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RADIO/OPTICAL/STRAPDOWN INERTIAL GUIDANCE STUDY FOR ADVANCED KICK STAGE APPLICATIONS

FINAL REPORT

Volume III — State-of-the-Art Surveys
Part I — Electro-Optical Sensors
Part II — Strapdown Inertial Instruments and Sensor Assemblies

30 JUNE 1967

Contract No. NAS 12-141

Prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

ELECTRONICS RESEARCH CENTER

Cambridge, Massachusetts

TRW SYSTEMS GROUP

ONE SPACE PARK · REDONDO BEACH, CALIFORNIA

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FOREWORD

This report presents the results of a nine-month study of "Radio/Optical/Strapdown Inertial Guidance Systems" for future NASA unmanned space missions, conducted by TRW Systems for NASA/Electronics Research Center, Contract NAS 12-141.

The broad objectives of this study were to:

- Establish the guidance requirements for a selected group of future NASA unmanned space missions.
- Investigate possible guidance concepts based on the appropriate use of radio, strapdown inertial, and optical techniques, with the further objective of establishing the proper functional role, the capabilities, limitations, and constraints of each of these elements in the overall guidance system concept.
- Define feasible radio/optical/strapdown inertial guidance system design concepts and equipment configurations.
- Perform analyses to establish the feasibility (performance) of the selected design concepts.
- Indicate areas of technology where state-of-the-art advances are necessary.

Volume I summarizes the entire study, conclusions, and recommendations. Volume II describes the detailed findings that support these conclusions. Supplementary material is presented in Volume III (surveys of electro-optical sensors and of inertial instruments) and in Volume IV (classified sensor data).

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Part I: SURVEY OF ELECTRO-OPTICAL SENSORS

VOLUME III, PART I

1. INTRODUCTION AND SUMMARY

In this part of Volume III, TRW Systems presents the results of a survey of the state-of-the-art in electro-optical instruments for use in optically aided strapdown inertial guidance systems for future unmanned space missions The four missions considered were:

1) Synchronous Earth orbiter

2) Mars orbiter

3) Lunar orbiter

4) Solar probe using a Jupiter assist.

The survey considered state-of-the-art electro-optical devices potentially feasible for these missions, including body-fixed star mappers (for three-axis attitude determination), planetary horizon sensors, and the more conventional sensors.

The electro-optical devices augment the strapdown inertial system by:

1) Updating the inertial system alignment and bounding the errors due to gyro drift.

2) Functioning as navigation sensors in which data from the electro-optical sensors is processed in an on-board computer to provide complete or partial updating of the vehicle position, velocity, and alignment data.

The specific candidate sensors selected and the guidance configurations defined were analyzed to determine the feasibility of meeting the accuracy requirements of the four specified missions. (These analyses are presented in Volume II.)

1) Sun sensors, including both nulling devices and solar-aspect sensors.

2) Earth sensors, including both horizon sensors for use in earth orbit and long-range sensors for use in interplanetary flight.

3) Star trackers, including both gimbaled and strapdown subsystems, using both mechanical and electronic scanning, and photoelectric or solid-state optical radiation detectors.

4) Star field sensors, using photoelectric and solid-state detectors, with either mechanical or electronic scanning techniques.

5) Planet sensors, for terminal approach or orbit, employing both mechanical and electronic scanning.

The information for this survey has been obtained directly from manufacturers and research laboratories, in which both the current state-of-the-art and projected advancements in the near future have been determined. Additional information was also obtained from the Aerospace Corporation. Only nonproprietary data has been included.

In addition, previous information compiled under the USAF Standardized Space Guidance System studies was also reviewed. It was found that a considerable number of subsystems covered by the SSGS surveys were not applicable for use in spacecraft. For example, numerous instruments have been developed for experimental programs conducted in laboratories, balloons, rocket probes, and high-altitude research aircraft. Instruments which were found to be nonemployable because of their excessive weight, volume, power, or complexity have been deleted. In particular, a large number of star tracking subsystems were found to be nonapplicable. A summary table of these is included, with the basis for rejection defined.

Earth and planetary landmark trackers were excluded from this survey by mutual agreement of NASA-ERC and TRW Systems representatives on 18 August 1966.

1.1 SUN SENSORS

Numerous sun sensors have been developed, with many being space-qualified. The wide variety of existing devices includes analog null sensors, digital solar aspect sensors, and attitude sensors for spinning spacecraft. Although some of these devices have been configured for a specific spacecraft application, a considerable number of off-the-shelf items are available and were considered for use in the subsequent design phases of this study. Where several nearly identical items have been found, only representative types have been included. For the Earth, Lunar, and Mars missions, the use of existing devices was anticipated. For the Jupiter mission, however, the sensitivity of sun sensors at the mean distance from the Sun of 5.20 AU required examination to determine if the sensitivity was adequate.

1.2 EARTH HORIZON SENSORS

Although a very large number of earth horizon sensors have been developed, the parameters of these devices are largely dependent upon the orbital altitude of the spacecraft. In addition, evolutionary changes have been made as additional knowledge of the radiance and spatial characteristics of the earth's atmosphere has been obtained. The most recent devices have been designed for use in the 12.5- to 16-micron carbon-dioxide absorption band, with the spectral passband being cut off below 14 microns. Instruments with inherently high accuracy have been developed, the most accurate having performance characteristics which are classified. However, in operation, the predominant error source is the spatial nonuniformity of the earth's atmosphere. Except for TRW's Reliable Earth Sensor and the Barnes Lunar and Planetary Horizon Sensor, earth horizon sensors utilize either mechanical devices or spacecraft spin to accomplish scanning. The direct application of existing equipment to the guidance systems configured in this study is dependent primarily upon the altitude of earth orbit, and accuracy and reliability requirements.

1.3 INTERPLANETARY EARTH SENSORS

Only two sensors have been developed for sensing the earth from interplanetary distances: the NASA/JPL long- and short-range earth sensors used for the Ranger and Mariner programs. Both employ end-on photomultiplier tubes, the short-range sensor utilizing static shadow-mask modulation and the long-range sensor employing a vibrating reed scanner. The maximum range limitations are one and fifty million miles, respectively. The application of these specific sensors requires consideration of the range and the reliability limitations of existing equipment.

For missions in the near-future, a star tracker using an image dissector, modified for Earth sensing, may be considered. For the solar probe with Jupiter assist, development schedules may permit consideration of a solid-state device.

1.4 STAR TRACKERS

An extremely large number of star tracker subsystems have been developed for military and experimental research programs. Most of these are not applicable to space missions because of their excessive complexity, or weight, volume, and power requirements. However, a number of additional systems specifically developed for spacecraft are space-qualified or operational. In the survey, these sensors were divided into two categories: those using mechanical scanning and those employing electronic scanning. The former category includes the equipment used on the OAO and Surveyor programs, as well as several subsystems using solid-state detectors. The latter category includes strapdown subsystems used on the Ranger, Mariner, and Lunar Orbiter programs, all of which utilize image-dissector-type photoelectric-photomultiplier tubes.

It is anticipated that an image-dissector subsystem may fulfill the star tracker requirements of the near-future missions. However, the development schedule will permit the consideration of devices currently in the research or developmental stages. The solid-state image sensor being developed by RCA Princeton Laboratories appears attractive, if the detectivity can be improved. The cadmium sulfide panel, interrogated by an electroluminescent matrix (being developed by Belock Instrument Corporation), may also be applicable.

1.5 STAR FIELD SENSORS

The earliest development in this field was the Star Angle Comparator developed by the Kearfott Division of General Precision, Inc., for the USAF Avionics Laboratory. One experimental model was developed and the feasibility of star identification by measurement of angular separation was demonstrated. The equipment was rather large, however, being designed only for laboratory evaluation. Also, it employed large rotating components and would require considerable additional development for space flight.

General Electric Company developed an experimental unit in 1962, utilizing simple pattern correlation. However, any misalignment greater than 20 percent between the pattern mask and the optical image of the star field could not be tolerated.

A breadboard unit developed by the Advanced Technology Division of American Standard, in which identification was to be accomplished by analysis of the electronic frequency spectrum of the star field pattern, has not proved to be successful.

The ITT "Starpat" equipment is a scanner only and does not accomplish self-contained star identification. Ground data analysis is required. The Honeywell Passive Star Scanner equipment utilizes vehicle spin to accomplish scanning and also requires ground data reduction.

Control Data Corporation has developed a very compact unit designed for use with an on-board computer. However, either a rotating slit reticle or spacecraft rotation is required for scanning.

The Berlock Instruments Corporation's solid-state device mentioned in 1.1.4 above is the only equipment which can be considered as completely solid-state. Currently in the developmental stage, this device may be applicable.

The RCA solid-state image sensor may also be considered for future missions if the detectivity can be improved.

In general, the applicability of star field sensors for spacecraft attitude control will be the subject of additional investigation during the subsequent phases of this study program.

1.6 PLANET SENSORS

Development of sensors for use in planetary approach or orbital guidance has been relatively limited. One subsystem in operational use on the Mariner spacecraft is manufactured by Barnes Engineering Company. Both Barnes and Northrop-Nortronics have developed experimental equipment for use in lunar orbit. Lockheed Missiles and Space Company is currently developing an experimental model of a fine alignment scanner for use on the IR-OAO, to be used for scanning of Venus, Mars, and Jupiter from Earth orbit.

To date, no planet sensor has been developed for use in approach guidance to Jupiter.

2. INSTRUMENT SURVEY

2.1 SUN SENSORS

The sun sensors described in this section are representative of the equipment presently available, in use, or intended for use on contemporary space vehicles. The survey results do not include information on all sun sensors available from each manufacturer. However, the characteristics of representative devices are defined.

2.1.1 General Discussion

Sun sensors fall into three general categories:

1) Null seekers
2) Aspect sensors
3) Sun present sensors

In some cases a combination of one or more is employed. Each category is discussed briefly below.

2.1.1.1 Null Seekers

A null-seeking sun sensor is characterized by two analog output signals which accurately define the attitude of the line to the sun about two axes of rotation by providing two electrical null signals. A single-axis null seeker indicates when the line to the sun lies in a plane. When the sun is off-axis, the output signals primarily provide phase information. In some cases the signals are relatively linear with angular displacement. The linear characteristic is usually used to derive the rate of change of angular displacement by electronic differentiation rather than to provide an accurate measure of the angular displacement.

The most common application of a null seeker is as the primary sensor in a servo control system which accurately points an energy conversion device (or a solar experiment) towards the sun. Where a large field-of-view is required for initial acquisition of the sun, both coarse and fine sensors may be used. This is required when the total field-of-view and null accuracy requirements are not compatible.

2.1.1.2 Aspect Sensors

Aspect sensors measure the attitude of the line-of-sight to the sun with respect to spacecraft coordinates. To achieve the required accuracy, aspect sensors are normally digital devices, and in some cases employ moving parts. The latter type normally consists of a narrow field-of-view null seeker mounted on a gimbal structure with digital encoders for measurement of gimbal angles.

2.1.1.3 Sun-Presence Sensors

These sensors provide an electrical signal which indicates when the sun is within the optical field-of-view. They do not provide information defining the location of the sun. The output signal can be either digital or analog. A sun-presence sensor employing a photosensitive silicon-controlled rectifier provides a digital output in the form of a two-state output signal.

2.1.1.4 Types of Detectors Used in Sun Sensors

Since 70 percent of the solar radiation falls in the wavelength band between 0.3 and 1 micron, all of the detectors used in sun sensors have their maximum spectral response in this band. The most commonly used detector used is the silicon solar cell. It is normally used in the photovoltaic mode and therefore requires no input power. Other detectors are cadmium-sulfide or cadmium-selenide photoresistors and photosensitive silicon-controlled rectifiers.

2.1.2 Sun Sensor Summary

Presented below is a brief discussion of each sun sensor. Detailed information on the associated electronics is not provided since signal processing and logic circuits are normally designed to fit a specific application and system configuration. The electronics are usually not contained within the sun sensor package.

Most of the available sun sensors are of the analog null seeker type and are used as primary sensors in spacecraft attitude control systems. The digital aspect sensors manufactured by Adcole Corp. and Bendix, Eclipse-Pioneer Division, are the only all-digital devices known to exist at this time. The Bendix and Honeywell AOSO fine sun sensors are examples of very high accuracy devices. These are more sophisticated than the simple null seekers and employ moving parts. Characteristics of the surveyed sun sensors are summarized in Table 2-I.

2.1.2.1 Digital Solar Aspect Sensors

Manufacturer: Adcole Corporation
Waltham, Massachusetts

Functional Description:

Adcole Corporation manufactures a large number of digital solar aspect sensors. Physical and performance data of several are given below. The sensors consist of detector heads and electronics. A single-axis detector head consists of a Gray-coded reticle, silicon photocells, and a housing. The Gray-coded reticle is a small oblong block of fused quartz with a slit centered along the top surface and a Gray-coded pattern on the bottom surface. Sunlight passing through the slit is screened by the pattern to either illuminate or not illuminate the photocells below. The outputs of the photocells comprise a digital word representative of the solar aspect angle about one axis. An additional photocell is usually included which is always "ON" when the sun is within a field-of-view of the detector head. The output of this cell is used as an AGC signal to compensate for the photocell outputs as a function of solar angle. This permits accurate angular determination of the transition between resolution elements. A two-axis detector head combines two of the above assemblies in a single package.

Physical and Performance Data:

Model	1301	1401	1402
Field-of-View	$128^{\circ} \times 1^{\circ}$	$128^{\circ} \times 128^{\circ}$	$64^{\circ} \times 64^{\circ}$
Resolution	1°	$1/2^{\circ}$	$1/64^{\circ}$
Accuracy*	$1/4^{\circ}$	$1/4^{\circ}$	2 arc min
Output	one 7-bit word	two 8-bit words	two 12-bit words
Operating Temperature Range		$-70^{\circ}C$ to $+100^{\circ}C$	

*Accuracy of angle determination at transition between resolution elements.

	Manufacturer	Designation	Developmental Status	Output Characteristic	Detector	Spectral Range	Optics
1	Adcole Corp.	Aspect Sensor No. 1301	Space Qualified	Digital One 7-Bit Word (Single Axis)	One Silicon Cell/Bit	0.4 to 1.1 Micron	Slit Apertures and Gray Code Mask
2	Adcole Corp.	Aspect Sensor No. 1401	Space Qualified	Digital Two 8-Bit Words (Two Axes)	One Silicon Cell/Bit	0.4 to 1.1 Micron	Slit Apertures and Gray Code Mask
3	Adcole Corp.	Aspect Sensor No. 1402	Space Qualified	Digital Two 12-Bit Words (Two Axes)	One Silicon Cell/Bit	0.4 to 1.1 Micron	Slit Apertures and Gray Code Mask
4	Ball Brothers Research Corp.	OSO Coarse Sun Sensor No. CE-3	Operational	Analog Cosine For Acquisition (Single Axis)	One Silicon Solar Cell	0.6 to 1.1 Micron	Lens
5	Ball Brothers Research Corp.	OSO Fine Sun Sensor NO. FE-3	Operational	Nulling, Analog Error Signal (Single Axis)	One Silicon Solar Cell	0.6 to 1.1 Micron	Lens
6	Ball Brothers Research Corp.	Disabling Eye No. TE-4	Space Qualified	Analog On-Off (Indicates Sun Presence)	One Silicon Solar Cell	0.6 to 1.1 Micron	Lens
7	Bendix Corp. Eclipse - Pioneer Division	Digital Aspect Sensor No. 1818775	Space Qualified	Digital (Single Axis)	Silicon Cell Array	0.4 to 1.1 Micron	Slit Aperture and Encoded Mask
8	Bendix Corp. Eclipse - Pioneer Division	Fine Angle Sensor No. 1818823	Space Qualified	Nulling, Analog Error Signal (Two Axes)	Coarse and Fine Silicon Quadrant Arrays	0.4 to 1.1 Micron	Lens
9	Bendix Corp. Eclipse - Pioneer Division	Wide Angle No. 1771858 and 1818787	Space Qualified	Nulling, Analog Error Signal (Two Axes)	Coarse and Fine Silicon Cell Arrays	0.4 to 1.1 Micron	Shadow Structure (Coarse) Square Aperture (Fine)
10	H.H. Controls Co., Inc.	Refractosyn Sun Sensor No. S-4	Space Qualified	Nulling, Analog Error Signal (Single Axis)	Two Silicon Cells	0.4 to 1.1 Micron	Two Critical Angle Prisms
11	Honeywell Aeronautical Division	AOSO Fine Sun Sensor	Experimental	Nulling, Analog Error Signals (Two Axes)	Two Silicon Solar Cells	0.4 to 1.1 Micron	Counter Rotating Wedges and Two Critical Angle Prisms
12	ITT Fed. Labs	Nimbus "D" Sun Sensors	In Development	Null, Analog Error Signal (Two Axes)	NS	0.2 to 0.3 μ	NS
13	NASA/JPL and Northrop-Nortronics	Mariner and Ranger Sun Sensors	Operational	Nulling, Analog Error Signals (Two Axes)	Cadmium Sulfide Photo-conductors	0.55 Micron Peak	Shadow Structure
14	TRW Systems	OGO Sun Sensor	Operational	Nulling, Analog Error Signals (Two Axes)	Radiation Tracking Trans-ducer* (Fine) Silicon Cells (Coarse)	0.4 to 1.1 Micron	Pinhole (Fine) None (Coarse)
15	TRW Systems	Pioneer Sun Sensors	Operational	Bistable Sun Present Indication	Photo-sensitive Silicon Controlled Rectifiers	0.4 to 1.1 Micron	Shadow Structure
16	TRW Systems	Intelsat III Sun Sensor	In Development	Analog Pulses (One Axis)	One Silicon Solar Cell	0.4 to 1.1 μ	Slit Apertures and Pinhole
17	TRW Systems	VASP Sun Sensors	In Development	Null, Analog Error Signal (Two Axes)	Six Silicon Solar Cells Per Assembly	0.4 to 1.1 μ	None (B Sensor) Lens (A and C Sensors)
18	TRW Systems	823 Sun Sensor	Operational	Bistable Sun Present Indication	Photo-sensitive SCR	0.4 to 1.1 μ	Shadow Structure

NOTE: NS = NOT SPECIFIED NA = NOT APPLICABLE *REG. TRADEMARK I

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Table 2-I. Sun Sensor Survey

Modulation Method	Optical Field of View	Accuracy	Weight	Volume	Power	Remarks	References
Spin of Spacecraft	$128^{\circ} \times 1^{\circ}$	15 ARC-MIN	1.5 oz.	1.33 IN^3	None Required	Shift register for serial output requires 0.5 watt of power	"Sun Sensor Survey" TRW Memo 9354.3 - 448 by P.B.Hutchings - 14 July 1965
NONE	$128^{\circ} \times 128^{\circ}$	15 ARC-MIN	3.5 oz.	3.2 IN^3	None Required	Shift register for serial output requires 0.5 watt of power	"Sun Sensor Survey" TRW Memo 9354.3 - 448 by P.B.Hutchings - 14 July 1965
NONE	$64^{\circ} \times 64^{\circ}$	2 ARC-MIN	5.1 oz.	8.7 IN^3	None Required	Shift register for Serial output requires 1.0 watt of power	"Sun Sensor Survey" TRW Memo 9354.3 - 448 by P.B.Hutchings - 14 July 1965
NONE	2π Steradians	±5 DEG.	6.5 Grams	2.0 IN^3	None Required	Data is for single matched pair. Use with No. FE-3 for complete system	"Sun Sensor Survey" TRW Memo 9354.3 - 448 by P.B.Hutchings - 14 July 1965
NONE	$\pm 10^{\circ}$ (Pair)	±1 ARC-MIN At Null	6.0 Grams	2.5 IN^3	None Required	Data is for single matched pair. Have flown on OSO-1. To be used on OAO and AOSO.	"Sun Sensor Survey" TRW Memo 9354.3 - 448 by P.B.Hutchings - 14 July 1965
NONE	$\pm 6^{\circ}$ Cone	NA	6.0 Grams	2.2 IN^3	None Required		"Sun Sensor Survey" TRW Memo 9354.3 - 448 by P.B.Hutchings - 14 July 1965
Spin of Spacecraft	$\pm 64^{\circ}$ One Axis	±0.5 DEG.	2 oz.	1.36 IN^3	None Required		"Sun Sensor Survey" TRW Memo 9354.3 - 448 by P.B.Hutchings - 14 July 1965
NONE	$\pm 10^{\circ}$ Cone (Total) ±20 MIN Cone (Fine)	5 ARC-SEC.	30 oz.	50.5 IN^3	None Required		"Sun Sensor Survey" TRW Memo 9354.3 - 448 by P.B.Hutchings - 14 July 1965
NONE	2π Ster. (Coarse) 20° Cone (Fine)	NS	2.5 oz.	5.4 IN^3	None Required		"Sun Sensor Survey" TRW Memo 9354.3 - 448 by P.B.Hutchings - 14 July 1965
NONE	$\pm 100^{\circ}$	15 μA/ARC-MIN At Null	1 Gram	0.04 IN^3	None Required	20-sec response time requires external amplifiers	"Sun Sensor Survey" TRW Memo 9354.3 - 448 by P.B.Hutchings - 14 July 1965
Vibrating Aperture	$\pm 5^{\circ}$ Square (Pointing) ±20 ARC-MIN SQ. (Scanning)	±1.3 ARC-SEC.	12 lb (Pair)	315 IN^3 (Pair)	1.5 W. MIN 13 W. PK. (Pair)	Rotating prismatic wedges for line of sight deviation. Sensors used in pairs.	"Sun Sensor Survey" TRW Memo 9354.3 - 448 by P.B.Hutchings - 14 July 1965
NONE	$\pm 90^{\circ} \times \pm 60^{\circ}$	$\pm 1^{\circ}$	NS	NS	None Required	Six sensors required per S/C to obtain 4π steradian field-of-view on two axes	NASA RFP PC-732-85148-207
NONE	4π Ster. (Coarse) $\pm 1.5^{\circ}$ Sq. (Fine)	±0.1 DEG.	11 oz.	21.5 IN^3	0.8 W	Two coarse sensors and four fine sensors comprise one set	Northrop - Nortronics Brochure No. NORT 64-360 16 December 1964
NONE	4π Ster. (Coarse) $\pm 17^{\circ}$ Cone (Fine)	±0.2 DEG. (Fine) ±3.0 DEG. (Coarse)	2.1 lb	108 IN^3 (Coarse) 54 IN^3 (Fine)	None Required	Sensor consists of two packages; one located at extremity of two solar arrays	"Sun Sensor Survey" TRW Memo 9354.3 - 448 by P.B.Hutchings - 14 July 1965
Spin of Spacecraft	(a) 45° AZI $\times 85^{\circ}$ EL. (b) 90° AZI $\times 20^{\circ}$ EL. (c) 2° AZI $\times 40^{\circ}$ EL.	$\pm 2.5^{\circ}$ $\pm 2.5^{\circ}$ ±0.3 DEG.	0.9 lb (Set of 5)	77 IN^3 (Set of 5)	0.08 W MAX.	Four wide coverage sensors and one narrow angle "pipper" comprise one set on spinning spacecraft	"Sun Sensor Survey" TRW Memo 9354.3 - 448 by P.B.Hutchings - 14 July 1965
Spin of Spacecraft	$\pm 80^{\circ}$ (Aspect) $\pm 45^{\circ}$ (Azimuth)	±0.3 DEG. (Aspect)	0.4 lb	6 IN^3	None Required	Used on spinning S/C to measure angle between the SA and the sun vector by measuring time between analog pulses	W. N. Palser TRW Systems
Spin of Spacecraft	$\pm 80^{\circ} \times \pm 2^{\circ}$ (B Sensor) $\pm 70^{\circ} \times \pm 20^{\circ}$ (A & C Sensors)	$\pm 0.5^{\circ}$ B Sensor $\pm 0.9^{\circ}$ A & C Sensor	0.13 (Excluding Leads) (One Assembly)	18 IN^3 (One Assembly)	None Required	Two assemblies used on spinning spacecraft	R. N. Wagner TRW Systems
Spin of Spacecraft	$\pm 12.5^{\circ}$ (EL) $\times \pm 20^{\circ}$ (AZ)	±3° (EL) ±2° (AZ)	43 Grams	8 IN^3	<.004 Watts	One sensor per spinning spacecraft	W. A. Massey TRW Systems

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Model	1301	1401	1402
Size	1-3/16" x 1-13/16" x 17/32"	2-3/8" x 2-3/8" x 9/16"	3-1/4" x 3-1/4" x 13/16"
Weight	1.5 oz	3.5 oz	5.1 oz
Power	-----	None Required	-----

Additional Comments:

Physical data on the electronics for the sensors was not included above because it can vary appreciably depending on the particular application. An example, however, is that the electronics package needed to process and identify six detector heads weighs 2 lbs, is 7-3/4" x 4-1/2" x 3-7/8" in size, and consumes 0.8 watts of power.

Adcole Corporation digital solar aspect sensors have been qualified for two space programs.

2.1.2.2 Analog Sun Sensors

Manufacturer: Ball Brothers Research Corporation
Boulder, Colorado

Functional Description:

BBRC has developed a large number of analog sun sensors with individual characteristics that are used to build up sun sensor systems. BBRC sun sensors have flown on the OSO-1 spacecraft and are being developed for use on the OAO and AOSO spacecraft. Physical and performance data on representative BBRC sensors is defined below. The "coarse eyes" which have almost hemispherical acquisition capability are silicon solar cells suitably filtered and covered with a diffuser to form a cosine-law sensor. The "fine eyes" which are used in pairs to provide a single-axis electrical null, each consist of an objective lens, knife-edge reticle, filter, and silicon solar cell. The "disable eyes" are "sun presence" type sensors similar in construction to the "fine eyes" except that the knife-edge reticles are replaced with circular apertures.

Physical and Performance Data:

Type	Coarse Eye	Fine Eye	Disable Eye
Model	CE-3	FE-3	TE-4
Field-of-View	2π steradians	$\pm 10^{o}$ (pair)	$\pm 6^{o}$ (cone)
Peak Output* (short circuit current)	0.5 ma	1/5 ma	1.5 ma
Response Time	-----	20μ sec	-----
Operating Temp Range	-----	$-20^{o}C$ to $85^{o}C$	-----
Spectral Response (determined by filter)	-----	0.6 to 1.1μ	-----
Scale Factor* (at null)	NA	$35\mu a$/arc min	NA
Linear Range	NA	$\pm 1^{o}$ (pair)	NA
Accuracy (at null)	NA	±1 arc min (pair)	NA.
Size	0.6"D x 0.7"L	0.6"D x 1.1"L	0.6"D x 0.9"L
Weight	6.5 grams	6.0 grams	6.0 grams
Power	-----	None Required	-----

*Into a $\leq$ 50-ohm load

2.1.2.3 Digital Solar Aspect Sensor - Type 1818775

Manufacturer: Bendix Corporation
Eclipse-Pioneer Division
Teterboro, New Jersey

Functional Description:

This sensor is intended for use on rotating space vehicles, providing positive, digital indication of the sun angle about a single axis. Eight separate channels are used on a semicircular mask to encode the sun angle in a Gray-code format. The mask provides alternate obstruction or admission of sunlight to silicon solar cell detectors located behind a slit at the center of curvature of each light window. The outputs of the photocells comprise a digital word representative of the solar angle about one axis. An additional channel located at 90° to the sensitive axis, may be used as a "pipper" for determining vehicle rotation rate and/or for triggering electronic circuits to convert from parallel to serial readout.

Physical and Performance Data:

Field-of-View	$\pm 64^{\circ}$ (one axis)
Resolution	1°
Accuracy	$\pm 0.5^{\circ}$
Operating Temperature Range	$-55^{\circ}C$ to $+85^{\circ}C$
Weight	2 oz
Size	1-5/8" x 1-3/8"
Power	None required

2.1.2.4 Fine Angle Sun Sensor - Type 1818823

Manufacturer: Bendix Corporation
Eclipse-Pioneer Division
Teterboro, New Jersey

Functional Description:

This null-seeker type sun sensor provides two analog error signals locating the line to the sun about two axes over a narrow field-of-view. It consists of an objective lens, a coarse silicon solar cell quadrant array, a magnifier lens, a fine silicon solar cell quadrant array, and a housing. When the line to the sun and the optical axis of the sensor are nearly aligned, the objective lens projects the sunlight through a hole in the center of the coarse quadrant array and brings it to a focus behind the coarse array. The image is then magnified by the magnifier lens and projected onto the fine quadrant array which produces a stable null and linear output signals about the null in two axes. Only the fine quadrant contributes to the output near null, while the coarse quadrant cell array gradually intercepts the focused rays as the angular deviation increases. The coarse and fine quadrant cell arrays are electrically interconnected and physically located to produce continuous output signals.

Physical and Performance Data:

Total Field-of-View	±10° (cone)
Fine Quadrant Field-of-View	±20 arc min (cone)
Null Stability	5 arc sec
Output (maximum)	4.0 ma (per axis)
Impedance	100 ohms (per axis)
Scale Factor (at null)	10μa/arc sec
Linear Range	±5 arc min
Monotonic Increasing Range*	1.5°
Operating Temperature Range	-55°C to +50°C
Size	2-11/16"D x 8-15/16"L
Weight	30 oz
Power	None required

*After which the outputs saturate at 4.0 ma out to ±10°.

2.1.2.5 Wide Angle Sun Sensor - Type 1771858 or 1818787

Manufacturer: Bendix Corporation
Eclipse-Pioneer Division
Teterboro, New Jersey

Functional Description:

This null seeker type sun sensor provides two analog error signals locating the line to the sun about two axes over a wide field-of-view. It consists of a fine sensor and a series of eight coarse detectors. The fine sensor consists of a quadrant array of silicon solar cells located behind a square aperture. The silicon solar cell coarse detectors (four per axis) are located around the periphery of the sun sensor structure to provide the wide angle coverage. The appropriate fine and coarse detector outputs are summed electrically and the transition from coarse to fine is achieved geometrically. The coarse detectors and fine sensor are mounted on a common structure and are hermetically sealed in a glass dome. Signals are available through an eight-pin connector.

Physical and Performance Data:

Total Field-of-View	2π steradians
Fine Sensor Field-of-View	20° cone
Scale Factor (each axis at null)	0.2 ma/deg
Output Characteristic (each axis)	
Fine Sensor	Linear
Coarse Detectors	Sine function
Output	0-5 ma
Load Resistance (per axis)	100 ohms
Operating Temperature Range	$-70^{\circ}C$ to $+50^{\circ}C$
Size	1-7/8"D x 2"L
Weight	2.5 oz
Power	None required

2.1.2.6 Refractosyn

Manufacturer: H H Controls Co., Inc.
Cambridge, Massachusetts

Functional Description:

The Refractosyn is a single axis null seeker type sun sensor consisting of an isosceles prism, two silicon solar cell detectors, and a housing. The detectors are mounted on the sides of equal length of the prism which is out at the critical angle (41.5°). At the null position, the sun's rays are essentially totally reflected at the critical angle of incidence. Angular movement of the sun about the null causes abrupt refraction of light onto one or the other of the detectors. The detector outputs are differenced electrically to produce a plus or minus output about the null.

Physical and Performance Data:

Total Field-of-View	$\pm 100^{\circ}$ (one axis)
Null Sensitivity	$15\mu a$/arc min
Peak Output	4.4 ma at 25° from null
Response Time	20μ sec
Accuracy	Not Available

Operating Temperature Range	$-20^{\circ}C$ to $+85^{\circ}C$
Spectral Response	0.4 to 1.1 microns
Weight	~1 gram
Size	11/32" x 9/32" x 3/8"
Power	None required

2.1.2.7 AOSO Fine Sun Sensor

Manufacturer: Honeywell-Aeronautical Division
Boston, Massachusetts

Functional Description:

This sensor is being designed and developed for application in the attitude control system of the Advanced Orbiting Solar Observatory (AOSO). This sensor provides two analog error signals locating the line to the sun about two axes relative to a reference axis in the sensor which can be deviated by a series of counter-rotating optical wedges. The sensor consists of a pair of coarse counter-rotating wedges, two pairs of fine counter-rotating wedges, two critical angle prisms, two silicon solar cell detectors, two reed modulators, electronics, and structure. The wedges are driven by bidirectional stepper motors and their positions are measured by digital optical encoders which use the sun as a source of light. Solar radiation, after passing through the wedges, passes through two critical angle prisms and is modulated and directed onto two silicon solar cell detectors. The outputs of the detectors are demodulated to produce two analog error signals. The sensor also contains an occult sensor and a medium null-seeker type sensor ($\pm 6^{\circ}$ FOV) which use the same optical path as the fine sensor.

Physical and Performance Data:

Fine Point Accuracy (within a $\pm$20 arc min FOV)	$\pm$1.3 arc sec
Scan Accuracy (within a $\pm$20 arc min FOV)	$\pm$1.3 arc sec
Coarse Point (within $\pm 5^{\circ}$ FOV)	$\pm$45 arc sec
Coarse Point ($\pm 5^{\circ}$ FOV) and Scan ($\pm$20 arc min FOV)	$\pm$45 arc sec (resolution $\pm$1.3 arc sec)

Size*	315 cu in.
Weight*	12 lb
Power*	1.5 to 13 w depending on stepper motor activity

*Includes two complete redundant sensors in a single hermetically sealed package

2.1.2.8 Nimbus "D" Sun Sensors

Manufacturer: ITT Federal Laboratories
San Fernando, California

Functional Description:

This sensor is being developed for application in the attitude control system of the Nimbus "D" spacecraft. Each sensor will provide a single axis analog error signal. Six sensors will be connected in two arrays, a solar paddle array and a yaw sensor array. A sensor with a stable null will be mounted on the front side of each of two solar paddles. A sensor with an unstable null will be mounted on the back side of each array. The control system acting on the combined sensor outputs will orient the solar paddle in the pitch axis to face the sun. An array of two sensors mounted on the spacecraft will be used to give yaw position information relative to the sun. The sensor will require no power and will operate in the 2000 to 3500 Å spectral region in order to passively minimize the null position errors due to sunlight reflected from the earth.

Physical and Performance Data:

Field-of-View		
Primary Axis	$\pm 90^{o}$ min	$\pm 95^{o}$ max
Orthogonal Axis	$\pm 60^{o}$ min	
Null Position Accuracy	$\pm 1^{o}$ for orthogonal axis position = 0^{o} $\pm 5^{o}$ for orthogonal axis position = $\pm 45^{o}$	
Scale Factor	$\geq 10\ \mu a/deg$ into a short circuit over $\pm 30^{o}$ in the primary axis for the orthogonal axis position = 0^{o}	
Linearity	$\pm 20\%$	

Cross Coupling	Primary axis output when orthogonal axis position = $\pm 45^{\circ}$	$\geq 50\%$ primary axis output for orthogonal axis position $= 0^{\circ}$
Spectral Band	2000 to 3500 Å	
Time Constant	<10 sec	
Size	Not yet established	
Weight	Not yet established	
Power	0 w	

2.1.2.9 Mariner and Ranger Sun Sensors

Manufacturer: NASA Jet Propulsion Laboratory
Pasadena, California and

Northrop-Nortronics
Palos Verdes Peninsula, California

Functional Description:

The Ranger and Mariner sun sensor assemblies consist of body-mounted sensors that provide a 4π steradian field-of-view and have two output control axes, nominally pitch and yaw. Two secondary and four primary sensors constitute a set. The secondary sensors provide a coarse indication of the location of the sun. These sensors provide a saturated signal that determines the direction of the angle through which the spacecraft would be rotated to bring the sun within the field-of-view of the primary sensors. The sun sensor subsystem has an unstable null located 180 degrees from the true system sensor null. If the sun is on either side of this null, the spacecraft is commanded to rotate away from this point. The time for spacecraft erection is minimized, since the spacecraft is rotated through the smallest possible angle to acquire the sun. The unstable null represents a very small region, and any spacecraft movement will cause the sun direction to shift from this anomalous point.

Two primary sensors are used on each axis to provide fine control about the sun sensor null. The null accuracy is ±0.1 degree, one-sigma. The field-of-view of each primary sensor covers a quadrant approximately 160 degrees in azimuth and 45 degrees in elevation. The secondary sensors also have a 160-degree field in azimuth. The coverage in elevation overlaps that of the primary sensors and provides complete coverage for acquisition.

Each primary sensor utilizes one cadmium-sulfide detector and each secondary sensor contains four cadmium-sulfide detectors. Thus, the complete sun sensor subsystem has a total of 12 detectors.

Physical and Performance Data:

Null Accuracy	$\pm 0.1^{\circ}$, (1σ)
Field-of-View	4π steradians
Detector	Cadmium sulfide photocells
Output (analog)	16 volts per deg for 0.5°, Saturates at ± 2.0 deg, saturation voltage 16 v
Weight	11 oz*
Size	21.5 cu in.*
Power	800 mw (primary power)
MTBF	100,000 hr
Reliability	0.92 for 8000-hr continuous operation

*One complete set of sensors without power supply

2.1.2.10 OGO Sun Sensor

Manufacturer: TRW Systems
Redondo Beach, California

Functional Description:

This null-seeker type sun sensor was designed and developed expressly for application in the attitude control system of the Orbiting Geophysical Observatory (OGO). The sensor provides two analog error signals locating the sun about two axes of rotation over a field-of-view of 4π steradians. The sensor consists of two packages which are located at the extremities of two solar array panels in order to obtain an unobstructed large field-of-view. One package contains a fine sensor and three silicon solar cell coarse detectors; the other, three coarse detectors.

The fine sensor consists of a silicon Radiation Tracking Transducer (RTT),* pinhole optic, and sun shade. The pinhold optic images the sun onto the surface of the RTT which generates two analog error signals and a "sun-present" signal when the sun is in the field-of-view determined by the sun shade. The "sun-present" signal is used to switch from the coarse detectors to the fine sensor. The sun sensor is passively temperature controlled.

Physical and Performance Data:

Total Field-of-View	4π steradian
Fine Sensor Field-of-View	$\pm 17^{\circ}$ (cone)
Scale Factor (at null)	0.75 /deg into 922-ohm load
Null Stability	$\pm 0.2^{\circ}$
Output Characteristic	Sine function
Output (maximum)	
Sun position signal	104 μa into 518-ohm load
Sun present signal	129 μa into 387-ohm load
Size	3" x 4" x 4-1/2" and 6" x 4-1/2"
Weight	21 lb (total)
Power	None required

2.1.2.11 Pioneer Sun Sensors

Manufacturer: TRW Systems
Redondo Beach, California

Functional Description:

These sensors were designed and developed for application in the attitude control system of the Pioneer spacecraft which is spin stabilized. These sun-present type sensors each consist of four photosensitive silicon controlled rectifiers (PSCR) mounted in a line and electrically connected in a redundant quad, and a sun shade which defines the field-of-view. When the sun is in the field-of-view, the PSCR's conduct as diodes.

†Registered trademark by Electro-Optical Systems, Inc., Pasadena, Calif.

When the sun is not in the field-of-view, the PSCR's essentially act as open circuits. The light threshold is set by selecting the value of a resistor between the PSCR gate and cathode; temperature compensation is accomplished by connecting a thermistor in parallel with a resistor. Temperature control of the sensors is accomplished passively.

Physical and Performance Data:

Sensor Type	A or C (Wide Field)	B or D (Wide Field)	E ("PIPPER")
Field-of-View			
Right Azimuth	$+22.5^{\circ} \pm 2.5^{\circ}$	$+45^{\circ} \pm 2.5^{\circ}$	$+1^{\circ} \pm 0.3^{\circ}$
Left Azimuth	$-22.5^{\circ} \pm 2.5^{\circ}$	$-45^{\circ} \pm 2.5^{\circ}$	$-1^{\circ} \pm 0.3^{\circ}$
Upper Elevation	$+80^{\circ}$ to $+85^{\circ}$	$+10^{\circ} \pm 4^{\circ}$	$+20^{\circ} \pm 4^{\circ}$
Lower Elevation	$0^{\circ} \pm 0.2^{\circ}$	$-10^{\circ} \pm 4^{\circ}$	$-20^{\circ} \pm 4^{\circ}$
Size*	4" x 2-3/4" x 3-1/2"	2-3/4" x 3" x 2-1/2"	4" x 2-1/4" x 2-1/2"
Weight	0.21 lb	0.16 lb	0.17 lb
Power	In the "ON" condition the voltage drop across a PSCR is ~1.7 v and the current should be limited to <200 ma. In the "OFF" condition a PSCR draws <10μa at 15 v.		
Reliability	-----	0.997 for 6 months in space	-----
Accuracy	±2.50	±2.50	±0.30

*approximate

2.1.2.12 Intelsat III Sun Sensor

Manufacturer: TRW Systems
Redondo Beach, California

Functional Description:

This sensor is being designed and developed for the Intelsat III satellite, a spin-stabilized vehicle. The sensor measures the angle between the spin axis of the satellite and the sun vector (aspect angle). It consists of a single silicon solar cell optically cemented to a quartz substrate. The side of the substrate opposite the solar cell contains a vaccum deposited nickel coating that is photo etched to produce a slit aperture array. The aperture array consists of three slits, i.e., reference slit, polarity slit, and measurement slit. The entrance angle of the sunlight into the slit aperture changes as the spacecraft rotates causing the refracted illumination to sweep across the active surface of the solar cell. The resulting output of the solar cell is a series of pulses. The aspect angle is determined by measuring the time, or rotational angle, between the reference pulse and measurement pulse. The polarity pulse is present for negative angles only. The sensor is capable of operation between solar aspect angles of ±80 degrees and at spin rates of over 200 rpm.

Physical and Performance Data:

Field-of-View (Aspect Angle)	$\pm 80^{o}$
Accuracy	$\pm 0.2^{o}$
Scale Factor	
0-10 deg	approx 1.5 rot deg/aspect deg
10-60 deg	approx 1 rot deg/aspect deg
60-80 deg	approx 0.8 rot deg/aspect deg
Pulse Output Voltage	0.250 to 0.450 v peak
Required Load Impedance	50 K ohms
Operating Temperature	0^{o}F to 150^{o}F
Size	2" x 2" by 1-1/2"
Weight	0.4 lb
Power	None required
Reliability (5 year flight)	0.997

2.1.2.13 VASP Sun Sensor Assembly

Manufacturer: TRW Systems
Redondo Beach, California

Functional Description:

The VASP sun sensor assembly was designed and developed for use in the attitude control system of the VASP which is a spinning spacecraft. Each sun sensor assembly consists of three functionally separate sensor units, designated the A sensor, B sensor, and C sensor. Each of these sensor units are null-seekers consisting of a pair of physically back-to-back N on P silicon solar cells connected electrically in parallel-opposition so that the combined output of a pair of cells is zero when the sun lies in a plane (null plane) parallel to the photovoltaic surfaces of the cells. The A and C sensors employ immersion lenses bonded to the cells to enhance the scale factor at null. The A and C sensor units each have a null plane useful over a wide field-of-view in altitude and their null planes are separated by 90° in azimuth. The B sensor unit has a null plane orthogonal to the null planes of the A and C sensor units and provides a linear error signal for a few degrees either side of null. The B sensor provides altitude information when the sun lies within a few degrees of the C sensor unit null plane.

Physical and Performance Data:

Sensor Unit	A[1]	B[2]	C[1]
Null Plane Orientation	90° (azimuth)	0° (altitude)	0° (azimuth)
Measurement Field-of-View			
Altitude	-70° to +70°	-80° to +80°	-70° to +70°
Azimuth	30° to 150°	-2° to 2°	-35° to +35°
Scale Factor at Null greater than	240 mv/deg of azimuth (Altitude = 0°)	48 mv/deg of altitude (Azimuth= 0°)	270 mv/deg of azimuth (Altitude = 0°)
Linearity (near null)	±20%	±20%	±20%
Maximum Output	±360 mv	±500 mv	±360 mv
Null Accuracy	±0.8°	±0.5°	±0.8°
Temperature Control	-----	Passive	-----
Operating Temperature Range	-----	+20°F to +75°F	-----
Nonoperating Temperature Range	-----	-135°F to +150°F	

1) All values given are for a 250-ohm external load

2) All values given are for a 100-ohm external load

Power	----- None Required -----
Size	----- 2.5 x 2.0 x 3.6 in -----
Weight	--0.125 lb not including external-- lead wires
Reliability	--0.998 for launch and 18 months-- in space

2.1.2.14 Vela III Sun Sensor

Manufacturer: TRW Systems
Redondo Beach, California

Functional Description:

This "sun present" type sensor was designed specifically for the Vela III spacecraft. Its purpose is to provide signals to the attitude control system of the spin-stabilized satellite during an attitude re-orientation maneuver which aligns the solar array panels relative to the sun. The sensor consists of a detector, a mask, two resistors, one thermistor, and a housing assembly. The detector is a photosensitive silicon controlled rectifier which acts as a switch, changing abruptly from open to short when the illumination on the detector exceeds a threshold value. The mask is a trapezoidal pattern of deposited aluminum on a glass substrate; the mask determines the sensor field-of-view. The two resistors and the thermistor set the light threshold of the sensor and provide thermal compensation for the temperature dependence of the detector. The housing assembly is machined from epoxy-glass laminate stock, providing a light weight structure which minimizes electrical interference with the nearby communications antenna. The sun sensor temperature control is accomplished passively by means of a white epoxy-base paint which is applied to the exterior surfaces.

Physical and Performance Data:

Field-of-View	
Azimuth	$-20^{\circ} \pm 2^{\circ}$ to $+20^{\circ} \pm 2^{\circ}$
Elevation	$+15^{\circ} \pm 3^{\circ}$ to $+50^{\circ} \pm 3^{\circ}$
Size:	2" x 2.5" x 0.75 (approximate)

Weight	0.10 lb
Power	In the "ON" condition the detector acts as a diode with 1.7 volts maximum voltage drop in the forward direction. In the "OFF" condition, leakage current is less than 10μa for the nominal bias voltage
Reliability (assessed)	0.997 for 100 hr operation after 6 months in orbit

2.2 EARTH HORIZON SENSORS

During the earth sensor survey, data was obtained on approximately 50 different devices. In this section seventeen of these devices are described in detail. The sample was reduced by including only sensor systems which are either flight qualified or in active development, and then by including only one representative sensor from each group.

2.2.1 General Discussion

The earth sensing devices surveyed fall into three main classes.

2.2.1.1 Mechanical Horizon Scanners

Each member of this class incorporates a deflectable optical component which provides spatial modulation by scanning across the earth-space radiance discontinuity. In most cases, the center point of the scanning pattern is brought into angular coincidence with this discontinuity and the angular offset from a fixed reference is read out from the deflection device. A combination of three or more such devices is required to sense the offset from local vertical about two axes. In some devices this is done by incorporating a peripheral or azimuth edge scan in a single head assembly.

2.2.1.2 Disc Scanning Earth Sensors

This type of sensor mechanically scans the optical line of sight through the radiant disc of the earth. The three most common scanning methods are:

1) Single axis deflection of a mirror

2) Conical scan, where the mirror is offset at a fixed angle from the axis of a motor

3) Passive scan, where the angular motion of a spin-stabilized spacecraft provides the scanning motion

The offset from the local vertical is computed by comparing the length of scans from two or more different sensors and measuring their time phasing.

2.2.1.3 Nonscanning Earth Sensors

The third class of sensor does not employ mechanical scanning. Two basic approaches have been utilized, electronically scanned detector arrays and radiation balancing sensors. In the former case the earth horizon or disc image is focused on an array of detectors sequentially interrogated electronically to locate the discontinuity or measure the disc width. The radiation balance sensor also incorporates multiple detectors. In this case, however, the amplitudes of the signals on the several detectors dissecting the image are continuously compared. The balancing array is generally configured such that the net signal is zero when there is no attitude offset.

2.2.2 Earth Sensor Summary

This subsection briefly describes the unclassified devices considered, and Table 2-II summarizes the physical and performance data on each sensor. (Volume IV presents data on three classified earth sensors.)

2.2.2.1 OGO Horizon Sensor

Manufacturer: Advanced Technology Division, American Standard

Contracting Agency: TRW Systems Group

Sensor Function:

The OGO horizon sensor system provides angular offset data for each of its four horizon track points. This data is combined to compute the roll and pitch offsets of the OGO spacecraft.

Manufacturer And Model	Development Status	Sensor Type	System Configuration	Output Characteristics	Detector	Spectral Range	Optical System
Advanced Technology Division - American Standard - OGO	Operational	Edge Tracking, Three Points	Two Dual Tracker Heads; One Electronics Assembly	Analog Pitch and Roll	GE-Immersed Bolometer	8 -22 Microns	Plane Scanning Mirror and Refractive Telescope
Advanced Technology Division - American Standard Advanced OGO	Being Space Qualified	Edge Tracking Three Points	Two Dual Tracker Heads; One Electronics Assembly	Analog Pitch and Roll	GE-Immersed Bolometer	14 - 16 Microns	Plane Scanning Mirror and Refractive Telescope
Advanced Technology Division - American Standard Long Life Earth Sensor	Acceptance Tested	Chord Scan; Two Orthogonal Planes	One Dual Scanner Assembly	Analog Pitch and Roll	GE-Immersed Bolometer	13.5 - 21 Microns	Plane Scanning Mirror and Refractive Telescope
Advanced Technology Division - American Standard GEMINI	Operational	Peripheral Edge Tracker	One Scanner Head and One Electronic Assembly	Analog Pitch and Roll	GE-Immersed Bolometer	13.5 - 22 Microns	Two Axis Plane Scanning Mirror, Refractive Telescope
Barnes Engineering Co. Mercury - 13-130A	Operational	Conical Scan, Two Scan Cones	Two Scanner Assemblies	Analog Pitch and Roll	GE-Immersed Bolometer	1.8 - 18 Microns	Prism Scan, Refractive Telescope
Barnes Engineering Co. Tiros - 13-200	Operational	Circular Scan Generated by Vehicle Spin	Two Fixed Sensors in "V" Configuration	Telemetered Horizon Pulses And Camera Trigger	GE-Immersed Bolometer	1.8 - 20 or 7.5 - 20 Microns	Refractive Telescope
Barnes Engineering Co. Apollo Antenna Pointing	Developmental	Radiation Balance, Fixed Field	One Dual Field Sensor	Analog Pitch and Roll	Four Element Thermopile	14 - 24 Microns	Refractive Telescope and Cone Condensers
Barnes Engineering Co. "FIRM"	Experimental	Edge Tracking, Four Points	Four Sensor Heads and Power Supply	Analog or Digital Pitch and Roll	GE-Immersed Thermistor Bolometer	14 - 16 Microns	Plane Scan Mirror, Piezoelectric Driven "FIRM" Prism and Refractive Telescope
Barnes Engineering Co. Thermopile Edge Tracker	Experimental	Edge Tracker, Three Points	Three Sensor Heads	Analog Pitch and Roll	Thermopile	14 - 20 Microns	Refractive Telescope
General Electric Co. NIMBUS	Operational	Conical Scan, Two Scan Cones	Two Sensor Heads	Analog Pitch and Roll	GE-Immersed Bolometer	12 - 18 Microns	Refractive Telescope with Prism Scan
Honeywell - APL Sensor	Space Qualified	Radiation Balance, Static	Single Sensor Unit	Analog Pitch and Roll	Thermistor Bolometer	7 Microns to Germanium Roll Off	Refractive Lens and Reflecting Cone
Lockheed Missiles and Space Co. - P-11	Operational	Circular Scan, Generated by Spacecraft Spin	Two Sensors In "V" Configuration	Earth Horizon Crossing Indication	GE-Immersed Bolometer	14 - 16 Microns	Refractive
Martin Co. Saturn V	Space Qualified	Edge Tracking, Four Points	Four Heads In One Canister	Digital Pitch and Roll	GE-Immersed Bolometer	13 - 17 Microns	Plane Tracking Mirrors and Refractive Telescopes
Northrop-Nortronics Short Range Earth Sensor	Operational (Ranger)	Radiation Balance	Single Sensor Package	Analog Pitch and Roll	Three Dumont K2103 Photo-multipliers	S-11	Shadow Mask
Northrop-Nortronics Long Range Earth Sensor	Operational (Mariner)	Radiation Centroid Tracker	Single Package	Analog Pitch and Roll	One Dumont K2103 Photo-multiplier	S-11	50 MM. f/1.2 Refractor, Vibrating Reed Scanner
TRW Systems Reliable Earth Sensor	Space Qualified	Radiation Balance	Single Unit	Analog Pitch and Roll	Metal Bolometers, Eight Element Array	12 - 18 Microns	Refractive

(1) Published by the Institute of Science and Technology, University of Michigan, under Navy Contract NOnr 1224(12)
Authors: John Duncan, William Wolfe, George Oppel, James Burn

NOTE: NS = NOT SPECIFIED
NA = NOT APPLICABLE

Table 2-II. Summary of Earth Sensors

Field of View ACQ.	Field of View TKG.	Field of View INST.	Quoted Instrument Accuracy	Tracking Time Constant or Bandwidth	Altitude Range	Quoted Reliability For One Year Of Operation	Volume IN^3	Weight Lbs	Power Watts	Data Source
±45°	±2°	1° x 1°	±0.10°, 3σ At Null	50 msec	100-70,000 n.mi.	0.86	270 (Total)	13.2 (Total)	11, Max	IRIA[1] "State of the Art Report on Infrared Horizon Sensors" No. 2389-80-7 April, 1965
±45°	±2°	1° x 1°	±0.05°, 3σ At Null	50 msec	120-80,000 n.mi. or 50-60,000 n.mi.	0.95	300 (Total)	16.8 (Total)	12, Max	Report ATD-R-1393 "Description of Advanced OGO Horizon Sensor System" 11 July, 1966
67°	NA	1° x 1°	±0.21° At Null	3 Sec.	6000-19,270 n.mi.	0.92	200	7.7	7	Report ATD-R-1387 "Description of Long-Life Earth Sensor System" 1 May, 1966
70° x 160°	± 2° x ±80°	1.4° x 1.4°	±0.10° At Null	3 Sec	50-220,000 n.mi.	0.78	209	11	11 Max	IRIA S.O.A. Report
110° Cone		2° x 8°	±1.0°	NS	50-300 Miles	0.50	154	6	7	IRIA S.O.A. Report
NA	NA	1.3° x 1.3°	±0.5°	NA	ANY	0.94	10	0.75	0.26	IRIA S.O.A. Report
NS	10°	10°	±1.5° at 8000 mi. ±0.28° at Lunar Range	NS	8000 mi. to Lunar Range	0.999	120	3.5	1.5	"Proc. of First Symposium on IR Sensors for Spacecraft Guidance and Control", Barnes - May 1965
NS	90°	1° x 2°	±.01°	10 cps	NS	NS	500	12	20	Barnes - Symposium Proceedings
NS	10°	10° Cone	±0.115°	0.5 Sec	6000-100,000 mi.	0.97	96 Per Head	3.5 Per Head	0.10	Barnes - Report No. P1016, April 1, 1966
90° Cone	NS	4° x 8°	±0.2°	60 msec	500 mi.	NS	26 Per Head	3.5 Per Head	4.5	IRIA S.O.A. Report
40° Cone	20° Cone	2.5° x 19°	±0.65°	0.015 cps	300-600 n.mi.	0.40	9.2 Head Only	2.8 lb Head Only	1.5	IRIA S.O.A. Report
NA	NA	1° x 1°	NS	NA	250-500 n.mi.	NS	18	0.75	0.5	LMSC Data Submitted to TRW Systems
90°	90°	.5° x 3°	±0.05, 3σ	10°/sec Track Rate	80-22,000 mi.	0.65	1360	35.0	20	IRIA S.O.A. Report
20° x 40°	20° x 40°	20° x 40°	±0.2° Pitch ±0.3° Roll (At Null)	14 msec	20,000 to 1,000,000 mi.	NS	72	2.5	3.5	Northrop Report No. 63-273 June, 1963
4° x 10°	4° x 10°	4° x 10°	±0.2° (At Null)	NS	1,000,000 to 50,000,000 mi.	NS	160	6.5	6.5	Northrop Report No. 63-273 June, 1963
12° cone	±12° cone	±12° cone	±1.0°, 1σ 3 Yr. Period	60 sec	6000-19,200 n.mi.	0.97	216	6.2	5	Barnes Symposium Proceedings

Sensor System Description:

The system consists of four tracking infrared telescopes, each utilizing an electromagnetically driven scan mirror, refractive lens, bolometer, and tracking electronics. The deflection device (Positor) has a total angular range of $\pm 45^{\circ}$ about its reference position and dithers $\pm 2^{\circ}$ about the track point. The angular offset of the track point from the Positor reference position is sensed by the modulation induced upon a reference coil in the magnetic field of the device. The attitude computer, a simple switching matrix, generates pitch and roll error signals from three of the tracking heads. Four independent attitude computations can therefore be made from the tracking data.

Limitations:

The major limitation of the OGO horizon sensor system is its inability to discriminate against radiance gradients of the atmosphere in the 8 to 22-micron band. This has been corrected in later models of this sensor.

2.2.2.2 Advanced OGO Horizon Sensor

Manufacturer: Advance Technology Division, American Standard

Contracting Agency: NASA Goddard

Sensor Function: Same as Paragraph 2.2.2.1, OGO Horizon Sensor

Generally identical with description given in Paragraph 2.2.2.1, except for the following:

1) The spectral acceptance band has been changed to 14-16 microns

2) A coated optical window is provided to seal the unit

3) Improved acquisition logic

4) Improved Positor

5) Improved system performance with regard to accuracy, reliability and RFI susceptibility

2.2.2.3 Long Life Earth Sensor System

Manufacturer: Advanced Technology Division, American Standard

Contracting Agency: NASA Goddard

Sensor Function:

Sense vehicle attitude error on board the Advanced Technology Satellite to be launched during 1967.

Sensor System Description:

The sensor system consists of two scanners, each containing an infrared telescope and bolometer. The system is aligned with respect to the vehicle so that its null axis and the vehicle yaw axis are collinear. The two scanners independently scan along orthogonal paths across the earth disc detecting displacements along the two orthogonal axes (pitch and roll). The Positor deflection system, similar to that of the OGO sensors, is used to generate the chord scan.

Limitations:

Long time constant; limited to altitude of 6000 nmi or higher.

2.2.2.4 Gemini Earth Sensor

Manufacturer: Advanced Technology Division, American Standard

Contracting Agency: McDonnell Aircraft Corporation

Sensor Function:

The Gemini horizon sensor measures the attitude of the Gemini Spacecraft relative to the local vertical and provides analog pitch and roll error signals to the attitude control system.

Sensor Description:

The Gemini horizon sensor tracks the earth's edge by scanning across the horizon in elevation and along the horizon in azimuth. The error signal is generated by sampling the azimuth and average elevation of the scanning Positor. The Positor deflection unit is similar to that of OGO but its base is suspended by a yoke which is oscillated in azimuth.

Limitations:

Like the OGO sensor, the Gemini scanner views the earth in the 8- to 22-micron band wherein it is difficult to discriminate against atmospheric earth radiance gradients.

2.2.2.5 Mercury Earth Sensor

Manufacturer: Barnes Engineering Co. (Model 13-130A)

Contracting Agency: McDonnell Aircraft Corporation

Sensor Function:

The Model 13-130A-1 was used in the automatic attitude stabilization system of the Mercury capsule. It is representative of a large group of conical scan earth sensors produced by Barnes.

Sensor Description:

The sensor system consists of two separate scanning heads and an electronic assembly. The scan is of the conical type, generated by rotating a prism about the detector axis. The prism refraction angle determines the apex angle of the scan cone. The phase and duration of the separate earth-crossing pulses may be compared to yield the relative azimuth and elevation angles of the horizon. These angles are proportional to spacecraft pitch and roll angles.

Limitations:

The extremely broad spectral bandpass of this sensor (1.8 to 18 microns) makes it subject to atmospheric gradient and fluctuations. The lifetime of this sensor system (~1000 hr MTBF/head) limits its application to short term missions. Later model conical scanners are improved in this respect.

2.2.2.6 TIROS Radially Oriented Horizon Sensor

Manufacturer: Barnes Engineering Co. (Model 13-200)

Contracting Agency: Radio Corporation of America

Sensor Function:

Attitude sensing and timing signal generator for spin stabilized vehicles. Initial production units used on TIROS weather satellite.

Sensor Description:

This sensor consists simply of a lens, bolometer detector and transistorized amplifier. It generates a pair of horizon pulses as the sensor axis crosses the earth's disc. The sensor may be used to measure spacecraft spin rate or to measure the magnitude of spacecraft attitude error. The sensors have been built in dual configurations to make possible resolution of the error signals on both the pitch and roll axes. The raw sensor data is telemetered to the ground for processing.

Limitations:

The sensor is usable only on a spin-stabilized vehicle.

2.2.2.7 Apollo Antenna Pointing Sensor

Manufacturer: Barnes Engineering Co.

Contracting Agency: NASA

Sensor Function:

This two-axis tracking sensor serves to point the Apollo communications antenna toward the earth. It is designed to track the earth from distances between 8000 miles and lunar distance.

Sensor Description:

The Apollo sensor works on the radiation balance principle, utilizing a quadrant array of thermopile detectors for a narrow field channel and a similar array for the wide field channel. The optical system for each of the channels consists of a refractive objective lens, interference filter and a set of four adjacent reflective cone condensers, used to separate the earth image into quadrants and to funnel the energy to their respective thermopiles. The weight, power consumption and reliability characteristics of this sensor are very favorable.

Limitations:

Moderate accuracy.

2.2.2.8 "FIRM" Horizon Sensor

Manufacturer: Barnes Engineering Company

Contracting Agency: In-house development

Sensor Function:

This sensor is a feasibility model designed and fabricated by Barnes to demonstrate a unique modulation scheme.

Sensor Description:

The term FIRM is an acronym for Frustrated Internal Reflection Modulator. Applied to the horizon sensor task, the FIRM cell consists of an optical wedge attached to a prism in the sensor optical path. By varying the spacing between the wedge and the prism with a piezoelectric drive, the optical path alternately passes through the wedge. As a result, the sensor optical axis is switched back and forth over a small angle ($\sim 10^{\circ}$), sufficient to modulate the earth horizon. Barnes has configured a high accuracy feasibility model based upon this principle, using the same basic scheme as in the thermopile edge tracker described in Paragraph 2.1.2.9.

2.2.2.9 Barnes Thermopile Edge Tracker

Manufacturer: Barnes Engineering Company

Contracting Agency: NASA, Manned Spacecraft Center

Sensor Function:

Horizon edge tracking for attitude measurement, unspecified mission. This sensor has been developed to the engineering model stage.

Sensor Description:

The edge tracking system consists of three optical heads separated by 90° in azimuth on board the vehicle. Each tracking head has two fields-of-view separated by a small fixed angle, each using a thermopile detector. Each sensor head is capable of pivoting in elevation to track the horizon. Tracking is accomplished by applying a downward dc signal to the head elevation servo. Tracker equilibrium occurs when the lower field-of-view senses enough of the earth's radiance to counteract the dc drive. Reliability is claimed to be extremely high.

2.2.2.10 Nimbus Earth Sensor

Manufacturer: General Electric Co.

Contracting Agency: NASA

Sensor Function:

Provide pitch and roll attitude error signals to the attitude control system of the Nimbus meterological satellite.

Sensor Description:

The Nimbus earth sensor system consists of two scanning heads. The scan pattern is conical with a 90^{o} cone angle generated by a rotating prism at a scan rate of 16.2 rps. The detector element in each sensing head is a germanium-immersed bolometer.

Limitations:

The sensor is designed to meet the Nimbus specification of a nominal 500 nmi altitude and a lifetime of 90 days.

2.2.2.11 Applied Physics Laboratory Horizon Sensor System

Manufacturer: Minneapolis-Honeywell

Contracting Agency: Applied Physics Laboratory
Johns Hopkins University

Sensor Function:

Attitude sensing instrument for the Transit space package.

Sensor Description:

The Honeywell - APL horizon sensor is a static device having no moving parts with the exception of a chopper. The imaging system consists of a unique stainless steel cone whose axis is nominally aligned with the local vertical and a germanium lens. The earth image, radially inverted by the cone optic, is sensed by a thermistor bolometer array. An updated all solid-state horizon sensor of similar specifications has been developed by Honeywell.

Limitations:

The APL sensor is limited to low altitude application and has a relatively long-time constant.

2.2.2.12 P-11 Earth Sensor

Manufacturer: Lockheed Missiles and Space Company

Contracting Agency: USAF

Sensor Function:

Provide horizon crossing pulses for telemetry to ground from the P-11 spacecraft.

Sensor Description:

The P-11 earth sensor consists of a germanium refractive optic, a three-element spectral filter, immersed thermistor bolometer detector and processing electronics. The sensor electronics are designed to provide a positive logic voltage at each horizon crossing of the optical axis. The sensor field-of-view is fixed relative to the spacecraft and the circular scan generated by the spin of the spacecraft.

2.2.2.13 Saturn V Horizon Sensor

Manufacturer: Martin Company

Contracting Agency: NASA, MSFC

Sensor Function:

To provide earth-oriented vehicle attitude to the Saturn boost guidance system during second-stage firing.

Sensor Description:

The Saturn V horizon sensor system consists of four tracking heads mounted in a sealed and pressurized canister. The basic optical elements of each head are a tracking mirror, and oscillating mirror modulator and a fixed infrared telescope. The scanning mirror is mounted on a flexural pivot and electromagnetically driven in a manner similar to that of the ATD Positor. The system tracks the horizon at four different points separated by 90° in azimuth, the tracking drive being supplied by a torquer connected to the first mirror of each head. Error signals are obtained from a resolver mounted on the same shaft.

2.2.2.14 Short-Range Earth Sensor

Manufacturer: Northrop - Nortronics

Contracting Agency: NASA - JPL

Sensor Function:

Provide attitude error signals for pointing the Ranger and Mariner spacecraft antenna.

Sensor Description:

The Ranger short-range earth sensor is a static device consisting of a three-element shadow mask, three end-on photomultipliers, power supply and processing electronics. The mask is configured so that an angular deviation of the earth from the sensor axis generates an unbalance in the photomultiplier signals. The device has a total field-of-view of $20^{\circ} \times 40^{\circ}$ and requires that the earth be illuminated by the sun since the photomultiplier spectral response is in the near-visual range of the spectrum.

2.2.2.15 Long-Range Earth Sensor

Manufacturer: Northrop Nortronics

Contracting Agency: NASA-JPL

Sensor Function:

Provide attitude error signals for pointing of the Mariner spacecraft antenna.

Sensor Description:

The Mariner long-range earth sensor consists of a refractive objective lens, a modulating mask mounted on a vibrating reed, a photomultiplier detector and electronics. The vibrating mask, approximately triangular in configuration, generates error signals which are linearly proportional to the two axis offset of the earth from the optical axis over a $4^{\circ} \times 10^{\circ}$ field. The vibrating reed scanner mechanism has proven sufficiently reliable for a three year lifetime in space. The device functions at ranges from one to fifty million miles, from which the earth is essentially a point source when illuminated by the sun.

2.2.2.16 Reliable Earth Sensor

Manufacturer: TRW Systems Group

Contracting Agency: NASA, Goddard

Sensor Function:

The reliable earth sensor is designed to provide two-axis attitude error signals for spacecraft missions extending over extremely long time periods.

Sensor Description:

The reliable earth sensor consists of an f/1 germanium objective lens, an array of eight platinum bolometers arranged in an error sensing bridge, and amplifying electronics. Since the sensor contains no moving parts, the probability of failure has been reduced to that of the electronics. The sensor has been configured in two forms, one for incorporation into an attitude control system, the other to provide telemetered error signals.

Limitations:

Moderate accuracy and long detector time constant.

2.3 STAR TRACKERS

During the survey of the star tracking subsystems, the characteristics of approximately fifty devices were reviewed. A large number of these trackers have been developed for aircraft navigation, missile guidance, and for the control of high altitude experiments in balloons and research aircraft.

Only those subsystems directly applicable to spacecraft were considered in detail. These have been divided into subsystems using mechanical scanning and those using electronic scanning. Subsystems not applicable for use in space are summarized, with comments difining the characteristics which eliminate them from consideration.

2.3.1 Discussion of Contemporary Star Trackers

The primary element defining the configuration of a star tracker is the type of quantum detector which is used to sense the stellar radiation. Over 99 percent of the navigational stars have effective radiation temperatures between 16,000 K^o and 2900 K^o, approximating black body radiators with peak radiance between 0.18 and 1.0 micron. Consequently, the quantum detectors used in star trackers must be sensitive within this spectral range. The detectors most frequently used are photoelectric photomultipliers and image dissectors, sensitive to radiation between 0.3 and 0.7 micron, and solid-state cadmium-sulfide and silicon devices. Cadmium-sulfide detectors have relatively narrow spectral response, with maximum sensitivity from 0.5 to 0.7 micron, while silicon detectors have a broad spectral response, from 0.4 to 1.1 microns, being maximum at approximately 0.8 micron.

Photomultipliers have the distinct advantage of extremely low internal noise, being limited (in the space application) by the shot noise associated with the thermally-induced dark current (Reference 1). In addition, high responsivity is provided by secondary electron multiplication in the dynode structure, providing essentially noise-free application,

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Reference (1) V.K. Zworykin and E.G. Ramberg, Photoelectricity and its Application, John Wiley and Sons, Inc., New York, N.Y., pp. 148-150, 258, 1949.

normally in the order of 10^6. Inasmuch as photomultipliers are not imaging devices, spatial modulation of the incident stellar radiation must be provided by a mechanical modulating device.

The EMR type 568A and 571A photomultipliers, however, provide coarse spatial modulation by the use of a photocathode which is divided into four quadrants which may be sequentially interrogated by electronic switching. However, since the method of scanning consists of balancing the radiance on opposite quadrants of relatively large area, trackers using this sensor are highly susceptible to unbalance in null due to solar glare, galactic background, and static or dynamic unbalance in the photoemission of the photocathode quadrants. Trackers using this type of photomultiplier are the Honeywell Radiation Center Advanced Star Tracker, Nortronics Star-Sun Tracker, and EMR Star Sensor for the Apollo X-Ray Astronomy Experiment.

Image dissectors offer two advantages over the photomultipliers: electronic scanning can be accomplished by deflection of the electron image, and dark noise is reduced due to the masking of the major portion of the electron image by the dissecting aperture. In addition, electronic gimballing may be employed where accuracy and field of view requirements are compatible, enabling the use of a strapdown tracker configuration. The Barnes Engineering Co. Canopus Tracker, used on the NASA/JPL Ranger vehicle, is representative of several trackers on contemporary spacecraft programs which utilize the image dissector.

Cadmium-sulfide detectors offer the advantage of increased responsivity through quantum gain, which may be as high as 10^4 (Reference 2). However, since they are limited by a relatively long-time constant, modulation may be accomplished only in the low-frequency range of the spectrum where current (1/f) noise predominates, reducing the detectivity. The General Precision Inc., Miniature Stellar Sensor, under development for the Air Force Avionics Laboratory is representative of this type of system. Spatial modulation is accomplished by lateral oscillation of the twin-V detector array.

Reference (2) A. Rose, "Performance of Photoconductors," Proc. of the IRE, December 1955, pp. 1830-1869.

The responsivity of silicon detectors is extremely low, as silicon does not have quantum gain. (An exception is the recently developed silicon-avalanche device, in which increased responsivity is obtained at the expense of an increase in noise level) (Reference 3). Under high background illumination, current noise predominates. In the space application, the noise limit may be either the thermal noise level or shot noise induced by junction current, if bias is used (Reference 4). In either case, both are comparable in magnitude to the thermal noise level of the preamplifier, and narrow-band modulation techniques are required to obtain a usable signal-to-noise ratio. The Kollsman KS-187 subsystem, in development for space applications, utilizes this modulation method with oscillating scanner mechanisms.

Vidicon tracking systems, although more complex than most other types, offer the capability of tracking in the presence of solar-induced glare caused by either scatter within optical elements or scatter from window surfaces contaminated by fuel exhaust products. The characteristic of integration of the radiation-induced electronic charge on the photoconductive target enables relatively rapid acquisition and tracking with a reasonable signal-to-noise ratio (Reference 5). A distinct disadvantage, however, is the line-of-sight stabilization which is required to prevent movement of the optical star image over several lines of the raster during a frame interval, which drastically reduces the signal amplitude. An example of this type of system is the Nortronics NCN-121 equipment which has been developed for the Apollo Range Instrumentation Program.

Reference (3) L. A. D'Asaro and L. K. Anderson, "At the End of the Laser Beam, A More Sensitive Photodiode," Electronics, May 30, 1966, pp. 94-98.

Reference (4) R. L. Williams, "Fast High-Sensitivity Photodiodes," Journal of the Optical Society of America, Vol. 52, No. 11, November 1962, pp. 1237-1244.

Reference (5) N. P. Laverty, "The Comparative Performance of Electron Tube Detectors in Terrestrial and Space Navigation Systems," IEEE Transactions on Aerospace and Navigational Electronics, Vol. AME-10, No. 3, September 1963, pp. 194-205.

The General Electric bilinear mosanic, consisting of two orthogonal linear arrays of detectors, is less complex than a complete mosaic. In the detailed discussion following, note that a relatively large optic is required (4 in. dia), and that detection capability is limited to stars of +2 magnitude and brighter. Presumably, mechanical scanning of the telescope is required for star acquisition and tracking.

2.3.2 Future Developments

High resolution solid-state imaging detectors, using electronic interrogation, offer the greatest potential from the standpoint of reliability for future system applications. These devices are being developed by RCA Princeton Laboratories, Westinghouse Electric Corporation, the Autonetics Division of North American Aviation, General Precision, Inc., Molectro Corporation, and Belock Instruments Corporation, under Government contracts or independent research and development programs. With the exception of the Belock and RCA equipment, details of most of these instruments are no included in this survey. The feasibility of star detection has not been demonstrated. Details of design and performance are considered to be proprietary by the manufacturer in some cases.

2.3.2.1 Belock Star Field Mapper

This device, applicable to both wide-field star mapping and narrow-field star tracking, is in development for the Applications Technology Satellite for the NASA-Goddard Space Flight Center.

A thin-film layer of cadmium-sulfide photoconductive material is used as the optical radiation sensor. With stellar radiation incident on one side, the opposite side is scanned by illumination from an electronically switched matrix in which illumination is produced by excitation of phosphor. When the illumination from the electroluminescent matrix is incident on the element of the photoconductor upon which the stellar radiation is focused, complete photoconductivity through the cadmium-sulfide is established and the resultant current flow forms the electronic star signal.

2.3.2.2 RCA Solid-State Image Sensor

Under contract from the USAF Avionics Laboratory, RCA Princeton Laboratories have developed a solid-state image sensor for use as a television transducer. A high gain photoconductive material (CdS) is used for the photosensitive elements. Electronic charge storage, to increase sensitivity, has been investigated using two techniques: elemental capacitive storage, and nonlinear rectifying electrode materials (i.e., gold). The elemental size is approximately 2 x 2 mils, and a mosaic of 80 x 180 elements has been developed. The mosaic is fabricated by vacuum deposition.

Interrogation is accomplished using synchronized horizontal and vertical thin-film networks. The operation of each is similar to a serial chain of one-shot multivibrators. Coupling to the photosensitive mosaic by both diodes and TFT's (thin-film transistors) has been used. The former is currently in use. Continuous operation for approximately one year has been obtained without failure. The size of each (h and v) network is approximately one inch square.

This contract has been underway for 3 years. Within the immediate future, a complete solid-state television camera is to be delivered to AFAL by RCA. Although not specifically intended for use as a sensor of stellar radiation, if the detectivity of this device can be increased it will be extremely promising for future applications.

Table 2-III summarizes contemporary trackers using mechanical scanning. These employing electronic scanning are outlined in Table 2-IV. Systems which were considered but found not applicable for use in spacecraft are presented in Table 2-V.

2.3.3 Star Trackers Using Mechanical Scanning

The following pages contain a detailed discussion of each instrument outlined in Table 2-III.

	Manufacturer	Designation	Program	Developmental Status	Detector	Spectral Response	Scanning Method	Optical System	Field of View	
									Total	Instan
1	General Electric Co.	Precision Star Tracker	IR & D	One Prototype Developed	1P21 Photo-multiplier	S-4	Vibrating Reeds	3.8 in. dia. f/4.75 Cassegrain	NS	NS
2	General Precision Inc. Keartott Division	Miniature Stellar Sensor	R & D USAF	IN Development	Cadmium Sulfide	0.4-0.65 Microns	Single-Axis Movement of Cell Array	2.75 in. Dia. f/5.0 Catadioptric	15 x 30 Arc-Min	Twin Slit Cell Arra
3	Hughes Aircraft Co. (SBRC)	Canopus Star Tracker	NASA Surveyor	Operational	1P21 Photo-multiplier	S-4	Scanning Mirror and Rotating Reticle	Refractive	4° x 5°	NA
4	Kollsman Instrument Corp.	KS-137	NASA OAO	Space Qualified	1P21 Photo-multiplier	S-4	Vibrating Reed	1.25 in. dia. f/4.0 paraboloid	1° x 1°	NA
5	Kollsman Instrument Corporation	KS-187 Solid-State Tracker	R & D	IN Development	Silicon	0.4-1.1 Microns	Notation (Two-axis Oscillating Mechanism)	2.5 in. dia. f/1.25 refractive	1° Circular	NA
6	Sylvania Electric Products Inc.	Fine Bore-sight Tracker	NASA OAO	Space Qualified	1P21 Photo-multiplier	S-4	Rotating Chopper	20.5 in. dia. 410 in. f.l	±4 Arc-Min	0.3 Arc Slit

NOTE: NS = NOT SPECIFIED

NA = NOT APPLICABLE

Table 2-III. Summary of Star Trackers Using Mechanical Scanning

taneous	Type of Output	Gimbal Freedom	Stated Performance (Star Mag.)	Accuracy	Weight	Volume	Power	Remarks	References
	NS	±30° Cone	+6.0 Mag.	10 Arc-Sec	25 lb	1.0 ft.^{3}	23 W.		TRW SSGS Study Vol. III and ARS Paper No. 1930-61 "A Precision Star Tracker Etc" by D. R. McMurrow Et Al
.-V y	Two Axes Digital	None	+2.2 Mag. 900 FT-L	3 Arc-Sec	NS	49.5 in.^{3}	NS		
	Single Axis Analog	±15° One Axis	-0.44 Mag.	6 Arc-Min	4.9 lb	NS	5 W.		IRIS Proc. Vol. 8 No. 3 August 1963, Pg. 5 "The Surveyer Canopus Sensor" by A. H. Sochel & E. W. Peterson
	Two Axis Digital	±43°	+2.0 Mag.	22 Arc-Sec Per Axis 30 Arc-Sec Total (1σ)	45.1 lb	3.3 ft.^{3}	15.4 W.	Six Used on OAO.	Mfr. Brochure Dated Dec. 1965
	Two Axis Digital	±60° Two Axes	+2.5 Mag.	15 Arc-Sec RMS 1σ Total Both Axes	24 lb	NS	NS	In Development for Space Applications	Telecon with Kollsman Instrument Corp. 9-27-66
Sec	Two Axis Analog	None	+7.0 Mag.	0.1 Arc-Sec	NS	NS	NS	Uses OAO Stellar Spectrometer Optical System	AIAA Paper No. 63-211 "Fine Guidance Sensor for High Precision Guidance of the OAO" by N. A. Gundersen June 1963

	Manufacturer	Designation	Program	Developmental Status	Detector	Spectral Response	Scanning Method	Optical System	Tota
1	Barnes Engineering Co.	Canopus Tracker	NASA/ JPL Mariner	Operational	CBS CL-1147 Reconnotron	S-11	Electronic	1 in. dia. f/0.8 Cassegrain	4° Rol 30° Pi
2	Bendix Corp. Eclipse-Pioneer Division	OAO Star Tracker Subsystem	NASA OAO	Space Qualified	ITT FW 143 Photo-multiplier	S-20	Electronic	1.5 in. dia. f/5.3 Refractor	1° x 1
3	Electro-Mechanical Research Inc.	Model 569A	NASA Apollo	Being Qualified	ASCOP 571A-01-14 Quadrant Photo-multiplier	S-11	Electronic	3.0 in. dia. f/2.5 Refractor	3° dia.
4	Bendix Corp. Eclipse-Pioneer Division	Star Tracker	NS	Experimental	Bendix Channeltron Photo-multiplier	NS	Electronic	NS	30 arc-mi
5	General Electric Co.	Star-Planet Tracker	NASA Ames	Experimental	Electrostatic Vidicon	NS	Electron Beam	Dual Field Cassegrain/ Refractor	80-155 annulu and 2°
6	General Electric Co.	Bilinear Photomosaic Detector System	NS	Developmental	Two Linear Arrays of 360 El. Ea.	NS	Electronic	Effective Aperture dia <4.0 in.	N/S
7	Honeywell Radiation Center	Advanced Star Tracker	R & D	Developmental	ASCOP 568A Quadrant Photo-multiplier	S-11	Electronic	NS	1.5° x
8	ITT Federal Laboratories	Electro-optical Sensing Head	NASA OAO	Space Qualified	ITT FW 143 Photo-multiplier	S-20	Electronic	1.5 in. dia. f/5.3 Refractor	1° x 1
9	ITT Federal Laboratories	OAO Boresighted Star Tracker	NASA OAO	Space Qualified	ITT FW 143 Photo-multiplier	S-20	Electronic	2.85 in. dia. f/1.85 Refractor	10 arc-min
10	ITT Federal Laboratories	Canopus Tracker	NASA Lunar Orbiter	Operational	ITT FW 143 Photo-multiplier	S-20	Electronic	20 mm. dia. f/1.0	16° x
11	ITT Federal Laboratories	Dual Mode Star Tracker	NASA GSFC	Operational	ITT FW 143 Photo-multiplier	S-20	Electronic	2.0 in. dia. f/1.5	8° x 8 (Searc
12	Northrop-Nortronics	Star and Sun Tracker	R & D NASA MSFC	One Laboratory Model Delivered	EMR 568A Quadrant Photo-multiplier	S-11	Electronic	2.0 in. dia. f/9.0 Cassegrain	30 arc-dia. (S Tracki 1° dia. (Sun T
13	Northrop-Nortronics	Range Instru-mentation Ship Tracker	NASA Apollo	Production	RCA C73496 H Vidicon	0.4-0.6 Micron	Electron Beam	3.5 in. dia. f/16.0 Cassegrain	10 x 10 arc-mi

NOTE: NS = NOT SPECIFIED

NA = NOT APPLICABLE

Table 2-IV. Summary of Star Trackers Using Electronic Scanning

Field of View		Type of Output	Gimbal Freedom	Stated Performance (Star Mag.)	Accuracy	Weight	Volume	Power	Remarks	References
	Instantaneous									
ch	0.86° Roll 11° Pitch	Single Axis Analog	None	+0.6 Mag.	±0.1°	6 lb	220 In.^{3}	6.0 W. (Pk.)		"Star Sensor Survey" TRW Memo 9353.3-434 by P. B. Hutchings 28 June 1965
	NS	Two Axis Analog	±60°	+2.5 Mag.	25 Arc-Sec on each Axis	14.5 lb (Inst.) 12.0 lb (Elect.)	990 In.^{3} (Inst.) 705 In.^{3} (Elect.)	10.0 W. (Inst.) 4.0 W. (Elect.)	Alternate for Kollsman Subsystem on OAO. Six Units/Set.	Mfr. Brochure "Bendix OAO Star Tracker Subsystem"
	3° dia.	Two Axis Analog	None	+2 Mag.	30 Arc-Sec (m = +2) 12 Arc-Sec (m = 0)	5 lb	117.5 In.^{3}	3 W.	For Apollo X-Ray Astronomy Experiment	EMR Brochure Sept. 1966
ı	NS	Digital	None	+3.0 Mag.	9.0 Arc-Sec	3.0 lb	95 In.^{3}	5.0 W.		TRW SSGS Study Vol. III, Pg. III-30 29 May 1964
cone	NS	Two Axis Digital	None	NS	3.0 Arc-Min (Wide Field) 2.0 to 5.0 Arc-Sec (Fine)	23 lb	0.5 Ft.^{3}	20.0 W		AIAA Paper No. 63-15 "An Advanced Optical - Inertial Space Navigation System" by J. D. Welch June 1963
	NS	Two Axis Digital	None	+2 Mag.	NS	NS	NS	NS		P. B. Hutchings Op. Cit.
1.5°	1.5° x 1.5°	Two Axis Analog	±20° One Axis	+1.0 Mag.	±27 Arc-Sec (Roll) ±5 Arc-Min (Pitch)	5.5 lb	185 In.^{3}	7.0 W. Av. 10.0 W. Pk		P. B. Hutchings Op. Cit.
	NS	Two Axis Analog	None	+2.5 Mag.	±9 Arc-Sec	o.0 lb	155 In.^{3}	4.5 W.	Used in Bendix OAO Star Tracker Subsystem	ITT Data Sheet
	NS	Two Axis Analog	None	+6.0 Mag.	1.5 Arc-Sec RMS (m = +4) 10 Arc-Sec RMS (m = +6)	9.0 lb Head 14.0 lb Elect.	135 In.^{3} (Head) 660 In.^{3} (Elect.)	7.7 W.	Boresighted with OAO Primary Telescope	ITT Data Sheet
.2°	16° x 1°	Single Axis Analog	None	+0.08 Mag.	±50 Arc-Sec RMS	7.0 lb	264 In^3	8.0 W		P. B. Hutchings Op. Cit. and ITT Data Sheet
ı	32 x 32 arc-min (Track)	Two Axis Analog	Electronic 8° x 8°	+3.0 Mag.	5 Arc-Sec RMS (m = 0) 12 Arc-Sec RMS (m = +3)	9.5 lb	260 In.^{3}	8.0 W.	Used on Aerobee Rocket Probes	P. B. Hutchings Op. Cit. and ITT Data Sheet
nin ır g) ıcking)	30 arc-min dia. 1° dia.	Two Axis Analog	None	+2.5 Mag.	10.0 Arc-Sec RMS	9.0 lb (Flight Model)	100 In.^{3} (Flight Model)	8.0 W. (Flight Model)		Mfr. Brochure
	NA	Two Axis Digital	None	+2.5 Mag. 1000 Ft.-L +3.5 Mag. (Nite)	2.8 Arc-Sec (Elev.) 1.0 Arc-Sec (AZ1)	9.5 lb (Flight Model)	265 In.^{3} (Flight Model)	12.0 W. (Flight Model)	Acquisition Time = 5 Sec used on Stabilized Platform for Alignment of SINS IMU	Mfr. Brochure "NCN-121 Star Tracker" 26 July 65

Table 2-V. Star Tracking Systems Not Considered for Space Application

Manufacturer	Type of Star Tracker	Program	Characteristics Preventing Use In Space Applications
AC Spark Plug	Photoelectric - Mechanical Single-Axis Scan	USAF Stellar - Inertial Guidance System	Not developed beyond prototype stage
Astrionics Corp. of America	Photoelectric - Rotating Reticle	NASA/University of Wisconsin X-15 Star Photography Experiment	Single purpose device, limited versatility, For Aircraft Experiment
EMI, LTD.	Vidicon	Concorde Aircraft	In experimental stage of development
Farrand	Image Orthicon, Shipborne	U. S. Navy	Extremely large size
General Precision Inc. Librascope Division	Photoelectric	NASA Stratoscope	Single purpose device for balloon experiment
General Precision Inc. Librascope Division	Astroguide Space Vehicle Navigation System	IR & D	Breadboard system, very large image dissector requires ASN-24 computer (or equiv) excessive weight, volume, power
Honeywell Radiation Center	PCM	USAF Avionics Laboratory R & D	Insufficient detection capability
IMPRO Corp.	Venus/Canopus	JHU Balloon Experiment	Extremely large size
Kollsman Instrument Corp.	KS-37, KS-50, KS-85, KS-120, KS-140	USAF	Aircraft Astrosextants, Excessive weight, volume, and power
Kollsman Instrument Corp.	KS-177, KS-192	IR & D	Aircraft Astrosextants, Excessive weight, volume, and power
Kollsman Instrument Corp.	Photoelectric	NASA Apollo	Accessory for manually operated navigation sextant
Kollsman Instrument Corp.	Photoelectric	NASA Gemini	Not developed
Kollsman Instrument Corp.	Image Dissector Strapdown	NASA Gemini	Not developed
Litton Industries	Solid State Mechanical Scan	USAF SIDS	Primarily for aircraft application
North American Aviation Autonetics Division	N2C, N3A, N3B, Photoelectric	USAF	For aircraft application, excessive weight, volume, and power
North American Aviation	AVN-2, Solid-State	USAF	For aircraft application, excessive weight, volume, and power
Northrop-Nortronics	Photoelectric	USAF Snark Missile	Excessive Weight, Volume, and power
Northrop-Nortronics	Photoelectric	USAF Skybolt Missile	Extremely complex, excessive weight and power
Northrop-Nortronics	Vidicon, Model A-8	USAF R & D	Experimental only. Excessive weight. Electronics in breadboard form.
Northrop-Nortronics	Vidicon, Model A-11	IR & D	Complete system for missile application. Excessive weigt, volume and power.
Northrop-Nortronics	Vidicon, Precision Navigation System (PNS)	USAF	For aircraft application, excessive weight, volume, and power
Northrop-Nortronics	Vidicon	USAF Staff Program	Excessive Weight and Power
Perkin-Elmer Corp.	Photoelectric, Rotating Wedge	NS	Used for pointing of ground based astronomical telescopes
Raytheon	Image Dissector	U.S. Navy, Polaris Missile Development	Excessive Weight

NOTE: NS = NOT SPECIFIED

2. 3. 3. 1 Precision Star Tracker

Manufacturer: General Electric Co.
Missile and Space Vehicle Dept.
Philadelphia, Pa.

A prototype model of a Precision Star Tracker was developed by the General Electric Co. under an independent research and development program. Development has not been carried beyond the prototype stage. The overall system consists of six major components: an optical telescope, a scanner, a detector, the gimbal and housing structure, the gimbal pick-off devices, and the associated servo subsystem (electronics package). The system consists of two packages - the tracker proper and the electronic package.

The telescope is mounted in a pair of gimbal rings arranged to permit motion about the pitch and yaw axes. Sufficient gimbal action is provided to allow the telescope to be trained anywhere in a cone with an apex angle of 30 deg. A mechanical scanning system at the focal point of the telescope modulates the light received by the photosensitive detector. Suitable electronics are provided to amplify the signal from the photodetector, to discriminate it from the background noise, to determine whether the signal is from a star of the proper magnitude, and to order the telescope to resume the search if it is the wrong magnitude. Servos (dc torque motors) are provided to drive the gimbals, with the gimbal pickoff devices to generate error signals proportional to the gimbal angular displacements.

The tracker has two modes of operation: search and track. The search begins on command; it can be either a 15-deg spiral about the center position, or a 5-deg spiral about any other position at gimbal angles between zero and 10 deg. When a star enters the telescope field-of-view, a signal appears at the photomultiplier output. The signal is fed to the electronics package which temporarily interrupts the search, feeds the signal to a control loop which finely positions the telescope until the star is exactly in the center of its field, and determines whether the star is within the required magnitude range. If it is within the required range, the search is discontinued and the track mode is initiated; if it is not, the search mode is resumed.

When a star of the proper magnitude enters the field of the telescope, the tracker electronics package switches the servo subsystem to the track mode, locking the telescope on the star. Any displacement of the star from the telescope axis will generate photoelectric error signals which are amplified and fed to the torquers to reposition the gimbals to keep the star on the telescope axis. The orientation of the telescope (and thus the "star line") relative to the vehicle is given by the electrical outputs of the gimbal-angle transducers.

Physical and Performance Data:

Modes of Operation	Pointing (Command) Search Track
Gimbal Freedom	30° cone
Tracking Rate	36°/min
Sensitivity	+ 6 star magnitude
Optical System	Cassegrain, 3.8 in. dia f/4.75
Scanner	Two vibrating reeds driven at 400 cps
Detector	RCA 1P21 Photomultiplier
Gimbal Encoders	19 - Bit Phasolvers
Accuracy:	
Encoders	2.5 arc sec 3σ
Servo Threshold	2.0 arc sec 1σ
Scanner Stability	4.5 arc sec
Overall	10 arc sec
Temperature Limits	+ 40 to + 120°F
Pressurization	1 atmosphere, dry nitrogen
Weight	25 lb
Volume	1 ft^3
Power	23 w (search) 18 w (track)
MTBF	One year in orbit

2.3.3.2 Miniature Stellar Sensor

Manufacturer: General Precision Inc.
Kearfott Division
Little Falls, New Jersey

Functional Description:

This solid-state sensor has been designed to detect and locate stars during extremely bright sky background conditions. The sensor provides two digital signals which locate a star within the field-of-view about two axes.

The sensor consists of an optical system which collects and focuses the star energy. A unique mechanical scanning device, impervious to vehicular disturbances, crosses the field-of-view and completely determines image position in a single scan. The detector and an optical linear encoder are rigidly attached to the scanner; the encoder, which is a continuous position monitor, permits reading in digital form the position of the detector at the moment of detection.

The photoconductive detector is fabricated in a twin-V geometric configuration with electrodes interposed between the active photoconductive line elements. The electrodes are so connected as to form two electrical bridge circuits which remain balanced for all conditions of fluctuating sky background brightness. The presence of a star image on any line element, however, instantaneously unbalances the bridge. The electrical pulse, as generated by the presence of a star image on any line element, is used in conjunction with pulses generated by the optical position encoder to provide digital star position data with respect to the optical line of sight.

Physical and Performance Data:

Star Visual Magnitude (m_v)	+2.2	+1.2	+0.2	-4 (Planet)
Sky Background (foot lamberts)	900	1500	3500	3500
Signal-to-Noise Ratio	6:1	6:1	6:1	240:1
Field-of-View in arc minutes	15 x 30			
Optical System	2.75 in. dia/F5.0 Catadioptric			
Scan rate in sec/field:	4			
Inherent accuracy in arc sec:	3			
Size	3"D x 7"L*			

* Present prototype model includes optics, detector, scanner, and some electronics.

2.3.3.3 Canopus Tracker

Manufacturer: Santa Barbara Research Center
Goleta, California

Functional Description:

The Canopus Tracker will provide roll orientation information for midcourse guidance of the Surveyor lunar landing spacecraft. In operation, the longitudinal axis of the Canopus Tracker is parallel to the spacecraft's roll axis, which is directed at the sun by auxiliary sensors. A mirror on the tracker permits prelaunch setting of the acquisition field of view of $\pm 2.3^{o}$ which will accommodate a launch date uncertainty as high as ±48 days. Star radiation collected by an objective, first is phase-modulated in the roll direction, then frequency-modulated by one band of a two-frequency chopper, and finally focused onto a photomultiplier detector.

A unique method of star magnitude discrimination is used. Since the longitudinal axis of the tracker is aimed at the sun, a precisely known fraction of the solar irradiance at the instrument can be collected by an auxiliary optical path. The solar radiation passes through a second frequency band on a chopping disc and falls on the photomultiplier. The output of the photomultiplier is amplified, and the star and sun frequency components are isolated by bandpass filters. The solar signal controls the dynode voltages of the photomultiplier, and thus controls the gain of the star channel. In this way the star signal is proportional to the ratio of stellar irradiance to solar irradiance and the navigation star is selected by knowing this ratio.

When the appropriate star signal is selected, the modulation envelope is detected and its phase compared with reference pips generated during roll phase modulation. Thus a proportional roll angle error signal is generated over $\pm 2^{o}$ to control the roll attitude of the spacecraft.

Physical and Performance Data

Field-of-View	4 X 5 deg
Gimbal Freedom	± 15 deg (one axis)
Sensitivity	-0.44 star magnitude
Accuracy (Roll Angle	6 arc min
Weight	4.9 lb
Power	5 w

2.3.3.4 KS-137 Satellite Star Tracker

Manufacturer: Kollsman Instrument Co.
Syosset, New York

Functional Description:

Kollsman is currently manufacturing the KS-137 for the stabilization and control system of NASA's Orbiting Astronomical Observatory (OAO). The KS-137 system consists of the tracker, tracking electronics and the pick-off electronics. The tracker consists of a telescope, two positioning gimbals, precision angle pick-offs, and a housing. The telescope contains two sets of reflective optics, two vibrating reed scanners, and a 1P21 photomultiplier tube. It modulates the light entering it to produce basic analog tracking and error signals. A sun shield is provided to permit operation as close as 30° to the and 12° to the earth's horizon.

Each gimbal is provided with a direct drive DC torque motor and an incremental angle encoder called a Phasolver. The torque motors can respond to both position command signals and the telescope-derived error signals. The tracking electronics package contains all the circuitry needed to operate the star tracker other than that required for the output angular measurements. The pick-off electronics contain all the Phasolver drive and output electronics.

Physical and Performance Data:

Total Accuracy (each axis)	±22 arc sec 1σ
Phasolver accuracy (each)	± 5 arc sec
Telescope Field-of-View	$1^{\circ} \times 1^{\circ}$
Gimbal Travel (each)	$\pm 43^{\circ}$
Sensitivity	±2.0 M and brighter
Acquisition Rate	$\leq 1^{\circ}/5$ sec
Optics	1.25"D f/4 reflective
Weight	
Tracker	23.5 lb
Tracking Electronics	19.0 lb
Pick-off Electronics	2.6 lb

Size	
Tracker	17.5" x 16.8" x 16.3"
Tracking Electronics	16.0" x 11.0" x 4.0"
Pick-Off Electronics	12.0" x 7.8" x 2.6"
Power (average)	
Tracker	1.6 w
Tracking Electronics	11.3 w
Pick-Off Electronics	2.5 w
Reliability (1 year in space)	0.92

2.3.3.5 KS-187 Solid State Tracker

Manufacturer: Kollsman Instrument Co.
Syosset, New York

Functional Description:

Kollsman Instrument Co. is presently developing a solid-state star tracker for use in spacecraft. The design is based upon techniques employed in the KS-177 and KS-192 star tracking systems which have been developed for aircraft navigation. To date, all development work on these three systems has been conducted under independent research and development programs.

The KS-187 tracker consists of both the tracking instrument and associated electronics in a single housing. The tracking telescope has freedom in both level and cross-level, and is mounted in pitch and roll gimbals. The optical system is refractive, and the optical star image is nutated on the silicon cell detector by the use of two oscillating mechanisms which are 90° out of phase. A sun shield will permit operation of an angle of 25° to the sun and 1° to the earth's horizon. Each gimbal is provided with a dc torque motor drive. Phasolver angle encoders are used. The torque motors can respond to both command signals and error signals from the star tracker.

Physical and Performance Data:

Optical System	2.5 in. dia , 3.125 in. f.l. refractor
Optical Field-of-View	1 deg dia
Sensitivity	+2.5 star magnitude

Acquisition Time	0.5 sec.
Signal Bandwidth	1 cps
Gimbal Freedom	$\pm 60^{o}$ (two axes)
Accuracy	15 arc sec rms 1σ (both axes combined)
Weight	24 lb
Volume	not specified
Power	not specified

2.3.3.6 OAO Fine Guidance Sensor

Manufacturer: Sylvania Electronic Systems
Waltham, Mass.

Functional Description:

The Fine Guidance Sensor utilizes the optical system of the high resolution stellar spectrometer in the Princeton Experiment on the OAO.

The 80 cm Dia Cassegrain optical system focuses the starlight on the slit of the spectrometer which has a spacing of 24 microns, corresponding to 0.3 arc sec. The diameter of the star image is approximately 50 microns in diameter. The light passing through the slit is used for spectrographic analysis and the light reflected from the slit jaws is used for the Fine Guidance Sensor.

The Fine Guidance Sensor measures the angular error across the slit width by comparing the relative intensity of the energy falling on each of the slit jaws. A biprism, located in the path of the reflected light, provides a means for measuring the angular error along the length of the slit.

After passing through a fused silica field lens, the light is reflected from a motor driven plane mirror with one opaque and one reflective sector. At any instant, light from only one quadrant is incident on the photomultiplier detector. The range in star magnitude varies from zero to seventh magnitude, with AGC being used to vary both photomultiplier and amplifier gain. The AGC is also used as a star presence signal.

Redundancy is provided through the use of two complete guidance sensor systems which may be selected by ground command.

Physical and Performance Data:

Total Field-of-View	±4 arc min (two axes)
Optical System	Cassegrain, 50 cm dia , f/20
Accuracy	± 0.1 arc sec (normal to spectrometer slit)
Error Signal Noise	0.005 arc sec
Sensitivity	0 to +7 star magnitude
S/N Ratio	50/1 at null (+7 star mag.)
Time Constant (Fine Guidance Mode)	5 sec
Reliability	0.9382 for one year in orbit

2.3.4 Discussion of Star Trackers Using Electronic Scanning

The following pages contain a detailed discussion of each instrument outlined in Table IV.

2.3.4.1 Canopus Star Tracker

Manufacturer: Barnes Engineering Co. Stanford, Connecticut and Jet Propulsion Laboratory Pasadena, California

Functional Description:

This tracker was developed at the Barnes Engineering Company and the Jet Propulsion Laboratory of the California Institute of Technology for use on NASA's Mariner spacecraft. The tracker provides one analog error signal locating the line to the star Canopus about one axis. The tracker is a single package which contains optics, an image dissector tube detector (CBS Reconotron, CL-1147), sun sensor and shutter, and the electronics needed to operate the tracker.

Physical and Performance Data:

Total Field-of-View	4^{o}(roll) x 30^{o}(pitch)
Instantaneous Field-of-View	0.86^{o}(roll) x 11^{o}(pitch)
Gimbaling	All electronic
Roll	Continuous sinusoidal sweep
Pitch	Six positions - 4.6^{o}/position

Sensitivity (threshold settings)	+0.6M to -2.4M
Null Offset	±0.1°
Error Gradient (at null)	8 v/deg
Equivalent Noise	0.013° peak-to-peak
Time Constant (roll axis)	0.5 sec
Optics	1" D f/0.8 semi-solid Cassegrain Schmidt
Size	4" x 5" x 11"
Weight	6 lb
Power	1.8 w (average)
	≈6.0 w (peak)

2.3.4.2 OAO Star Tracker Subsystem

Manufacturer: The Bendix Corporation
Eclipse-Pioneer Division
Teterboro, New Jersey

Functional Description:

The subsystem was designed, developed and tested under NASA Contract NAS 5-2018 for application to the Orbiting Astronomical Observatory (OAO). This development is a backup to the Kollsman KS-137 Satellite Star Tracker. The subsystem consists of a star tracker assembly and electronics. The star tracker assembly consists of a tracking head supported by two gimbals each with ±60° freedom. The gimbals are driven with direct coupled dc torque motors and the gimbal positions are read with 16-bit optical encoders. A labyrinth seal and lubricant reservoir are employed to provide lubrication of the gimbal bearings in the space environment.

The tracking head is manufactured by ITT Federal Laboratories. The tracking head is capable of tracking a selected star within its field-of-view and providing analog voltages which are proportional to the angle between the optical axis and the star line.

The electronics include the gimbal servo electronics, the gimbal command or track networks, power control circuit, resolver circuits, and telemetry circuits. The tracking head electronics are included in the tracking head.

Physical and Performance Data:

Gimbal Readout Accuracy (each)	25 arc sec
Gimbal Angle (each)	±60°
Tracking Accuracy	±9 arc sec
Tracker Head Field-of-View	1° x 1°
Sensitivity	±2.5 m and brighter
Optics	1.5"D f/5.3 refractive
Weight	
Star-Tracker Assembly	14.5 lb
Electronics	12 lb
Size	
Star-Tracker Assembly	11" x 9.5" x 9.5"
Electronics	11" x 16" x 4"
Power	
Star-Tracker Assembly	10 w
Electronics	4 w (average)

2.3.4.3 Star-Tracker Sensor for Apollo X-Ray Astronomy Experiment

Manufacturer: Electro-Mechanical Research, Inc.
Princeton, New Jersey

Functional Description:

A photomultiplier tube with a quadri-sected cathode, designed for use as a null sensor in star tracking or other image-locating systems, has been developed. For the Apollo X-Ray Astronomy Experiment, this device was chosen as the basic sensor in a local attitude reference system mounted with the experiment's prime sensor. The attitude sensor includes the quadrant tube, collecting optics, boresight adjustment provisions, a shutter for protecting against direct sunlight exposure (with an auxiliary sun sensor), power supplies, switching and decommutation circuits, and an automatic gain control loop for normalization of behavior when tracking stars of different brightnesses. Seven primary outputs are provided: X-deflection, Y-deflection, the individual decommutated outputs of the four quadrants, and a star-brightness indication derived from the AGC loop. The intended application being in a relatively slow feedback loop closed by manual control of body attitude, the star image is deliberately defocused, so as to provide a shallow slope near null and quasi-linear outputs over an extended range of error angles. The spot size chosen is 0.080 inch, which provides approximately-proportional error-angle indication within ±15 arc minutes of null. The signal bandwidth is two cycles per second, chosen to optimize the tradeoff between dynamic tracking errors and noise filtering. Increasing aperture and/or focal length can be used to improve this figure, as can reduction in bandwidth if system dynamics permit.

When not operating, the unit meets Apollo Service Module launch environment requirements. When operating, it tolerates temperatures from $-55^{\circ}C$ to $+71^{\circ}C$, and pressures from sea level to orbital ambient, including intermediate pressures. Thequadrant tube in the Apollo tracker has an antimony-cesium photocathode. The dark current limits the performance at $71^{\circ}C$ so that precise tracking is possible only on stars of zero magnitude or brighter. Other cathode materials can be used to restore high-temperature dynamic range. Additional growth capability resides in the applicability of beam-splitting optics to the achievement of fractional arc second null precision without sacrifice of response speed.

<u>Physical and Performance Data:</u>

Optical System	3-in dia f/2.5 refractor
Total Field-of-View	3-deg dia
Instantaneous Field-of-View	3-deg dia
Detector	ASCOP 571A-01-14 quadrant photomultiplier tube
Spectral Response	S-11
Scanning Method	Sequential interrogation of photomultipler photocathode quadrants
Output	Two analog voltages on two axes
Accuracy	*12 arc sec on 0 mag star 20 arc sec on +1 mag star 30 arc sec on +2 mag
Weight	5 lb
Volume	117.5 in^3 (approx.)
Power	3 w

* 5 arc sec with defocused image

2.3.4.4 Channeltron Star Tracker

Manufacturer: Bendix
Eclipse-Pioneer Division
Teterboro, New Jersey

Functional Description

An experimental tracking system is being developed by Bendix utilizing the Channeltron photomultiplier tube, in which the conventional dynode structure is replaced by small narrow glass tubes coated internally with a resistive coating capable of secondary electron emission. With voltage applied along the wall of the tube, current gains as high as 10^6 are possible. Photocathodes with both S-9 S-11 spectral response have been made. Through the use of multiple channels with the ends in close proximity to the photocathode, with individual electrical connections, zonal recognition or "scanning" of the photocathode is possible.

Physical and Performance Data:

Optical System	Not specified
Field-of-View	30 arc min
Accuracy	9.0 arc sec
Scanning	Electronic
Sensitivity	+3.0 star magnitude
Weight	3.0 lb
Volume	95 in^3
Power	5.0 w

2.3.4.5 Star-Planet Tracker

Manufacturer: General Electric Company
Defense Electronics Division
Johnson City, New York

Functional Description:

The development over the past three years of this tracker has been funded by GE and NASA/Ames. The system is now operable and has demonstrated its tracking capability. Target location is provided in digital form. The capabilities of the tracker have been expanded to locate multiple targets and extended bodies.

The tracker consists of a dual-field optical system, vidicon and pre-amplifier assembly, T. V. camera electronics, data processor, and power supply. A one inch electrostatic vidicon is the detector and a unique reticle pattern is employed to provide precise location of the target and largely eliminate sweep drift and nonlinearity.

Physical and Performance Data:

Optic	Dual-field Cassegrainian/refractor
Fields-of-View	80-155 deg annulus (wide field) 2 deg dia (narrow field)
Accuracies	3.0 arc min (wide field) 2.0 to 5.0 arc sec (narrow field)
Weight*	23 lb
Volume*	0.5 ft^3
Power*	20.0 w

*Estimated flight package based on existing equipment

2.3.4.6 Bilinear Photomosaic Detector System

Manufacturer: General Electric Co.
Defense Electronics Division
Johnson City, New York

Functional Description:

The bilinear photomosaic tracker system utilizes two linear arrays of solid state photosensitive elements to establish the X and Y image plane coordinates of a point source target and provides this information as an output in direct digital form.

The system differs from conventional, two-dimensional arrays or mosaics in that the conventional devices have a separate detector for each resolution element. The task of mechanizing such a system, of even moderate resolution capability, is formidable. In the bilinear system, however, the image is optically resolved into its X and Y coordinates and 2n elements replace the n^2 elements required by the conventional mosaics. Thus 720 elements provide 129,600 bits of information.

Laboratory tests of a breadboard system have demonstrated the practicality of the basic concept, and have established that the sensitivity of the detector arrays is sufficient to track 2nd magnitude stars ($+2M_V$) with optics of less than 4" effective aperature. The bilinear mosaic concept offers: the increased reliability inherent in solid-state static devices, significant reductions in size, weight, and power, high sensitivity and large dynamic range, off axis and extended body-tracking capability, and no moving parts.

2.3.4.7 Advanced Star Tracker

Manufacturer: Honeywell, Radiation Center
Boston, Mass.

Functional Description:

This device is a single-gimbal two-axis star tracker designed for use in space vehicles. It consists of a combined reflecting and refracting optical system, a quadrant multiplier phototube with HV power supply, a quadrant switching network, a data processing system to produce analog error signals in pitch and roll, and a sun sensor shutter system to protect the phototube.

The sensor used in the star tracker is a Type 568A quadrant multiplier phototube manufactured by ASCOP, a division of Electro-Mechanical Research, Inc. This tube is a multiplier phototube which has its photocathode divided into four sectors. Each photocathode sector is activated by switching a 170-v potential between it and the first dynode of the tube. If a photocathode quadrant is shorted to the first dynode, that sector of the photocathode will effectively be turned off even though an optical image may be imaged on it. By sequentially switching each sector off and on, an image scanning method is achieved, and error signals may be derived by appropriate circuitry.

Physical and Performance Data:

Field-of-View	$1.5^\circ \times 1.5^\circ$ ($1.5^\circ \times 40^\circ$ with gimbal)
Linear Range (roll)	±15 arc min
Null Accuracy (roll)	±27 arc sec
Pitch Error (max)	±5 arc min

Sensitivity (threshold gates)	+1.0 to -3.0 M
Size	8.7" x 5.3" x 4.0"
Weight	5.5 lb
Total Power	7.0 w (average) 10.0 w (peak)
MTBF	85,000 hr

2.3.4.8 Electro-Optical Sensing Head

Manufacturer: ITT, Federal Laboratories
San Fernando, California

Functional Description:

The electro-optical sensing head is a compact accurate star tracker which is presently employed as the tracking head of the Bendix OAO Star Tracker Subsystem. The head consists of an optical system, and ITT FW 143 multiplier phototube detector, detector electronics, sun sensor, and sun sensor electronics. The head provides two analog error voltages which locate the star within the field-of-view about two axes.

Physical and Performance Data:

Star Magnitude Sensitivity	+2.5
Field-of-View	1^{o} x 1^{o}
Tracking Accuracy	± 9 arc sec
Bandwidth	10 cps
Error Gradient	0.167 v/arc min
Optics	1.5"D, f/5.3 refractive
Photo Surface	S-20
Gimbaling	electronic
Size	5 5/8" x 5 1/4" x 5 1/4"
Weight	6 lb
Power	4.5 w

2.3.4.9 OAO Boresighted Star Tracker

Manufacturer: ITT, Federal Laboratories
San Fernando, California

Functional Description:

The OAO boresighted star tracker is presently used for orientation of the OAO and is accurately aligned to the experiments. The tracker consists of an optical system, an ITT FW 143 multiplier phototube detector, electromagnetic detection circuitry, sun shield, offset tracking electronics, and vehicle attitude logic electronics. The tracker provides two analog error signals which locate the star in the field-of-view about two axes.

Physical and Performance Data:

Star Magnitude Sensitivity	+6
Field-of-View	10 arc min
Tracking Accuracy	1.5 arc sec rms (+4 star mag); 10.0 arc sec rms (+6 star mag)
Error Filter Bandwidth	1/2 cps
Error Gradient	1 v/arc min
Photo Surface	S-20
Optics	2.85"D, f/1.9 refractive
Gimbaling	electronic $+1.5^{\circ}$ in 15 arc min steps
Weight (total)	25 lb
Power	7.7 w
Size	
Sensor	3" x 15"
Electronics	5" x 11" x 12"

2.3.4.10 Canopus Tracker

Manufacturer: ITT Federal Laboratories
San Fernando, California

Functional Description:

The tracker is presently being developed to be used in the attitude control system of NASA's Lunar Orbiting Photographic Spacecraft. The tracker provides one analog output signal which is proportional to the angular displacement of the line to Canopus about one axis. The tracker is a single package consisting of optics, an ITT FW 143 multiplier phototube detector, and electronics. The electronics contain detector electronics, scan logic, deflection electronics, and power supplies.

Physical and Performance Data:

Stellar Sensitivity (as set by threshold gates)	-1.92 M to 0.08 M
Total Field-of-View	$8.2^{\circ *} \times 16^{\circ}$
Instantaneous Field-of-View	$1^{\circ} \times 16^{\circ}$
Null Stability	50 arc sec rms
Equivalent Angular Noise	15 arc sec rms
Error Bandwidth	10 cps
Error Gradient (over 2°)	1 v/deg
Optics	20 mm f/1.0
Weight	7 lb
Size	4" x 5.5" x 12"
Power	8 w

*Axis of control

2.3.4.11 Dual Mode Star Tracker

Manufacturer: ITT, Federal Laboratories
San Fernando, California

Functional Description:

The dual mode star tracker provides two analog output signals which are directly proportional to the angular displacement of the star from each of two orthogonal planes where their intersection is the optical axis of the sensor. The tracker consists of an optical system, an ITT FW 143 multiplier phototube detector, detector electronics acquisition scan electronics, track scan electronics, control logic electronics, and processing electronics. This tracker has been developed for NASA Goddard Space Flight Center and used on the Aerobee Rocket Probe.

Physical and Performance Data:

Star Magnitude Sensitivity	+3
Field-of-View	
Acquisition Mode	$8^{o} \times 8^{o}$
Track Mode	32' x 32'
Tracking Accuracy	5 arc sec rms (m = 0), 12 arc sec rms (M= +3)
Acquisition Time	$\leq$ 1 sec
Bandwidth	5 cps
Error Gradient	0.3 v/arc min
Optics	2"D, f/1.5 refractive
Gimbaling	electronic
Size	5" x 10.5" x 5"
Weight	9.5 lb
Power	8 w

2.3.4.12 Star and Sun Tracker

Manufacturer: Northrop - Nortronics
Palos Verdes Peninsula, Calif.

Functional Description:

This subsystem was designed and developed for the Astrionics Laboratory of the NASA Marshall Space Flight Center. One engineering model has been delivered for laboratory evaluation.

This optical system is a Wright-corrected Cassegrain telescope which is used for star tracking. For tracking the sun, a small refracting lens is used in conjunction with a shadow mask. In both cases the incident radiation is imaged onto the segmented photocathode of an EMR type 568A quadrant photomultiplier tube by a cluster of four field lenses.

Scanning is performed electronically by sequentially interrogating the four quadrants of the photomultiplier photocathode at a rate of 4 kc.

The signal detection circuits are analog and the outputs consist of two analog voltages defining the position of the star or sun with respect to the two coordinates orthogonal to the optical axis. In addition, a signal indicating detection of the celestial body is supplied.

Functional and Performance Data:

Tracking Accuracy	10 arc sec rms (each axis - star tracking
	10 arc sec rms (each axis - sun tracking)
Sensitivity	+1.8 to +2.5 star magnitude (depending upon density of galactic background)
Optical Field-of-View	30 arc min (star tracking) 60 arc min (sun tracking)
Weight	9 lb (flight model)
Volume	100 in^3 (flight model)
Power	8 w (flight model)
MTBF	37,000 hr (estimate, using high reliability component parts)

2.3.4.13 NCN-121 Star Tracker Subsystem

Manufacturer: Northrop - Nortronics
Palos Verdes Peninsula, Calif.

Functional Description:

The subsystem was designed and developed under the Apollo Range Instrumentation Ship Program for NASA and the U.S. Navy. The tracking head assembly is gyro-stabilized on two axes orthogonal to the line of sight and is mounted on a remote platform which is optically linked to the Inertial Measurement Unit of the SINS navigation system. The primary function is updating the azimuth alignment of the SINS IMU. In addition, star elevation angles may be measured to obtain a position fix.

Although this subsystem has been designed for shipborne application, the circuit design utilizes integrated circuits which may be compactly packaged for space applications. The total electronic component count is 1100. Several sizes of optical systems have been developed for shipborne, aircraft, missile, and space applications.

A miniature 1/2-in. diameter RCA C-73496H electromagnetic vidicon is used to sense stellar radiation. The sweep voltages are generated in digital form. Signal detection circuits are analog, and the error signal logic circuits are digital.

The optical system is a Wright-corrected Cassegrain telescope. Contained within the optical head assembly are a sun sensor and protective shutter mechanism, and a signal preamplifier. The primary feature of this subsystem is rapid star acquisition in the presence of scattered solar radiation either within the atmosphere or within the optical window of a spacecraft.

Physical and Performance Data:

	Ground or Shipborne	Spacecraft
Optical System		
Weight	9.5 lb	4.5 lb
Volume	240 $in.^3$	160 $in.^3$
Power	2.75 w	2.0 w
Field-of-View	10 x 10 arc min	30 x 30 arc min
Electronics		
Weight	30 lb	5 lb
Volume	3800 $in.^3$	105 $in.^3$
Power	25 w	10 w
Accuracy	2.8 arc sec (El) 1.0 arc sec (Az)	8.4 arc sec (El) 3.0 arc sec (Az)
Acquisition Time	5 sec	5 sec
Star Magnitude	+2.5 (day) +3.5 (night)	+2.5
Sky Brightness	1,000 ft - 1	0 ft - 1
Operating Temperature	-15^{o} to $100^{o}C$	-54^{o} to $125^{o}C$
Humidity	100%	100%
Vibration	MIL-STD-202	Random 0.05 g^2/cps
Altitude	Sea Level	80,000 ft
MTBF	6,000 hr	14,800 hr

2.4 STAR FIELD SENSORS

2.4.1 General Discussion

The process of surveying available hardware of the star-field sensor type is somewhat more difficult than it is for more conventional sensing hardware. The concept of scanning, identifying, and making attitude or navigation measurements from a star field has been frequently explored, but hardware mechanizations have been few, and in most cases are only of an experimental nature. Table VI and the following descriptions do not include all proposed devices but only those which have been developed at least to the breadboard stage.

The star-field sensor differs from the usual star tracker in that the characteristics of a group of stellar targets are utilized simultaneously for attitude sensing or navigation. The star-tracker instrument, by contrast, generally locks upon a single star in a narrow field-of-view, obtaining multiple observations by sequential operation or by means of multiple tracker heads. The operations involved in star-field attitude sensing are:

1) Star-Field Detection

2) Star-Field Identification

3) Attitude and/or Position Computation

Most star-field sensing instruments, either proposed or developed, perform only the first function, supplying detection indications and coordinates to a computer. The slit scanning devices recently developed by Control Data Corporation operate in this mode. Similar schemes, based upon simultaneous electronic imaging of a wide field instead of slit scanning, have been proposed and mechanized. One version of such a proposal was presented by N. S. Potter as early as 1960[1]. Several other mechanizations have been proposed subsequently with the emphasis being upon minimizing the size and complexity of the detection element.

[1] Potter, Norman S.; "Orientation Sensing in Inertial Space by Celestial Pattern Recognition Techniques," presented at the ARS 15th Annual Meeting, Shoreham Hotel, Washingtion, D. C., Dec. 1960 (paper No. 1482-60)

A few star-field sensors have been proposed or mechanized that are entirely autonomous in that they utilize some form of internal logic or stored analog data to identify and track the star field without depending upon a digital computer. One device of this type is described in this section, the Star Field Tracker developed by General Electric Company.

Generally speaking, star-field sensors measure only the geometrical characteristics of the star field, i.e., the star "pattern." One rather unique scheme, however, measures the spectral characteristics of the star field. This sensor has been proposed and experimentally proven by the Federal Scientific Corporation under contract to the Air Force.[2,3] The system is based upon the U-B-V wide band stellar photometry classification and is described in some detail later in this section.

2.4.2 Star-Field Sensor Summary

A summary of the star-field sensors which were surveyed is presented in Table 2-VI, followed by a discussion of each sensor.

2.4.2.1 Stellar Attitude Measurement System

Manufacturer: Advanced Technology Division, American Standard

Contracting Agency: NASA-Goddard

Function: Provide inputs for spacecraft attitude computation

Sensor Description:

The Stellar Attitude Measurement System is a star-field scanner designed around the Positor, an electromagnetic deflection unit used in the ATD earth sensors. The system consists of a refractive optical system, a deflecting Positor for each scanning axis and a pair of orthogonal

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[2] "Star Identification by Optical Radiation Analysis," Report No. ALTDR 64-13, prepared for the AF Avionics Laboratory, Wright Patterson Air Force Base by Federal Scientific Corporation under subcontract to Polarad Electronics Corporation, January 1964.

[3] "Experimental Verification of Star Identification by Optical Radiation Analysis," Report No. AL-TDR-64-251, February, 1965 (source same as Reference 2).

Manufacturer and Model	Function	Development Status	Detector	Scanning Technique	Optical System Type	Optical F of View
Advanced Technology Division - American Standard	Attitude Sensing	Breadboard Evaluated for NASA/GSFC	Photomultiplier	Mechanical, Two Axis "Positor"	Nikon 50 mm f/1.4	20°
Belock Instruments Corp.	Attitude Sensing	In Development for NASA/ GSFC	Cadmium Sulfide Panel	Electro-luminescent Panel	Refractive	10° x 10°
Control Data Corporation WACR-I	Attitude Sensing	Experimental Model Developed for USAF Avionics Laboratory	Photomultiplier	Double Parallel Rotating Slits	100 mm f/1.1 Refractive	46° x 16° Annulus
Control Data Corporation WACR-II	Attitude Sensing	Experimental Model Developed for USAF Avionics Laboratory	Photomultiplier	Triple Rotating Slits	50 mm f/1.1 Refractive	38° x 16 Annulus
General Electric Co. Star Field Tracker	Attitude Sensing	R and D	ASCOP 541A Photomultiplier	Star Field Mask, FM Modulator	35 mm f/1.2 Refractive	46°
General Precision Star Angle Comparator	Attitude Sensing	Experimental Model Developed for USAF Avionics Laboratory	Photomultiplier	V-Slit Scanning Reticle	2.4" Diameter f/2.5 Catadioptric	8°
Minneapolis-Honeywell Passive Star Scanner	Spin Rate and Attitude Sensing Rocket Probes	Operational NASA/ LANGLEY Project SCANNER	ASCOP 543D Photomultiplier	Spacecraft Motion - V-Slit Reticle	2.75" Diameter f/5.0 Refractive	6° x 6°
ITT Federal Laboratories "STARPAT"	Star Pattern to Determine Rocket Probe Attitude	One Model Delivered to NASA/CSFC for Aerobee Rocket	ITT FW 143 Photomultiplier	Electronic Digital	2" Diameter f/5.0 Refractive	10° x 10
Federal Scientific Star Radiation Analyzer	Star Identification and Tracking	Experimental Model Field Tested	Photomultiplier	Filter Wheel and Chopping Reticle	2" Diameter Cassegranian, 30" Focal Length	70 Arc-Min (Search) 8 Arc-Min (Identification)

Table 2-VI. Summary of Star Field Sensors

ield	Scan Rate	Threshold Star Magnitude	Accuracy	Volume In^3	Weight lbs	Power Watts	Accessory Equipment	Source of Data
	0.25 Sec Frame Time	+5.0	0.1°	250	10	8	Computer and Electronic Spectrum Analyzer	Communication from ATD to TRW Systems Sept. 1966
	30 msec	+3.0	±2 min	200	9	10	Computer	Binary Digital Output for Each Star in Field of View
	1.5 Sec or 15 Sec Per Scan	+4.0	6 Sec to 60 Sec	NS	NS	NS	Computer	Report AFAL-TR-66-10 "Feasibility Investigation of a Wide Angle Celestial Reference for Space Navigation", Control Data Corp., April 1966
	1.5 Sec or 15 Sec Per Scan	+3.0	6 Sec to 60 Sec	NS	NS	NS	Computer	Report AFAL-TR-66-10 "Feasibility Investigation of a Wide Angle Celestial Reference for Space Navigation", Control Data Corp., April 1966
	NS	+3.0	30 Sec	NS	10	NS	None Initial Alignment within 10° Required	P. J. Klass, "Star Tracker Gives Attitude Data", Aviation Week, June 18, 1962 pp 52-53
	Hemispherical Scan in 60 Seconds	+2.0	60 Sec	NS	NS	NS	Computer	Report ABD-TDR-63-586 "Design, Fabrication and Test of a Feasibility-Model Relative Star Angle Comparator", S. Young Kearfott Div., General Precision, Inc., May, 1963
	0.75 RPS	+3.0	0.02°	846	20	1.4	Ground Computer Reduction	NASA/LANGLEY R 4 Q dated 15 Nov. 1963
	Variable	+3.0	Digital Resolution ~0.15°	101 (Head) 67.5 (Elect.)	7	5.5	Computer	ITT Data Sheet
	21 Sec (Worst Case Measurement Time)	~+2.5	15 Arc-Sec Per Axis at Rate of 1°/Sec	NS	NS	NS	Computer	USAF Reports Referenced in Section 2.4

NOTE: NS = NOT SPECIFIED

rectangular reticles. The star-field coordinate and intensity data is modulated at the Positor frequency, amplified and supplied to the sensor output. The star-field information is extracted from the output with an external laboratory spectrum analyzer. The star field is identified and attitude offset calculated by means of a computer program. An experimental model has been developed and tested.

2.4.2.2 Solid-State Star-Field Sensor

Manufacturer: Belock Instrument Corp.

Contracting Agency: NASA-Goddard

Function: To obtain data defining spacecraft attitude with respect to celestial coordinates

Sensor Description:

This solid-state device utilizes a thin layer of cadmium-sulfide photoconductor as a sensor of stellar radiation. The star-field pattern is focused on one side of the photoconductor. The opposite side of the photoconductor is scanned on two axes by either a line or spot of illumination produced by excitation of a phosphor. The phosphor is excited by an electrically switched matrix. When the illumination from the phosphor is coincident with the element of the photoconductor upon which a star image is focused, complete photoconductivity through the cadmium sulfide is established and the resultant current flow constitutes the star signal. Sequential signals for each star, in binary digital form, are produced.

2.4.2.3 Wide-Angle Celestial Reference - I

Manufacturer: Control Data Corporation

Contracting Agency: USAF - Avionics Laboratory, Wright Field

Function: Experimental sensor to demonstrate the star field scanning and identification principle for space navigation

Sensor Description:

The Wide-Angle Celestial Reference, Version I, consists of an objective lens, motor-driven two-slit reticle with angle encoder, fiber optic coupling to a field lens photomultiplier, and amplification and threshold detection electronics. Polar coordinates, relative to the optical axis, are measured for two or more stars by means of the scanning reticle and encoder. This data, together with the brightness of each star in the field, is input through a digital interface to a general purpose computer. The computer is programmed to identify the star field. An experimental model has been developed and proven in field tests.

2.4.2.4 Wide-Angle Celestial Reference – II

Manufacturer: Control Data Corporation

Contracting Agency: USAF - Avionics Laboratory, Wright Field

Function: Experimental sensor to demonstrate the star field scanning and identification principle for space navigation

Sensor Description:

The WACR-II is basically identical with WACR-1 with some modifications made in the interests of overall simplicity. The slit configuration for the WACR-II involves three openings instead of the two used in WACR-I, the objective lens for WACR-II has been reduced in diameter and the fiber optics and field lens deleted. An experimental model has been developed and proven in field tests.

2.4.2.5 Star-Field Tracker

Manufacturer: General Electric Company

Contracting Agency: General Electric Research and Development Program

Function: Experimental device to prove the principle of star-field tracking by means of an analog correlation technique

Sensor Description:

The General Electric star-field tracker uses a photoetched mask with holes corresponding to the location of stars in the field to be tracked. The mask is placed behind the focal plane of the objective lens so that the defocused star images have a finite size. A collecting lens follows the mask to focus the total transmitted star-field energy upon a photo-multiplier detector. The system has a peak-output signal only when the star field exactly matches the mask holes and provides a proportional signal $\pm 10^{\circ}$ from this exact correlation. The radiant star signal is modulated by an FM chopper.

2.4.2.6 Star Angle Comparator

Manufacturer: General Precision Inc., Kearfott Division

Contracting Agency: USAF - Avionics Laboratory

Function: Breadboard instrument to demonstrate the feasibility of star-field identification by measurement of star-field geometry.

Sensor Description:

The Kearfott Star Angle Comparator utilizes a mechanically deflected narrow field telescope with a "V" slit reticle to scan a wide field-of-view. The sensing system, consisting of a photomultiplier detection circuitry and processing electronics detects and records the coordinates of the stars encountered. A computer program is then used to positively identify the detected stars using only their relative separation angles. The wide field (up to hemispherical) scanning process requires considerable time but reduces the length of the star list. An experimental model has been developed, tested, and delivered to the Air Force Avionics Laboratory.

2.4.2.7 Passively Scanned Star Telescope

Manufacturer: Minneapolis-Honeywell

Contracting Agency: NASA - Langley

Function: The PSST is used to determine the spin rate and attitude in space of spin-stabilized rocket probes.

Sensor Description:

The PSST is a fixed field device, consisting of an objective lens, slip reticle, photomultiplier detector and detection electronics. The sensor axis is pointed normal to the rocket spin axis and detects stars as they pass across the reticle in the objective focal plane. The time history of the star detections is telemetered to ground for digital computer computation of the attitude as a function of time. This equipment is currently in use in Project Scanner, in which high altitude rocket probes are being tested at Wallops Island, Va.

2.4.2.8 "STARPAT"

Manufacturer: ITT Federal Laboratories

Contracting Agency: NASA-Goddard Space Flight Center

Function: The "STARPAT" is a simple digital star-field scanner which digitally encodes the incremental elements of the field-of-view electronically and senses the presence of stars for determination of the attitude of a space probe.

Sensor Description:

The "STARPAT" consists of an optical system, an ITT deflectable photomultiplier, deflection circuitry, high voltage power supply, and digital logic circuitry. The sensor is self-contained and has no moving parts. The field-of-view is $10^{\circ} \times 10^{\circ}$ and the electronic scan is capable of dividing this field into a matrix of 64 x 64 elements. One unit has been delivered to NASA GSFC for use on the Aerobee Rocket Probe. The star-field information is processed externally to derive the desired spatial attitude data.

2.4.2.9 Star Optical Radiation Analyzer

Manufacturer: Federal Scientific Corporation

Contracting Agency: USAF - Avionics Laboratory

Function: The sensor has been designed to detect, identify and track upon a single star in a narrow field-of-view.

Sensor Description:

The Optical Radiation Analyzer is a gimbaled tracking radiometer which detects and locks onto a star with stability sufficient to make 1% intensity measurements in three spectral bands. These intensity measurements, made in the V, B, V stellar photometry bands, are processed in a computer to identify the tracked star. Although this sensor is properly a star tracker rather than a star-field sensor, it is included here since it also is used in the star identification mode. An experimental model has been field tested and evaluated.

2.5 PLANET SENSORS

2.5.1 General Discussion

The instruments described in this section have been designed primarily for sensing planets other than the earth. To date, effort has been concentrated on sensors to be used in Mars approach and flyby, and Lunar approach and orbit. An exception is the fine alignment subsystem being developed for the IR-OAO for scanning the planets Venus, Mars, and Jupiter, from earth orbit.

The Barnes planet sensor, operational on the Mariner program, utilizes a bolometer detector and mechanical scanning. Barnes and Northrop-Nortronics have developed similar equipment for Lunar horizon sensing using electronically-interrogated thermopile arrays.

The Lockheed Missile and Space Co. planet sensor for the IR-OAO utilizes an image dissector with electronic scanning.

To date, no equipment has been developed for approach guidance to the planet Jupiter.

2.5.2 Planet Sensor Summary

Table 2-VII contains a summary of the planet sensor survey, followed by a brief description of each instrument.

2.5.2.1 Mariner Planet Seeker

Manufacturer: Barnes Engineering

Contracting Agency: NASA-Jet Propulsion Laboratory

Function: The sensor was used to measure the attitude of the Mariner spacecraft relative to the target planet for direction of the scientific instrument package.

Sensor Description:

The Mariner planet seeker is a single-sensor head system using two contra-rotating germanium prisms to sweep a 70° cone. The four lobe rosette scan pattern inscribed within the cone is used for planet acquisition and tracking. The objective optic is a germanium refractor, and the detecting element is a germanium-immersed thermistor bolometer. The Mariner seeker is capable of operating over a planetary subtense range of 2° to 55°.

Manufacturer and Model	Development Status	Sensor Type	System Configuration	Output Characteristics	Detector	Spectral Range	Optical System
Barnes Mariner Planet Seeker	Operational (NASA/JPL Mariner	Disc Scanner (Rosette Pattern)	Single Sensor Package	Analog Pitch and Roll	Ge-Immersed Bolometer	1.8-20.0 Microns	Refractive Objective an Two Scannin Prisms
Barnes Lunar and Planetary Horizon Scanner	Experimental Model Built for NASA/JPL	Electronic Scan	4 Optical Heads and One Electronic Package	Separate Horizon Crossing Pulses for Each Head	Thermopile Array	14-50 Microns	Uncorrected Schmidt
Lockheed Planet Scanner	Breadboard being Developed for NASA/AMES IR-OAO	Periphery Scanner	Single Scanner	Two Axis Analog Error Signals	CBS Reconatron	S-20	IR/OAO Primary Op 38 in. Diam 1500-2000 i f.l
Northrop Nortronics Lunar Horizon Tracker	Experimental Model Delivered to NASA/MSFC	Electronic Scan	Single Package	Two Axis Digital (Coarse) Analog (Fine)	Thermopile Array	0.5-35 Microns	4 Bouwers Optical Systems

Table 2-VII. Summary of Planet Sensors

	Field of View			Quoted Instrument Accuracy	Tracking Time Constant or Noise	Planet Subtense Range	Quoted Reliability for One Year Operation	Volume In^3	Weight lbs	Power Watts	Data Source
	Acq.	Pkg.	Inst.								
	70° Cone	NS	1/2° x 1/2°	0.025°	Dual Mode: .05 cps and 700 cps	2° to 55°	0.50	315	12	3 Av.	IRIA "State of the Art Report on Infrared Horizon Sensors" No. 2389-80-7 April 1965
	10 x 81° (Per Head)	NA	0.9° x 10° (per detector element)	±0.5°	300 m sec	22° to 170°	NS	275 (Per Head)	25 (Total System)	13.5	"Proc. of the First Symposium on IR Sensors for Spacecraft Guidance and Control" Barnes Engr. May 1965
c ter	NS	4.5 Min Cone	NS	±1.6 Sec	10 cps	10 to 65 arc sec Venus, Mars, and Jupiter	NS	980	5 (Design Goal)	15 max.	NASA AMES RFP A-8516 May 22, 1964
	2π Sterad.	NS	3° x 90° Per Head	0.1°	14 m sec.	15° to 140°	NS	340	13	10	Mfr. Brochure

NOTE: NS = NOT SPECIFIED
NA = NOT APPLICABLE

2.5.2.2 Lunar and Planetary Horizon Scanner

Manufacturer: Barnes Engineering

Contracting Agency: NASA-Jet Propulsion Laboratory

Function: The LPHS is a prototype all solid-state horizon scanner. It has been developed as a possible universal horizon scanner.

Sensor Description:

The LPHS consists of four sensor heads, each having a 90° field-of-view in one axis provided by an uncorrected Schmidt optical system. The field-of-view of each head is divided into 90 sectors by an array of thermopile detectors. An electronic scan is used to sequentially interrogate the detector arrays, locating the image of the planet-space interface occurring on one element. The disadvantage of this scanning concept is the requirement for precise matching of the responsivity of the many detectors.

2.5.2.3 IR-OAO Planet Scanner

Manufacturer: Lockheed Missiles and Space Co.

Contracting Agency: NASA-Ames

Function: This sensor, in its final flight version will be used to generate attitude stabilization and scanning signals for the IR-OAO spacecraft while the planets Venus, Mars, and Jupiter are being observed with the primary telescope.

Sensor Description:

The Lockheed planet scanner has been developed in the breadboard stage. A commercial Catadioptric telescope and relay lens simulates the extremely long focal length of the IR-OAO telescope, into which the scanner will be incorporated. The planet is scanned on the faceplate of a CBS "Reconotron" image dissector and its deviation from the telescope optical axis is measured with high precision. In the present form, this device is not suitable for use in a strapdown guidance system, but the basic concept is applicable.

2.5.2.4 Lunar Horizon Sensor

Manufacturer: Northrop Nortronics

Contracting Agency: NASA-Marshall Space Flight Center

Function: The lunar horizon sensor detects and measures the position of the lunar horizon from a lunar orbiting spacecraft.

Sensor Description:

The Nortronics lunar horizon sensor consists of four optical heads, each using a wide-angle Bowers optical system and a linear array of thermopile detectors subtending a $3^{\circ} \times 90^{\circ}$ field-of-view. The heads are mutually orthogonal and are normal to the nominal vertical. Each thermopile element in the array subtends $3^{\circ} \times 3^{\circ}$ and consists of a detector pair shaped to permit linear interpolation on the pair. The arrays are interrogated electronically to determine the position of the horizon-space image. The signal from each head assembly consists of an indication of which elementary pair observes the horizon image and the location of the horizon within the element. The device has been fabricated and tested in the experimental stage.

Part II: SURVEY OF STRAPDOWN INERTIAL INSTRUMENTS AND SENSOR ASSEMBLIES

VOLUME III, PART II

1. INTRODUCTION

In this part of Volume III, TRW Systems presents the results of a survey of the state-of-the-art in strapdown inertial guidance systems and the instruments from which a conceptually designed system might be constructed. The information gathered during the equipment survey provides a basis for characterizing a system, or systems, optimal to each of the four missions under study. (Techniques relevant to the design of these systems are described in Volume II.) An example is the refinement of the electrical-torquing loops pertinent to strapdown systems. Work on pulse-torquing techniques, recently completed, will provide more accurate rebalance loops for inertial instruments. (See Volume II.)

Section 4 of Volume II comprehensively treats error models for conventional sensors used in a strapdown configuration and specifies detailed error models for two strapdown sensor package configurations. Additionally, Volume II provides the basic error analysis philosophy for a strapdown guidance system.

During the course of the study, the error models synthesized for each selected instrument configuration were applied to mission trajectory sensitivities for accuracy evaluation. These results together with the size, weight, power, cost, and reliability have been traded-off to permit recommending a system implementation for each mission.

In this investigation of strapdown hardware, first, the field of inertial instruments was surveyed for devices suitable for strapdown applications compatible with the missions under study. Second, the field of inertial sensor assemblies was surveyed for existing or conceptual designs applicable to the missions under study. Table 1-I shows the three major categories of concern in the makeup of a strapdown inertial sensor assembly. The greatest developmental emphasis is placed on an instrument configuration consisting of three single-degree-of-freedom gyros and three accelerometers. The systems realizable now or in the near future are:

1) Systems made up of instruments selected from the gyros and accelerometers listed in the tabulations showing parameters of interest in Section 4.

2) Systems consisting of electrostatically suspended gyros (ESG) and electrostatically suspended gyro accelerometers (ESG/A).

Table 1-I. Sensor Assembly Configuration Selection

Strapdown Inertial Sensor Assembly

Angular Rate Sensors	Acceleration Sensors	Electrical Torquing Loops
Single-Degree-Of-Freedom RI Gyros	Pendulous	Pulse Torquing Fixed Pulse Variable Pulse
Two-Degree-Of-Freedom Free-Rotor Gyros G10B	Integrating PIGA VM4A	Analog Torquing
Single-Axis Platform	Electrostatically Suspended Gyro Accelerometer	
Electrostatically Suspended Gyro		

2. SUMMARY AND CONCLUSIONS

The equipment survey and the initial evaluation of the trends in the field of strapdown systems indicate the following:

Several selected gyros and accelerometers described in Section 3 appear promising for use in strapdown systems. Effort to develop and to improve their performance parameters is underway.

ESG and ESG/A are in the advanced hardware development stage. Systems employing ESG's and ESG/A's may provide, in the future, the next increment in the advancement of the state-of-the-art of guidance systems, especially as applied to missions of long duration.

It should be recognized, however, that the ESG/ESG/A reliability, in its present and near future state, falls far short of the "conceptual" capability often associated with a device with but one moving part suspended in a true free-rotor configuration. It is as true in general, as it is in particular relative to the ESG/ESG/A, that inertial sensor electromechanical complexity is no longer the pacing MTBF criterion. Through evolutionary strides in the precision instrument field, we have arrived at the point where the complexity and reliability of the electronics bound the MTBF parameter. As an example, the ESG MTBF is but 40,000 hr and is solely a function of sensor electronics. Tungsten filaments discounted, the predicted MTBF does not exceed 60,000 hr. Certainly, the state-of-the-art SDF gyro is not outclassed by the ESG in this regard.

While it is no panacea for extended-time space mission reliability, the ESG/ESG/A class of sensors does offer a useful option in mechanizing guidance systems wherein the tradeoffs between mission accuracy and reliability are a function of the need for updating and the techniques employed to update. The fully calibrated ESG/ESG/A, with its extremely low long-term drift, should be considered seriously when optimizing a system for specific missions.

Regardless of the improvements in inertial sensor reliability with the advent of microminiaturized electronics, gas-spin bearings, electromagnetic and electrostatic gimbal and free-rotor suspension schemes, and the overall engineering maturity of the precision inertial instrument industry of today, the requirements for ultra-high strapdown systems reliability in future missions will probably be satisfied through the use of redundancy techniques, i.e., redundant arrays of proven sensors with failure detection and reorganization capability. Numerous mechanizations are possible involving both deterministic and statistical concepts to establish majority voting logic.

Pulse-torquing techniques are replacing the analog method in the gyro and accelerometer rebalance loops, as a means of increasing accuracy. An in-depth discussion of pulse torquing is given in Appendix A of Volume II.

Specific information on the predicted performance of strapdown systems under development by other companies is unavailable because of the proprietary nature of these projects.

3. INSTRUMENT SURVEY

The inertial instrument survey was conducted to establish a list of components which by virtue of their capabilities and performances, present or projected, could be applied in a strapdown mode to the missions under study. The limits of performance have been tentatively established subject to updating as more detailed information becomes available from the manufacturers.

3.1 CANDIDATE STRAPDOWN GYROS

Analysis on other strapdown programs conducted at TRW has shown that three basic categories of gyros should be explored for strapdown systems application. These are the SDF, TDF, and the ESG. Because of the emphasis placed on SDF gyros by companies developing strapdown systems, this class was given much attention. The results of this phase of the survey, including comments concerning their application, are given in Table 3-I. The particulars concerning the TDF and the ESG are treated separately as classes of gyros rather than in a tabular comparison of manufacturers.

3.1.1 Selected SDF Gyros

Based on the information available to date, three gyros were chosen as outstanding candidates. These gyros were selected on the basis of size, weight, power, torquing rate (15 deg/sec minimum) as well as performance characteristics. The performance of these selected gyros is given in Table 3-II. The definition of the performance parameters is as follows:

1) Fixed Drift, Maximum

   The fixed (g-insensitive) gyro drift is measured during the calibration period, and an appropriate compensation term is established for computer compensation. The maximum permissible value of fixed drift determines the range of compensation required for the scaling of the equations in the computer. Note that this term is not an error source.

		Physical Characteristics		Construc		
Gyro Model No.	Gyro Manufacturer	Size (in.)	Weight (lb)	Type	Rotor Bearing Type	Gimbal Suspension
GI-MI	Nortronics	2(dia) x 3	0. 9	SDF*	Gas	Flotation + electro-static
GG327	Honeywell	2. 3(dia) x 4. 7	1. 3	SDF	Gas	Flotation + pivot and jewel
G70 2590	Kearfott	2. 06(dia) x 3. 1	1. 0	SDF	Gas	Flotation + pivot and jewel
GG87	Honeywell	2(dia) x 4	1. 1	SDF	Ball	Flotation + pivot and jewel
25IRIG Mod II	MIT (Design)	2. 42(dia) x 3. 7	1. 5	SDF	Ball	Flotation + magnetic
SYG 1000	Sperry	1. 83(dia) x 2. 79	0. 72	SDF	Ball	Flotation + pivot and jewel
2566	Kearfott	2. 6(dia) x 2. 8	1. 2	SDF	Ball	Flotation + pivot and jewel
AB-5	NASA/Bendix	3(dia) x 2	2. 0	SDF	Ball	Hydrostatic gas bearing
GI-K7	Nortronics	1. 6(dia) x 2. 5	0. 5	SDF	Ball	Flotation + pivot and jewel
2542 2543	Kearfott	1. 4(dia) x 2. 3	0. 39	SDF	Ball	Flotation + pivot and jewel
2564	Kearfott	2. 0(dia) x 2. 8	0. 9	SDF	Ball	Flotation + pivot and jewel
GG 334A	Honeywell	2. 5(dia) x 4. 7	1. 4	SDF	Gas	Flotation + pivot and jewel
RI 1139B	Norden	2. 7(dia) x 3. 6	1. 5	SDF	Ball	Flotation + pivot and jewel

*Single-degree-of-freedom (SDF)

tion Characteristics										
					Motor			Pickoff		
Angular Momentum (CGS)	Gimbal Freedom	Gyro Gain	Pickoff (Type)	Torquer (Type)	Volts	Freq (cps)	Power (w)	Volts	Freq	Vo
2. 4 x 10^5	±3 deg max	15	Capacitive	Perm magnet	26	3ϕ 1600	6. 5 Start 3 Run	5 to 100	2. 5 to 10 kc	
2. 0 x 10^5	Up to ±4 deg	0. 4	Microsyn	Perm magnet	50 Start 26 Run	3ϕ 800	4. 5	5	32 kc	28 115
3. 5 x 10^5	±2 deg	8. 8	Microsyn	Perm magnet	26	3ϕ 800	15 Start 8 Run	6 to 9	4 kc to 20 kc	28 115
1 x 10^5	±4 deg	0. 3	Dualsyn moving iron	Perm magnet	26	3ϕ 400	3		400 cps	0-3 AC,
4. 3 x 10^5	±1 deg	1. 0	Microsyn	Microsyn	28	2ϕ 800	8 Start 4. 5 Run	28 8 25	800 3. 2 kc 20 kc	115
1 x 10^5	±4 deg	0. 5 to 14	Moving coil	Perm magnet	26	3ϕ 400	4. 2 Start 3. 0 Run		400~ to 10 kc	115 30
. 27 x 10^5	±2 deg	1. 0	Microsyn	Perm magnet	26	3ϕ 400	3. 7 Start 2 Run	15	2 to 20 kc	
. 6 x 10^6	±3 deg	Very Large	Moving coil	Moving coil	26	3ϕ 400	8	10	4. 8 kc	No Hea
. 9 x 10^5	±3 deg	1. 0	Microsyn	Perm magnet	26	3ϕ 400	3. 25 Start 3. 0 Run	2	1. 6 to 8 kc	
. 3 x 10^5 . 6 x 10^5	±2 deg	1. 0	U-bridge air core	Perm magnet	26	3ϕ 800	2	8	800~	28 28
. 27 x 10^5	±3 deg	0. 3 to 250	U-bridge air core	Perm magnet	26	3ϕ 400	4. 5 Start 3. 5 Run	13	0. 4 kc to 20 kc	28
x 10^5	±3 deg	0. 5	Microsyn	Perm magnet	26	2ϕ 800	7. 0 Start 3. 5 Run	5	32 kc	28
. 5 x 10^5	±1 deg	0. 25	Moving coil	Perm magnet	26	3ϕ 400	6. 0 Start 3. 0 Run	8. 3	10 kc	

Excitations				Torquer	
	Control Heater				
ts	Freq	Max Power (w)	Other	Range	60 Da Scale Fa Stability (
			PSF 24 mv/mr TSF 5. 0 deg/hr/ma	0-1000 deg/hr (0. 278 deg/sec) 0-6300 deg/hr available (1. 75 deg/sec)	100 pp
	AC/DC AC/DC	10 100	PSF 20 mv/mr TSF 10 deg/hr/ma and 1200 deg/hr/ma	0-180, 000 deg/hr 50 deg/sec 0-1000 deg/hr (0. 278 deg/sec)	100 pp
	AC/DC AC/DC	20 80	PSF 0. 45 v/deg to 0. 62 v/deg TSF 1 to 50 deg/hr/ma	0-11, 000 deg/hr 30 sec	
DC		55	TSF 800 deg/hr/ma	0-60 deg/sec 0-30, 000 deg/hr	300 pp
		40	Magnetic Suspension 3. 5 v 4800 ~0. 1 w	25 deg/sec max	
	AC/DC AC/DC	100 30	PSF 5. 7 mv/mr TSF 134 deg/hr/ma	0-28, 000 deg/hr cont 67, 000 deg/hr (8 sec max)	100 pp
			PSF 300 mv/deg TSF 900 deg/hr/ma	25 deg/sec cont	100 pp
er	No Heater	No Heater		0-360 deg/hr	
		250 90	PSF 16. 7 mv/mr TSF 600 deg/hr/ma	17 deg/sec cont 60, 000 deg/hr	100 pp
	AC/DC AC/DC	40 20	PSF 4. 9 mv/mr TSF 272 deg/hr/ma TSF 136 deg/hr/ma	20 deg/sec cont 30 int 10 deg/sec cont 20 int	
	AC/DC	30	PSF 8. 75 mv/mr at 800 cps exc TSF 1 to 220 deg/hr/ma	20 deg/sec cont	100 pp
	AC/DC	10	PSF 20 mv/mr TSF 1200 deg/hr/ma	114 deg/sec cont	100 pp
	AC/DC	10	PSF TSF 450 deg/hr/ma	28 deg/sec cont	100 pp

		Performance					
		Drift deg/hr					
r		g-insensitive		g-sensitive		g^2-sensitive	Ran
:tor nax)	Linearity	Magnitude	60-day Stability (1σ)	Magnitude	60-day Stability (1σ)	Magnitude	OA Ve
n	<0.01%	<0.2	<0.05	<0.2	<0.05	<0.02	0.001 to 0.005
n	0.01% (to 10 deg/sec) 0.01%	0.5 to 0.1	0.01	0.1	0.01	0.04	0.005
	0.01%	0.1	0.03	0.15	0.05	0.03	0.001
n	0.05%	0.4	0.2	0.7	0.3	0.02	0.02
	0.05%	(Refer to Volume IV of this report for these data.)					
n	0.5% to ±3 deg	0.7	0.1	0.2	0.2	0.01	0.005
n		0.7	0.2	0.8	0.2	0.02	0.005
		(Refer to Volume IV of this report for these data.)					
n	0.05%	0.5	0.3	1.0	0.3	0.06	0.05
	0.05%	1.0 0.5	0.5 0.3	0.6 0.3	0.5 0.2	0.015 0.015	0.05 0.025
n	0.2%	1.0	0.17	0.1	0.33	0.015	0.05
n	0.01% (to 20 deg/sec) 0.05% (to 114 deg/sec)	0.1	0.15	0.1	0.2	0.04	0.005
n		0.3	0.06	0.3	0.33	0.01	0.02

Table 3-I. Strapdown, Single-Degree-of-Freedom Rate Integrating Gyros

dom Drift		Cost Prod, $ (100 lots)	Usage	Other: Remarks
't	OA Horiz			
	0.004	15K	Company development	Very early development, ceramic construction, torquing rate is too low for strapdown usage.
	0.007	15K		Superseded by GG 334
	0.001	15K		Torquing rate is too low for strapdown usage.
	0.04	7.5K	Early strap-down systems	Superseded by GG 334
		16K	Polaris IG system	A modified version with a permanent magnet torquer is used in the MIT prototype system, constant temperature control (even during transportation) is required. (Designated 25 PIRIG)
	0.007	10K		Torquing rate is too low for strapdown usage. Modified version with air bearing and high torquer rate (~100 deg/sec) designated SYG1440.
		13K	Proposed for LM abort guidance system	High torque modification of 2565 Alpha Series Gyro. However, this is a conceptual design and no hardware development is planned.
		10K	Saturn IG system	From "A General Description of the ST 124-M Inertial Platform System NASA TN D2983," September 1965 H. E. Thomason.
	0.05	4.5K	Vela, Lorv, Flip, Arpat	Three units modified for 30 deg/sec continuous torquing are being fabricated for TRW Systems.
	0.07 0.04	3.5K 5K	Company development	2543 torquing rate is too low for strapdown usage. 2542 is "similar in price to Nortronics GI-K7, but has poorer drift stability and no production history."
	0.05	8.5K	Rags, Tars, Mariner C	This unit is similar in price and performance to the Honeywell GG334A, but it was designed for platform usage and does not have the high torquing rate capability of the 334A.
	0.005	9K	Honeywell strapdown system	This unit was specifically designed for strapdown usage.
	0.007	15K	LM abort system	The high cost of this unit and its low torquing rate in comparison to the 334A make it a less attractive unit.

2) <u>Fixed-Drift Stability</u>

The change in fixed drift for a specified period after calibration is the fixed-drift stability and is the term that actually contributes to the system inaccuracy.

3) <u>Mass Unbalance, Maximum</u>

The mass unbalance (spin-reference axis, MUSRA, and input axis, MUIA) is also measured during calibration for computer compensation. The maximum permissible value of mass unbalance determines the range of computer compensation required and is not an error source.

4) <u>Mass Unbalance, Stability</u>

The change in MUSRA/MUIA for a specified period after calibration is the MUSRA/MUIA stability.

5) <u>Torquer Scale-Factor Stability</u>

The change in torquer scale factor for a specified period after calibration is the torquer scale-factor stability.

Table 3-II. Selected SDF Gyros

Performance Parameters	Kearfott 2564/2565	Nortronics Mod GI-K7	Honeywell GG 334A
Bias:			
Stability [1]	0.5 deg/hr (max)	1.0 deg/hr (max)	0.45 deg/hr (max)
Temperature Sensitivity	0.01 deg/hr °F	0.02 deg/hr °F	————
Mass Unbalance:			
Stability [1]	1.0 deg/hr-g (max)	1.0 deg/hr-g (max)	0.6 deg/hr-g (max)
Temperature Sensitivity	0.05 deg/hr-g °F	0.08 deg/hr-g °F	————
Scale Factor:			
Stability [1]	————	100 ppm (max)	100 ppm (max)
Temperature Sensitivity	40 ppm/°F	100 ppm/°F	————

Notes:
1. Stability is for 60 days with shutdown, unless otherwise noted.
2. Source: Performance data provided by manufacturer
3. Both Kearfott and Nortronics gyros are ball-bearing designs, while the Honeywell is designed with a hydrodynamic gas-spin bearing.

3.1.2 Two-Degree-of-Freedom (TDF) Free-Rotor Gyro

The TDF hydrodynamic gas-bearing gyro is a familiar component of inertial guidance systems for ballistic missiles; a large number of them are used on the Minuteman program. If this considerable reservoir of manufacturing and operational experience could be extended to a design more applicable to strapdown applications, an inertial system of proven performance and reliability could result. To this end, a miniaturized version of the G6 free-rotor gyro with an increased torquing-rate capability is under development. Identified as the G10B, its characteristics are discussed in detail below. Among these characteristics, the most significant from the point of view of performance are long-term, fixed-drift stability and warmup time. The ability to limit fixed drift for periods up to one year not only permits long mission times but also simplifies prelaunch operations by allowing instrument calibration, off the vehicle, a considerable time prior to launch with the assurance that this calibration will be maintained. The freedom from flotation problems permits a faster warmup time than would otherwise be possible for those applications in which this is significant. An additional characteristic, which might be significant for specific applications, is the capability of operating the gyro at considerably reduced spin-motor speed with a consequent reduction in power dissipation but with some improvement in performance†. The physical characteristics of the G10B are summarized in Table 3-III.

The accuracy of the vehicle rate and position depends upon the error sources of the gyro. The error model for the G10B gyro can be described

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†The following comments may be made with regard to performance at reduced wheel speed.

1) The random drift is improved by reducing the wheel speed (e.g. $0.02^{\circ}/hr$ to $0.005^{\circ}/hr$ rms for 240 rps to 60 rps).

2) The term does increase at reduced speed however during the launch phase the gyro can be run at 240 rps (for necessary bearing stiffness), and then reduced in orbit (near zero g environment). The g^2 term is expected to be stable to about $0.01^{\circ}/hr/g^2$ and might be compensated for if vehicle accelerated significantly while at low rotor speed.

3) The maximum torquing rate is 20 deg/sec at a spin motor speed of 60 rps.

Table 3-III. G10B Gyro Characteristics

Environmental Conditions	
Storage temperature range	-65 to +165°F
Ambient operating temperature range	-30 to +150°F
Acceleration	+6 g steady-state dc at 60 rps +22 g steady-state dc at 240 rps
Radiation-hardened	yes
Input Parameters	
Pickoff excitation	20 v ± 2% rms 19.2 kc ± 1% rms
Gyro motor normal	5 vrms nominal
Operating voltage	400 cps ± 0.05% 3-phase quasi-square wave. Voltage shall be regulated for rotor speed control.
Start voltage	30 vrms nominal 400 cps 3-phase quasi-square wave not to exceed 5 sec
Gyro motor power normal	0.7 w at 60 rps
Operating	6.0 w at 240 rps
Starting	50.0 w for 5 sec
Pulse-Torquing Signals	
Current levels:	
High rate Low rate	113.12 ma 20 ma
Torquer Scale Factor	
High rate Low rate	180.1 deg/hr/ma 32 deg/hr/ma
Maximum Torquing Rate	
High rate Low rate	20,480 deg/hr 640 deg/hr
Torquer linearity	0.05%
Torquer Power Requirements	
High rate Low rate	3.0 w 0.1 w
Output Signal	
Pickoff	19.2 kc suppressed carrier modulated. Normal near zero level signal with maximum 100 mv rms.
Pickoff Scale Factor	500 mv/mr $^{+20\%}_{-10\%}$ uncompensated
Gyro Parameters	
Rotor moment of inertia	945 $gm\text{-}cm^2$
Rotor angular momentum	3.6×10^5 $gm\text{-}cm^2/sec$ at 60 rps 1.44×10^6 $gm\text{-}cm^2/sec$ at 240 rps
Size	3.3 x 3.5 in.
Weight	2 lb

by the following equation: (See Reference 6).

$$\dot{\phi} = N_1 + CA_1 - DA_2 - BA_1 A_3 - EA_2 A_3$$

$$\dot{\phi} = N_2 + CA_2 + DA_1 - BA_2 A_3 + EA_1 A_3$$

where:

1, 2 refer to the input axis

3 refers to the spin axis

A_1, A_2, A_3 are the accelerations along the 1, 2, and 3 axes, respectively.

N = total nonacceleration dependent drift, deg/hr (bias)

C = drift due to rotor mass unbalance, deg/hr/g

D = drift due to structural compliance of rotor, deg/hr/g

B = drift due to gas-bearing incompressible effect, deg/hr/g^2

E = drift due to gas-bearing compressible effect, deg/hr/g^2

The bias or N term of the above expression includes the following error sources at a rotor speed of either 60 or 240 rps:

	60 rps	240 rps
Compensated bias	0.1 deg/hr rms	0.1 deg/hr rms
Random drift (9-min interval)	0.005 deg/hr rms	0.02 deg/hr rms
Bias stability* (60 days)	0.3 deg/hr rms	0.3 deg/hr rms
Temperature sensitivity	0.005 deg/hr/°F	0.005 deg/hr/°F

*Including shutdown to 0°F and storage at -65°F.

The values of the other error model terms are:

	60 rps	240 rps
C =	0.5 deg/hr/g	0.5 deg/hr/g
D =	0.4 deg/hr/g	0.4 deg/hr/g

$$B = 9.6 \text{ deg/hr/g}^2 \qquad 0.4 \text{ deg/hr/g}^2$$

$$E = 1.6 \text{ deg/hr/g}^2 \qquad 0.4 \text{ deg/hr/g}^2$$

3.1.2.1 Torquer Axis Alignment

The "A" and "B" torquer axis of the gyro can be aligned normal and/or parallel to the plane defined by the gyro alignment surfaces to within 1 arc min. The gyro-spin axis can be aligned normal to the "A" and "B" torquer axis to within 1 arc min.

Torquer scale-factor errors such as torquer eccentricity, unbalanced turns, etc., are fully compensated when pulse-width modulation techniques are employed.

Spring-rate coupling at the low torquing rate will be less than:

Direct coupling:	0.002 deg/hr/0.1 sec pickoff angle
Cross coupling:	0.02 deg/hr/0.1 sec pickoff angle†

3.1.2.2 Additional Notes

- Speed Control: The G10B gyro has an induction-type spin motor and cannot hold a constant speed. The speed is a function of applied voltage and gas-bearing loading. In order to keep the motor voltage sensitivity drift of the gyro low, a regulated spin-motor power supply is required. However, even with a regulated power supply, a drift will be apparent because of acceleration loading of the gas bearing. To overcome these problems, Autonetics has developed a gyro-rotor speed-control technique in which the rotor speed is held constant by varying the magnitude of the spin-power supply. This is done by modulating the pickoff signal as a function of rotor speed and then comparing the modulation frequency with a reference frequency and using the difference signal to control the magnitude of the spin-motor voltage. The complexity of the speed-control circuitry is about twice that of a regulated power supply, and one speed-control loop is needed for each gyro.

- Pulse Torquing: Autonetics has pulse-width-modulation torqued the gyro up to a fundamental frequency of 400 cps. The nutation frequency of the gyro is about 350 cps, and the

†It is the nature of the coefficient to be quite stable and easily measured. If the servo-loop transfer function of pickoff angle versus input-angular rate were known, then a major portion of this error could be compensated for.

sensitive bandwidth of the Autonetics pulse-torque rebalance loop with the gyro was tested on an oscillating rate table and was measured as 15-cps maximum.

- The price of the G10B gyro without any compensation for motor voltage sensitivity, temperature sensitivity, scale-factor tolerance, null restraint, or speed control modifications is about $6K.

3.1.3 Electrostatically Suspended Gyro (ESG)

The characterisitcs for the ESG discussed here are representative of a class of instruments rather than any specific model since these gyros are still in the latter stages of their development cycle rather than available as completed designs. As an example, Honeywell, a major supplier of these units, has completed approximately 50 of them to date. Because the ESG design is unique, many performance parameters are attributable to the design approach rather than design specifics for any one model, and it is possible to establish anticipated performance limits for any representative of the class provided that there is no major change in that design approach (as would result from a cryogenic ESG, for example).

The major performance parameter which leads to the consideration of ESG's is low drift. While it is certainly possible to achieve performance in the range below 0.001 deg/hr in a relatively benign (low-g) environment using an ESG, this is accomplished by identifying some 23 error terms and applying computer compensation for them. Since some of these terms have very long periods (on the order of 24 hr), such an operation implies not only extensive computer capability for compensation but an appreciable prelaunch calibration period. Honeywell predicts performance in the 0.005 deg/hr range, using an abbreviated 7-term compensation scheme, and performance of 0.05 deg/hr using a 3-term calibration scheme. Since this latter performance is within the capability of many gyros of conventional design, the application of ESG's to requirements at this level does not appear warranted on the basis of drift alone.

Two other aspects of ESG performance justify its selection for particular applications: long life and low power. The gyro-rotor suspension involves no physical contact and, therefore, no wear. The gyro life is more dependent upon the number of starts and stops than continuous operating time. Once the rotor is up to speed, spin-motor power is removed and the gyro coasts in normal operation, with a consequent reduction in power dissipation. This reduces the problem of dissipating the heat generated by the gyro in a space application. Therefore, the choice of an ESG for a particular application may be dictated by the need for low power or long life. It is pertinent to note that while the ESG has a very long wearout life its MTBF, primarily a function of electronics reliability, is currently estimated at 40,000 hr.

3.1.3.1 Instrument Limitations and Characteristics

The pacing components limiting the ESG MTBF are the ion-getter vacuum pump filament, and the incandescent lamp tungsten filament used in the optical pickoff system. The other major area affecting reliability is the suspension system circuitry which is composed of numerous transistors, diodes, and other solid-state elements. The quoted MTBF is based on the more complex circuit required for high-g (boost survival) capability. The contribution of the suspension system to the MTBF value is significantly reduced when the low-g suspension circuitry (for space flight applications) is employed. It is estimated that a 3 to 1 reduction in complexity (parts count) results by using the low-g circuitry.

In addition to the complexity of its readout as a possible deterrent to the application of an ESG is its limited ability to be torqued. The torquing limitation arises because the ESG rotor operates, of necessity, in a hard vacuum. The application of energy to torque raises its temperature, with a consequent degradation in performance, and the hard vacuum prevents easy dissipation of this heat. Just as the torquing limitations are imposed by the inability to put energy into the rotor, readout techniques are limited by the inability to take energy out of it. Current practice is to use a photoelectric readout of a pattern imprinted on the rotor. This readout does not produce a direct measurement of case angle, but the case angle may be calculated from the readout signals. However, this calculation imposes a severe requirement on computer speed.

The limiting factor in obtaining readout accuracy is the resolution of the engraved pattern on the sphere (pattern "edge" uncertainty). The use of the straight line (great circle) instead of an explicit direction-cosine readout pattern gives superior accuracy. When coupled with a 20-bit/sec sample rate, the straight line pattern will yield a 20-arc sec 3-σ accuracy.[†] Work is in progress on a multiple-straight-line pattern, which is estimated to yield a 5-arc sec 3-σ accuracy. However, the multiple-line pickoff increases the computer interface complexity (signal processing between gyro and computer). In the all-attitude angle pickoff for strapdown application, three optical circuits will be required to yield a pickoff angular freedom of ±50 deg from the equator; because the region within ±40 deg from the rotor poles to pattern is an optical dead zone. Additional operating characteristics pertinent to the ESG applications for space guidance system are covered in the following paragraphs:

- Temperature Sensitivity

  The ESG is relatively temperature-insensitive for small variations (±10°F) and may be operated over a very wide ambient temperature range with some degradation of performance. The error factor here is the change in vacuum gap due to thermal expansion coefficient differences between the alumina ceramic case and beryllium free rotor (approximately 2 to 1 mismatch).

- Power Requirements

  The stated strapdown ESG 5-w power requirement does not include temperature control power, but does include the necessary circuitry for all gyro axes (suspension, startup, pickoff).

- Reaction Time

  The reaction time is a function of reaching suspension system equilibrium and is decreased as g-level preloading is reduced. For the low-g suspension in a strapdown mechanization, a reaction time of 15 min is predicted.

- Shock Capability

  It is estimated that the ESG has a 200-g unsuspended survival capability. This implies a nonoperative condition for applications where the staging shock exceeds

[†]Verbal communication from Honeywell.

the 15-g operative limit and would require space startup. Honeywell states this is feasible since they claim demonstration of long-term calibration repeatability with shutdowns.

3.1.3.2 Performance Considerations

The ESG has been probably subjected to the most thorough and rigorous continuing theoretical study in the history of inertial sensors. These efforts have produced a series of revisions in design and fabrication goals, resulting in a steady improvement in performance. Since there is only one moving part in the gyro, an extremely high degree of predictability in performance is achieved.

Because of geometric and mechanical simplicity, there are only four significant torque effects in an ESG:

- Mass-Unbalance Torque

  This torque is common to all gyroscopic devices. In the ESG, mass-unbalance torque is caused by a noncoincidence of the geometric centroid of the rotor with its center of mass.

- Electric Torque

  The basic phenomena of electric torque is an interaction between the electric-field support force and an aspherical rotor.

- Gas-Drag Torque

  Gas molecules present between the rotor and electrode assembly can cause both rundown and precession torque.

- Magnetic Torque

  Since the ESG rotor is a spinning, conducting sphere, it is susceptible to a magnetic torque arising from the presence of a magnetic field.

Proper design has reduced most of these torques to negligible levels. Gas drag has been virtually eliminated by operating the gyro in a hard vacuum, while magnetic torque has been sufficiently reduced by an external magnetic shield and internal electronic shielding. Mass unbalance and electric torque are the only effects which require continued attention. These effects are discussed in the following paragraphs.

1) Mass-Unbalance Torque Mass unbalance may be conveniently divided into two components: axial (i.e., a component along the spin axis) and radial (normal to the spin axis). Radial mass unbalance torque is minimized in two ways: 1) by minimizing the quantity of radial mass unbalance in the rotor during fabrication; and 2) by designing the support electrodes so that their areas satisfy a fundamental design criterion. This electrode design actually constrains the radial mass-unbalance torque so that it is parallel to the spin axis, thus causing only a small speed change rather than a precession torque.

   Axial mass unbalance causes a precession torque in the presence of an inertial acceleration of the gyro and may be compensated for by two different methods. When the gyro operates in a strapdown mode, the external acceleration field may be measured with accelerometers and the resultant gyro precession calculated with the aid of a computer, as with floated gyros.

2) Electric Torque There is no electric torque present if the rotor is a perfect sphere. Present machining accuracies in rotor fabrication produce an almost "perfect" sphere. However, the rotor tends to bulge at the equator at operating spin speeds due to the centrifugal effect. In turn, the electric support force produces a couple on the ellipsoidal rotor. In a strapdown mode, this torque may be reduced to negligible levels by two methods: 1) computer compensation; or 2) by precise rotor centering and properly controlling electrode voltages. In gimbaled mode, the torque may also be nulled by: 1) controlling electrode tolerances, precise centering of the rotor, and good gimbal servo follow-up; 2) machining a prolate rotor so that it deforms into a sphere at operating speeds. The Navy ESG Monitor has used the second method with excellent results.

3) Gyro Torquing A related subject is the question of torquing this type of gyro during use. Rather than being "limited in its usefulness because it cannot be torqued," as has often been asserted, the ESG possibly offers a wider range of engineering choices for torquing than does the conventional gyro. On the one hand, the rotor being completely unconstrained in its angular degrees of freedom can function without any torquing, and since it is practical to allow for precession of the spin vector in inertial space by data processing of the readout signals rather than by

applying compensating torques, the gyro can be useful without torquing and, in fact, will tend to exhibit optimum drift characteristics in such a torque-free, minimum-disturbance condition. If, on the other hand, there is a mandatory reason for torquing the ESG, that can be done at moderate rates by exciting the same coils as are used for spinup, while keeping the torque normal to the spin vector. The limitations on torquing are rotor overheating and degradation in drift performance. As a typical example, an ESG spin vector locked to a level platform would have to be torqued continuously at approximately earth's rate. To precess a typical ESG at earth's rate requires 0.8 dyne-cm. The corresponding dissipation in the rotor is 1.2 mw which on a continuous basis will raise the rotor temperature 2°C assuming only radiative thermal transfer and 1/20th black body emissivity. The precision with which the torque vector must be controlled in this case is (drift rate)/(earth's rate) or 1/15,000 for 0.001 deg/hr drift rate, a precision requirement substantially the same as with conventional gyros and for the same reason. Torquing at high vehicle roll, pitch, or yaw rates would overheat the ESG rotor and degrade the drift performance because of torquer precision limitations, but the drift performance degradation likewise affects conventional gyros. Torquing the ESG at rates smaller than earth's rate, as for example to compensate for precession in inertial space, implies correspondingly milder rotor temperature rise and milder torquer precision requirements.

General Motors Defense Research Laboratories is another company actively involved in the development of electrostatically suspended gyros. Report No. TR 65-12, dated April 1965, gives the results of performance and environmental tests performed on their Mod 0 (August 1964) electric vacuum gyro (EVG). A later report to the Air Force, written jointly by AC Electronics Division and GM-DRC in early 1966 (a copy is available at NASA/ERC), provides information on the improvements made since the earlier report. One of these improvements would be development of a process to apply a readout pattern to the rotor which would materially reduce the errors caused by pattern torque. Another development resulted in an improved method of heterodyning the readout signal by modulating the light source.

Meaningful drift performance and sensitivity numbers from testing have been difficult to obtain. Numbers have been taken from various articles, reports, and telephone conversations with individuals from AC, Honeywell, JPL and University of Illinois. The validity of the numbers is not known nor is the procedure of the test. In most cases, there may even be some confusion in the definition of the term being numerically charterized; since ESG's by their nature cannot be described in the manner of conventional gyros — and not everyone agrees on how to specify them. The available information has been filtered and summarized in Table 3-IV. In addition, predictions of future performance (from the same sources) have been listed after subjective interpretation.

Table 3-IV. TRW Estimate of ESG Performance

Error Source	Space Environment Magnitude	
	Present	1971
Spin-axis drift - rms	0.003°/hr	0.0001°/hr
Uncompensated spin-axis drift	0.5°/hr	0.001°/hr
Short-term† stability spin-axis drift	0.003°/hr	0.001°/hr
Long-term†† stability spin-axis drift	0.001°/hr	0.001°/hr
Short-term stability g proportional coefficient	0.0005°/hr/g	
Long-term stability g proportional coefficient	0.001°/hr/g	
Short-term stability g^2 proportional coefficient	$0.0003^\circ/\mathrm{hr}/g^2$	
Long-term stability g^2 proportional coefficient	$0.002^\circ/\mathrm{hr}/g^2$	

† 24 hr
†† 1 month with shut-downs

There is such a wealth of information available on the ESG class of gyros in the form of error modeling, performance figures, descriptive material, circuits, test data, and reliability data, that to include it all would be beyond the scope of this report. Most of this information (some of which is classified) is already available in reports. These reports, if not in NASA files, are available from Honeywell, Inc., Minneapolis, Minnesota; and GM Defense Research Laboratories, Santa Barbara, California.

3.2 CANDIDATE STRAPDOWN ACCELEROMETERS

The conventional accelerometers surveyed as possible candidates for strapdown system application are listed in Table 3-V. These instruments can be categorized in two basic classes: 1) the force balance type, and 2) the integrating type. The choice depends on the mission requirements and many factors which will be considered as the study progresses.

On the near horizon, in the category of unconventional accelerometers, the Honeywell ESG/A (an electrostatically suspended three-axis accelerometer) appears to offer the greatest likelihood of space mission employment.

The first prototype of this instrument is being tested now, or will be soon. The fundamental concept is utilization of a digital suspension scheme with pulse-weight control from which the output difference count per unit of time, between opposing electrodes, is directly proportional to acceleration input along the electrode axis. Three-axis capability is, of course, achieved by the hexahedral electrode array used in the latest ESG/ESG/A designs.

Extensive hardware development has been performed to perfect the ESG/A, which is a direct outgrowth of the ESG concept. The predicted prototype ESG/A capability is given as:

Dynamic range:	1 part in 10^5
Threshold with a low (0.1 g) suspension preload:	$\sim 10^{-6}$ g

The problem of ESG/ESG/A operation during boost and orbital or coast phases of space missions is being approached from two directions: 1) the capability of nonoperative boost and staging survival with fast-reaction space startup and a low-suspension preload operating point; 2) the use of an adaptive control system to obtain a variable-suspension preload through the entire mission acceleration and shock profile. The adaptive technique under study is intended to provide an operable range of <0.1 to 30 g. However, the impact on system reliability due to the significantly increased electronics complexity of an adaptive control

Accel Model No.	Accelerometer Manufacturer	Size (in.)	Weight (lb)	Operating Temperature (°F)	Type	Proof Mass Suspension
16 PIGA Mod G	MIT (design)	2.42 x 3.9	1.8	125 ±0.1	PIGA	Flotation + magnetic
GG162 GG226	Honeywell	4.9 x 6.2	5.1		PIGA	Hydrostatic
KAIG	Kearfott	1.75 x 2.5	0.7	130 ±0.1	PIGA	Flotation + dual flexure
AB-3	NASA/Bendix	3.25 (dia) x 5	2.7	104 ±6	PIGA	Hydrostatic gas bearing
D4C	ARMA	1.9 x 2	0.75	0 to 70°C±1°C	VSA	String
MOD VII	Bell	1.75 (dia) x 1.15	<0.4	120 ±0.1	Torque balance	Hinged pendulum
GG177	Honeywell	1.5 (dia) x 1.8	0.3	170 ±10	Torque balance	Hinged pendulum
2401 -005	Kearfott	2 x 1.0 x 1.3	0.2	150 ±5	Torque balance	Hinged pendulum
GG116	Honeywell	2 (dia) x 1.8	0.5	180	Torque balance	Pivot and jewel
4310	Donner	1.4 x 3.0 x 1.2	0.5	-40 to +200	Torque balance	Pivot and jewel
VM4A	Autonetics	2.55 x 4.8	2.3	67.5 ±2	Force balance pendulum	Hydrostatic fluid bearing
16 PIPA	MIT (design)	1.6 (dia) x 1.8	0.4	133.5	Pulsed integrating pendulum	Flotation + magnetic

Pendulosity	Proof Mass	Proof Mass Material	Proof Mass Freedom	Velocity Storage	Pickoff Type	Torquer Type
2 gm-cm	--	Beryllium	±1.5 deg	15-47 ft/sec	AVDT	Perm magne servo-motor
25 gm-cm	--	--	--	--	Optical	Perm magne servo-motor
8.63 gm-cm	24 gm	Beryllium	±0.3 deg	--	Capacitive	Perm magne servo-motor
20 gm-cm	--	Beryllium	±0.3 deg	DNA	Moving coil	Perm magne servo-motor
--	--	Beryllium	--	--	Magnetic	--
0.16 gm-cm	--	Aluminum	±2 mrad	--	Capacitive	Perm magne
1.6 gm-cm	--	--	±6 mrad	--	Differential transformer	Perm magne
--	--	--	--	--	Capacitive	Perm magne
--	--	Aluminum	--	--	Microsyn	Perm magne
--	--	Aluminum	--	--	Capacitive	Perm magne
--	--	Manganin	--	--	Capacitive	Eddy current
0.25 gm-cm	--	--	±8 deg	--	Microsyn	Microsyn

	Other Unique Features	Suspension			Pickoff		
		Volts	Freq	Power	Volts	Freq	Power
t	Gyro H = 2000 gm-cm^2/sec	5	4.8 kc	250 ma	2	4.8 kc	
t	Gas bearing MIG (GG215) ceramic gimbal						
t	Gyro H = 10,000 gm-cm^2/sec				9	20 kc	NIL
t	Double integrating PIG				10	4.8 kc	NIL
					Perm magnet		
t					1.5	8 kc	
t					2	100 kc	0.26 w
t							
t							
t							
	Uses rotating magnet to generate restoring torque	20	3.5 cps sq wave	0.7 w	100	4.8 kc	5 w
		2	3.2 kc		2	3.2 kc	

Torquer			Velocity Resolution	Threshold	Other	Bias	
Volts	Freq	Power				Magnitude	Stability
- Not used -			0.12 ft/sec			(Refer to Volume	
			0.0156 ft/sec	10 µg			
- P.M. -			0.1 ft/sec	10 µg		500 µg	50 µg
- Not used -			0.15 ft/sec	(Refer to Volume IV of this repor			
				1 µg	2500 µg deadband near origin (0 g)	± 0.1g*	60 µg
				1 µg		200 µg	60 µg
				1 µg		100 x µg	80 µg
				0.1 µg		400 µg	80 µg
					Accuracy < 0.02%	500 µg	300 µg
			0.12 ft/sec pulse	(Refer to Volume IV of this repor			
			0.16 ft/sec pulse				

Scale Factor Stability	Linearity		Heater			Otl
	Quadratic Magnitude	Cubic Magnitude	Voltage	Freq	Power	
:V of this report for these data.)			28	DC	2.6 w	Gyrowheel 12 v 800 ‹
16 ppm						
50 ppm	Not measured	Not measured	28	DC 60 or 400 cps	20 w	Total power includes Gyrowheel 7.5 v 400
for these data.)				- None -		
15 ppm	$8\ \mu g/g^2$	$27\mu g/g^3$				Total power includes feedback amplifier fc
90 ppm	$>10\ \mu g/g^2$	$0.1\ \mu g/g^2$		- None -		
80 ppm						
30 ppm	$2\ \mu g/g^2$	$0.1\ \mu g/g^2$				
100 ppm	$10\ \mu g/g^2$	NM				
for these data.)				- None -		

ıer	Range	Range Factor	Usage	
ps, 2ϕ 100 ma/phase	±30 g	1.0 rad/sec/g	Advanced Minuteman	16 PIGA performa 11 PIGA .
	0 ±10 g	0.25 rad/sec/g		Developm well feels
heater - 15 w cps 1 w	(To ±100 g 0 ±20 g)	0.86 rad/sec/g		In early c
	0 to > 10 g	0.2 rad/sec/g	Saturn inertial guidance system	From "A form Sys
heaters 10-w positive r string excitation	± 50 g	127 cps/g		Exhibits Stabilitie
	±2.7 g	1 ma/g	LM abort guidance system	Only acce strapdow
	±60 g	1.5 ma/g	Centaur	
			Subroc, Flip	
	±100 g			
	± 20 g	69 pps/g	Minuteman	Large qu
	±11 g		Apollo and LM primary G&N system	A modifie used by M gram. H room ten handling Kearfott .

Table 3-V. Strapdown Accelerometers

Remarks	Accel Model No.	Accelerometer Manufacturer
√od J in development-range extended to ±100 g and ıce design goals better by a factor of 5. Smaller .lso under development Mod G is in production.	16 PIGA MOD G	MIT (design)
ent work on this unit has been stopped. Honey- that enough PIGA designs exist.	GG 162 GG 226	Honeywell
evelopment stages.	KAIG	Kearfott
General Description of the ST 124M Inertial Plat- em NASA TND2983" Sept. 1965, H. E. Thomason.	AB-3	NASA/Bendix
arge creep characteristics as function of time. s are for 10 days.	D4C	ARMA
lerometer being used in a production ı system.	Mod VII	Bell
	GG 177	Honeywell
	2401 -005	Kearfott
	GG 116	Honeywell
	4310	Donner
ntities produced.	VM4A	Autonetics
d version with a permanent-magnet torquer is IT and Sperry in their in-house strapdown pro- ›wever, this instrument must be kept warm (above perature) even during shipment. This creates a ›roblem not encountered if the Bell VII or '401-005 is used.	16 PIPA	MIT (design)

scheme must be carefully considered for an optimized mechanization.

a) Scale-Factor Stability

The time-dependent shift of the mean value of scale factor taken in the initial calibration period from that of any other subsequent calibration measurements taken during a specified interval.

b) Bias Stability

The time-dependent shift of the mean values of bias taken in the initial calibration period from that of any other subsequent calibration measurements taken during a specified interval.

c) Scale-Factor Temperature Sensitivity

The change of scale factor due to an operating temperature change of the instrument.

d) Bias Temperature Sensitivity

The change of bias due to an operating temperature change of the instrument

e) Alignment Stability

The time-dependent shift of the mean value of alignment taken in the initial calibration period from that of any other subsequent calibration measurement taken during a specified interval.

3.2.1 Pendulous Force Balance Accelerometers

Two accelerometers described in Section 4 are outstanding candidates based on information obtained to date. The performance parameters of these accelerometers are presented in Table 3-VI, with definition of the performance parameters as follows:

3.2.2 Integrating Accelerometers

Instruments such as the Autonetics VM4A and the MIT-designed 16 PIGA are considered high-quality instruments. Their drawback is that they weigh four to five times as much and are twice as large as the selected pendulous accelerometers. Also a premium is paid for the high accuracy, since a 16 PIGA costs at least twice as much as the instruments described in Table 3-III.

Table 3-VI. Selected Accelerometers

Performance Parameters	Bell Model VII	Kearfott 2401-005
Bias:		
Stability	20 μg (max)	30 μg (1 σ)
Temperature sensitivity	3 μg/°F (1 σ)	10 μg/°F (max)
Scale Factor:		
Stability	20 ppm (max)	30 ppm (1 σ)
Temperature sensitivity	25 ppm/°F (1 σ)	20 ppm/°F (max)
Integrating Accelerometer Stability	0.5 arc sec (max)	1.2 arc sec (max)
Threshold	1.0 μg (max)	0.1 μg (max)
Cross-coupling coefficient	4 $\mu g/g^2$ (max)	5 $\mu g/g^2$ (max)
Cross-axis sensitivity	———	20 μg/g (max)

Notes:
1. Stability is for 30 days with shutdown.
2. Source: Kearfott: performance data furnished by Kearfott.
3. TRW in-house evaluation of Bell VII has been primarily limited to determination of nonlinearities by precision centrifuging.

The 16 PIGA and VM4A are two of the most accurate acceleration sensors made and provide digital output signals. The gyroscopic principle is utilized in the MIT-designed 16 PIGA. The instrument is composed of a gyro on a single-axis platform. An unbalance is purposely built into the gyro so that an acceleration-sensitive torque is produced. The single-axis platform is servo-driven so that a turning rate is imposed on the pendulous gyro. The platform rate is controlled by the gyro pickoff output and becomes a null-seeking device where the acceleration-sensitive torque caused by the unbalance is exactly balanced by the gyroscopic torque which results from the platform. The turning rate is proportional to acceleration, and the total angle traversed is an indication of the velocity or the time integral of acceleration. The instrument is typically read out by means of a shaft encoder which produces an electrical pulse for each angular increment of the output shaft. The frequency of the signal is, therefore, proportional to acceleration, and each pulse represents an incremental change in velocity. The Bendix AB-3 accelerometer is a PIGA with a hydrostatic gas-bearing output axis instead of the floated, magnetically centered system used on the 16 PIGA. It should be noted that gyroscopic velocity meters are also sensitive to input angular rates relative to inertial space. This consideration is especially important if these instruments are body-mounted as in a strapdown guidance system.

The Autonetics VM4A velocity meter utilizes a torque-balance principle to measure acceleration. The instrument employs a hydrostatic fluid bearing to suspend a pendulous manganin cup which surrounds a separately mounted, rotatable magnet assembly. Rotation of the magnet shaft produces on the manganin cup a torque which is a linear function of shaft speed. This torque is controlled so as to balance exactly the pendulous torque produced by acceleration. Under these conditions, an encoder on the magnet shaft produces a frequency proportional to acceleration, and the total angle the shaft traversed is a measure of the change in velocity.

4. STRAPDOWN SENSOR ASSEMBLIES

Though many companies are involved in strapdown guidance, the details relative to the depth of their involvement and their future plans are, in many cases, proprietary. This information can be obtained only at the specific direction of NASA.

A survey of sensor assemblies is presented in Table 4-I. The gyros and accelerometers listed in the table are those which the indicated company has used in actual strapdown testing. It must be realized that instruments are selected to meet the specific requirements of the system. Such factors as cost, size, weight, power, performance, availability, and operating life are derived from system requirements and are used as criteria for instrument selection.

Complete information is available on only three of the systems listed in Table 4-I. These are the United Aircraft system which is used in the LM Abort Guidance System, and the TRW Systems, Inc. Models TG-166 and TG-266. The UAC system is known, of course, because TRW Systems, Inc. produces the LM/AGS system, with UAC providing the sensor assembly under a subcontract to TRW.

Although a complete assembly of the TRW Model TG-166 has not been fabricated, considerable design and analytical work has been accomplished. The error models are well known and have been applied to environmentally induced errors as well as to a particular mission trajectory. This system, which uses a modified version of the Nortronics GI-K7 gyro and the Kearfott Model 2401-005 analog rebalanced accelerometer, is considered a moderate-accuracy, low-cost system.

The TRW Model TG-266 system, which is in the design and analysis phase, will have increased accuracy (but somewhat higher cost) by the use of a more accurate gyro and by employing a newly developed, pulse-torquing technique. The system can be adapted to use either the Honeywell GG334A gas-bearing gyro or the Kearfott 2564/2565 ball-bearing gyro in a pulse-torqued mode.

Manufacturer	Size (in.)	Weight (lb)	Power (not heaters)	Model
United Aircraft Corp. Systems Center (UACSC) (TRW Systems Design Control Spec SS2-1, Rev. D for LM/ASA)	13.5 x 9.0 x 5.1	25	28 v, 128 kc clock, 43 w	Norden RI-1139B
TRW Systems Model No. TG166	10 x 8.5 x 6.2	23	40 w	Nortronics GI-K7 (Modified version)
TRW Systems Model No. TG266	11 x 8.5 x 6.2	25	40 w	Honeywell GG334A or Kearfott 2564/2565
MIT Instrumentation Laboratory*	----	--	---	25 IRIG MOD II (Modi- fied with permanent magnet torquer and capable of 100 deg/se torqing rates). Desig nated 25 PIRIG
NASA-Huntsville	21 diameter x 14 length	--	---	NASA-Bendix AB-5
Honeywell*	----	--	---	Honeywell GG334A
RCA (Burlington)*	----	--	---	Honeywell 8001 and Nortronics K7
Sperry*	----	--	---	Gas bearing, high torq ing rate (permanent magnet torquer) gyros fabricated by Sperry. Designated SYG 1440

*Best known estimate; information unofficial

Gyros	Accelerometers		
Capture Loop	Model	Capture Loop	Deve
Pulse torque, constant power, pulse width modulated, two current levels (plus and minus)	Bell VII	Pulse torque, constant power, pulse width modulated, two current levels	Proc back
Pulse torque, constant power	Kearfott 2401-005	Analog torque	Deve
Pulse torque, constant power	See Volume IV	Pulse torque, constant power	Desi phas
Pulse torque, constant power ternary (plus, zero and minus) system utilizing a dummy load in the gyro, current is passed through the dummy load during the zero state.	16 PIPA (Modified with permanent magnet torquer)	Pulse torque, constant power, fixed pulse, two charge states	One
Gyro is mounted on a servo motor driven turntable. Turntable output is a theodosyn.	NASA-Bendix AB-3	Accelerometer turntable uses a theodosyn to provide a digital output	Sing tests earl
Pulse torque, ternary system variable power	Honeywell GG177	----	Prot but I all in prop
Pulse torque, ternary, variable power	Honeywell GG116	Pulse torque, two state	Unkr
Advanced pulse-torquing scheme (details proprietary)	16 PIPA (Thrust axis PIPA is modified with a permanent magnet torquer) or Bell VII	----	Unkr

Table 4-I. Strapdown Sensor Assemblies

:lopment Status	Comments
luction for LM up	Specification Requirements: (All stability measurements are for 120 days, 3 σ, with temperature variations between +60 to +160°F Gyro Channel: Bias stability = 2.8 deg/hr Mass unbalance stability = 0.6 deg/hr-g Scale factor stability = 270 ppm Accelerometer Channel: Bias stability = 200 μg Scale factor stability = 260 ppm
lopment phase	Low cost, moderate performance system. Uses proven inertial components
gn and analysis e	High performance system. Uses state-of-the-art components and pulse torque techniques.
prototype system	
e-axis breadboard . Prototype in ' design	
ɔtype systems exist, :oneywell considers .formation to be :ietary	
ɔwn (proprietary)	The indicated instruments have been utilized in single axis testing. Further effort is dependent upon receipt of a contract.
ɔwn (proprietary)	Interfaces with UNIVAC 1800 series computers.

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7. Mettler, E., "Strapdown ESG/ESA Capabilities," TRW Systems IOC No. 7223.3-71 and No. 7223.3-225, 11 March - 29 July 1966.