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# TECHNICAL NOTE

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# STUDY OF A PROPOSED INFRARED HORIZON SCANNER FOR USE IN SPACE-ORIENTATION CONTROL SYSTEMS

By Norman M. Hatcher and Ernst F. Germann, Jr.

Langley Research Center
Langley Air Force Base, Va.

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## SUMMARY

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An attitude-sensing device for space vehicles which detects the thermal radiation discontinuity at opposite horizons of a planetary body to produce an attitude error signal is described. The planetary body may be the Earth, its Moon, Mars, or Venus. The sensor is expected to have an accuracy of $0.25^{\circ}$ for the Earth, a long continuous operating lifetime, a wide altitude range, a wide capture capability, and an inherent ability to produce signals indicating vehicle altitude. An experimental model incorporating many of the features of the proposed sensor indicates that the proposed sensor will be low in weight, volume, and power consumption. The sensor's altitude range, accuracy, lifetime, and sensitivity to radiation from the Moon and planets are discussed.

## INTRODUCTION

Many space-vehicle missions will require that either the entire vehicle, or a part of it, be pointed toward the Earth, its Moon, or some other planetary body. Examples of vehicles which will require such orientation are weather observation satellites, certain types of communications satellites, manned and unmanned planetary landing craft, and planetary surveillance satellites. An on-board instrument is required, therefore, which will sense the attitude of the vehicle with respect to the planetary body and produce a signal indicating the direction and magnitude of pointing error. With this information, the attitude control system can, by the use of reaction jets, flywheels, or other suitable means of obtaining torque, keep the vehicle or its components properly oriented.

Different missions will impose different requirements upon the sensor, although certain requirements will apply for almost all missions. These are accuracy, low power consumption, light weight, small size, the ability to discriminate against bodies other than the desired planetary body, nighttime as well as daytime operation, and an output signal which can easily be used by the attitude control system. Specific missions may, in addition, require that the sensor have a long operational lifetime, the ability to produce a useful attitude error signal for large angles of misorientation, a wide altitude range, or the ability to produce vehicle altitude information. Although several other attitude sensors have been designed and a few units are commercially available, it is felt that a need still exists for a sensor which can effectively meet a larger number of these requirements. It is the purpose of the proposed sensor to fill this need.

There are several methods by which attitude information could be obtained aboard a space vehicle, such as by radar, by sensing the gravity
gradient, and by sensing the planetary-reflected visible radiation. How- L
ever, all these methods would impose serious limitations upon the sensor. 1
On the other hand, all the above-mentioned requirements can be met by 6
utilizing the planetary-emitted infrared radiation. The sensor which 2
this report describes therefore uses this infrared radiation in pro- 5
ducing attitude and altitude information. In operation, the sensor scans by means of rotating mirrors from space across the planetary body and notes the change in amount of infrared radiation which it receives upon crossing opposite horizons.

The present report describes the sensor and the principles upon which it operates. In addition, its sensitivity, altitude range, and accuracy are quantitatively discussed.

## SYMBOLS

For conversion of units, 1 statute mile = 1,609.344 meters.

| | |
|---|---|
| $A$ | detector area, $cm^2$ |
| $A_1$ | area of radiating source, $cm^2$ |
| $A_2$ | effective optical aperture, $cm^2$ |
| $B$ | bandwidth of amplifier, cps |
| $D$ | distance from detector to radiating source, statute miles |
| $E$ | total emissive power, $\frac{\text{watts}}{\text{cm}^2}$ per hemisphere |
| $e$ | amplifier input noise, volts |
| $h$ | altitude above center of planetary body, statute miles |

$k$ Boltzmann's constant, $1.38 \times 10^{-23}$ joules/$^{o}K$

$N$ total electronic noise, volts

$P$ radiant power, watts

$R$ resistance, ohms

$r$ radius of planetary body, statute miles

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$t_h$ time at horizon interception

$t_r$ time at reference signal generation

$t_{r,1}, t_{r,2}$ times at generation of successive reference signals

$S$ signal

$T$ temperature, $^{o}K$

$\gamma$ scan arc from switch-on to sensor axis

$\delta$ scan arc from beginning of scan to first interception of horizon

$\epsilon$ emissivity

$\epsilon_1$ emissivity of radiation source

$\epsilon_2$ emissivity of detector

$\varphi$ angular pointing error

$\sigma$ Stefan-Boltzmann constant, $5.7 \times 10^{-12}$ watt/cm$^2$/deg$^4$

$\tau$ detector time constant, sec

Subscript:

min minimum

## PRINCIPLES OF OPERATION

The operation of the sensor is based upon the following principles. All bodies emit radiation at a level proportional to the fourth power of

their absolute temperatures in accordance with the Stefan-Boltzmann law, $E = \epsilon\sigma T^4$. At the temperatures of the Earth, Mars, Venus, and the Moon, most of this radiation falls within the infrared region of 3 to 30 microns. Since the effective temperature of space is almost absolute zero, a significant infrared radiation discontinuity exists at the horizon of the planetary body. If the axis of the sensor is pointed toward the center of some spherical planetary body, the angles at the sensor between this axis and any two opposite points on the horizon of the body are equal. In figure 1, the fields of view of two radiation detectors are scanning synchronously in one plane through large opposite scan arcs from space
across the horizon of the planetary body. Since a pointing error exists, L
one detector in the error plane will detect an increase in radiation at 1
one horizon at some time before the second detector in this plane detects 6
a corresponding radiation increase at the opposite horizon. The scan 2
rate is constant; thus, the angular pointing error is directly propor- 5
tional to the time interval between interception of opposite horizons. The direction of pointing error is determined by noting which of the two detectors first detects the radiation discontinuity.

Two additional detectors scanning in a perpendicular plane can, in a similar manner, detect the magnitude and direction of pointing error in the second plane.

## Optical System

Figure 2 shows the optical system of the sensor. It consists of four germanium hemispherical lenses, four parabolic reflector segments, and four flat mirrors rotated by a slow-speed alternating-current motor. As the mirrors are rotated, the $1.5^{\circ}$ by $6^{\circ}$ fields of view of the detectors are caused to scan through arcs from space across the planetary body horizons. Incoming radiation is reflected from the rotating mirrors to the parabolic reflector segments which, in turn, reflect and converge the radiation upon the lenses. The lenses focus the radiation upon thermistor detectors attached to the rear surfaces of the lenses.

The refractive index of germanium is about 4. For an equivalent field of view, this reduces both the width and length of the detector by factors of 4. Since the responsivity (Voltage output/Radiation input) of the thermistor is inversely proportional to the square root of its area (ref. 1), a responsivity increase of 4 is theoretically obtainable. However, due to reflection losses from the germanium, responsivity increases of only about 3.5 can actually be realized with the lenses. The lenses also function as filters for removing radiation with wavelengths below 1.8 microns and thus eliminate most of the Sun's unwanted direct and planetary-reflected radiation.

The two mirrors in one scanning plane are made to rotate 90° out of phase with the mirrors in the second plane. This phase relation is maintained as they are driven by an intermeshing gear set. The mirrors are also faced on both sides, which allows them to make two scans per revolution. The scan cycles of the mirrors in the two perpendicular planes are therefore separated by 90° of mirror rotation, or 180° of scanning. Because of this feature, only one electronic circuit is required for alternate processing of the signals in both planes.

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## Circuit Functions

The basic function of the electronic circuit of the attitude sensor is to convert the optical input to the thermistor detectors to an electrical output signal. This output signal indicates the direction and magnitude of vehicle pointing error. In order to obtain this output, the optical input is processed by the circuit shown in figure 3. The character of the signal at various points in the diagram is shown in figure 4.

The change in the amount of intercepted infrared radiation produced by scanning over the horizon is converted to an electrical signal in the thermistor detector bridge circuit of figure 3. This electrical signal is then amplified and turns on a triggered bistable transistor such as the Trigistor (ref. 2). The triggered bistable transistor is the circuit equivalent of the bistable multivibrator circuit. Opposing horizons operate independent bistable transistors. The two bistable transistors are so connected that their outputs subtract. Since the bistable transistor outputs are of equal magnitude, an output signal exists only from the time one thermistor detects the horizon until the other thermistor in the same scan plane detects the opposite horizon. If a pointing error exists, an output signal will be produced with a duration proportional to the pointing error. The direction of the pointing error is determined by noting which of the two detectors first intercepts a horizon. The bistable transistors are then reset by a reset signal and thus readied for the next scan cycle. The next scan cycle will be in a plane perpendicular to the one just completed and the input and output signals are switched accordingly. The switching arrangement permits the use of the same electronic signal processing circuit for both planes of scanning. Transistorized electronics are used to minimize power drain, which is expected to be less than 1.5 watts.

A Sun signal eliminator circuit is provided to prevent the Sun from injecting a false error signal into the sensor output.

From signal pulses existing in the processing circuitry, the altitude of the vehicle can be computed.

# ELECTRONICS

## The Bridge Circuit

The bridge circuit converts the optical input to the scanner into an electrical signal. A thermistor detector is used as the varying bridge branch, as shown in figure 3. Thermistors were selected as radiation sensing elements for the present study because of their ruggedness, fast response, small size, and good sensitivity in the desired infrared range without cooling. The thermistor has a high negative temperature-resistance coefficient so that an increase in the infrared radiation incident upon it produces a decrease in its resistance. The bridge branches are so adjusted that the bridge is approximately in balance when the thermistor is viewing space. The change in resistance produces an output in the bridge circuit into which the detector is connected. An additional thermistor is mounted near the active thermistor and is shielded from incoming infrared radiation. This thermistor is part of the bridge circuit and is used to compensate for changes in characteristics of the bridge circuit due to changes in local ambient temperature. The sensor uses four such bridge circuits, two for each axis, one for each horizon. The detectors in only one scanning plane are biased at any one time and thus only one pair of bridges is activated simultaneously. The intensity of the radiation focused on the detector surface by the optical system increases abruptly when the field of view crosses over the horizon and thus produces a signal in the bridge output.

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## The Amplifier Circuit

The bridge output is connected to an alternating-current amplifier which amplifies the transient pulse produced by the horizon crossover. The gain of this amplifier must be sufficient to produce a signal with enough power to operate the subsequent bistable transistor circuit shown in figure 5. The amplifier rise time should be less than the time constant of the thermistor detector, so that it will not introduce any errors into the system. Since the amplifier is an alternating-current amplifier it will not respond to slow and long-term variations in the input signal, as might be caused by ambient temperature changes. Low-noise silicon transistor amplifiers can be obtained for this portion of the circuit.

## The Triggered Bistable Transistor Circuit

The bistable transistor circuit is based on the bistable properties of the recently developed four-layer transistor (ref. 2). The bistable transistor circuit of figure 5 has two stable states, either "on" or "off." It can be switched from one state to the other by applying a signal to its gate terminal. A positive gate signal puts the bistable transistor in the "on" condition and a negative signal puts it in the "off" state. The amplifiers produce a positive signal pulse when the detectors in their input circuit perceive a horizon. This signal turns on the bistable transistor and the bistable transistor then remains on until it is cut off by a pulse from the reset signal multivibrator shown in figure 3. The bistable transistor output is a constant amplitude signal. The output signals of the two bistable transistors are subtracted in the circuit and their difference appears across the bistable transistor circuit output terminals. If both detectors produce a signal at the same time, that is, when the sensor is perfectly alined, there will be no output because the two signals cancel. If, however, the field of view of one detector intercepts the horizon before the other, due to sensor misalinement, one bistable transistor will be turned on first. Thus, a signal will appear at the output terminals which is either positive or negative, depending on the misalinement direction. This signal will persist until the second bistable transistor is turned on by horizon interception of the second detector and the output then disappears. The net output is thus a pulse with a polarity determined by the error direction and a width proportional to the angular pointing error. This output error signal can be used directly or, if necessary, it can be integrated by an integrator circuit (ref. 3) such as those shown in figures 6 and 7. This integrator circuit has complete restoration between pulses. If the restoring signal is applied at the beginning of the scan cycle, the output of the integrator will consist of a direct-current signal whose amplitude is proportional to the error angle. This signal will persist for a large part of the scan period. The output of the integrator must be coupled to the load by a high impedance circuit, such as a cathode follower circuit, to prevent draining the charge off the integrator capacitors.

## The Reset Signal Circuit

At the end of each scan cycle a negative reset pulse is applied to the bistable transistor gate which turns both bistable transistors off and thus prepares them for the next scan cycle. This reset signal is produced by a monostable multivibrator which is triggered by magnetic pickups, as shown in figure 8. The magnetic pickup is a coil wound around a permanent magnet. As a magnetic material is passed in the vicinity of this magnet the reluctance of the magnetic path changes and a change is produced in the number of magnetic lines of flux linking the coil. This change in flux linkages induces a current in the coil. Two pickups are mounted on the frame in such a position that a corner of the scanning mirrors passes by them at the beginning of each scan cycle. Tiny permanent magnets are mounted on one mirror in each of the two scan axes at the point where they pass the pickup coil. These magnets are oriented so that their magnetic fields aid that of the pickup coil magnet and thus the signal produced is larger than that which would be obtained by using nonmagnetized ferromagnetic material. The location of the magnetic pickups with respect to the angular position of the mirrors determines the position of the detector fields of view at the beginning of the active portion of the scan cycle. The pulse produced by the magnetic pickup triggers the monostable multivibrator and causes it to go into the "on" state. It then remains in the "on" condition for a period of time which is determined by the circuit elements. This "on" time is adjusted to produce a pulse of sufficient width to keep the bistable transistor turned off until any switching transients that may exist disappear from the amplifier output. The switching transients could be produced by a small unbalance existing in the bridge circuit at the time of scan axis switching.

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The active portion of the scan cycle, or that portion during which the detectors are battery biased, may begin when the detector fields of view are at any angular position below about $30^{\circ}$ above the horizontal (where the horizontal is perpendicular to the sensor axis). The detectors remain biased until the mirrors have rotated through $90^{\circ}$ from switch-on, at which time the bias is switched to the detectors scanning in the perpendicular plane. The active portion of each detector's scan arc will exist only until a signal is produced at horizon crossover. Subsequent signals will not be processed by the electronics.

## The Switching Signal

The magnetic pickups also trigger a conventional bistable multivibrator circuit so that it is turned on during the scan period in one plane and turned off during the scan period in the perpendicular plane. The output of this multivibrator is used for switching the battery bias and the output signals from one plane to the other. In order to help insure a long operating lifetime, switching is accomplished by electronic rather than by mechanical relays.

## Elimination of False Signals

When the Sun enters the detector field of view it produces a signal in the circuit. Since the infrared radiation received from the Sun is many times more intense than that received from the planetary body, it produces a much larger amplifier output. By the use of Zener diodes which will not conduct below their Zener breakdown voltage, this larger signal can be used to operate the reset multivibrator. The reset multivibrator thus inactivates the sensor until the Sun passes out of the field of view. In order to do this the time duration of the reset pulse must be at least as long as the time required for the sensor to scan across the Sun. The functions of the sensor are thus not normally impaired by the presence of the Sun.

There are, however, several rare conditions which may persist for several scan cycles during which the Sun signal eliminator circuit will not prevent the sensor from being operative. One such situation occurs when the Sun covers only a very small fraction of the field of view so that the amount of incident radiation is approximately equal to that received from the reference body. Under this condition the Sun will trigger the multivibrator and the sensor will produce a false error output. The probability for the existence of this condition will be extremely low. An undesirable condition also occurs when the Sun is in such a position that it appears to be touching the edge of the planetary body. In that case the Sun eliminator circuit will inactivate the sensor for a short period of time and when the sensor is reactivated it will already have scanned over the horizon and will thus miss a scan cycle.

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The Earth could also interfere with the operation of the Moon sensor. Even at the Moon's distance from the Earth, the Earth could produce a strong enough signal to trigger the electronics. The relative intensities of the radiation inputs from these planetary bodies may be seen in figure 9. There are several ways to alleviate the problem, however. For example, the active portion of the scan cycle could be restricted so that it would begin when the detector fields of view are at some angular position below the horizontal and it could end several degrees before the fields of view have reached the vertical (with respect to the sensor axis). This would prevent the Moon sensor from stabilizing a vehicle toward the Earth and it would lessen the probability of Earth interference.

A method of preventing the Moon sensor from triggering on the Earth signal would be to modify the sensor so that the detector output would not reach the triggering level during the time required for the sensor to scan across the Earth. This could be accomplished by increasing the scan rate, by using a detector which has a longer time constant, or both. Accuracy would be seriously degraded by this method, however, and further work should be done on Earth signal elimination if very accurate, uninterrupted attitude information is required from the Moon sensor.

## Altitude Signal

The signal processing of the attitude sensor under study is of such a nature that altitude can readily be determined from signals in the circuit. The geometry of altitude determination is shown in figure 10.

The position of the magnetic pickups, and thus the angle at which the reset signal occurs, is known. From the time between successive reset signals the exact scan rate is obtained. From the time between the reset signal and the horizon crossover signal, the angle $\gamma - \delta$ of the horizon can be determined (fig. 10). The horizon angle determines the altitude of the sensor by the relation $h = r\left[\csc(\gamma - \delta) - 1\right]$. All the information necessary for determining altitude can be obtained by recording two signal pulses, one at the reset time and one at the crossover time.

## OPERATING CHARACTERISTICS

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### Radiation From the Moon and Planets

Figure 9 shows the approximate amounts of radiation which will be received by the detectors from the Earth, Mars, Venus, and the Moon at varying distances from these planetary bodies. The curves are based on the use of a germanium hemispherical lens which filters out a large percentage of the incident radiation, an effective optical aperture of 5 $cm^2$, and a 1.5° by 6° field of view. The lunar and planetary radiation characteristics used in the appendix and also to obtain the curves in figure 9 are reviewed in the following paragraphs.

Because of its tenuousness and lack of infrared-absorbing water vapor, the Martian atmosphere will probably not present a large enough radiation discontinuity to trigger the electronics. The sensor should therefore detect the surface of Mars, instead of its atmosphere. The surface varies in temperature between the extremes of about 175° K and 300° K (ref. 4, p. 77-98). It has an average emissivity of 0.9 (ref. 4, p. 150). With increasing altitude, a larger surface area of the planet will be included in the detector fields of view. Thus, the average detected temperatures of these larger areas will be progressively less extreme with increasing altitude due to an averaging effect of the hotter and colder portions of the planet. The increasing minimum average temperature with altitude is responsible for the increase in the minimum radiative power curve of figure 9 for Mars out to an altitude of 39,500 statute miles, where the maximum average temperature is about 270° K and the minimum is about 200° K. Beyond 39,500 miles, Mars does not fill the entire fields of view of the detectors and the minimum intercepted radiative power curve starts to decrease with increasing altitude. The maximum radiative power curve for Mars decreases with increasing altitude from the surface of Mars because of the averaging effect of the colder areas.

For the Moon, the temperature varies between about 375° K at the subsolar point and 123° K in some shaded areas (ref. 5). Its average emissivity is about 0.9 (ref. 6). As with Mars, the average detected surface temperatures will become less extreme with increasing altitude out to 20,000 miles (the highest altitude at which the Moon completely fills the field of view), where the average maximum temperature is 345° K and the average minimum temperature is 150° K.

The effective radiating portion of the Earth's atmosphere radiates as a black body at temperatures between 225° K and 270° K. These are assumed to be the maximum and minimum Earth temperatures detected by the sensor at any altitude. The Earth does not completely fill the detector fields of view beyond 73,000 statute miles. Thus, the radiation curves for the Earth decrease past this point.

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The dense carbon dioxide atmosphere of Venus radiates as a black body at a constant temperature of about 226° K (ref. 7). Beyond 71,500 statute miles, Venus does not fill the entire field of view, so that the radiation decreases beyond this point.

## Altitude Range

The radiation discontinuity at the horizon of the planetary body must be sufficiently strong to produce a signal which is distinguishable from random currents, or noise, in the electronics. The maximum operational altitude is, therefore, limited by electronic noise. An analysis of the electronic noise is made in the appendix.

If the assumptions in the previous section are valid and with an amplifier gain of about 74,000 (see appendix), the sensor could conceivably operate at distances of about a million miles from the Earth and Venus, about 570,000 miles from Mars, and about 135,000 miles from the Moon. These values are determined from the interception of the minimum radiative power curves with the minimum operational power level line of figure 9. The minimum operational power level represents the smallest amount of radiated power which could produce a detectable signal in the electronic circuitry. The sensor's capture capability, or probability that the planetary body would be scanned across by at least one detector during a scan cycle to produce a useful attitude error signal, would be poor at extremely high altitudes, however, due to the small angular size of the planetary bodies. Should higher altitude capability be required, the optical components could be enlarged to collect more radiation and thus produce a stronger signal.

The minimum operational altitude will be either that at which the time duration of the radiation discontinuity is so long that the resulting slowly rising thermistor output cannot be amplified as a pulse by the alternating-current amplifier, or that at which errors due to this unclearly defined atmospheric horizon become intolerable.

## Capture Capability

In general, a large capture capability is desirable since it will permit quick alinement of the vehicle over a wide range of injection attitudes. Also, in the event that the vehicle becomes significantly misalined, it will permit rapid realinement. The present sensor has a large capture capability due to its large scan arc.

If, upon injection or at some later time, the vehicle has an attitude such that the planetary body is in the field of view of one detector during the detector's entire scan arc, there will be no horizon crossover signal from this detector. Such a condition could occur at altitudes at which the angular size of the planetary body at the sensor is larger than the active portion of the scan arc. Without a signal the sensor's capture capability could be seriously limited. This problem will possibly not be important for an Earth or Mars oriented sensor, however; radiation discontinuities over the surfaces of these planets may be sufficient to trigger the electronics and thus produce an error signal.

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On the other hand, Venus and possibly the Moon will not provide these surface radiation discontinuities and, at low altitude some additional means will be required to detect the vehicle's attitude. Under these conditions the input radiation levels into opposite detectors might be compared and the stronger input used to trigger the electronics.

## Accuracy

Imperfections in the construction of gears and optics and variations in characteristics of electronic components could produce inaccuracies amounting to $0.05^{\circ}$. However, the principal limiting factors in sensor accuracy will probably be those associated with the radiating source itself. These are oblateness, differences in altitude of the effective radiating source, and variations in radiation intensity along the horizon of the source.

The Earth's oblateness and variations in altitude of the effective radiating source in the atmosphere could cause a maximum error of about $0.1^{\circ}$ for Earth oriented vehicles. Differences in radiation intensity at opposite horizons of the planetary body, as caused by emissivity and temperature variations, will produce signals having different slopes. Thus, the time required for the signals to reach the fixed triggering level will be proportionately different and an additional error will be produced. This additional error should be less than $0.1^{\circ}$ for Earth oriented vehicles, where the amounts of radiation received from opposite horizons differ by less than the expected maximum factor of 2. These combined factors could, then, cause a maximum error of about $0.25^{\circ}$ in the sensor's determination of the vehicle's attitude with respect to the geometrical center of the Earth. Errors of less than 0.50° can be expected in the determination of the direction toward the centers of Mars, Venus, and the Moon.

### Lifetime

If it is assumed that the level of cosmic radiation is insufficiently high to produce marked changes in the characteristics of electronic components, the principal factor limiting sensor lifetime will probably be gear and bearing wear. Wear is minimized by the use of a slow-speed motor driving the mirrors as slowly as 90 rpm. Lifetime can be further increased by lubricating the moving parts and pressurizing the sensor to keep the lubricant from evaporating. This would require a cover similar to the one shown in figure 11.

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## EXPERIMENTAL SENSOR

In order to verify the feasibility of the sensor described in the preceding sections, construction was begun on a prototype experimental sensor. This construction is complete except for portions of the circuitry still under development. Although the sensor cannot yet be tested as a complete system, some initial tests have been made by using the optical system, detectors, amplifiers, and mechanical drive system, while the instrument scanned a known thermal discontinuity. All the tested components performed well. The relatively heavy (5-ounce) drive motor consumed 1.8 watts, although a smaller motor such as would be used in the proposed sensor is expected to weigh 2 ounces and to consume less than 1.5 watts. The experimental sensor has a diameter of 4 inches, a volume of about 60 cubic inches, and a weight of about 32 ounces, complete with the unassembled electronic components. The experimental sensor is identical to the proposed sensor except that the rotating mirrors of the experimental model are driven by a worm (figs. 12 and 13) instead of by the proposed pinion drive. The latter arrangement was chosen after construction had begun and is considered better in that it will have less friction, and, with the drive motor moved to its proposed location, the maximum operational altitude limitation which the motor would have imposed is removed.

## CONCLUSIONS

An attitude sensor for orienting space vehicles toward planetary bodies has been designed and construction on a slightly different version is partially complete. Theoretical and experimental evaluations indicate that the sensor will have the following important characteristics:

1. It should be accurate to 0.25° (Earth scanning version).

2. It should be capable of operating continuously for long periods of time.

3. It should be capable of operating at very high as well as low altitudes.

4. It should have a wide capture capability.

5. It should be capable of indicating vehicle altitude.

6. It should weigh about 1.8 pounds.

7. It should have a volume of about 60 cubic inches.

8. It should require less than 3 watts of power.

9. It should, except on rare occasions, be able to reject false signals from the Sun.

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Langley Research Center,
National Aeronautics and Space Administration,
Langley Air Force Base, Va., October 24, 1961.

# APPENDIX

## RADIATION TRANSFER AND SYSTEM SENSITIVITY

The radiation discontinuity which the sensor detects between space and the horizon of the planetary body may be obtained in the following manner:

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The radiant power received by the detector from the planetary body is given by the following formula (which may be obtained by using ref. 8)

$$P = \frac{A_1 A_2 \epsilon_1 \epsilon_2 \sigma T^4 (\text{Optical efficiency})}{\pi D^2} \tag{1}$$

where $A_1$ is the area of the radiating source, $A_2$ is the effective optical aperture of the sensor (about 5 $cm^2$), $\epsilon_1$ is the emissivity of the radiation source, $\epsilon_2$ is the emissivity of the detector (0.8), $\sigma$ is the Stefan-Boltzmann constant, T is the absolute temperature of the radiating source, and D is the distance from the detector to the radiating source.

The field of view of the sensor is about $1.5^\circ$ vertically by $6^\circ$ horizontally, or about 0.0027 steradian. Thus, the projected area of the radiating source is $0.0027D^2$.

The optical efficiency of the sensor is determined by the surface quality and reflectivity of the optical components and the percentage of incident radiation which the germanium lens transmits. Germanium is opaque to nearly all radiation except that within the wavelength region of 1.8 to 25 microns. Thus, the fraction of the total incident energy which the lens transmits depends upon the temperature of the planet, due to a shift in the wavelength of maximum emission with temperature. If the lowest expected planetary temperatures are assumed, the optical efficiencies for the Earth-, Moon-, Mars-, and Venus-sensors are about 0.27, 0.12, 0.24, and 0.28, respectively.

Substituting the appropriate values in equation (1) gives

$$P_{min} = 1.2 \times 10^{-5} \text{ watt}$$

This is the minimum signal expected from the Earth at altitudes below 73,000 miles. Since essentially no radiation is received from space, this is the discontinuity which the sensor will detect upon crossing the Earth's horizon. Thermistor detectors with the desired characteristics have a responsivity of about 175 volts per watt of radiation input. The detector output signal is, then,

$$(175 \text{ volts/watt})(1.2 \times 10^{-5} \text{ watt}) = 2.1 \times 10^{-3} \text{ volt}$$

The detector bridge output is half this value or

$$\text{Bridge output} = 1 \times 10^{-3} \text{ volt}$$

The electronic noise is found as follows:

The noise equivalent power of the detector is determined from the empirical formula (obtained from ref. 9)

$$\text{Noise equivalent power} = 2 \times 10^{-10}\left(A\,\frac{B}{\tau}\right)^{1/2} \tag{2}$$

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For a field of view of about 1.5° by 6° the dimensions of each detector are 0.02 centimeter by 0.08 centimeter which gives them areas $A$ of $1.6 \times 10^{-3}$ cm$^2$.

The detector time constant $\tau$ should be such that the detector will respond directly to the radiation increase as its field of view crosses over the horizon. The scan rate is 3 cps with a resulting cross-over time of 0.0014 second. The detector time constant $\tau$ should, therefore, be no more than 0.0014 second.

The amplifier bandwidth $B$ must be wide enough to accommodate the sharply increasing bridge output signal which is produced at horizon crossover. The bandwidth is obtained from the following relation (ref. 10):

$$B = \frac{0.45}{\tau} = 320 \tag{3}$$

Substituting equation (3) in equation (2) gives the detector noise equivalent power as

$$\text{Detector noise equivalent power} = 3.8 \times 10^{-9} \text{ watt}$$

This value corresponds to a noise voltage of $(175 \text{ volts/watt})(3.8 \times 10^{-9} \text{ watt})$, or

$$\text{Detector noise voltage} = 0.67 \text{ microvolt}$$

Theoretically, the thermal noise generated in the input resistance of the amplifier is (ref. 11)

$$e^2 = 4RkTB \tag{4}$$

The equivalent input resistance $R$ of the amplifier is approximately 80,000 ohms, $k$ is the Boltzmann constant, $T$ is the maximum expected operating temperature ($323^{\circ}$ K), and the bandwidth $B$ is 320 cps (from eq. (3)). Therefore

$$e = 0.68 \text{ microvolt}$$

The "noise figure" is the ratio of the actual noise in an amplifier to the theoretical thermal noise. For low-noise silicon transistor amplifiers, the noise figure is about 1 decibel or 1.12. Thus, amplifier noise voltage is 0.76 microvolt.

The noise generated in the input leads and in the bias battery is expected to be less than 0.5 microvolt if proper shielding is used. The total electronic noise $N$ is the sum of the detector noise, the amplifier noise, and the lead and battery noise or 1.9 microvolts.

The minimum signal-to-noise ratio for the Earth sensor out to 73,000 miles is, therefore,

$$\frac{S_{min}}{N} = \frac{1 \times 10^{-3} \text{ volt}}{1.9 \times 10^{-6} \text{ volt}} = 525 \tag{5}$$

For reliable operation of the sensor, the signal should be about 5 times the total electronic noise, or 9.5 microvolts. This corresponds to a minimum operational radiation power level of $1.09 \times 10^{-7}$ watt, as indicated by the horizontal line in figure 9.

It is desirable to keep the triggering point of the bistable transistor at a fraction of the full bridge output level in order to keep errors brought about by differences in intensity of opposing horizons at a minimum. On the other hand, the triggering level of the Earth sensor should be high enough so that the Moon will not accidentally trigger the circuit. For a selected triggering level of one-quarter of the full bridge output, with the minimum radiative power from the Earth being assumed, the Moon will not interfere with Earth sensors operating within a few thousand miles from the Earth's surface. For operation at higher altitudes, the triggering level would have to be increased proportionately to reject the Moon signal.

The triggering voltage for selected bistable transistors is 0.7 volt. For best sensitivity the maximum usable gain is $\dfrac{0.7 \text{ volt}}{9.5 \times 10^{-7} \text{ volt}} = 74{,}000.$ A higher gain than this cannot be used, because the noise might then be amplified to such an extent that it would cause random triggering of the circuit. For a chosen triggering level for the Earth sensor of one-fourth the full bridge output, or 250 microvolts, the amplifier should be designed for a gain of

$$\frac{0.7 \text{ volt}}{2.5 \times 10^{-4} \text{ volt}} = 2,800$$

Low-noise silicon transistor amplifiers are available with gains of 8,000. This gain can be adjusted to 2,800 or, if a higher gain is necessary, as in the Moon sensor, additional stages of amplification can be used.

The minimum available signal from the Moon for altitudes up to 65,000 miles (fig. 9) is $4.2 \times 10^{-7}$ watt. For a responsivity of 175 volts/watt the corresponding thermistor output is $7.4 \times 10^{-5}$ volt. Again, the bridge output is half this value or 37 microvolts and the minimum detectable signal-to-noise ratio is 5. Horizon detection can thus be achieved at a thermistor output voltage of 5 times the noise voltage, or 9.5 microvolts. Triggering can thus take place when 9.5/37 or about one-fourth of the field of view is covered by the Moon. Under these conditions, the required amplifier gain for the Moon sensor will be 74,000.

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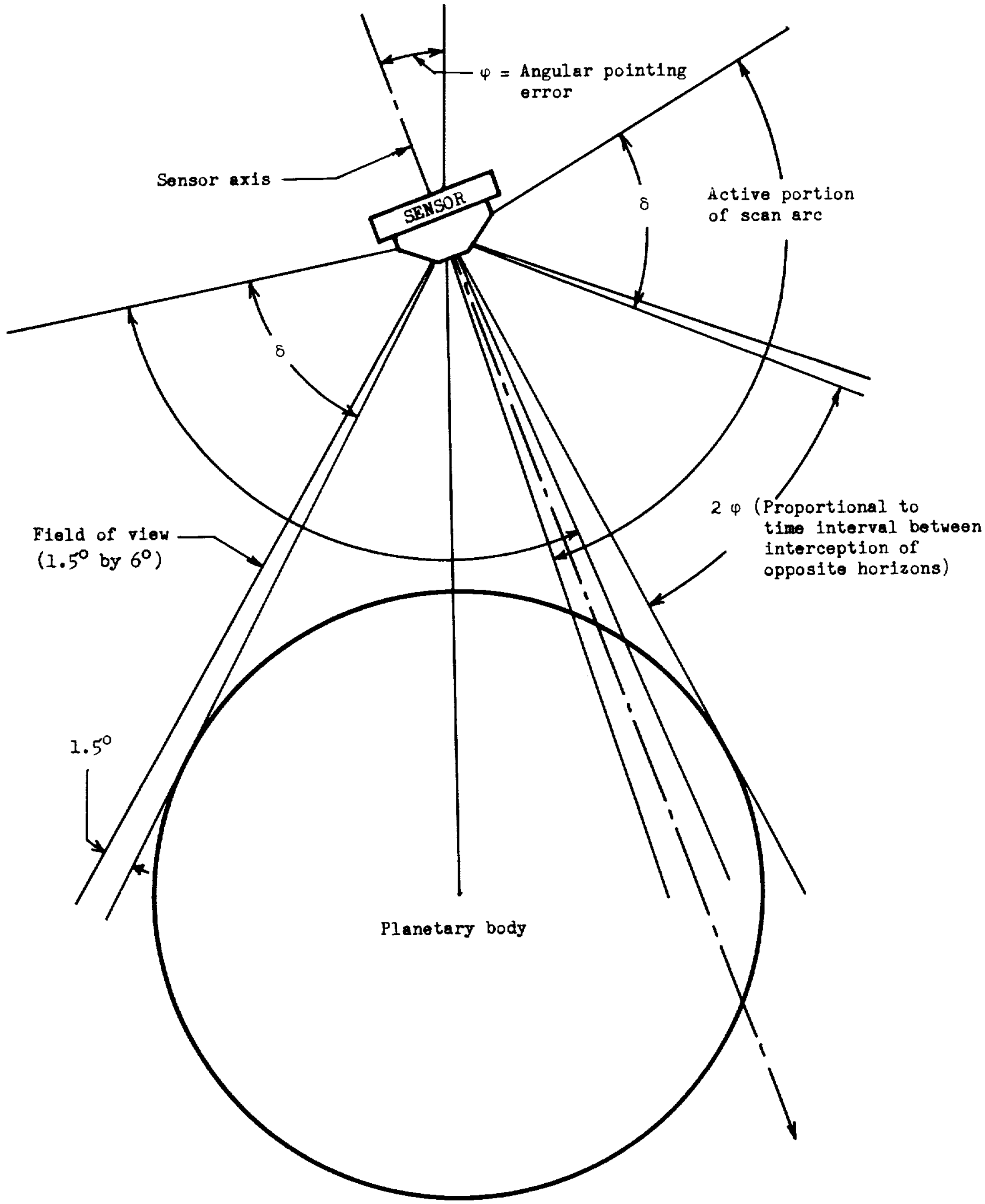


Figure 1.- Pointing-error determination.

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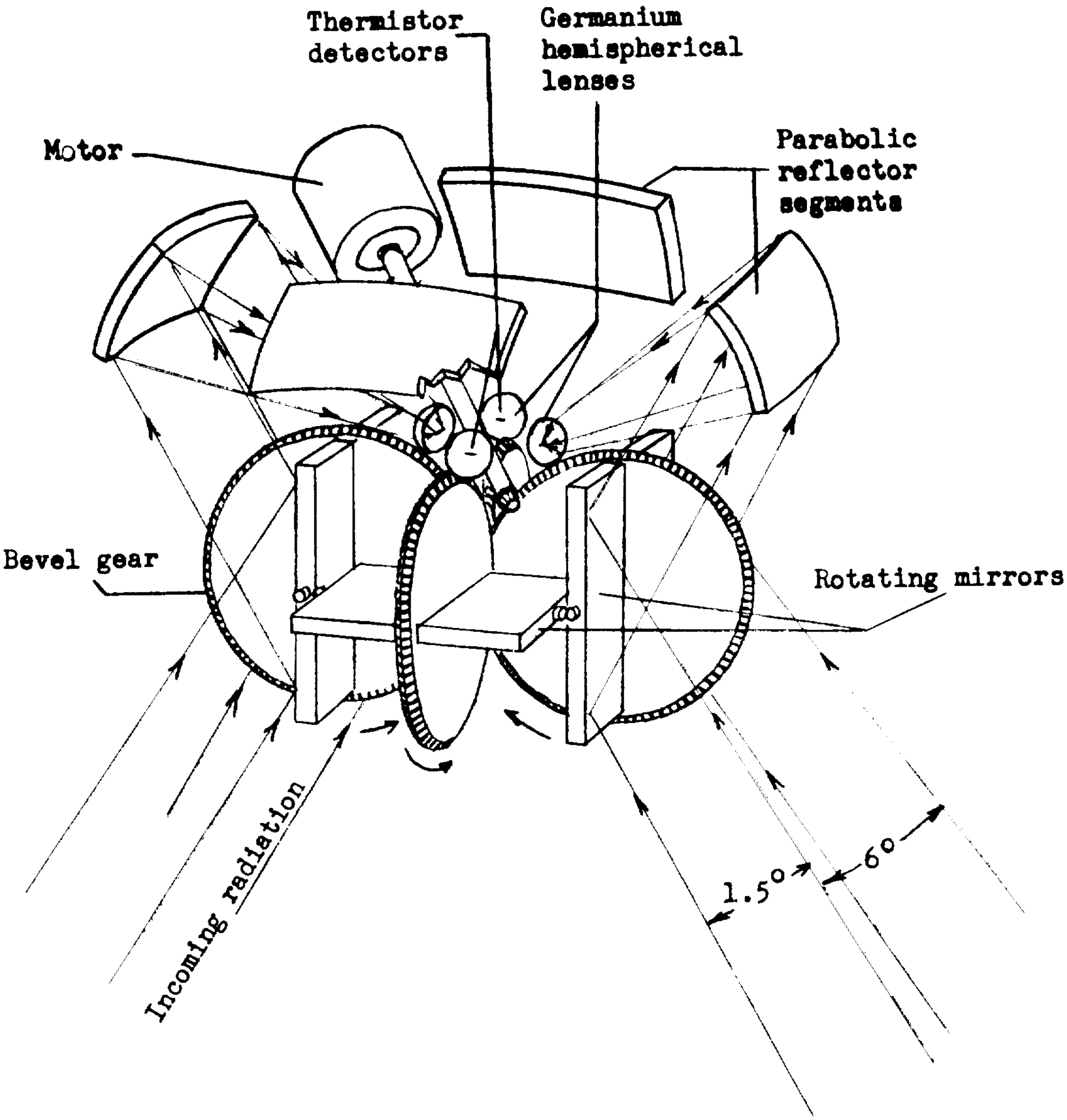


Figure 2.- Optical system.

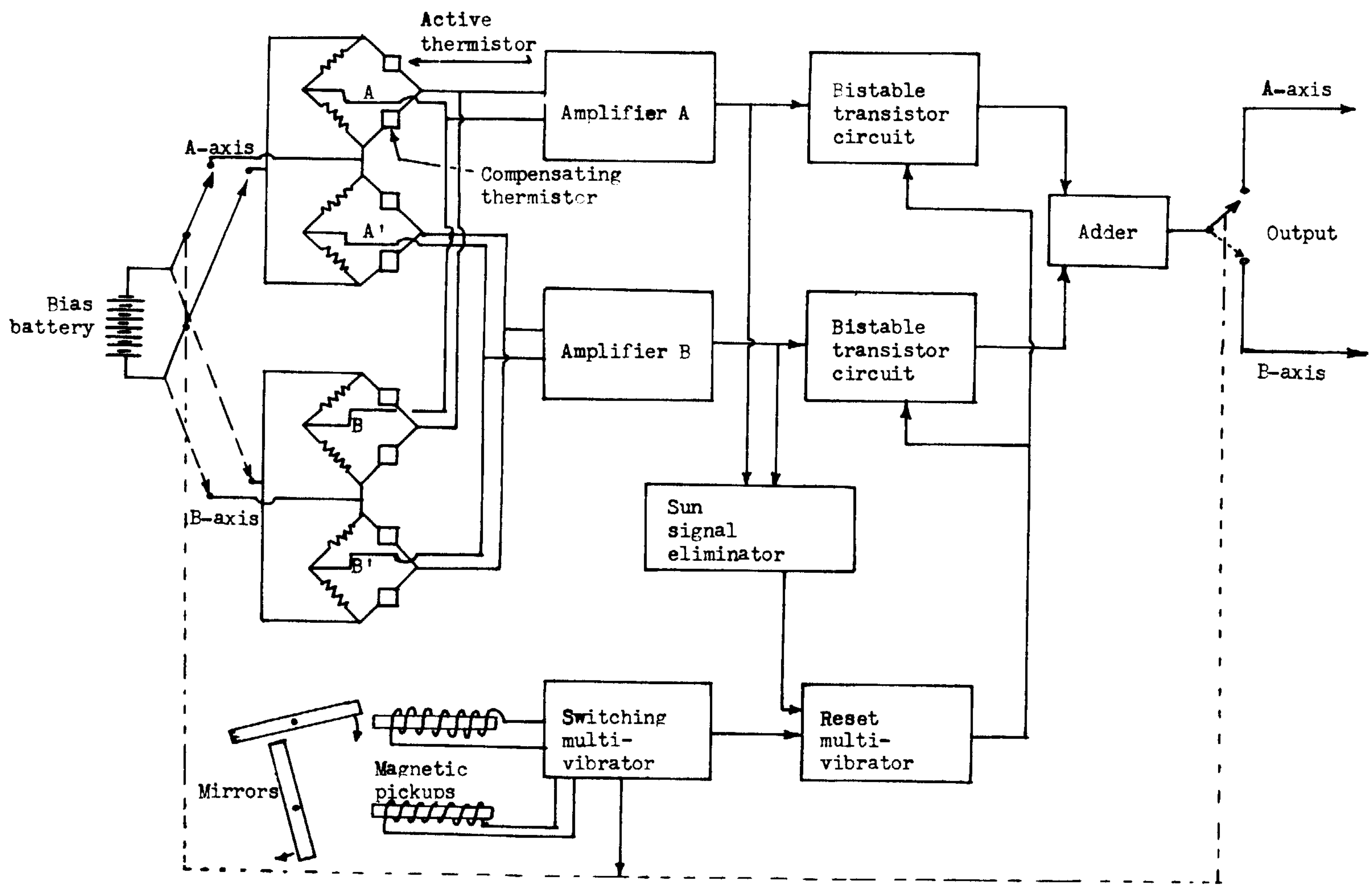


Figure 3.- Circuit block diagram.

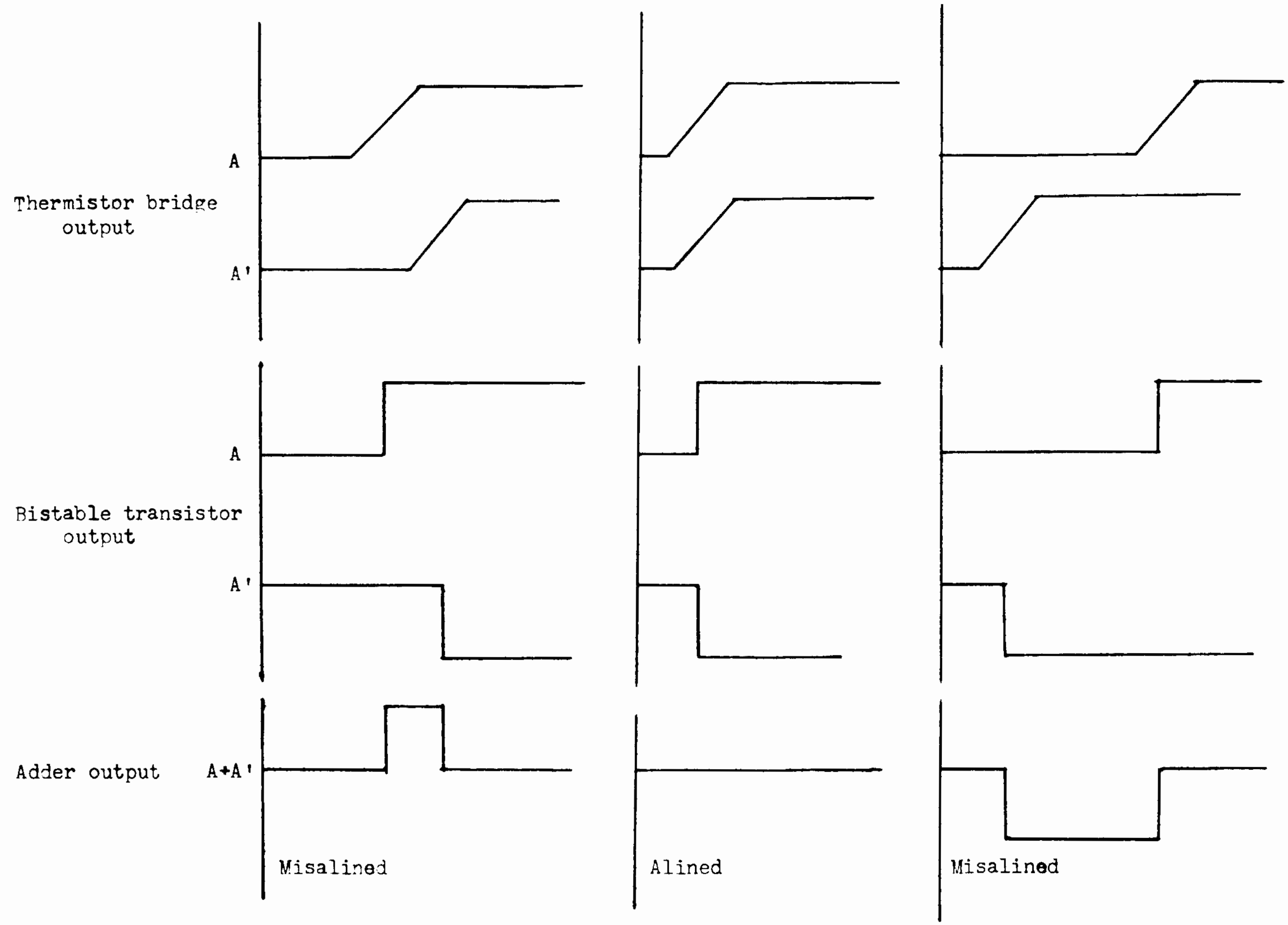


Figure 4.- Sensor signals for axis A-A'.

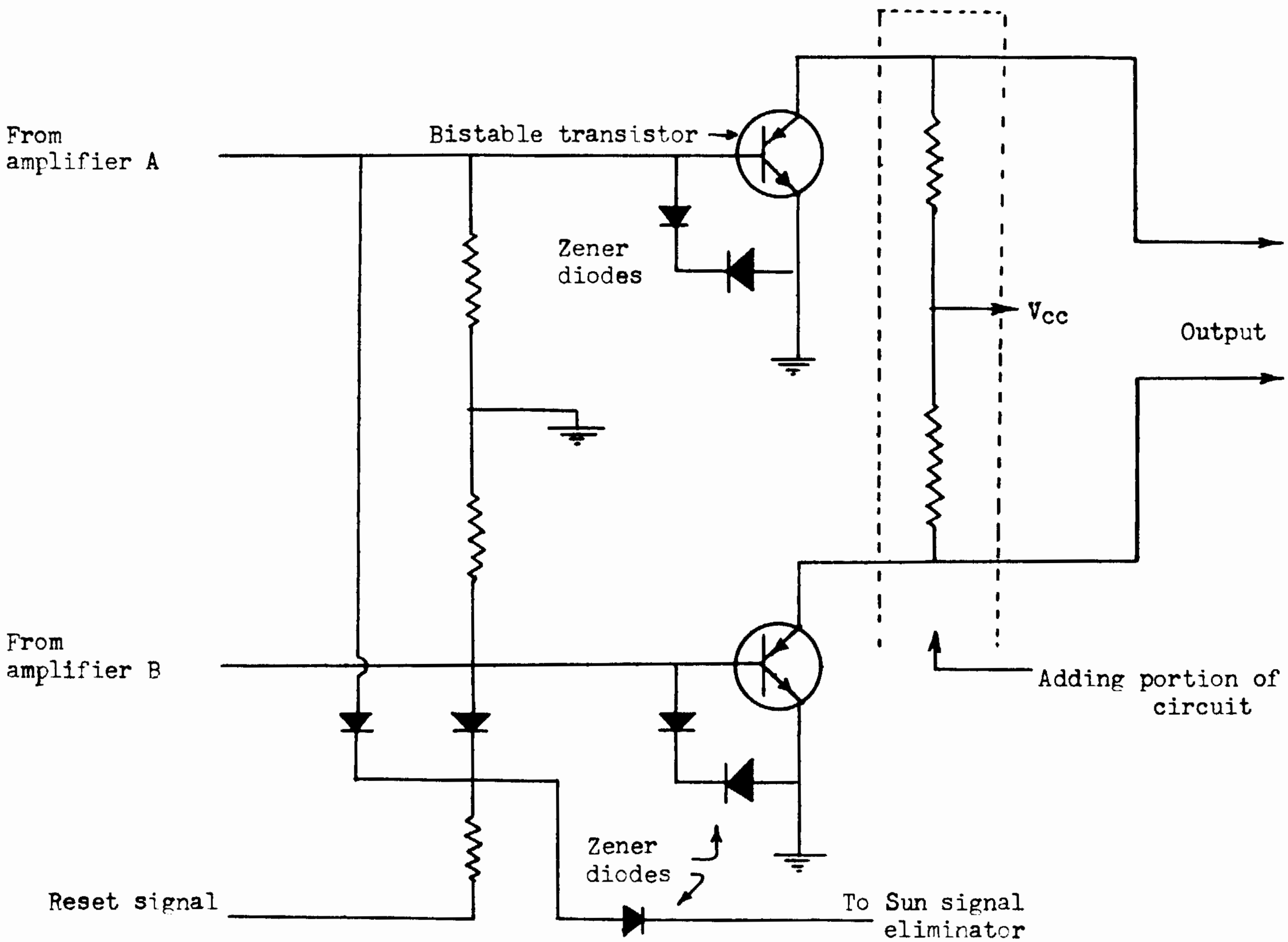


Figure 5.- Triggered bistable transistor circuit with adder.

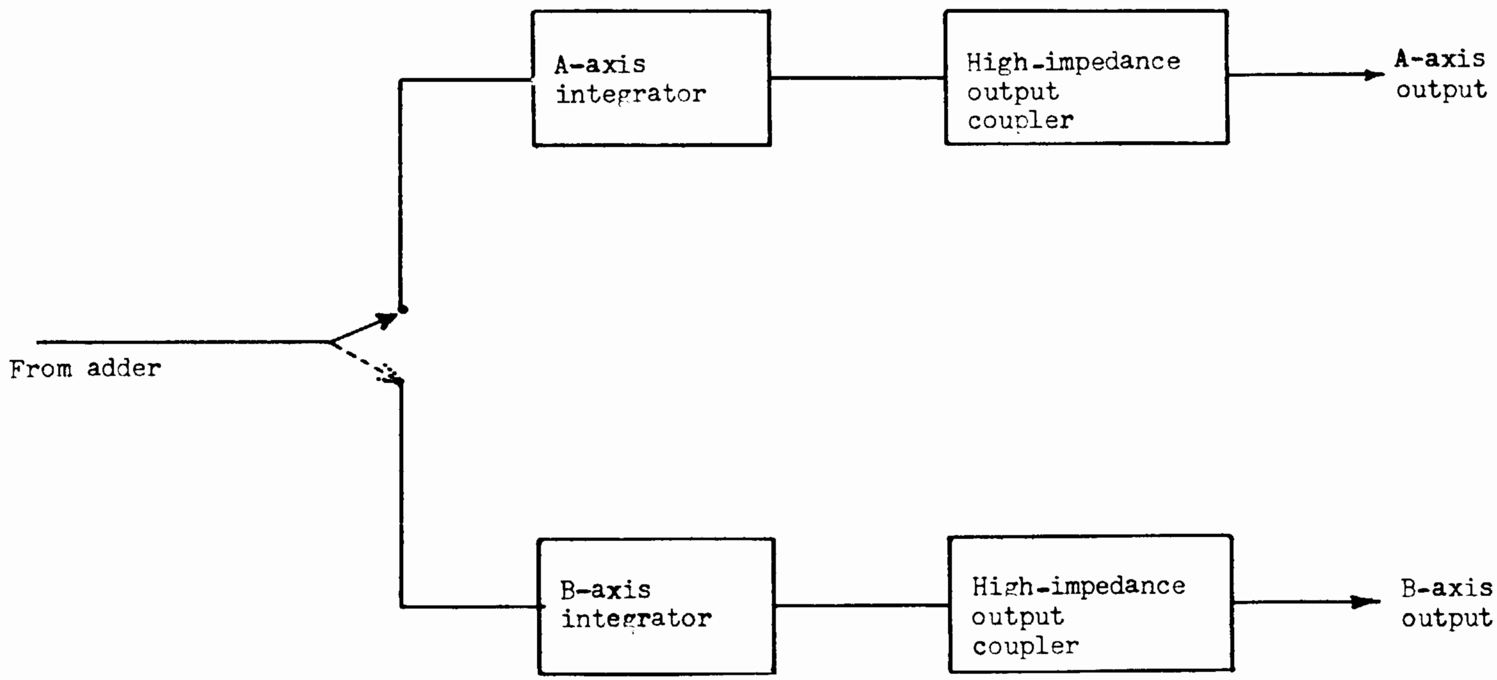


Figure 6.- Block diagram of pulse integrator and output coupler.

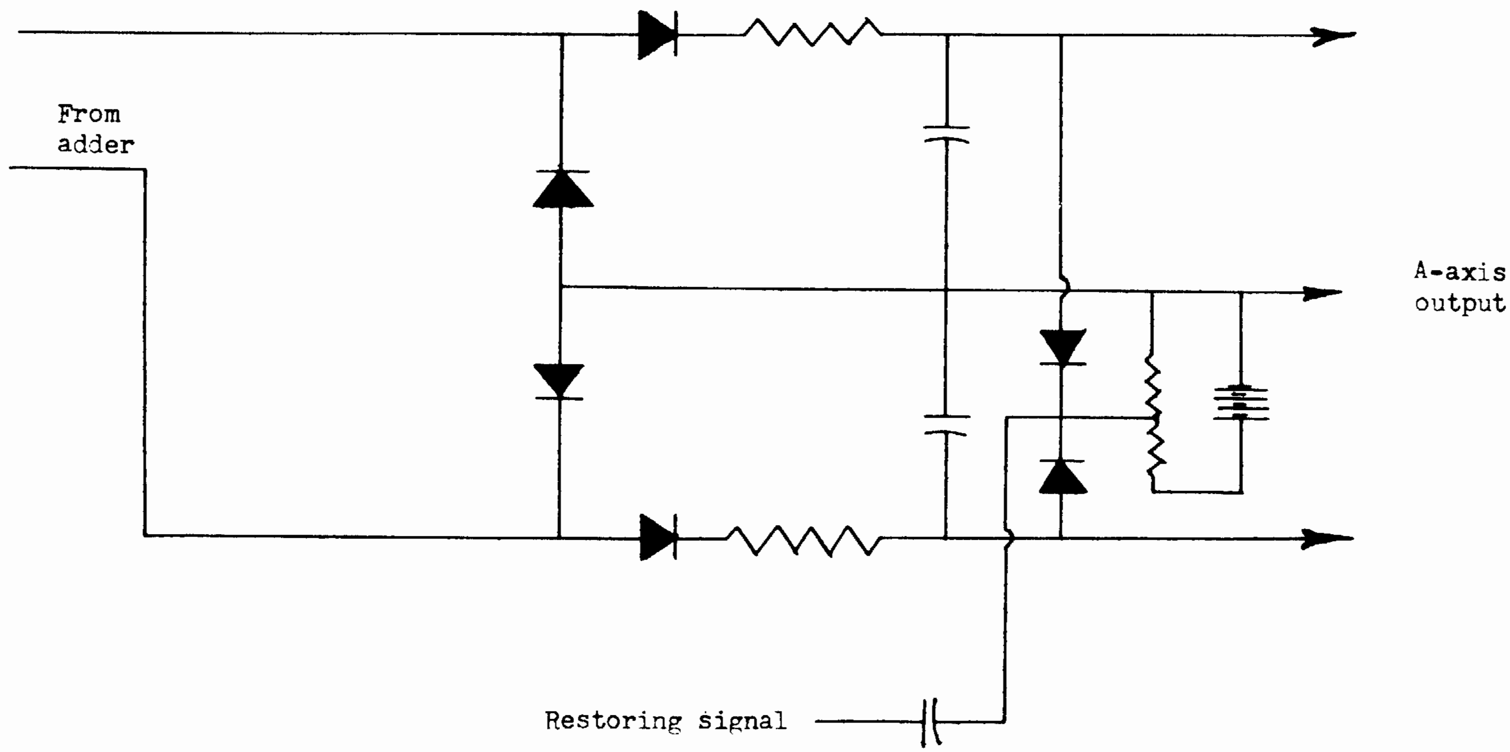


Figure 7.- Integrator circuit.

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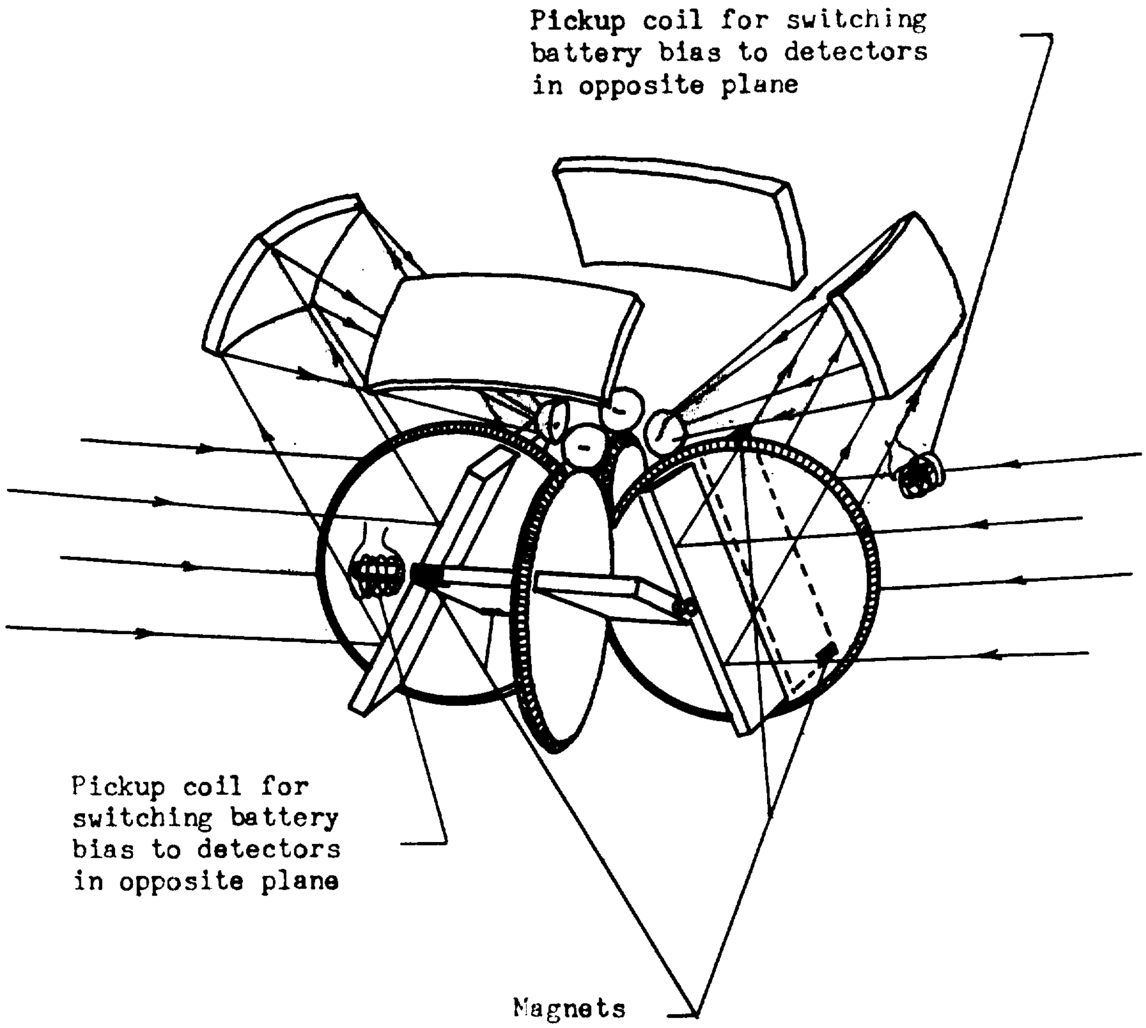


Figure 8.- Location of magnets and pickups for providing a switching signal.

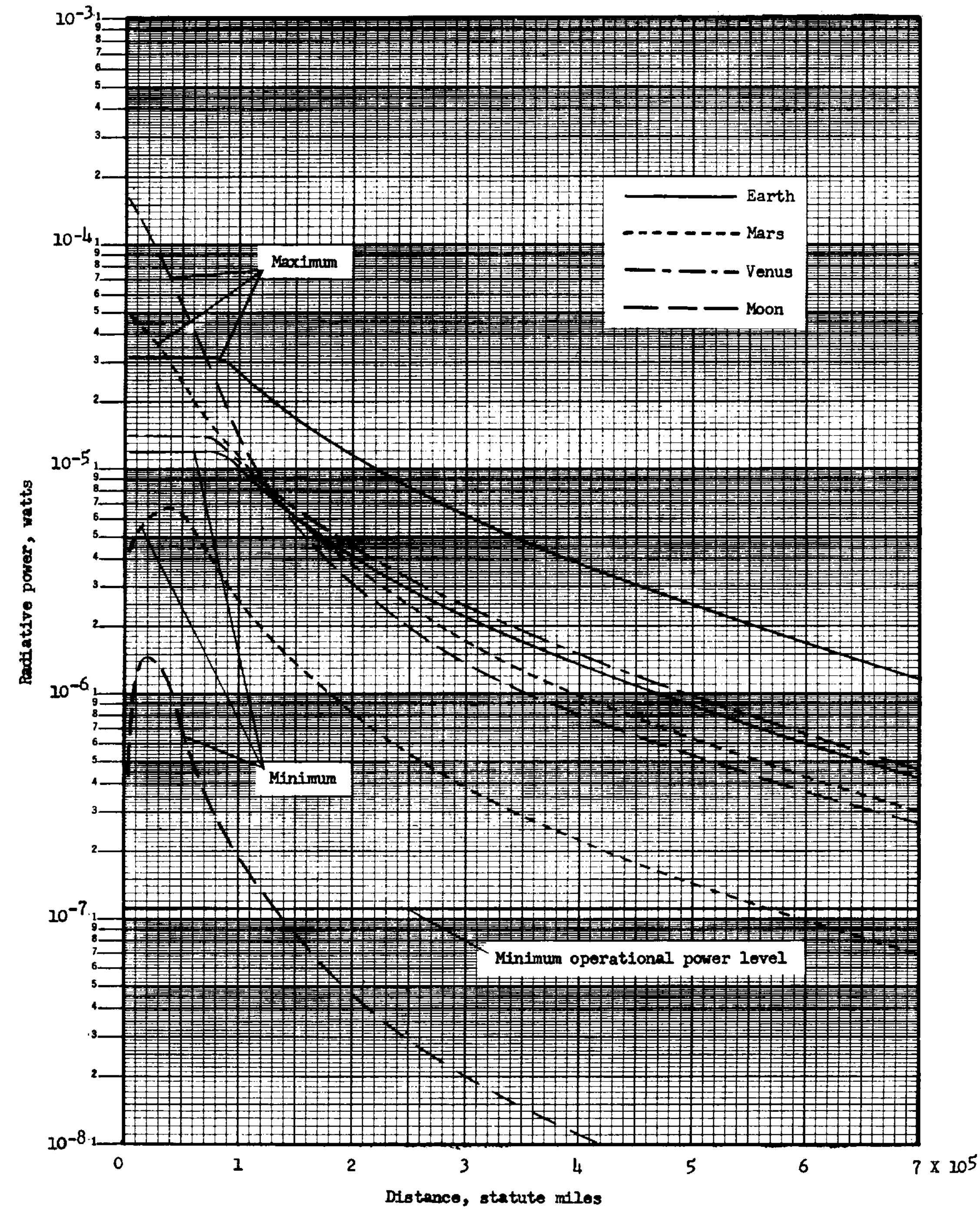


Figure 9.- Variation of maximum and minimum radiative power received by detector with distance from center of planetary body. Average curve given for Venus.

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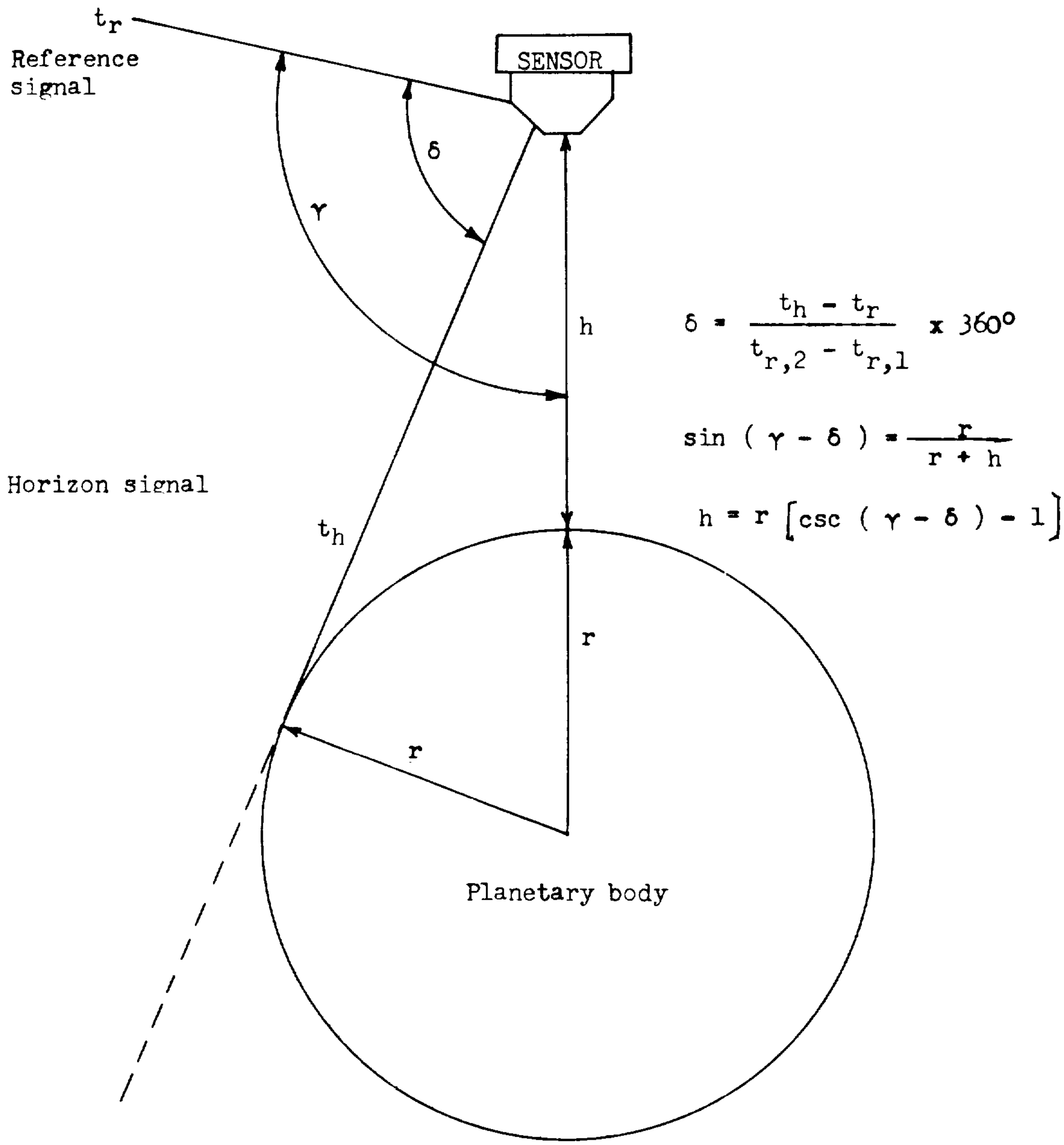


Figure 10.- Altitude determination. $(t_{r,2} - t_{r,1}$ is the time between successive reference signals.)

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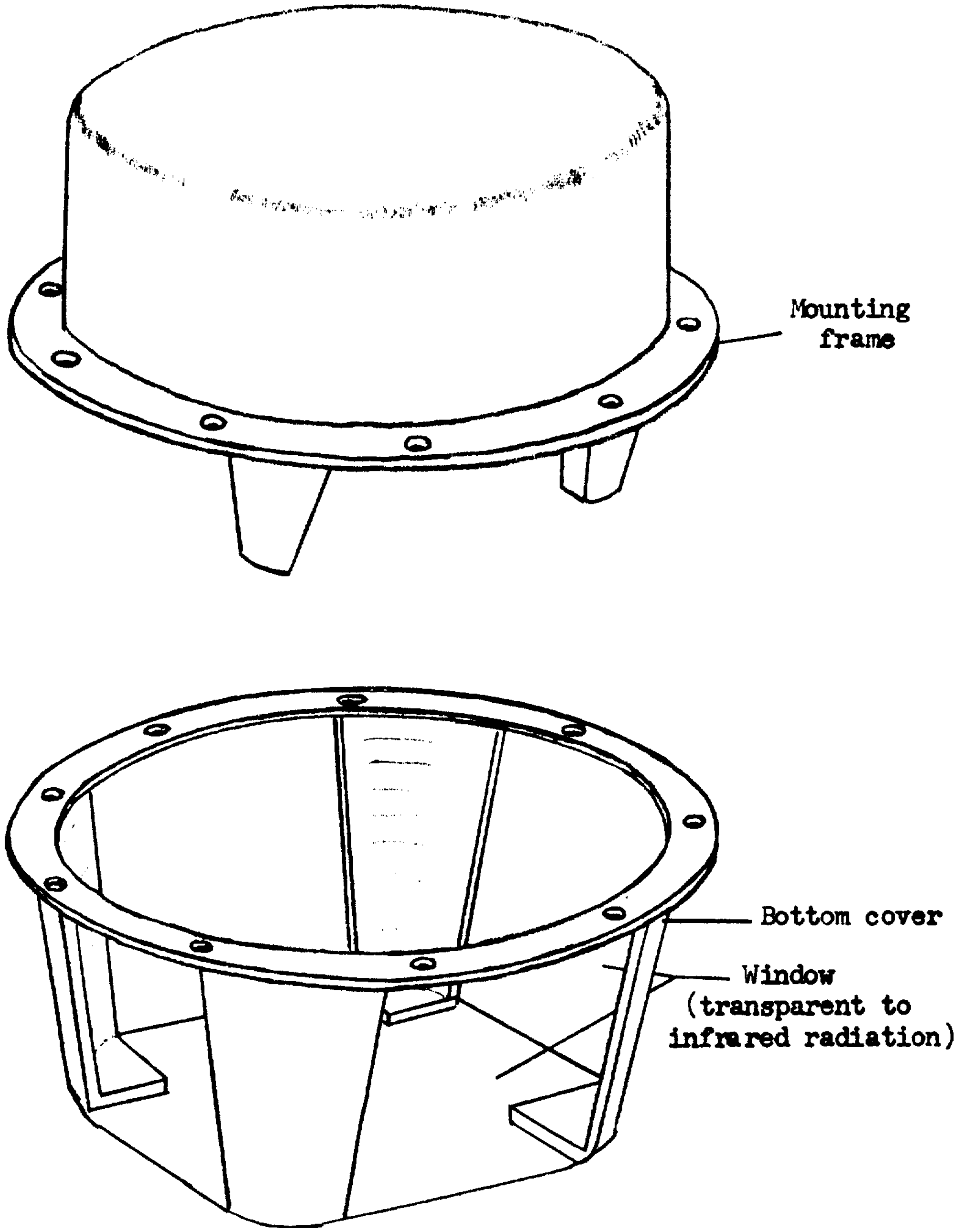


Figure 11.- Bottom-cover configuration.

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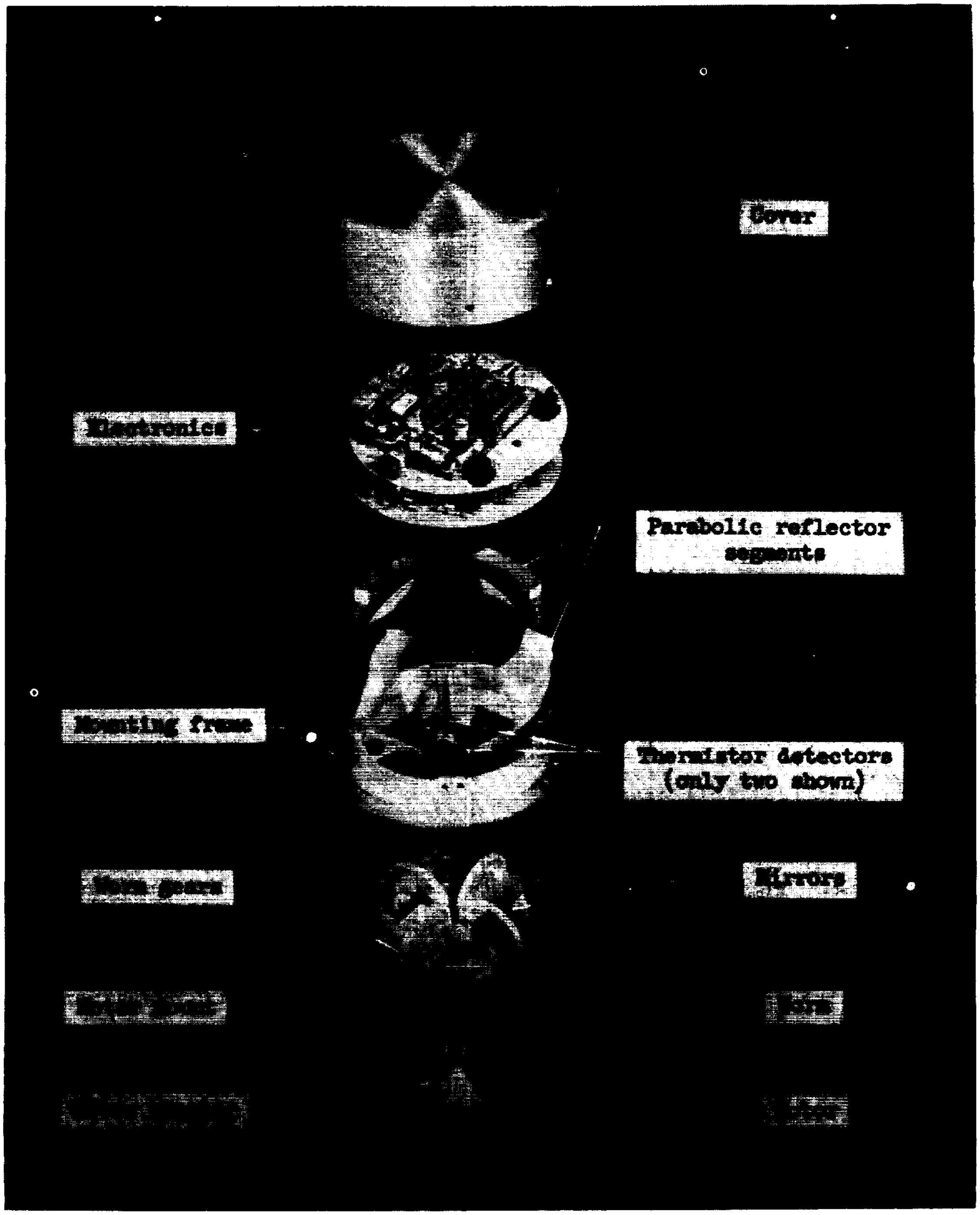


Figure 12.- Exploded view of experimental sensor. L-60-6511.1

Figure 13.- Experimental sensor. L-60-6509

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