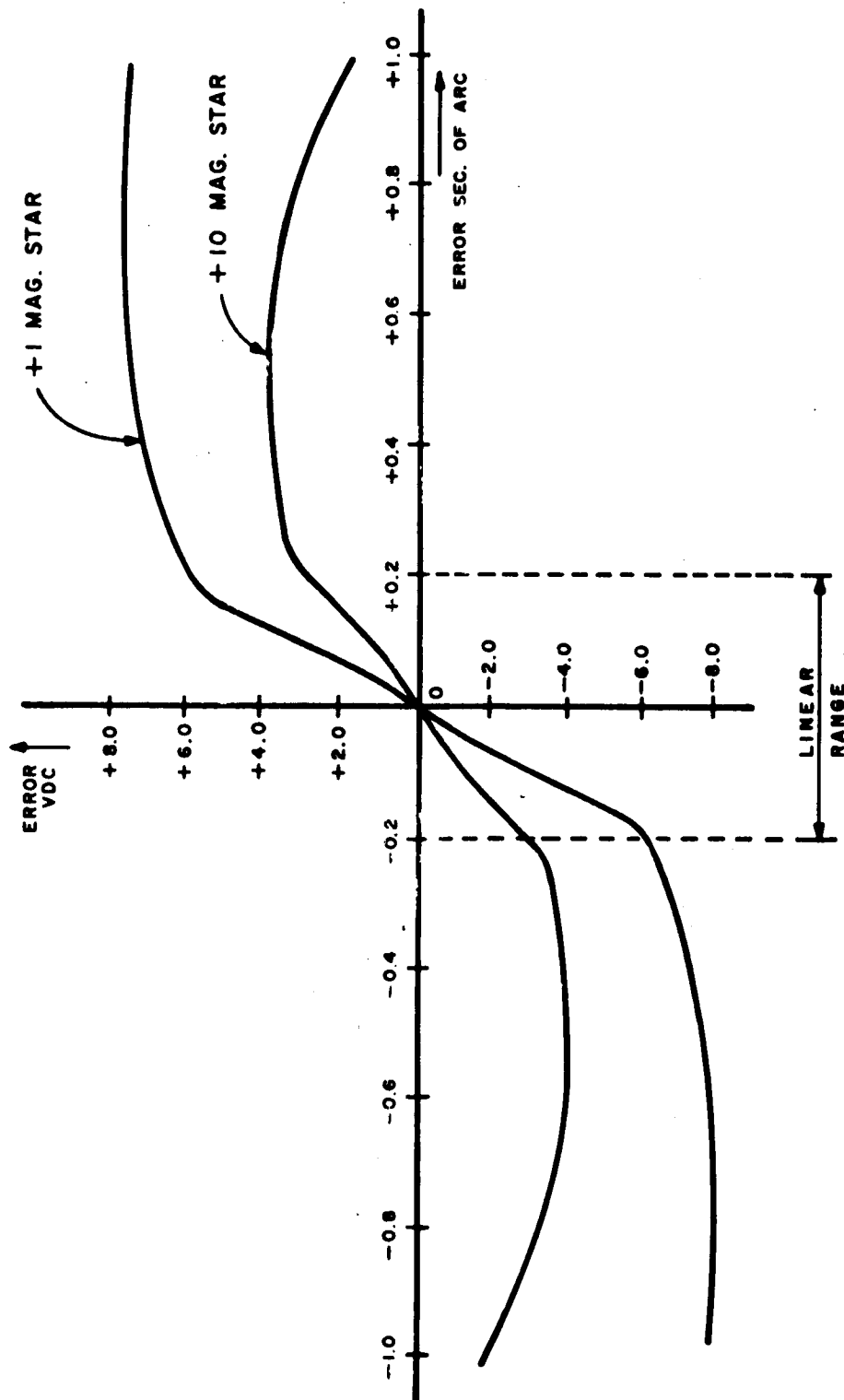


Figure 10.4.1.4-2 Single Axis Error Transfer Function, Crossed Aperture Image Position Encoder

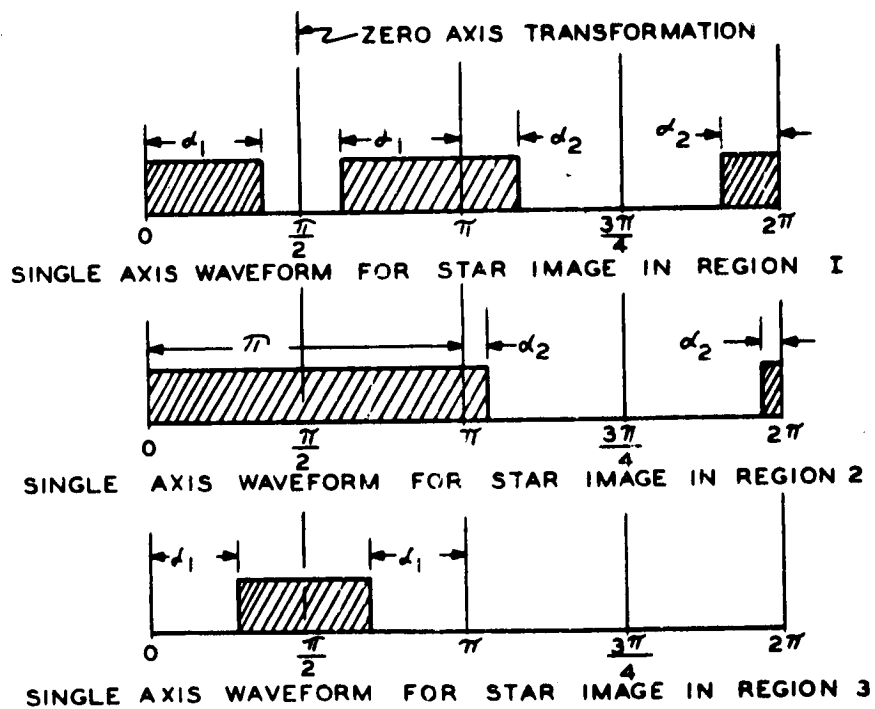
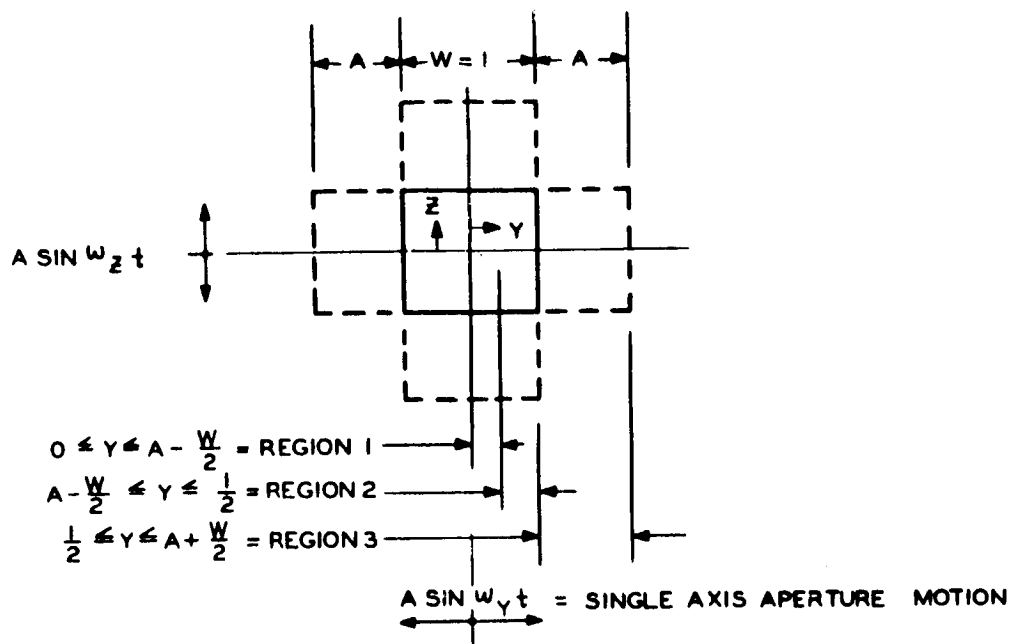


Figure 10.4.1.4-3 Single Axis Waveforms

In region 1:

$$\alpha_1 = \sin^{-1} \frac{(y + W/2)}{A}$$

$$\alpha_2 = \sin^{-1} \frac{(W/2 - y)}{A}$$

In region 2:

$$\alpha_2 = \sin^{-1} \frac{(W/2 - y)}{A}$$

In region 3:

$$\alpha_1 = \sin^{-1} \frac{(y - W/2)}{A}$$

By axis transformation the developed signal waveforms, shown in Figure 10.4.1.4-3, represent an even function having the Fourier series:

$$\tau_y(t) = f_0(y) + f_1(y) \cos \omega_y t + f_2(y) \cos 2 \omega_y t + \dots$$

where:

$f_0(y)$ = dc component of transmitted energy

$f_1(y)$ = fundamental coefficient

$f_2(y)$ = second harmonic coefficient

$\tau_y(t)$ = displacement of the star image from null measured along the y axis, with respect to time

The motion of the z axis is superimposed upon the light flux energy ϕ in the same fashion as described for the y axis. Proper selection of ω_y and ω_z enables coordinate transformation of the z axis to coincide with the y axis such that both the y and z functions become even.

Therefore:

$$\tau_{yz}(t) = \tau_y(t) \cdot \tau_z(t)$$

$$\begin{aligned} \tau_{yz}(t) = & f_0(y) f_0(z) + f_0(z) f_1(y) \cos \omega_y t \\ & + f_0(z) f_2(y) \cos 2 \omega_y t + f_0(y) f_1(z) \cos \omega_z t \\ & + f_0(y) f_2(z) \cos 2 \omega_z t \\ & + \text{sum and difference terms} \end{aligned}$$

The significant flux components that pass the narrow band limitation in y and z are:

	<u>Signals used for</u>	
$f_0(y) f_0(z)$	average flux	
$f_0(y) f_1(z) \cos \omega_z t$	= z axis	position and recognition
$f_0(y) f_2(z) \cos 2 \omega_z t$	= z axis	recognition
$f_0(z) f_1(y) \cos \omega_y t$	= y axis	position and recognition
$f_0(z) f_2(y) \cos 2 \omega_y t$	= y axis	recognition

The results of this Fourier analysis for the signal components are given in figure 10.4.1.4-4. Figure 10.4.1.4-5 indicates the plot of the d-c fundamental and second harmonic components as a function of star displacement (y) using a single aperture.

Superposition of the two apertures, one for each axis, modifies this response as shown in figure 10.4.1.4-6.

A study of the second harmonic component in region 1, reveals that its maximum occurs at a reed amplitude: $A = \sqrt{\frac{W}{2}}$.

For identical apertures of a width w oscillating in a plane of two orthogonal axes with an amplitude A, the coverage in each axis is $2A + W$; this is also the length of the complementary aperture. A total field of $(2A + W) \times (2A + W)$ is successively scanned by an instantaneous field opening $w \times w$.

10.4.1.5 Signal-to-Noise Analysis

The object of the Fine Guidance Experiment is to test telescope pointing capabilities of 0.01 arc second using target stars of +10 magnitude or brighter. This technology will be required for the 3 meter diameter (or larger) future space telescope. The following analysis will establish the collecting aperture needed for noise limited resolution of 0.003 sec of arc using the scanning and detection method previously described. This analysis, when applied to the 3 meter telescope will provide a higher sensitivity with resolution capability comparable to the Fine Guidance experiment.

The experiment telescope proposed for the Fine Guidance experiment will provide a ± 1 arc second field of view using the scan method analyzed in subsection 10.4.1.4.2 with aperture dimensions $W = 0.32 \times 10^{-3}$ inch and scan double-amplitude $2A = 0.46 \times 10^{-3}$ inch.

	Region 1	Region 2	Region 3
Limits	$0 \leq y \leq A - \frac{W}{2}$	$A - \frac{W}{2} \leq y \leq w/2$	$w/2 \leq y \leq A + \frac{W}{2}$
DC Signal	$\left[\sin^{-1} \frac{w/2 - y}{A} + \sin^{-1} \frac{y + w/2}{A} \right] \frac{1}{\pi}$	$1/2 + \frac{1}{\pi} \sin^{-1} \frac{w/2 - y}{A}$	$1/2 - \frac{1}{\pi} \sin^{-1} \frac{y - w/2}{A}$
Fundamental Component	$\frac{2}{\pi} \left[\sqrt{1 - \frac{(w/2 - y)^2}{A^2}} - \sqrt{\frac{(w/2 + y)^2}{A^2}} \right]$	$\frac{2}{\pi} \sqrt{1 - \frac{(w/2 - y)^2}{A^2}}$	$\frac{2}{\pi} \sqrt{1 - \frac{(y - w/2)^2}{A^2}}$
Second Harmonic Component	$\frac{2}{\pi} \left[\frac{y + w/2}{A} \sqrt{1 - \frac{(y + w/2)^2}{A^2}} + \frac{w/2 - y}{A} \sqrt{1 - \frac{(y - w/2)^2}{A^2}} \right]$	$\frac{2}{\pi} \left[\frac{w/2 - y}{A} \sqrt{1 - \frac{(w/2 - y)^2}{A^2}} \right]$	$\frac{2}{\pi} \left[\frac{w/2 - y}{A} \sqrt{1 - \frac{(w/2 - y)^2}{A^2}} \right]$

Figure 10.4.1.4-4 Fourier Analysis Results of Single Axis Modulation

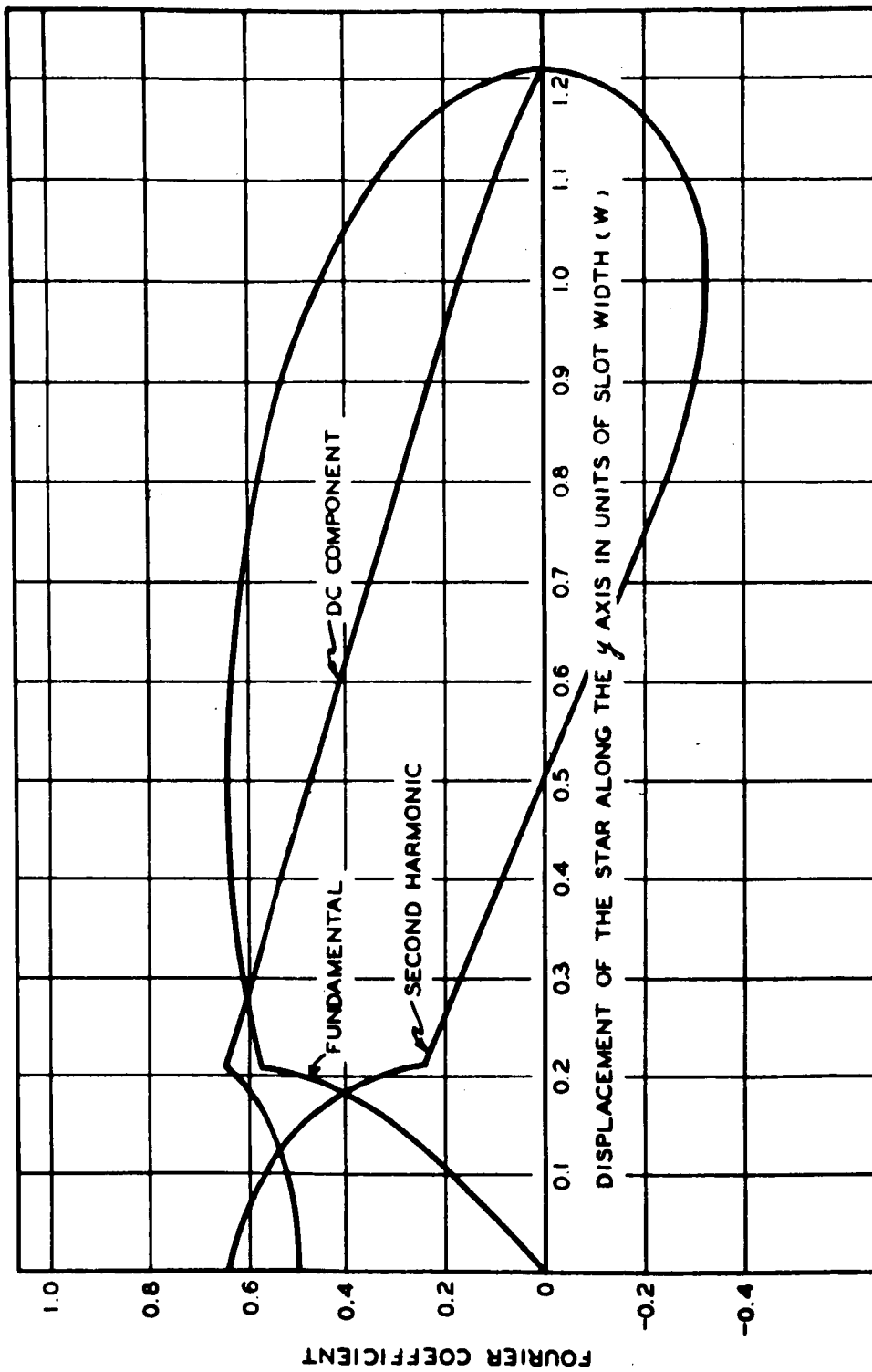


Figure 10.4.1.4-5 DC, Fundamental and Second Harmonic Amplitudes, Single Axis

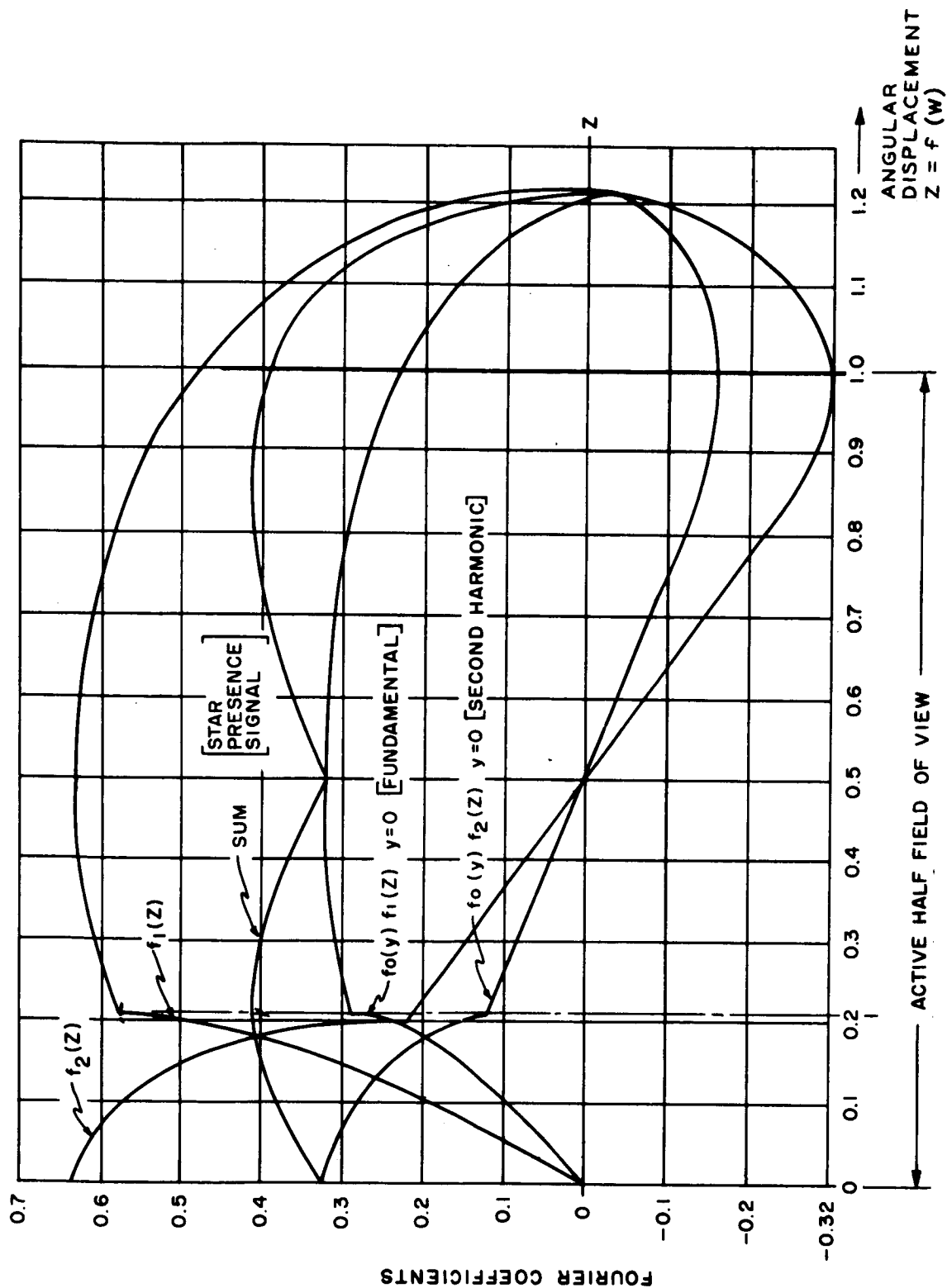


Figure 10.4.1.4-6 Two Axis Scanner, Waveform Amplitudes

Actual transfer functions for the +1 and +10 magnitude stars as shown in figure 10.4.1.4-2 were obtained by scaling down the field size in the Kollsman built Fine Guidance for the Goddard Experiment Package which was designed for the Orbiting Astronomical Observatory B.

For our computation the linear approximation of the transfer function as shown in figure 10.4.1.5-1, will be used. It shows a linear error range from -0.2 arc second to +0.2 arc second and is comparable to the theoretical transfer characteristic shown in figures 10.4.1.4-5 and 10.4.1.4-6. At an error $E = 0.2$ arc second the signal current I_s is maximum and the signal-to-noise ratio S/N is also maximum. It is desirable to have a one sigma resolution limit of ϵ resolution = 0.003 arc second, for which the signal-to-noise ratio $S/N = 1$. This requires a maximum signal-to-noise $S/N_{\max} = 67$ (at the end of the linear range).

The $S/N_{\max}$ will be computed for an A0 spectral class star ($T \cong 11,000$ K), and a four-alkali (extended S-20) photomultiplier tube with a UV transmitting window. The photocathode spectral response $\sigma_k(\lambda)$ is shown in figure 10.4.1.5-2.

$$S/N_{\max} = \frac{I_{s \text{ max rms}}}{\sqrt{2 e \Delta f \Sigma I_{dc \text{ noise}} + I_{tn \text{ rms}}^2} \sqrt{\Delta f}}$$

where

$I_{s \text{ max rms}}$ = maximum photocathode ac error signal output

$e = 1.6 \times 10^{-19}$ (coulombs), electron charge

$\Delta f = 1$ Hz system closed loop bandwidth

$\Sigma I_{dc \text{ noise}} = I_{dc \text{ sn}} + I_{dc \text{ bn}}$ Noise generating current due to signal and due to background

$I_{tn \text{ rms}}$ = Photomultiplier tube noise per 1 Hz bandwidth

Signal Current

Calculated for a +10 magnitude star

$$I_{s \text{ max rms}} = \frac{A_e}{\sqrt{2}} A \eta P_{\max} \int_0^{\infty} \left[\frac{P(\lambda)}{R(\lambda)} \right]_{\max} \sigma_k(\lambda) d\lambda$$

where $A_e = 0.29$ is the modulation coefficient as described in subsection 10.4.1.4.2 and obtained from figure 10.4.1.4-6 for the fundamental at the end of the approximately linear range.

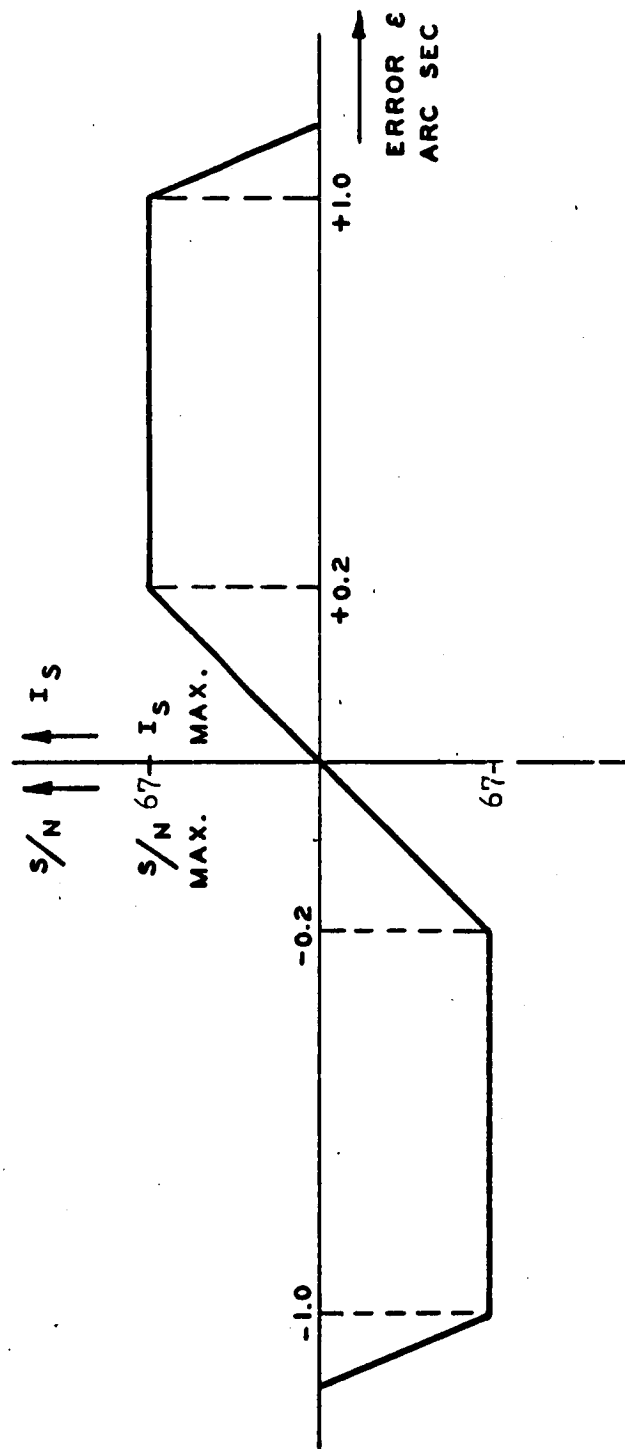


Figure 10.4.1.5-1 Linear Approximation of Error Transfer Function

A = collecting area of the Fine Guidance Experiment telescope assigned to fine pointing (cm²).

$\eta = 0.8$ average efficiency of the telescope

$$P_{\max} = \frac{1}{A_{\lambda}} \times P_{\lambda} \quad \text{peak black-body star radiation}$$

$A_{\lambda} = 0.39$ coefficient for 11,000°K star (see Figure 10.4.1.5-2)

at $\lambda = 5400 \text{ \AA}$

for $\lambda = 5400 \text{ \AA}$, $P_{\lambda} = 3.8 \times 10^{-7} \text{ (erg/cm}^2\text{/sec/100\AA)}$ for a 0 mag star outside earth atmosphere

for a + 10 mag star $P_{\lambda} = 1.9 \times 10^{-17} \text{ (W/cm}^2\text{/500\AA)}$

$P_{\lambda} / P_{\lambda \max}$ is the A0 star black body relative radiation as a function of wavelength is plotted in figure 10.4.1.5-2.

$\sigma_K(\lambda)$ is the photomultiplier cathode spectral response (see Figure 10.4.1.5-2, for Type 641E EMR tube). For this calculation the typical values were used.

A graphical summation of $P_{\lambda} / P_{\lambda \max} \Delta\lambda \times \sigma_K(\lambda) \Delta\lambda$ was performed in place of the integration, using a

$$\Delta\lambda = 500\text{\AA}:$$

$$\sum_{\lambda = 2500\text{\AA}}^{\lambda = 7500\text{\AA}} P_{\lambda} / P_{\lambda \max} \Delta\lambda \times \sigma_K(\lambda) \Delta\lambda = 0.115 \text{ amp} \times 500\text{\AA}/\text{watt}$$

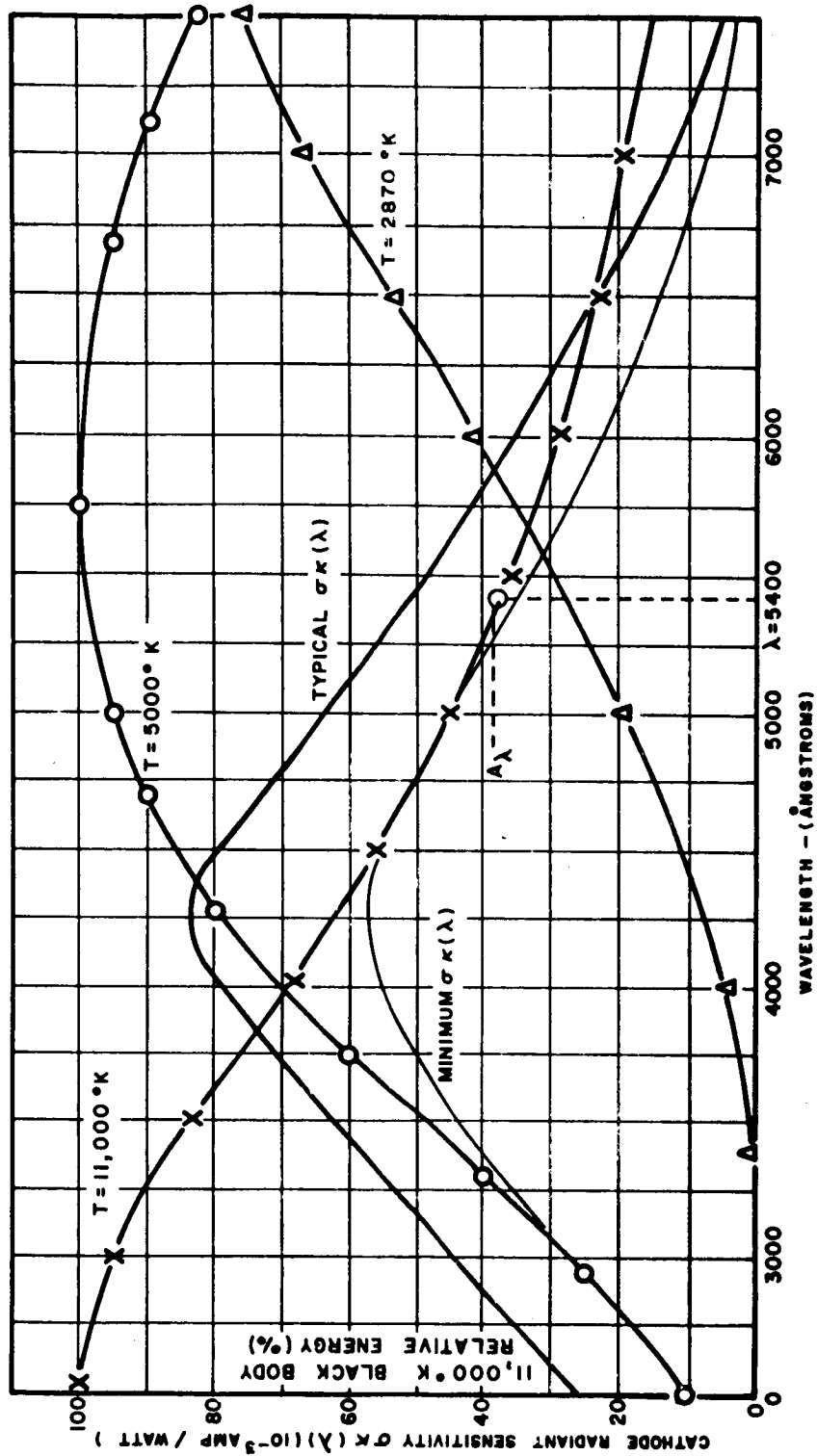
Thus,

$$I_{S \max \text{ rms}} = 0.9A \times 10^{-18} \text{ (Amps)}$$

where A is the collecting area in cm²

$$I_{\text{dcs}} = I_{S \max \text{ rms}} \cdot \sqrt{2} \times \frac{A_{\text{av}}}{A_e}$$

$$I_{\text{dcs}} = 1.75 A \times 10^{-18} \text{ (Amp)}$$



X-X $T = 11,000^\circ\text{K}$
 O-O $T = 5,000^\circ\text{K}$
 Δ-Δ $T = 2,870^\circ\text{K}$

$P_\lambda/P_\lambda \text{ Max.}$ Black Body Radiation vs. Wavelength

Figure 10.4.1.5-2 Photomultiplier Spectral Response

From the graph in figure 10.4.1.4-6 the average star flux factor

$$A_{av} = 0.4.$$

Background Illumination Noise

Background illumination impinging upon the photocathode generates a d-c current:

$$I_B = B \omega A \eta \sigma_k (5000^\circ\text{A})$$

$$= 0.59 A \times 10^{-23} \text{ amps with } A \text{ in } \text{cm}^2$$

where

$$B = 9.1 \times 10^{-12} \text{ (8) watt/Sterad/cm}^2 \text{ mean space background radiation}$$

ω = solid angle subtended by the Fine Pointing Sensors

$$= 2.3 \times 10^{-11} \text{ (steradian)}$$

Using the star distribution graph in figure 10.4.1.5-3 the mean background star class was established. Using the Herzsprung diagram in figure 10.4.1.5-4, this mean spectral class indicated an equivalent black body radiation at $T = 5000^\circ\text{K}$. Then, $\sigma_k = 35 \times 10^{-3} \text{ A/W}$, from figure 10.4.1.5-2 with 5750 Angstroms peak spectral response.

Star Noise

Noise contribution of star current is a function of the dc value of the star signal. This d-c signal is computed in the Fourier analysis of the light modulator in subsection 10.4.1.4.2.

Tube Noise

The photomultiplier tube equivalent noise input (ENI) is defined as the luminous flux input at 2870°K chopped on and off (50% duty cycle) at a low audio frequency, which will produce a signal of the same rms value, in a 1 Hz bandwidth, as does the noise. This noise is a function of PMT current amplification and ambient temperature.

$$P \text{ noise at } 4200^\circ\text{A} = 3.5 \times 10^{-16} \text{ Watt for } \Delta f = 1 \text{ Hz,}$$

$$\text{for } 10^6 \text{ current amplification at } 20^\circ\text{C}.$$

(8) Computed from C.W. Allen: Astrophysical Quantities, The Athlone Press, London, 1963.

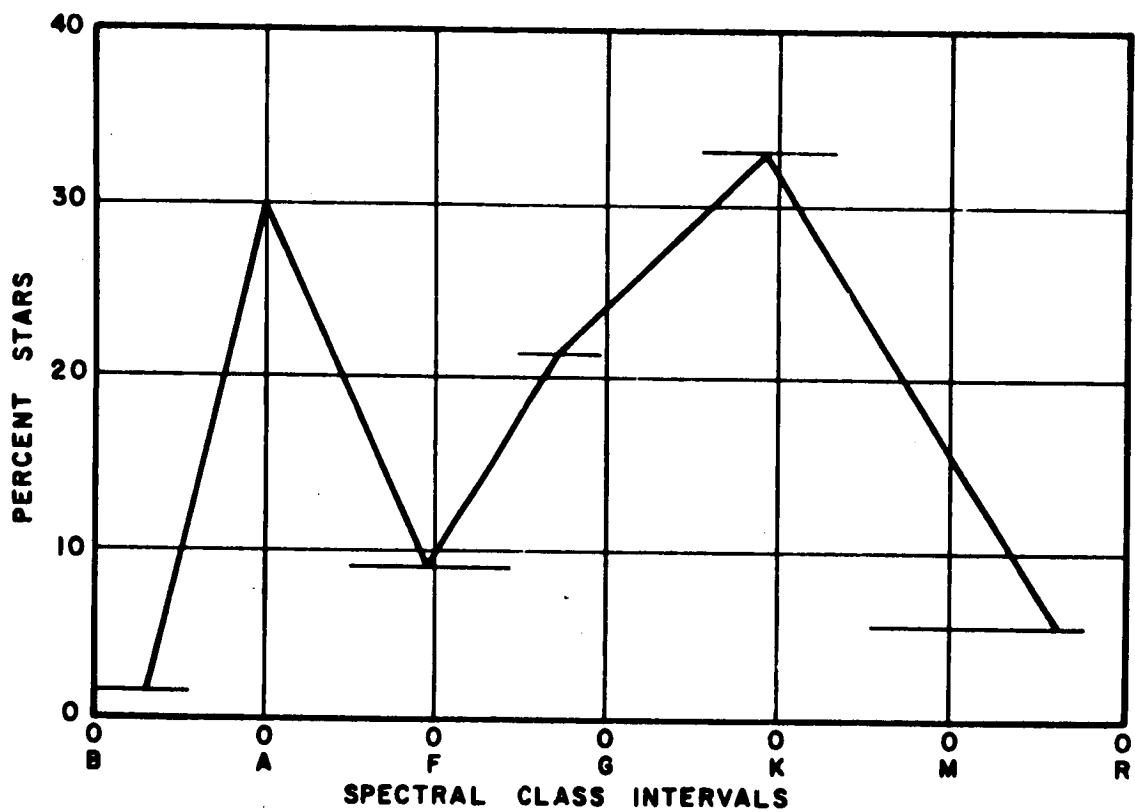


Figure 10.4.1.5-3 Star Distribution as a Function of Spectral Classes (Based on Henry Draper Catalogue - Using 222, 750 Brightest Stars)

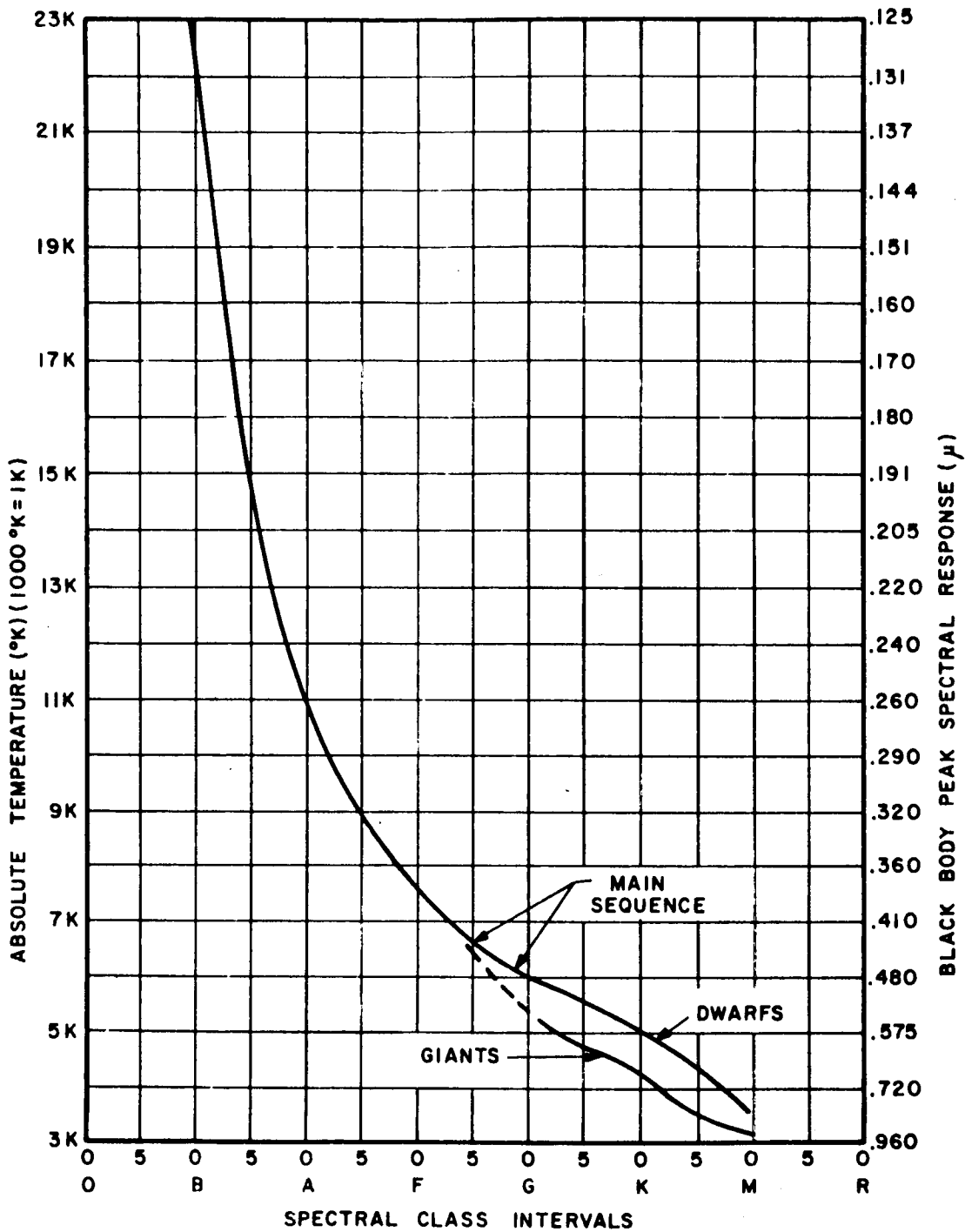


Figure 10.4.1.5-4 The Color Temperatures of the Stars

(From: Astronomy by Russel, Stewart and Dugan, Vol. II, Rev. Ed. 1955)

A cathode average sensitivity at 2870°K of $\sigma_K = 7.88 \times 10^{-3}$ Amp/Watt was graphically computed using the $R_N/R_{N_{max}}$ curve for $T = 2870^\circ\text{K}$ in Figure 10.4.1.5-2. The rms tube noise current for a square wave with 50% duty cycle is

$$I_{tn \text{ rms}} = P_{\text{noise}} \times \frac{2}{\pi} \times \frac{1}{\sqrt{2}} = 1.3 \times 10^{-18} \text{ Amp}$$

$$S/N_{\text{max}} = \frac{I_{s \text{ rms max}}}{\sqrt{2e \Delta f \sum I_{dc \text{ noise}} + I_{tn}} \sqrt{\Delta f}}$$

= 49.7, using previous values

This is computed for a +10 magnitude star in a 1 Hz bandwidth and using 40 percent of the total collecting area of the proposed fine guidance experiment telescope. Twenty percent of the light is utilized by the intermediate sensor, and the remaining 80 percent is divided equally between the two fine error sensors. (The total area is 4575 cm², allowing (0.4) (4575) = 1830 cm² for each fine sensor.)

For a magnitude change of -0.7, both the RMS signal current, $I_{\text{rms max}}$, and the d-c current due to starlight, I_{dcs} , are increased by a factor of $2.51^{(0.7)} = 1.82$, resulting in a signal-to-noise ratio of 69 at a star magnitude of +9.3. Thus only stars of +9.3 magnitude and brighter will full-fill the postulated pointing requirement. An extrapolation for the large future telescope (120 inch diameter) in which up to 50 percent collected light energy may be used for error sensing⁽¹⁾ shows that under this condition the Fine Guidance can achieve 0.01 arc second pointing for stars down to +11.3 magnitude. For dimmer target stars, the offset pointing method will be applied using the brightest star within 1 to 2 degrees from the target. The same technology provided by this experiment will be applicable. The additional requirement will be placed upon angle readout instruments with high resolution and stability compatible with the required pointing.

(1) Boeing Report, *ibid.*

10.4.1.6 Dynamic and Structural Analysis of Fine Guidance Telescope

A preliminary analysis has been performed on the FGT to estimate the dynamic responses and resulting dynamic stresses induced during launch, to substantiate the design of the FGT structural configuration.

a. Dynamic Model

The FGT has been analyzed in two parts, considering cantilevered sections forward and aft of the gimbal ring (which is assumed fixed to displacement and rotation) as shown in figure 10.4.1.6-1

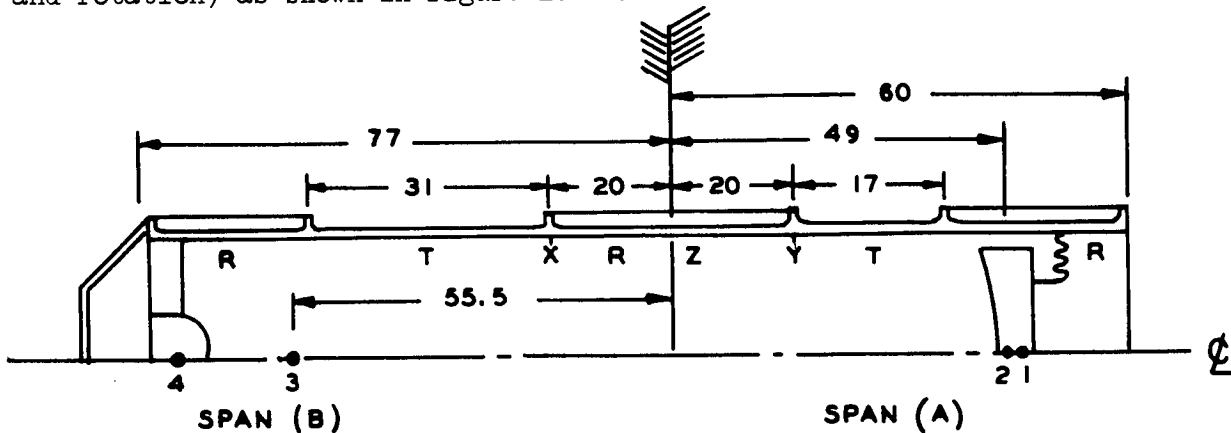


Figure 10.4.1.6-1 Dynamic Model of FGT

A two-degree of freedom lumped mass analysis is performed to predict the fundamental natural frequencies and dynamic loads in spans (A) and (B) of figure 10.4.1.6-1. In span (A), point (1) represents the weights of the telescope structure, error sensors and star tracker, and point (2) the weights of the primary mirror and supporting structure. In span (B), point (3) represents the weights of the star tracker, baffle mechanism, and part of the telescope structure, and point (4), the weights of the secondary mirror and supporting structure, cover, and part of the telescope structure.

The telescope structure comprises two cross-sections, labeled R (ribbed tube) and T (tube). Using the cross-section dimensions specified in section 10.2.1.3.5, the area moments of inertia were computed as:

$$I_R = 4072 \text{ in}^4 \quad (1)$$

$$I_T = 807 \text{ in}^4 \quad (2)$$

(b) Dynamic Analysis

To compute the fundamental natural frequencies of spans (A) and (B), the Eigenvalue problem was solved using matrix analysis. The Eigenvalue problem is expressed as

$$\begin{bmatrix} C \end{bmatrix} \begin{bmatrix} W \end{bmatrix} \{V\} = \frac{386}{\omega^2} \{V\} \quad (3)$$

where

$\begin{bmatrix} C \end{bmatrix}$ = influence coefficient matrix

$\begin{bmatrix} W \end{bmatrix}$ = weight matrix

$\{V\}$ = modal vector (Eigenvector)

ω = frequency - rad/sec = $\frac{f(\text{cps})}{2 \pi}$

The influence coefficient matrix $\begin{bmatrix} C \end{bmatrix}$ is composed of elements C_{ij} ($i = 1, N; j = 1, N$), which are the displacements at a mass point i resulting from a unit load applied at a mass point j . N is the total number of mass points. The weight matrix $\begin{bmatrix} W \end{bmatrix}$ is a diagonal matrix consisting of elements W_i ($i = 1, N$), which are the total weights lumped at the individual mass points.

Span (A)

The influence coefficient matrix is

$$\begin{bmatrix} C \end{bmatrix} = \begin{bmatrix} C_{11} & C_{12} \\ C_{21} & C_{22} \end{bmatrix} = 10^{-6} \times \begin{bmatrix} 2.7 & 2.7 \\ 2.7 & 5.4 \end{bmatrix} \text{ (in/lb)} \quad (4)$$

The weight matrix is

$$\begin{bmatrix} W \end{bmatrix} = \begin{bmatrix} W_1 & 0 \\ 0 & W_2 \end{bmatrix} = \begin{bmatrix} 176.8 & 0 \\ 0 & 157 \end{bmatrix} \text{ (lb)} \quad (5)$$

Solving the Eigenvalue problem of Equation (3) by iteration, the fundamental natural frequency of Span (A) is

$$f_n = 92 \text{ cps.} \quad (6)$$

$$\text{The modal vector is } \{V\} = \begin{Bmatrix} 0.631 \\ 1.0 \end{Bmatrix} \quad (7)$$

Span (B)

The influence coefficient matrix is

$$[C] = \begin{bmatrix} c_{33} & c_{34} \\ c_{43} & c_{44} \end{bmatrix} = 10^{-6} \times \begin{bmatrix} 3.56 & 5.64 \\ 5.64 & 11.0 \end{bmatrix} \quad (\text{in/lb}) \quad (8)$$

The weight matrix is

$$[W] = \begin{bmatrix} w_3 & 0 \\ 0 & w_4 \end{bmatrix} = \begin{bmatrix} 120 & 0 \\ 0 & 89.7 \end{bmatrix} \quad (\text{lb}) \quad (9)$$

The fundamental natural frequency of Span (B) obtained by numerical iteration of Equation (3) is

$$f_n = 84.4 \text{ cps} \quad (10)$$

$$\text{the modal vector is } \{V\} = \begin{Bmatrix} 0.545 \\ 1.0 \end{Bmatrix}$$

c. Response Analysis

The dynamic acceleration input to the gimbal ring is taken as the LEM Random Vibration Specification:

$$W_0 = 0.04 \text{ g}^2/\text{cps} \quad (49-950 \text{ cps})$$

To estimate the dynamic responses resulting from this input, the single degree of freedom response relationship is used, which assumes "white noise" random vibration over the entire frequency spectrum. This relationship is given by the expression

$$\ddot{y}_{rms} = \sqrt{\frac{\pi}{2} f_N Q W_0} \quad (11)$$

$\ddot{y}_{rms}$ = rms response acceleration - g's

f_N = natural frequency - cps

W_0 = power spectral density - g^2/cps

Q = amplification factor - assumed = 20

For most applications it is assumed that $\ddot{y}_{rms} = \sigma$ = standard deviation from the mean. The peak response is assumed as $\ddot{y}_{peak} = 3\sigma$.

Span (A)

Substituting the numerical quantities, $f_n = 92$ cps; $Q = 20$; $W_0 = 0.04$ g^2/cps ; into Equation (11),

$$(\ddot{y})_{rms} = \sigma = 10.8 \text{ g's}$$

$$\ddot{y}_{peak} = 3\sigma = 32.4 \text{ g's} \quad (12)$$

Span (B)

Substituting the numerical quantities

$f_n = 84.4$ cps; $Q = 20$; $W_0 = 0.04$ g^2/cps ; into Equation (11),

$$(\ddot{y})_{rms} = \sigma = 10.3 \text{ g's}$$

$$\ddot{y}_{peak} = 3\sigma = 30.9 \text{ g's} \quad (13)$$

d. Stress Analysis

The bending stresses, resulting from the dynamic loads computed in Equations (12) and (13), are computed at two locations in Spans (A) and (B) due to the change in cross-section. These locations are Y and Z in Span (A) and X and Z in Span (B), as shown in figure 10.4.1.6-1. The allowable bending stress is taken as $S_A = 20,000$ psi.

Span (A)

The total bending moment at Y, noting the load factor (LF) in Equation (12), is $M_y = (W_1 + W_2) l_{1,2-Y} (LF) = (176.8 + 157) (29) (32.4) = 313,638 \text{ lb-in.}$

The bending stress at Y, noting Equation (2), is $S_y = \frac{M_y C}{I_T} = \frac{313,638(16.031)}{807} = 6230 \text{ psi}$

The margin of safety is

$$\text{M.S.} = \frac{S_A}{S_y} - 1 = \frac{20,000}{6,230} - 1 = 2.2$$

At station Z,

$$M_Z = (W_1 + W_2) l_{1,2-Z} (LF) = (333.8) (49) (32.4) = 529,941 \text{ lb-in}$$

Noting Equation (1),

$$S_Z = \frac{M_Z C}{I_R} = \frac{529,941 (16.125)}{4072} = 2100 \text{ psi}$$

$$\text{M.S.} = \frac{S_A}{S_Z} - 1 = \frac{20,000}{2100} - 1 = 8.5$$

Span (B)

The total bending moment at X, noting the (L.F.) in Equation (13) is

$$M_x = [W_3 l_{3-x} + W_4 l_{4-x}] (L.F.) = [120(35.5) + 89.7(57)] (30.9)$$

$$= 290,000 \text{ lb-in}$$

$$S_x = \frac{M_x C}{I_T} = \frac{290,000 (16.031)}{807} = 5760 \text{ psi}$$

$$\text{M.S.} = \frac{S_A}{S_X} - 1 = \frac{20,000}{5,760} - 1 = 2.5$$

At station Z,

$$M_Z = \left[W_3 \ell_{3-Z} + W_4 \ell_{4-Z} \right] (L.F.) = \left[120(55.5) + 89.7(77) \right] (30.9)$$

$$= 419,200 \text{ lb-in}$$

$$S_Z = \frac{M_Z C}{I_R} = \frac{419,200 (16.125)}{4072} = 1660 \text{ psi}$$

$$M.S. = \frac{S_A}{S_Z} - 1 = \frac{20,000}{1660} - 1 = 11$$

The margin of safety is 2.2 in Span (A) and 2.5 in Span (B), thus validating the structural design.

10.4.2 Isolation Experiment

10.4.2.1 Requirements Analysis

The purpose of this subsection is to explain how the results presented in subsection 10.2.2.4.1 were derived. Basically, the several elements in the dynamics system, the spacecraft, the isolation system, the telescope control system, and the fine control system, are treated as isolated linear filters, and as illustrated in the sketch of that subsection only a single axis is considered. The derivation of the linear transfer functions, each of which is characterized by a static gain, a damping ratio and a natural frequency, is straight-forward. Only the disturbance input torques merit detailed discussion.

The starting point for deriving the power-density spectra illustrated in figure 10.2.2.4.1-1 is the experimental results recently obtained by Douglas⁽¹⁾ in a study of man-produced disturbances in a space vehicle. In that study subjects under simulated zero-g conditions were required to execute body and limb motions typically associated with routine housekeeping and experimental tasks of a future orbital research laboratory. The force resulting from each type of motion was recorded as a function of time. About 15 discrete types of movements were studied in this way. The experimental force time curves were analyzed for frequency content by means of Fourier analysis in that study. The results were presented in the form of tabulated sine and cosine coefficients for the first seven harmonics of the Fourier expansion for each type of movement. That is, the force is described by:

$$F(t) = A_0 + \sum_{n=1}^{\infty} A_n \cos n\omega t + B_n \sin n\omega t \quad (1)$$

where

ω is the fundamental frequency, A_0 , A_n , B_n are the tabulated coefficients.

This equation properly describes the measured force only over the time interval between $t = 0$ and $t_f = 2\pi/\omega$ that the measurement is taken. However, body and limb movements are inherently cyclic in nature. Therefore, it is assumed that each of the individual movements are repeated such that equation (1) describes the force at each point in time that is of interest, whether or not the point happens to fall within the fundamental period.

Equation (1) can be rewritten as

-
- (1) Experimental Study of Dynamic Effects of Crew Motion in a Manned Orbital Research Laboratory (MORL), Douglas Aircraft Company, Inc., Missile and Space Systems Division, Huntington Beach, California, October 1966.

$$F(t) = \sum_{n=1}^{\infty} \bar{C}_n e^{jn\omega t} \quad (2)$$

where

$$\begin{aligned} \bar{C}_n &= |\bar{C}_n| \angle \phi \\ |\bar{C}_n| &= \sqrt{A_n^2 + B_n^2} \\ \phi &= \tan^{-1} B_n/A_n \end{aligned}$$

We have about fifteen different types of disturbances, each of which is described by a different set of seven C_n coefficients. We are interested in describing the overall disturbance environment contributed by all disturbances occurring in a random manner. The most meaningful single number for this description, it would seem, is the rms value of the resulting disturbance. This is the approach taken here.

The total force contributed by M disturbances is assumed to be

$$F_T(t) = \sum_{i=1}^M \sum_{n=0}^7 \bar{C}_{in} e^{jn\omega t} \quad (3)$$

and in evaluating equation (3) phase information is discarded. The rms value of the force is

$$F_T \text{ (rms)} = \sum_{i=1}^M \sqrt{\frac{\sum_{n=1}^7 |\bar{C}_{in}|^2}{2}} \quad (4)$$

since the products of terms of different frequencies produce no average power.

The tabulated coefficients from the Douglas study include constant or d.c. terms. And these are used in obtaining the curves of figure 10.2.2.4.1-1. However, this is probably due to instrumentation errors and limitations to their technique. For an astronaut in a spacecraft to induce a net momentum into the spacecraft, without doing the same to himself, would appear to violate the momentum conservation law. Therefore, in deriving the results used here the zero-frequency terms are omitted.

In converting the results of equation (4) into disturbance torque data, certain assumptions were made. First, it is assumed that the moment arm through which each of the forces act is 6 feet, an average value for the baseline spacecraft configuration considered here. Second, two astronauts are assumed to be engaged in the typical motions for which the disturbances are defined.

One is involved in "body locomotion" in which the body as a whole is moved between control panels in the spacecraft cabin. The other astronaut is engaged in console operation and simple-pendulum arm movements, head movements, etc. associated with console operation.

In presenting the results, three cases are considered, as illustrated in figure 10.2.2.4.1-1. For the first case only one astronaut is involved, and he is restricted to movements that a constrained astronaut may execute, such as console torquing, limb movements, etc. That is, the "free" astronaut is immobilized. For the second case, the "free" astronaut moves but is restricted to velcro walking. For the third case, the "free" astronaut is free to move at will but the first astronaut is constrained. These three cases are thought to be representative of the practical situation that might prevail in a spacecraft.

The results of figure 10.2.2.4.1-2 were obtained by applying the discrete disturbance frequencies to the spacecraft control system. If the disturbance torque is described as follows

$$\bar{T}_D = \bar{T}_D(j\omega_1) + \bar{T}_D(j\omega_2) + \dots$$

where

T_D = disturbance torque, ft-lbs

$\omega_{1,2,\dots}$ = individual discrete frequencies

$$T_D(\text{rms}) = \sqrt{\frac{1}{2} \left[|T_D(j\omega_1)|^2 + |T_D(j\omega_2)|^2 + \dots \right]}$$

and the transfer function of the spacecraft control system is $G_{s/c}(j\omega)$

where

$$\theta(j\omega) = G_{s/c}(j\omega) T_D(j\omega)$$

then

$$\theta(\text{rms}) = \frac{1}{2} \sqrt{| \theta(j\omega_1) |^2 + | \theta(j\omega_2) |^2 + \dots}$$

It is assumed that the spacecraft transfer function is

$$G_{s/c}(s) = \frac{1/I_{s/c}}{s^2 + 2\zeta\omega_{s/c}s + \omega_{s/c}^2}$$

where

$I_{s/c}$ = moment of inertia = 15,000 lb-ft-sec²

$\omega_{s/c}$ = natural frequency, rad/sec

z = damping ratio

For numerical results, z is assumed to be 0.707 and $\omega_{s/c}$ variable.

Translation of the telescope support point on the spacecraft is

$$\mu_o \text{ (rms)} = \ell \theta \text{ (rms)} \quad (5)$$

where

μ_o = linear translation, feet

ℓ = distance from spacecraft center of rotation to telescope attach point, feet.

For numerical results ℓ is assumed to be 6 feet.

The upper curve of figure 10.2.2.4.1-2 is for single impulse having the momentum (about 15 ft-lb-sec) of the greatest disturbance (free soaring). This is to show the effect of lower frequencies on the response amplitude and to present a more realistic picture. The free soaring data given in the Douglas report, for example, is for virtually zero soaring time. That is, the astronaut pushes himself off and then stops his movement in one continuous motion. In practice, however, a soaring time of two seconds might prevail, in which case a fast spacecraft control system could respond to limit the amplitude excursion. Thus, the upper curve is plotted to show the maximum amplitude of the spacecraft excursion as a function of control system bandwidth. The curve is derived by taking the inverse transform of $G_{s/c}$ and finding its maximum.

The other dynamic elements in the system are treated in a similar manner. The transmissibility of the suspension is derived as follows. Let the spacecraft end of the suspension undergo the forced motion μ_o given by equation (5), the telescope of mass $M_{t/s}$ the motion μ , and the suspension system be represented as a spring constant K and viscous damping B .

The motion is described by the following differential equation.

$$M_{t/s} \frac{d^2 \mu}{dt^2} + B \left(\frac{d\mu}{dt} - \frac{d\mu_o}{dt} \right) + K (\mu - \mu_o) = 0$$

The transfer function of the suspension is

$$\frac{M(s)}{M_o(s)} = \frac{(B/M_t/s) S + K/M_t/s}{S^2 + (B/M_t/s) S + K/M_t/s}$$

where

M, M_o are the Laplace transforms of μ, μ_o and S is the complex variable.

In terms of the small signal parameters,

$$\frac{M(s)}{M_o(s)} = \frac{2z \omega_I S + \omega_I^2}{S^2 + 2z \omega_I S + \omega_I^2}$$

where

$$\omega_I^2 = K/M$$

$$2z \omega_I = B/M$$

For steady-state motion, where $s = j$

$$\frac{M(j\omega)}{M_o(j\omega)} = \frac{1 + j2z(\omega/\omega_I)}{1 - (\omega/\omega_I)^2 + j2z(\omega/\omega_I)}$$

The magnitude of this function is the transmissibility T .

$$T = \frac{M(j\omega)}{M_o(j\omega)} = \sqrt{\frac{1 + (2z\omega/\omega_I)^2}{[1 - (\omega/\omega_I)^2]^2 + (2z\omega/\omega_I)^2}}$$

The above equation is plotted in figure 10.2.2.4.1-3 for a wide range of damping ratios. All the curves approach a negative slope of 20 db/decade of frequency at higher frequencies, but the attenuation there is a function of the damping ratio, z , as shown in figure 10.2.2.4.1-4.

The telescope response μ (rms) is determined by applying each discrete $\mu_o(j\omega_n)$ to the transmissibility $T(\omega_n)$ and then summing the squares as before.

It is assumed that the telescope response motion μ causes an unbalance torque T_T given by

$$T_T = d M_t/s \ddot{\mu}$$

where

T_T = telescope torque, ft-lbs

d = offset, feet

$M_{t/s}$ = telescope mass, slugs

$\ddot{u}$ = telescope acceleration, ft/sec²

Numerically, $d = 1$ inch and $M_{t/s} = 50$ slugs for the results presented in figure 10.2.2.4.1-6.

The image stability curves of figure 10.2.2.4.1-7 are derived in an analogous manner under the assumption that

$$\phi(j\omega) = G_{t/s}(j\omega) T_T(j\omega)$$

and

$$\psi(j\omega) = G_m(j\omega) \phi(j\omega)$$

where

$\phi(j\omega)$ = telescope stability angle, rad

$$G_{t/s}(S) = \frac{1/I_{t/s}}{S^2 + 2z\omega_{t/s} S + \omega_{t/s}^2}$$

$I_{t/s}$ = moment of inertia, 250 slug-ft²

$\omega_{t/s}$ = natural frequency, 0.47 rad/sec

z = damping ratio, 0.707

$\psi(j\omega)$ = image stability, rad

$$G_m(S) = \frac{S^2}{S^2 + 2z_m \omega_m S + \omega_m^2}$$

And for the case with fine control

$$z_m = 0.7$$

$$\omega_m = 4.5 \text{ rad/sec}$$

whereas for the case without fine control

$$\omega_m = 0.$$

10.4.2.2 Orbital Forces

The orbital frame of reference is $\bar{i}$, $\bar{j}$, $\bar{k}$. $\bar{i}$ is tangent to the orbit and points in the direction of orbital travel, $\bar{k}$ points to the geocenter, and $\bar{j}$ completes the right-handed rectangular set. The origin is the spacecraft center of mass.

The X, Y, Z axes are the same spacecraft body axes used in the design weight analysis. The telescope suspension axes (with origin at the telescope center of mass) are defined by the three unit vectors.

$$\bar{i}_{xs}, \bar{i}_{ys}, \bar{i}_{zs}$$

parallel, respectively, to -X, Y, and -Z. The satellite center of mass is located relative to the telescope center of mass by the vector

$$-35.5 \bar{i}_{xs} + 64.0 \bar{i}_{ys} - 10.2 \bar{i}_{zs}$$

whose length is

$$[(35.5)^2 + (64.0)^2 + (10.2)^2]^{\frac{1}{2}} = 73.89$$

The space system is composed of two parts: a spacecraft with mass M_1 and a telescope with mass M_2 . The space system center of mass is the origin of the orbital reference and r_1 and r_2 are, respectively, vectors from this origin to the M_1 and M_2 centers. The external forces that act on each mass are:

$\bar{F}_g$, gravitational attraction of the earth

$\bar{F}_i$, impact forces - aerodynamic and solar

$\bar{F}_a$, mutual attraction forces

$\bar{F}_r$, restraining forces

We note that

$$\bar{F}_{a1} = \bar{F}_{a2}$$

$$\bar{F}_{r1} = \bar{F}_{r2}$$

$$\bar{F}_{i2} = 0, \text{ interior telescope location}$$

$$\bar{F}_{a1} + \bar{F}_{r1} + \bar{F}_{i1} + \bar{F}_{g1} = M_1 \bar{a}_1$$

$$\bar{F}_{a2} + \bar{F}_{r2} + \bar{F}_{g2} = M_2 \bar{a}_2$$

$$\bar{F}_{g1} + \bar{F}_{g2} + \bar{F}_{i1} = (M_1 + M_2) \bar{a}_0$$

$$\bar{F}_{g2} + \bar{F}_{g2} = (M_1 + M_2) \bar{g}_0$$

where $\bar{a}_0$, $\bar{a}_1$, $\bar{a}_2$ are, respectively, the accelerations of the satellite about the geocenter, the spacecraft, and the telescope; and g_0 is the acceleration of the system mass center due to earth's gravitational attraction. The main interest here is $\bar{F}_{r2}$ which is given by:

$$\bar{F}_{r2} = M_2 \bar{a}_2 - (\bar{F}_{a2} + \bar{F}_{g2})$$

We wish to calculate $\bar{F}_{r2}$ for various orientations in two circular orbits; 200 nmi altitude and synchronous altitude. The satellite, of course, is fixed with respect to the orbital reference frame.

Calculation of $\bar{a}_2$:

$$\begin{aligned} \bar{a}_2 &= \bar{a}_0 + \dot{\omega}(x_2 \bar{k} - z_2 \bar{i}) - \omega^2 (z_2 \bar{k} + x_2 \bar{i}) \\ x_2 \bar{i} + y_2 \bar{j} + z_2 \bar{k} &= \bar{r}_2 \end{aligned}$$

ω is the satellite's angular velocity about the geocenter

$\omega = 2\pi$ radians $\div$ orbital period in seconds

$\dot{\omega} = 0$, circular orbits

$$\bar{a}_0 = \bar{g}_0 + \bar{F}_I / (M_1 + M_2)$$

$\bar{g}_0$ is the acceleration of the satellite due to the gravitational attraction of the earth, so

$$\bar{g}_0 = (\mu/R_0^2) \bar{k}$$

$$\mu = 1.404 \times 10^{16} \text{ ft}^3/\text{sec}^2$$

R_0 is the satellite's radius vector

$M_1 + M_2$ is the mass of the whole satellite

$\bar{F}_I$, impact forces on satellite from dynamic pressure and solar wind
(see below)

Calculation of $\bar{F}_I$:

$$\bar{F}_I = (N/\sin a) \bar{i} + (c - N \tan a) \bar{U}$$

$\bar{U}$ is a unit vector, origin at system's center of mass, along the cylinder axis.

c is the chord force, flat plate face of cylinder

$$C = k f (\pi r^2) \cos^2 \alpha$$

N is the normal force, cylindrical side

$$N = 4/3 k f r L \sin^2 \alpha$$

$$k = 2$$

$$f = \rho v^2 + 9.56 \times 10^{-8} \text{ lbs/ft}^2$$

r = cylinder radius

L = cylinder length

ρ = aerodynamic density

v = orbital velocity

Calculation of $\bar{F}_{a2}$:

η and ξ are, respectively, the magnitudes of horizontal and vertical components of the attractive force. These depend on the ratio of the masses of spacecraft to telescope which is 19.87/1. The total attractive force is

$$(-\eta \cos \tau) \bar{l}_{ys} + (-\eta \sin \tau) \bar{l}_{xs} + \xi \bar{l}_{zs}$$

where

$$\tau = \tan^{-1} 64.0/35.5 \text{ (angle between } \eta \text{ and the } \bar{l}_{ys} - \bar{l}_{zs} \text{ plane).}$$

Calculation of $\bar{F}_{g2}$:

$$\bar{F}_{gs} = -g_0 \left[\bar{r}_2/R_0 - (1 + 3z_2/R_0) \bar{k} \right] M_2$$

g_0 has been calculated previously

R_0 is the orbit radius

$$\bar{r}_2 = x_2 \bar{i} + y_2 \bar{j} + z_2 \bar{k}$$

Other calculations:

Orbital Velocity

$$V = \sqrt{\mu/R_0}$$

Orbital angular velocity

$$\omega = \sqrt{\mu/R_0^3}$$

$$\text{Orbital Period} = 2\pi\sqrt{R_0^3/\mu}$$

Orientation sequences and transformations:

Two orientation sequences are used. One, the telescope is rotated through a full circle about its "natural position" in orbit. Second the telescope is pointed at the forward horizon and then rotated to the rear horizon. These two sequences give a reasonable simulation of pointing the telescope at astronomical objects and at terrestrial points.

First orientation sequence:

This sequence begins with the suspension axes parallel to the orbital reference axes. To accomplish the "full circle" rotation,

$$\bar{l}_{xs} \text{ is left parallel to } \bar{i}$$

and

$$\bar{l}_{zs} \text{ is rotated about } \bar{l}_{xs} \text{ through } 360^\circ$$

Therefore the transformation from the orbital reference to the suspension axes is

$$\begin{bmatrix} x \\ y \\ z \end{bmatrix} = \begin{bmatrix} 1 & 0 & 0 \\ 0 & \cos a & \sin a \\ - & -\sin a & \cos a \end{bmatrix} \begin{bmatrix} a \\ b \\ c \end{bmatrix}$$

where $a\bar{i} + b\bar{j} + c\bar{k} = (a, b, c)^T$, $x\bar{l}_{xs} + y\bar{l}_{ys} + z\bar{l}_{zs} = (x, y, z)^T$, and a is the angle between the starting position of $\bar{l}_{zs}$ and its position after rotation, $0^\circ \leq a < 360^\circ$.

Second orientation sequence:

Let θ be the angle subtended by the earth at the space system center of mass and ϕ be the angle subtended by the vector r_2 at the geocenter. Then $\theta - \phi$

is the angle between k and l_{zs} when the telescope points the forward horizon and $\theta + \phi$ is that angle when the telescope points the rear horizon.

For the 200 nmi altitude, these angles are

$$\theta = 70^\circ 55'$$

$$\phi = 1.5947 \times 10^{-5} \text{ degrees}$$

and for the synchronous orbit, these are

$$\theta = 8^\circ 40'$$

$$\phi = 2.5460 \times 10^{-6} \text{ degrees}$$

The orientation sequence begins with the telescope pointing the forward horizon, then the telescope is rotated back through the geocenter to the rear horizon. Thus the transformation from the orbital axes to the suspension axes, rotating the telescope about l_{ys} , is

$$\begin{bmatrix} x \\ y \\ z \end{bmatrix} = \begin{bmatrix} \cos \alpha & 0 & -\sin \alpha \\ 0 & 1 & 0 \\ \sin \alpha & 0 & \cos \alpha \end{bmatrix} \begin{bmatrix} a \\ b \\ c \end{bmatrix}$$

where $a \bar{i} + b \bar{j} + c \bar{k} = (a, b, c)^T$, $x \bar{l}_{xs} + y \bar{l}_{ys} + z \bar{l}_{zs} = (x, y, z)^T$, and $-(\theta + \phi) \leq \alpha \leq \theta - \phi$.

Numerical values:

For the calculation of the mutual attraction forces the ratio of the telescope mass to the spacecraft system mass was calculated to be 1: 19.87, where the mass of the telescope was 287 lbs and the mass of the spacecraft system was 15,638 lbs.

The length and radius of the spacecraft, assumed to be a cylinder, was approximated to be 20.00 feet and 16.67 feet, respectively.

Standard numerical values were used for various other orbital characteristics.

Atmospheric Density: 200 Nautical mile orbit, $\rho = 2.015 \times 10^{-14} \text{ slugs/ft}^3$
Synchronous orbit, $\rho = 1.000 \times 10^{-24} \text{ slugs/ft}^3$

Gravitational constant = $1.404 \times 10^{16} \text{ ft}^3/\text{sec}^2$

Earth radius = 3,440 Nautical miles

Synchronous orbit radius = 19,360 Nautical miles

Angular velocity: 200 Nautical mile orbit $\omega = 1.149 \times 10^{-3}$ radians/sec

Synchronous orbit, $\omega = 6.9737 \times 10^{-5}$ radians/sec

Orbital velocity: 200 Nautical mile orbit, $V = 2.5187 \times 10^4$ ft/sec

Synchronous orbit, $V = 1.0063 \times 10^4$ ft/sec

Gravitational acceleration:

200 Nautical mile orbit, $g_o = 28.663$ ft/sec²

Synchronous orbit, $g_o = 7.3056 \times 10^{-1}$ ft/sec²

10.4.2.3 Spring Suspension

10.4.2.3.1 Spring Design Trade-offs

K = individual spring constant

$k_{(eff)}$ = effective spring constant

M = mass of telescope

ω_I = resonant angular frequency

d = wire diameter

N = number of turns/spring

D = coil diameter

$$k_{(eff)} = 2K = M\omega_I^2$$

Since

$$K = \frac{Gd^4}{8ND^3}$$

$$M\omega_I^2 = \frac{Gd^4}{4ND^3}$$

solving for D, N, and d in turn

$$D = \left[\frac{G}{4M\omega_I^2} \right]^{1/3} \times \left[\frac{d^4}{N} \right]^{1/3}$$

$$N = \frac{G d^4}{4M\omega_1^2 D^3}$$

$$d = \left[\frac{4M\omega_1^2}{G} \right]^{1/4} N^{1/4} D^{3/4}$$

The values of M and G used were

$$M = 1600 \text{ lb} = \frac{1600}{32 \times 12} \text{ slug ft/in.}$$

$$G = 11.5 \times 10^6 \text{ lb/in.}^2$$

10.4.2.3.2 Springs as Conductors

P_e = power loss in watts (or per cent of 300 watts)

V = supply voltage = 28 volts d.c.

$R_{(eff)}$ = effective resistance of spring conductors

R = individual spring resistance

ρ = resistivity of Beryllium-copper wire = 2.085×10^{-5} ohm-inch

N = number of turns/spring

D = coil diameter

d = wire diameter

$K_{(eff)}$ = effective spring constant = 4.17×10^{-4} lb/in.

K_o = individual spring constant

ND = solid height of spring

G = 6×10^6 lb/in.² for beryllium-copper wire

For a given d.c. voltage, the power loss is

$$P_e = \frac{V^2}{R_{eff}}$$

For three springs in parallel connected in series with three more in parallel,

$$R_{(eff)} = \frac{2R}{3}$$

the resistance of one spring is

$$R = \frac{(N\pi D)}{\pi \frac{d^2}{4}} = \frac{4\rho ND}{d^2}$$

$$K_{(eff)} = 2 K_o = 2 \left(\frac{Gd^4}{8ND^3} \right)$$

so,

$$P_e = \frac{3}{2} V^2 \frac{d^2}{4\rho ND} = \frac{3V^2 K_{(eff)}}{2\rho G} \frac{D^2}{d^2}$$

and

$$P_e^2 = \frac{9V^4 K_{(eff)}}{16\rho^2 G} \frac{D}{N}$$

thus,

$$Nd = \frac{9V^4 K_{(eff)} Dd}{16 \rho^2 G P_e^2} = \frac{9V^4 K_{(eff)}}{16 \rho^2 G P_e^2} \sqrt{\frac{3V^2 K_{(eff)}}{2 \rho G P_e}} x D^2$$

simplifying,

$$Nd = \frac{9\sqrt{3} V^5 K_{(eff)}^{2/3} D^2}{2 16 \rho^{5/2} G^{3/2} P_e^{5/2}} = \frac{1.245 x 10^3 D^2}{P_e^{5/2}}$$

10.4.2.3.3 Constraint Torques of Cantilevered Spring

x = horizontal component of spring length

y = vertical deflection

δ = axial spring displacement

θ = angular displacement of spring axis from vertical at free end

M = moment acting on spring at free end

F_y = vertical force on spring

F_x = horizontal force on spring

L = unstressed spring length

d = wire diameter

N = number of turns

D = coil diameter

ν = Poisson's ratio

R = radius of curvature of spring axis

For $\theta < 1^\circ$ and $\frac{y}{x} < .1$,

$$-\frac{y}{x} \doteq \tan \left(\frac{\theta}{2} \right)$$

$$F_{yx} = -M$$

$$\frac{M}{\theta} = \left(\frac{2}{2+\nu} \right) \left(\frac{Ed^4}{64ND} \right) = \text{spring constant due to angular deflection of free end of spring.}$$

Thus,

$$F_y \doteq -\frac{M}{x} = -\frac{1}{x} \left(\frac{2}{2+\nu} \right) \frac{Ed^4}{64ND} \times 2 \text{ arc tan } \left(-\frac{y}{x} \right)$$

$$\delta = R - L = \frac{x\theta}{\sin\theta} - L$$

$$\delta = \frac{x^2 + y^2}{y} \text{ arc tan } \left(\frac{y}{x} \right) - L$$

Since

$$\nu = \frac{E}{2G} - 1 ,$$

$$F_y = \left(\frac{4}{1 + \frac{E}{2G}} \right) \frac{Ed^4}{64ND} \frac{1}{x} \text{ arc tan } \left(\frac{y}{x} \right)$$

Since for small θ and small $\frac{y}{x}$,

$$\tan \frac{y}{x} \doteq \frac{y}{x}$$

$$\delta = \frac{x^2 + y^2}{y} \left(\frac{y}{x} \right) - L = x + \frac{y^2}{x} - L$$

By neglecting y^2/x ,

$$\delta = x - L = \text{horizontal displacement}$$

Thus,

$$F_y = \frac{4E d^4}{(1 + \frac{E}{2G}) 64ND} \frac{y}{x^2}$$

$$F_x = \frac{G d^4}{8 ND^3} (x-L)$$

If ψ = Angular displacement of telescope in plane of F_x and F_y

and r_o = radius of telescope

then the torque on the telescope is,

$$\text{for } \psi < 10^\circ$$

$$T\psi = F_y r_o = K_\psi \psi$$

by requiring that k_o = axial spring constant

be constant,

$$k_o = \text{constant} = \frac{G d^4}{8 ND^3}$$

then

$$F_y = \frac{\frac{E}{2G} D^2}{(1 + \frac{E}{2G})} \left(\frac{G d^4}{8 ND^3} \right) \frac{y}{x^2}$$

$$F_y = \frac{\left(\frac{E}{2G} \right)}{(1 + \frac{E}{2G})} D^2 k_o \frac{y}{x^2}$$

For the spring suspension, the effective spring constant $k_{(eff)}$ is twice k_o ; thus,

$$F_y = \frac{\frac{D^2}{2} \times \frac{E}{2G} \times k_{(eff)}}{(1 + \frac{E}{2G})} \times \left(\frac{y}{x^2} \right)$$

For most spring materials,

$$\frac{E}{2G} = \frac{30}{23}$$

$$y = r_o \sin \psi$$

$$x = L + r_o (1 - \cos \psi)$$

$$= L + 2r_o \sin^2 \left(\frac{\psi}{2} \right)$$

so for $\psi < 1^\circ$

$$\frac{y}{x^2} = \frac{r_o \psi}{\left(L + r_o \frac{\psi^2}{2} \right)^2} = \frac{\psi}{r_o \left(\frac{L}{r_o} + \frac{\psi^2}{2} \right)^2} ;$$

thus,

$$T_\psi = F_y r_o = \frac{D^{2\frac{1}{2}} \frac{30}{23} k_{(eff)}}{\left(1 + \frac{30}{23} \right)} \times \frac{\psi}{\left(\frac{L}{r_o} + \frac{\psi^2}{2} \right)^2}$$

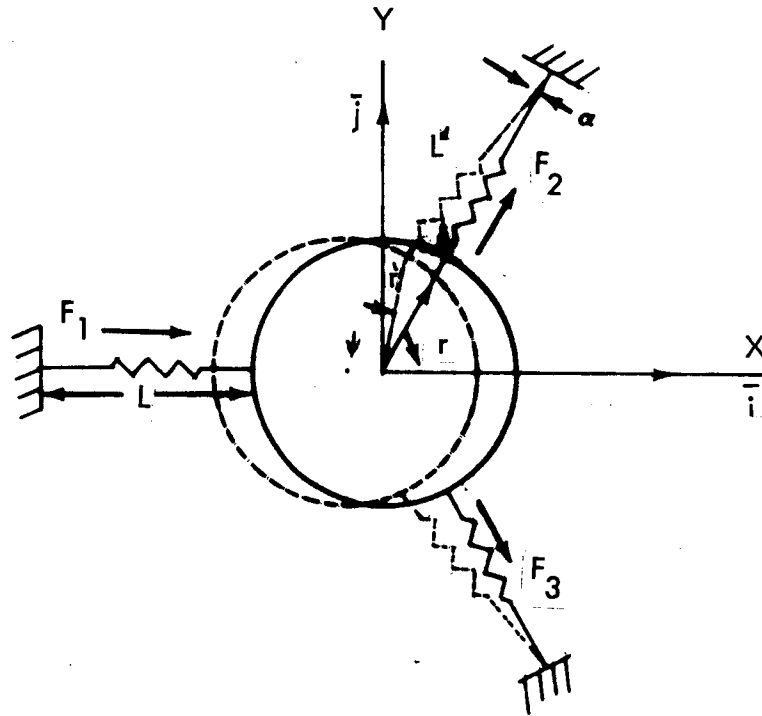
For $D = 5.0"$, $r_o = 23"$, $k_{(eff)} = 2k_o = 2(.1) \text{ lb/in.}$

$$T_{total} = 2T_\psi = \frac{2 \times 25 \times (1.3) \times .1 \times 1.745 \times 10^{-2} \times 1/12}{(2.30) \times \left(\frac{L}{23} + 1.523 \times 10^{-4} \psi^2 \right)^2} \psi$$

$$T_{total} = \frac{4.108 \times 10^{-3} \text{ ft-lb/deg}}{\left(\frac{L}{23} + 1.523 \times 10^{-4} \psi^2 \right)^2} \psi$$

10.4.2.3.5 Large Spring Deflections

The change in effective spring constants for large angles is derived below.



The sum of the forces is

$$\bar{F} = \bar{F}_1 + \bar{F}_2 + \bar{F}_3 \quad (1)$$

$$F = K [\bar{i} (\delta_1 + 0.5\delta_2 + 0.5\delta_3) + \bar{j} (.866\delta_2 - .866\delta_3)]$$

where

$\delta_1, \delta_2, \delta_3$ = actual spring deflections.

For infinitesimal displacements at an angle θ (measured C.W. from - X axis)

$$\delta_1 = \delta \cos \theta \quad (2)$$

$$\delta_2 = \delta \cos (\theta + \pi/3) \quad (3)$$

$$\delta_3 = \delta \cos (\theta - \pi/3) \quad (4)$$

where

δ = telescope displacement.

Inserting equations (2), (3), (4) into (1) gives,

$$\frac{F}{\delta K} = 1.5 (i \cos \theta - j \sin \theta)$$

which shows that for infinitesimal displacements, the effective spring constant is 1.5 times the spring constant of a pair independent of the angle θ .

For larger displacements, the geometry for the displaced configuration must be solved. Referring to the sketch,

$$r' = (r^2 + \delta^2 - r\delta)^{\frac{1}{2}} \quad (1)$$

$$\sin \psi = \delta/r' (.866) \quad (2)$$

$$L' = [(r')^2 + (r + L)^2 - 2 r' (r + L) \cos \psi]^{\frac{1}{2}} \quad (3)$$

$$\sin \alpha = 0.866 \quad \delta/L' \quad (4)$$

$$\frac{F}{K\delta} = 1 + 2 \cos^2 (60 - \alpha) \quad (5)$$

Numerical solution of the above equations is plotted in figure 10.2.2.4.2.2-4.

Assuming that the springs have no angular stiffness, resistance to the telescope being displaced angularly (say θ) about its axis is due to elongation of the springs. Denoting the restoring forces of the three spring pairs by $\bar{F}_1$, $\bar{F}_2$ and $\bar{F}_3$, the restoring torque for the θ displacement is

$$\sum_{i=1}^3 \bar{R}_i \times \bar{F}_i$$

where $\bar{R}_i$ is the radius vector to the point of action of $\bar{F}_i$. Thus the effective rotational stiffness, K (net), of the system is given by

$$\frac{d}{d\theta} \left| \sum_{i=1}^3 \bar{R}_i \times \bar{F}_i \right|$$

Note that

$$r = |\bar{R}_1| = |\bar{R}_2| = |\bar{R}_3|$$

and, for small θ displacements,

$$|\bar{F}_i| = K_r \Delta L$$

where r denotes the telescope radius, ΔL denotes the increment of length of a spring pair, and K_r denotes the radial stiffness of a spring pair. Resolving the vectors into components yields

$$\frac{K \text{ (net)}}{K_r} = r (r + L) \frac{d}{d\theta} \left[\frac{\Delta L \sin \theta}{L + \Delta L} \right]$$

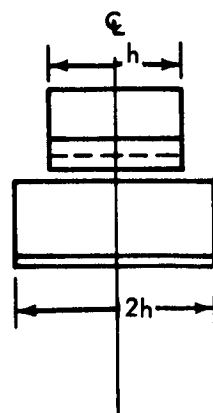
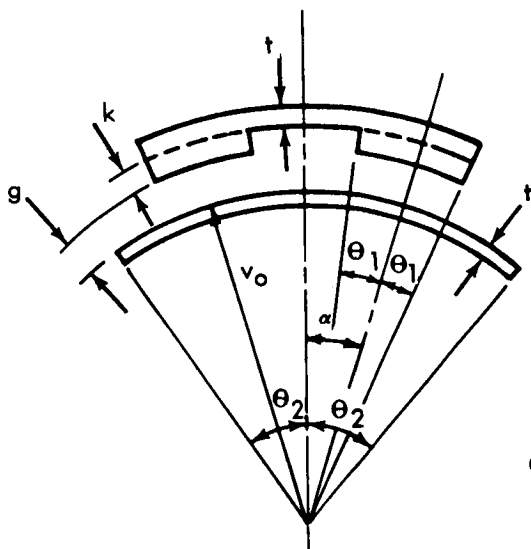
Now, using successively the cosine law to express ΔL , a two-term approximation for the radical, the usual derivative formulas, and two-term series truncations for $\sin \theta$ and $\cos \theta$ yields

$$\frac{K \text{ (net)}}{K_r} = r^2 (r + L)^2 \left[\frac{\theta^2/2}{L^2 + r (r + L) \theta^2/2} + \frac{L^2 \theta^2}{[L^2 + r (r + L) \theta^2/2]^2} \right]$$

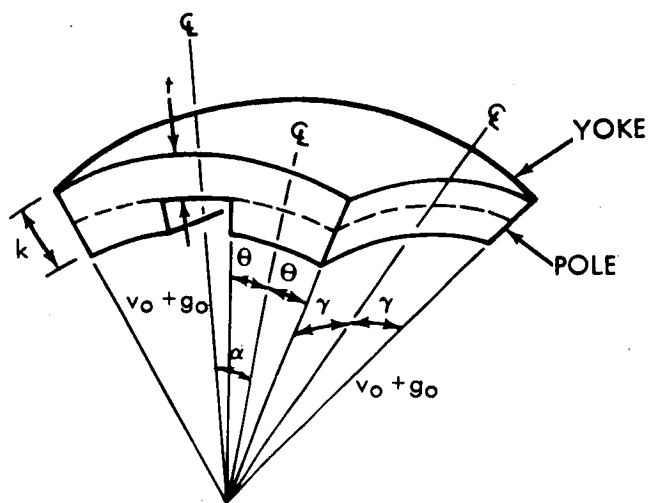
10.4.2.4 Magnetic Suspension

10.4.2.4.1 Weight & Core Loss Optimization

The following relationships are chosen to give the magnet in figure 10.4.2.4.1-1, a relative configuration that may be scaled up or down by varying the single parameter t .

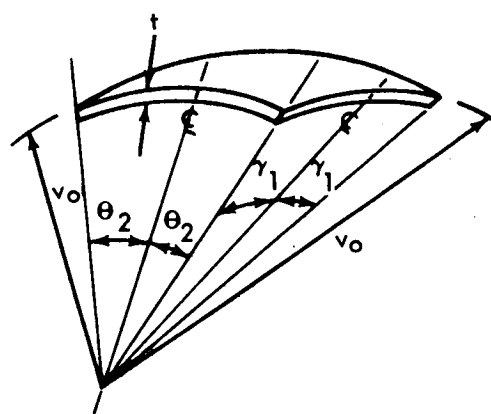


CYLINDRICAL CASE



MAGNET

SPHERICAL CASE



CAP

Figure 10.4.2.4.1-1. Magnet Geometries.

$$h = 5 t$$

$$k = 2 t$$

$$\theta_1 = \frac{5t}{2 (r_o + g_o)}$$

$$a_1 = \frac{15t}{4 (r_o + g_o)}$$

$$\theta_2 = \frac{35t}{4 (r_o + g_o)}$$

The two additional parameters γ and γ_1 in figure 10.4.2.4.1-1 are related to t by

$$\gamma = \frac{1}{2} \text{ arc sin } \frac{5t}{r_o + g_o}$$

$$\gamma_1 = \frac{1}{2} \text{ arc sin } \frac{10t}{r_o + g_o}$$

The core losses are given by

$$P = (V_y B_{my}^2 + V_p B_{mp}^2 + V_c B_{mc}^2) f^2 \times \left[\frac{\rho_o^k}{B_o^2 f_o^2} \right] \text{ watts}$$

The weight of the magnet unit is given by

$$W = \rho (V_p + V_y + V_r)$$

where

$$V_y = \frac{125}{2} t^3 \left[1 + \frac{3t}{2 (r_o + g_o)} \right] \text{ volume of yoke}$$

$$V_p = 50 t^3 \left[1 + \frac{t}{2 (r_o + g_o)} \right], \text{ volume of both poles}$$

$$V_c = \frac{125}{2} t^3 \frac{(r_o - t/2)}{r_o + g_o}, \text{ volume of cap segment under magnet}$$

$$V_r = 175 t^3 \frac{(r_o - t/2)}{r_o + g_o}, \text{ volume of cap}$$

f = frequency of supply

ρ = density of material = .76 lb/inch³

k_o = .15 watts/lb

B_o = 4000 gauss

f_o = 60 hz

$B_{my} = \frac{\sqrt{2} \phi_1 \left(\frac{A'}{A}\right)}{0.9 \times 6.45 A_y}$, maximum flux density in yoke, 0.9 stacking factor

$B_{mp} = \frac{\sqrt{2} \phi_1 \left(\frac{A'}{A}\right)}{0.9 \times 6.45 A_{ag}}$, maximum flux density in poles

$B_{mc} = \frac{\sqrt{2} \phi_1 \left(\frac{A'}{A}\right)}{0.9 \times 6.45 A_c}$, maximum flux density in cap

$\frac{A'}{A} = \left(1 + \frac{2g_o}{5t}\right)^2$, fringing flux correction

$A_y = 5 t^2$ = cross sectional area of yoke

$A_c = 5 t^2$ = cross sectional area of cap segment

$A_{ag} = 25 t^2 \frac{(r_o + g_o/2)}{(r_o + g_o)}$, cross sectional area of airgap

$\phi_1 = \left\{ \frac{\theta_1}{\sin \theta_1} \times \frac{1}{\cos a_1} \times (f_x \text{ in newtons}) \frac{\mu_o A_{ag}}{2 Z_n} \left(\frac{1}{Q_o - 2} \right) \right.$
 $\left. \times \left[1 + 2 (Q_o - 1) Z_n^2 + \frac{1}{4} (1 + 2 (Q_o - 1)^2 + (Q_o - 1)^4) Z_n^4 \right]^{\frac{1}{2}} \right\}$,
working flux/pole

The following values were taken:

$\rho = 0.276 \text{ lb/in}^2$

$r_o = 30.0 \text{ inch}$

$k_o = 0.15 \text{ watts/lb}$

$g_o = 2.50 \text{ inches}$

$B_o = 4000 \text{ gauss}$

$f_o = 60 \text{ hz}$

$\left(\frac{\theta_1}{\sin \theta_1} \times \frac{1}{\cos a_1} \right) = 1.100$

$$\mu_o = 4\pi \times 10^{-7} \text{ b/A-m}$$

$$f_x = \frac{0.010}{0.225} \text{ newton}$$

$$Z_n = 0.185$$

$$Q_o = 5$$

Substituting all of this into the weight and power loss equations yields, after some simplification,

$$P = (62.54 t + t^2) (1 + 1/t)^4 f_x^2 \times 3.891 \times 10^{-8} \text{ watts}$$

$$W = (285.0 + t) t^3 \times 2.654 \times 10^{-1} \text{ lbs}$$

It was found that this cylindrical approximation to the spherical case shown in figure 10.4.2.4.1-1 was valid.

The volumes and areas for the spherical case were found by ordinary integration in spherical polar coordinates, and these resulting formulas were evaluated at $t = 0.5$ ". From these, and the values given above for the constants, the core losses and weight were computed:

cylindrical case:

$$t = 0.5 \text{ in.}$$

$$f = 400 \text{ hz}$$

$$f = 60 \text{ hz}$$

$$P = 15.9 \text{ watts}$$

$$P = 0.358 \text{ watts}$$

$$W = 9.50 \text{ lb}$$

$$W = 9.50 \text{ lb}$$

spherical case:

$$t = 0.5 \text{ in.}$$

$$f = 400 \text{ hz}$$

$$f = 60 \text{ hz}$$

$$P_s = 15.75 \text{ watts}$$

$$P_s = 0.354 \text{ watts}$$

$$W_s = 9.07 \text{ lb}$$

$$W_s = 9.07 \text{ lb}$$

10.4.2.4.2 A.C. Suspension Incremental Spring Constant

$$F_x = \frac{-\phi_o^2}{2\mu_o A_o} \frac{4 (Q_o - 2) Z_n \cos a_o \frac{\sin \theta_o}{\theta_o}}{1 + 2 (Q_o - 1) Z_n^2 + \frac{1}{4} (1 + 2 (Q_o - 1)^2 + (Q_o - 1)^4) Z_n^4}$$

For

$Q_0 = 5$, quality factor of winding

$\phi_0 = 4.77 \times 10^{-5}$ weber, flux in airgap

$A_0 = 47.1 \text{ inch}^2$, pole face area

$\mu_0 = 4 \pi \times 10^{-7}$ wb/A m

$\theta_0 = .2$ radian

$\alpha_0 = .17$ radian

Z_n = normalized displacement

F_x = force in x-direction

x = radial displacement

$$F_x = \frac{-(7.611 \times 10^{-2} \text{ lb}) Z_n}{1 + 8 Z_n^2 + 72.25 Z_n^4}$$

$$Z_n = \left[.3942/\text{inch} \right] x$$

$$\frac{d F_x}{d Z_n} = \frac{dx}{d Z_n} \quad \frac{d F_x}{d x}$$

$$\frac{d F_x}{d x} = \frac{d Z_n}{d x} \quad \frac{d F_x}{d Z_n} = - 3.00 \times 10^{-2} \text{ lb/inch} \left[\frac{1 - 8 Z_n^2 - 216.75 Z_n^4}{(1 + 8 Z_n^2 + 72.25 Z_n^4)^2} \right]$$

10.4.2.4.3 Fringing Effects

The A. C. tuned support must also allow the telescope and certain limited amounts of rotational freedom while restraining any translational movement. Due to the large air gap, there is a significant amount of flux that is not directly beneath the pole faces, but is instead out to the sides of the magnet. If the armature beneath the magnet is displaced to one side, the symmetry is broken and the fringing flux around the magnet will produce a net torque on the armatures and so on the telescope itself. This torque may be reduced by (a) making the armatures have a spherically shaped surface with their radii having the center of mass of the telescope as their origin, and (b) extending the edges of the armatures far enough beyond the pole faces so that nearly all the fringing flux passes through the top surfaces of the armatures. These two measures give the system a much stronger symmetry under rotation and sharply decrease the torques due to the fringing flux.

There are two ways to compute the value of the fringing flux of a specific configuration. The first is the calculation of the permeance of the probable flux paths. This consists of dividing the volume containing the fringing flux into geometrically simple volumes bounded by the flux lines computing the permeance of each smaller volume, adding up these permeances in series or parallel, and then computing the leakage coefficient of the system. From this can be obtained a probable value of the fringing flux and so the value of the net torque on the displaced armature. The inherent inaccuracy of this method is that flux lines do not usually follow mathematically simple paths. The geometrically simple volumes are only an approximation, the error decreasing as the total volume is more and more finely divided into smaller and smaller segments.

The second method is called "field mapping" and is entirely graphical. It consists of sketching the distribution of flux lines and equipotential lines in such a manner that the total volume of the field is broken up into smaller unit volumes each having the same permeance. This permeance is computed from direct measurements of the small volumes, and the total permeance is obtained by adding the permeances of the unit volumes in series and parallel until the entire volume of the field is covered. The chief difficulty of this method is that it is generally not easy to draw, with the proper distribution, accurate flux and equipotential lines, especially in a three-dimensional field surrounding a pole having sharp corners. This method is more difficult and not as nearly accurate as the first one. As an example, consider the following bipolar magnet with a flat plate as the armature as shown in figure 10.4.2.4.3-1.

The force tending to center the plate beneath the magnet is given by

$$F = 1/2 F_a^2 \frac{dP_t}{ds}$$

where

F_a = magnetomotive force of magnet

s = displacement

P_t = total permeance of system

This may be approached in two ways: first, to find the total permeance P_t as an analytical function of the displacement s , and second, to compute ΔP_t and Δs separately for a given small displacement. Either approach is tedious and involves a large number of computations, thus decreasing the accuracy. Consider the permeance of only the right hand end of the system: since permeances in parallel and in series add linearly and inversely, respectively,

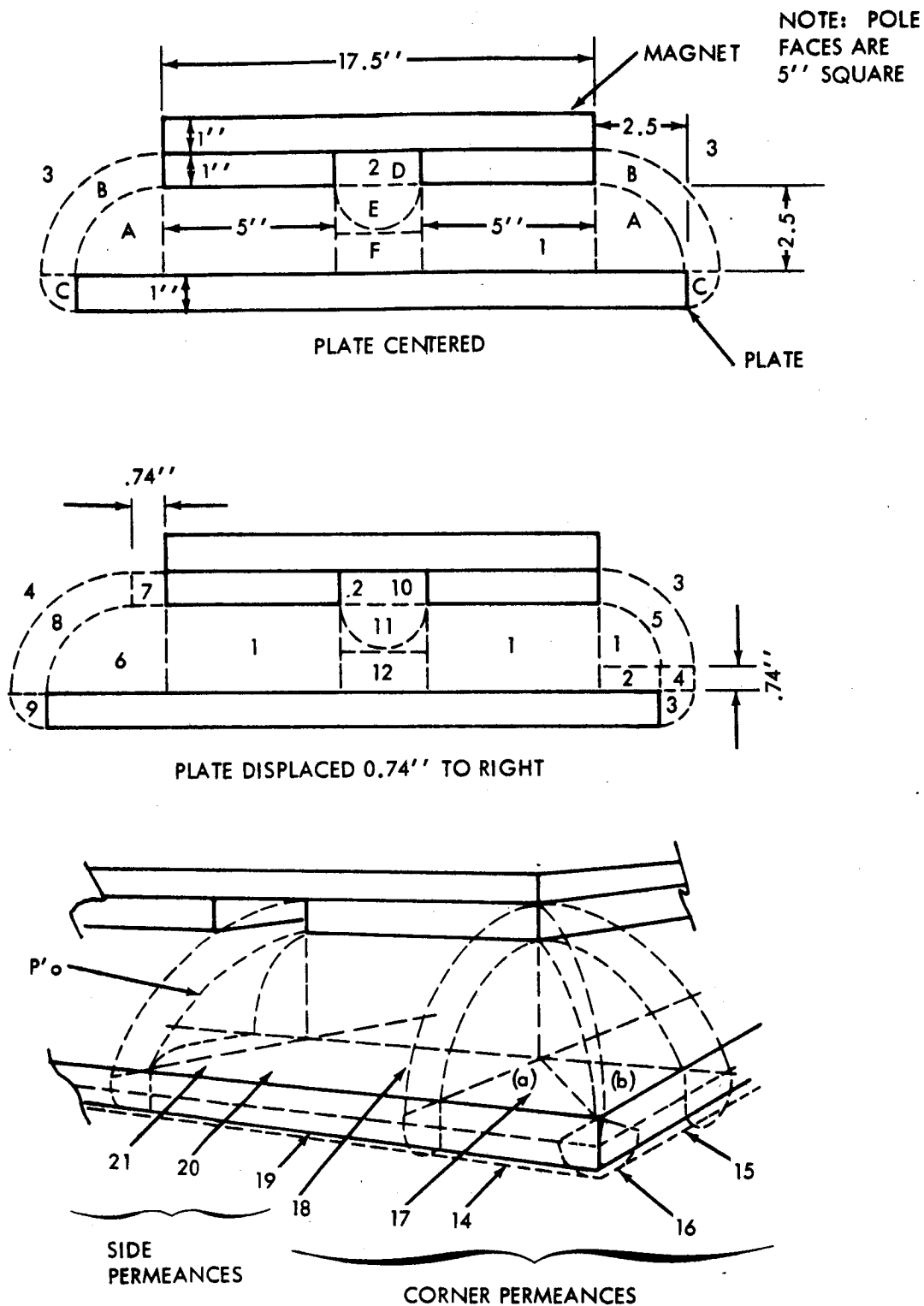


Figure 10.4.2.4.3-1. Illustration of Fringing Geometry.

$$P_{R.H. END} = \frac{1}{\frac{1}{P_3} + \frac{1}{P_4} + \frac{1}{P_5}} + \frac{1}{\frac{1}{P_1} + \frac{1}{P_2}} + \frac{2}{\frac{1}{P_{18a}} + \frac{1}{P_{14}}} + \frac{2}{\frac{1}{P_{18b}} + \frac{1}{P_{15}}} + 2(P_{17a} + P_{17b})$$

$$P_1 = 5 \times 0.52 \mu = 2.60 \mu$$

$$P_2 = \mu \frac{5 \times 1.76}{0.74} = 11.9 \mu$$

$$P_3 = \mu 5 \times 0.52 = 2.60 \mu$$

$$P_4 = \mu \frac{1 \times 5}{0.74} = 6.76 \mu$$

$$P_5 = \frac{2 \times 5 \mu}{\pi} \ln \left[1 + \frac{1}{1.76} \right] = 1.44 \mu$$

$$P_{17} = 0.308 \mu \times 2.86 = 0.881 \mu$$

Note: This volume has been replaced by $\frac{1}{2}$ of a spherical quadrent having the same volume. The equivalent radius is 2.86 in.

$$P_{18} = \frac{\pi}{8} \frac{[(3.50)^2 - (2.50)^2]}{(5.53)^2} = 0.069 \mu$$

Note: The mean path length of 5.53" is taken as the average of the mean arc lengths of each side.

Note that P_{17} and P_{18} are taken as approximations to P_{17a} , P_{17b} and P_{18a} and P_{18b} .

P_{14} and P_{15} may be found by the same formula as P_3 .

So,

$$P_{R.H. END} = \frac{1}{\frac{1}{1.44 \mu} + \frac{1}{6.76 \mu} + \frac{1}{2.60 \mu}} + \frac{1}{\frac{1}{2.60 \mu} + \frac{1}{11.9 \mu}} + \frac{2}{\frac{1}{.069 \mu} + \frac{1}{1.30 \mu}} + \frac{2}{\frac{1}{.069 \mu} + \frac{1}{1.30 \mu}} + 2(.881 \mu + .881 \mu)$$

$$P_{R.H. END} = 6.40 \mu$$

Similar computations yield values for the permeances of the sides, lefthand end, and for the case where the plate is centered. From these, the value of

$$\frac{\Delta P_t}{\Delta s} \text{ is found to be}$$

$$\frac{\Delta P_t}{\Delta s} = 1.255 \times 10^{-7}$$

For $F_a = 100$ amp-turns, the side force is

$$F = 4.43 \times (100)^2 \times 1.255 \times 10^{-7} \text{ lb} = 5.56 \times 10^{-3} \text{ lb}$$

If the force is computed by finding P_t as a function of s and differentiating, the value of the side force is, for the same F_a ,

$$F = 5.84 \times 10^{-3} \text{ lb}$$

The torque induced by this fringing flux force is, at a 30.0 inch radius, 1.43×10^{-3} lb-ft for one magnet. The entire force between the magnet and plate, however, is only 10×10^{-3} lb for this linear system, so the side force is over 50 per cent of the total force. This example shows the necessity of eliminating as much of the fringing effects as is possible.

10.4.2.4.4 Servo Analysis

Referring to the block diagram of figure 10.2.2.4.2.3-12, let the forward transfer function representing the actuator and inertia be called G_2 and the accelerometer feedback loop transfer function called H . Then

$$G_2 = \frac{K_p K_c / M_T \tau}{S^2 (1 + s)}$$

$$H = \frac{K_a \omega_f S^2}{S + \omega_f}$$

and the closed loop transfer function is

$$\frac{G_2}{1 + G_2 H} = \frac{(K_p K_c / M_T \tau) (S + \omega_f)}{S^2 [S^2 + (1/\tau + \omega_f) S + \omega_f/\tau (1 + r_m)]}$$

where

$$r_m = \frac{K_p K_c K_a}{M_T}$$

K_p = power amplifier gain, amp/volt

K_c = actuator gain, lbs/amp

K_a = accelerometer gain, slugs

M_T = telescope mass, slugs

ω_f = filter bandwidth, rad/sec

τ = actuator time constant, seconds

In practice, $r_m \gg 1$; otherwise the accelerometer loop will not serve its purpose. Assume $r_m = 100$. Then the second-order factor will produce complex poles for $\omega_f < \frac{400}{\tau}$. Since τ is about 0.1 second, this case will hold.

The effective bandwidth of the filter is about $\omega_f r_m$. This must be kept small because of noise in the accelerometer. The closed-loop transfer function from the viewpoint of the suspension system (i.e., the suspension transmissibility) is

$$\frac{\mu}{\mu_0} = \frac{(K_e K_1 K_p K_c / M_T \tau) (S + \omega_f) G_c(s)}{S^2 [S^2 + (1/\tau + \omega_f)S + \omega_f r_m / \tau + \frac{K_e K_1 K_p K_c}{M_T \tau} (S + \omega_f) G_c(s)]}$$

where

$G_c(s)$ = transfer function of a compensation network yet to be determined.

K_e = error sensor scale factor, volts/inch.

K_1 = amp gain, volts/volt.

The closed-loop transfer function for a disturbance input F_D is

$$\frac{\mu}{F_D} = \frac{\frac{1}{M_T} (S + 1/\tau) (S + \omega_f)}{S^2 [S^2 + (\omega_f + 1/\tau) S + \frac{\omega_f}{\tau} r_m + \frac{K_p K_c K_e K_1}{M_T \tau} (S + \omega_f) G_c(s)]}$$

For a noise input to the accelerometer loop, the closed-loop transfer function is

$$\frac{\mu}{N} = \frac{r_m \omega_f / \tau}{S^2 [S^2 + (1/\tau + \omega_f) S + \frac{\omega_f}{\tau} r_m + \frac{K_p K_c K_e K_1}{M_T \tau} (S + \omega_f) G_c(s)]}$$

For the purpose of synthesis, the inner and outer loops may be separated because of the greatly different dynamic properties. It is desired for the outer loop to have a very low natural frequency and damping (typically 0.01 rad/sec and 1 per cent critical) whereas for the inner loop the bandwidth should be limited only by the accelerometer noise, and the damping should be chosen on the basis of the time response to disturbance inputs.

Therefore, looking at the inner loop first and assuming the gain of the outer loop is zero ($G_c = 0$), the acceleration response to a disturbance force is

$$\begin{aligned}\ddot{\mu}(s) &\cong \frac{\frac{F_d(s)}{M_T} (s + 1/\tau) (s + \omega_f)}{s^2 + (\omega_f + 1/\tau) s + \frac{\omega_f}{\tau} r_m} \\ &= \frac{F_d(s)}{M_T} \left[1 - \frac{(r_m - 1)\omega_f/\tau}{s^2 + (1/\tau + \omega_f) s + \frac{\omega_f}{\tau} r_m} \right]\end{aligned}$$

For a step input of disturbance

$$F_d(s) = F_m/s$$

the final value of acceleration is

$$\ddot{\mu}_{ss} = \lim_{s \rightarrow 0} \ddot{\mu}(s) = \frac{F_M}{M_T r_m} = \frac{F_M}{K_A}$$

where

$$K_A = K_p K_c K_a$$

is the "effective mass".

The initial acceleration for the same disturbance is

$$\ddot{\mu}_i = \lim_{s \rightarrow 0} s \ddot{\mu}(s) = \frac{F_M}{M_T}$$

The time required to go from the initial to the final value is

$$t_r \cong \frac{8\tau}{1 + \omega_f \tau}$$

This is assuming, of course, that the outer loop gain is zero. In practice, the disturbance acceleration will be reduced by the factor $\frac{1}{r_m}$ in the time

t_r ; then over a much longer period will be reduced to zero by the action of the outer loop.

To minimize the disturbance torque resulting from F_D , t_r should be minimized. However, before choosing a value for t_r , it is necessary to look at the effect of this choice on the noise response, which is

$$\ddot{\mu}(s) = \frac{(r_m \omega_f / \tau) N(s)}{s^2 + (1/\tau + \omega_f) s + \frac{\omega_f}{\tau} r_m}$$

For white noise distributed with a constant power spectral density, the rms value of $\ddot{\mu}$ is determined from

$$\ddot{\mu}^2(\text{rms}) = \frac{1}{2\pi} \int_{-\infty}^{\infty} |G_n(j\omega)|^2 \Phi_{nn}(j\omega) d\omega$$

where

$$\Phi_{nn}(j\omega) = k^2 = \text{power spectral density (ft/sec}^2\text{)}^2/\text{rad/sec}$$

$$G_n(s) = \frac{r_m \omega_f /}{s^2 + (1/\tau + \omega_f)s + \frac{\omega_f}{\tau} r_m}$$

$$\text{Representing } |G_n(j\omega)|^2 \text{ as } G_n(s) G_n(-s) \Big|_{s=j\omega}$$

and evaluating the integral by the residue theorem⁽¹⁾ gives

$$\ddot{\mu}(\text{rms}) = \frac{k}{\sqrt{2}} \sqrt{\frac{\omega_f r_m}{(1 + \omega_f \tau)}}$$

To determine numerical values, assume

$$r_m = 100$$

$$\tau = 0.1 \text{ second}$$

$$k = 10^{-7} \text{ g}/\sqrt{1 \text{ cps}}$$

and require that

$$\ddot{\mu}(\text{rms}) = 2.4 \times 10^{-6} \text{ ft/sec}^2$$

which will produce about 10^{-5} ft-lb of torque for a 1 inch moment arm and a 50-slug mass.

This gives

$$\omega_f = 0.1 \text{ rad/sec}$$

(1) Truxal, J. G., Automatic Feedback Control System Synthesis, McGraw-Hill, 1955

Thus, from noise consideration the filter cut-off should be no greater than 0.1 rad/sec.

To determine the effect of this choice on the disturbance response, it is noted first that the response time becomes

$$t_r \cong 0.8 \text{ sec}$$

Generally, this is less than the response time of the telescope control system. A step disturbance of force then will appear approximately as an impulse of torque that injects an angular momentum ΔH into the telescope:

$$\begin{aligned} \Delta H &= \int_0^{t_r} F_M d (e^{-t/2\tau}) dt \\ &= \frac{2 F_M d}{e} \end{aligned}$$

The error response θ of the telescope control system will reach a maximum of

$$\theta_{\max} \cong \frac{\Delta H}{2 I \omega_T}$$

where

ω_T = control frequency, rad/sec

I = telescope moment of inertia, slug-ft²

For $F_M = 1$ ounce, $I = 300$ slug-ft², and $T = 1$ rad/sec, the maximum error is

$$\theta_m \cong 1 \text{ arc-sec}$$

which is the maximum allowed for the telescope intermediate loop before loss of fine loop acquisition (see subsection 10.2.1).

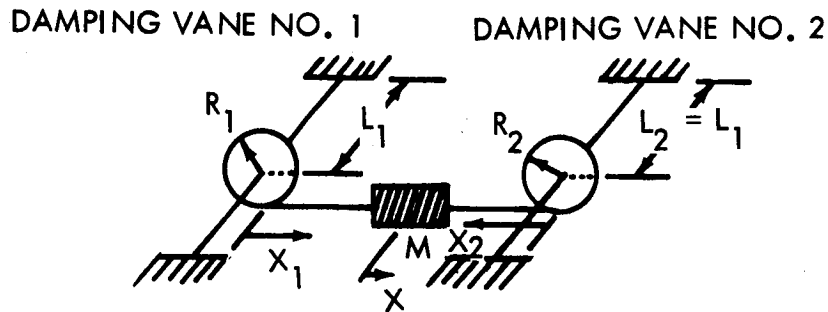
Returning now to the outer suspension loop, let the open-loop transfer function be called G . Then

$$G = \frac{(K_e K_l K_p K_c / MT\tau) (S + \omega_f) G_c(s)}{S^2 [S^2 + (1/\tau + \omega_f)s + \frac{\omega_f r_m}{\tau}]}$$

The poles and zeros of G are plotted in figure 10.2.2.4.2.3-13b, where a lead compensation network $G_c(s)$ is also plotted. No cascade compensation is actually required since the zero at $-\omega_f$ will prevent the loci for the outer loop crossing the imaginary axis. However, the operational procedure requires operation of the outer loop with the inner loop disabled for one operational mode. The root-loci for this case is shown in figure 10.2.2.4.2.3-13a.

10.4.2.5 Damper Analysis

The equations which determine the relationship between the damper components are derived below, after which the relationship between the material stressed to magnetic field induced is shown to meet the specific damping requirement.



The equation describing torque loading of a structural member is

$$T/J = \frac{G\theta}{L}$$

To wind wire onto damping vane #2 (see sketch) when displacement X is in the positive direction an offset wind is required at equilibrium. The same condition must be imposed on damping vane #1 to wind excess wire during displacement of 1000# mass in the $-X$ direction

$$R_1 \theta_0 = 1/2" = \text{maximum displacement of 1000# mass}$$

$$\theta_0 = \frac{1}{2R_1} \text{ radians}$$

For the first wheel,

$$X_1 = R_1 (\theta_1 - \theta_{01}) = R_1 \left(\frac{T_1 L_1}{2G_1 J_1} - \theta_{01} \right)$$

or

$$X_1 = R_1 \left(\frac{F_1 R_1 L_1}{2G_1 J_1} - \theta_{01} \right)$$

and for the second wheel,

$$X_2 = R_2 \frac{(F_2 R_2 L_2 - \theta_{o2})}{2G_2 J_2}$$

But

$$X_1 = -X_2 = X = \text{distance moved from equilibrium}$$

$$L_1 = L_2 = L = \text{length of torsion wire}$$

$$R_1 = R_2 = \text{radius of damping vanes number one and number two}$$

$$G_1 = G_2 = G = \text{modulus of rigidity of the material}$$

$$J_1 = J_2 = J = \text{second moment of area of the torsion wire}$$

$$\theta_{o1} = \theta_{o2} = \theta_o = \text{equilibrium offset angle}$$

$$\theta = \text{torsion wire twist angle}$$

$$\therefore F_1 = \left[\frac{X}{R_1} + \theta_o \right] \frac{2GJ}{R_1 L}$$

$$F_2 = \left[\frac{-X}{R_1} + \theta_o \right] \frac{2GJ}{R_1 L}$$

At equilibrium ,

$$F_1 (X = 0) = F_2 (X = 0) = \frac{2G\theta_o J}{R_1 L} ; F_T = 0 = F_1 - F_2$$

Off Equilibrium,

$$F_T = F_1 - F_2 = \frac{4GJ}{R_1 2L} X$$

$$F_T = 0$$

where F_T is the total restoring force acting on the 1000 lb mass

$$\therefore K = \frac{4GJ}{R_1 2L}$$

$$\text{for a round wire } J = \int_0^R r^2 dA = \int_0^R 2\pi r^3 dr = \pi/2 R^4$$

where

R is the torsion wire radius

$\therefore K = \frac{2\pi GR^4}{R_1 2L}$ which is shown in the nomograph of figure 10.2.2.4.2.4-2.

When

$$G = 11.5 \times 10^6, R_1 = R_2 = 2.5 \text{ in.}, R = .01423$$

and

$$L = 4.53 \text{ then } K = .1 \text{ lb/in.}$$

Lossed energy during the first cycle with a $\frac{1}{2}$ " displacement of the telescope will be found. Steinmetz's law is $W_h = \eta B^{1.6}$ and holds for values of flux in the range 500 to 15,000 gauss in iron.

W_h is ergs lossed per cycle per cu cm,

B is magnetic field in gauss, and

η is a constant of the material used

Each damper must dissipate 5.5 per cent of 1.4×10^4 ergs during the first half cycle of the $\frac{1}{2}$ " telescope displacement. This corresponds to 7.7×10^2 ergs per damper.

For a $\frac{1}{2}$ " telescope displacement, the damping vanes that satisfy our damping requirements for particular B fields will be found.

The weight of damping material which must undergo a complete cycle of the B-H curve for the first half cycle to achieve the required damping are displayed in table 10.10.4.2.4-1.

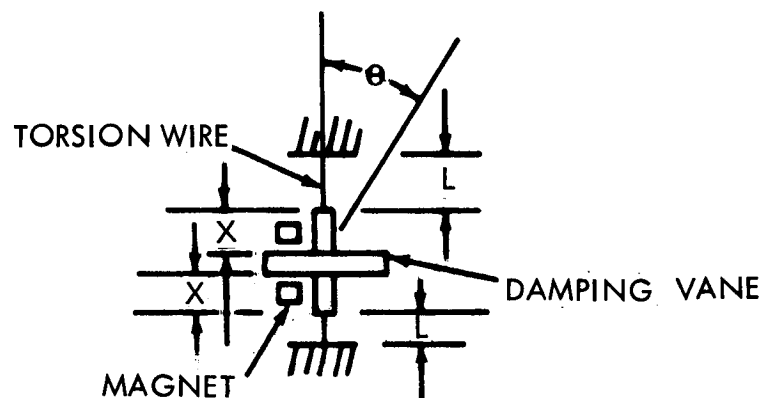


TABLE 10.4.2.4-1

<u>Material</u>	<u>B in gauss</u>	<u>Volume Stressed Damping Vane Material</u>	<u>Weight of Stressed Damping Material</u>
99.9% Fe	500	9.15 cm ³	.158 lb
Cold rolled 50%	750	4.82 cm ³	.0835 lb
	1000	3.02 cm ³	.0522 lb
7.87 gram/cm ³	1250	2.12 cm ³	.0367 lb
	1500	1.58 cm ³	.0274 lb
99.9% Fe	500	30.6 cm ³	.53 lb
Annealed 900°C	750	16.7 cm ³	.289 lb
	1000	10.1 cm ³	.174 lb
7.87 gram/cm ³	1250	7.12 cm ³	.123 lb
	1500	5.27 cm ³	.912 lb

The same information is graphically displayed in figure 10.2.2.4.2.4-3.

The maximum tension on the torsion wire will be 30 per cent of its tensile strength which is 330×10^3 psi for .02846 in diameter music wire.

The maximum lateral rotation is $\theta_m = 1/16$ th radian so that the damping vane will not come into contact with the magnets. See sketch preceding page. Therefore, without the teflon stops shown in figure 10.2.2.4.2.4-1, the maximum torque exerted on the damping vane by the magnets is

$$T_M = F_{o1} + F_{o2} + \frac{2X^2\theta^2 E R^2}{L^2}$$

$$\left(\cos \tan^{-1} \frac{X\theta}{L} \right) (X\theta) + \left(\sin \tan^{-1} \frac{X\theta}{L} \right) (X)$$

11.0 LARGE OPTICS

A telescope in space has two principal advantages over a telescope on the ground: it permits observation of ultraviolet radiation which is absorbed by the atmosphere and it is free of the effects of scattering and turbulence in the atmosphere. To make significant advances by taking advantage of the freedom from atmospheric effects, the telescope must have a large aperture, a diffraction limited optical system, and yet be light enough to be compatible with the payload of the launch vehicle. New technology is required to achieve these objectives. Two flight experiments are described in this section which will provide some of the required technology. These experiments are: Thin Mirror Nesting Principle and Primary Mirror Figure Test.

11.1 PRIMARY MIRROR CONFIGURATION ALTERNATIVES

The various alternatives available in the construction of a large space primary are depicted in **figure 11.1-1**. **Principle problem areas of the alternatives** are also shown. If the mirror is conceived to be of one piece construction (non-segmented) the discussion of passive approaches refers to active as well. The passive alternatives are shown in further detail in **figure 11.1-2**.

Of the construction approaches listed, the first (thin or membrane, $t/D = 1/50-1/100$) is applicable primarily to metals. Metals have the requisite high allowable stress levels to prevent permanent deformations in manufacture and handling, with very thin cross-sections. In the weightless environment of space only the topmost layer of a mirror, which carries the reflecting surface, is theoretically required. The thin mirror construction is an approach to this ideal but, of course, special supporting means are required in manufacture and launch, while in the 1 or more g terrestrial and launch environments.

The high t/D of approximately $1/6$ for the solid alternatives derives from the low allowable stress levels of the traditional vitreous materials employed, and when such large diameters as 3 meters are considered, the weight and volume increase rapidly. Further, as the mass of a conventional vitreous mirror becomes large it becomes more vulnerable to thermal shock, due to the low thermal conductivity. Lengthy and gradual stress relief techniques are required to avoid the possibility of cracks.

The "eggcrate" construction approach is associated with vitreous materials and is a refinement of conventional solid construction. Large depth is retained (much in the manner of an I-beam) to keep stresses (hence strain) within allowable levels, but weight is sharply reduced by removal of interior material. The same vulnerability to thermal shock exists, made more acute at the fused boundaries, and leads to the possibility of separation either immediately or later in the life of the mirror. Should the flange thickness be made too small minute deformations can occur during the polishing process resulting in "show through".

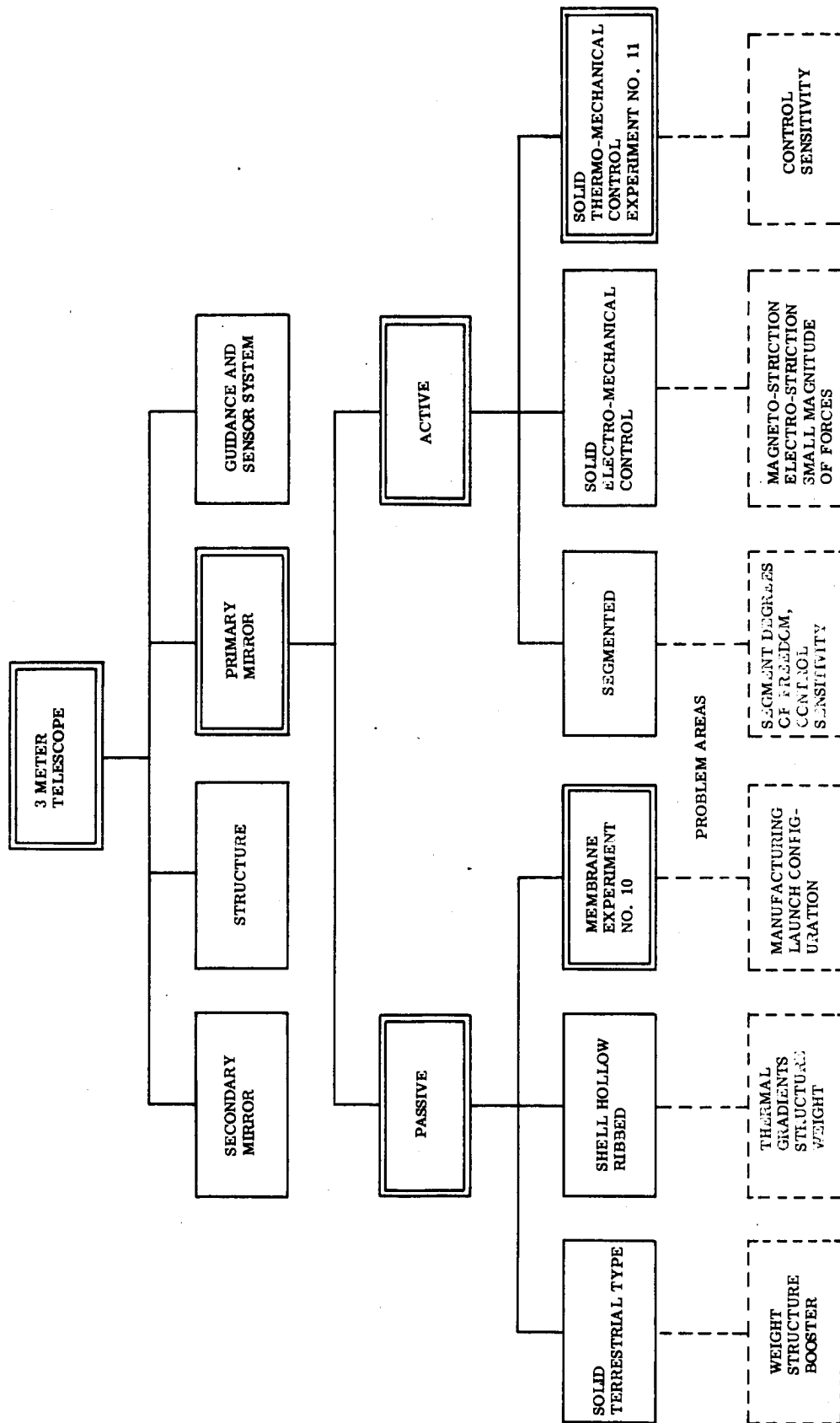


Figure 11.1-1. Survey of Mirror Construction Alternatives.

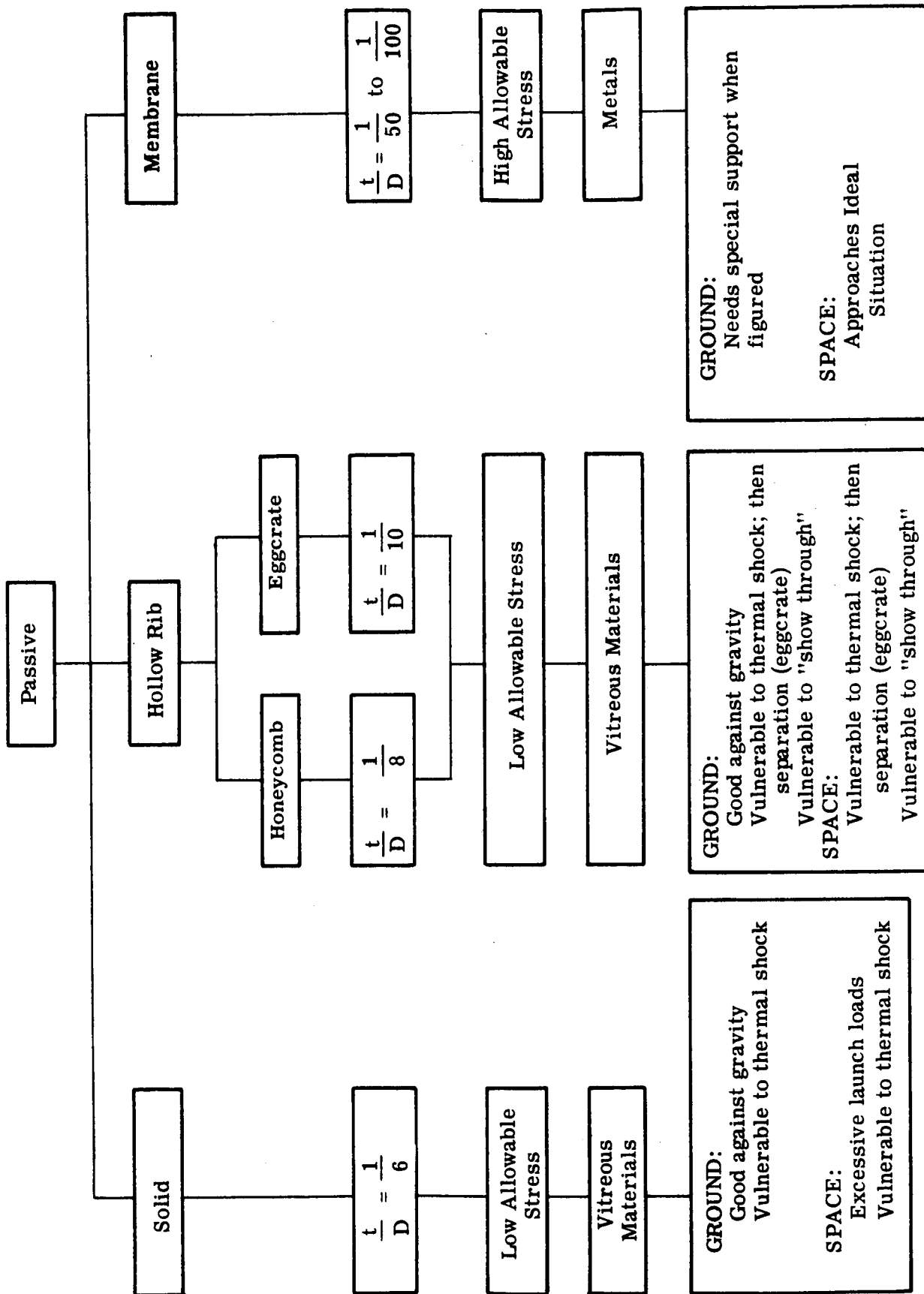


Figure 11.1.1-2. Passive Construction Approaches to Large Space Primaries.

Honeycomb construction is also associated with vitreous materials but due to asymmetry is not as efficient structurally as eggcrate, resulting in greater depth and weight though still less than conventional. Honeycomb is one-piece construction, however, which eliminates the separation possibility. "Show through" can still occur if the flange is thin enough.

Regarding active approaches to mirror construction, the segmented approach if successful could divide a large mirror into component sizes where vitreous materials are more effective. Optically, it is required that the movement, as a unit, of a large section of a mirror contribute to an increase in figure accuracy. Servo loops of sufficient precision and accuracy must also be implemented.

The two space experiments recommended in the section include a Thin Mirror Passive and Thermo-Mechanical Active Solid approach.

11.2 JUSTIFICATION

The need for large aperture (100 to 150 in. dia.) diffraction limited optics in space, to extend the frontiers of astronomy, has been well

identified. (1), (2), (3), (4) Such high performance optics should make possible system resolution limits in order of .01 arc second with which details of stars could begin to be discerned. Stability to better than this magnitude over several hours is a necessity; the study of this is the subject of another experiment (section 10.2.1).

Given the need for large apertures, thin mirror primaries become desirable. Given the need for "diffraction limited" performance, correctable primaries may be necessary.

11.2.1 Contribution and Need

The major advantage of the thin mirror concept is basically a lower unit weight. Section 11.5.4.2 describes a comparison of designs for a rigid

-
- (1) Space Research - Directions for the future, report of a study by Space Science Board, Woods Hole, Mass., 1965.
 - (2) Space Program in the Post Apollo Period, Report of President's Science Advisory Committee, Feb. 1967.
 - (3) Applications in Astronomy Suitable for Study by means of Manned Orbital Observations and Related Instrumentation and Operational Requirements, edited by Lawrence W. Frederick, NASA Grant No. NSG-480, Oct. 1963.
 - (4) A system study of a Manned Orbital Telescope, Oct. 1965, Aerospace Group, The Boeing Co., Seattle Washington.

(Goddard Experiment Package metal mirror) system for a one "g" environment during building and launch vs that for a zero "g" environment.

Table 11.5.4-2 shows that the saving in weight of large optics is expected to exceed 80% of the rigid design. Thus the larger the optics the more desirable the thin shell approach becomes. The major contribution of the thin mirror experiment is the determination of the permanent strain deformation, and hence maximum stress a mirror may experience and still remain within figure tolerance for diffraction limited performance. Thus, the maximum stress level (which is related to the Precision Elastic Limit or P.E.L.) will be determined with a satisfactory confidence level, so that with appropriate factors of safety large (in excess of 3 meters) thin diffraction limited mirrors may be designed, manufactured and launched into orbit. The requisite data will be obtained from a scale model of the ultimate 3 meter mirror by extrapolation in accordance with a precise scaling law. A secondary contribution will be the confirmation or indication of new techniques for orbital assembly and alignment.

The main contribution of the active optics experiment will be a technique for correction of mirror figure to diffraction limited quality, if prior to use, the mirror undergoes excessive figure changes resulting from thermal and mechanical sources. The results of this experiment will provide the needed technological data for the ultimate design of large actively controlled optical systems. Major areas of data needed or to be gathered are:

- a. the effect of the weightless condition of space on the mirror and correction system.
- b. the effect of the various thermal conditions of space on the mirror correction system.
- c. the precision to which a mirror surface can be actively corrected and maintained at diffraction limited quality.

11.2.2 Need for Space Testing

The necessity of gathering technology data from experiments in a space environment is based on two important factors:

- a. zero "g" ground test simulation techniques are inadequate for determining figure deviations in diffraction limited mirrors; to achieve diffraction limited quality the mirror surface must be figured to approximately $\lambda/30 = 0.8 \times 10^{-6}$ inch, as shown in OTAES part I section 4.10.5.7, and thus measurements of this order of magnitude must be made. Section 11.2.2.1 discusses the possible zero "g" simulation techniques, and their individual disadvantages.
- b. the precision elastic limit (P.E.L) (See section 11.5.4.4) must be determined with the purpose of establishing allowable design stresses for configuring large thin mirrors. The minute permanent

deformations (plastic strains) which occur at or below the P.E.L. **in a large mirror will impart an exceedingly small additional** deflection when compared to the effect of gravity (ratio greater than 1:1000) and is therefore not reliably measured on Earth. Section 11.2.2.2 discusses the P.E.L. consideration and its effects on thin mirror design.

11.2.2.1 Limitations of Ground Based Zero-G Testing

An earthbound simulated space laboratory to test optical mirrors for ultimate use in space should have a zero gravity or a uniform support pressure capability. Changes in pressure and gravity are changes in applied loads on the mirror and alter its characteristics. These parameters must be nullified on earth to simulate a space environment. Some approaches to compensate for these loads and their merits are reviewed.

The design baseline mirror to be tested has a uniform section of 10 millimeters (3/8 inch) thick, 1.3 meters (50 inches) in diameter and is constructed of beryllium. For a mirror with a spherical shape and a radius of 3.2 meters (125 inches) the sagittal dip is approximately 6.35 cm (2-1/2 inches). The weight of this dish at 1 g is 22.6 kilograms (50 pounds). The surface area is 12,600 cm² (1960 in²).

a. Water Flotation Method

The mirror may be floated on water with its reflective surface up and central aperture closed. The test instruments are rigidly supported above and independent of the mirror. The surface figure is tested as the mirror free-floats below.

The mirror and test instruments alignment will be difficult to maintain in a paraxial focus condition:

- (1) the free-floating mirror will shift due to seismic tremors.
- (2) the mirror is subject to bending at the unsupported ends for the section above the water-level. Analysis of this situation, as shown in section 11.5.1, indicates that a 3.35 inch overhang will exist, resulting in a maximum deflection under its own weight of approximately 7.42×10^{-4} in. This is far in excess of the maximum figure deviation of $.8 \times 10^{-6}$ inch allowed for diffraction limited performance.

b. Mercury Flotation Method

The mirror is floated on a mercury pool as in "a" above. Stability between test instruments and mirror surface cannot be obtained because:

- (1) Seismic tremors will shift mirror axis with respect to test instruments and a stable interferogram cannot be obtained for $1/20\lambda$ or better.

- (2) Mercury will support the mirror over a smaller area than water and permit a larger mirror overhang, since mercury has specific gravity = 13.6 and beryllium has a specific gravity = 1.85. This increases the bending moment at the unsupported ends. Analysis of this situation, as shown in section 11.5.1 indicates that a 13.7 inch overhang will exist, resulting in a maximum deflection under its own weight of approximately 3.08×10^{-3} inches, far in excess of the maximum figure deviation of 0.8×10^{-6} inch allowed for diffraction limited performance.

NOTE: Attaching the test apparatus to the mirror to offset item (1) aggravates item (2).

c. No Overhang Flotation Method

In this technique the mirror rests on a fluid which just supports the entire back face of the mirror. The mirror is held against the fluid by 10 gm spring loaded limit stops. This method is unfeasible since the density of the fluid would have to be $\gamma_f = 0.55 + 0.75 \times 10^{-3} \text{ gm/cm}^3$, as is shown in section 11.5.2; instability would exist if the density of the fluid exceeds the positive tolerance ($0.75 \times 10^{-3} \text{ gm/cm}^3$), since the load on each unit stop would exceed 10 gms; a fluid density less than γ_f (0.55 gm/cm^3) would result in submersion of the mirror blank.

d. Submersion Method

Ideally, the mirror should be totally submerged in a fluid whose specific gravity is equal to that of the mirror (1.85). A review of known fluids does not indicate any fluid near the specific gravity of beryllium which would also be clear enough for satisfactory light transmission.

e. "Zero-g" Aircraft Method

Zero-g aircraft tests can be achieved for periods up to about one minute; however the time required to obtain a good interferogram study is about one hour. In consequence the one minute test time available in airborne tests is inadequate to obtain satisfactory results.

f. Pneumatic Flotation Methods

The mirror is floated on a cushion of air. Except for the loads of the 10 gram limit stops, the mirror is essentially a free floating body.

Disadvantages of this method are:

- (1) There is a degree of uncertainty in obtaining a completely uniform support, since the pressure of the air during the test period must be controlled to $17.65 + 0.0237 \text{ mm}$ of water, as shown in section 11.5.3. If the positive tolerance exists, that is the pressure is 17.6737 mm of water, bending deformation

occurs in the mirror blank, resulting in a maximum deflection at the center of 5.4×10^{-6} inch. This is in excess of the tolerable figure deviation of 0.8×10^{-6} inch.

- (2) The temperature of the supporting gas must be uniform.

11.2.2.2 Precision Elastic Limit Consideration

In an ideal situation a thin membrane (shell mirror) can be figured to optical tolerance in a strain free supporting nest while in the gravitational field of one "g", since the composite structure simulates a thicker mirror which is easily figured. Once removed from the nest for assembly in a telescope, gravitational effects will distort the mirror **figure; however**, when placed in orbit (zero "g") the mirror will revert to the strain free condition thereby returning to the figure tolerances that were attained on earth.

In a real situation this is not completely true. It is well known that a material experiences a permanent "set" or deformation when its elastic limit is **exceeded**; accordingly equipment is designed to operate at stress levels well below the elastic limit of the component materials. However, in precision instrumentation, it has been found that even when designs have been engineered substantially below the elastic limit, unexplained errors have occurred.

It has been established that in order to maintain performance specifications in precision systems, an exact determination of proportional limit must be made. Minute permanent deformations occurring in materials below the conventional proportional limit now require a new limit. This limit has been termed the Precision Elastic Limit (P.E.L.) and is defined as "the stress which produces a residual strain or deformation of one micron per meter (or one micro inch per **inch**"). This strain is a permanent set in the material and is sufficient in magnitude to change the diffraction limited figure of an optical element.

A discussion of the P.E.L. as it applies to beryllium is given in Section 11.5.4.4. In order to define clearly the thickness-to-diameter ratio for large thin mirrors it is necessary to define precisely the material properties, including the P.E.L. The critical factor requiring definition in this experiment is the maximum stress level that will not impart detrimental strains to the mirror. Accuracies required for diffraction limited performance restrict permanent deformation to the order of magnitude of 0.02 micron. The deformation in a one "g" environment of a thin mirror due to its own weight is of the order of 20 microns. In view of this ratio of more than 1000 to 1 the permanent deformations are not accurately measurable in the one "g" environment on earth. A method of determining and evaluating the maximum stress level is discussed in OATES Part I Section 4.11.5.1. Once the P.E.L. or maximum allowable stress is determined with a satisfactory confidence level, appropriate factors of safety may be applied to thin mirror design and fabrication.

11.3 LARGE OPTICS EXPERIMENTS

11.3.1 Thin Mirror Nesting Principle, Erection, Alignment and Testing of Large Optics in Space

In this experiment, a scale model telescope of 1.3 meter (50-inch) diameter employing a thin beryllium mirror will be erected, aligned and tested in space. In orbit the mirror figure will be evaluated by using interferograms. Mirror figure patterns will be obtained prior to and during orbit. These patterns will be compared. The results of this experiment will serve as a guide for the ultimate goal of the design and fabrication of a 2.5 to 5 meter (100 to 200 inch) aperture reflector type telescope for optical use.

The primary advantage of a thin mirror design is the weight saving gained throughout the entire telescope system, and an analysis of these savings is shown in section 11.5.4.2. This design, of course, has the attendant problems of support during manufacture, handling and launch. Manufacture and test of a thin mirror in a "nest", where the mirror is uniformly supported at all times provides a condition that can be called a simulated zero gravity condition. Simulated zero gravity is defined as a situation where the mirror is uniformly supported in such a way as to eliminate any deformation due to its own weight. This type of support permits the construction of a thin mirror on earth so that it will perform adequately in the weightlessness of space.

Selection of the mirror material becomes dependent on such factors as stiffness to density ratio (E/ρ), thermal warpage index (TWI), dimensional stability, micro-creep, thermal expansion inhomogeneity, and polishing characteristics, which also affect the possible construction methods of the mirror. These factors are discussed in detail in the text and a comparison is made between likely mirror materials, including beryllium and quartz

A thin beryllium mirror has been conceptually designed based upon the known precision elastic limit (P.E.L.) of this material. This is a stress below the conventional elastic limit of beryllium, at which a minute residual (plastic) strain of 1 micro-inch/inch occurs. This strain is a permanent set in the material and is sufficient in magnitude to change the diffraction limited figure of an optical element. These minute permanent deformations in a large mirror will impart an exceedingly small additional deflection when compared to the effect of gravity (ratio greater than 1000:1) and therefore cannot be reliably measured on earth. The ultimate design of large thin mirrors (120-200 inch in diameter) will depend on the knowledge of an appropriate design stress, so that the mirror can be fabricated, launched and placed in orbit while still maintaining its figure within diffraction limited tolerances. Therefore, the P.E.L. must be determined with greater accuracy and related to the residual strain that just causes the mirror figure to be out of tolerance. The thin model mirror is designed so that all induced stresses will be limited to a margin of safety of 1/19 of the P.E.L.

Prior to launch the mirror will be uniformly loaded to the P.E.L. and the load released. In orbit the effect of any induced residual strains on mirror figure will be determined; the location on the mirror surface which is out of figure tolerance will be noted and the P.E.L. determined. The results of this model test will be extrapolated, to the design of larger aperture mirrors using a scaling technique.

Optical testing for mirror figure will be performed using a scatter plate interferometer. A description of this instrument as well as its implementation is given in the text.

11.3.1.1 Thin Mirrors Experiment Design

11.3.1.1.1 Experiment Objective

This experiment will obtain the necessary data for the structural design of 3 meter (120 inch) or larger, thin shell primary mirrors for future spaceborne telescopes. It will also define some techniques necessary for in space alignment of these spaceborne systems.

The primary objective is to gather data on the Precision Elastic Limit (P.E.L.), i.e., the stress at which permanent deformations due to strain will place the mirror out of figure tolerance when used in space. These data obtained from a scale model of the ultimate 3-meter mirror will permit determination of those design parameters most sensitive to extrapolation.

The secondary objective of this experiment is to investigate the ability to automatically align and test a large optical system in orbit.

11.3.1.1.2 Experiment Plan

The thin mirror concept as discussed in section 11.3.1.1.3 results in a light weight primary mirror requiring only temporary support for fabrication and launch and minimum support in orbit.

The fabrication and testing, on earth, of a thin shell primary mirror will be in a nesting fixture. Essentially, this "nesting technique" is such as to provide uniform mechanical support and thermal continuity throughout the initial manufacture of the thin shell primary mirror, so that it is completely undeflected while in its nest. OTAES Part I section 4.10.5.4 describes this method in detail.

Prior to launch, the mirror will be removed from the nest and rim mounted. It will be subjected to a uniform transverse load in order to stress the mirror to the P.E.L. of 175 kg/cm^2 . This load will then be relieved and the mirror prepared for launch. This is the prestress condition used to determine the allowable design stress for future diffraction limited mirrors in space. OTAES Part I section 4.11.5.1 describes and reviews the stressing effects of this load.

After removal from the prestressing fixture the mirror will be placed into the launch protection configuration. A feasibility study, OTAES Part I section 4.11.5.6 has been made to show that the mirror may be protected from the forces of vibration and shock in all directions (i.e., translation and rotation of mirror surface, and translation parallel to the mirror surface) as well as from the fundamental "free-free" mirror resonance (existing at a frequency of approximately 75 Hz).

The means of protection envisioned is the placement of pneumatic bladders on the front and rear surfaces of the mirror and a toroidal bladder about the rim. The spring constants and requirements of these bladders are described in OTAES Part I section 4.11.5.6. Thus, it can be assumed to an acceptable high confidence level that the stress applied to the mirror during launch and insertion into orbit will also be safely below the P.E.L. (or below the design stress of 8.75 kg/cm^2).

Once in orbit the bladders are deflated and the thin primary mirror is held against kinematic mounts by spring loaded clamps at three locations 120 degrees apart. Then the primary and secondary mirrors will be aligned and focussed as described in OTAES Part I section 4.11.5.7.

The mirror will then be tested for "figure" by interferometric means. OTAES Part I section 4.11.5.8 outlines how the mirror figure will be determined by this method. The interferometer data will be used to determine the mirror shape and hence how far from the manufactured shape it has gone. After determining the new shape, the strain induced by the controlled stressing at the P.E.L. of 175 kg/cm^2 can be calculated and plotted as a function of radius from the center. The stress (also a function of the radius) is calculated as it varies from a maximum of 175 kg/cm^2 at the center to a lesser value at the rim so that the deformations may be directly compared to the stress. Thus by reviewing the results and comparing with the maximum allowable deformation for a diffraction limited mirror, a maximum P.E.L. can be established. This P.E.L., with an appropriate safety factor, can then be used in the design of large diffraction limited mirrors using the scaling law developed in section 11.5.5.

11.3.1.1.3 Thin Mirror Design

The design of the thin mirror hinges on determining the appropriate material, minimum allowable thickness and the allowable thickness to diameter ratio.

Beryllium S-200-C is currently favored for the baseline thin mirror material on the basis of thermal, physical, stiffness-to-mass ratio and manufacturing considerations. Section 11.5.4.5, "Comparison of Mirror Materials", reviews these considerations. Of special interest is Table 11.5.4-3 of that section. A minimum thickness of 0.125-0.2 inches has been established for beryllium mirror application, based upon the considerations discussed in section 11.5.4.1. The basis of determining the diameter to thickness ratio is the P.E.L. discussed in Section 11.5.4.4. At any selected P.E.L., a family of mirrors may be designed for various diameters with minimum thickness. Since the

primary purpose of this experiment is to determine the P.E.L. more precisely than is presently known, a factor of safety of 19 will be used in this experiment. After determining the allowable P.E.L. by means of this experiment more realistic factors of safety permitting even thinner mirrors will be possible. OTAES Part I section 4.10.5.2 develops the means of determining the mirror thickness vs. diameter, and table 4.10-1 of this section tabulates mirror thickness and weight vs mirror diameter.

Based upon this, the recommended mirror size for the experiment is 1.3 meters (50 inches) and thickness of 10 millimeters (0.38 inches) of Beryllium S-200-C.

Since the same P.E.L. or maximum stress allowable is used for each mirror diameter, the following relations hold true:

- a. The maximum deflection of each mirror is the same. (OTAES Part I section 4.10.5.2)
- b. The stress value and variation from the center to the outer edge is the same for each diameter when comparing stress vs. the ratio of r/a , where r is the radius from the center to the point of measurement and a is the maximum radius of the mirror (one half the diameter). For a description of stress variation across the mirror see OTAES Part I section 4.11.5.1.

The shape best suited and selected for manufacturing in this experiment is a sphere. It is much simpler to figure than an aspheric surface and, in addition, it is possible to take advantage of the condition of unity magnification of a spherical form during mirror figure evaluation, i.e., it permits Foucault and interferometer tests to be performed at the center of curvature. This is realistic since a sphere varies little from a paraboloid for the case we are considering and will satisfy the experiment requirements. (OTAES Part I section 4.10.5.3).

The fabrication and stress relieving of this mirror will be carefully planned and executed so that all handling and manufacturing stresses imparted to the mirror will have been within $1/19$ of the P.E.L. (or 8.75 kg/cm^2) and high confidence that the mirror has not experienced any stresses near or beyond the selected P.E.L. of 175 kg/cm^2 (2500 psi) is possible.

11.3.1.2 Experiment Measurement Objective

This section describes the parameters to be measured and how these measurements relate to specific technology advances. These measurements are:

- a. Measurement of mirror figure deviation as a function of residual strain. The resultant technological advances will be
 - (1) Accurate determination of the P.E.L.
 - (2) Determination, with a high level of confidence, of reasonable design stress factors of safety, to produce a lightweight, high quality mirror for large space optics (i.e., 120-200" mirror) requiring near diffraction limited performance.

- b. Measurement of mirror figure deviation as a function of stability of a thin mirror. The technological advance will be the determination of the mechanical and thermal stability of a large thin mirror as a function of time in orbit. This will aid the determination of the minimum thickness (as a function of diameter) for thin mirrors.
- c. Measurement of thermal gradients. These measurements will aid in the determination of possible figure distortion as a function of thermal differentials. These distortions must be separated analytically from mechanical strains to evaluate the effects of residual strain and mechanical and thermal instability.

11.3.1.3 Experiment Equipment Design

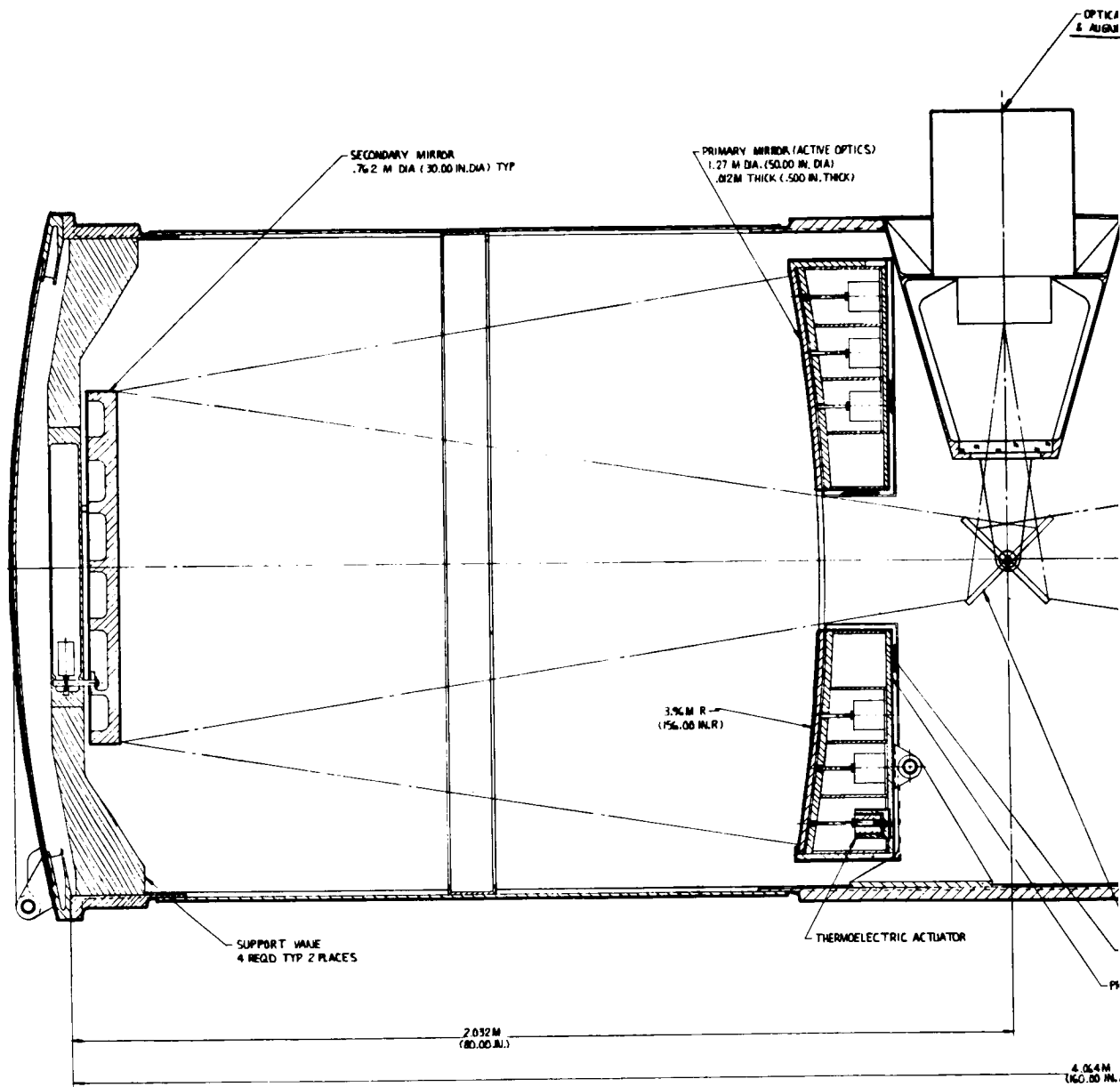
11.3.1.3.1 Configuration of Mirror Test Well

The primary mirror test well is a cylindrical tubular housing common to both the thin mirror and active optics experiments as shown in figures 11.3.1-1 and 11.3.2-1. The configuration and relative locations of the secondary mirrors are identical and the optical measurement instrumentation is common to both experiments. The active optics experiment design is discussed in section 11.3.2.3.1.

The thin primary mirror, designed and fabricated as described in section 11.3.1.1.3 will be launch protected by a pneumatic bladder system as described in OTAES Part I section 4.11.5.6. Once in orbit the primary mirror will be automatically erected in the following manner. The bladders will be deflated slowly by remote bleeding, while the kinematic mounts located in back of the primary are automatically released. As the mirror is forced against the kinematic mounts, spring loaded clamps will automatically rotate into position to secure the mirror at three points on the periphery. The mirror cover doors, securing the mirror face bladders during launch, are automatically retracted against the structural wall.

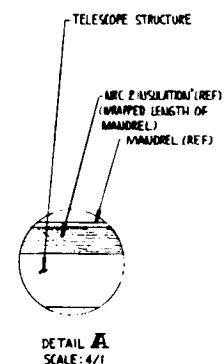
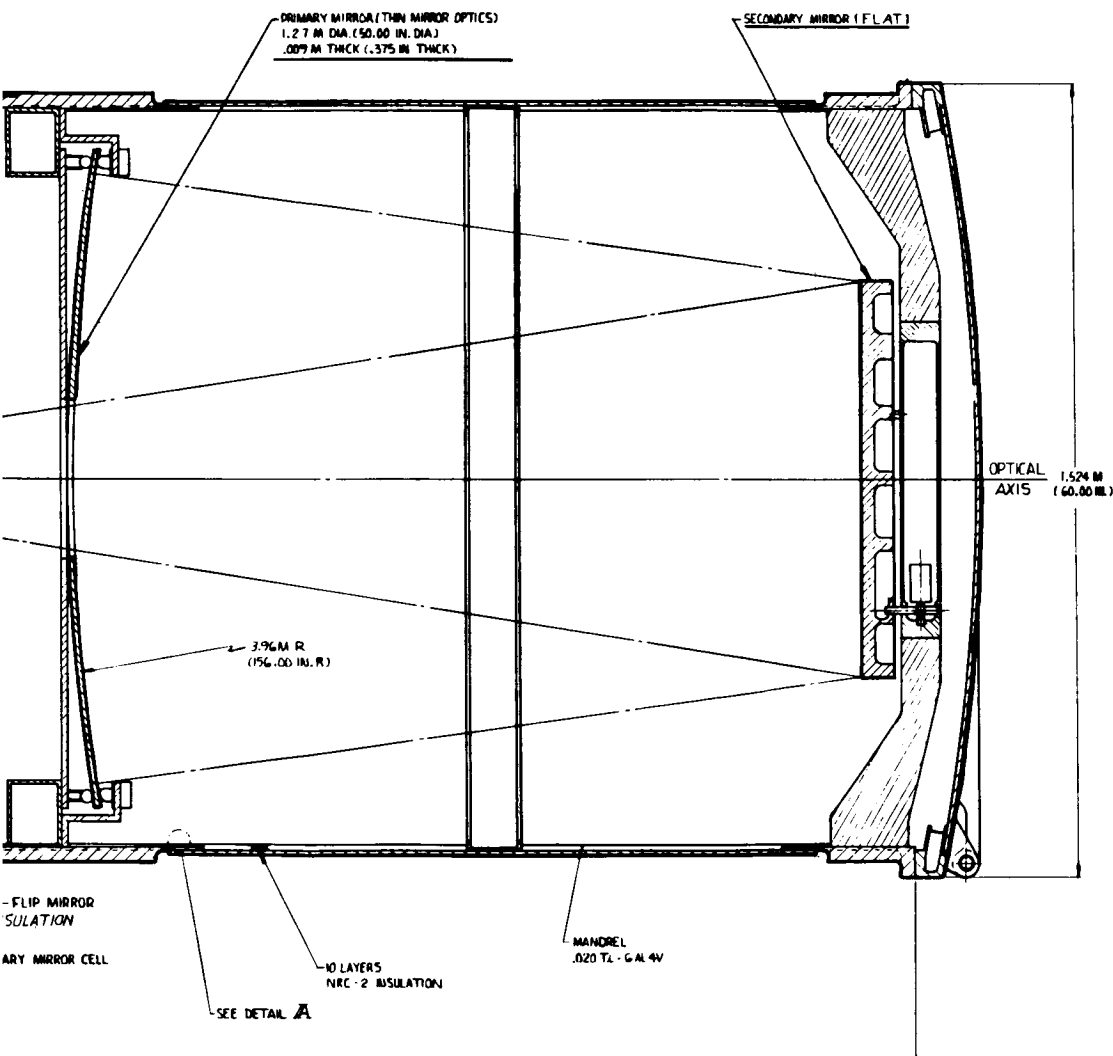
Alignment of the optical axis is achieved by automatic bi-directional tilt of the plano-secondary mirrors with respect to the primary using autocollimation. A flip mirror, common to both the thin mirror and the active optics experiments, is located on and aligned at 45 degrees to the optical axis in the center of the well and links the optical test and alignment instrumentation to each of the experiments. The test instruments, described in section 11.3.1.3.2 are located at the center of the curvature of the primary mirrors, by insertion through an opening in the mirror test well.

The secondary mirrors, which are identical, are mounted on four vanes (90 degrees apart) which secure to reinforced ribbed rings. Hatch covers at both ends of the well, which allow the experiments to be exposed to space, are also hinge connected to the ribbed rings. The central part of the well in which both primary mirrors are supported and which interfaces with the spacecraft is a reinforced structure similar to the ribbed rings. The reinforced structure is 0.25 inch thick aluminum shell, with 30 axial



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FIGURE TEST
INSTRUMENTATION



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Figure 11.3.1-1. Primary Mirror Experiment Well-Thin Mirror Nesting Principle.

0.75 inch x 0.25 inch ribs., having a maximum O.D. of 58 inches. Connecting the reinforced sections is aluminum tube 0.0625 inches thick, 56 inches in O.D. The tube is further reinforced to prevent buckling by inserting channel rings midway in the tube spans. A preliminary structural analysis has been performed and is shown in section 11.5.6.

Each half of the test well is insulated by inserting a tubular mandrel (.020 inches thick, ti-6Al4V) covered with 10 layers of NRC-2 (aluminized-mylar) just inside the wells. The back of the active optics primary is further insulated to prevent heat dissipation to the thin mirror primary.

11.3.1.3.2 Description of Supporting Instrumentation

The optical instrumentation used in the mirror experiments consists of an autocollimator, a Foucault tester and the scatterplate interferometer. Each of these instruments will have a common adapter so that it can be secured to the same mounting structure in the pressurized compartment at the center of curvature of the primary mirror.

The autocollimator will be used to align the secondary mirror to the optic axis of the primary by reflecting an image back on itself using collimated light, and aligning the crosshairs of the return image with the fixed crosshairs of the autocollimator reticle. This is accomplished by remote bi-directional tilt of the secondary mirror.

The Foucault tester is used to determine the center of curvature of the primary mirror. The tester consists of a pinhole light source and a sharp opaque vertical (knife) edge. The observer sights through the tester at a point just behind the knife edge and repositions the knife edge axially until the entire mirror appears to darken uniformly. This position of the knife edge then represents the center of curvature of the primary mirror.

The scatter plate interferometer, also mounted at the center of curvature, is used to check the figure of the primary mirror. A description of the instrument is given in section 11.3.1.4.1.

11.3.1.4 Data Management

The principal data to be obtained from this experiment will be the information required to evaluate primary mirror figure. Additional data will be taken to determine the thermal gradients in the primary mirror which will indicate the existence of possible thermal distortions.

Mirror figure deviations will be measured using the scatterplate interferometer (section 11.3.1.4.1), with the scatterplate located at the center of curvature of the primary mirror as defined by the Foucault test, and tilted slightly to achieve a vertical line fringe pattern. The interferogram obtained will be monitored by a vidicon T.V. camera, and displayed in the spacecraft T.V. monitor. It will also be transmitted to the ground station for interpretation (section 11.3.1.4.1), and comparison with the reference interferogram of the corresponding diffraction limited mirror.

To determine the thermal gradients, temperature sensors will be mounted at prescribed locations on the back surface of the thin mirror. By observing the thermal control meter, which will have readout capability for each of the sensors, the thermal profile will be determined. Inserting this profile into a thermal distortion analysis computer program (Ground Based) the calculated thermal distortion will be compared to the figure deviations as determined by the scatterplate interferometers. This will provide a technique for separating thermally induced distortions from the distortions resulting from plastic strain.

11.3.1.4.1 Optical Measurement Technique

An interferometer method has been developed that tests a mirror at or near the radius of curvature by means of a scatterplate. ⁽¹⁾ The scatterplate interferometer has been used to verify the figure of large aperture mirrors to tolerances of 1/50th of a light wave. The same instrument will be used during ground and space tests to eliminate instrument error.

In using the scatterplate interferometer to measure mirror figure changes, a fringe pattern results. A method of relating figure changes to the resulting fringe pattern is shown in OTAES Part I, section 4.11.5.8.

Figure 11.3.1-2 is a schematic of the scatterplate interferometer. A laser beam is projected by lens 1 to focus on the center of the spherical mirror at point E. A scatterplate intercepts this beam (Point A) so that part is transmitted undeviated to point E. Part of the beam is deviated or scattered to Point F. The mirror reflects both beams which re-combine (Point C) on the scatterplate. A difference in length exists between A E C

(1) "Scatter-Fringe Interferometry" by J. M. Burch, Journal of Optical Society of America, Vol. 52, page 600, 1962.

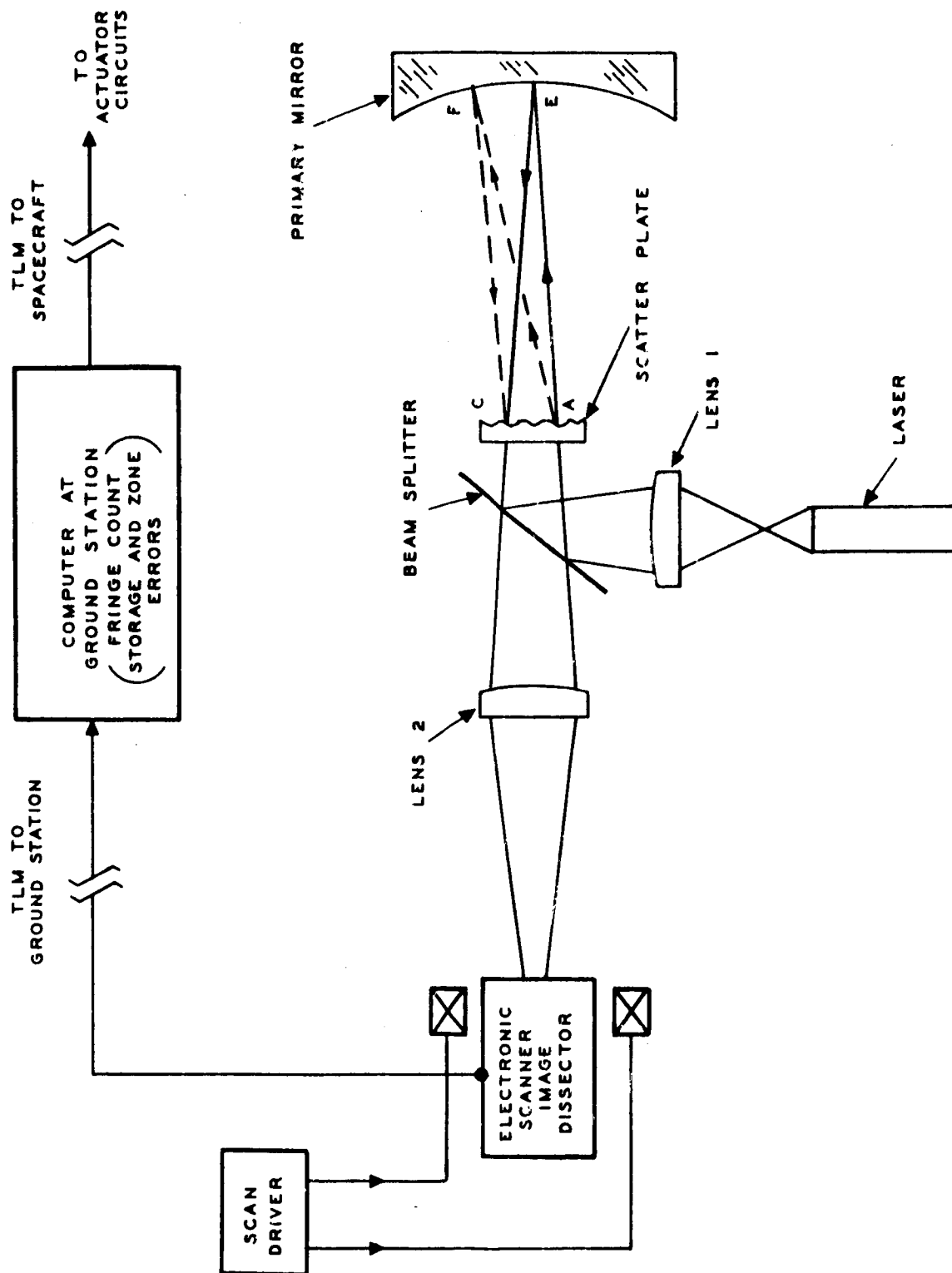


Figure 11.3.1-2 Scatterplate Interferometer with Fringe Readout and Error Computation

and A F C if the mirror is not a perfect sphere. This path length difference will create a pattern of dark interference fringes. These are projected by Lens 2 onto the electronic scanner, which reads out the pattern to a ground station.

The fringe pattern consists of vertical bands and is obtained by tilting the wave front (i.e., rotating the scatterplate slightly). This is preferred over the concentric circle interference pattern because the scatterplate remains in the focal plane, while defocusing is required to obtain concentric fringes, and fringe definition is thereby degraded. The effect of fringe definition becomes important in data reduction, and a vertical fringe pattern allows linear interpolation of fringe order using a rectangular grid. Thus measurement sensitivity is enhanced relative to the concentric fringe pattern, which requires interpolation based on a non-linear quadratic function.

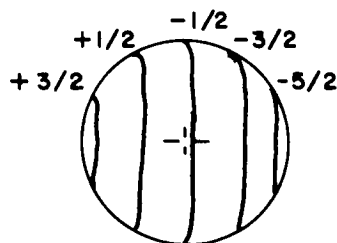
Using this information as input data to a computerized mathematical procedure, the output gives the departure of the actual tested surface from the reference mathematical surface, as based on a least squares fit of the input data. This method is applicable to many types of surfaces (spherical, parabolic, ellipsoidal, etc.), since the least squares fit can be made to each of the surfaces. This program has recently been used at Kollsman to evaluate a parabolic mirror which was optically figured using an ion beam. Correction of the surface was achieved to an Rms level of $\lambda/22$.

A sketch of this interferogram, shown in figure 11.3.1-3, is included to illustrate the general method.

In order to reduce the data from a scatterplate interferogram the following procedure must be followed: The paraxial radius of curvature of the surface must be measured and supplied as input data to the program. Secondly, the interferogram of the surface is broken up into a grid of points which should represent approximately equal areas per point.

A grid of points on the interferogram sketch of a paraboloid is shown in Figure 11.3.1-3. There is a one to one correspondence between the grid points and the actual surface coordinate. A scale factor which must be given as input data is the ratio of the actual surface diameter to the diameter of the interferogram. The order of the fringes are also determined and labeled.

If the scatterplate is just inside the center of curvature of the mirror and is displaced slightly to the right, the fringe order is as shown below.



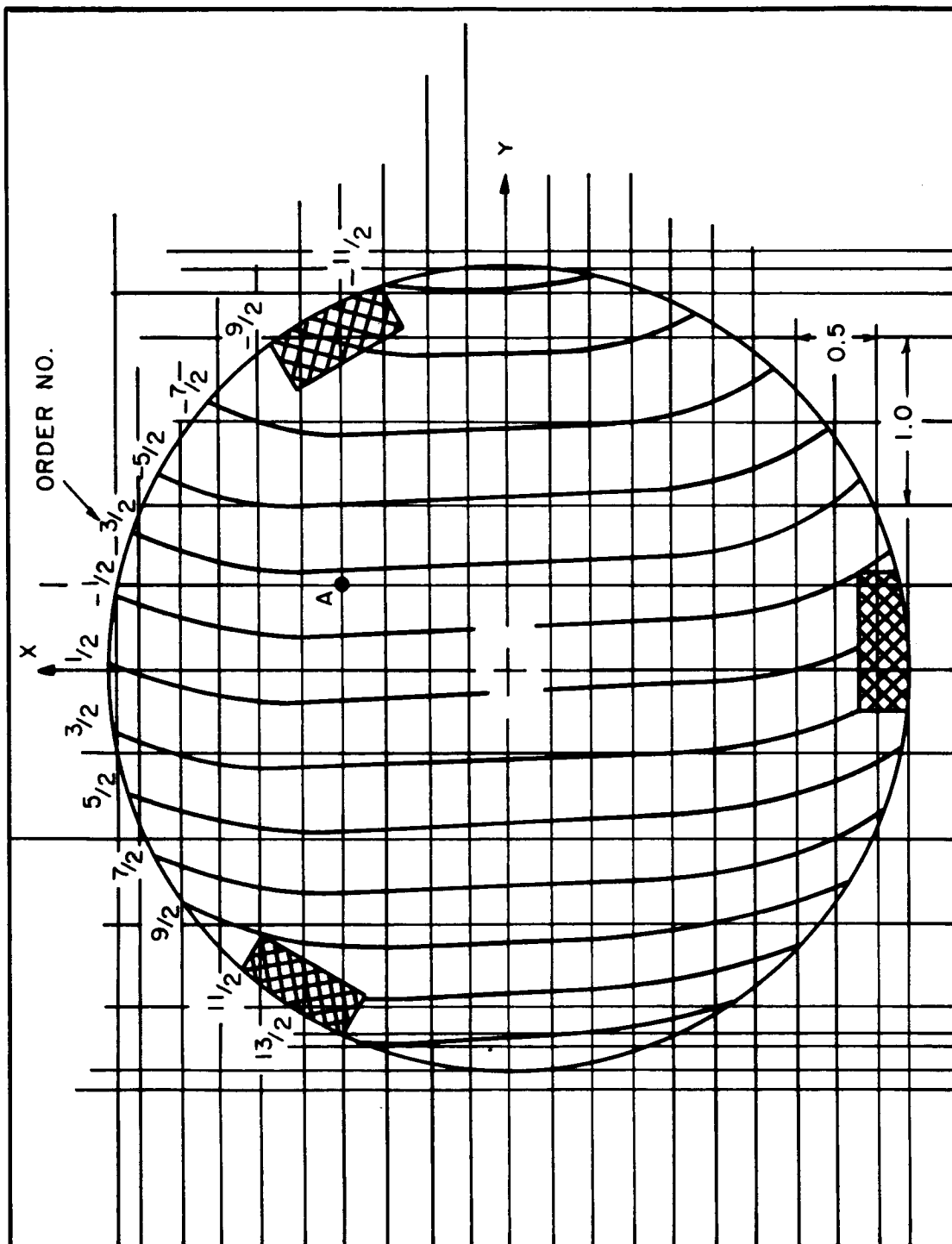
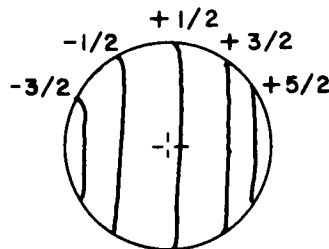


Figure 11.3.1-3 Sketch of Scatterplate Interferogram

If the scatterplate is just inside the center of curvature of the mirror and is displaced slightly to the left, the fringe order is as shown below.



If the scatterplate were just outside the center of the curvature the above situation would reverse.

Input data to the program are the interferogram coordinates (X and Y) of each point along with the order corresponding to that point. If a point lies between two fringes as does point A, in figure 11.3.1-3, its assigned order is linearly interpolated from its distances between the two orders. In this case for point A, $X=1$, $Y=0.5$, order ≈ -1.25 . For every point on the grid a similar set of data is entered. This information, with the scale factor, radius of curvature of the surface, and the total number of points N is then processed in a "Surface Analysis Program" and the following output is given. For each input point on the interferogram, the corresponding actual surface coordinates are listed along with the departure, in + inches and + wavelengths of light, from the reference to the actual surface being analyzed. Thus, if a sufficiently small grid spacing is chosen (large number of points) a contour map of the surface is easily obtained. Also, a figure of merit for the surface is given in terms of the r.m.s. departure from a perfect surface, (the root square deviation of all points on the surface). This number is directly connected to the expected image quality capabilities of the surface being measured.

The procedure for supplying input to the computer program can be one of the following alternatives; a technician using manual techniques; partial automation using a manually scanning comparator with a digital readout and card punch; complete automation using a programmed scanning mechanism with the capability of either digital punchout or direct writing on binary tape. The complete manual reduction takes about two hours, semi-automatic about 15 minutes and completely automatic several seconds.

11.3.2 Primary Mirror Figure Test and Correction

In this experiment, a thin spherical 1.3 meter beryllium mirror is attached to a specially constructed base structure. The surface and figure of this mirror will be controlled utilizing thermo-mechanical actuators located between the mirror and its base structure. Evaluation of the mirror figure will be accomplished by obtaining interferograms of the mirror surface, which will be obtained during final figuring on Earth and during orbit. These patterns will be used to determine the precision of the correction capability and this can be used as a guide for the design of large, figure correctable, reflectors. The interferogram will be interpreted electronically on the spacecraft and transmitted to a ground based computer. Calculations will then be performed and the results transmitted back to the spacecraft in the form of commands indicating the thermo-actuator locations and forces required for correcting the figure. The force required will be an analog of thermal expansion or contraction of the actuator rods. Only in the zero-g environment of space, where the gravity effects of the rods and other attachments to the mirror are nullified, can the proposed technique be investigated and design data for a larger mirror be obtained.

The figure correction technique has been examined analytically using a digital computer program. Assuming an initial figure error pattern in which the mirror has deformed in the shape of a paraboloid, correction to within a diffraction limited figure deviation of $\lambda / 30 = 0.8 \times 10^{-6}$ inch has been shown to be feasible, (section 11.5.7.).

11.3.2.1 Experiment Design

11.3.2.1.1 Experiment Objective

The object of this experiment is to investigate the possible active control of the figure of a lightweight mirror, in order to achieve and maintain diffraction-limited optical performance in orbit.

A secondary purpose of this experiment is to gather technological data in order to establish the requirements for a 3 meter or larger primary mirror that can be figure corrected and maintained in orbit. This objective is obtained by ground and space testing a model with selected design parameters, so that the output data can be extrapolated to provide design specifications for all large mirrors utilizing similar construction concepts.

11.3.2.1.2 Experiment Plan

The experiment (see figure 11.3.2-1) contains a relatively thin (section 11.3.2.3.1.1) spherical beryllium mirror "fixed" at selected points to a stiff lightweight reference structure.

A matrix of mechanical actuators is placed between the mirror back and the reference structure. These change the shape of the front surface by applying forces between the mirror and the reference structure. The actuators

are individually controlled and capable of continuous operation for mirror figure control. The test and evaluation instruments are placed at the center of curvature of the spherical mirror as noted in figure 11.3.2-1.

The most significant test instrument is the scatterplate interferometer. The components necessary for this type interferometer will be fabricated in accordance with the design suggested by the inventor, J. M. Burch, as discussed previously in section 11.3.1.4.1.

A conceptual design of a 1.3 meter diameter spherical mirror is shown in figure 11.3.2-1. This mirror is a scale model of 3 meter and larger telescope designs. The thin mirror design is reviewed in section 11.3.1.4.1.

Since the experiment is to determine the ability to control an optical surface, the shape of the surface should permit ease of manufacture and tests. Therefore, a spherical surface has been selected taking advantage of the condition of unity magnification during mirror figure evaluation, which permits Foucault and interferometer tests to be performed at the center of curvature. This is also realistic because there is little variation between a sphere and a paraboloid, (OTAES, Part I, section 4.10.5.3). Hence, the results of figure tests on a spherical mirror are applicable to a paraboloid and the experimental results can later be applied to the common aspheric surfaces used for telescopes.

The fabrication and testing, on earth, of a thin shell primary mirror will be in a nesting fixture. Essentially, this "nesting technique" is such as to provide uniform mechanical support and thermal continuity throughout the initial manufacture of a thin shell primary, so that it is completely unstrained while in its nest. OTAES Part I, section 4.10.5.4, describes this further.

Beryllium S-200-C was selected as the mirror material for this experiment on the basis of thermal, physical, stiffness-to-mass ratios and manufacturing considerations. Section 11.5.4.5. reviews these considerations. Of special interest is table 11.5.4-3 of this section.

11.3.2.2 Experiment Measurement Objective

This section describes the parameters to be measured and how these measurements relate to the specific technology advances sought.

- a. Measurement of mirror figure as controlled by thermal-mechanical actuators. The resultant technological advances will be:
 - (1) Determination of correctional effects and efficiencies (operational, power and response) of thermo-mechanical actuators applied to large optics in space.
 - (2) Determination of optimum controls to maintain a thin mirror close to diffraction limited performance.

- (3) Determination of maximum manufacturing tolerances allowable for a large correctable mirror with requirements close to diffraction limited performance. The increase in tolerance permitted will significantly reduce mirror figuring time.

- b. Measurement of stability of mirror figure as corrected. The resulting technological advance will be determination of the stability of the correctional elements as a function of orbit, time, power, etc., and the effects upon the actuator matrix selections.

11.3.2.3 Experiment Equipment Design

11.3.2.3.1 Description of Experiment Well

The test well structure and secondary mirror with its supporting structure are similar to those of the thin mirror experiment as discussed in Section 11.3.1.3.1. The difference lies in the primary mirror with its supporting reference structure and thermo-mechanical actuators (Figure 11.3.2-1).

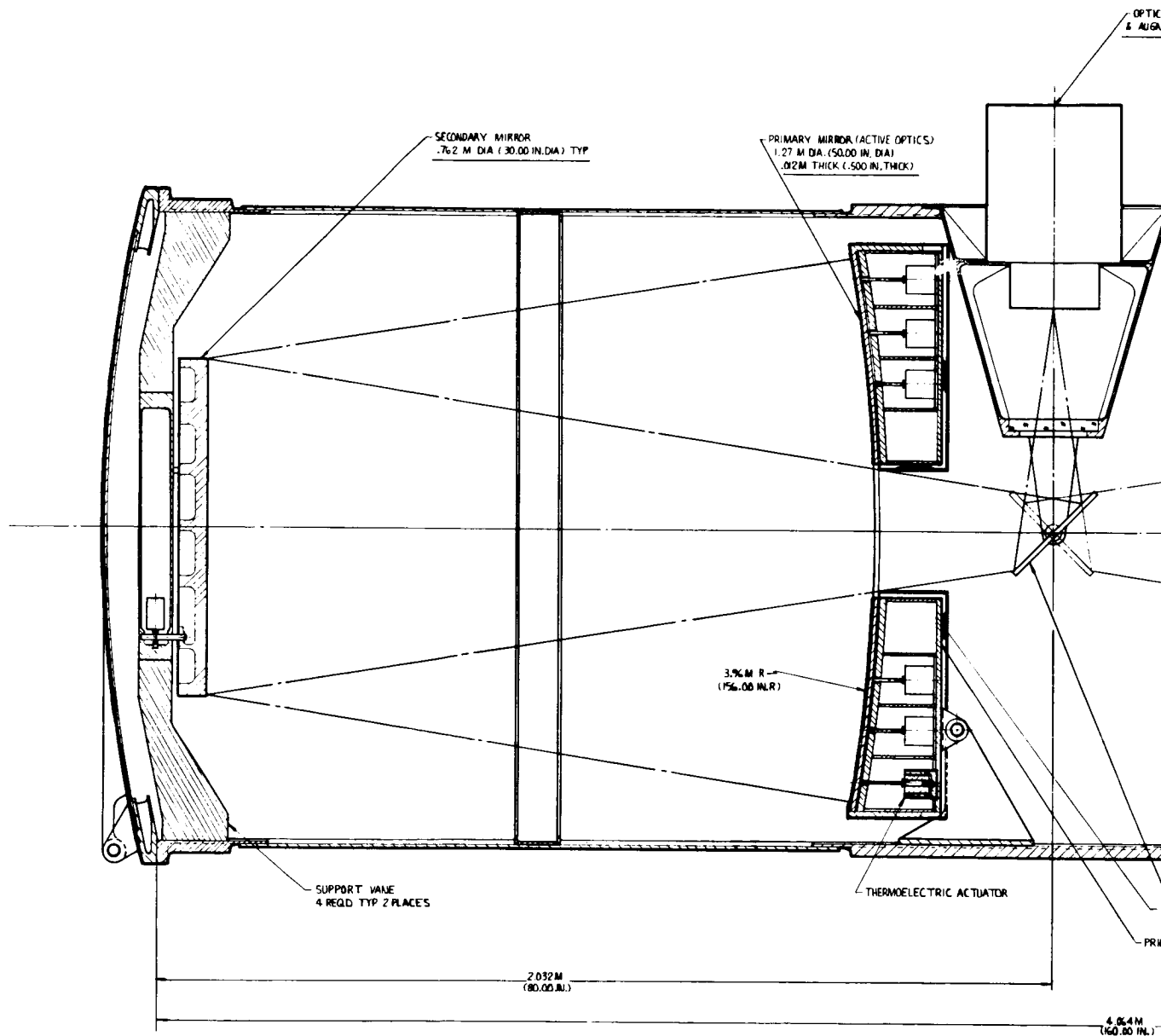
11.3.2.3.1.1 Primary Mirror

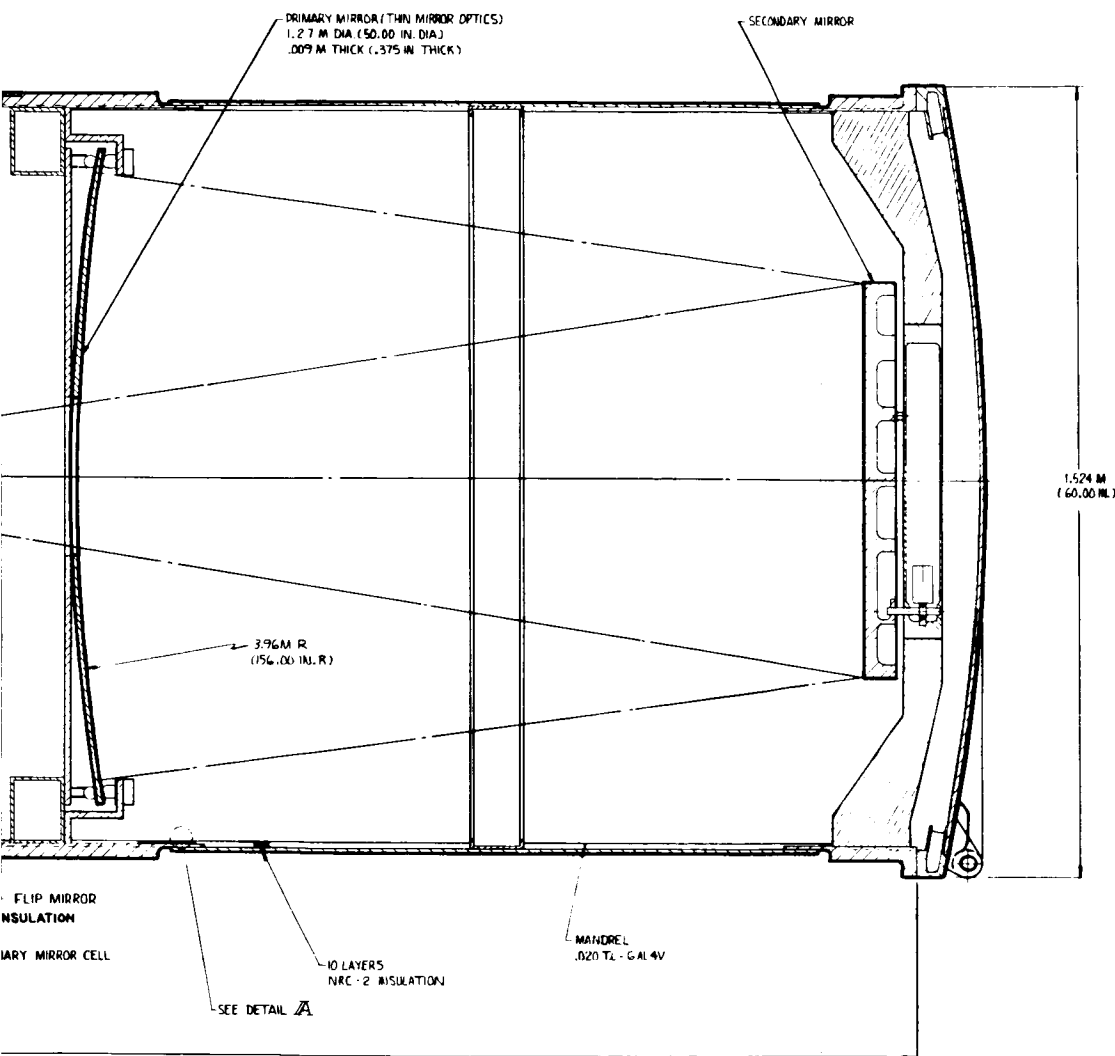
The primary mirror will be a relatively thin (0.5-1.0 inch thick), 50 inch diameter beryllium mirror. The thickness of the mirror will depend primarily on two factors (a) Penetration distance required of the interface connection between the actuators and the rear surface of the primary as illustrated in figure 11.3.2-4. (b) Relation of thickness to force capacity of the thermo-mechanical actuators for correcting mirror figure. Stiffness of the mirror is directly related to thickness, and applied force required is a function of stiffness; that is a thicker mirror will require larger actuator forces to restore a specific figure error pattern. A detailed figure error correction analysis has been performed for a mirror thickness of 0.5 inches and is shown in Section 11.5.7. Both the above mentioned thickness considerations must be studied in greater detail. Additional mirror thickness considerations are discussed in Section 11.3.1.1.3

11.3.2.3.1.2 Reference Structure Design

The reference structure is aluminum welded in an egg-crate type design with 50 millimeter (2-inch) thick plates top and bottom as in figure 11.3.2-2. The top plate of this reference structure will support the actuators which are in contact with the rear surface of the mirror. The top plate curvature will match the rear surface of the mirror. However, during the polishing, figuring, and testing phase, a spherical back plate is used as a nest to uniformly support the mirror on the table of a polishing machine. Circumferential restrainers will be used to keep the mirror in position. After mirror figuring, the reference structure is used for storage and installation into the spacecraft.

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MIRROR FIGURE CALLOUTS UNDERLINED

Figure 11.3.2-1. Primary Mirror Experiment Well-Mirror Figure Test and Correction.

Three mounting pads are located on the base plate of the reference structure. These pads are placed at a .707 radial distance as in figure 11.3.2-2. The top plate contains clamps around its circumference for retaining the mirror rim. It should be noted at this time that the reference structure will be designed to isolate mirror surface changes from the surrounding spacecraft structure.

11.3.2.3.1.3 Thermo-Mechanical Actuator Design Approach

The most important part of the figure control mechanism is the actuators. Figures 11.3.2-3 and 11.3.2-4 illustrate how force will be applied to the rear of the mirror via an actuator. An aluminum tube is shown with a wire heater inside and insulated on the outside by 20 layers of aluminized mylar. All surfaces are in contact before heat is applied to the tube. The top plate described will be stiffer than the mirror and can be considered as the reference plane. The expansion of the thermal-actuator will transmit a force to the rear of the mirror and deflect it in proportion to the spring constants of the two materials and the heat applied.

The mirror is attached to the top plate at its outer edge. Therefore, the force required per centimeter deflection will vary with the change in location of the applied force. The outer edge supports are considered as pinned joints and cannot transmit a moment. A preliminary calculation of forces required shows that a .45 kilogram force will produce an 8.0×10^{-7} cm deformation at a point near the axial region of a test mirror that is simply supported at its outer edges. The required sensitivity for controlling the thermal actuators will be established through analysis and pre-tests. The change in the mirror figure before and after orbit is not expected to exceed 2 fringes (one wavelength) or about 20×10^{-6} inch. Further, for the measuring or correcting to one wavelength, instruments sensitive to $1/2$ wavelength are required. The surface tolerance is $1/32$ wavelength and instruments sensitive to $1/60$ wavelength are needed. These requirements are within the practical limits of optical measuring instruments.

The coefficient of thermal expansion of the aluminum in the actuator is linear within the anticipated temperature range of actuator operation. The aluminum in the actuator is equivalent to a 1 inch rod which will elongate 12×10^{-6} inches when its temperature is raised 1.0 degree F. A temperature control to 0.1 F will be required to obtain elongations within the optical tolerances previously specified.

11.3.2.3.2 Description of Supporting Instrumentation (Refer to section 11.3.1.3.2)

11.3.2.4 Data Management

The primary data from this experiment will concern measurement of mirror figure as controlled by thermo-mechanical actuators.

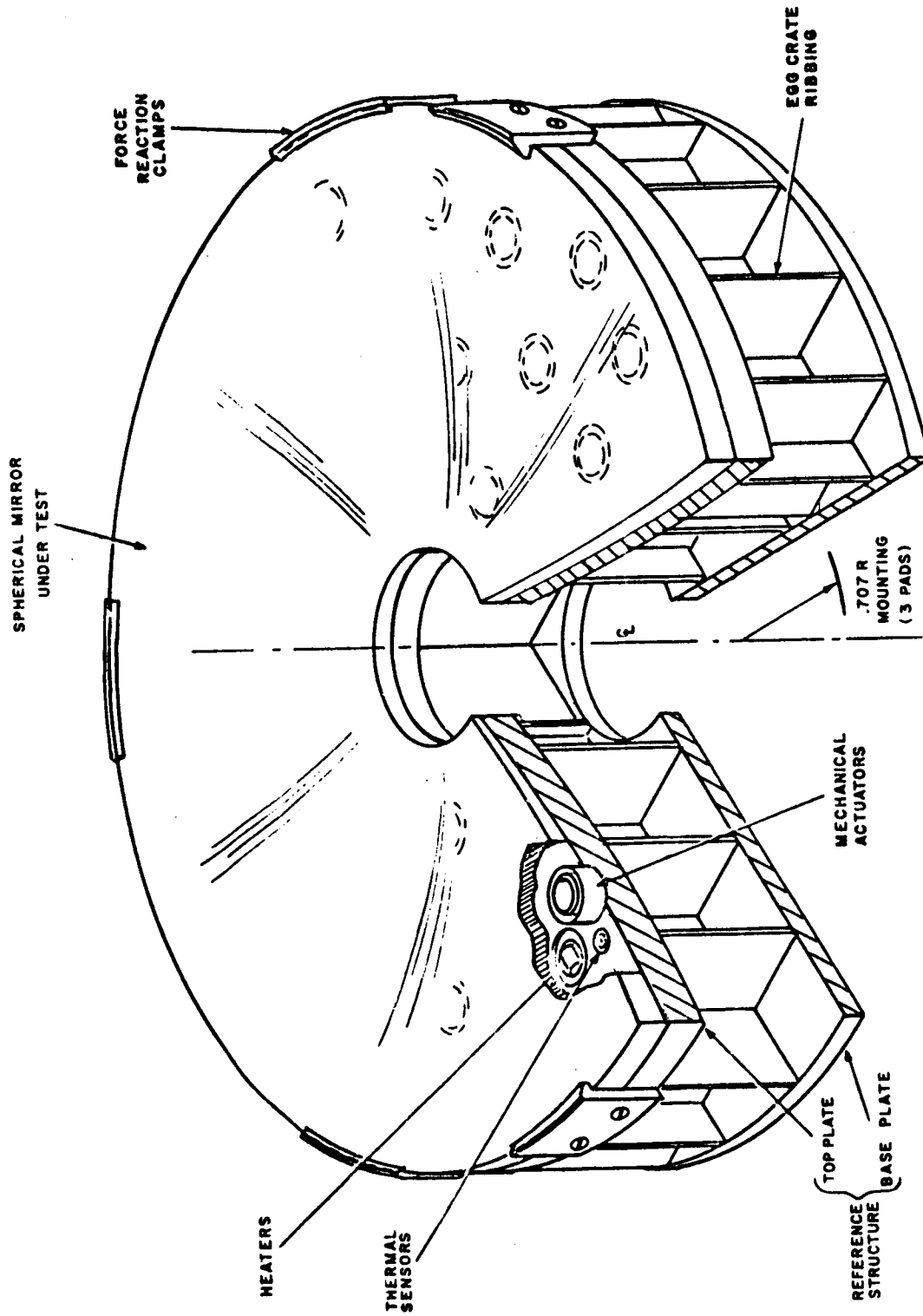


Figure 11.3.2-2 Test Mirror Mechanization

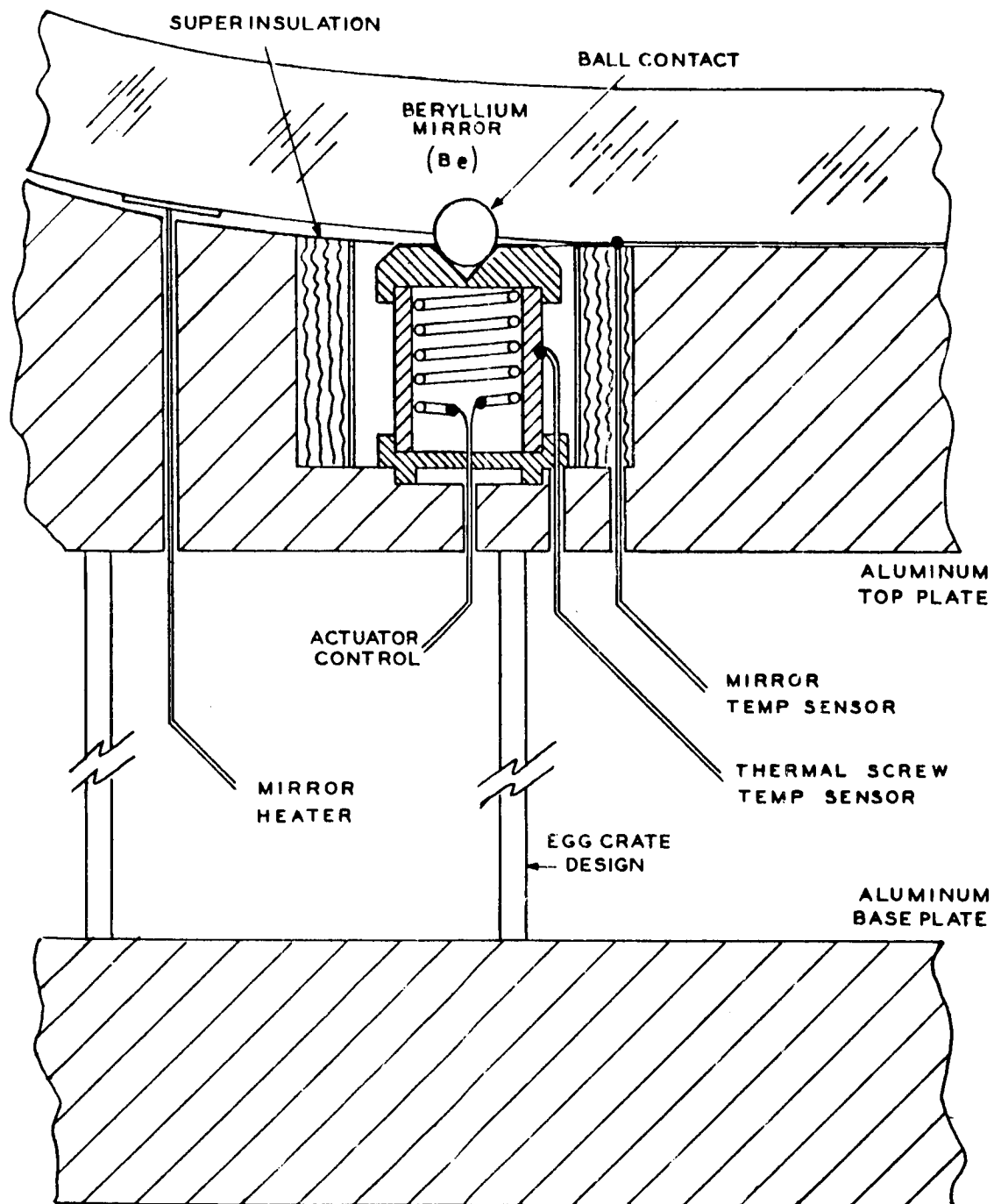


Figure 11.3.2-3 Unidirectional Actuator Schematic

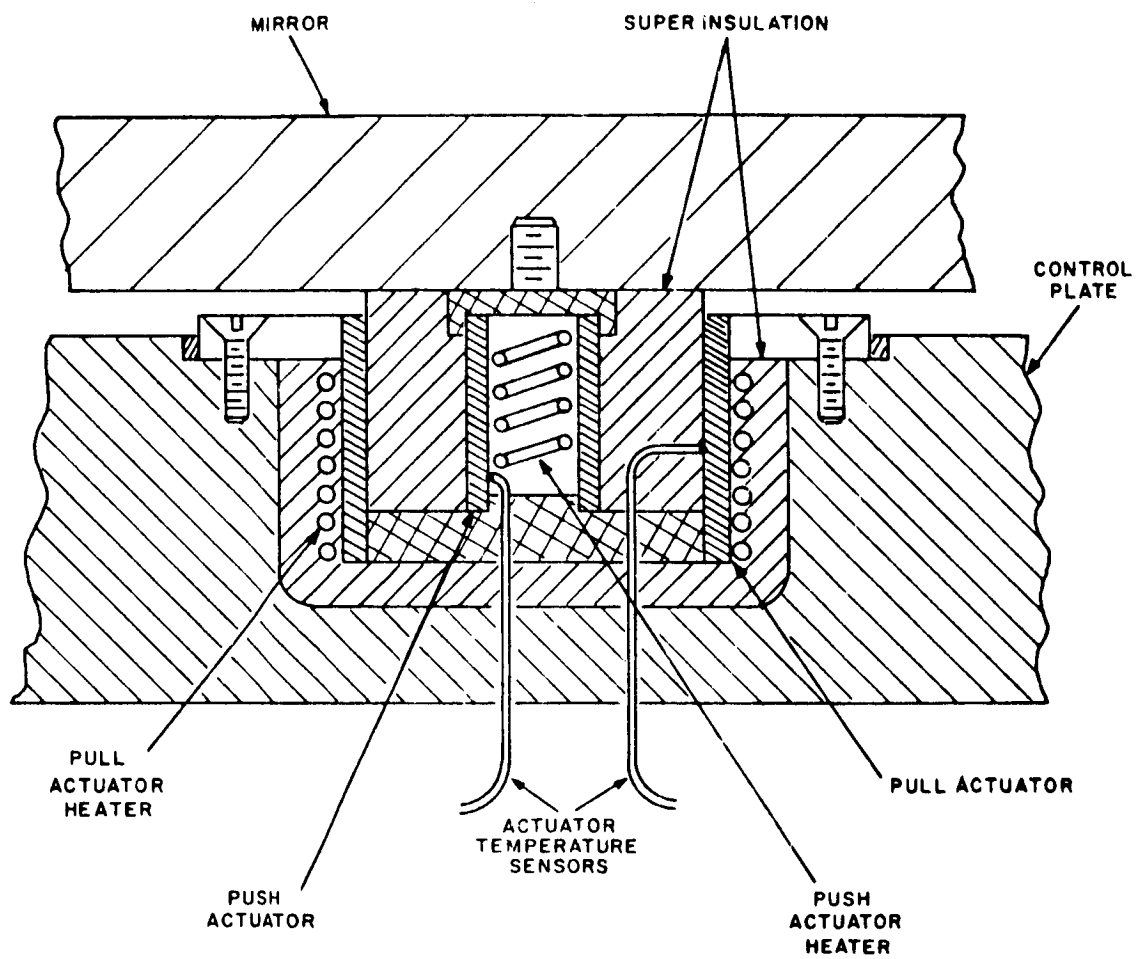


Figure 11.3.2-4 Bidirectional Actuator Schematic

A block diagram outlining the major elements of mirror control and test is shown in figure 11.3.2-5. Two modes of mirror control are presented. These are manual mode and automatic mode. The mirror figure is tested with scatter plate interferometer that provides a fringe pattern readout as described in section 11.3.1.4.1. This fringe pattern is compared with a reference fringe pattern. In the automatic mode, the fringe pattern is converted to digital information by the scanner mechanism. This data is sent via telemetry to a ground based general purpose computer. A comparator compares both patterns and develops digital control signals which are transmitted to the spacecraft. These signals are used to operate the mechanical actuators which apply forces to the rear surface of the mirror to alter the figure. The temperature sensing section of the diagram depicts a secondary feedback loop for maintaining the mirror isothermal (at uniform temperature). The mechanics of evaluating mirror figure in conjunction with the ground station is discussed in section 11.3.1.4.

The mirror surface will be monitored by thermal sensors to determine the temperature distribution. Each of the sensors will have a readout on a temperature meter. Isothermal control will be provided by actuating a predetermined number of heaters in the array, which are monitored by an astronaut within the cabin.

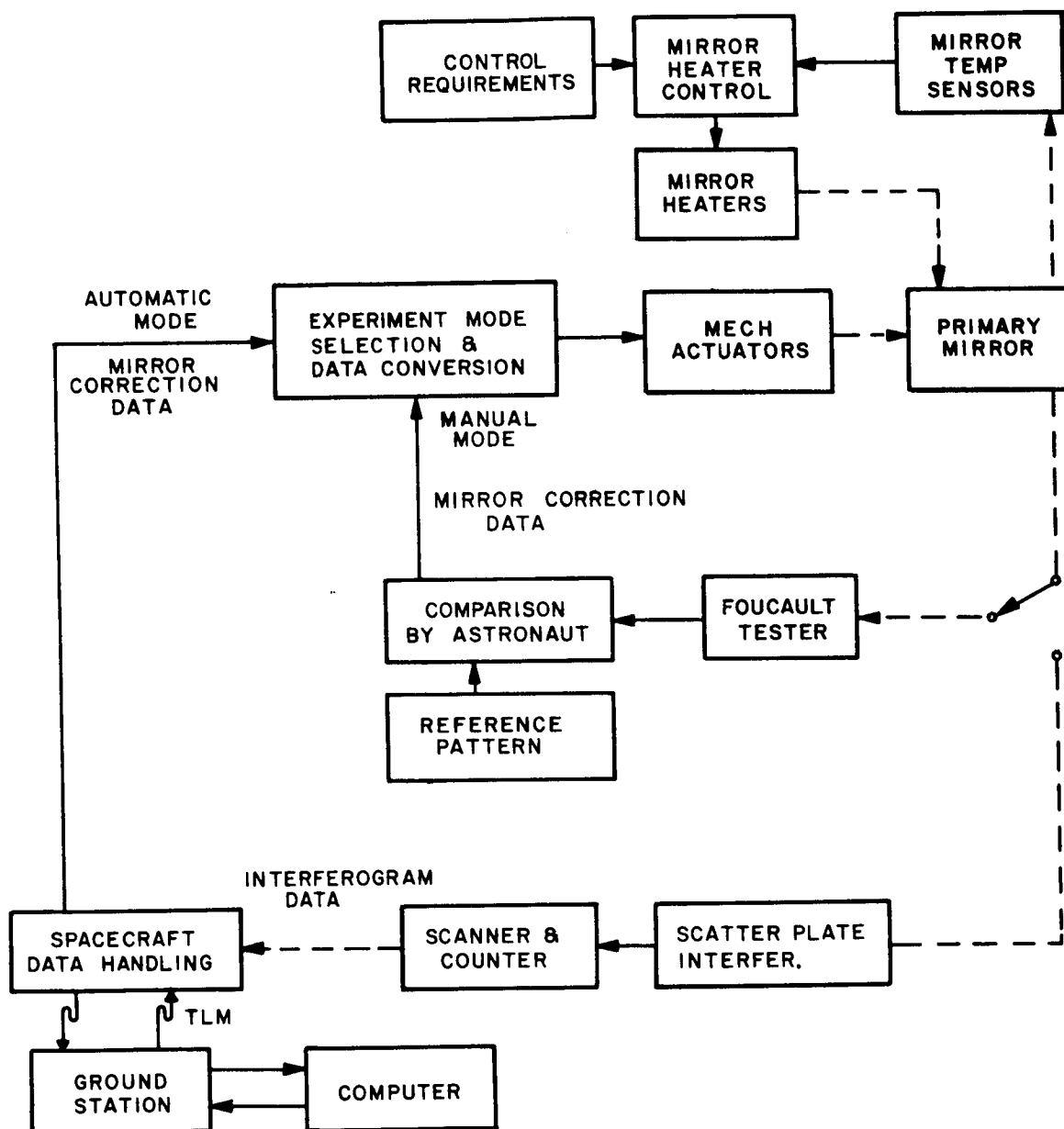


Figure 11.3.2-5. Design Implementation

10 min. d. Eyepiece is removed and stowed.

11.4.1.4 Center of Curvature Determination (Foucault Test)
(See 11.5.8.2)

- 10 min. a. Remove foucault tester from stowage and position on the autocollimator mount at the established image plane.
- 1 min. b. Switch on the light source located within the tester and illuminate the surface of the primary mirror.
- 60 min. c. Sight through the tester at a point just behind the knife edge and reposition the knife edge axially using the adjustment knob on the tester until the entire mirror surface appears to darken uniformly. This position of the knife edge is the center of curvature of the primary mirror.
- 10 min. d. Foucault Tester is removed and stowed.

11.4.1.5 Installation of Scatterplate Interferometer Module

- 10 min. a. Remove the scatterplate interferometer module from stowage and install in same mount as foucault tester.
- 10 min. b. Position the interferometer scatterplate at the center of curvature established by task 11.4.1.4(C).

11.4.1.6 Thin Mirror Figure Test (See section 11.5.8.3)

30 min.

inc. warm-up

- a. Switch on the interferometer laser source.
- 30 min. b. Verify the center of curvature by noting that no return image appears through the scatterplate (assuming the scatterplate is initially perpendicular to the optical axis). If the scatterplate is not at the center of curvature, a set of concentric interference rings will be observed.
- 30 min. c. Position the scatterplate to allow a mirror figure measurement. Move it slightly off axis (one mil) and slight tilt it through an angle depending on the curvature of the mirror using the appropriate adjustment knobs on the module. The image now seen will be a series of wavy vertical interference lines.
- 30 min. d. Attach vidicon T.V. camera to interferometer.
- 30 min. e. Switch on interferogram T.V. monitor in cabin; compare resolution of interferogram with that presented by monitor. For figure evaluation the vertical interference lines are compared to a reference interferogram.

- 15 min. f. Switch on spacecraft to ground telemetry and send interferogram to ground station via vidicon.

(End of Astronaut Procedure)

- g. The interferogram will be interpreted by ground station and figure changes will be compared by computer with a perfect reference mirror. Areas of the primary which are out of figure tolerance (greater than $\lambda/30$ distortion) will be evaluated to determine residual strains, so that a maximum allowable stress (P.E.L.) can be established to design a mirror that will retain its figure in orbit.

11.4.2 Primary Mirror Figure Test and Correction Procedure

The following tasks are performed by an astronaut within access compartment.

11.4.2.1 Alignment of Optics (See section 11.5.8.1)

- 10 min. a. Remove and stow scatterplate interferometer used in thin mirror experiment.
- 10 min. b. Remove the autocollimator from stowage and secure it in position on the mount provided.
- 10 min. c. Switch the autocollimator on and sight onto the central flat and crosshairs of the primary mirror.
- 1 min. d. Switch on the power to secondary mirror fine adjustment mechanism drive motor.
- 60 min. e. Adjust the secondary mirror tilt by manipulating control knob, adjust autocollimator reticle moving lens until crosshair at the center of the primary aligns with autocollimator reticle. The image presented to the astronaut will be inverted in relation to the actual crosshairs. With the crosshair and reticle superposed, axial alignment between the primary mirror and secondary mirror flat will have been established.
- 10 min. f. Detach and stow autocollimator.

11.4.2.2 Image Plane Check (Focus Adjustment)

- 10 min. a. Remove eyepiece from stowage and position in the autocollimator mount near calculated focal plane.
- 1 min. b. Switch on the point light source, positioned in the image plane.
- 30 min. c. Sight through the eyepiece and reposition secondary for maximum target definition (best focus). Since the image will be inverted and reverted, any off-axis translations will also be

reversed; i.e., left will be right and top will be bottom.

10 min. d. Remove eyepiece and stow.

11.4.2.3 Center of Curvature Determination (Foucault Test)
(See section 11.5.8.2)

10 min. a. Remove the Foucault tester from stowage and position on the autocollimator mount at the established image plane.

1 min. b. Switch on the light source located within the tester and illuminate the surface of the primary mirror.

60 min. c. Sight through the tester at a point just behind the knife edge and reposition the knife edge axially, using a fine adjustment knob on the tester until the entire mirror surface appears to darken uniformly. This position of the knife edge is the center of curvature of the primary mirror.

10 min. d. Remove the Foucault tester and stow.

11.4.2.4 Installation of Scatterplate Interferometer Module

10 min. a. Remove the scatterplate interferometer module from stowage and install in same mount as Foucault tester.

10 min. b. Position the interferometer scatterplate at the center of curvature established by task 11.4.2.3(C).

30 min.-incl.
warm-up c. Pre-test check and alignment

(1) Switch on the Laser source.

(2) Verify the center of curvature by noting that no return image appears through the scatterplate (assume the scatterplate is initially perpendicular to the optical axis). If the scatterplate is not at the center of curvature a set of concentric interference rings will be observed.

30 min. (3) Position the scatterplate to allow a mirror figure measurement. Move it slightly off axis (one mil) and slightly tilt it through an angle depending on the curvature of the mirror using the appropriate adjustment knobs on the mount. The image now seen will be a series of wavy vertical interference lines.

- 30 min. (4) Attach vidicon T.V. camera to interferometer.
- 30 min. (5) Switch on interferogram T.V. monitor in cabin; compare resolution of interferogram with that presented by monitor. For figure evaluation the vertical interference lines are compared to a reference interferogram.
- 15 min. (6) Switch on spacecraft to ground telemetry and send interferograms to ground station.

11.4.2.5 Figure Test and Correction Procedure (See section 11.5.8.3)

- 15 min. a. The Interferogram is forwarded to ground station as per ground time step 11.4.2.4(C) and is compared to reference pattern to obtain mirror figure change from earth to space environment. The results of the ground based computer analysis, indicating deviations from desired figure, are telemetered back to the spacecraft; the figure correction procedure follows:
 - b. Obtain Isothermal Mirror Condition
- 10 min. (1) Switch on primary mirror thermal sensor monitor and note temperature of mirror based upon the zonal distribution of sensors.
- 90 min. (2) Record temperature distribution and telemeter to ground based computer to determine the required input to the mirror heaters to achieve an isothermal surface.
- 240 min. (3) Apply required heat using thumb wheels corresponding to particular zones.
- 10 min. (4) Repeat step b-1.
- 10 min. to 330 min. (5) If Isothermal condition has been achieved to within specified accuracy, go to step (6); otherwise repeat step (2) through step (5).
- 10 min. (6) Recheck curvature by repeating step 11.4.2.4(C-2).
- 15 min. (7) Obtain Interferogram of Isothermal mirror and telemeter to ground based computer for comparison with reference pattern.
- indeterminate (8) If figure is still out of tolerance; continue, otherwise correction has been achieved, indicating the residual distortions (plastic strains) of the mirror surface are within the allowable figure tolerance ($\lambda / 30$).

c. Apply Thermo-Mechanical Actuators

- | | |
|---------|--|
| 10 min. | (1) Switch on actuators thermal sensors monitor. |
| 10 min. | (2) Switch on mirror thermal sensors monitor to note any changes in mirror temperature due to leakage from actuator heaters. |
| 30 min. | (3) Apply heat using thumb wheels, proportional to actuator array recommended by ground based computer based upon interferogram of step b-7. |
| | (4) Note figure changes by observing interferogram. |
| 15 min. | (5) Telemeter interferogram for re-evaluation of quantity and array of actuators and the forces required if figure tolerance has not been achieved; if figure tolerance has been achieved, terminate test. |

11.5 SUPPORTING ANALYSIS

11.5.1 Zero "G" Ground Test Simulation Using Water and Mercury Flotation

Nomenclature

γ_f = fluid density - lb/in³

ρ_m = mirror density - lb/in³

F_B = bouyant force

R = radius of curvature of spherical mirror

a = radius of mirror

b = radius of submerged part of spherical surface

S = sagitta of spherical mirror = height of spherical segment

h = sagitta of submerged part of spherical surface

W_M = weight of mirror

V_{seg} = volume of spherical segment - in³

t = thickness of mirror

A_{seg} = area of curved surface of spherical segment

$(S.G.)_M$ = specific gravity of mirror

$(S.G.)_F$ = specific gravity of fluid

ω = uniform load = lb/in²

$m = \frac{1}{\nu}$

ν = poisson's ratio

E = elastic modulus

y_{max} = tip deflection of mirror overhang

Figure 11.5.1-1 shows the spherical mirror floating on a fluid with density γ_f . The spherical mirror is buoyed up by the fluid with a force F_B equal

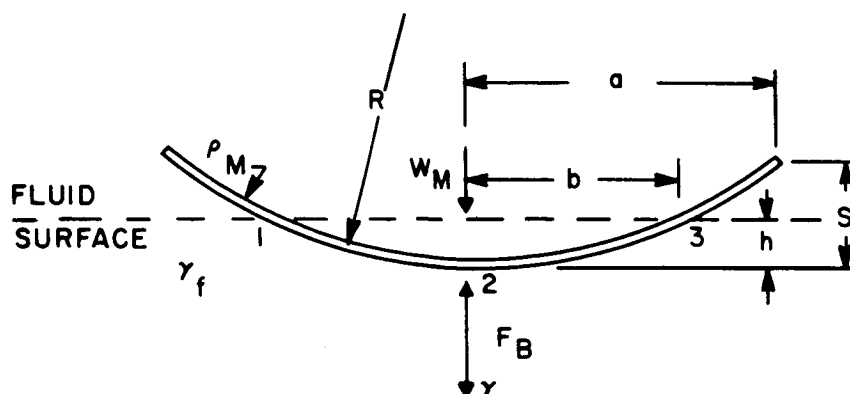


Figure 11.5.1-1 Fluid Flotation Schematic

to the weight of the mirror W_M . The buoyant force F_B is equal to the weight of an imaginary volume of fluid enclosed by the submerged part of the spherical surface. This corresponds to the weight of fluid in the spherical segment 1-2-3. The radius of curvature of the mirror is R and the mirror is assumed to submerge a distance h ; therefore the buoyant force is

$$F_B = \gamma_f V_{\text{seg}} = \gamma_f \frac{\pi}{3} h^2 [3R - h] \quad (1)$$

The radius of curvature is assumed to be much greater than the submerged distance h ; therefore, $3R \gg h$ and

$$F_B = \gamma_f \pi h^2 R \quad (2)$$

The weight of the spherical mirror of thickness t , and density ρ_M , is given by the expression

$$W_M = \rho_M t A_{\text{seg}} = \rho_M t (2\pi RS) \quad (3)$$

Using Figure 11.5.1-2, where a is the radius of the spherical segment, the equation for the sagitta S is obtained as

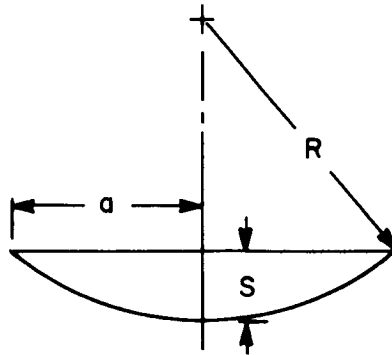


Figure 11.5.1-2 Spherical Mirror Geometry

$$R^2 = (R - S)^2 + a^2$$

or

$$a^2 = 2 RS - S^2$$

For chords that are short in comparison with the radius (R), S^2 will be negligibly small with respect to $2 RS$ and the length of the sagitta becomes

$$S = \frac{a^2}{2R} \quad (4)$$

Substituting the result of Eq. (4) into (3) for S , the weight of the mirror becomes

$$W_M = \rho_M t \pi a^2 \quad (5)$$

Since $F_B = W_M$, equating (2) and (5)

$$\gamma_f \pi h^2 R = \rho_M t \pi a^2$$

or

$$h = \sqrt{\frac{\rho_M}{\gamma_f} \cdot \frac{a^2}{R}} \cdot t \quad (6)$$

The submerged distance h is the sagitta of the submerged spherical segment. Substituting h for S , and b for a in Eq. (4), the radius b of the submerged spherical segment becomes

$$b = \sqrt{2Rh} \quad (7)$$

Two fluids were considered as flotation media, water and mercury.

The numerical data for the spherical mirror are

$$a = 25 \text{ inches}; \quad R = 125 \text{ inches}; \quad t = 0.38 \text{ inch}$$

Substituting these values into Equations (6) and (7) and noting that the density ratio $\frac{\rho_M}{\rho_f}$ is equal to the ratio of specific gravities $\frac{(S.G.)_M}{(S.G.)_f}$

$$h = 1.38 \sqrt{\frac{(S.G.)_M}{(S.G.)_f}} \quad (6)'$$

$$b = 15.8 \sqrt{h} \quad (7)'$$

The results using (6)' and (7)' are tabulated below; the specific gravity of the mirror material, beryllium, is 1.85.

Flotation Medium (Fluid)	$(S.G.)_M$	$(S.G.)_f$	h (inches)	b (inches)	Overhang $a - b$ (inches)
Water	1.85	1.0	1.875	21.65	3.35
Mercury	1.85	13.6	0.51	11.3	13.7

Due to the dished shape of the spherical mirror, the unsubmerged part of the mirror (see $a - b$ in Table) will be unsupported and thus will be free to deflect under its own weight. This condition can be idealized, conservatively, as an overhung plate fixed circumferentially at a radius b from the center, as shown in figure 11.5.1-3.

The expression for maximum deflection at outer radius (a) is given in Roark, "Formulas For Stress and Strain" (pg. 200, Case 21); this expression has been rearranged into the following form:

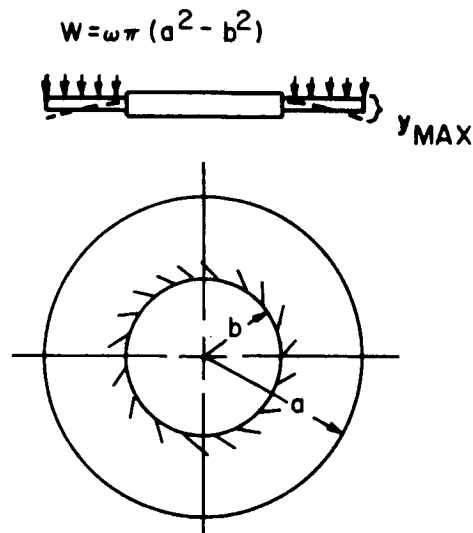


Figure 11.5.1-3 Idealized Model of Mirror Overhang

$$y_{\max} = \frac{3 \omega (m^2 - 1) a^4}{16 m^2 E t^3} \left[\frac{(7m+3) + \left(\frac{b}{a}\right)^6 (m-1) - \left(\frac{b}{a}\right)^2 (m+7) - \left(\frac{b}{a}\right)^4 (7m-5)}{(m+1) + \left(\frac{b}{a}\right)^2 (m-1)} - \frac{4 \left(\frac{b}{a}\right)^2 \left[(5m-1) + \left(\frac{b}{a}\right)^2 (m+1) \right] \log \frac{a}{b} - 16 \left(\frac{b}{a}\right)^2 (m+1) \left(\log \frac{a}{b} \right)^2}{(m+1) + \left(\frac{b}{a}\right)^2 (m-1)} \right] \quad (8)$$

The numerical quantities associated with the beryllium mirror are as follows:

$$a = 25 \text{ inches}; \quad m = 50; \quad E = 42 \times 10^6 \text{ psi}; \quad t = 0.38 \text{ inch};$$

$$\rho = 0.066 \text{ lb/in}^3; \quad \omega = \rho t = 0.0251 \text{ lb/in}^2$$

Substitution of the numerical parameters reduces the deflection equation to

$$y_{\max} = 798 \times 10^{-6} \left[f \left(\frac{b}{a} \right) \right] \quad (9)$$

where $f\left(\frac{b}{a}\right)$ is the bracketed expression of Equation (8)

Using water as the flotation medium, $b \doteq 21.65$ inches (from the table), so that

$$\frac{b}{a} = \frac{21.65}{25.0} = 0.867; \quad f\left(\frac{b}{a}\right) = 0.93$$

$$(y_{\max})_{\text{water}} = 742 \times 10^{-6} \text{ inches}$$

Using mercury as the flotation medium, $b \doteq 11.3$ inches, and

$$\frac{b}{a} = \frac{11.3}{25.0} = 0.451; \quad f\left(\frac{b}{a}\right) = 3.86$$

$$(y_{\max})_{\text{mercury}} = 3080 \times 10^{-6} \text{ inches}$$

Requirements of mirror figure measurements (diffraction limited performance) are that deviations from the spherical pattern should not exceed approximately $\frac{\lambda}{30} = \frac{24 \times 10^{-6}}{3.0} = 0.8 \times 10^{-6}$ inch. The deflections computed above will

result in deviations from the spherical pattern. Comparing these values (742×10^{-6} inches and 3080×10^{-6} inches) to the allowable figure error 0.8×10^{-6} inches, indicates that substantial figure error will result using the flotation media discussed above.

11.5.2 Zero "G" Ground Simulation Using No Overhang Flotation

To alleviate the overhang condition resulting from flotation of a spherical thin mirror on water or mercury as discussed in section 11.5.1, the fluid density required to just support the entire rear surface of the mirror is determined. To prevent instability, the periphery is retained by three 10 gm spring loaded limit stops, (120° apart) as shown in Figure 11.5.2-1

The total bouyant force F_B is equal to the weight of the mirror W_M and the force applied by the limit stops $F_{L.S.}$.

$$F_B = W_M + F_{L.S.} \quad (1)$$

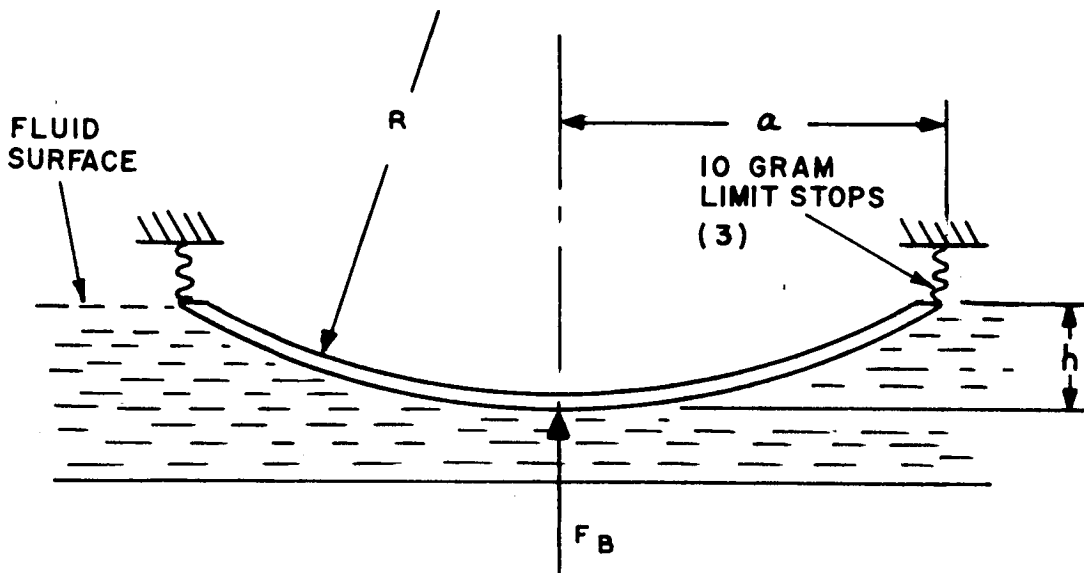


Figure 11.5.2-1 No Overhang Flotation Schematic

But F_B is equal to the weight of the volume of fluid enclosed by the submerged part of the spherical surface; or from Eq. (2), section 11.5.1

$$F_B = \gamma_f \pi h^2 R \quad (2)$$

where:

γ_f = density of fluid

h = submerged distance - sagitta

R = radius of curvature of sphere

From Equation (5) section 11.5.1

$$W_M = \rho_m t \pi a^2 \quad (3)$$

where:

ρ_m = density of mirror

t = thickness of mirror

a = radius of mirror

Substituting Equations (2) and (3) into Eq. (1), and solving for γ_f

$$\gamma_f = \frac{\rho_M t a^2}{h^2 R} + \frac{F_{L.S.}}{\pi h^2 R} \quad (4)$$

Substituting the numerical quantities,

$$\begin{aligned} a &= 25 \text{ in}; & R &= 125 \text{ in}; & t &= 0.38 \text{ in}; & h &= \frac{a^2}{2R} = 2.5 \text{ in} \\ \rho_M &= 0.066 \text{ lb/in}^3; & F_{L.S.} &= 3 \times 10 \text{ gm} = 0.066 \text{ lbs}, \\ \gamma_f &= 0.0202 + 0.027 \times 10^{-3} \text{ lb/in}^3 \end{aligned} \quad (5)$$

or

$$\gamma_f = 0.55 + 0.75 \times 10^{-3} \text{ gm/cm}^3$$

The extreme precision in fluid density required to maintain stability makes this technique an impractical method of zero "G" ground simulation.

11.5.3 Zero "G" Simulation by Pneumatic Flotation

In this technique the mirror is floated on a cushion of air similar to the method shown in the accompanying figure. By this method the mirror is free from deformations due to the effects of gravity; however, due to the spring loaded limit stops and the difficulty of maintaining the pressure precisely to balance the mirror weight, deformations of the mirror may result. Even if the resulting deformations can be controlled to within tolerable limits, the necessary tolerances on pressure are extremely small. The following analysis shows quantitatively the results discussed above.

The mirror is assumed to be restrained at three locations (120° apart) by spring loaded limit stops as shown in figure 11.5.3-1. Each spring applies a load of $F_s = 10 \text{ gm} = 0.022 \text{ pounds}$ to restrain the mirror on the kinematic mounts.

If the air pressure is just enough to balance the mirror weight the static (1 "G") deflection due to its own weight will be eliminated; assuming for this preliminary analysis the mirror can be represented by a flat circular plate, the balancing pressure must be

$$P = \rho t = (0.066 \text{ lb/in}^3)(0.38 \text{ in}) = 0.0251 \text{ lb/in}^2$$

where:

ρ = density of beryllium

t = thickness of mirror

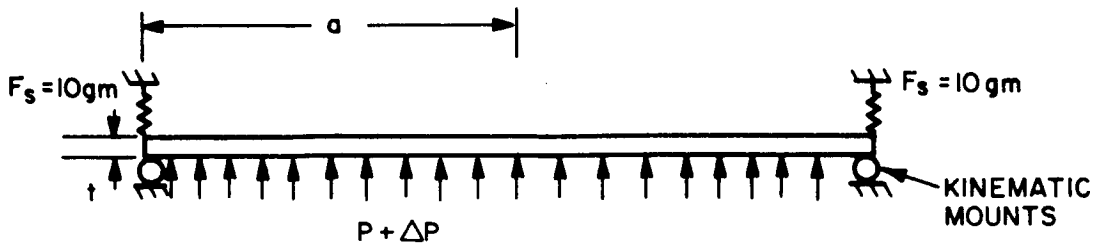


Figure 11.5.3-1 Pneumatic Flotation Schematic

The positive tolerance on this pressure would be the differential pressure that just causes the mirror to lift off the kinematic mounts, since past this point the mirror will be unstable. This maximum differential pressure corresponds to the total limit stop spring load $3 (10 \text{ gm}) = 0.066 \text{ lbs}$, that is,

$$\Delta p = \frac{0.066 \text{ lbs}}{\pi a^2} = \frac{0.066}{\pi (25)^2} = 3.37 \times 10^{-5} \text{ lb/in}^2$$

where:

a = radius of the plate

The maximum deflection at the center of a uniformly loaded circular plate on three supports (located 120° apart), is given approximately in Timoshenko's, "Plates and Shells", pg. 271 by the expression:

$$y_{\max} = \frac{0.307 W a^2}{E t^3}$$

where:

W = total uniform load = $\Delta p \pi a^2 = 30 \text{ gms} = 0.066 \text{ lbs}$

E = modulus of elasticity = $42 \times 10^6 \text{ lb/in}^2$.

Substituting the numerical values, the maximum deflection caused by the maximum differential pressure Δp

$$y_{\max} = \frac{0.307 (0.066) (25)^2}{(42 \times 10^6) (0.38)^3} = 5.41 \times 10^{-6} \text{ in.}$$

Summary of Results

The pressure to float the mirror must be

$$\begin{aligned} p &= 0.0251 + 3.37 \times 10^{-5} \text{ lbs/in}^2 \\ &= 0.695 + 9.33 \times 10^{-4} \text{ inches of water} \\ &= 17.65 + 0.0237 \text{ mm of water} \end{aligned}$$

There is a degree of uncertainty in controlling the pressure to within $\Delta p = 9.33 \times 10^{-4}$ inches of water and hence the possibility exists of lifting the mirror off its kinematic mounts, resulting in instability.

The maximum deflection at the center of the plate due to the differential pressure Δp is

$$y_{\max} = 5.41 \times 10^{-6} \text{ inches,}$$

which is greater than the maximum allow deflection of 0.8×10^{-6} inches required for diffraction limited performance. This value can be reduced by decreasing the value of Δp , but it has already been concluded that the above value of Δp will be difficult to control.

The above results indicate that the pneumatic flotation technique is an uncertain method for equivalent zero "G" testing for mirror figure.

11.5.4 Materials Analysis

11.5.4.1 Minimum Thickness of Beryllium Mirror

From a qualitative standpoint, the minimum thickness has been established as 0.2 inch based upon practical considerations. Specifically, experience with beryllium, especially with respect to its brittleness and notch sensitivity, leads to the view that machining and handling of this material in mirror blanks having a thickness less than 0.2 inch presents serious practical problems.

Information from manufacturers of beryllium (Reference 1) also, indicates that machining beryllium in blanks of less than 0.125 inch thickness may yield undesirable properties in the finished mirror surfaces. In the process of machining the thin blank "bowing" distortion (local buckling) of the surface may occur. When the resulting internal stresses are relieved by heat treatment the surface may not return to the desired prefinished shape.

The brittleness (lack of yield or plastic flow region in stress-strain curve) of beryllium becomes accentuated when large diameter mirrors with less than 0.125 inch thickness are machined. There is a greater chance of cracking the specimen, as the ultimate stress in flexure is more easily attained through bending deformations due to impact. This results in an increased scrap rate during both the machining process and during handling.

11.5.4.2 Weight Saving Considerations

The launch weight of a large optical system for astronomical purposes is a function of the design approach. For example, compare the launch weight of a rigid design capable of withstanding the rigors of launch and one "G" environment, to that of a thin mirror designed for a zero "G" environment. Full cognizance can be taken of the fact that the equipment need be aligned and operating only when weightless or during orbital flight.

a. One "G" Design

A weight estimate for a rigid system may be realistically approached by starting with values derived from the QAO Goddard Experiment Package modified to achieve diffraction limited performance. For comparison purposes a rigid quartz primary mirror optical system is also considered. Table 11.5.4-1 shows the resulting weights for a 38-inch diameter telescope and values corresponding to 50- and 120-inch optics having the same general configuration.

The estimates shown in Table 11.5.4-1 assume the following conditions:

-
- (1) a. Communication with Dr. Denny, Beryllium Corporation, December 21, 1966.
 - b. Communication with H. Boehling, Brush Beryllium Company, December 22, 1966.

- (1) The imaging system incorporating such optics must survive the stresses of launch without losing its surface figure or coarse alignment.
- (2) Camera weight is excluded.
- (3) Extrapolation to the larger (50 and 120 inch) telescopes is made in accordance with specific rules for weight growth, designated at the bottom of Table 11.5.4-1, and discussed at the end of this section.

b. Zero "G" Design

A weight saving is to be expected from a redesign of the optical elements for zero g application, specifically a reduction in the mirror thickness and supporting structure. This weight advantage may be most directly appreciated by comparison with a rigid, preassembled system designed to similar performance requirements. The last two columns of table 11.5.4-2 shows these savings, when compared to rigid designs employing both beryllium and quartz primary mirrors. Weights of the lightened primary mirrors are based on the thickness to diameter ratios obtained in OTAES Part I, section 4.10.5.2; i.e., 50 inch diameter, 0.375 inch thick; -120 inch diameter, 2.13 inches thick.

The governing parameter used in establishing the weight growth equations for the optical elements is the Rayleigh criterion that "the optical path length over the entire aperture of the element must not vary more than a quarter wavelength". For a specific optical surface this imposes the restriction that any deflection caused by the element's own weight and other earth environment forces should not exceed a small fraction of the wavelength. Thus, the mirror must be designed to have the same deflection regardless of diameter. Applying this constraint to the deflection equation results in a weight growth as the fourth power of the diameter.

A different criterion has been applied to the structure supporting the optical elements; namely, that the shell buckling stress will remain constant as the size increases. Using this premise, the weight of the structure increases as the cube of the diameter.

Relative to the weight of an element and the material of which it is made, the weight varies with the three-halves power of the density and inversely as the one-half power of the modulus of elasticity. For beryllium and quartz, the materials which are of interest in construction of secondary mirrors, a beryllium secondary would weigh approximately one-third as much as a quartz secondary of equal stiffness and size.

An examination of table 11.5.4-2 emphasizes the large influence of the optical elements on the weight of the telescope portion of an imaging system, and upon the system itself.

TABLE 11.5.4-1

WEIGHT SUMMARY OF RIGID (PREASSEMBLED SPACEBORNE IMAGING SYSTEMS

TELESCOPE WEIGHTS FOR GROUND ASSEMBLY AND ALIGNMENT
MINIMAL ADJUSTMENT IN ORBIT

Primary Dia (Inches)	Primary Weight Beryllium (pounds)	Primary Weight- Quartz egg-crate (pounds)	Secondary Weight Beryllium (pounds)	Secondary Weight Quartz (pounds)	Structure and Mech. Weight (pounds)	Total Weight Beryllium primary	Total Weight Quartz primary
38*	290	-****	-	15	370	675	-
38	-	200	-	15	255	-	470
50**	850	-	-	50	800	1700	-
50**	850	-	15	-	800	1665	-
50**	-	610	-	50	580	-	1240
120**	27000	-	-	1500	11800	40300	-
120**	27000	-	500	-	11800	39300	-
120**	-	20200	-	1500	8000	-	29700
BASIS OF SCALING	4 (Dia)	4 (Diam)	1/3 QUARTZ	4 (Diam)	3 (Dia)		

Goddard Experiment Package
(Present 130
hardware)

330

NOTE:

- * Numerical values in this row obtained by modifying present GEP (last row) to achieve Diffraction Limited performance.
- ** Numerical values obtained by scaling numbers in first row in accordance with BASIS OF SCALING row.

*** All telescopes F/5

**** Based upon OAO primary - Princeton experiment; 32 inches diameter, with weight of 100 lbs.; values based on (Diam)⁴ scaling.

TABLE 11.5.4-2
WEIGHT SUMMARY OF SPACE ASSEMBLED IMAGING SYSTEM

THIN PRIMARY AND LIGHTWEIGHT STRUCTURE

Primary Dia (Inches)	Primary Weight (Pounds)	Secondary Weight (Pounds)	Structure Weight (Pounds)	Alignment Equipment (Pounds)	Approximate Stowage Equipment (Pounds)	Total Weight (Pounds)	Saving From 1 "g" Design Beryllium *	Saving From 1 "g" Design Quartz *
50	50	15	400	150	30	645	1020 #	600 #
120	1615	330	2200	720	180	5045	34,255 #	24,700 #

NOTE:

1. Telescope Assembled and Aligned in Orbit

*Launch weight savings

11.5.4.3 Beryllium Mirror Manufacturing Process

The mirror will be manufactured from a hot-pressed compact produced from S-200-C sintered beryllium powder. S-200-C denotes a blend of virgin material and powder made from chips recovered from previous sinterings.

The manufacture of the beryllium mirror will follow essentially a similar process to that of the GEP primary mirror. This process involved the following steps:

- a. Rough machining to approximate size and shape
- b. Stress relieving
- c. Finish machining to 0.001 inch of theoretical figure
- d. Kanigen plating - 0.006 inch maximum
- e. Final figuring and polishing
- f. Aluminizing (vapor deposition) and overcoat

11.5.4.4 Theory of P.E.L. in Beryllium

Studies of the precision elastic limit have been made. Perhaps the most applicable to our purpose is the study conducted for the Navy⁽²⁾ in 1960. This study verified that minute deformations were occurring in materials below the conventional proportional limit. The standard used in this study was the stress at which the permanent deformation of one millionth of an inch per inch was evidenced. Since even this much strain can be detrimental to diffraction limited performance of a mirror, new limits must be experimentally determined for our application.

For Beryllium Block, fabricated from sintered powder by hot pressing, the PEL is a function of grain size, percent B_eO (Beryllium Oxide), location within the block and whether the induced plastic strain is parallel to or perpendicular to the direction of pressing. Common S-200C (Brush Beryllium Designation) Powder with grains of 20 microns and containing 1 to 2 percent B_eO yields a PEL of approximately 2500 psi. Decreasing the grain size to 10 microns and recycling the beryllium powder to increase the B_eO content to 4 to 5 percent B_eO , increases the PEL to approximately 8500 psi (Brush Designation I-400). Variations of PEL as low as 1200 psi for

(2) Dr. T.J. Hughe, "An Investigation of the Precision Mechanical Properties of Several Types of Beryllium," G.M. Research Labs, 1960.

specimens having relatively large grain size and small B_eO content, to as high as 16000 psi for specimens having smaller grain size and relatively higher B_eO content are noted in Reference (2).

11.5.4.4.1 Conventional Tensile Specimen Testing

Almost all known testing for determination of the PEL has been performed on Tensile Bars, and the stress, after repeated dead weight load-unloading procedures, which first causes a detectable positive residual (plastic) strain 1.0 microinch/inch is taken as the PEL. This should not discount the possible existence of plastic strains less than 1.0 microinch/inch, which could induce significant figure deviations in terms of diffraction limited performance of the primary mirror. Section 11.5.4.4.2 discusses the possible dependency of tolerable plastic strain on mirror geometry, based on an allowable figure deviation of the mirror surface.

11.5.4.4.2 Geometric Dependence of Tolerable Plastic Strain

Reference (3) indicates that for an allowable spherical distortion (maximum deflection δ) of a flat circular disc, induced by a plastic strain a distance C from the neutral surface, the plastic strain ϵ_o is related to the diameter and thickness of the disc in the following manner:

$$\epsilon_o = \frac{8 \delta C}{d^2} \quad (1)$$

Where δ = maximum allowable deflection (figure deviation) from center to edge of disc, C = one-half thickness, d = diameter of disc; applying this relationship to the thin mirror geometry selected in this study, i.e., $d = 50$ inches; $2C = 0.375$ inch, and selecting a tolerable figure deviation of $\delta = 1 \lambda = 20 \times 10^{-6}$ inches, the tolerable plastic strain would be $\epsilon_o = 0.014 \times 10^{-6}$ in/in.

The expression for ϵ_o is based upon simplifying assumptions and must be experimentally checked for its applicability and validity. The results of an experimental study may result in empirical relationships or suggest additional analytical investigation. An experimental check becomes even more imperative, if interpretation of the stress corresponding to the value of ϵ_o is to be made.

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- (3) Barnes Jr., W.P., "Considerations in the Use of Beryllium for Mirrors," Applied Optics, Vol. 5, No. 12, July 1966.

An apparent difference exists between the stress versus plastic strain data presented in References (3) and (4). (Reference (4), cites Reference (5). Comparisons of test specimens of various types of beryllium in Reference (3) indicate that the stress at which plastic strain occurs approaches zero as plastic strain approaches zero (Figure 11.5.4-1). In Reference (4), the data indicates that plastic strain remains at zero up to a stress of approximately 2000 psi.

To evaluate the implications of the former case, the data in Reference (3) were extrapolated to the narrow plastic strain range near zero as shown in figure 11.5.4-1. Corresponding to the plastic strain computed above, i.e., $\epsilon_0 = 0.014 \times 10^{-6}$ in/in the stress would be approximately 150 psi, for S-200-C and 850 psi for I400. The latter case (Reference (4)) implies that a plastic strain of $\epsilon_0 = 0.014 \times 10^{-6}$ in/in would probably not exist and that the plastic strain corresponding to 2000 psi occurs rather abruptly.

11.5.4.5 Comparison of Mirror Materials

A general evaluation of possible mirror materials is given based upon comparison of their thermal and mechanical characteristics. The field is narrowed to two materials, Beryllium and Quartz, which are compared as above and in fabrication characteristics.

11.5.4.5.1 Conventional Material Properties

11.5.4.5.1.1 Thermal Considerations

The thermal effects encountered by mirrors in space is a major consideration in the selection of a mirror material. To this end an index, called the Thermal Warpage Index, has been used and tabulated for evaluating the thermo-physical merits of optical and structural materials. This index is independent of geometric configuration and is purely a function of the physical properties of the materials.

The Thermal Warpage Index, TWI, is a qualitative measure of the ability of a material to resist a change in its dimensions with a temperature gradient

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- (4) Maringer, R.E., "Review of Dimensional Instability in Metals," DMIC Memo 213, June 23, 1966.
 - (5) Jennings C.G. etc., "Some Factors in the Selection of Beryllium for Engineering Applications," Report No. T4-311/3110, North American Aviation/Autonetics, Anaheim, California, (Undated).

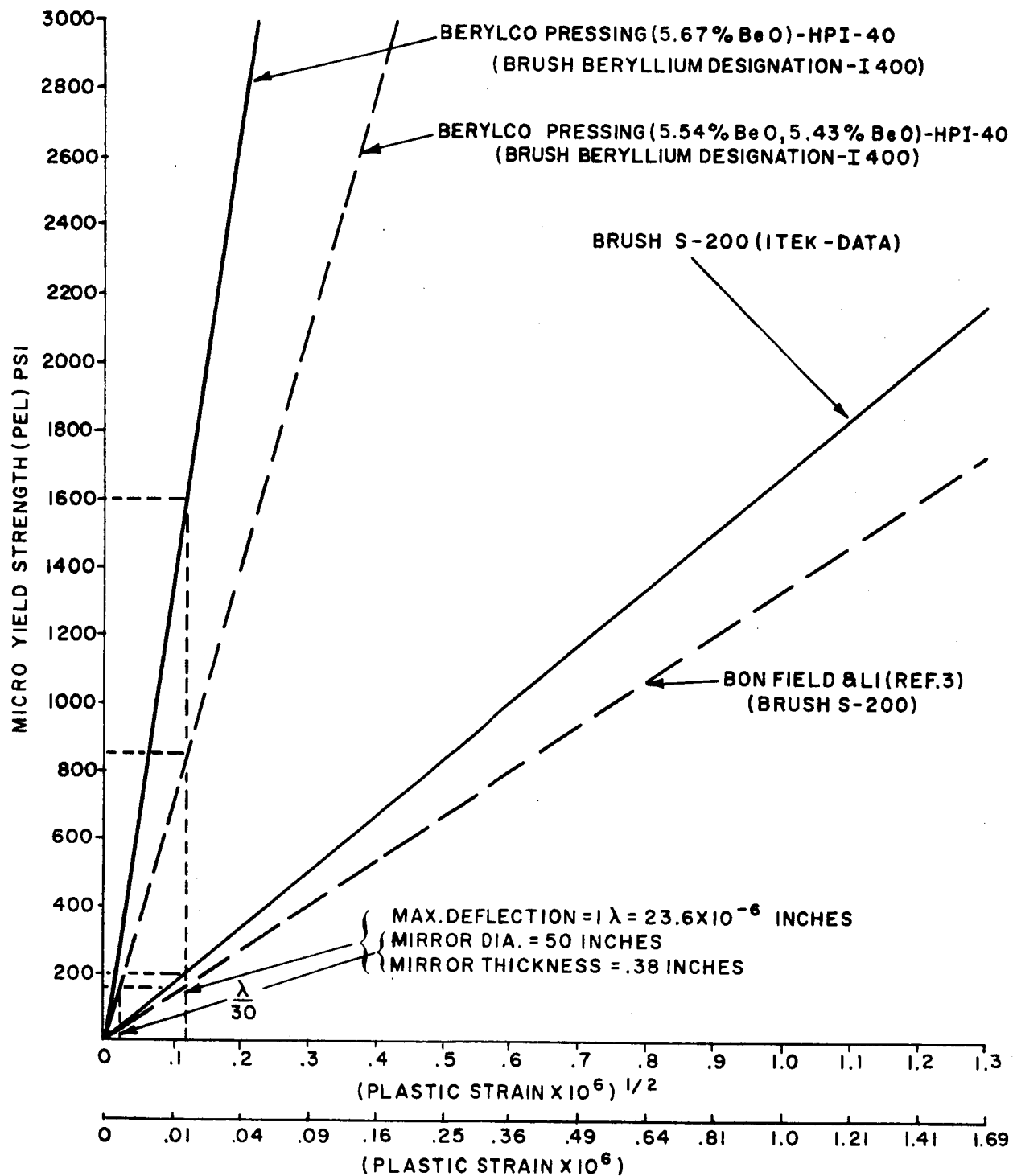


Figure 11.5.4-1. Micro Yield Strength (P.E.L.) Versus Plastic Strain for Beryllium
(Data Extrapolated from Reference (3))

applied across the material. The TWI is the ratio of the coefficient of expansion to the thermal diffusivity or:

$$TWI = \frac{\alpha C_P \rho}{K} \quad (1)$$

where:

$$\text{coefficient of expansion} = \alpha \quad (2)$$

$$\text{thermal diffusivity} = \frac{K}{C_P \rho} \quad (3)$$

K = thermal conductivity

C_P = specific heat

ρ = density

Thus the TWI is proportional to the coefficient of expansion, specific heat, density and inversely proportional to the thermal conductivity. A derivation of formula (3) may be found in texts on heat transfer such as "Elements of Heat Transfer and Insulation" by Jacob and Hawkins.

The values of TWI for various materials are listed in table 11.5.4-3 and may be compared numerically (e.g., TWI = 187, Beryllium; TWI = 450, Quartz). The materials having the lower TWI will have a lesser tendency to support temperature gradients and will stabilize quicker; therefore, lower TWI is favorable.

This is as important in the fabrication stage as in application and use. During fabrication, one of the time consuming factors is the stabilization required between figuring and testing, i.e., the material must be dimensionally stabilized before a measure of mirror figure can be made. Figure 11.5.4-2 is a photograph of a large lightweight beryllium mirror (38-inch diameter, 1-1/2 inches thick). Experience with this mirror and others has shown that the time required to stabilize and test beryllium as compared to quartz was significantly less and resulted in a more favorable manufacturing time cycle.

The problem of dimensional changes and distortions resulting from thermal gradients can be essentially eliminated by the selection of a material with zero or near zero coefficient of expansion, i.e. $\alpha \approx 0$. Two materials can have this characteristic: certain types of Cervit, and quartz at a temperature of -80°C .

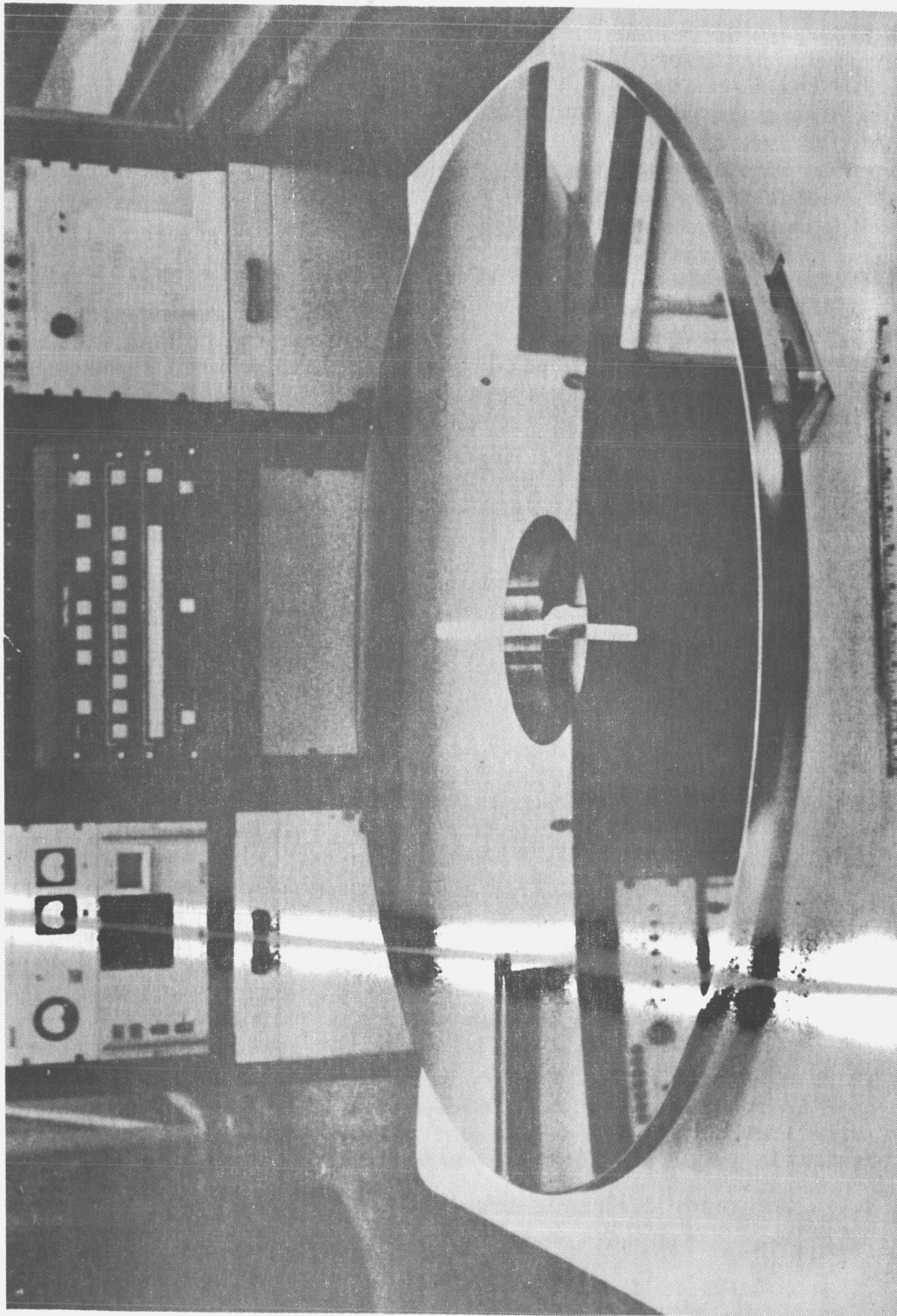


Figure 11.5.4-2. 38" Beryllium Mirror (Space Application).

Cervit is a proprietary, partially vitrified ceramic material made by Libbey-Owens-Ford. It is of the same class of materials as Corning's Pyroceram. LOF has been able to produce Cervit with a zero temperature coefficient of expansion over a range of greater than 50°C. The material appears to be dimensionally stable, as Pyroceram is not. However, it is uncertain at this time that this material will be sufficiently developed in time to be useful for the problem at hand. It may take many more years of development for its application in 120 inch diameter mirrors.

To maintain a quartz mirror at -80°C will require active thermal control with all its additional complexities. With the optical processing electronics in proximity to the mirror, the low temperature environment will be severe enough to degrade performance of the electronics. The expansion coefficient curve for quartz has a slope at -80°C, so that deviations from this temperature will result in a finite and variable expansion characteristic for this material in any event.

11.5.4.5.1.2 Mechanical Considerations

Material properties and dimensions measured at initial assembly versus those after long term usage have been known to change. Thus proper material selection and consideration of these possible changes are very important. The material selected for each mirror must be evaluated by metallurgical and radiographic tests to assure the required homogeneity, internal microstructure, surface conditions, etc. Proper stress relief control must be maintained in order to control and effectively eliminate the effects of aging and creep. Thus, a complete history must be maintained of the mirror material from ingot or press, through manufacture-assembly and on through its useful life. This history will be the basis for an accurate quality control of mirror figure as well as providing much needed data for future designs of better and larger mirrors.

Table 11.5.4-3 tabulates four groups of materials investigated as candidates for mirror material.

Group A lists the materials that will be considered for mirror fabrication based upon availability, practicality, experience and desirability. These materials, headed by beryllium, are ranked in order of lowest TWI. A column indicating the stiffness to mass ratio is also included,

$$\text{stiffness to mass ratio} = \frac{E}{\rho}$$

where:

E = Young's Modulus

The higher the stiffness ratio the more desirable the material is, in that a lighter weight mirror can be fabricated with equal stiffness. Comparing stiffness ratios of beryllium and quartz from table 11.5.4-3,

TABLE 11.5.4-3

COMPARISON OF MIRROR MATERIALS

Group	Material	Coefficient of expansion $\alpha \times 10^{-6}$ (in/in°F)	Density ρ (lbs/in ³)	Specific Heat C_p (Btu/lb°F)	Thermal Conductivity k (Btu-in./ft ² hr °F)	Young's Modulus $E \times 10^6$ (lb/in ²)	(1) Thermal Warpage Index $\frac{TWI \times 10^{-12}}{(\text{hr ft}^2/\text{of in}^4)}$	(2) Stiffness to Mass ratio $\frac{E}{\rho} \times 10^6$ (psi/lb/in ³)
A	Beryllium	6.4	.066	.46	1040	42.0	187	636
	Aluminum 356A-T6	11.9	.097	.23	1056	10.3	253	106
	Invar (36% Ni)	0.80	.290	.12	93.5	21.0	298	72.5
	Quartz (anneal)	0.30	.082	.18	9.7	10.0	450	122
	Pyrex	1.8	.081	.20	6.7	9.3	4400	115
B	Pyroceram (9609)	0.10	.095	.185	25.2	17.3	69	182
	Cervit-C-100	0.05	.091	.20	6.7	13.3	135	146
C	Beryllium oxide	4.4	.109	.24	1500	45	77	412
	Copper (pure)	9.0	.322	.094	2760	12	97	37.3
D	Meehanite GA	6.87	.268	.13	350	21	680	78.3
	416 SS (anneal)	5.50	.28	.11	174	29	980	103
	Inconel	6.1	.286	.10	90	31	1940	108
	Monel (as cast)	9.17	.312	.13	186	23	2000	74
	303 SS	9.6	.29	.12	112	28	2980	96.5

(1) Thermal Warpage Index ($TWI = \frac{\alpha \rho C_p}{k}$)(2) Stiffness to mass ratio (E/ρ)

$$(E/\rho)_{\text{bery.}} = 636 \text{ and } (E/\rho)_{\text{quartz}} = 122.$$

From the consideration of TWI and E/ρ , beryllium appears more favorable as a mirror material on this basis.

Group B lists two new "glass" materials with very low coefficients of expansion and very favorable TWI indices. However, at the present time we have not considered these materials for baseline design of large mirrors, as previously discussed. It is also noted that the E/ρ ratio is relatively good except when compared to beryllium.

Group C lists two additional non-ferrous metals. These have been excluded from Group A because of fabrication problems with the materials and the E/ρ ratio indicates beryllium will produce a much lighter mirror.

Group D lists several members of the iron-steel family. Their exclusion from Group A is readily apparent when considering the TWI values. In fact, previous experimental efforts with these materials have demonstrated the lack of desirable properties for the manufacture of mirrors.

11.5.4.5.1.3 Comparison of Beryllium and Quartz

The major desirable properties required of a good thin mirror material are:

- a. High modulus of elasticity
- b. Low density
- c. High conductivity and thermal diffusivity
- d. Dimensional stability of mirror figure (see section 11.5.4.5.2.2)
- e. Fabricability

It has been shown previously that beryllium has more favorable thermal and mechanical properties than quartz for thin mirror application. Some implications of the above mentioned properties follow.

Relative to the thermal properties of the materials, which are markedly different, we must decide which set is preferable. Taking quartz first, we can say that it will change shape very slightly as the temperature changes, provided that the whole mirror is at the same temperature. If the heat flow to or from the mirror is not uniformly distributed, then quartz's very poor conductivity will allow it to support thermal gradients, which can seriously degrade optical performance. This is especially true if the heat flow is not symmetrical with respect to the optical axis; thus quartz does not have favorable thermal behavior.

Beryllium, on the other hand, has a relatively large coefficient of thermal expansion (6 PPM/°F) but a thermal conductivity that is one third that of copper. This means that large heat flows are needed to cause significant gradients. In a very rough way beryllium is better by a factor of 5 considering that it expands 20 times more than quartz and its conductivity is 100 times better. Beryllium's higher specific heat further limits the rate of change of temperature, which contributes to the reduction of gradients. The change in curvature caused by bulk temperature changes can cause defocusing of the optical system, but this can be compensated for by focus control without figure degradation.

Relative to mechanical properties, a major source of deflection in a mirror blank is its own weight. From this point of view the elastic modulus divided by the material density is a strong figure of merit, and beryllium is better than quartz by a factor of 5.2, in this respect. This assumes of course, that the two mirrors would be geometrically similar. However, honeycomb or egg crate structures are possible with quartz but not with beryllium. This gives quartz a geometric advantage which can be used to compensate for its lower figure of merit. Egg crate construction has been used on quartz mirrors as large as 32 inches. A disadvantage of this construction is that the ribs can "show through" in the optical figure because of nonuniform strains and recovery from strain occurring during polishing. To attempt to overcome this "show through" by reducing the spacing between the ribs reduces the weight saving.

The final factor to be considered is the polishing characteristics of the materials. Here quartz is clearly superior. In fact beryllium is so poor it must be plated with Kanigen (nickel-nickel phosphide) to secure adequate polishing behavior. However, beryllium with a Kanigen coating is of sufficient quality to perform well down to about 1100 Angstroms. This is well into the vacuum ultraviolet where overall performance is limited by the reflective efficiency of the aluminum-dielectric coating system, the actual optical surface for both mirror materials.

11.5.4.5.2 Advanced Material Considerations - Beryllium

The level of mirror figure tolerances required for diffraction limited mirror performance makes important some particular precision properties of beryllium. Such properties include microcreep, dimensional instability and thermal expansion inhomogeneity.

11.5.4.5.2.1 Microcreep

The "Microcreep Limit" is defined in Reference (2) as the stress just sufficient to cause progressive increase in residual (plastic) strain on three successive loadings to the same stress level. Experimental results documented in Reference (2) indicate that the microcreep limit occurs at about twice the PEL stress of beryllium. It is believed that this large difference in stress at which beryllium first shows positive residual strain and the stress at which microcreep begins indicates the tendency of this material to strain harden very rapidly.

It is often difficult to relieve microcreep stresses even after high temperature anneal. Thus the philosophy of restricting the stresses induced in a beryllium blank during thin mirror fabrication to 1/19 PEL (OTAES Part I, section 4.10.5.2) should avoid microcreep and allow stress relief to be performed.

11.5.4.5.2.2 Dimensional Instability

Dimensional instability (defined in Reference (4)) refers to changes in dimensions that occur over a period of time in a specimen without external loading. These dimensional changes may be attributed to (1) metallurgical phase changes in the material; e.g., the martensite-austenite transformation in steels results in a volume increase; (2) relaxation of residual stresses produced during previous history of the blank.

A detailed discussion of the effects of metallurgical phase changes on some materials is given in Reference (4). Reference (3) indicates that there is no evidence of metallurgical phase changes in beryllium below 800°C. Residual stresses are most frequently introduced during fabrication and heat treatment. Machining appears to induce relatively high stresses (as high as 39,000 psi) at the surface of turned beryllium, as pointed out in Reference (4) [Citing Reference (6)]. These residual stresses concentrated near the surface can contribute significantly to dimensional instability if they are sufficiently high to induce microcreep. It is often difficult to remove these stresses even after a high temperature anneal. The machining surface stress of 39,000 psi corresponds to a machining cut of 0.006 inch [(Data in Reference (3))] and since the microcreep limit stress of S200-C is 7000 psi machining cuts of this magnitude will induce microcreep in a beryllium blank. However, fine machining cuts of 0.002 inch will reduce the surface stress to about 5000 psi, which should allow residual stress to be eliminated by heat treatment. Another form of residual stress can be set up as a result of changes in temperature if the material is anisotropic in its thermal expansion, that is, an appreciable amount of stress can be built up between adjacent grains as the temperature of a polycrystalline aggregate changes. Curing such a situation may become complex, since the thermal expansion itself is influenced by both heat treatment and by method of fabrication.

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- (6) Bonfield, W. et al, "The Effect of Surface Conditions On the Micro Strain of Beryllium," Trans. AIME, 227, pp 669-673 (June 1963).

11.5.4.5.2.3 Thermal Expansion Inhomogeneity

Pressings of beryllium powder have characteristics of random anisotropy both locally and from point to point in a particular blank. The effect on thermal expansion homogeneity for particular pressings is to create a difference in thermal expansion in orthogonal directions. In Reference (4) it has been shown that the mean difference in thermal expansion for beryllium hot pressings containing 2 percent B_eO maximum (S-200-C) is 0.2×10^{-6} in/in/°F.

The effect of 4 percent B_eO minimum content (I400) is to reduce the mean difference to 0.07×10^{-6} in/in/°F. This inhomogeneity in thermal expansion is shown in Reference (3) as able to cause a strain in a mirror blank, as a function of temperature changes by assuming a linear variation of α through the blank and using $\Delta\alpha = 0.2 \times 10^{-6}$ in/in/°F, a mean $\bar{\Delta\alpha} = 0.1 \times 10^{-6}$ in/in/°F results. If the tolerable strain for mirror blank deformation of 20×10^{-6} inches is $\epsilon_0 = 0.014 \times 10^{-6}$ inch as discussed in section 11.5.4.4.2, the allowable temperature change in the mirror could then be

$$\Delta T = \frac{0.014 \times 10^{-6} \text{ in/in}}{0.1 \times 10^{-6} \text{ in/in}} = 0.14^\circ\text{F}.$$

While temperature sensitivity to this extreme appears unlikely, it has been suggested that inhomogeneity in large blanks can be controlled by maintaining symmetry in mirror configuration and production operations. Thus each mirror blank would be fabricated from a single beryllium pressing, and the center of symmetry of the finished mirror would be made to coincide with the center of symmetry of the pressing.

11.5.4.5.3 Conclusion

There does not appear at the present time to be any one material which clearly stands out as the material for large space primary mirror applications. However, based upon the considerations previously given, and on the subcontractor's successful experience with beryllium for large mirrors, this material was chosen as a baseline material for the thin mirror experiments. It is considered to have a high probability of success as a material for large, lightweight diffraction limited telescope optics for space applications.

11.5.5 Extrapolation from 50 Inch Mirror Model Data to 120 Inch Prototype Using Scaling Law

The theory of models is based upon scaling laws formulated such that adherence to them will cause identical circumstances to obtain for both model and prototype. Extrapolation of data produced on the model then will allow accurate prediction of the prototype behavior. The scaling laws are usually expressed as dimensionless ratios of the parameters involved, these "numbers" then representing the particular conditions to be held constant for model and prototype. Familiar examples from fluid flow are the Reynolds Number indicating laminar or turbulent flow, and Mach Number indicating subsonic or supersonic regimes.

In the case of the thin mirror nesting experiment, the dimensionless ratio holds the stress the same in both model and prototype, thus indicating the dimensions applicable to the prototype based upon model data. Therefore a relationship exists between the dimensional and mechanical properties of the model and prototype. For this preliminary analysis the conservative assumption is made for the prediction of bending stresses using flat plate theory. Hence the static model discussed in OTAES Part I, section 4.11.5.1 is employed. Then the following relationship is obtained, by equating maximum bending stresses in the model and prototype:

$$\frac{3}{8} \frac{a_m^2 \rho_m}{h_m} \left[3 + \nu_m \right] = \frac{3}{8} \frac{a_p^2 \rho_p}{h_p} \left[3 + \nu_p \right] \quad (1)$$

of as a set of dimensionless ratios

$$\frac{h_p}{h_m} = \left(\frac{a_p}{a_m} \right)^2 \left(\frac{\rho_p}{\rho_m} \right) \left(\frac{3 + \nu_p}{3 + \nu_m} \right) \quad (2)$$

where:

h_p, h_m = thickness of prototype and model

a_p, a_m = radius of prototype and model

ρ_p, ρ_m = density of prototype and model

ν_p, ν_m = Poissons ratio for prototype and model

Since both the prototype and model are to be beryllium mirrors, $\rho_p = \rho_m$ and $\nu_p = \nu_m$ in Equation (2). The only factor which could have a significant effect on the material properties of beryllium is temperature, that is if the prototype and model were to operate at significantly different temperatures. For example, a change in operating temperature could affect the value of the elastic modulus as indicated by the relationship

$$E = 44.8 \times 10^6 - 4.5 \times 10^3 T \quad (1)$$

However the effect of temperature on the density (ρ) and Poissons ratio (ν) would be negligible within the accuracy limit of the stress level. As a result these values will remain essentially constant throughout the operating temperature range of both the model and prototype. If it were required to know the stress level very accurately, minor corrections could be made in the course of data reduction based on the measured temperature.

Noting the foregoing equalities, Equation (2) reduces to

$$\frac{h_p}{h_m} = \left(\frac{a_p}{a_m} \right)^2 = \left(\frac{D_p}{D_m} \right)^2 \quad (3)$$

where

D_p, D_m = diameter of prototype and model

That is, if the ratio of diameters, and the thickness of the model h_m are known, the thickness of prototype h_p will be such that the maximum stress of the prototype will be similar to that of the model. The thickness of the prototype based on the results of space testing of the model is given by the relationship

$$h_p = \frac{3}{8} \frac{a_p^2 \rho_p}{S_A} [3 + \nu] \quad (4)$$

(1) Extrapolated from "Beryllium in Aerospace Structures," Brush Beryllium; Approximately valid for temperature range $80^\circ\text{F} > T > -60^\circ\text{F}$.

where

S_A is the allowable design stress based upon the precision elastic limit (S_{PEL}) of beryllium, determined from the space experiment, which will not appreciably degrade diffraction limited performance of the primary mirror. That is

$$S_A = \frac{S_{PEL}}{1 + M.S.}$$

where

M.S. = assumed margin of safety.

11.5.6 Preliminary Structural Analysis of Test Well

The following preliminary analysis applies to the active optics and thin mirror experiment test well since they are identical except for the primary mirror configurations. Of interest are the fundamental bending natural frequency and maximum dynamic stress of each test well tube with respect to the central ribbed ring as shown in figure 11.3.1-1.

Assuming the central ribbed ring is rigid compared to the tubular structure supporting the secondary mirror, each of the test wells can be idealized as a cantilevered beam with an end mass. To simplify the preliminary analysis the stiffness of the secondary mirror vane system is assumed greater than the tube stiffness, and the weight of the vanes is added to the end mass. This assumption is conservative, since the computed natural frequency and resulting stress will be higher than would be obtained from a more rigorous analysis.

11.5.6.1 Dynamic Analysis

The cantilever model is shown in the figure below.

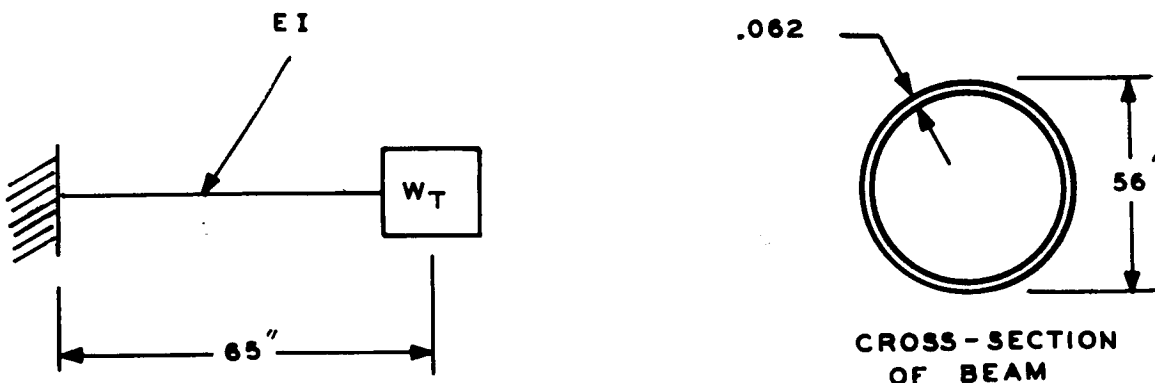


Figure 11.5.6-1 Mathematical Model of Mirror Test Well

The appropriate numerical quantities are indicated in section 11.3.1.3.1 as

R = radius of tube = 28 inches

t = thickness of tube = 0.0625 inch

ℓ = length of cantilever = 65 inches
(distance from central ring to secondary mirror support)

$E = 10 \times 10^6$ lb/in²

$I = \pi R^3 t = 4280$ in⁴

W_T = weight of end mass = 189.4 lbs (this weight includes the secondary mirror and supporting structure, the cover and 25% of the effective structural tube)

The deflection of the tip due to a unit load is

$$C = \frac{\delta}{P} = \frac{\ell^3}{3EI} = \frac{(65)^3}{3(10^7)(4280)} = 2.14 \times 10^{-6} \text{ in/lb}$$

The fundamental natural frequency is

$$f_n = \frac{1}{2\pi} \sqrt{\frac{g}{CW_T}} = \frac{1}{2\pi} \sqrt{\frac{386}{(2.14 \times 10^{-6})(189.4)}}$$

$$= 156 \text{ cps}$$

11.5.6.2 Response Analysis

The random vibration input (PSD) to the fixity is taken from the LEM Vibration Specification as $W_0 = 0.04 \text{ g}^2/\text{cps}$ (49 to 950 cps). Assuming a single degree of freedom response, the RMS response acceleration is given by the expression

$$G_{\text{rms}} = \sqrt{\frac{\pi}{2}} f_n W_0 Q \quad (1)$$

where

Q = magnification factor (assumed = 20)

Substituting the numerical quantities

$$f_N = 156 \text{ cps}; \quad W_0 = 0.04 \text{ g}^2/\text{cps}; \quad \text{and } Q = 20,$$

the root mean square (rms) response of the end mass is

$$G_{\text{rms}} = \sqrt{\frac{\pi}{2} (156) (0.04) (20)} = 14 \text{ G's RMS}$$

The RMS acceleration is usually considered equal to the standard deviation (σ) and the peak acceleration is taken as three times this value or 3σ . The peak acceleration of the end mass is therefore

$$G_{\text{peak}} = 3\sigma = 3 \times 14 \text{ G} = 42 \text{ G}$$

11.5.6.3 Maximum Bending Stress at Fixity

The bending stress is given by the expression

$$S_{\text{max}} = \frac{Mc}{I}$$

Moment at fixity is

$$M = W_T \ell \times G_{\text{peak}} = (189.4) (65) (42) = 515,000 \text{ lb-in}$$

$$S_{\text{max}} = \frac{515,000 (28)}{4280} = 3360 \text{ psi}$$

The maximum allowable stress is usually taken as $S_A = 20,000\text{-psi}$ for aluminum; therefore the margin of safety is

$$\text{M.S.} = \frac{S_A}{S_{\text{max}}} - 1 = 4.95$$

11.5.7 Mirror Correction Analysis

To implement the thermo-mechanical actuators for correcting mirror figure deviations from diffraction limited quality, a relationship between the error patterns and the arrays, (i.e., number and location) of the actuators must be determined. The actuators are connected to the back surface of the primary at specified locations; the number of actuators will be determined by analysis and test based upon anticipated error patterns.

The error patterns are simulations of anticipated deviations of the mirror surface from desirable diffraction limited figure. These deviations can result from thermal distortions due to temperature gradients and changes from earth to orbit, as well as launch induced residual stresses or plastic strains. The distortion of the mirror surface could be relative to the rigid reference frame or may result from a compound thermal distortion of the primary mirror, actuators, and reference frame. For diffraction limited image resolution, actuator correction of the error patterns must be such that the deviations from the spherical mirror radius do not exceed $\pm \lambda/30 = \pm 0.8 \times 10^{-6}$ inch.

As a first attempt at spherical mirror correction, an analytical technique has been developed for correcting a flat circular plate which has been initially deformed to a defined error pattern. This analytical procedure is presented in section 11.5.7.1. The results of the analysis which is performed on a digital computer, yield the quantity and location of the thermo-mechanical actuators for correcting a particular error pattern, the forces which must be applied by the actuators, and the distortion pattern of the mirror after the actuators are applied.

The mathematical mirror simulation was a beryllium circular plate 50 inches in diameter and 0.5 inch thick (uniform), simply supported on the outer boundary. Two error patterns have been considered thus far as illustrated in figure 11.5.7-2; a paraboloid error pattern and sinusoid-paraboloid error pattern. The peak displacements in the error patterns were assumed to be $1\lambda = 20 \times 10^{-6}$ inch. The results of the analysis indicate that an array of four actuators, shown in figure 11.5.7-4 will reduce the maximum distortion of the mirror to 1.1×10^{-6} inch ($\lambda/18$), for the case of a paraboloid error pattern. For the actuator cases considered thus far, the best correction of a sinusoid-paraboloid error pattern was -2.95×10^{-6} inch ($\lambda/7$), produced by the actuator array shown in figure 11.5.7-3. Details of the computer analysis are discussed in sections 11.5.7.2 to 11.5.7.5.

Although cases were considered involving 4, 8, and 12 actuators, some results were inconclusive, and further evaluation is necessary to determine the actuator array or set of arrays that can be utilized to correct for all anticipated error patterns. Additional study is required to determine if sensitivities are inherent in the mathematical method or in the computer

solution. Once this has been established, and suitable adjustments made, the case of the combined primary mirror, actuators, and reference frame subsystem will be analyzed. Coupling the elasticities of the mirror, actuators and reference frame will result in more accurate determination of the actuator forces and resulting corrected deformation pattern of the mirror.

11.5.7.1 Analytical Procedure

The purpose of this study has been to minimize the deflections of a circular mirror, resulting from an induced error pattern, with the application of restoring force actuators. The method of obtaining the values of these restoring forces as well as determining the resulting deflections of the plate is described below:

Consider a circular plate simply supported at the outer boundary as shown in figure 11.5.7-1(a). This represents an idealization of the mirror configuration considered in this analysis, but the following discussion is applicable to any support condition. The positive X, Y axes are as shown, with the origin of coordinates being the center of the plate. The Z axis is the direction of plate deflection and has the positive direction shown in figure 11.5.7-1(a).

If the plate has an initial deflection pattern $Z_0 = f(x, y)$, the deflections at all points on the plate indicated by the coordinates (x, y) will be defined by this function. A method of reducing the deflections of these points, from those defined by the function $Z_0 = f(x, y)$ is to force one or more points on the plate to a reduced value of deflection. Usually restraint forces are applied opposite to the direction of initial deflection such that the displacements at the location of these forces are reduced to zero. The deflections at the remaining points on the plate are computed by subtracting the deflection at these remaining points, resulting from the applied (restraint) forces, from the values of the initial deflections Z_0 at these points using the principle of superposition.

To aid this approach the plate is divided into an array of discrete points, as indicated by the gridwork intersections shown in figure 11.5.7-1(a). Supposing there are m discrete points on the plate, n of which are the locations of the restoring (restraint) forces. For explanatory purposes let $n = 4$, where the locations of the forces are as indicated in figure 11.5.7-1(a). It is required that the displacements at the coordinated (x_1, y_1) , (x_2, y_2) , (x_3, y_3) and (x_4, y_4) be reduced to zero.

The deflections of $(Z_1)_F$ of the $n = 4$ force application points, due to the $n = 4$ applied forces F_j , can be written as a set of simultaneous equations in the following manner;

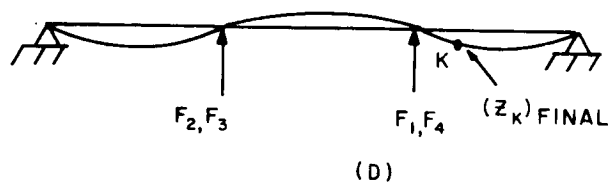
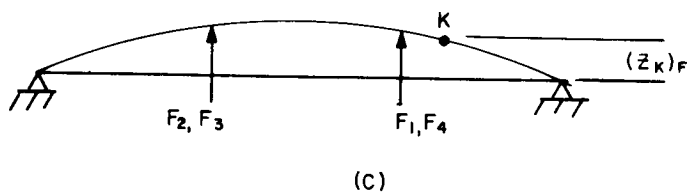
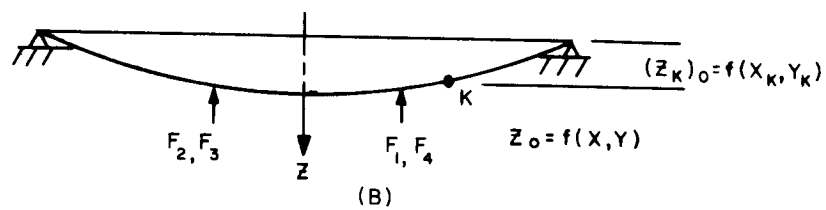
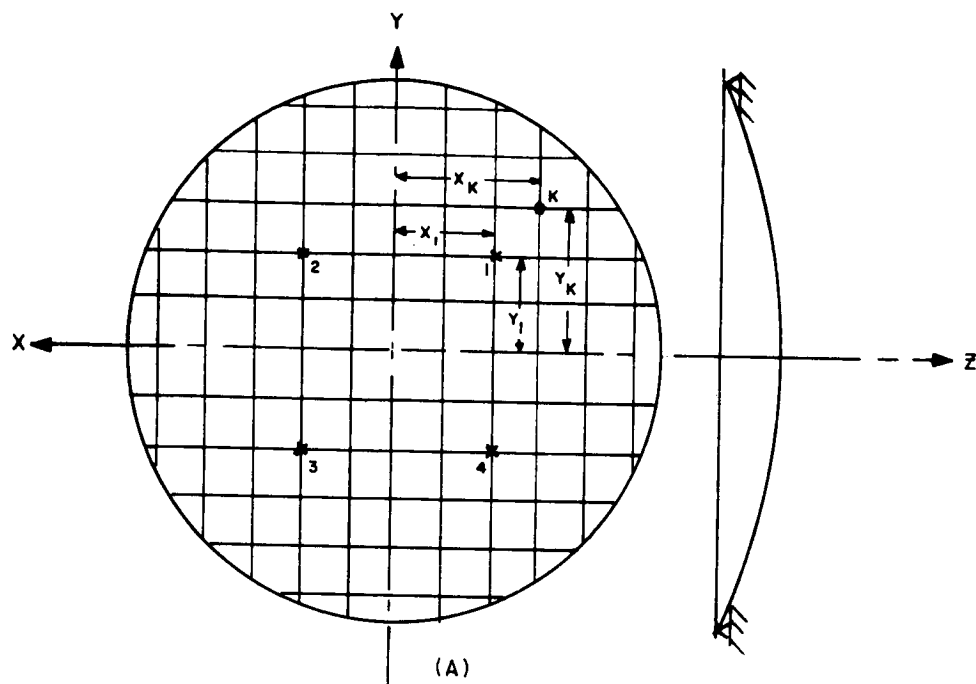


Figure 11.5.7-1 Rim Supported Circular Plate

$$\begin{aligned}
(Z_1)_F &= \alpha_{11} F_1 + \alpha_{12} F_2 + \alpha_{13} F_3 + \alpha_{14} F_4 \\
(Z_2)_F &= \alpha_{21} F_1 + \alpha_{22} F_2 + \alpha_{23} F_3 + \alpha_{24} F_4 \\
(Z_3)_F &= \alpha_{31} F_1 + \alpha_{32} F_2 + \alpha_{33} F_3 + \alpha_{34} F_4 \\
(Z_4)_F &= \alpha_{41} F_1 + \alpha_{42} F_2 + \alpha_{43} F_3 + \alpha_{44} F_4
\end{aligned} \tag{1}$$

where the α_{ij} ($i = 1, 4; j = 1, 4$) are influence coefficients, defined as the deflection at a point i due to a unit force applied at a point j .

Equation (1) can be written in the reduced matrix form

$$\begin{Bmatrix} (Z_1)_F \\ (Z_2)_F \\ (Z_3)_F \\ (Z_4)_F \end{Bmatrix} = [A] \times \begin{Bmatrix} F_1 \\ F_2 \\ F_3 \\ F_4 \end{Bmatrix} \tag{2}$$

where $[A] = \begin{bmatrix} \alpha_{11} & \alpha_{12} & \alpha_{13} & \alpha_{14} \\ \alpha_{21} & \alpha_{22} & \alpha_{23} & \alpha_{24} \\ \alpha_{31} & \alpha_{32} & \alpha_{33} & \alpha_{34} \\ \alpha_{41} & \alpha_{42} & \alpha_{43} & \alpha_{44} \end{bmatrix}$ = influence coefficient matrix for the force application points

To reduce the deflections at points $i = 1, 4$ to zero, the displacements $(Z_i)_F$ caused by the applied forces at these points must be equal to the initial displacements $(Z_i)_0 = f(x_i, y_i)$; in this case

$$\begin{aligned}(Z_1)_F &= (Z_1)_0 = f(x_1, y_1) \\ (Z_2)_F &= (Z_2)_0 = f(x_2, y_2) \\ (Z_3)_F &= (Z_3)_0 = f(x_3, y_3) \\ (Z_4)_F &= (Z_4)_0 = f(x_4, y_4)\end{aligned}\tag{3}$$

Substituting equations (3) into equation (2) for $(Z_i)_F$,

$$\begin{Bmatrix} (Z_1)_0 \\ (Z_2)_0 \\ (Z_3)_0 \\ (Z_4)_0 \end{Bmatrix} = [A] \times \begin{Bmatrix} F_1 \\ F_2 \\ F_3 \\ F_4 \end{Bmatrix}\tag{4}$$

To solve for the values of the restraint forces F_j , both sides of equation (4) are pre-multiplied by the inverse $[A]^{-1}$ of the matrix $[A]$. The restraint forces F_j are equal to the product of $[A]^{-1}$ and the initial deflection column matrix $\{(Z_i)_0\}$; that is,

$$\begin{Bmatrix} F_1 \\ F_2 \\ F_3 \\ F_4 \end{Bmatrix} = [A]^{-1} \times \begin{Bmatrix} (Z_1)_0 \\ (Z_2)_0 \\ (Z_3)_0 \\ (Z_4)_0 \end{Bmatrix}\tag{5}$$

The restoring forces are opposite in direction to the initial displacement pattern $Z_o = f(x, y)$ as shown in **figure 11.5.7-1(b)**. Therefore, the effect of these forces will result in deflections of the plate in the negative Z direction as shown in **figure 11.5.7-1(c)**.

If the deflection of an arbitrary point K on the plate is considered, the initial deflection of this point is

$$(Z_K)_o = f(x_K, y_K) \quad (6)$$

The deflection of the point K due to the restoring forces F_j is as follows:

$$(Z_K)_F = - \left[\beta_{K1} F_1 + \beta_{K2} F_2 + \beta_{K3} F_3 + \beta_{K4} F_4 \right] \quad (7)$$

where the β_{Kj} ($j = 1, 4$) is the deflection of the arbitrary point K due to a unit applied restraint force F_j . Since the values of F_j have been computed by equation (5), these values are substituted into equation (7) for the deflection $(Z_K)_F$.

The final deflection $(Z_K)_{FINAL}$ at the point K is given by the sum of equations (6) and (7), using the principle of superposition.

$$\begin{aligned} (Z_K)_{FINAL} &= (Z_K)_o + (Z_K)_F \\ (Z_K)_{FINAL} &= f(x_K, y_K) - \left[\beta_{K1} F_1 + \beta_{K2} F_2 + \beta_{K3} F_3 + \beta_{K4} F_4 \right] \end{aligned} \quad (8)$$

A typical final deflection pattern is shown in **figure 11.5.7-1(d)**, where the deflections at the force application point are shown to be zero. The forces F_j will be felt as reactions by supports at these zero deflection locations.

In the general case there can be m discrete points and n force application points; therefore a summation of this analytical technique will be given in the general sense;

Computation of restraint forces:

$$a. \quad \left\{ Z_i \right\}_F = [A] \quad \left\{ F_j \right\} ; (i = 1, n; j = 1, n)$$

where

$$[A] = \begin{bmatrix} \alpha_{11} & \alpha_{12} & \cdots & \alpha_{1n} \\ \alpha_{21} & \alpha_{22} & \cdots & \alpha_{2n} \\ \vdots & \vdots & \ddots & \vdots \\ \alpha_{n1} & \alpha_{n2} & \cdots & \alpha_{nn} \end{bmatrix} \quad \begin{array}{l} \text{influence coefficient matrix} \\ = \text{for force application points} \\ (n \times n) \end{array}$$

The matrix symbol $\left\{ \right\}$ indicates a column matrix of elements.

b. Condition of zero displacement at force application points

$$\left\{ (Z_i)_F \right\} = \left\{ (Z_i)_O \right\} = \left\{ f(x_i, y_i) \right\}$$

NOTE: It is sometimes advantageous not to reduce the displacement at the force application point to zero, but to some arbitrary value ΔZ_i . For this case the above relationship is modified as follows:

$$\left\{ (Z_i)_F \right\} = \left\{ (Z_i)_O \right\} \pm \left\{ \Delta Z_i \right\} ,$$

which reduces to the above relationship for $\Delta Z_i = 0$

c. Solution for restraint forces

$$\left\{ F_j \right\} = [A]^{-1} \left\{ (Z_i)_O \right\}$$

or, as noted in step b.

$$\left\{ F_j \right\} = [A]^{-1} \left\{ (Z_i)_O \pm \Delta Z_i \right\}$$

Computation of final plate deflections:

a. Initial deflections of all discrete points K on plate

$$\left\{ (Z_K)_O \right\} = \left\{ f(x_K, y_K) \right\} ; (K = 1, m)$$

b. Deflections due to restraint forces F_j

$$\left\{ (Z_K)_F \right\} = - [B] \left\{ F_j \right\} ; (K = 1, m; j = 1, n)$$

where:

$$[B] = \begin{bmatrix} \beta_{11} & \beta_{12} & \dots & \beta_{1n} \\ \beta_{21} & \beta_{22} & \dots & \beta_{2n} \\ \vdots & \vdots & \ddots & \vdots \\ \beta_{m1} & \beta_{m2} & \dots & \beta_{mn} \end{bmatrix} = \begin{matrix} \text{influence coefficient matrix} \\ \text{deflections of } m \text{ discrete} \\ \text{points due to } n \text{ applied unit} \\ \text{restraint forces } F_j \end{matrix}$$

c. Final deflections of plate

$$\left\{ (Z_K)_{\text{FINAL}} \right\} = \left\{ (Z_K)_O \right\} + \left\{ (Z_K)_F \right\}$$

$$\left\{ (Z_K)_{\text{FINAL}} \right\} = \left\{ f(x_K, y_K) \right\} - [\tilde{B}] \left\{ F_j \right\}$$

11.5.7.2 Mirror Correction Analysis

For the mirror correction analysis the OTAES 50 inch mirror was idealized as a uniform thickness circular plate, simply supported at the outer boundary. The plate was divided into a set of discrete points corresponding to possible location of the force actuators, of the push-pull type, as shown in figure 11.5.7-3. These points also define the final deflection pattern of the plate for the different force actuator arrays considered in this study.

Two error deflection patterns were considered in this study; paraboloid and sinusoid-paraboloid. The paraboloidal deflection pattern shown in figure 11.5.7-2(a) is given by the function

$$(Z_i)_O = 20 \times 10^{-6} \left[1 - \frac{r_i^2}{a^2} \right] \quad (1)$$

$$\text{where: } r_i^2 = x_i^2 + y_i^2$$

x_i, y_i = coordinates of discrete points

a = radius of the plate

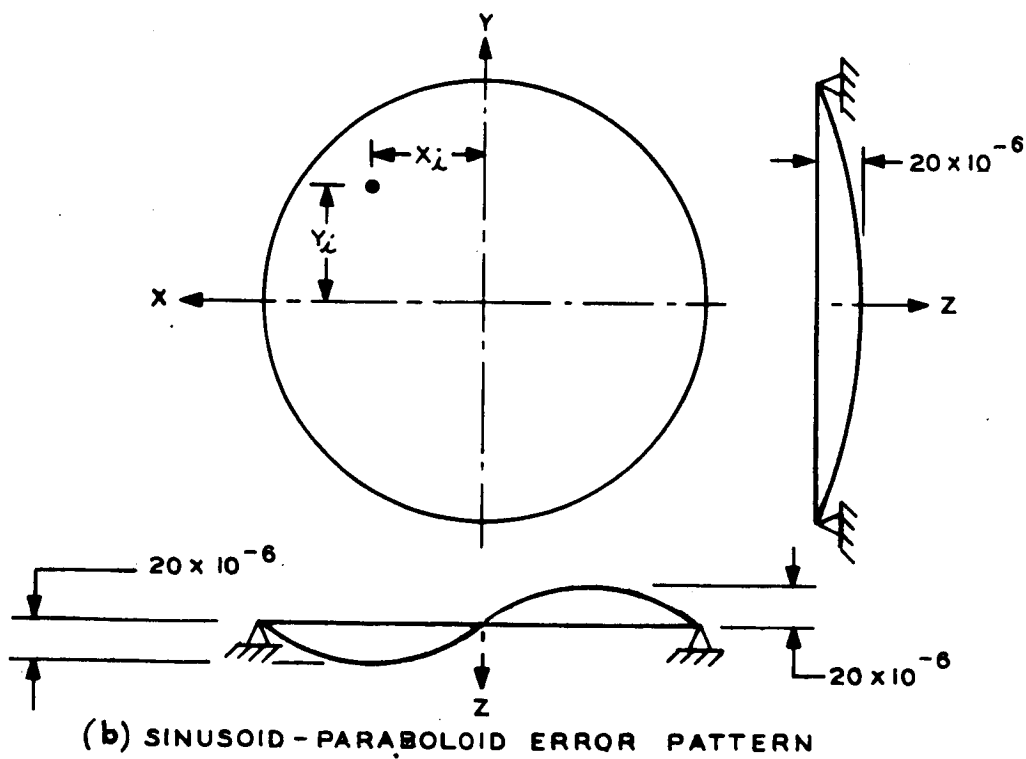
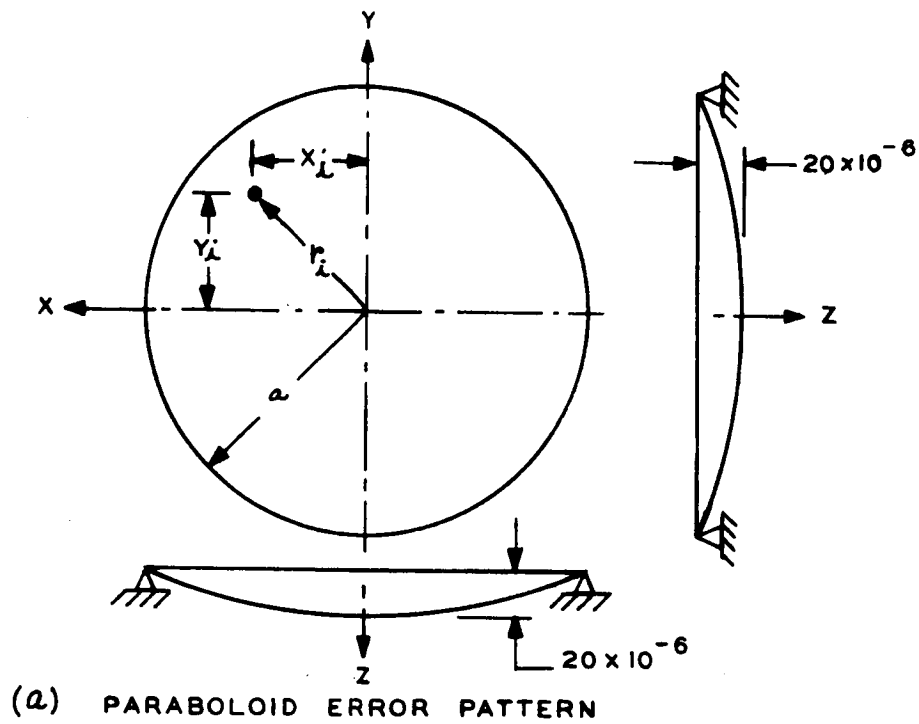


Figure 11.5.7-2 Error Patterns

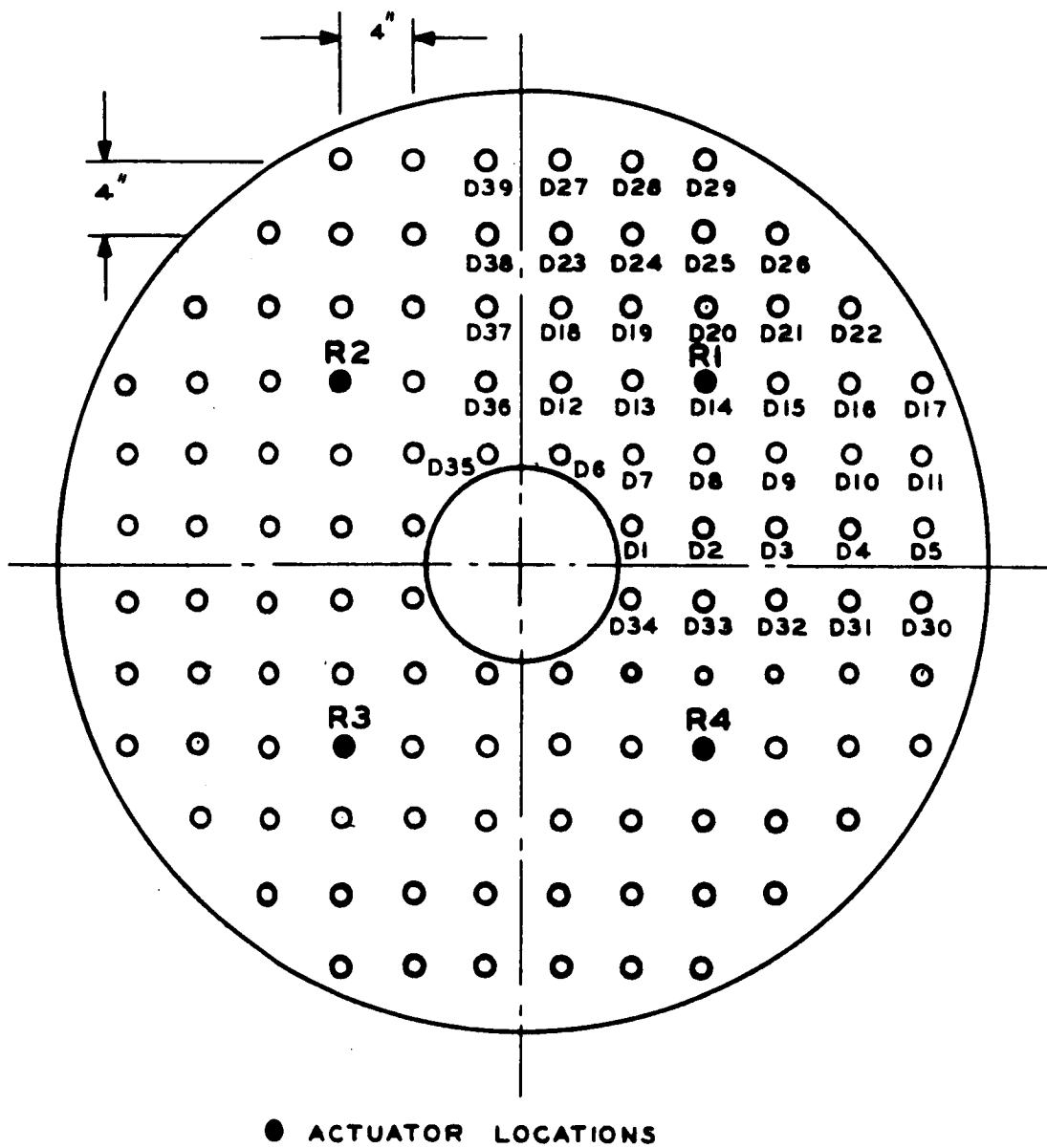


Figure 11.5.7-3 Schematic of 50" Mirror Showing Actuator Locations - Case 1

The sinousoid-paraboloid deflection pattern shown in figure 11.5.7-2(b) is given by the function

$$(Z_i)_o = 20 \times 10^{-6} \left[1 - \frac{y_i^2}{a^2 - x_i^2} \right] \sin \frac{x_i}{\sqrt{a^2 - y_i^2}} \quad (2)$$

The equation defining the deflections of a uniform thickness simply supported circular plate due to discrete applied forces, was obtained from Reference (1). This equation applies for a uniform plate without a hole in the center. Although the plate analyzed has a hole in the center, it is anticipated that the errors incurred by using this formula will not affect the accuracy necessary to evaluate the best force actuator array for minimizing the deflections resulting from the initial error pattern. The deflection equation in Reference (1) is used for obtaining the influence coefficients of the force actuator points, which are included in the [A] matrix, (see section 11.5.7.1, and the influence coefficients of the deflection points due to the actuator forces which are included in the [B] matrix (see section 11.5.7.1).

11.5.7.3 Computer Analysis

Both the analytical procedure discussed in section 11.5.7.1 and the deflection equation given in Reference (1) were programmed for the digital computer which has a core memory of 8000 bits. Due to core memory limitations the maximum number of force actuators which could be considered was $n = 12$. The maximum number of points which could be used to define the final deflection pattern was $m = 50$. To facilitate the deflection analysis, force actuator arrays were selected to be symmetric with respect to the x and y axes so that a single quadrant of $m = 39$ points could be used to describe the deflection pattern of the restrained circular plate. In this manner, the deflection patterns in the remaining quadrants are defined using the results obtained in the single quadrant. This is discussed further in section 11.5.7.4.

The initial computer runs performed to check out the computer program indicated a sensitivity in the inversion of the [A] matrix (see section 11.5.7.1) for greater than $n = 5$ force actuators. That is, the computer program sub-routine for matrix inversion based on the Count matrix inversion technique; treated any [A] matrix greater than a size 5 by 5 as a singular matrix, indicating that a solution for the forces either did not exist or was not unique. This was possible due to rounding off errors in the computer

- (1) Raymond J. Roark, "Formulas for Stress and Strain," Third Edition, 1954, p. 195 (case 4).

computation of the matrix inversion. The sensitivity was apparent since the elements of the $[A]$ matrix were not well separated, resulting in a determinant solution $|A|$ which was small, but not near zero as would be the case in a singular matrix.

To achieve a solution for the actuator forces the matrix equation given by equation 4 in section 11.5.7.1 was treated in expanded form as a set of linear simultaneous equations which were solved using the relaxation method described in Reference (2). This method is not as sensitive to rounding off errors in computing the actuator forces and allows a larger number of force actuators to be considered.

The following numerical quantities were used in the mirror correction analysis.

a = radius of plate = 25 inches

t = thickness of plate = 0.5 inches

E = modulus of elasticity = 42×10^6 psi

ν = Poisson's ratio = .02

In one typical case for which a computer solution was obtained, force actuators were assumed located at points $(-6, 6)$, $(6, 6)$, $(6, -6)$, $(-6, -6)$ (i.e., $n = 4$) where the terms in the parenthesis are the x and y locations (in inches) of the force actuators. This case is shown in figure 11.5.7-4 where the actuator points are labeled R_1, R_2, R_3, R_4 and the $m = 39$ deflection points are labeled $D_1, D_2, \dots, D_{39}$. The influence coefficients of the actuator points and the deflection points due to applied unit loads F_J ($J = 1, 4$) at the actuator locations, were computed. The solution for the actuator forces in this case is $F_1 = F_2 = F_3 = F_4 = .0775$ lbs. Finally, the solution for the final plate deflections was computed.

11.5.7.4 Results

In the computer study, arrays of 4, 8 and 12 push-pull actuators were considered to minimize the paraboloid and sinusoid-paraboloid error patterns. The results of this study are tabulated in Table 11.5.7-1, 11.5.7-2, and 11.5.7-3. Each of the cases in the tables corresponds to an actuator array indicated by the appropriate figure number. For instance, case (2) in Table 11.5.7-1 corresponds to the actuator array shown in figure 11.5.7-4. Both the paraboloid (P) and Sinusoid-paraboloid (S-P) error patterns are considered

(2) M.G. Salvadori and M.L. Baron, "Numerical Methods in Engineering," Second Edition, p. 36, 1961.

TABLE 11.5.7-1

NUMBER OF ACTUATORS = 4

Case	Error Pattern	Figure	Actuator Forces (lb)				Maximum Pos. Deflection		Maximum Neg. Deflection	
			F ₁	F ₂	F ₃	F ₄	Point	Value (In)x10 ⁻⁶	Point	Value (In)x10 ⁻⁶
1	P	4.10-17	0.115	0.115	0.115	0.115	21	0.61	34	-1.685
	S-P		-0.842	0.842	0.842	-0.842	32	2.87	24	-2.95
2	P	4.10-18	0.078	0.078	0.078	0.078	4	1.1	34	-0.4
	S-P		-0.867	0.867	0.867	-0.867	22	1.8	24	-3.33
3	P	4.10-19	0.16	0.16	0.16	0.16	24	0.68	34	-2.5
	S-P		-0.8	0.8	0.8	-0.8	22	1.85	24	-3.31
4	P	4.10-20	0.071	0.071	0.071	0.071	20	1.26	34	-0.045
	S-P		-0.474	1.34	0.474	-1.34	22	1.66	24	-3.32

TABLE 11.5.7-2

NUMBER OF ACTUATORS = 8

Case	Error Pattern	Figure	Actuator Forces (lb)								Maximum Pos. Deflection		Maximum Neg. Deflection	
			F ₁	F ₂	F ₃	F ₄	F ₅	F ₆	F ₇	F ₈	Point	Value (In)x10 ⁻⁶	Point	Value (In)x10 ⁻⁶
5	P	4.10-21	.564	.554	.55	.556	-.218	-.189	-.191	-.207	23	3.03	33	-2.4
	S-P		-.869	.861	.87	-.862	.405	-.20	-.21	+.204	22	1.85	24	-3.31
6	P	4.10-22	1.094	1.178	1.095	1.09	-0.48	-0.58	-0.5	-0.5	18	4.34	3	-2.63
	S-P		-1.116	1.115	1.773	-1.773	0.19	-0.19	-2.316	2.316	22	1.84	34	-3.31

TABLE 11.5.7-3

NUMBER OF ACTUATORS = 12

Case	Error Pattern	Figure	Actuator Forces (lb)												Max. Pos. Defl.		Max. Pos. Defl.	
			F ₁	F ₂	F ₃	F ₄	F ₅	F ₆	F ₇	F ₈	F ₉	F ₁₀	F ₁₁	F ₁₂	Point	Value x10 ⁻⁶	Point	Value x10 ⁻⁶
7	P	4.10-23	9.5	7.49	9.62	7.44	9.53	7.56	9.43	7.61	-10.74	-10.73	-10.72	-10.75	10	.64	11	-.83
	S-P		25.	-63.5	63.3	-24.5	-35.	73.6	-73.2	34.4	0	0	0	0	0	-	-	-

in each case. The actuator forces ($F_1, F_2 \dots$) required to create a zero deflection at the actuator locations are given. The actuator locations ($R_1, R_2 \dots$) are shown in the appropriate figure. Also included in the tables are the maximum positive and negative deflections of the plate due to the restraint of a particular actuator array, and the points in the appropriate figure where these deflections exist.

Deflection contour plots (showing lines of constant deflection) for cases 2, 5 and 7 are shown for a single quadrant in figures 11.5.7-10 through 11.5.7-14. For the paraboloid error pattern symmetry of the contour plot exists across both the x and y axes. For the sinusoid-paraboloid error pattern symmetry exists across the x axis and asymmetry exists across the y axis.

The analytical results indicate that a mirror correction for the paraboloid error pattern near $\lambda/30 = 0.8$ times 10^{-6} inch can be achieved with four actuators corresponding to case (2) in table 11.5.7-1. The maximum deflection of the plate is 1.1 times 10^{-6} inch which is obtained by applying 0.078 pounds at each of the actuator locations shown in figure 11.5.7-4. The deflection contour plot shown in figure 11.5.7-10 indicates the distortion pattern of the 50 inch mirror when the force actuators are applied to correct for the paraboloid error pattern. A reduction in the deflections of the plate to less than $\lambda/30$ can also be achieved with twelve actuators as indicated in table 11.5.7-3. The actuator forces required are of the order of 10 pounds, resulting in a maximum deflection of -0.83 times 10^{-6} inch, as shown in the approximate contour map in figure 11.5.7-14, and can at best be taken as an order of magnitude of the solution due to the present uncertainty of this solution. However, the error correction due to the four actuators shown in figure 11.5.7-4, has a high confidence level.

The best correction for the sinusoid-paraboloid error pattern with the limited number of actuator arrays considered was -2.95 times 10^{-6} inch with an actuator array as shown in figure 11.5.7-3. This corresponds to case 1 of table 11.5.7-1, which requires actuator forces of $F_1 = F_4 = -0.842$ pounds and $F_2 = F_3 = 0.842$ pounds. However, all cases considered using four and eight actuators, resulted in approximately this maximum deflection as indicated in tables 11.5.7-1 and 11.5.7-2. It is anticipated, that of these cases only case 7, table 11.5.7-3 could have resulted in a smaller deflection. However, the results in this case were uncertain at this time, since the solutions for the actuator forces did not converge. Typical contour plots of the actuator corrected deflection pattern are shown in figures 11.5.7-11 and 11.5.7-13, for the case of the sinusoid-paraboloid error pattern.

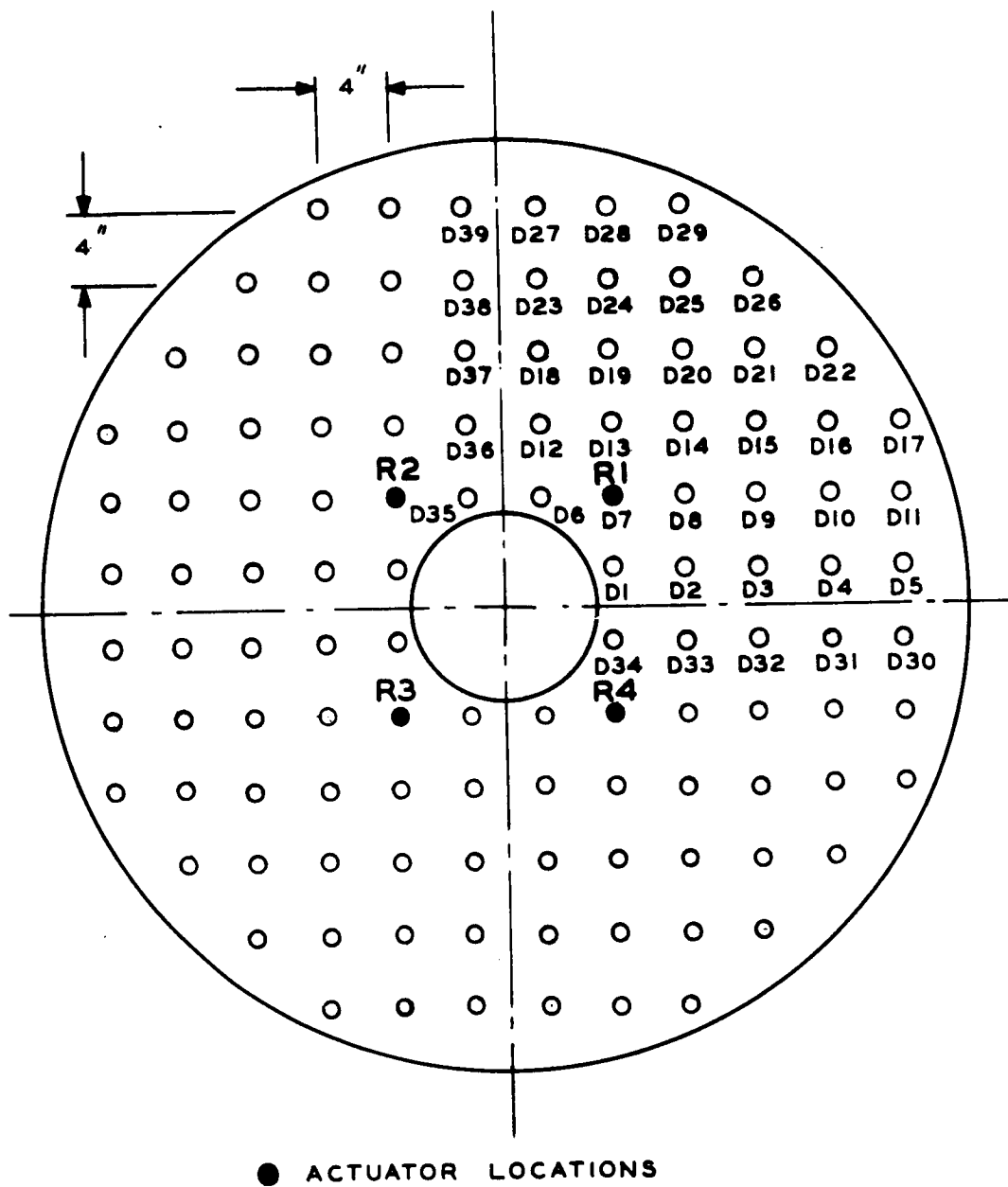


Figure 11.5.7-4 Schematic of 50" Mirror Showing
Actuator Locations - Case 2

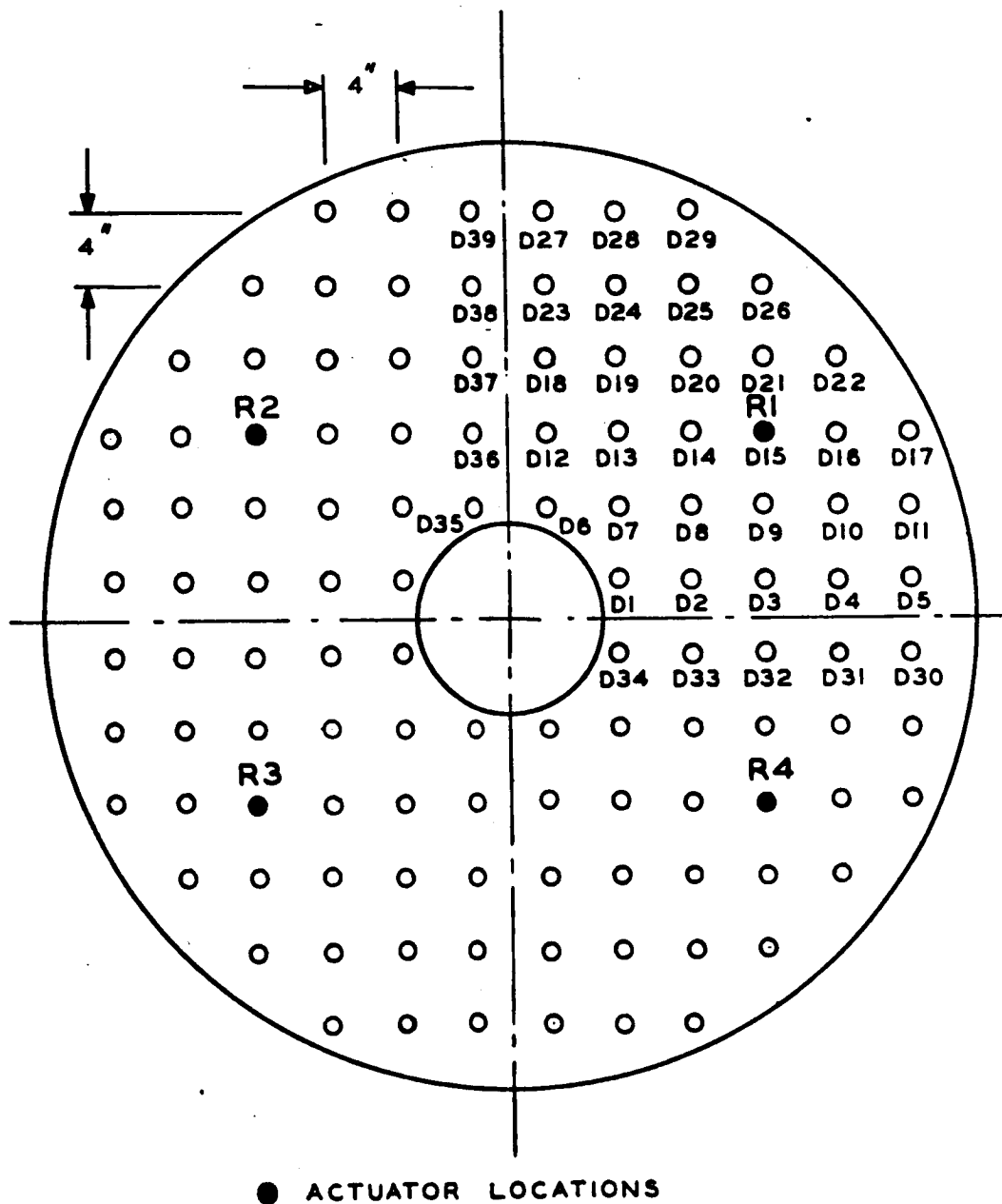


Figure 11.5.7-5 Schematic of 50" Mirror Showing
Actuator Locations - Case 3

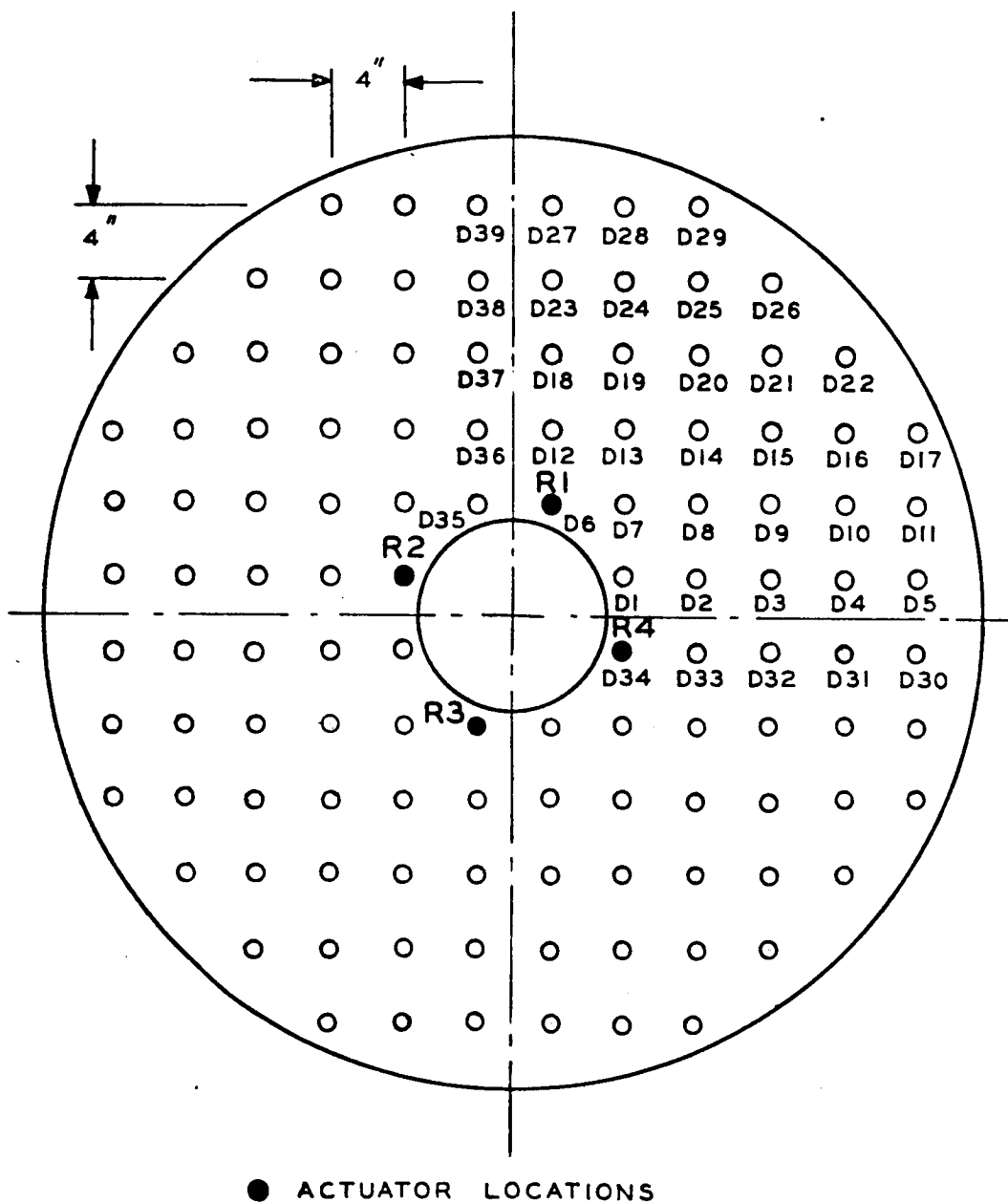


Figure 11.5.7-6 Schematic of 50" Mirror Showing
Actuator Locations - Case 4

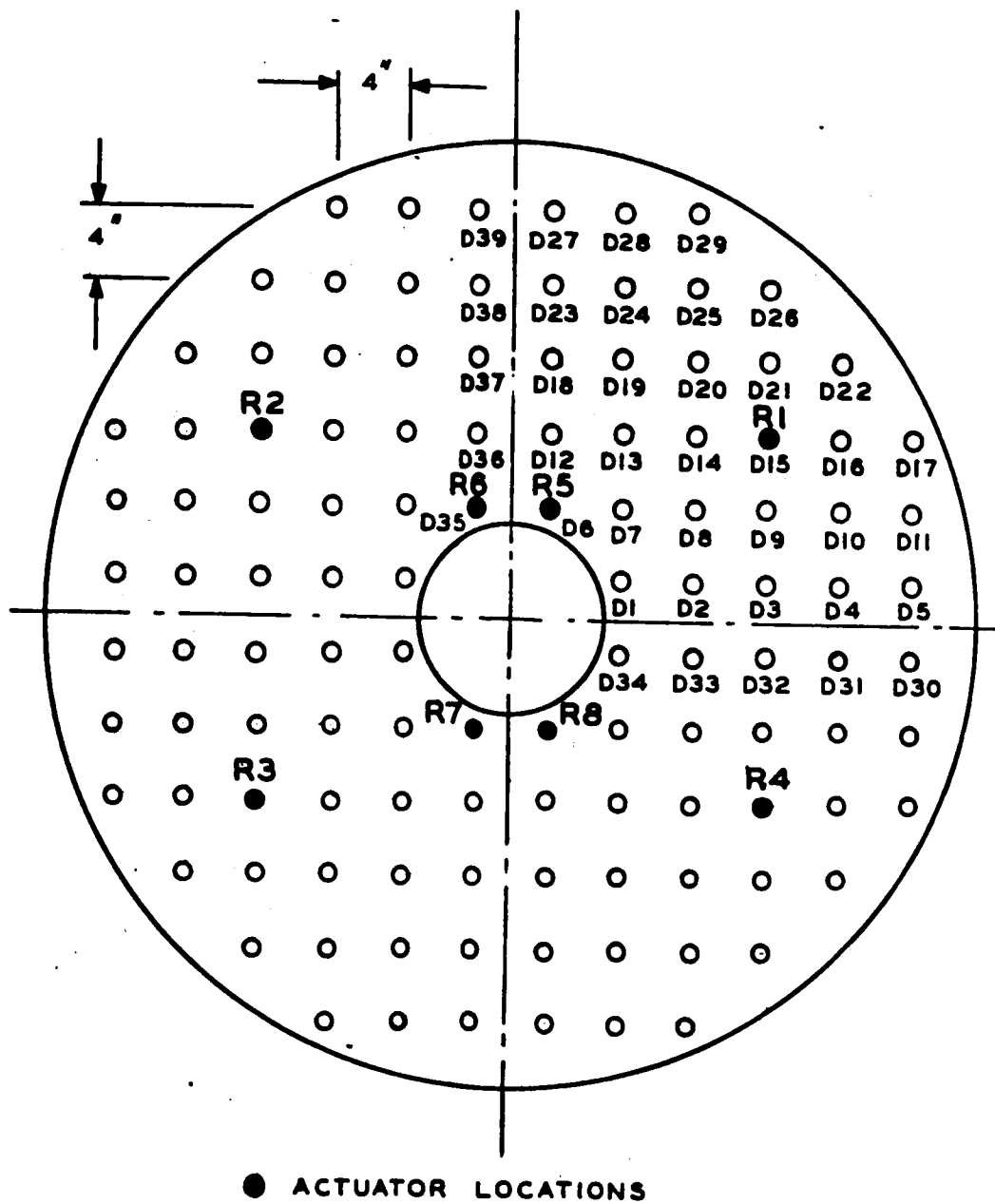


Figure 11.5.7-7 Schematic of 50" Mirror Showing Actuator Locations - Case 5

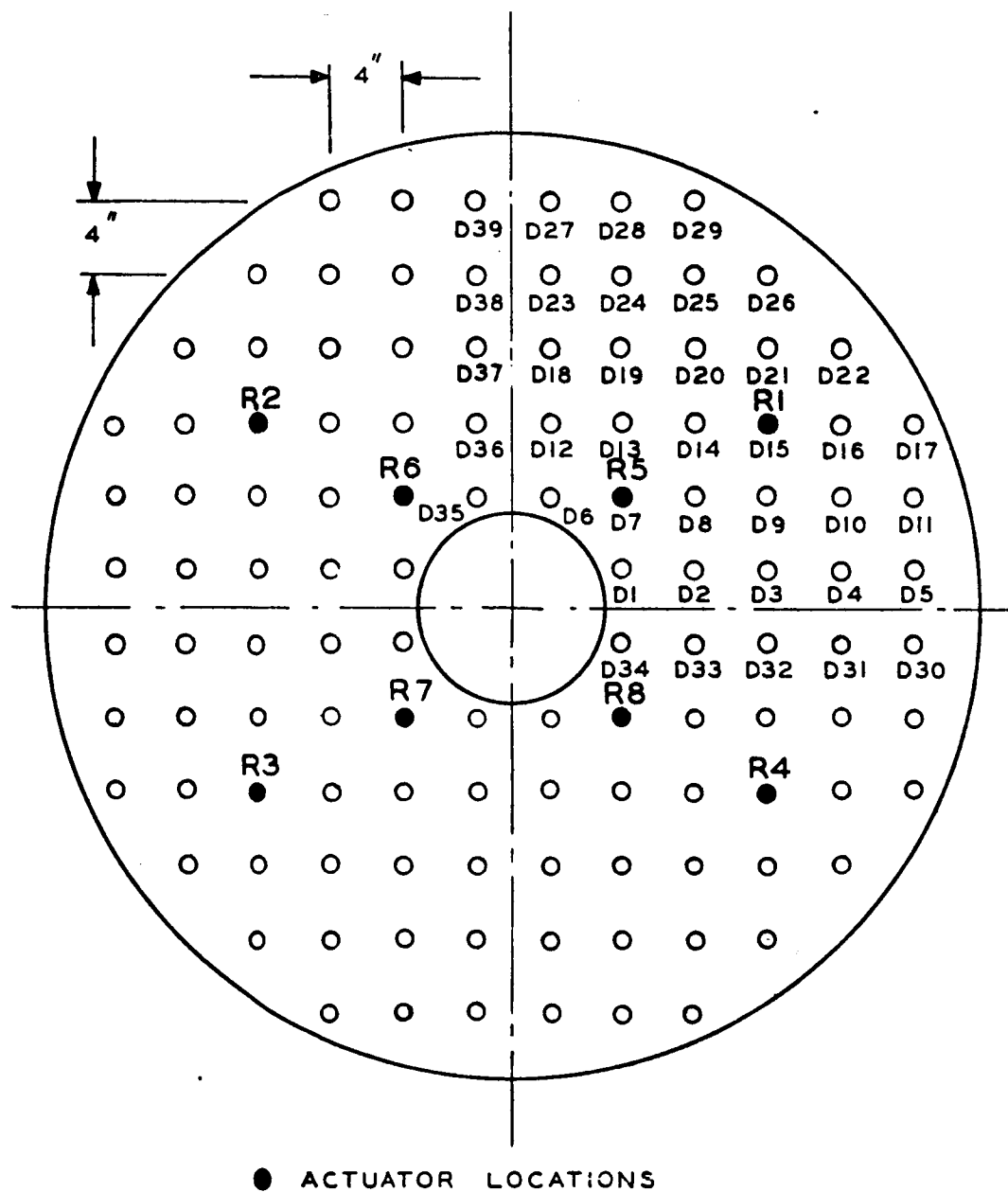


Figure 11.5.7-8 Schematic of 50" Mirror Showing
Actuator Locations - Case 6

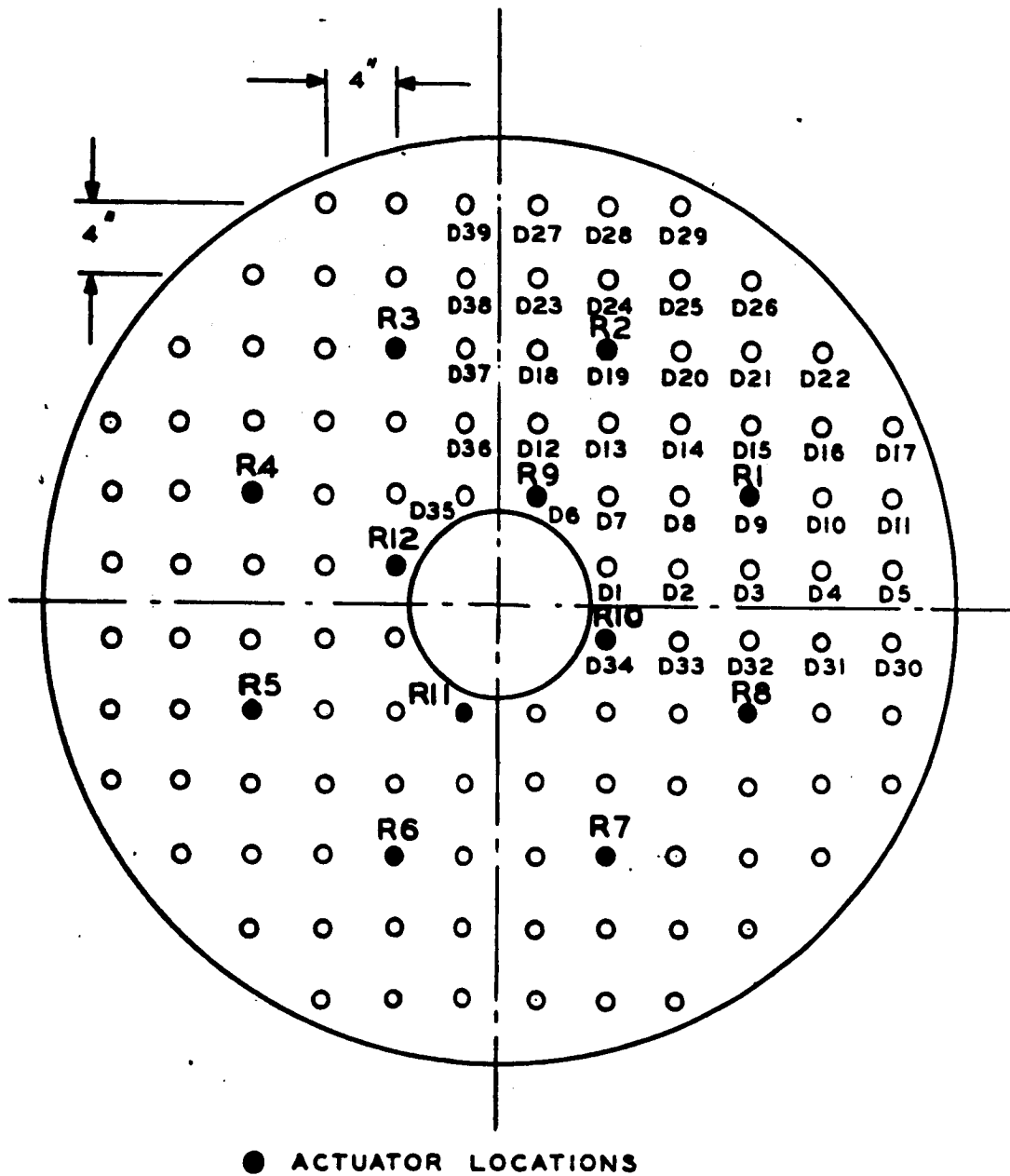
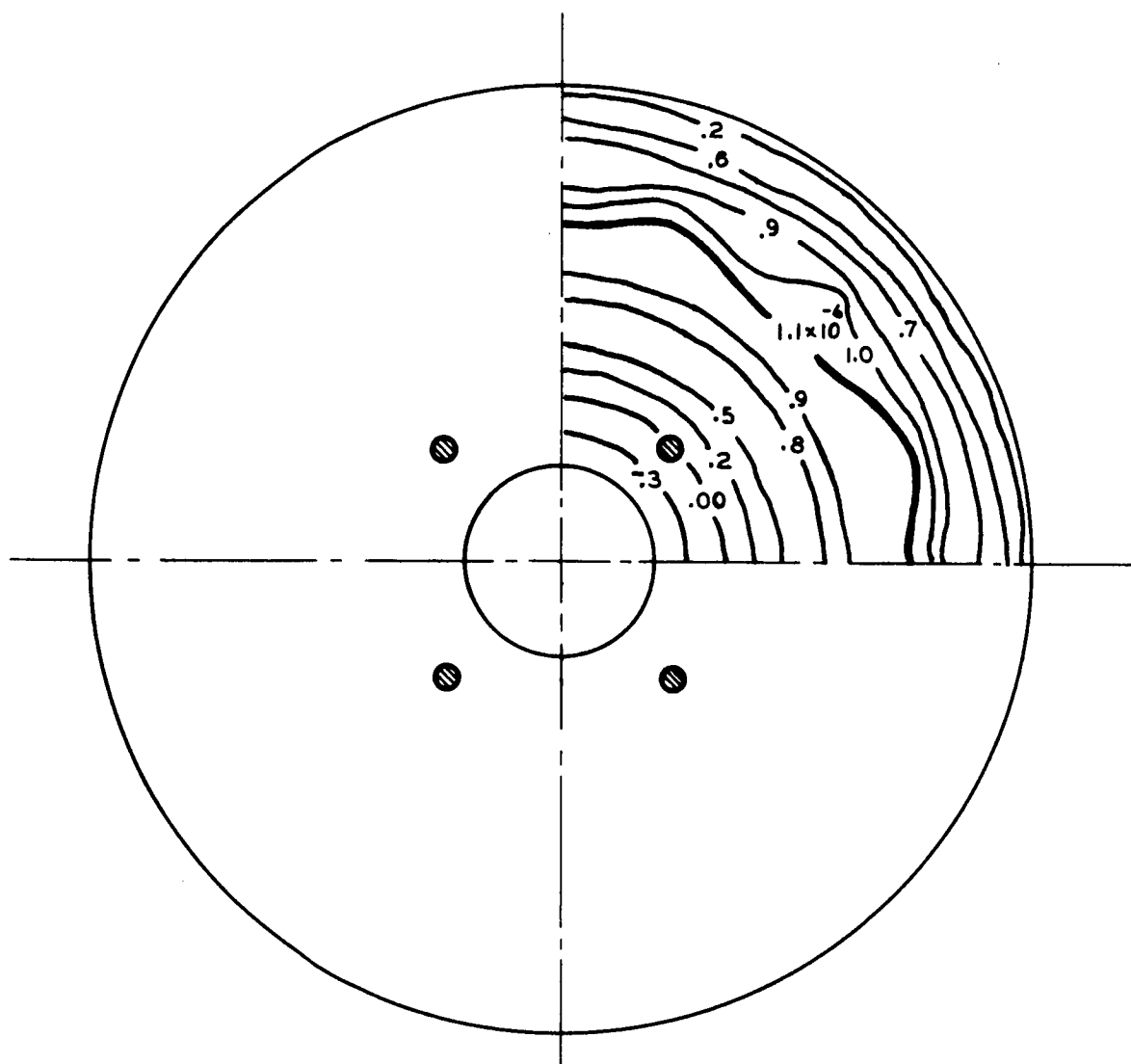


Figure 11.5.7-9 Schematic of 50" Mirror Showing
Actuator Locations - Case 7



⊗ ACTUATOR LOCATIONS

Figure 11.5.7-10 Deflection Contour Plot of Correction to Paraboloid Error Pattern - Case 2

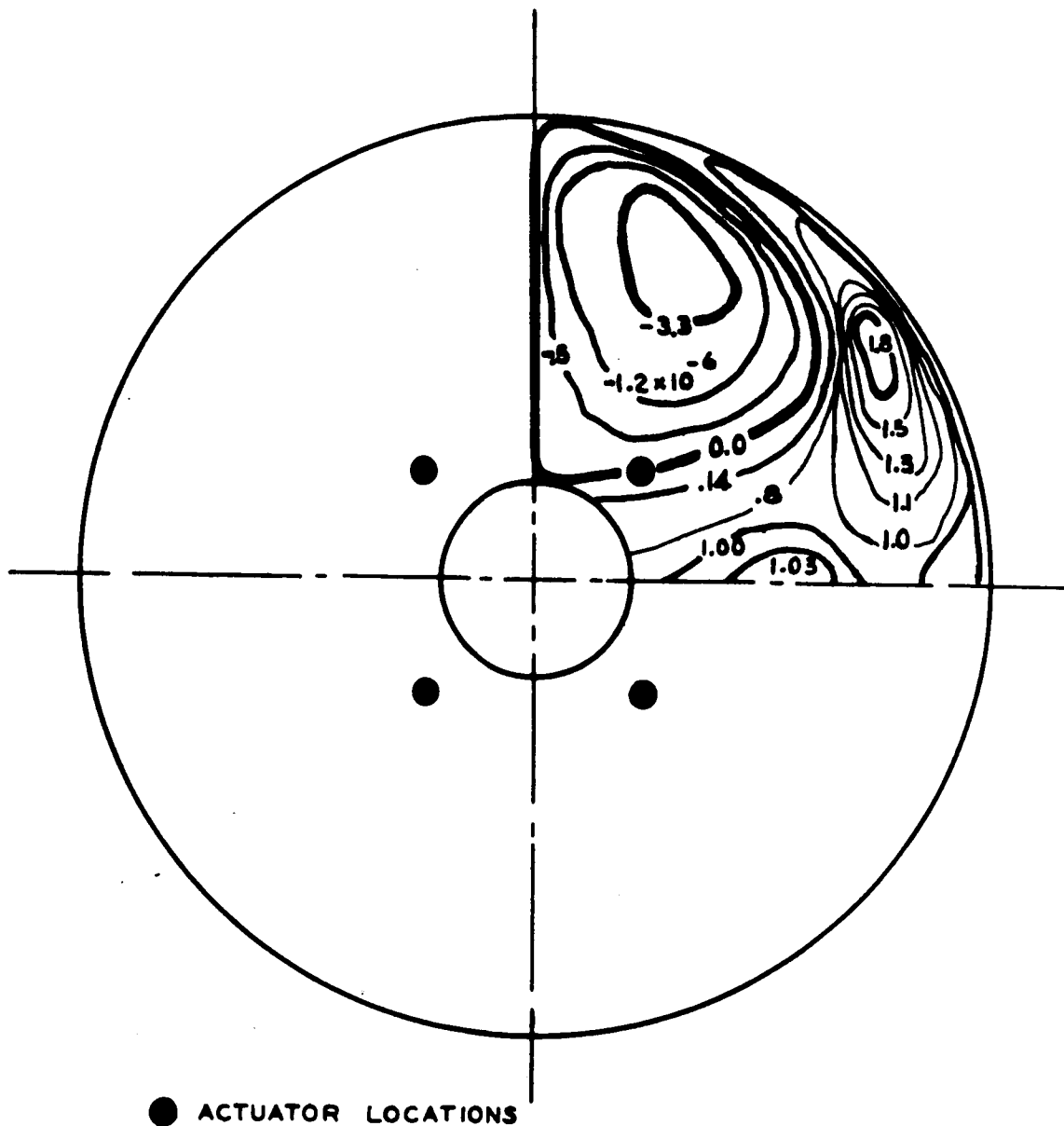


Figure 11.5.7-11 Deflection Contour Plot of Correction to Sinusoid Error Pattern - Case 2

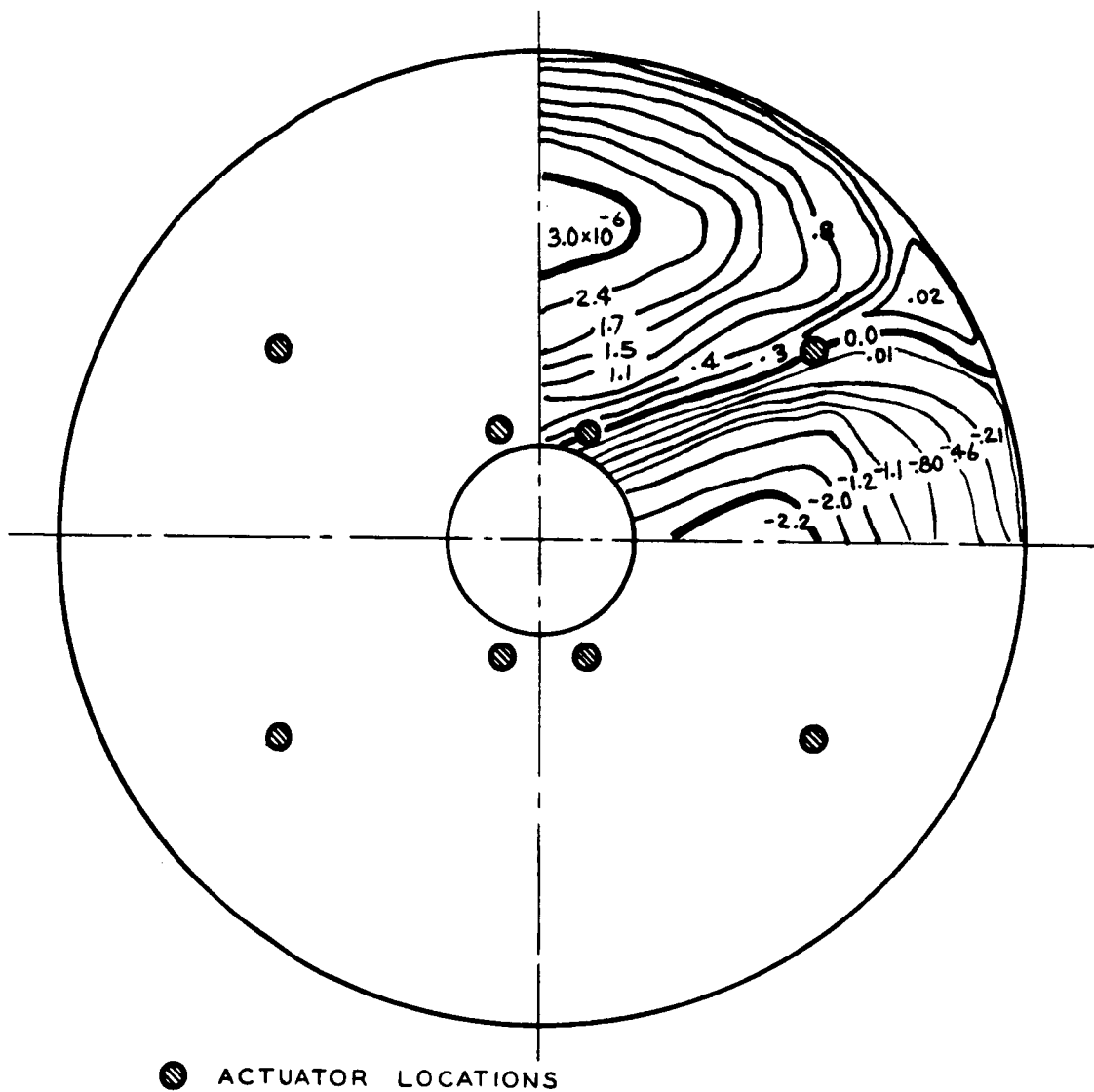


Figure 11.5.7-12 Deflection Contour Plot of Correction to Paraboloid Error Pattern - Case 5

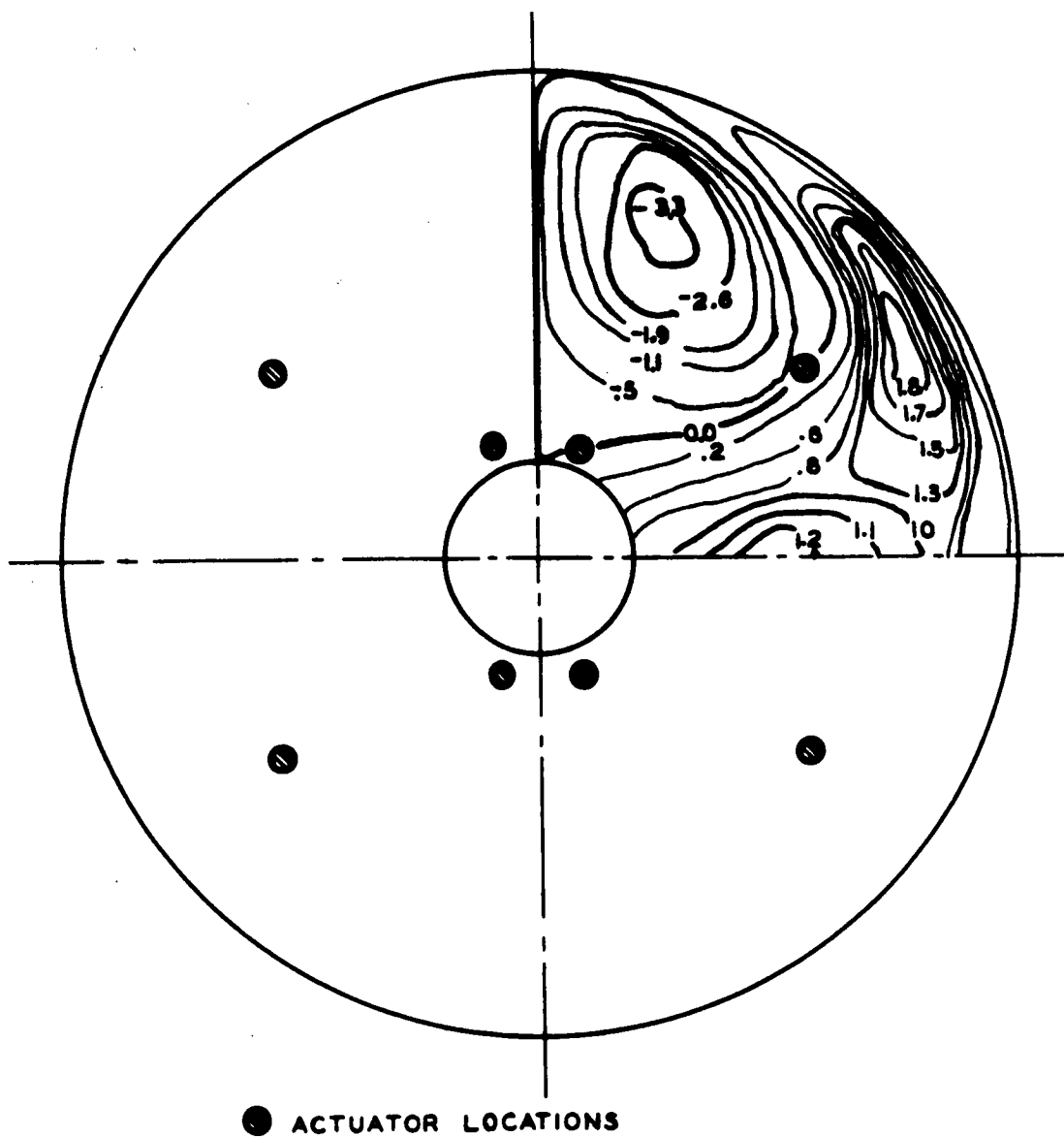


Figure 11.5.7-13 Deflection Contour Plot of Correction
to Sinusoid Error Pattern - Case 5

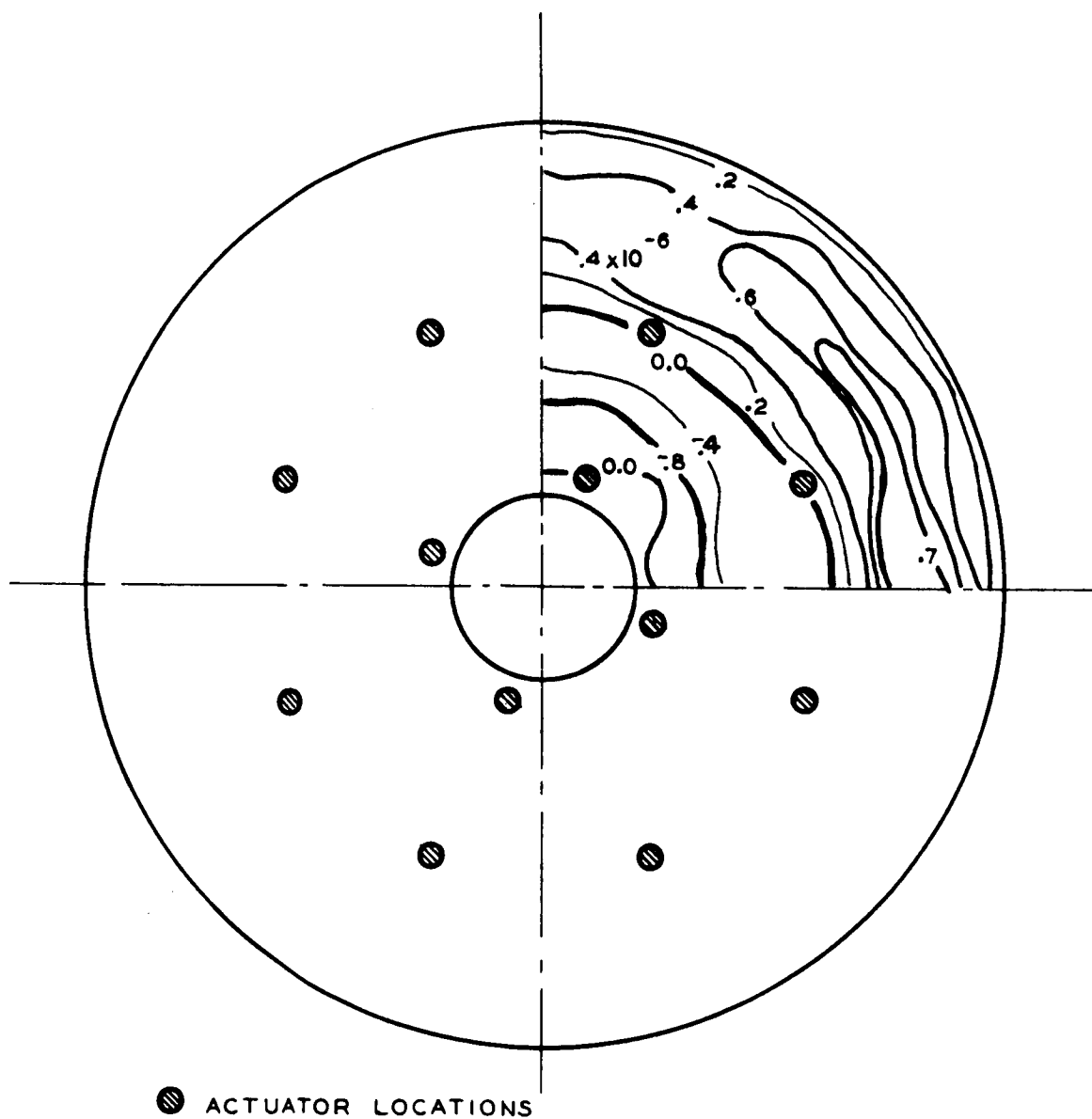


Figure 11.5.7-14 Deflection Contour Plot of Correction to
Paraboloid Error Pattern (Approx.)
- Case 7

11.5.7.5 Conclusions

As a result of the matrix inversion problem discussed in section 11.5.7.2, which delayed a more complete study, a limited number of error patterns and actuator arrays were considered. From these results certain conclusions and trends have been established, which encourage further work in this area.

For example, as the size of the actuator arrays become large, the solution for the actuator forces, using the relaxation method (Reference (2)) becomes sensitive to the number of iterations or successive approximations to the solution. That is, an increased number of iterations are required to converge on the solution for the actuator forces. This is indicated in tables 11.5.7-2 and 11.5.7-3, where the solutions for the actuator forces have not quite converged to the actual values. A study should be performed to establish convergence or other mathematical limitations of the computer analysis.

The results also indicate that as the error pattern becomes more complicated, such as an increased number of waves in the pattern, the amount of force required by the actuators will increase. Depending on the error pattern, this will also be the case if a large number of actuators are involved. It therefore seems plausible that mirror correction should be achieved with a minimum number of force actuators.

It has been established that the array of four actuators shown in figure 11.5.7-4 can correct the paraboloid error pattern to about $\lambda/30$.

Since the maximum positive deflection is 1.1×10^{-6} inch and maximum negative deflection is -0.4×10^{-6} inch, as shown in table 11.5.7-1, the maximum deflection in the plate can be further reduced by forcing the displacements at the actuator locations to values other than zero. An increase in the deflection of the actuator locations to 0.4×10^{-6} inch decreases the maximum deflection of the plate to about 0.9×10^{-6} inch. To ascertain whether the sinusoid-paraboloid error correction can be reduced from the value of -2.95×10^{-6} inch attained by case 1 of table 11.5.7-1, requires further investigation. Since the study was limited by development of a workable computer program, it is felt that additional work is necessary to evaluate the effects of other actuator patterns for mirror correction.

11.5.8 Optical Testing Procedures

11.5.8.1 Autocollimation

Autocollimation consists essentially of sending an image back on itself. The reflected image (crosshairs) will be just as thick as the actual crosshairs, since the magnification is 1:1. (They will usually appear to be slightly fainter, however, than the actual crosshairs.)

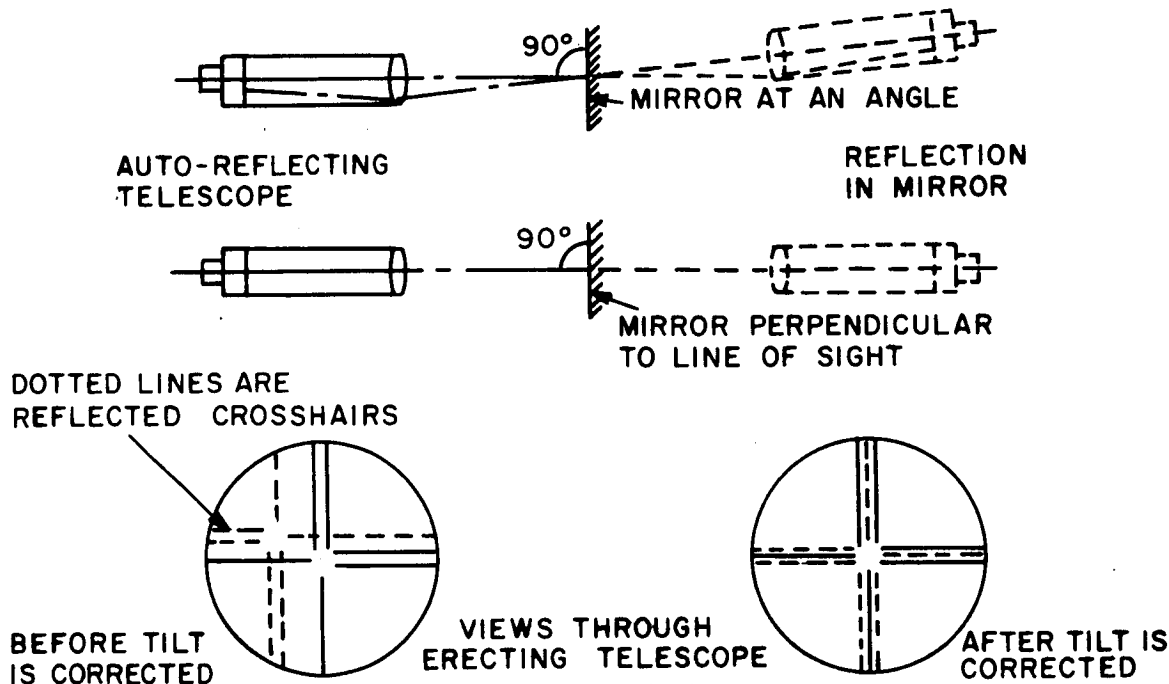


Figure 11.5.8-1 Autocollimation. The reflected image (crosshairs) will be seen as inverted

Figure 11.5.8-1 shows the principles involved in autocollimation. The crosshair pattern in the autocollimation telescope will be as indicated. The image which appears in the mirror is shown in dotted lines. To move the image downward, the astronaut must tip the mirror backward. The result of adjusting the mirror properly and the views seen through the telescope are also shown in the lower part of the figure. The image presented to the astronaut of the crosshairs will be inverted in relation to the actual crosshairs.

In the mirror experiment a collimated beam of light is used to project the image of crosshairs which is accurately located on the axis of the primary mirror's central aperture. This image is then aligned with the reticle crosshairs in the autocollimator eyepiece.

11.5.8.2 Foucault Test

Center of curvature evaluation for the mirror experiments will be accomplished by use of the Foucault knife-edge or shadow test. The following is a brief discussion of this test.

The astronaut will insert a sharp, opaque vertical edge in a beam of light close to the image plane. The astronaut's eye, positioned behind this edge (in the access well), views the mirror. Any internal displacement of the light rays from their normal position will be translated into a shadow appearing in the defective area. The astronaut will see a very striking image; the defects will appear in true relief, as though the surface were illuminated by grazing light with the defects amplified about a million times.

11.5.8.2.1 Nature of the Foucault Test

A light source is located within the radius of curvature of the mirror which reflects an image. If the source is positioned exactly at the center of curvature, the image formed by the reflected rays would coincide with the source. Therefore, the source must be slightly displaced to one side of the axis. Thus, the image will shift an equal amount in the opposite direction. The astronaut's eye will be positioned immediately behind the image, while the knife-edge will be inserted immediately in front of the eye. For the sake of orderly discussion assume that the source has been displaced to the left of the axis as the astronaut faces the mirror and that the knife-edge is inserted from the right.

11.5.8.2.2 Judgements and Image Interpretations

Case I - Perfectly Spherical Mirror

The laser source being close to the center of curvature will cause negligible aberration since all reflected rays will converge on the same reflected point. If the knife enters the beam in front of this point, the astronaut will see a shadow moving across the mirror in the same direction as the knife. If the knife enters behind this point the shadow appears at the left and moves in the opposite direction. If the knife enters exactly at that point of convergence every part of the mirror surface will contribute equally to the image and the entire mirror will appear uniformly dark. The change through the above image motions occurs gradually, and the astronaut can determine when the knife-edge enters exactly in the image plane. As the knife is inserted he need only to compare the brightness at the right and left-hand edges of the mirror. If the right is somewhat darker, he draws the knife back to a position slightly farther from the mirror. If the light is somewhat darker, the astronaut brings the knife forward. Using this trial and error method, the astronaut will arrive quite rapidly at the point of uniform darkening.

Case II - A Nonspherical Mirror

In this case the defects would appear as raised or depressed circular zones concentric with the rim of the mirror.

If the knife enters the beam at the convergence point of rays, a wide circular band or zone appears. This zone of the image accordingly appears to be a flat portion (of the relief map) to the astronaut. At the mirror edge, however, and at the center, are zones of somewhat larger radius of curvature. Rays reflected from these zones converge at a slightly different focus, and therefore cannot be entirely intercepted by the knife at the same time as rays from the intermediate zone. Rays originating from the mirror "slopes" turned toward the knife-edge are intercepted first. Thus, the astronaut would see these as the first regions to darken. By the same token, slopes inclined the other way would appear to remain lighted longest. In other words, the shadows suggest in relief the differences of height between the defective zones. The appearance may also be described as that of a molded surface seen under grazing illumination. The astronaut needs only to know whether the light is coming from the left or from the right to know whether any given defect is raised or depressed.

11.5.8.3 Scatterplate Interferometry

The procedure in establishing the center of curvature and obtaining interference patterns of a mirror requires meticulous accuracy. In establishing the center of curvature of the mirror the scatterplate is moved back and forth (micrometer adjustment) over the distance of a fraction of the wavelength of light.

If the astronaut has obtained a perfect sphere and has placed the scatterplate at the center of curvature of the mirror, no image will be seen. When the scatterplate is moved further away than the center of curvature, a series of concentric black circles will be seen. The farther from the center of curvature, the more circles will appear. As the scatterplate is brought back to the center of curvature, the concentric circles will disappear one by one until no circles appear at the center of curvature. When the scatterplate is moved to a point within the center of curvature, the order of appearance of the concentric circles will be reversed from that described above and the number of circles will increase the closer the astronaut moves the scatterplate to the mirror.

After the astronaut has established the center of curvature, he moves the scatterplate slightly off axis (one mil) and slightly tilts the scatterplate. The angle of tilt depends on the curvature of the mirror. The image now seen will be a series of vertical wavy lines as shown in figure 11.3.1-3. The astronaut will map the lines via a coordinate system and compare them to theoretically derived lines of a perfect mirror. The astronaut will map the lines by placing a standard linear coordinate grid over the obtained pattern. The vertical grid lines will be ruled while the horizontal lines will provide 100 specified comparison points/line.

Therefore, each wavy line obtained will be described by its linear distance from the vertical grid line over 100 specified points. The resulting numbers will be fed into a computer where each series of numbers (one hundred numbers/line) will be compared to the numbers derived from a theoretically perfect mirror. Thus a difference between theoretical and experimental configurations may be obtained.

12.0 INTEGRATED EXPERIMENT REQUIREMENTS

The purpose of this subsection is to present the integrated requirements and constraints imposed by the experiment modules upon the subsystems and spacecraft. The individual experiment requirements and their grouping into four telescope modules is presented in subsections 9.0, 10.0 and 11.0. The laser communication experiments (number 1 through 9) are housed in two telescope modules. The first module contains two gimballed telescopes, a 1-meter and a strapped 0.3-meter telescope, and the second module contains a gimballed 0.3-meter telescope. The primary mirror figure experiments (numbers 10 and 11) are mounted in a 1.525-meter telescope module and the fine guidance and isolation comparison experiments (numbers 12 and 13) are placed into a 1.702-meter telescope module.

The requirements of the experiments in each module are summarized in table 12.0-1. The table shows the requirements imposed by the modules on ~~some~~ spacecraft subsystems. Primary emphasis in this phase is placed on subsystem design for data management, electrical power and design integration. A detailed electrical power load profile appears in figure 25.2.1-1. The structural and mechanical design, guidance navigation and control, and reaction control subsystem requirements are directly paced by experiment development and their specific requirements will be developed during a parametric trade-off study in Phase B.

TABLE 12.0-1
INTEGRATED EXPERIMENT REQUIREMENTS

Telescope Module	Experiment Number	Experiment Name	Data Requirements			Total No. of Signals	Electrical Power Requirements (Watts)	Design Integration Requirements		
			No. of Signals Having Sampling Rate Greater Than 100 S/S	Between 2 S/S and 100 S/S	Between 1/20 S/S and 2 S/S	Less than 1/20 S/S	Standby (Watts)	Diameter (meter)	Length (meter)	Weight (kg)
1.0 meter gimballed and 0.3 meter strapped Laser Communication	1	Optical Heterodyne Detection on Earth	25	5	25	150	78.5	157		
	2	Optical Heterodyne Detection in Space	30	5	30	150	17	34		
	3	Direct Detection on Earth	25	10	40	60	78.5	157		
	4	Megahertz Optical Communication	-	-	-	-	88.5	177		
	5	Precision Tracking of a Ground Station	-	10	10	60	63.5	127		
	6	Point-ahead, and Space-Ground-Space Loop Closure	-	10	5	30	53.5	107		
	7	Transfer Tracking from One Ground Station to Another	-	5	5	25	53.5	107		
	8	10M Phase and Amplitude Correlation	15	5	10	45	78.5	157		
	9	Pulse Distortion Measurements	5	5	10	60	53.5	107		
Summary		Totals	100	55	135	780	870	1.42 } .68 }	3.56 } 2.24 }	1,189.77 } 233.15 }
0.3 meter gimballed laser Communication	2	Optical Heterodyne Detection in Space	30	5	30	150	17	34		
	3	Direct Detection on Earth	25	10	40	60	78.5	156		
	5	Precision Tracking of a Ground Station	-	10	10	60	63.5	127		
Summary		Totals	55	25	80	270	430			

TABLE 12.0-1 (CONTINUED)
INTEGRATED EXPERIMENT REQUIREMENTS

Telescope Module	Experiment Number	Experiment Name	No. of Signals Greater than 100 S/S	Data Requirements No. of Signals Having Sampling Between 2 S/S and 100 S/S	Requirements: Less than 1/20 S/S	Total No. of Signals	Electrical Power Requirements Standby (Watts)	Operational (Watts)	Diameter (meter)	Length (meter)	Weight (kg)
Primary Mirror Figure	10	Primary Mirror Figure Test and Correction	-	-	5	60	39	78			
	11	Thin Mirror Nesting Verification	-	-	5	100	33	66			
Summary		Totals	-	-	10	160			1.52	4.07	758.40
Fine Guidance	12	Fine Guidance	-	10	5	80	62.5	125			
	13	Telescope Isolation	-	25	30	10	32.5	65			
Summary		Totals	-	35	35	90			1.6 middle .87 ends	1.53 3.50	956.17

13.0 OTHER EXPERIMENTS

This section presents those experiments considered in addition to the thirteen "hard core" experiments previously discussed.

13.1 DISCARDED EXPERIMENTS

13.1.1 Experiments Discarded During Part I

During Part I of this contract some twenty-four experiments were initially proposed. Nine of these were discarded as not requiring space testing. These experiments are discussed in detail in the part I final report in subsection 5.0, Discarded OTAES Experiments.

These experiments, referenced to the OTAES Part I report section numbers, are:

- 5.1 Spectrograph development
- 5.2 Baffle systems comparison
- 5.3 Atmospheric absorption spectroscopy
- 5.4 Photon - Photon Scattering
- 5.5 Remote Manual Optical Alignment
- 5.6 Visual Tracking Rating
- 5.7 Cryogenic cooling
- 5.8 Photo-electro-optical experiment
- 5.9 Mirror coating

13.1.2 Experiments Discarded During Part II

One further experiment was discarded during Part II of this program. This was 4.14, Interferometer System. After further study it was concluded that space testing of this sytem could not be justified, and that the experiment should consequently be discarded.

13.2 NEW EXPERIMENTS

During Part II of the OTAES study, a numer of new experiments have been conceptually studied. In addition, one experiment from Part I is still in the new experiment category. This is experiment 4.15, Segmented Optics.

13.2.1 Segmented Optics

This experiment is not considered as one of the thirteen "hard core" experiments; however, because of its potential as a technique for the placing of large optical systems in space, it is considered to be a very strong candidate. This experiment is described in detail in the OTAES Part I final report, subsection 4.15.

13.2.2 Space Radiation Shielding Study

13.2.2.1 Summary

The most easily damaged material to be carried into space is photographic film, some of which is completely fogged by doses of few rads. According to the acceptable radiation tolerances and the specific application of the film, radiation shielding must be provided. Shielding will modify the radiation environment. Structural and inherent shielding will influence the effects of radiation. In some cases shielding eliminates damage problems by absorbing most of the incident radiation. In other cases, however, complications will arise from the creation of penetrating secondary radiation or from the degradation of energy to a level that is even more sensitive in causing damage.

In space there are protons and electrons of a wide range of energies and intensities. This environment cannot be directly simulated in a laboratory. There is even difficulty in simulating a certain portion of the radiation spectrum. Only a limited number of high energy proton accelerators are available.

The purpose of this experiment in space is to determine photographic effects of space radiation, dosage levels, particle depth of penetration and types of shielding required. Flight results will be correlated with ground radiation studies.

13.2.2.2 Contribution and Need

To our knowledge, there is only scant experimental information regarding the radiation tolerances as a function of particle type and energy for space film. Furthermore, laboratory experiments have been performed by irradiating film samples with one type of incident radiation, not with a multiple type of particle beam such as may be encountered in space missions. Recording of the acceptable radiation tolerances and the specific application of the film radiation shielding must be provided. Structural and inherent shielding will influence the effects of radiation. In some cases shielding eliminates damage problems by absorbing most of the incident radiation. In other cases, however, complications will arise from the creation of penetrating secondary radiation or from the degradation of energy to a level that is even more sensitive in causing damage. Shielding materials must therefore be selected on the basis of their attenuation properties, if possible, their minimum bremsstrahlung and characteristic x-ray emission.

Shield weight penalty associated with space radiation for long missions is severe enough to preclude or restrict the use of photographic film. To reduce or eliminate the shield weight penalty, advantage should also be taken by other possible modes of protection such as controlled temperature and humidity environmental storage.

The purpose of these experiments in space is to determine:

- a. Film sensitivity to ionizing radiation.
- b. Total exposure level during a mission.
- c. Particle depth of penetration and types of shielding required to reduce exposure dosage to an acceptable level.
- d. Test alternate modes of protection.

13.2.2.3 Need for Space Testing

In space there are protons and electrons of a wide range of energies and intensities. Radiation effects studies are at present made in research facilities that cannot simulate the space environment. First, there is the problem of duplicating the range and intensity of multiple types of radiation. Second, only a limited number of high energy accelerators are available. Furthermore, research facilities are not usually available for long-time testing. However, in usual laboratory work, the procedure is to irradiate a sample with either electrons or protons, not both at the same time. The difficulty is that two or more radiation-producing machines are needed to make different types of particle flow simultaneously toward a target.

An experimental evaluation of the space radiation effects upon a photographic emulsion and consequently determination of the protection requirements is the only way to ascertain that a stream of both electrons and protons hitting a photographic emulsion does not have an effect different from a stream of only one kind of particle. In this experiment, the results from detailed analysis of the radiation environment, the effect of this environment upon film emulsion, the modifications of this environment by shielding or other modes of protection will be applied to predict performance of photographic film for future applications.

13.2.3 Experiments on Space Myopia

13.2.3.1 Summary

Missions in the manned orbital telescope require astronauts to align and focus optical instruments and perform photography under a variety of dim-light conditions. One of the major psychological factors decreasing the accuracy of these operations during flight will be space myopia. Therefore, the proposed experiments are to analyze the environmental factors contributing to space myopia. By conducting these experiments, factors affecting the accuracy of operations of optical instruments will be assessed.

13.2.3.2 Contribution and Need

Previous research on airplane accidents of World War II and of commercial airliners indicates that the high rate of pilot errors is associated with

visual distortion, specifically space myopia. This type of myopia was also noticed in underwater photographers, during the experiments of Sea Lab II in 1965. However, quantitative analysis of space myopia in relation to the accuracy of performance has not been thoroughly studied.

Since the manned orbital telescope requires high accuracy of optical instrument operation, it is very important to investigate space myopia as a function of accuracy of operation during space flight. Furthermore, the longer the duration of flight the more the tendency to develop space myopia. Also, space myopia affects the degradation of other visual capabilities, such as visual acuity, distance judgment, legibility and readability. Therefore, the psychological experiments on space myopia during orbital flight become significant. Conducting such experiments on space myopia will help to remedy the space myopia temporarily, improve environmental factors affecting space myopia, and devise a new optical instrument. This would result in the precision of space performance in a manned orbital telescope.

13.2.3.3 Need for Space Experiment

In space flight there are two major environmental factors affecting space myopia. These are empty space field and weightlessness. These environmental factors increase the convexity of the lens of the eye which is the major factor affecting space myopia. The effect of empty space field on space myopia has been well studied under plane flight conditions. However, effects of weightlessness on space myopia has not been studied.

Space myopia is usually developed under one or more of the following conditions:

- a. looking at relatively distant objects
- b. infinite visual field
- c. fixating at near distant objects within a spacecraft

The eye has a tendency, when looking into space with no concrete objects on which to fixate, to automatically focus only a few feet in front of the individual. Also during space flight, the eye will be constantly fixating on instruments located at short distances, and it will automatically focus only a few feet in front of the individual. The eye can focus an object by changing the convexity and strength of the lens. This result, called accommodation, is accomplished by the ciliary muscles. If the object is more than six feet, the muscle is relaxed. If the object comes nearer than six feet, the muscle contracts more. As a result, the lens becomes more convex. As mentioned before, space myopia affects the degradation of other visual capabilities, such as visual acuity, distance judgment, legibility and readability. The longer the duration of flight, the more the tendency to develop space myopia. Therefore, the present proposed experiments are to be performed during space flights.

13.2.3.4 Experiment Description

The proposed experiment is to measure the magnitude of space myopia in a short time during space flight. The degree of myopia is normally measured by the Vernier Acuity Test, Snellen Eye Chart, Landolt C (broken lines). For space flight, however, a Simple Flip Eye Chart will be designed to measure the magnitude of the space myopia in a short time. This will be employed along with the modification of Vernier Acuity Test in which crews can judge whether two halves of a line are continuous or not.

13.2.4 Residual Atmosphere Scattering Experiments

13.2.4.1 Summary

Under the low pressure conditions and in the zero gravity condition of earth orbit, outgassing will occur and particulate matter will "float" in the vicinity of the spacecraft. The particulate matter may also contain residue from cold gas maneuvering thrusters, where insufficient velocity had been imparted to cause the material to leave the vicinity of the vehicle. During manned flights, astronauts have remarked on the appearance of material that "glinted" in the vicinity of the capsule at sunrise. The combination of the astronaut's vision being dark adapted and the particles being in the sunlight before the capsule entered the sunlight provided optimum opportunity for noticing this "contamination."

In the course of deep space astronomy from an orbiting telescope, the retention of outgassing products and particulate matter may become a serious problem. Sunlight may be scattered into the telescope causing an increase in the apparent background of the object under study.

At this time, the effect is only suspect and measurements need to be made to determine the nature and extent of the cloud of material in the vicinity of the spaceborne telescope. The measurements need to be taken over an extended time period to determine if the cloud is dissipated or compacted around the craft with the passage of time. Measurements need to be made in the vicinity of several large spacecraft, where different thrusting and stabilizing techniques are employed to aid in gathering data on the nature of the spacecraft cloud and to evaluate the problems that may be caused by the use of certain types of thrusters.

A light source radiating into space will be used to illuminate the scatterers (figure 13.2-1) and a receiver that is located some distance away will scan the beam to gather data at various wavelengths.

13.2.4.2 Experiment Objective

The objective of this experiment is to determine, by indirect measurement, the amount of sunlight that may be scattered in a telescope by material retained in the vicinity of the spacecraft and to determine the variation of scattered light with time. A further objective will be to determine the characteristics of the material causing the scattering.

Particulate matter and gas molecules carried in space in the vicinity of the spacecraft may become a serious limitation to deep space astronomy from above the earth's atmosphere. The scattering effect of this residual atmosphere must be measured in space to enable astronomers to estimate its effect on deep space astronomy. An intense light source illuminating a defined volume of this atmosphere will provide sufficient scattered light to a scanning receiver so that adequate data may be obtained for evaluation.

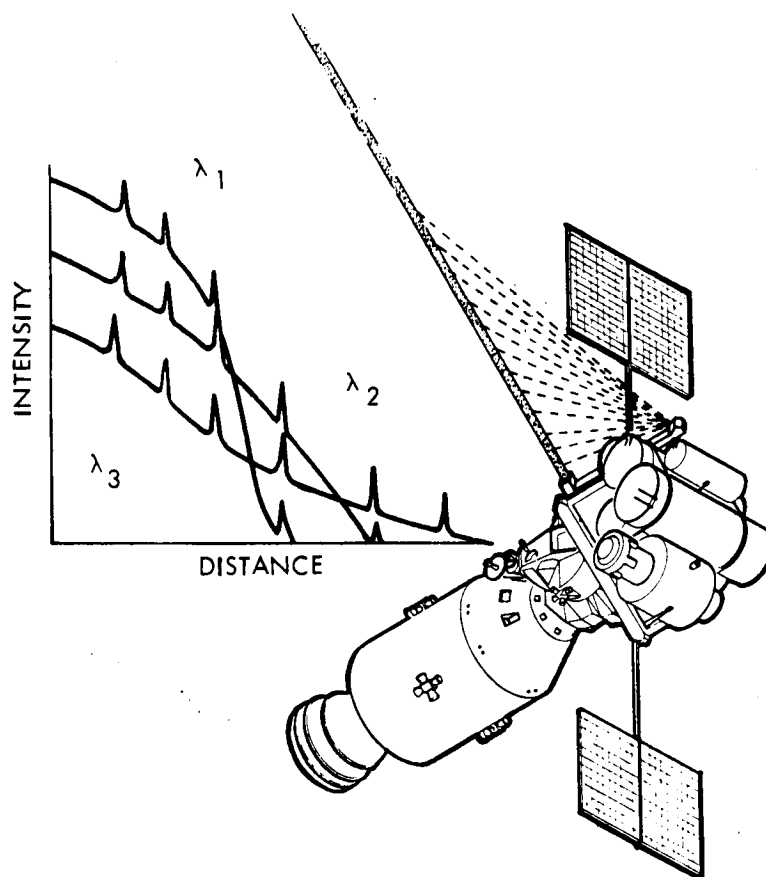


Figure 13.2-1. Residual Atmosphere Scattering Experiment.

13.2.4.3 Experiment Justification

13.2.4.3.1 Contribution and Need

The study of deep space objects from ground-based observatories is limited by several factors. One of the most critical is the effect of the instability, scattering, absorption of the atmosphere. The profound effect of scattering of sunlight restricts astronomical observation of stars to nighttime conditions and in locations away from large cities. Placing a telescope in a near-earth orbit will greatly reduce these effects because of the very tenuous atmosphere at orbital altitudes. Turbulence and absorption will be nonexistent, but scattering may still be a matter of concern for the following reasons.

Approximately 70 percent of the observing time of a near-earth orbiting telescope will be spent in the direct rays of the sun. Because of the short "night" of the telescope (70 minutes maximum at synchronous earth orbit), observations must be made during daylight as well as when the telescope is in the earth's shadow. The presence of scattering material around the vehicle will cause the sunlight to be scattered into the telescope (even though the sunshade operates properly).

Scattering can be divided into three general categories that are a function of the size of the scatterer.

- a. Rayleigh Scattering - where the scatterer's largest dimension is much less than the wavelength of incident light.
- b. Mie Scattering - where the scatterer's largest dimension is of the same order of magnitude as the wavelength of incident light.
- c. Particulate Scattering - where the scatterer's largest dimension is much greater than the wavelength of the incident light.

Theories have been developed based on special concepts of the scatterer's physical shape that enable computation of the amount of energy that can be scattered in a given direction. For Rayleigh scattering, where the scatterers are considered to be atoms or molecules of the predominant gases, oxygen and nitrogen, and spherical in shape, the computation can be made to determine the amount of sunlight that could be scattered into the open end of the telescope. For Rayleigh scattering, the variation of energy with scattering angle is small and may, for the purpose of this discussion, be considered constant; that is, the angle formed by the sun to scatterer and scatterer to telescope does not significantly affect the energy scattered into the telescope.

For Mie and particulate scattering, the effect of the angle from the sun to the particle and the telescope is variable and must be taken into consideration in determining the total scattered energy into the telescope.

The total energy scattered into the telescope over a period of time may exceed the total energy collected by the telescope for a star of 20th magnitude. The energy of a type A₀ star of 20th magnitude is $\approx 6.3 \times 10^{-17}$ ergs/sec/cm²/Å; and over the 2000 Å band between 0.4 μ and 0.6 μ, this gives 12.6×10^{-14} ergs/sec/cm². Solar energy at the earth distance above the atmosphere is 1.38×10^6 ergs/sec/cm². Approximately 27 percent of this energy lies between 0.4 μ and 0.6 μ, giving a 0.35×10^6 ergs/sec/cm².

Therefore, if the energy scattered into the telescope is $\frac{12.6 \times 10^{-14}}{0.35 \times 10^6} = 3.6 \times 10^{-19}$ times the sun's energy, the signal-to-noise ratio will be one.

13.2.4.3.2 Need for Space Testing

The purpose of this experiment is to determine the effect of scatterers in the vicinity of a spacecraft in orbit around the earth. The conditions that will affect these scatterers is the zero gravity, the very low atmospheric pressure, outgassing and thruster by-products, and the dilution of the scatterers with time. The same conditions cannot be adequately and simultaneously simulated on earth.

13.2.4.4 Active Contamination Measurements

13.2.4.4.1 Backscatter Experiment

An experiment for gathering data on scattering material must be able to differentiate the light scattered from nearby material and the light from distant sources such as the stars, moon, and earth. It should also be capable of providing distance and Nσ product (number of scatterers x scattering cross section) of the scatterers for a period of several months. More data on the nature of the scatterers could be obtained if a light source having a bandwidth of 0.3 μ to 0.7 μ were used to illuminate the cloud and measurements were made at several points in this bandwidth.

An intense source of light will be used to obtain sufficient scattered energy. To conserve power and to aid in discrimination of distant sources, a pulsed white light source such as a strobe lamp will be used, with the detector gated on only during the strobe period. Filters having bandwidths of 100 Å will provide the desired samples of the spectrum from 3000 Å to 7000 Å. The output of the detector will be quantized and relayed to the ground for evaluation, with the filter parameters and detector look angle.

13.2.4.4.1.1 Alternatives to a Spaceborne Experiment

The experiment was conceived, based on the need to make measurements of particles having random motion in the vicinity of the spacecraft and with velocity vectors equal to that of the spacecraft. With independent motion

of the scatters, there is no basis for correlation of their motion with that of the spacecraft. Similarly, there is no basis for estimating motion in a 1 g field on earth.

The vacuum of space can be simulated in a vacuum chamber, and measurements can be made of returns due to Rayleigh scattering. However, larger particles will be subject to the force of gravity and will not float in the chamber. Thrusters may be operated in the chamber to gather data on the scattering characteristics of their exhaust products. These characteristics would be useful in identifying returns obtained from spaceborne tests.

A zero gravity condition can be simulated by an aircraft flying a parabolic path for a 30-second time period. This time would not be sufficient to determine dispersion characteristics of the particles or to obtain a distribution of gravity-affected particles. Inclusion of a high-vacuum chamber in an aircraft flying a zero gravity path does not seem to be practical.

There is no alternative to a spaceborne experiment, but useful data can be obtained by preliminary ground tests.

13.2.4.4.1.2 Experiment Design

13.2.4.4.1.2.1 Experiment Measurement Objective

The purpose of the experiment is to measure the amount of light scattered by materials that are retained in the vicinity of the spacecraft. The measurement of scattered light from a source on the spacecraft will be analyzed to deduce the amount of sunlight that would be scattered into the open end of a telescope by the residual atmosphere of the spacecraft.

13.2.4.4.1.2.2 Experiment Equipment Design

This experiment is comprised of two functional pieces of equipment, a transmitter, and a receiver. The concept of the transmitter and receiver is shown in figure 13.2-2. The arrangement of transmitter and receiver provides a scattering output that will be similar to the scattering of sunlight into a telescope aperture. This may be directly interpreted to give results of expected solar energy into the telescope.

The transmitter has a strobe lamp illuminating a parabolic reflector, providing a beam having minimum divergence. The reflector will be as large as is reasonable, about 0.3 meters in diameter, to illuminate as much volume as possible with a "cylindrical" beam. The transmitter lamp and reflector will be fastened firmly to the structure and pointed perpendicular to the outer surface of the spacecraft. The cylindrical beam is then positively located with respect to the spacecraft.

Since the strobe lamp is not a point source, but is a distributed source, the quality of the reflector may be considerably reduced from diffraction-

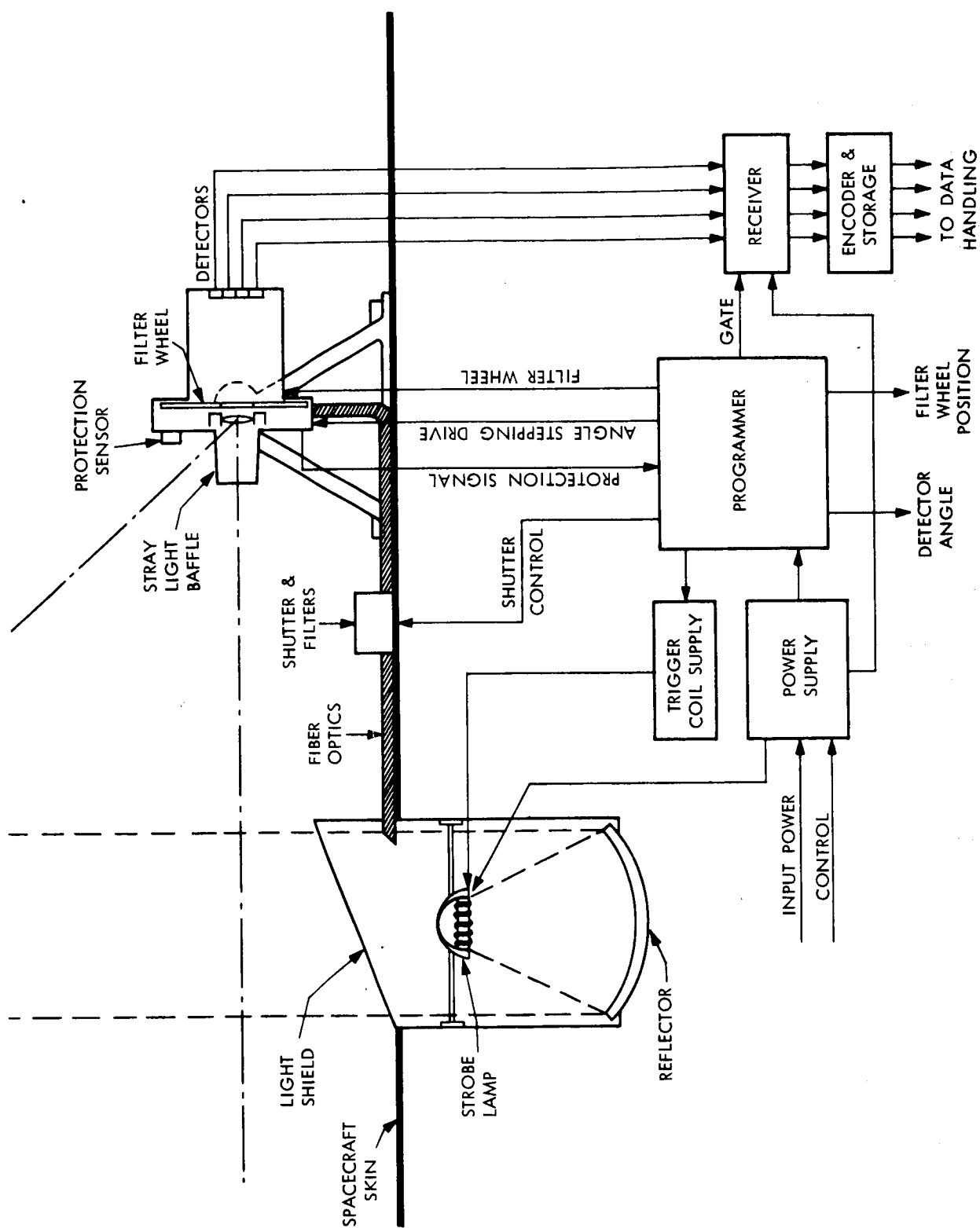


Figure 13.2-2. Residual Atmosphere Experiment Block Diagram.

limited quality without impairing performance. The lamp will be "fired" at 6-second intervals, and electronic synchronizing equipment will turn on the receiver just as the lamp is fired. The receiver will remain on for less than 10 microseconds after the strobe pulse has fallen to 10 percent of peak value. This will minimize the effect of any background illumination.

The receiver will be mounted outside the spacecraft skin and will be pivoted so that it can scan in steps along the beam to a distance of about 5 meters from the spacecraft. The aperture for the receiver will be selected to provide as great a reduction in image size with maximum aperture size. A filter wheel will be located behind the receiver aperture. It will have nine filters plus an opaque calibration aperture. Several detectors will be located in a row at the focus of the receiver optics. The row of detectors will be parallel to the transmitter beam. Each element will collect energy from that section of the beam imaged on it. With this technique, energy from several adjacent lengths of the beam can be simultaneously detected and recorded. Proper aperturing of each detector will minimize the collection of background radiation. There will be between 5 and 10 steps required to examine the length of the beam out to 5 meters. Each detector will have its own amplifier and signal encoder.

The filter wheel will be stepped after each pulse. When the opaque section is in front of the receiver aperture, the scattered light will be shut out, and the shutter in the fiber optics path will be opened. When the strobe lamp is fired, light will be piped through fiber optics and into a diffusing disk behind the opaque section. The light level will be adjusted to illuminate the detectors at about the level that they will be illuminated when the receiver is pointing at that part of the beam nearest the spacecraft.

If the field of view at any time includes direct energy from the sun, or reflected energy from the earth or moon, it will be necessary to protect the receiver from this energy. To provide a signal for this protection, a photocell will be mounted on the receiver, having a field of view twice as large as the detector subtense and aligned with the receiver optical axis. When the output exceeds a preset value, a signal is supplied to the programmer to drive the filter wheel to the calibrate position. Until the protection signal falls below the threshold, all equipment is kept in the standby mode.

13.2.4.4.1.2.3 Data Management

The interpretation of received data will require recording of the angular position of the receiver aperture, the output of each detector, the transmitter light output, and the filter in the optical path. With the transmitter providing a cylindrical beam having very small divergence, the energy in the beam will be nearly a constant value out to the maximum range of 5 meters. The arrangement of receiver and transmitter will, therefore, cause a maximum of 7.5 to 1 variation of energy out to 5 meters for a given particle size. The variation of energy with particle size is expected to

have a range of 10^6 , giving a total dynamic range requirement of 1×10^7 for the receiver, detector, amplifiers, and encoders. Since the amount of energy received must be greater than 9.5×10^6 photons/second to equal the sky background that provides a limiting magnitude of -22.2 per square seconds of arc, then the threshold may be set at that level and the range will determine the upper level.

After each strobe pulse, the data-handling system must sample and store each detector output. The encoders will be reset to zero and be ready for the next pulse.

13.2.4.4.1.2.4 Operational Procedures

This experiment will be nearly automatic in operation, requiring only that the power be applied, either from the astronaut's control panel or by command from the ground. When power is supplied, the programmer will control all operations. This will include operating the flash lamp at a pre-set rate, changing the angular position of the receiver in the desired sequence, positioning the filter wheel, and providing for calibration of the receiver with the strobe lamp output once for every cycle of the filter wheel. With this calibration, all receiver outputs will be normalized independent of light output of the lamp or variations of the detector sensitivity.

The programmer will also provide gate control for the detector output, permitting the amplifiers to receive signals only during the lamp output pulse, and control the readout to the time immediately following the gate pulse. The timing sequence for the filter wheel and receiver angular position is shown in figure 13.2-3.

Although man is not needed for operation of this experiment, his presence and extra-vehicular activity will form an important part of the experiment. Valuable data will be obtained regarding the extent and duration of contamination caused by man, the venting of the air locks, and material carried and released by his suit during EVA.

13.2.4.4.1.2.5 In-Flight Checkout Design

The transmitter assembly will be mounted below the surface of the spacecraft skin and will be ready for operation without any setup activity. The receiver will be mounted above the spacecraft surface for operational reasons and will also be ready without setup activity unless space limitations within the shroud requires that the receiver be demountable.

The only requirement for checkout will be to observe the performance of the calibration operation. Observation of power consumption will provide assurance of lamp operation.

The strobe lamp will be operated well below its rated power output and will provide over a thousand hours of life. However, it is a wear-out item that may need to be replaced.

RECVR. POSIT. #	FILTER	FILTER WHEEL POSITION	FLASH
	.30 - .31 μ	1	1
	.35 - .36 μ	2	2
	.40 - .41 μ	3	3
	.45 - .46 μ	4	4
	.50 - .51 μ	5	5
	.55 - .56 μ	6	6
	.60 - .61 μ	7	7
	.65 - .66 μ	8	8
	.70 - .71 μ	9	9
	CALIBRATE	10	10
	CALIBRATE DARK		NO FLASH

REPEAT FOR POSITIONS 2 - 10



Figure 13.2-3. Residual Atmosphere experiment Timing Sequence.

13.2.4.4.1.2.6 Supporting Subsystem Requirements

The equipment is self-contained and requires only prime power at spacecraft voltage for operation. The total average power consumed will be about 15 watts. Ten watts will be used by the lamp based on 10 pulses per minute with 40 joules per pulse. The receiver and data-handling circuits will use 5 watts.

13.2.4.4.2 Hologram Experiment

Both particle size and distribution in the atmosphere accompanying the spacecraft may be determined by applying hologram techniques.⁽¹⁾ Essentially a small volume is sampled at a time. The small volume is sampled when a Q-switched laser beam, diverging from a pinhole, is used to illuminate the particles in the sample. On the optics axis is a camera lens that receives the laser light with a film negative and shutter, all of which are housed in a container.

This hologram camera is deployed outside the spacecraft. However, experiment design may allow a mirror to be deployed so that camera may be located at a more convenient position. While holography is essentially a lensless technique, the lens is used to provide magnification of the fog particles. A series of frames may be recorded as the camera is moved to view various sample volumes.

After return to earth, the holograms of the particles are used to provide a means to reconstruct the sample volume of particles. Reconstruction occurs when the hologram is illuminated by a helium-neon gas laser and collimator. The film is transported on a motor-driven carriage along the axis of the collimated beam. Using a television screen for display, the reconstructed images are imaged onto a vidicon camera in a closed circuit television monitor. Thus the hologram reconstructor is used to scan and size the particles in the hologram frames.

13.2.4.5 Passive Contamination Measurement

In addition to the two active experiments proposed, a third experiment which makes use of solar radiation scattered into the OTAES telescopes is also suggested. There are a number of considerations to be borne in mind in this experiment.

(1) Brian O. Thompson, John H. Ward and William R. Zinky, "Application of Hologram Techniques for Particle Size Analysis," Applied Optics, (New York, March 1967), Vol. 6, No. 3, pp. 519.

- a. We are interested only in those particles which will scatter radiation into the telescope. Particles within the telescope tube which are not illuminated will not concern us here.
- b. As much information as possible should be given about the spatial distribution of such contaminants.
- c. An experimental system should be as simple as possible.
- d. In the event of a cloud of vapor being in the field of view, we should attempt to get information about this if possible.

13.2.4.5.1 Experiment Description

Given a telescope focused at infinity, the image of the star field will be formed at the focal plane of the lens. The cloud of contaminant of interest is localized near the entrance aperture of the telescope. It is possible to add a second lens at the focal plane of a power sufficient to permit the telescope/lens combination to focus close to this aperture. In this case, a second focal plane is obtained at the rear of the first, which will image the light scattered from the contaminants. Starlight will not be focused at this plane, since the second or field lens will not appreciably affect the rays emanating from the stars which are focused at the primary focal plane. Thus this light will appear as a more or less uniform background at the secondary focal plane.

If we now introduce a spatial filter in the secondary focal plane with some detector (photomultiplier tube) behind it, and oscillate this in the plane of the filter, we will modulate the light from the contaminant source, and the signal out of the photomultiplier will be a function of the concentration of contaminants. By moving the secondary lens through a small axial distance, or by using a zoom type system, it is possible to change the position of the area of the contaminant cloud under observation, thus enabling the measurement of the spatial distribution of contaminants.

13.2.4.6 Supporting Analyses

13.2.4.6.1 Energy Scattered from Oxygen and Nitrogen Molecules

The expression for Rayleigh-Gans scattering⁽²⁾ for natural light assuming spherical particles is:

$$I_s = I_o \frac{k^4 V^2 |m - 1|^2}{8\pi^2 r^2} \cdot G(2 \times \sin \frac{1}{2} \theta) \cdot (1 + \cos^2 \theta)$$

(2) Van De Hulst, H. C., Light Scattering by Small Particles, p. 89.

where

$$k = \frac{2\pi}{\lambda}; \quad x = \frac{2\pi a}{\lambda}$$

$$a = \text{radius of scatterer} = \frac{x\lambda}{2\pi}$$

$$r = \text{distance from particle to point of measurement}$$

$$V = \text{volume of scatterer} = \frac{4}{3}\pi a^3 = \frac{4}{3}\pi \left(\frac{x\lambda}{2\pi}\right)^3$$

$$\theta = \text{angle formed by direction of propagation of light, scatterer, and the observer}$$

$$G(2x \sin \frac{1}{2}\theta) = \text{Bessel function of } \frac{3}{2} \text{ order divided by the first term of the series}$$

$$m = \text{index of refraction} \approx 1.33$$

$$I_s = I_o (5.3 \times 10^{-3}) \left(\frac{x^6 \lambda^2}{r^2}\right) |m - 1|^2 \cdot G(2x \sin \frac{1}{2}\theta) \cdot (1 + \cos^2 \theta).$$

When x is smaller than 0.5, then $G(2x \sin \frac{1}{2}\theta)$ is greater than 0.9 and will be assumed to be 1. The diameter⁽³⁾ of oxygen and nitrogen molecules are:

$$O_D = 1.5 \times 10^{-10} \text{ m}$$

$$N_D = 1.6 \times 10^{-10} \text{ m}$$

$$\text{Then, } x = \frac{1 \times 10^{-9}}{\lambda} \text{ m for oxygen or nitrogen}$$

$$\therefore I_s = I_o \frac{5.3 \times 10^{-58}}{\lambda^4 r^2} (1 + \cos^2 \theta).$$

$$\text{at } \lambda = 0.5 \times 10^{-6}, \quad \theta = 90^\circ$$

$$I_s = \frac{0.85 \times 10^{-32}}{r^2} \quad I_o = \text{energy scattered per molecule}$$

where r is in meters.

(3) Handbook of Chemistry and Physics, Chemical Rubber Publishing Co., 44th Edition, p. 3506.

At low earth orbit, 100 miles above the earth's surface, the atmospheric pressure is $\cong 1 \times 10^{-5}$ Torr. The number of molecules, (N), per cubic meter at this pressure is 3.64×10^{17} . Using the dimensions of the transmitter for comparison calculations, we obtain a beam area (A) of 0.07 m^2 . The intensity of light scattered by solar energy (I_{solar}) from one meter in front of the collector out to infinity in the wavelength range of 0.4μ to 0.6μ will be:

$$\begin{aligned} I_s &= (0.85 \times 10^{-32}) (N)(A)(I_{\text{solar}}) \int_1^{\infty} \frac{d\gamma}{r^2} \\ &= 7.6 \times 10^{-11} \text{ ergs/sec/cm}^2. \end{aligned}$$

Comparing this energy to stellar energy, we may determine the magnitude of the star which this energy equals.

The sun is a star of type G, and we will compare on this basis. The sun has magnitude of -26.6, and its energy is $3.5 \times 10^5 \text{ ergs/sec/cm}^2$ in the wavelength range of 0.4μ to 0.6μ .

$$\text{Then, } 0.4 (m - n) = \log \frac{l_n}{l_m}$$

where

m = magnitude of the sun

n = magnitude of the star (to be determined)

l_n = star radiant energy

l_m = solar radiant energy

$$\begin{aligned} n &= m - 2.5 \log \frac{l_n}{l_m} = -26.6 - 2.5 \log \frac{7.6 \times 10^{-11}}{3.5 \times 10^5} = 12.5 \\ &= \text{magnitude of star, equivalent to scattered sunlight.} \end{aligned}$$

At synchronous earth orbit, the atmospheric pressure is $\cong 5.5 \times 10^{-15}$. At this pressure, the number of molecules per m^3 is 2×10^8 . Under the above conditions, the light scattered into the open end of the telescope is:

$$\begin{aligned} I_s &= 0.85 \times 10^{-32} (3.5 \times 10^5 \times 2 \times 10^8 \times 7 \times 10^{-2}) \int_1^{\infty} \frac{d}{r^2} \\ &= 4.15 \times 10^{-20} \text{ ergs/sec/cm}^2. \end{aligned}$$

This is equivalent to energy from a star of magnitude 36.0.

Figure 13.2-4 shows the relationship of intensity of scattered sunlight as expressed in stellar magnitudes as a function of atmospheric pressure.

13.2.4.6.2 Energy Scattered from Particles having Size in the Order of Wavelength of Light

The previous discussion concerned particles where the size was very much smaller than the wavelength of light. A comparison between these particles and those of a size of the wavelength of light is in order.

$$I_s = I_o (5.3 \times 10^{-3}) \left(\frac{x^6 \lambda^2}{r^2} \right) |m - 1|^2 \cdot G(2 \times \sin \frac{1}{2} \theta) \cdot (1 + \cos^2 \theta)$$

but recognizing that the Bessel function will now become significant, we obtain for a particle, whose radius is equal to the wavelength of light ($0.5 \times 10^{-6} \mu$):

$$I_s = \frac{I_o}{r^2} (8.1 \times 10^{-12}) G(2 \times \sin \frac{\theta}{2}) \cdot (1 + \cos^2 \theta)$$

and for a particle whose radius is twice the wavelength of light:

$$I_s = \frac{I_o}{r^2} (5.1 \times 10^{-10}) G(2 \times \sin \frac{\theta}{2}) \cdot (1 + \cos^2 \theta).$$

The plot of these expressions as a function of angle is shown on figure 13.2-5. The peak value between 80° and 100° for $a = 0.5 \times 10^{-6} \mu$ is

$$I_s = \frac{I_o}{r^2} (1.05 \times 10^{-14}) \text{ and for } a = 1 \times 10^{-6} \mu \text{ is } I_s = \frac{I_o}{r^2} (4.7 \times 10^{-14}).$$

The value for molecules is $I_s = \frac{8.5 \times 10^{-33}}{r^2} I_o$. Therefore, the energy scattered by particles having a size of the order of the wavelength of light will be 10^{18} times that scattered by a molecule.

Using incident sunlight (3.5×10^5 ergs/sec/cm² between $0.4 \mu - 0.6 \mu$) for I_o , and for $a = 0.5 \mu$, then $I_s = 3.7 \times 10^{-9}$ ergs/sec/cm² at a distance of one meter. The energy scattered by one particle having a radius of 0.5μ at a distance of one meter is equivalent to a star of magnitude 8.4

13.2.4.6.3 Comparison of Strobe Light Output to Solar Energy

For this experiment, it is desirable to have a source whose intensity will be greater than the solar energy by at least a magnitude. The output efficiency of a Xenon strobe lamp is about 25% between 0.4μ and 0.6μ . Therefore:

$$\text{Input} = 1 \text{ joule}$$

$$\text{Output} = 0.25 \text{ joule between } 0.4 \mu - 0.6 \mu.$$

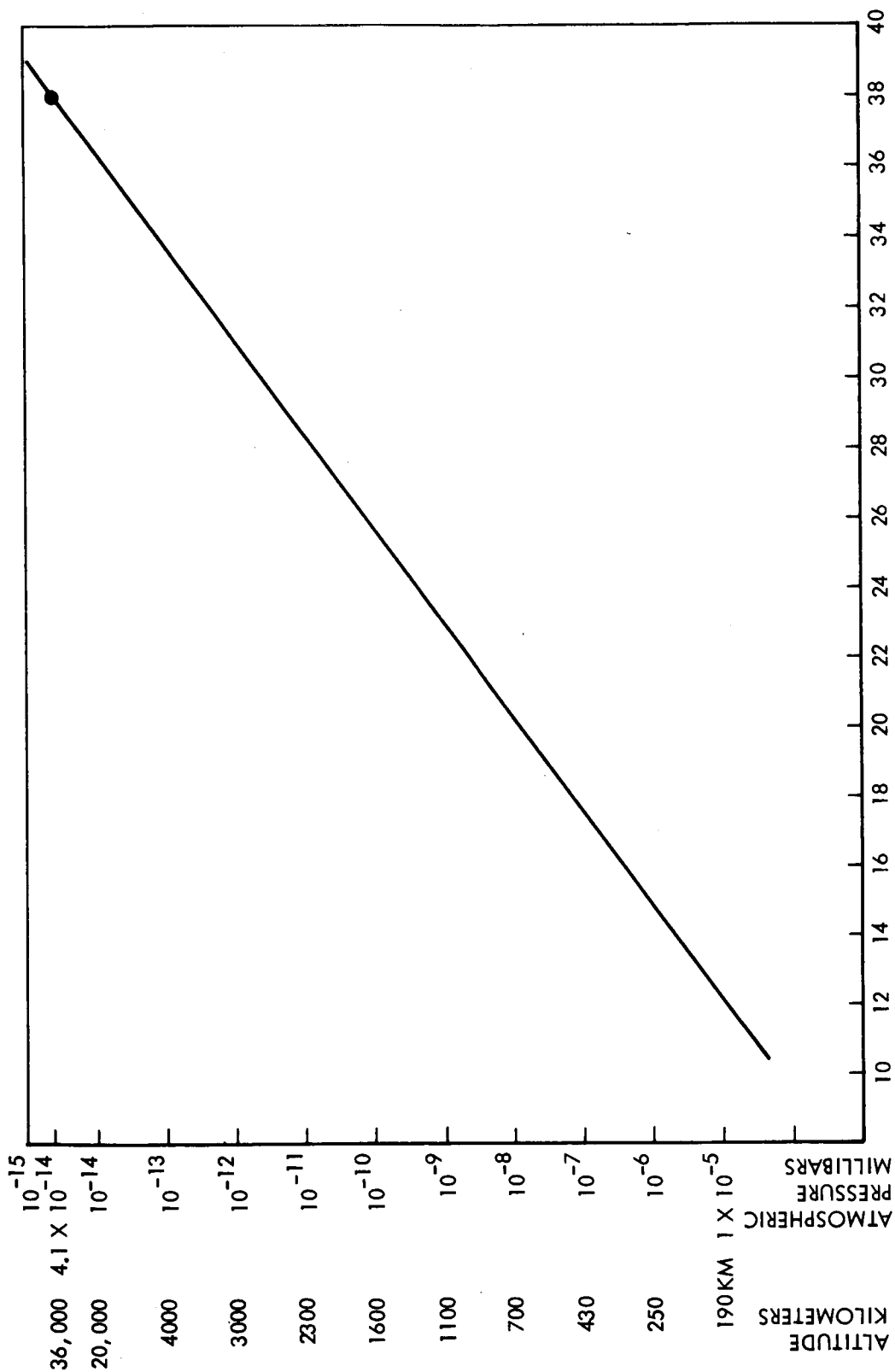


Figure 13.2-4. Equivalent Stellas Magnitude of Sunlight Scattered by Residual Atmosphere

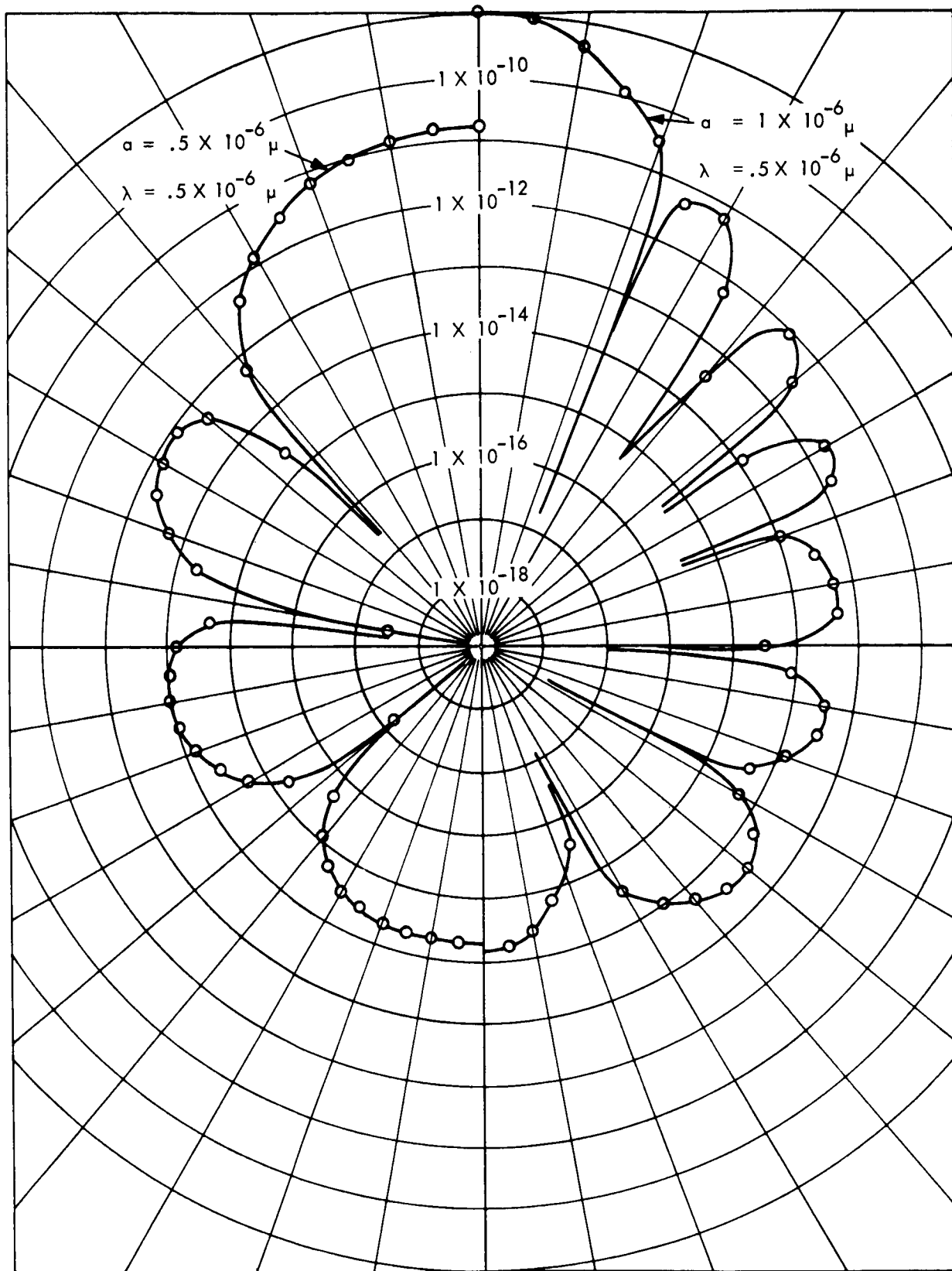


Figure 13.2-5. Scattering Intensity as a Function of Angle

With a beam diameter of 0.3 meter and 50% collection efficiency, $A = 2800 \text{ cm}^2$.

$$\text{Energy Density} = \frac{.125}{2800} = .445 \times 10^{-4} \text{ joules/cm}^2.$$

Since the energy is released in less than 1×10^{-5} seconds, the energy is equivalent to

$$\frac{.445 \times 10^{-4}}{1 \times 10^{-5}} = 4.45 \text{ watts/cm}^2.$$

The solar energy in the bandwidth 0.4μ to 0.6μ which is -3.5×10^{-2} watts/cm²(4) is 27 percent of the solar constant, giving a ratio of about 130/1 for a one-joule strobe light to solar energy if the receiver is gated. Strobe lamps are available for this purpose, having inputs as high as 10,000 joules; however, heat dissipation requires several minutes between pulses for this output.

13.2.4.6.4 Limiting Magnitude of the Fine Guidance Telescope

When the fine guidance telescope is used as recommended for astronomical applications, at the cassegrainian focus, it has the following characteristics:

$$\text{Aperture} = 0.75 \text{ m}$$

$$\text{EF} = 12.1 \text{ m} = 1210 \text{ cm}$$

$$\text{Image dia} = d = 0.002 + (\theta) \text{ EF} = -2.8 \times 10^{-3} \text{ cm}.$$

The limiting magnitude of a telescope is⁽⁵⁾:

$$m_l = M + 5 \log F - 2.5 \log d - 20.78$$

where

$$M = \text{limiting magnitude of the night sky} = 22.2$$

$$F = \text{equivalent focal length}$$

(4) Johnson, F. S., "The Solar Constant," Journal of Meteorology, VII, No. 6, December 1954, p. 436.

(5) Whipple, F. L., and Rubenstein, P. J., "The Limiting Magnitudes of Photographic Telescopes," Popular Astronomy, 50, January 1942, pp. 24-27.

d = image size on photographic film

$$\begin{aligned} m_1 &= 22.2 + 5 \log 1210 - 2.5 \log 2.8 \times 10^{-3} - 20.78 \\ &= 23. \end{aligned}$$

Therefore, from figure 13.2-4, the fine guidance telescope must be used above 1200 km in order to reduce the effect of scattered light so that it may operate at its limiting magnitude of 23.

13.2.5 Space Optics Thermal Control Experiments

13.2.5.1 Experiment Justification

One of the important technology areas in space optics is that of thermal control. The temperature variations and gradients that can be tolerated are in most cases orders of magnitude smaller than those permissible with spacecraft. Both active and passive methods of thermal control have been studied and proposed, and some experiments have been conducted. The passive approach, if it can be made to work, offers the best method for most considerations. Much can be learned about such systems in detailed analytical studies and ground testing in a simulated space environment, but the ultimate proof of its capability to furnish long time control must be demonstrated in space.

13.2.5.2 Contribution

The results would be used to demonstrate the feasibility of passive thermal control of an orbiting telescope. Results from parametric studies with this experiment would be used to arrive at a firm design application of these concepts.

13.2.5.3 Need for Space Testing

Much can be learned of the basic feasibility of such a concept in ground tests and studies. But the difficulty in accurately duplicating some aspects of the space environment as well as the inability to simulate others (i.e., long term zero g), make it imperative the experiments be conducted in space prior to actual application.

13.2.5.4 Experiment Design

Two experiments are proposed to demonstrate the concept of passive thermal control of a telescope system. These are discussed briefly and conceptually, and further work will be performed during Phase B in order to evaluate fully the implications of these concepts.

The purpose of these experiments is to take telescopes which incorporate passive thermal control and subject them to typical duty cycles for possible missions, and evaluate their ability to furnish acceptable optical performance.

13.2.5.4.1 Heat Pipe Experiment

This experiment would utilize the heat pipe concept in the design of the telescope tube, which conceivably would offer a uniform thermal environment on the inside of the telescope.

The specific method of approach is to form a heat pipe by the toroidal enclosure formed by the telescope tube wall and the telescope well. Tailored surface coatings of a specific absorptivity/emissivity ratio will be used on the exterior well surface and interior telescope tube wall. Such a design is expected to minimize the circumferential thermal gradient in the telescope well and tube walls thereby minimizing the thermal gradients in the mirrors. The heat pipe wick and working fluid will be selected from current state-of-the-art items.

Temperature transducers will be installed on the telescope tube wall and the telescope mirror to give both absolute temperature and gradients. Analytical simulation of the telescope will be a basic requirement to evaluate the departure of the analytic technique from space experimental results and ascertain the effects of orbit aging on the performance of the passive thermal control system. This type of experiment could also consider different types of fluids in the heat pipe, and could also be conducted in conjunction with complete passive control experiments.

13.2.5.4.2 Application of High Performance Insulation

This specific candidate passive control technique will be comprised of a state-of-the-art High Performance Insulation (HPI) such as NRC-2, GAC-1, GAC-2, or Linde SI-62 in conjunction with tailored surface coatings of a specific absorptivity/emissivity (α/ϵ) ratio. In order to eliminate the uncertainty associated with the thermal conductivity of structural joints where accurate assessment of thermal conductivity is a problem, welded joints will be employed where feasible.

The experiment plan is to use HPI on exterior side of the telescope tube wall and on the internal side of the telescope well wall. This configuration is expected to yield minimum thermal gradients. The application of micro-meteoroid protection as an integral part of the HPI will be the design method of approach. This configuration is expected to minimize the weight of the thermal control/micro-meteoroid protection system.

Temperature transducers will be located on the telescope tube wall and the telescope mirror to give both absolute temperature and gradients.

Analytical simulation of the telescope will be a basic requirement to evaluate the departure of the analytic technique from space environmental results and ascertain the effects of orbit aging on the performance of the passive thermal control system.

13.3 OPERATIONAL USES OF OTAES SYSTEMS

The existence in synchronous orbit of a high resolution telescope with accurate guidance and pointing capabilities (the fine guidance telescope) and an independent data link (provided by the laser communications experiments) is an opportunity in evaluating operational performance of such systems, in both manned and remote operating modes.

This subsection will discuss briefly some "operational" experiments which may be performed using OTAES systems on completion of the original hardcore experiment. It is realized that these uses will require some additional instrumentation, and may in some cases affect the performance of the basic experiment. A full analysis of these interfaces will be performed in Phase B, together with a more detailed study of experiments proposed.

13.3.1 Remote Sensing Applications

There is considerable interest among scientists concerned with remote sensing of environment in a capability for high resolution photography from synchronous orbit. Such a capability will provide synoptic coverage of a considerable portion of the earth's surface, and will be of major interest to meteorologists and oceanographers in particular.

It is proposed to equip the fine guidance telescope with a camera system in order to make multi-spectral photographs of the earth's surface at high resolution, and to use the laser experiment for altimetry and the determination of positional information concerning the spacecraft. The laser data link will also be utilized to transmit data to ground at the high rates required to make such an experiment worthwhile.

We can consider the physical characteristics of the OTAES system involved, namely, the fine guidance telescope. This has an aperture of 30 inches (0.75 meters) and on the basis of diffraction limited optics it can be shown that the ground resolution varies from approximately 100 feet at the center (on axis) to 700 feet at the edge of a 1° field of view (30 minutes off axis). This will provide adequate resolution for cloud cover photography, and the synchronous orbit allows a continuous picture of the growth of cloud formations to be taken. If we consider the lowest ground resolution, we have in the field of view a resolution element of 2.5 arc sec. Thus the field of view will contain $\frac{(60 \times 60)^2}{2.5 \times 2.5}$ resolution elements, or about 2×10^6 resolution

elements per picture. Using a grey scale of sixteen levels of grey implies a four bit word for each resolution element. Thus transmitting this information in digital form we have 8×10^6 bits per picture. The laser data link is projected

to have an information capacity of 10^7 bits/sec. Thus we can see that we can transmit the contents of such a picture in approximately one second.

Of course, it is also possible for earth resources applications to provide a correcting lens in the telescope (the Ross corrector) which will reduce the aberrations significantly, so that we may have ground resolution on the order of 200 ft at the edge of the field. This will, of course, increase the number of bits of information by a factor of about 30, giving approximately 30 seconds per scan of the field of view if the original information capacity is maintained.

This implies that complete coverage of the portion of the earth's surface visible from the spacecraft can be attained in about 100 pictures, taking approximately $1\frac{1}{2}$ hours for full coverage.

In consequence, extensive studies of the feasibility of using the fine guidance telescope for earth remote sensing should be conducted, and suitable specific applications be evaluated during the next phase of the program.

13.3.2 Astronomical Uses of the OTAES Fine Guidance Telescope

This section will consider ways in which the OTAES fine guidance telescope may be used for astronomical observations.

The fine guidance telescope is a diffraction-limited Cassegrain system with a magnifying relay lens. The diameter of the primary mirror is 0.75 meters. The effective focal ratios are $f/96$ with the lens and $f/16$ without the lens. The respective linear resolutions are 15 and 93 lines per millimeter at 5500 Å.

A focal ratio of $f/96$ is usually used in astronomy only for solar tower work. A linear resolution of 15 lines per millimeter is too coarse for suitable use with photographic film or television transducers.

An $f/16$ Cassegrain telescope is more suitable for general observations particularly medium-dispersion spectroscopy and long-focus astrometry. The linear resolution of 93 lines per millimeter at 5500 Å and 256 lines per millimeter at 2000 Å is suitable for use with photographic film or television transducers.

Astronomical objects of special interest for medium-dispersion spectroscopic would be O and B stars, emission and planetary nebulae, quasi-stellar sources, globular dusters, spectroscopic binaries, and stars with suspected planetary systems. Astronomical objects of special interest for long-focus astrometry are distant multiple galaxies, globular clusters, close visual binaries, and stars with suspected planetary systems.

The fine guidance experiment is designed to obtain pointing and guiding accuracies of 0.01 with no restrictions on the energy required to maintain

these accuracies. Since the angular resolution of f/16 Cassegrain is 0.03 at 5500 Å and 0.09 at 2000 Å, the pointing and guiding accuracies of 0.01 are sufficient to maintain a "diffraction-limited" system.

The long-focus astrometry and for medium-dispersion spectroscopy of extended objects, these pointing and guiding accuracies would have to be maintained for three orthogonal axes. For medium-dispersion spectroscopy of point sources, these pointing and guiding accuracies would have to be maintained in two axes, orthogonal to each other and to the direction of the spectro-scope slit. A lesser accuracy can be maintained in the third axis, providing the constant drift necessary along the slit in order to give spectral lines of sufficient length. For spectroscopy, a comparison spectrum of some sort will be required.

In summary, it is recommended that the 0.75 meter OTAES fine guidance telescope be used at the f/16 Cassegrain focus for second generation spectral and positional surveys of all astronomical objects in the ultraviolet and visual portions of the spectrum. Special emphasis on astronomical objects would be in accordance with the above listings and the results of first generation surveys from unmanned space astronomy observations.

Since this is a diffraction-limited reflective system with close pointing and guiding tolerances, observing times and performance should be limited only by the available energy from the particular spacecraft configuration, the orbital considerations, and/or the limitations of the detectors.

**OPTICAL TECHNOLOGY
APOLLO EXTENSION SYSTEM
PHASE A**

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EXPERIMENTS

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