

R-600

(Final Report on Task II of Contract NAS-9-6823)

VOL. III RADIATION SENSOR SYSTEMS

JANUARY 1968

INSTRUMENTATION LABORATORY •

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CONTROL, GUIDANCE, AND NAVIGATION FOR ADVANCED MANNED MISSIONS

(Final Report on Task II of Contract NAS-9-6823)

VOL. III RADIATION SENSOR SYSTEMS

JANUARY 1968

INSTRUMENTATION LABORATORY MASSACHUSETTS INSTITUTE OF TECHNOLOGY CAMBRIDGE, MASSACHUSETTS

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ABSTRACT

This is a Final Report on Advanced Manned Missions Guidance and Control Systems Study, NASA Contract NAS 9-6823.

The objective of this study is to develop and demonstrate via laboratory tests, advanced guidance and control techniques which, for future manned missions, will minimize spacecraft constraints, enhance mission flexibility, and eliminate the mission time dependency of guidance and control system reliability, while maintaining adequate performance to accomplish the required missions.

January 1968

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INTRODUCTION AND SUMMARY

Volume III of R-600 is concerned with the development of radiation sensors subsystems for advanced manned space missions. The work reported in this volume was performed under Part II, Task 5 of contract NAS-9-6823 between the NASA Manned Spacecraft Center and the Massachusetts Institute of Technology Instrumentation Laboratory.

General requirements for advanced manned missions upon Control, Guidance, and Navigation Systems are set forth in Volume I. In that volume, a report is made of a general study of the overall requirements for advanced manned missions involving a range of exploratory missions in the solar system following the Apollo and Apollo Applications Category of missions.

Volumes II and IV respectively direct themselves to the development of computer subsystems and inertial subsystems as applicable to advanced manned space missions.

In this volume, there is set forth in detail the results of a ten-month's effort to investigate various aspects of advanced technology applicable to radiation sensor subsystems for advanced manned missions.

The studies discussed herein are based upon manned planetary missions. Although other mission types were considered, such as extended earth orbital and space station missions, it was concluded that the planetary-type missions were the most demanding and that the earth orbital and space station missions could be performed by a system designed for planetary missions. It was also concluded that the alignment and navigation for many of the anticipated earth orbital and space station missions could be achieved with adaptations of Apollo generation hardware. The versatility, reliability, accuracy and operational characteristics required for a planetary mission appear in general to encompass the characteristics required for the other mission types considered.

The functions considered in this study which were to be performed by a radiation sensor subsystem during a planetary mission were:

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- 1. Spacecraft navigation
- 2. Inertial measurement unit (IMU) alignment
- 3. Rendezvous navigation and guidance

4. Probe navigation and guidance

5. Low-power, long-duration spacecraft attitude determination

Of these functions considered, the spacecraft navigation function is the most demanding upon the development of a radiation sensor subsystem. The RSS design model presented in Chapter 4, Section 6 is primarily designed to satisfy the spacecraft navigation requirements. This design model also has the capability of performing the IMU alignment function, as discussed in Section 4.6. These first two functions listed are the ones which can most readily be analyzed within the present state of definition of manned planetary missions.

The last three functions listed will be primary tasks for onboard radiation sensor subsystems and should be analyzed as soon as mission guidelines become available.

Rendezvous will probably be a major factor in manned planetary missions. Even for the Saturn V class of boosters, the large size of the spacecraft will necessitate its assembly in space. Also, rendezvous will be necessary in order to use recoverable probes for planetary exploration. Most presently anticipated rendezvous navigation and guidance tasks can be performed by the design mode.

Ejected probes will probably be the vehicle for many of the planetary experiments which serve as the motivation for planetary missions. Except during terminal guidance, the probes can be accurately tracked and guided using the spacecraft navigation system. Much of the navigation of probes can best be performed from onboard the spacecraft in order to simplify the probe system as much as possible and to avoid duplicating sophisticated equipment.

Low-power, long-duration attitude determination will be necessary to maintain spacecraft attitudes and rotations throughout extended planetary missions. Radiation sensors such as the star tracker, sun sensor, and scanner used in the design model would serve this function. There is a strong probability that in the not too distant future, low thrust atomic engines will become practical for planetary missions. To utilize these engines, long-duration guidance would be required which is best provided by radiation sensors. This function can also be provided by the radiation sensors used in the design model. Based upon the requirements for the last three functions listed, it is felt that most of these RSS tasks can be performed by appropriate adaptation of the design model.

Because planetary navigation is considered to be the most demanding and the presently best defined mission function, the bulk of this volume is thus directed toward the analysis and development of a planetary navigation system. There are four facets of this study which are presented in the four chapters of this volume. These four efforts were carried out in parallel because of the limited duration of the study. The results of each effort were fed into the other efforts as much as possible, but it should be noted that there is some work presented in each section which time would not allow to be fed back into the other sections. This study represents a sound framework for development, but a considerable amount of study remains yet to be done. There is sufficient material presented in each chapter to form the basis for a next generation of effort. At the end of this section, recommendations are given on the next steps which need to be taken toward the eventual development of a radiation sensor subsystem for advanced manned missions,

The first of the four chapters of this volume presents the phenomena which would be used as navigation and IMU alignment references, and their associated location uncertainties. The state of planetary knowledge is an important factor in determining the utility and location uncertainty of each of these phenomena. Some of the most significant phenomena uncertainties are discussed at the end of the chapter.

Sufficient fundamental scientific inform. tion on solar system reference features is available to permit accurate interplanetary onboard navigation (see Chapter 3), but this knowledge is limited and any new data from unmanned probes would enable the development of a more optimal system. Thus one future objective of manned navigation task groups would be to maintain a close communication with other groups associated with unmanned planetary probe missions, so as to be cognizant of all pertinent data. Another objective would be to have data needs reflected in the instrumentation of future unmanned probes. Future data would be desirable to verify existing planetary data. For some as yet unspecified mission it may also be desirable to reduce the uncertainties further. This could be accomplished during future unmanned or manned missions. It is anticipated that the uncertainties indicated in Chapter 1 can be considerably reduced over the next few years using anticipated future probes. The resulting lower uncertainties could then be utilized in the software of the first manned planetary missions without significantly effecting hardware designs.

Chapter 2 presents the navigation measurement techniques considered, which are based upon the phenomena discussed in Chapter 1. Profiles and detection error models are developed for planetary horizons. Error models are presented for the various navigation measurement techniques which use uncorrelated reference uncertainties and the angle measurement uncertainties discussed in Chapter 4. Correlated measurement models are presented which can provide major inflight improvements in the preflight horizon phenomena uncertainty

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values, thereby providing more accurately located planetary references and constituting a valuable planetary scientific experiment.

Collaboration between the navigation and scientific efforts of a mission can provide many mutual benefits. The sensitivity of navigation horizon locators to atmospheric phenomena is discussed in Section 2.3. Any information that scientific experiments can provide on atmospheric characteristics will reduce the phenomenon uncertainty in horizon navigation accuracy. Even if the terrestrial limb of the planet is used, rather than an atmospheric locator, scientific observation of planet diameter and topography can assist navigation. Observations of clouds, dust, aerosols, haze and other factors which might distort or obscure the horizon would certainly increase measurement reliability.

The navigation measurements can contribute data which is of scientific value for investigation of atmospheric phenomena. Section 3.3 presents some of the atmospheric parameter uncertainties which can be improved during the process of taking navigation measurements. This measurement system was designed solely to perform navigation measurements. If the scientific possibilities are considered in the design objectives, a system which yields considerably more scientific information could be designed. The combined navigation-scientific measurement concept outlined here would increase the real-time computer and astronaut participation over that required for the uncorrelated navigation measurement formulation. However, the additional effort would be very productive and should be investigated in greater detail, considering the optimal adaptation of the RSS design model.

An Earth orbital experiment has been proposed by MIT which would measure the atmospheric constituent densities as a function of altitude. It is presented in MIT/IL Report E-2220, <u>MIT Aeronomy Experiment</u>. The same experiment can be used for flyby trajectories as well as orbital trajectories. An optimum planetary experiment would be almost identical, with only modifications in the wavelengths used in the data analysis. Somewhat different wavelengths would be selected in order to adapt to the planetary atmospheric composition, and the atmospheric components of primary interest.

Chapter 3 presents a navigation error sensitivity analysis which assumes the error models discussed in Chapter 2. Volume I has presented the onboard navigation formulation which has been developed and which was used to evaluate the ACG & N requirements for several simulated representative missions. Chapter 3 presents the results of a limited analysis of several representative planetary mission phases and indicates the navigation accuracies which can be anticipated. An Earth-to-Venus leg, an Earth-to-Mars leg, a Venus flyby leg, a Mars flyby leg, and Mars orbital phase are simulated. A range of phenomena uncertainties, navigation instrument accuracies and measurement schedules are investigated to determine their effect upon navigation accuracy during the mission phases.

Navigation position and velocity determination accuracies are the principle performance indicators which have been evaluated. These accuracies have direct impact on mission cost functions such as: total mission Δv , probe navigation and guidance accuracy, atmospheric braking accuracy, atmospheric entry and landing accuracy, etc. These quantities constitute primary motivation for achieving given levels of system accuracy. However, these quantities were not used as performance indicators in this chapter because of insufficient time and resources to do so, and since they are more dependent upon mission definitions. To proceed further in using these cost functions it would be very helpful to have some more definite missions definitions, objectives, and constraints. Otherwise there is such a large number of parameters that it becomes prohibitive to evaluate all of the possible combinations. The evaluation of system performance requirements against mission cost functions is one of the most significant tasks necessary toward the development of a planetary navigation and guidance system.

The results presented in Chapter 3 indicate that navigation accuracies of a few miles can be achieved with onboard navigation. One point to note is that the onboard navigation is only susceptible in the "second order" to: astronomical units (AU), planet ephemeris, and planet geophysical uncertainties. The effect of these uncertainties, which, for example, cause errors in the initial conditions at interplanetary injection, are diminished to an insignificant amount as the spacecraft nears the vicinity of the planet. By sighting directly at the target planet, the propagated errors from these uncertainties are essentially eliminated. One advantage of onboard navigation then, is that the navigation becomes based in the reference frame of the target planet. Also for any atmospheric braking maneuvers, probe guidance or planetary experiments, the onboard sensors can directly determine the location of the significant phenomena in the spacecraft frame of reference.

An example is provided by a planetary atmospheric entry maneuver. In addition to requiring spacecraft trajectory knowledge, this task calls for accurate knowledge of the atmospheric density profile. The strict definition of the entry window is that the spacecraft should enter a specific density region at a prescribed angle and velocity. In cases where low entry angles are involved, unexpected density variations can jeopardize the mission even if the trajectory is accurate relative to the center of the planet. The advantage of optical

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navigation in this situation is that it actually measures the spacecraft position relative to the density profile, which is the parameter needed for precision entry.

Several tentative conclusions have resulted from the simulations runs thus far for certain phases of these two specific missions:

- 1. That the space sextant angle measurement uncertainties limit the navigation accuracies achievable during the interplanetary mission midcourse phase, and that the navigation accuracy during midcourse for a given measurement schedule, is approximately proportional to the sextant angle measurement uncertainties.
- 2. That the planetary phenomena location uncertainties relative to the planet, limit the ultimate navigation and guidance accuracies achievable during flyby and orbital mission phases and that navigational accuracy for a given measurement schedule is approximately proportional to the "average" phenomena location uncertainties.
- 3. That the ΔV required to achieve a given guidance accuracy at the planet (as allowed by the navigation accuracy limitations imposed by the planet locator phenomena uncertainties) is approximately proportional to sextant angle measurement accuracy. This conclusion is based upon portions of two specific missions and represents a limited amount of data. The ΔV 's performed do not necessarily represent an optimum schedule which could reduce the ΔV 's required. On the other hand, it is believed that the proportionality between the ΔV 's required and sextant accuracy is representative for these specific mission phases. Differences in the ΔV required will result from different guidance requirements and for different solar system and mission geometries. As previously stated, more analysis is required to generally evaluate this mission cost function.

Chapter 4 presents the RSS design studies which were used as the basis for the instrument accuracies and functional constraints which were incorporated into the error models presented in Chapter 2 and evaluated in Chapter 3. The first half of Chapter 4 presents, in approximately evolutionary order, the thinking that went into the formulation of the RSS design model. A ground rule for this study was that complete attitude isolation between the spacecraft and the sensor head be provided by the RSS mechanization so as to place a minimum of demands upon spacecraft attitude control. Several alternative designs are presented and discussed and the conclusion is drawn that an automatic system which uses electro-optical sensors and electrical transfer of information to the astronauts will provide the best sum of accuracy, versatility, reliability, independence from other spacecraft functions and an appropriate level of astronaut participation. Particular emphasis was placed upon devising a general purpose design model which could be used for planetary missions involving all anticipated relevant solar system bodies, and serve as many functions as practical. The design model presented uses a few versatile, general-purpose sensors rather than multiple specialized instruments. The sensors chosen can detect all of the star and solar system phenomena considered to be of significance for guidance and navigation.

It should be noted that backup systems or techniques are not discussed herein. There are several techniques which are simpler in mechanization than the design model, but which are seriously deficient in one or more of the previously listed performance characteristics. In the improbable event of catastrophic, irrepairable failure of the RSS system, manual measurement and/or computation techniques could be used as backups. Hand held space sextants, star occultation orbital navigation, visual attitude determination and alignment are a few of the possibilities. Except in emergency situations sole use of these techniques generally places prohibitive demands upon the astronaut, the total Δv and the other spacecraft systems.

The last half of Chapter 4 presents several specific topics related to the design of an RSS, such as folded optics design, stray light baffle design, automatic landmark tracking and bearing heat transfer studies. Also, flashing light visibility studies are presented which are of particular significance for rendezvous tasks. In addition, a navigation measurement mechanization is presented consisting of a scanner mated to a ring laser which can conceptually provide navigation angle measurements accurate to 1/10-arc second. For the missions considered thus far, this degree of accuracy does not appear to be required, but it should be noted that if the need does arise such accuracies are conceptually possible.

Radiation Sensor Subsystem Development Task

There are several "next step" key tasks which have evolved out of the effort reported in this document.

Research:

a. Continue correlation and assimilation of planetary geophysical data into navigation reference models.

- b. Communicate critical geophysical uncertainties to unmanned planetary probe projects so as to realize appropriate experiments that may be compatible with the package.
- c. Devise cost functions which are most representative of mission and system optimum.

Analysis:

- a. Update mission navigation simulation error models to represent all of the concepts reported in Chapters 1, 2, and 4.
- b. Perform analysis of limb phenomena locators and associated sensor designs to develop optimal combinations and to determine detection uncertainty functions.
- c. Perform an information and servo analysis of a design model instrument in a range of dynamic environments. One example would be the dynamic environment encountered during an artificial gravity rotation.
- d. Complete parameter studies of flyby and orbital missions.

Design:

- a. Design an engineering model of RSS based upon design model.
- b. Design a scanner-photometer sensor package.

Hardware Development;

- a. Build and evaluate an engineering model of design model optics.
- b. Build and evaluate an engineering model of a video tracker.
- c. Build and evaluate an engineering model of a ring laser angle encoder.

The following is a list of the general objectives relating to the design of Radiation Subsystems for advanced manned missions.

- 1. Conduct a study program to update knowledge of the potential navigational reference features in the solar system and their potential for meeting the mission and system requirements. Such phenomena as planet terrestrial or atmospheric limbs, planet radiation distribution centers, unknown landmark target spatial distributions, microwave and laser radar ranging, are involved.
- 2. Use these studies to develop sensors and related electronic and mechanical hardware to utilize the phenomena which have

been identified above in addition to star phenomena which will be used for celestial attitude reference.

- 3. Evaluate these sensors to the extent possible in laboratory or atmospheric test programs and also design and develop the necessary experiments which can be used to qualify these sensors on early manned and unmanned missions.
- 4. Design and develop Radiation Subsystems for advanced manned missions, based on the results of the above tasks.

This technical area is characterized by the need for geophysical research, associated analysis to determine optimal phenomena locator-sensor combinations, systems analysis, and engineering development experiments in the laboratory and in space environments.

Early effort would involve maintaining a design model of the Radiation Sensor Subsystem based upon best available scientific and engineering information. Concurrently, systems analysis and component systems development would be accomplished. Several experiments would be identified and designed involving materials, components, and sensors. These experiments would be suggested for inclusion on certain Apollo Applications (AAP) missions.

In parallel and prior to receiving all of this information a radiation subsystem will be developed for use and evaluation on extended orbital flights such as a space station. This interim subsystem would involve design and construction of an engineering model and a prototype for use in a test and qualification program. In the sequence suggested here, a set of subsystem drawings would be available for use in bringing an industrial contractor into the program.

A time line of this development plan is shown in the following figure.

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Radiation sensor subsystem timeline

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1. GEOPHYSICAL AND PLANETARY STUDIES

1.1 Motivation for and Nature of Study

Fundamental to the problem of interplanetary navigation is the maintenance of a sufficiently accurate spacecraft state vector. Data required to derive the spacecraft position and velocity with respect to a given planetary body can be obtained by a variety of angular measurement sequences accepted by a recursive filtering technique. Several measurement types may be employed, but the most accurate can be shown to utilize the angle between the line of sight to a selected star and a line of sight to a well-known position on some nearer body, e.g. a planet, the Sun, or perhaps a moon. The most useful feature for this latter line of sight appears to be a point on the limb of the planet (or Sun or Moon) in the plane containing the spacecraft, the planet center, and the selected navigation star. This point or locator on the limb would be some specified point in the horizon (e.g. that point with a radiative int, of one half the maximum for the planet), or conceivably, in the case of a planet with a very thin atmosphere, the edge of the solid body itself. Initially, at least, the ability to utilize both sunlit and dark limbs must be considered.

The uncertainty in any one elemental measurement will generally consist of the sum of an angular instrument uncertainty plus the distal uncertainties in the locator position. It is with these distal or phenomena uncertainties that this section is concerned. The objective here is to consider the various celestial bodies which could profitably be used for navigational information, and to determine just what can be said toward establishing the magnitude of phenomena uncertainties for locators associated with those bodies. It should be noted that these uncertainties are equally relevant when the body is employed purely as an attitude reference.

To delimit the size of this study and to provide a real element of applicability, navigational simulations of two specific missions have been considered in this report. The first, scheduled for launch in 1979, is a Mars flyby mission including return to Earth (shown in Fig. 1-1). The second mission is a triple planet flyby to be launched in 1977 (shown in Fig. 1-2). This would first go to Venus, then to Mars, back to Venus and on to Earth again.

Since for these missions navigation with respect to the Earth, Venus, and Mars are of primary importance, locators and phenomena uncertainties for these planets are of particular interest. To establish the inherent optical stability and associated uncertainty of a potential locator, good atmospheric models and data on meteorological and climatological variations are needed. Such data is largely available for the Earth, but is still highly speculative for both Venus and Mars. Given this information, however, one can develop appropriate scattering and absorption models to predict the radiative nature of a planetary horizon. MARS FLYBY: Projection of trajectory and planetary orbits into plane of the ecliptic; the vernal equinox (VE) of date is indicated by an arrow. The dates of occurrence of significant events are listed below; correlative spacecraft and planet positions are represented on the figure by the associated event number. Incremental spacing on the figure frame is in units of 0.1 astronomical units.

Event	Number	Date	
Transplanetary Injection	1	3 December 1979	
Mars Encounter	2	21 April 1980	
Return to Earth	3	9 October 1981	

Fig. 1-1. Mars flyby. (Sheet 1 of 2)



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TRIPLE PLANET FLYBY: Projection of trajectory and planetary orbits into plane of the ecliptic; the vernal equinox (VE) of date is indicated by an arrow. The dates of occurrence of significant events are listed below; correlative spacecraft and planet positions are represented on the figure by the associated event number. Incremental spacing on the figure frame is in units of 0.1 astronomical units.

Event	Number	Date
Transplanetary Injection	1	24 February 1977
Venus Encounter #1	2	18 June 1977
Mars Encounter	3	20 December 1977
Venus Encounter #2	4	26 August 1978
Return to Earth	5	1 January 1979

Fig. 1-2. Triple-planet flyby. (Sheet 1 of 2)



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For this purpose a critical survey of the available literature on the atmosphere of Venus and Mars has been undertaken. A brief condensation and interpretation of this data is presented later in this section as a basis for preliminary determinations of locator uncertainty on these planets. Atmospheric data for the Earth is both well-known and extensive and will not, therefore, be repeated in this document.

The data on Venus and Mars are frequently ambiguous and sometimes contradictory. The reader must then be aware that the models and figures presented here are neither final nor is there associated with them a high degree of confidence. Rather, they serve as a reference upon which various studies testing the sensitivities of horizon locator uncertainties may be based.

1.2 Potential Navigation Bodies and Phenomena

Any body in the solar system which is detectable and of known position may serve as a navigation or attitude reference for interplanetary missions. The planets, their satellites, asteroids and the Sun are all potentially useful, depending on the nature of the specific mission.

Since these bodies are seen with a finite angular subtense, two modes of resolution may be considered. Either the limb or edge of the apparent disc may be detected, or a known position on the viewed surface of the body may be used. In this latter category, known landmarks might be used at close range, while for very distant cases the point of peak intensity in the scattered sunlight or infrared thermal radiation would, perhaps, prove useful. The peak intensity detection scheme can, however, be quickly disregarded due to its sensitivity to local variations, meteorology and albedo, even if corrections for diurnal, seasonal, latitudinal, and phase angle effects were known and applied. As for known landmarks, there is presently insufficient data to select landmarks of meaningful accuracy on any body other than the Earth and our moon. It can be shown that in view of cloud cover problems on the Earth, geodetic uncertainties in landmark position, and the possibility for erroneous identification of the intended landmark, this technique presently holds little if any advantage over other methods.

Neither the radiation center nor known landmark scheme will then be further considered within this study. Limb or edge detection with respect to a large number of bodies still, however, remains practical, but many of these possibilities may also be dismissed. This is done by considering the trajectories of specific missions of interest and the planetary configurations at those times. Initially, at least, it is suggested that all asteroids and satellites, excluding the Moon, be eliminated because of insufficient data regarding their ephemerides. Detection could also be problematic in many cases. Second, there will be a finite amount of instrument uncertainty in determining the line of sight to the selected body. Therefore, the uncertainty in the derived information deteriorates in proportion to the distance from the spacecraft to that body. Because of this, the a priori assumption was made that no planet beyond Jupiter would be useful. Furthermore, Jupiter was to be considered only if it was within two astronomical units of the spacecraft, or if it was otherwise in a position such that its use could markedly reduce the uncertainty in the spacecraft state vector. Neither of these conditions was met for the mission being considered, and Jupiter was subsequently eliminated from further consideration. The navigation simulations, the results of which are discussed in Section 3, appeared to lend validity to all of these deletions.

Mercury was also eliminated from the study due to its proximity to the Sun and the resulting difficulty and occasional inability to detect that planet. It is expected that little information is lost here as the Sun itself can be used as a reference in that general direction. Finally, the Moon was eliminated since it offered little more information than the Earth. The horizon of the Earth can probably be more accurately referenced than a point on the edge of the lunar disc due to uncertainty of surface topography on the Moon. Continued mapping of the lunar surface from Orbiter photos may eventually give lunar landmarks a small margin of improvement over the Earth's horizon, but the difference is never likely to be large. Only while the spacecraft is in the Earth's sphere of influence are lunar references apt to be useful, and though favorable measurement directions may occur during this time, depending on lunar phase, the added information potential would not represent any gross improvement.

Remaining for use as navigation and attitude references are the Earth, Venus, Mars, and the Sun. In the case of the former two, the atmospheres are sufficiently thick that the edge of the solid planet cannot be "seen" anywhere in the ultraviolet (UV) to infrared (IR) spectrum. Therefore, only horizon locators in their atmospheres would be of value. Since the atmosphere of Mars is much thinner, either the edge of the solid disc or a horizon locator in its atmosphere might be used. Use of the Sun would probably employ the edge of the photosphere. Measurements of any of these features must necessarily consider uncertainties in altitude or radius due to atmospheric variations and instabilities, and uncertainties in radius and other geodetic parameters.

In addition to these "near" bodies, stars (excluding the Sun) will serve as very accurate, quasi-inertial attitude references. These would ultimately be selected according to the mission needs for spatial distribution. For the purpose of the present study, the thirty-seven navigation stars selected for the Apollo Guidance and Navigation System (Table 1-1) will be used. These stars are well distributed, having no obvious gaps, and are all of third magnitude or brighter.

1-7

No	Star Name	Vis. Mag.	Spectral Class	Approximate Right Ascension (1968)		Approx Declination	rimate on (1968)
140.				(Hr.	Min.)	(Deg.	Min.)
1	a Andromedae	2.1	AO	0	07	+28	55
2	β Ceti	2.2	K0	0	42	-18	10
3	γ Cassiopeiae	2.2	B0	0	55	+60	33
4	a Eridani (Achernar)	0,6	B5	1	31	-57	24
5	a Ursae Minoris (Polaris)	2.1	F8	2	02	+89	07
6	heta Eridani	3.4	A2	2	57	-40	26
7	a Ceti	2.8	M2	3	01	+03	58
8	a Persei	1.9	F5	3	22	+49	45
9	a Tauri (Aldebaran)	1.1	gK5	4	34	+16	27
10	β Orionis (Rigel)	0.3	cB8	5	13	-08	14
11	a Aurigae (Capella)	0.2	gG0	5	14	+45	58
12	a Carinae (Canopus)	-0.9	F0	6	23	-52	÷ 41
13	a Canis Majoris (Sirius)	-1.6	A0	6	44	-16	40
14	a Canis Minoris (Procyon)	0.5	F5	7	38	+05	18
15	γ Velorum	1.9	00	8	09	-47	15
16	Ursae Majoris	3.1	A5	8	57	+48	10
17	a Hydrae	2.2	K2	9	26	-08	31
18	a Leonis (Regulus)	1.3	B8	10	07	+12	09

Table 1-1. Navigational Stars (Sheet 1 of 2)

	Cten Nomo	Vis Mag.	Spectral Class	Dectral Approximate Right Class Ascension (1968)		Appro Declinati	ximate Ion (1968)
INO.	Star Name	4 1		(Hr.	Min.)	(Deg.	Min.)
19	6 Leonis	2.2	A2	11	47	+14	45
20	γ Corvi	2.8	B8	12	14	-17	22
20 21	a Crucis	1.6	B1	12	25	-62	55
22	a Virginis (Spica)	1.2	B2	13	24	-11	00
23	n Ursae Majoris	1.9	B 3	13	46	+49	28
24	A Centauri	2.3	K0	14	05	-36	13
25	a Bootis (Arcturus)	0.2	gK0	14	14	+19	21
26	a Coronae Borealis	2.3	A0	15	33	+26	49
27	a Scorpii (Antares)	1.2	gK0	16	27	-26	22
28	a Trianguli Austr.	1.9	K2	16	45	-68	58
29	a Ophiuchi	2.1	A5	17	33	+12	35
30	. a Lyrae (Vega)	0.1	A0	18	36	+38	45
31	σ Sagittarii	2.1	B3	18	53	-26	20
32	a Aquilae (Altair)	0.9	A5	19	49	+08	47
	6 Capricorni	3.2	F8	20	19	-14	53
24	a Paymois	2.1	B3	20	23	-56	50
35	a Cygni (Deneb)	1.3	A2	20	40	+45	10
36	e Pegasi	2.5	К0	21	43	+09	44
37	a Piscis Austr. (Fomalhaut)	1.3	A3	22	56	-29	48

Table 1-1. Navigational Stars (Sheet 2 of 2)

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As indicated in the preceding section, the above navigation and attitude references may be combined in sextant-type measurements to yield useful position and velocity data. Three other navigation techniques may also provide useful data during close approach and actual flyby or orbital phases of a mission. Star occultation by a planet, the tracking of unknown landmarks or surface irregularities, and the use of an onboard active radiation system such as radar, microwave or laser radiation will all be considered. These, as well as the previously mentioned measurements, do, however, imply a distal uncertainty in the extrapolated data due to uncertainties in the reference phenomena. Information relating to these phenomena uncertainties and their causes is presented for each reference body, respectively, in the following section.

1.3 Atmospheric and Geodetic Considerations

1.3.1 Stars

All stars (excluding the Sun) may be treated as point sources of radiation, contributing no resolvable phenomena uncertainty. This condition will generally hold for any mission within this solar system.

Since the stars are actually moving relative to this solar system, their attitudes are only quasi-inertial. In addition, the approximation that stars may be considered infinitely distant is invalid for some stars on missions to distant parts of the solar system. Therefore, regular updates on star attitudes will be required to account for this temporal drift and potentially for change of spacecraft position. These updates would be made at whatever frequency was necessary to maintain the star attitude reference file to the desired accuracy.

1.3.2 Sun

A reliable value for the uncertainty in the radius of any measurable feature of the solar disc is indeterminant. The chromosphere is strongly disturbed by the various forms of flare activity and is rendered useless as a reference. Photographs of the photosphere are significantly more promising and indicate an approximate uncertainty in the radius to the top of that region,

$1\sigma = 7500 \text{ km}$

This is still a rather large uncertainty, but there is considerable evidence that careful selection of wavelength could produce a locator in the solar limb, probably at the top of the photosphere, which would have a radius uncertainty of approximately 500 km (1 σ). Until conclusive data is obtained, however, the more conservative value will be applied as a reference value for navigation studies.

1.3.3 <u>Earth</u>

The <u>U. S. Standard Atmosphere</u>, 1962 is used in these studies as the reference atmosphere for the Earth. Abundant data is also available regarding daily, seasonal, and latitudinal variations. Recent studies are also improving the knowledge of cloud cover, ozone distribution, and albedo. Because this data is so readily available and relatively non-controversial, it will not be reiterated in this document. The combination of this information, however, permits the calculation of a horizon locator with a reasonable small uncertainty (see Section 2).

It does not appear that known landmarks would be used. The potential of these as indicated by their position uncertainty should, however, be indicated. These values vary significantly, depending on the continent concerned, but range from about 0.01 miles to 1.50 miles in the horizontal and rarely exceed 0.01 miles in the vertical. Average values are approximately 0.20 miles horizontal and 0.01 miles vertical.

The topography is, of course, well known and any altitude or radius figure could potentially be corrected to within a predetermined uncertainty provided that a sufficient portion of the topographic information had been coded among the accessible data.

1.3.4 Venus

A considerable amount of data has been collected regarding the Venusian atmosphere, yet it is far from being understood. This fact was graphically illustrated by the results of the recent Soviet probe and American [Jet Propulsion Lab (JPL)] flyby to Venus. While relatively consistent with each other, their data tended to contradict or otherwise invalidate the conclusions of a large percentage of previous experiments (including the Venus survey presented in Advanced Systems Memo #78-67). This study has accepted the limited, preliminary results of those recent missions. The earlier data has been re-examined in order to extract any data still considered valid, and to re-evaluate these so as to be consistent with the probe and flyby results.

Most striking and generally unexpected was the atmospheric composition reported by the new experiments. The Soviets reported 90-95 percent CO_2 while the JPL results indicated 75-85 percent CO_2 . The Soviet experiment reported no nitrogen present with a detection threshold of about 7 percent. Correlation of these values and consideration of IR absorption spectra and cosmic elemental abundances suggest a total composition as indicated in Table 1-2.

In addition to composition, the Russian probe also measured temperature and pressure at two points during descent and at the surface. The JPL

Carbon Dioxide	88%
Nitrogen	7%
Argon	5%
Water Vapor	0.1-0.7%
Oxygen and Ozone	Trace
Carbon Monoxide	Trace
Hydrocarbons	Trace

Table 1-2. Probable Composition of the Venusian Atmosphere

S-band occultation experiment provided refractivity as a function of distance from planet center. From this, temperature and pressure can be derived, and two points of data were provided on a preliminary basis from JPL. Using radardetermined radius information provided by the MIT Lincoln Laboratory, the JPL data was then correlated with the Russian data to produce a temperature profile. Phase angle and latitude correlations of the Soviet and American data were not made as the only available data was insufficient for those purposes. A priori comparison of their trajectories suggests, however, that data from both missions referred to a relatively moderate to cool region of the planet, about 55° beyond the sunset terminator in the Venusian evening for the JPL data and probably within 45° of the sunrise terminator for the Soviet data. The JPL data referred to an area at about 37°N. Lat.; the Soviet probe may refer to a region anywhere in the low to moderate latitudes. In spite of these variations, microwave and radar brightness studies of Venus with respect to phase angle and latitude tend to indicate that these differences may not be great. As an initial approximation these data are here correlated and integrated into one model which is suggested to be representative of a cool region of Venus.

This model (Fig. 1-3) does indicate one major feature not previously suggested. In particular, there is a probable temperature inversion in the middle to upper troposphere. Though there are significant temperature and altitude uncertainties in the data which has been used in this model, it appears to be rather difficult to eliminate completely such an inversion. As more information becomes available from the recent efforts, better control of this feature may be possible. It may also become reasonable to speculate on the mechanism causing this inversion if, indeed, it does exist. A number of processes, including absorption, convection, and phase transition associated with cloud formation may be involved.

Apart from this somewhat conjectural model, considerable data of a more definitive nature has also been accumulated. The adiabatic lapse rate may be calculated directly as a function of composition from the relationship

$$L = \frac{(\gamma - 1)gM}{R}$$

where L is the adiabatic lapse rate, g is the acceleration of gravity, M is the molecular weight, and R is the universal gas constant. Gamma (γ) is the ratio of the specific heat at constant pressure to that at constant volume; the value of γ is also expressible as a function of the number of degrees of freedom in the compositional gas molecules. For the probable range in composition of the Venusian atmosphere this implies an adiabatic lapse rate of about -10 to -15°K/km.



Fig. 1-3 'Temperature profile for tentative model of Venusian atmosphere.

(°)

The JPL experiment measured a lapse rate of -9 to -10° K/km, but the inconsistency in this may be attributed to their upper reference point lying slightly within an essentially isothermal stratosphere. Given a very high CO₂ content it does appear that it will be on the order of -10 to -12°K/km with the more extreme value of -15°K/km holding only if the CO₂ is diluted by approximately 25 percent argon.

Other minor constituents have been reported in the Venusian atmosphere. Water vapor in the amount of 0.1-0.7 percent was reported by the Soviet experiment. Earlier IR absorption studies had measured an upper limit of 250 microns of precipitable water above the cloud-tops, with 115 microns being the likely content. In addition, the presence of ice crystals in and above the cloud-tops has been strongly indicated by both an atmospheric halo effect and IR studies. An upper limit of 57 cm-atm of diatomic oxygen has been reported to exist above cloud-top. Monatomic oxygen has not been recognized, but the presence of ozone near the cloud-tops and high in the upper atmosphere has been suggested.

The existence of various hydrocarbons in the Venusian atmosphere has also been suggested. Thus far there is little evidence supporting this, though some IR absorption spectra have indicated the possibility of small amounts of formaldehyde low in the atmosphere. There have also been weak indications of CO; its abundance is thought to be on the order of 2 to 23 cm-atm.

Carbon dioxide is definitely the major component and nitrogen, neon, and argon are expected to be the prime dilutants. Of these nitrogen and argon are most favored because of cosmic abundances and mechanisms of atmospheric evolution. Very little data of any nature regarding these components is, however, presently available.

Reference was earlier made to the cloud-tops in the Venusian atmosphere. Indeed, the clouds appear to be one of the most striking elements of that planet. Their cover, though not necessarily total or uniform, is sufficient to prevent (to date) any observations of the Venusian surface within the ultraviolet, visible, or infrared spectrums. Because of the general opaqueness of these clouds, most observations of Venus have been made relative to the clouds. Infrared absorption studies are generally related to composition above the cloudtops, or occasionally a few kilometers or even a scale height into the clouds. In actuality the control on this is very poor. Cloud heights and opacity probably vary somewhat and the depth of penetration is in general a poorly known function of the wavelength being studied. As a consequence there is a considerable spread in the IR data. Furthermore, there is a large uncertainty as to how any of that data, even the norm, is actually related to the cloud tops. Nevertheless, a few

1-15

characteristic values have evolved. Most consistent has been a reported "cloudtop" temperature, usually 230-240°K. Associated pressures have ranged from 90-600 mb, but this bears further study in view of the recently revised composition estimates. Correlating these figures with the recent data, and the uncertainties therein, it seems probable that the altitude of this measured phenomenon could range from about 28 to perhaps 50 km.

There is also the problem of vertical structure and extent of the clouds. Again there is little data available, but indications are that two or three different cloud levels are possible. In addition to that level indicated above, there appears to be a thin, probably intermittent cloud level much higher in the atmosphere. The possible third layer is a low-altitude phenomena, occasionally seen through and perhaps confused with the somewhat higher layer first discussed.

Evidence for the highest layer exists primarily in the UV absorption indicated by spectral studies. This has been ascribed to possible dark ultraviolet clouds of the composition $(C_3O_2)_n$, probably at an altitude of 100-120 km. Alternate sources for this absorption may be found in possible aerosol or ozone distributions above the main cloud mass.

Any differentiation of the lower layers, if indeed there are two, is presently unrealistic, though much of the confusion may result from potential low altitude dusts and aerosols.

The composition of the main cloud mass itself is surprisingly unresolved. Ice crystals (H₂O) are quite possibly a major component. Water vapor is ruled out by absorption studies and non-gaseous phases of CO_2 , in general, seem unlikely due to the limits of their stability fields. It is conceivable, however, that sublimation of CO_2 to a solid phase could occur in vertical connective cells on the cold, dark side of Venus. Whether this dynamic state could maintain itself in metastable equilibrium is not known. The only other suggested components of these clouds have been hydrocarbon aerosols and various ill-defined dusts. None of these possibilities can presently be extended predominant support.

Surface temperature and pressure conditions were reported by the Soviet probe to be $540 \pm 10^{\circ}$ K and 20 ± 2 atm, respectively. It is believed that this temperature was measured at a cool region of the planet. How the pressure corresponds to an average surface pressure is not known since the relative altitude of the landing site is indeterminant. It is possible, though not probable, that the representative surface pressure could be greater by perhaps 6 atm. Though these pressure values are somewhat lower than most earlier calculations, the indicated temperature falls well within the expected region.
The actual distribution of surface temperature may vary considerably. The probable range is 450-700°K though extremes of 200-1000°K have been calculated from radar observations. Polar temperatures are thought to be on the order of 25 percent less than equatorial temperatures. The diurnal variation at the equator was measured to be 55°K. The uncertainty in this value is, however, large, and equatorial variations of 100-120°K have been proposed.

Stratospheric temperature was measured by the JPL flyby at $243 \pm 10^{\circ}$ K (Scale height = 5.4 ± 0.2 km). Earlier calculations produced values ranging from 180-235°K.

As for atmospheric circulation, little is known though a stable lowvelocity system has been predicted. One model with some empirical evidence suggests a lower atmospheric system of six large rotating cells positioned with radial symmetry about the sub-solar point.

The albedo of Venus is generally rather high in the region of the solar spectrum, ranging from about 0.53 in the UV to 0.87 in the red and near IR. In the visible the planet has a slight yellow-red tint which varies somewhat with phase angle.

Airglow phenomena do exist on Venus and several sources have been ascribed. Quite probable is an aurora uniformly distributed due to lack of a concentrating magnetic field. Glow discharges from the dissociation of CO_2 and other molecules are also possible. In addition, a very low-altitude nocturnal airglow associated with formaldehyde has been reported, but this remains unconfirmed.

Geodetically, there are several points of interest with respect to Venus. The JPL ephemeris presently assigns an RMS uncertainty in the position of Venus of approximately 200 km. This, however, should decrease rapidly in the future with expected refinements in the value of the astronomical unit. Knowledge of Venusian gravitational parameters is also improving rapidly with the observation of each flyby mission.

The equatorial radius of Venus has been determined by the MIT Lincoln Laboratory to be 6053 ± 3 km; this uncertainty is expected to be reduced to ± 1 km within a few years. Since the planet's diurnal period of rotation has been determined (250 ± 4 , -7 days) and the position of its rotational axis is known to lie within 8 to 10° of its orbital pole, the planet radius as a function of latitude can also be calculated with good accuracy. The uncertainty in the planet radius at any point on its surface is then predominantly a function of surface roughness and possible non-equilibrium of the crust. What little evidence exists thus far indicates that these factors may also be small and eventually predictable to, perhaps, 1 km (1 σ). A summary of much of the discussed data and information is presented in Table 1-3. It can be recognized that only an atmospheric locator can be used as a navigation or attitude reference due to the high density of the Venusian atmosphere and the opacity of its cloud cover. In determining a locator, the potential importance of Mie scattering from possible aerosols and solid dusts should not be dismissed in spite of the present deficiency of relevant data. Likewise, it can be recognized that the employment of an infrared locator may be rather problematic due to atmospheric composition, uncertainty in atmospheric stability, and the sensitivity of the IR absorbers to pressure and temperature deviations. The available data is distinctly insufficient to present a definitive model. Some reasonable indications of the navigation potential of this reference can still, however, be derived and are presented in Chapter 2.

1.3.5 <u>Mars</u>

There is considerably more agreement regarding the nature of Mars and its atmosphere than there is with respect to Venus. The low atmospheric surface density and the relative lack of clouds on Mars permit observation of the actual surface of that planet. A broad spectrum of infrared and ultraviolet absorption experiments, radar studies, and human observation have produced considerable data. More recent re-evaluations of some figures, particularly since the addition of the Mariner 4 flyby results, have produced atmospheric models of reasonable definition. Many unknowns, both atmospheric and geodetic, do, of course, remain.

The surface pressure, though once thought to be considerably higher, is now generally believed to lie in the range 4-10 mb. The tendency, largely based on the Mariner data and somewhat upon IR determinations of CO_2 content, is to accept figures in the 4-6 mb range. The corresponding range of surface density reported by Mariner 4 is 1.9 to $2.5 \times 10^{17} \text{ mol/cm}^3$; expressed as mass density this is 1.43 to $1.75 \times 10^{-5} \text{gm/cm}^3$. It is possible, however, that the occultation occurred at a somewhat elevated portion of the Martian surface. This then could make an 8 mb value more likely.

The composition of the Martian atmosphere is believed to consist primarily of carbon dioxide, probably on the order of 70-95 percent. Nearly all of the balance would be expected to consist of nitrogen, argon, and/or neon.

Several minor constituents are known to exist. Very high in the upper atmosphere both CO and O are produced by dissociation. Their abundances have been reported to be 0.2 cm-atm of CO and 250 cm-atm of O, with the latter figure being an upper limit. The O_2 content has been reported not to exceed 2 cm-atm. Water vapor has been detected in spectral analyses, and abundances

Atmospheric Molecular Weight	42.5 (+1.5, -2.5)
Surface Pressure:	
Measured Value (U.S.S.R.)	20 ± 2 atm.
Possible Range	5 to 40 atm.
Surface Temperature:	
Measured Value (U.S.S.R.)	540 ± 10 °K
Probable Range	450 to 700°K
Adiabatic Lapse Rate:	
Calculated Value (U.S.S.R.)	-10.4°K/km
Possible Range	-10 to -15°K/km
Altitude of Tropopause (probable)	$55 \pm 10 \mathrm{km}$
Stratospheric Scale Height (JPL)	$5.4 \pm 0.2 \text{ km}$
Stratospheric Temperature:	
Calculated Value	243 ± 10 °K
Probable Range	180 to 253°K
"Cloud-Top" Pressure	90 to 600 mb
"Cloud-Top" Temperature	$235 \pm 5^{\circ} K$
"Cloud-Top" Altitude	28 to 50 km
Average Near-Surface Wind Velocity	Unknown; believed to be quite low
Apparent Albedo:	
Probable Average for Visible	0.8
Range from UV to Near IR, Respectively	0,53 to 0,87
Equatorial Radius	$6053 \pm 3 \text{ km}$
Period of Rotation	250 (+4, -7) days, retrograde
Period of Revolution	225 days
Inclination of Axis to Pole of Orbit	8 to 10°
RMS Uncertainty in Ephemeris Position	~200 km
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Table 1-3. VENUS: Geophysical Data

of 10-20 microns of precipitable water with some seasonal dependence have been suggested. Only a trace amount of ozone (upper limit being 4 micron-atm) is believed to exist, its density decreasing monotonically with increased altitude.

Average surface temperatures have been determined by radar studies giving values generally in the range 150-200°K. Microwave studies indicated temperatures on the order of 215 ± 25 °K but these emissions are believed to originate slightly below the surface. Large latitudinal, seasonal, and diurnal effects are known on Mars. One study, conducted when Mars was at aphelion, indicated a sub-solar temperature of 273°K and a polar temperature of 201°K. Mariner 4 data, however, indicated a late-winter, late-afternoon temperature of only 175°K at 55°S. Lat. Furthermore, mid-winter temperatures at the poles have been calculated to drop to a minimum of slightly below 100°K. The ellipticity of the orbit of Mars can also cause an important variation, probably with an overall magnitude of about 35°K. As for the diurnal variation, sunrise temperatures lie in the range 170-210°K and a 100-150°K increase may be expected during the day.

The Martian tropopause is not a well-defined entity since the lapse rate generally undergoes such a small change. The height of the tropopause is believed to vary from about 15-30 km, being highest where the surface temperature is lowest. Mariner 4 occultation data indicated a nearly constant scale height of 8-10 km from the surface up to 30 km. This would then mean a nearly isothermal atmosphere of 175 ± 25 °K in that region. Emersion data from that same experiment is indicative of the summer and did suggest a significant tropospheric lapse rate. Calculations show that the adiabatic lapse rate for an atmosphere of the Martian composition is 5.5 ± 0.1 °K/km; the presence of local summer dust storms and their implications with respect to convective equilibrium, suggest that the actual temperature gradient may be higher. Above the tropopause the lapse rate has been calculated to be about 0.64°K/km. This continues to an altitude of approximately 100 km where the mesopause temperature minimum may be as low as 85°K.

Some difference of opinion exists here in that the Mariner 4 local winter data strongly suggest a temperature minimum of possibly 45°K at 60-75 km. Above this would be a slight increase in temperature to at least 100°K at 110-130 km.

A wide variety of dusts, clouds, aerosols, and hazes have been reported to be associated with the Martian atmosphere. Most notable are the clouds of yellow dust particularly prominent in the southern hemisphere during summer. Little is known about their frequency, but on one or two occasions they have grown to hemispheric extent. At such times the surface is obscured for ten days or so and finer elements of the cloud may remain suspended for months. It is estimated that these clouds may extend to 20 ± 10 km and consist of particulate matter as large as 10^{-2} cm in diameter. The dominant component which lingers for so long is most likely of the micron range or smaller in diameter.

In addition to the dusts, white clouds have been seen. These are rarely large and are commonly associated with the polar caps. They probably consist of solid crystals of CO_2 , water vapor, solid crystals of H_2O , or some combination of these; the first is certainly the most likely, and CO_2 crystals were tentatively detected by Mariner 4 existing in minor amounts as high as 100 km in the Martian atmosphere. For the most part, these white clouds are thought to be confined to the lower 5-10 kilometers of the atmosphere, but exceptions must be anticipated.

Many reports of a blue haze have been made, and the belief has been that this is actually an absorption effect from a possible high-altitude aerosol layer. If this exists, it is quite thin, probably in the 60-100 km region, and absorbing only in the blue end of the visible spectrum. There is some evidence, however, that the observed effects result purely from a wavelength dependence in the albedo of the Martian surface material.

Several studies of atmospheric circulation patterns have been made, but the results thus far are largely inconclusive. The existence of circulation on a planetary scale has been demonstrated by the evolution of the larger dust clouds. Likewise, the origin of these has indicated the probable occurrence of rather intense vortices and convective cells. Observation of dust clouds has demonstrated the presence of prevailing winds with velocities of 60 km/hr, sometimes reaching almost 100 km/hr. Theoretical models have indicated possible continuous surface velocities of 240 km/hr and peak values of about 600 km/hr. Evidence for these speeds, however, is thus far unfounded in actual observations of the planet.

Reference was made briefly to the albedo of Mars. The determination of the true material albedo is dependent upon resolution of the high altitude aerosol problem. The net apparent albedo has been calculated and is indicated by the following three values for different portions of the spectrum: 0.33 in the red, 0.18 in the green, 0.046 in the near UV. It is obvious that the albedo is significantly higher in the vicinity of the ice caps; an averaged value appropriate to those areas should be expected to lie in the range 0.7-0.9, depending on the possible short wavelength absorption. Geodetically, Mars is not known as well as Venus. Radar surveys by the MIT Lincoln Laboratory were made at Martian latitude $21^{\circ}N$ and indicated a radius of 3382 ± 7 km. This uncertainty is expected to be reduced to ± 2 km within a few years. Extrapolation to other latitudes is not done so readily, however. Since Mars has an appreciable rate of rotation, considerable dynamic polar flattening should be expected. Optical studies to date have had insufficient resolution to produce definitive conclusions, but there is some suggestion by comparison of these with theoretical calculations that Mars is in a state of non-equilibrium. The degree of this potential is not known, but a radius differential of several kilometers might be expected.

There is also considerable topographic relief on the Martian surface, as shown in part by the Mariner 4 television experiment. Differentials of perhaps 10-12 km must be anticipated in the extreme cases. Radar studies here indicate that total topographic plus non-equilibrium differentials may range to about 15 km.

As was the case with Venus, knowledge of the ephemeris position has an RMS uncertainty of about 200 km. Reduced uncertainty in the astronomical unit will improve this value. The Mariner 4 flyby mission also resulted in improved gravity data regarding Mars; significant additions to this information will be expected from future missions to that planet.

It would appear that either the solid edge of the Martian disc or an atmospheric locator could be used as a navigation reference. The fundamental atmospheric and geodetic data relevant to this problem is briefly reiterated in Tables 1-4 and 1-5. Consideration to such navigation references is given in Section 2. It should be recognized that the combined problems of possible (high altitude) near UV absorption, frequency and extent of dust clouds, and radius uncertainty are all presently critical; however, significant advances in each of these problems are possible.

1.4 Critical Uncertainties

There are, with respect to each of the potential reference bodies, certain dominant factors contributing to their total associated uncertainty. It is not within the scope of this report to suggest the most expedient means of reducing these uncertainties. Rather, they are presented as a motivation for further possible studies in the future.

Star attitude uncertainties appear to be negligible although improved longterm prediction of the apparent drift in this attitude may be required for extended missions. Additional information on star ranges will also be required for missions to the outermost planets of this solar system.

Carbon Dioxide	85%
Nitrogen, Argon, and Neon	15%
Water Vapor	<1%
Oxygen and Ozone	Trace
Carbon Monoxide	Trace

Table 1-4. Probable Composition of the Martian Atmosphere

Atmospheric Molecular Weight	41.9 (+2.1, -3.9)
Surface Pressure:	
Possible Range	4 to 10 mb
Probable Value	4 to 6 mb
Atmospheric Molecular Density (Surface Value)	1.9 to $2.5 \times 10^{1} \text{(mol/cm}^{3}$
Atmospheric Mass Density (Surface Value)	1.43 to 1.75 \times 10 ⁻⁵ gm/cm ³
Surface Temperature:	
Average Value	150 to 200°K
Probable Range	140 to 300°K
Magnitude of Diurnal Temperature	
Variation at Surface	100 to 150°K
Adiabatic Lapse Rate	$-5.5 \pm 0.1^{\circ} \text{K/km}$
Altitude of Tropopause (probable):	
Day	15 to 30 km
Night	~0.0 km
Stratospheric Scale Height	8 to 10 km
Stratospheric Temperature	175 ± 25 °K
Stratospheric Temperature Gradient	0.64°K/km
Mesopause Temperature Minimum	45 to 85°K
Maximum Altitude of Dust Clouds	20 ± 10 km
Near-Surface Wind Velocities:	
Sustained: Observed	60 km/hr
Theoretical	240 km/hr
Peak: Observed	100 km/hr
Theoretical	600 km/hr
Apparent Albedo:	
Red	0.33
Green	0.18
Near UV	0.046
Radius (21°N. Lat.)	3382 ± 7 km
Period of Rotation	1.03 days
Period of Revolution	687 days
Inclination of Axis to Pole of Orbit	23°59'
RMS Uncertainty in Ephemeris Position	~200 km
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Table 1-5. MARS: Geophysical Data

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As for the sun, more data regarding the stability of the photosphere, or any other selected limb locator, where associated with solar prominences will be most valuable. Both observational and theoretical studies would be applicable.

The most obvious, unresolved factor associated with Earth-referenced navigation is the stability of the infrared horizon. Considerable study must be given to cloud cover and vertical development of clouds, variations in water vapor content, and ground albedo, on all of which an IR horizon locator will be highly dependent. Though still in developmental stages, an adequate visible horizon locator and improved geodetic control grids are possible and under control with present knowledge and technology.

There is a great need for additional, reliable information regarding Venus. Its composition is relatively well known, but considerable refinement will be required. This is needed to define accurately both Rayleigh-scattering cross sections and the adiabatic lapse rate. The latter, along with measured vertical temperature profiles, will serve as an indicator of atmospheric stability. Since Rayleigh scattering is of prime interest, density profiles with accurate altitude control are necessary. Also important to these calculations are temperature variations with respect to phase angle and latitude, as well as altitude. Fundamental to any understanding of the Venusian atmosphere is some definition of cloud structure, distribution, composition, and source mechanism. Dusts, aerosols, or solid crystals may produce important Mie scattering; the abundance, composition, and distribution of such particles must also be determined. Considerable attention must be given to the presence of all IR absorbers, the pressure and temperature dependence of their absorption coefficients, and their spatial distribution and associated pressure and temperature on the planet. Carbon dioxide, water vapor, ozone, and various hydrocarbons may all be important in this respect. Finally, realistic circulation models would be useful in assessing the net stability of the Venusian atmosphere.

The needs with respect to Mars are somewhat similar in that reliable data on composition, vertical density profiles, systematic spatial variations in pressure and temperature, surface conditions, and atmospheric circulation are all needed. Both ultraviolet and infrared absorbers are known to exist in the Martian atmosphere and considerable definition of these may be required. Since the edge of the solid disc of Mars may be seen through the thin atmosphere, further information regarding clouds; haze, and atmospheric dusts which could observe that edge are most relevant. Their height, thickness, spatial distribution, and frequency of occurrence are all desired. Also important in this respect is good statistical data on surface roughness and spatial variation, systematic or otherwise, in the planet radius.

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Though other planetary bodies were not employed in the navigation studies described in this document, they should not be completely disregarded. Mercury, Jupiter, the moon, the satellites of Mars and Jupiter, the larger asteroids, and perhaps other planetary bodies may prove highly valuable in certain missions, provided that sufficient information regarding them has become available. Again atmospheric, surface, and ephemeris data are all relevant to this purpose.

It can, then, be recognized that significant improvements in the uncertainties associated with these navigational references is both possible and expected. The plan of this present study has been to use as a nominal reference, estimates derived from the best currently available data.

Sensitivity studies have been designed to test a range of phenomena uncertainties from the worst probable, as indicated by current data, to the best which could be expected with the input of future data. The nominal values and actual ranges of phenomena uncertainty considered for the several navigational and attitude references in this study are presented in Chapter 2.

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2. MODELS OF NÁVIGATION PHENOMENA

Chapter 1, preceding, has discussed the phenomena which would be used for optical navigation measurements. This chapter discusses the expected accuracy of these measurements. Section 2.1 describes several types of measurements which are useful for interplanetary navigation. The characteristic common to most optical measurements is that they specify the angular relationship between some inertial reference and some phenomenon on a nearby body such as a planet or the sun as viewed from a spacecraft. One of the error sources in the type measurement is that the location of the phenomenon is imperfectly known with respect to a useful navigation coordinate system, such as the center of the sun or planet. The most important planetary phenomena are described in Section 2.3. Since all of the near planets have optically significant atmospheres, meteorology and climatology are important aspects of phenomena error studies. Several basic atmospheric and radiation transfer equations are presented in Section 2.3.1. This topic is presented in greater detail than many of the other topics in this report in order to emphasize an important area which has often been superficially treated in the past. Mars offers the possibility of either a terrestrial or an atmospheric horizon. Venus offers only an atmospheric horizon.

Some feature of a planet's horizon must be chosen as a reference locator so that the measuring equipment can be programmed to detect it. Several such locators are discussed in Section 2.2. There are a variety of locators ranging from very simple to very sophisticated. The accuracy with which a locator can be specified with respect to a planet center and/or planet geoid, and the repeatability with which an instrument can detect it are functions of the amount of information digested by the measurement. A more sophisticated technique will produce a more accurate measurement at the expense of requiring more complicated equipment and a greater computer or astronaut workload.

A very simple horizon detection technique, the half-maximum intensity technique, is used as the basis for evaluating equipment detection errors, as discussed in Section 2.4. A few functional dependences are explained and the probable accuracy of a representative system is presented. Other types of instrument errors which produce navigation angle measurement error are described in detail in Section 4.3, 4.5 and 4.7.

The final section of this chapter presents a summary of the error model used in the computer statistical simulation of several representative space missions. The simulation is described in Volume I of this report and some of the important results are presented in Chapter 3 of this volume.

2.1 Measurement Concepts

This chapter describes several types of navigation measurements which can be made from onboard the spacecraft. The list, given below, is based upon the useful phenomena selected in Chapter 1, plus unknown landmark tracking and tracking of man-made objects such as satellites or probes. Many other possible measurements involving such things as moons and other bodies, are discussed in Chapter 1. Only those which show definite promise for planetary missions are listed below.

Navigation Measurement Types:

- 1. Planetary horizon-star
- 2. Planet diameter
- 3. Sun-star
- 4. Star occultation
- 5. Known landmark-star
- 6. Unknown landmark rate or tracking
- 7. Probes or satellite tracking

The phenomena uncertainties which contribute errors to each measurement type are described and some estimates are made of the magnitude of the resulting navigation uncertainties.

2.1.1 Star Horizon Measurement

The basic parameter in star horizon measurement is the angle between a star and a planet limb. Two separate observations define the lines of sight and an angle encoder measures the angle between them. The star line of sight provides the basic inertial reference for the measurement and the planet limb is located by a narrow-field-of-view scanning photometer. The instrument is described in detail in Chapter 4.

The photometer field of view is a rectangular slit aligned with its wide dimension parallel to the planet surface and its narrow dimension in the measurement plane defined by the star, the spacecraft and the center of the planet. Figure 2-1 depicts the measurement geometry. The optimum height of the field of view is a tradeoff between slit integration error (which is minimized by a small slit) and photosensor noise error (which is minimized by a large slit). The width of the slit is limited by planet curvature and telescope field of view.

Two types of horizon phenomena are useful for navigation – visible and near ultraviolet light from scattered sunlight and infrared radiation from thermal radiation. Both types have several advantages and disadvantages. Infrared is



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available on both the light and the dark side of a planet. The visible horizon is more accurate when it is available.

When viewed from space with an accurate instrument, the limb of a planet is not a sharp edge, but rather a profile of gradually diminishing intensity, such as shown by the horizon intensity profile in Fig. 2-1. The shape and altitude of the profile is determined by atmospheric density and the intensity is determined by a source function, which for infrared depends upon atmospheric composition, temperature, pressure and density, and for the visible profile depends upon density, sun angle and planet albedo.

A useful profile characteristic is that the shape of a profile remains constant despite changes in the source function. That is, even though the maximum intensity of the profile may vary, the altitude where the intensity is some fraction of the maximum intensity remains constant. This is not strictly true because several secondary effects tend to vary the shape when the source function changes; however, the principle is sufficiently valid to be useful.

This fact can be best utilized in a navigation system by having the instrument scan the profile from bottom to top, and as it scans it first identifies the maximum intensity and then it compares it to the tail of the profile where the intensity is, for example, half the maximum. Any fraction would do for a locator; however, the half-maximum has the advantage of being the point of maximum slope so it can be located with the greatest accuracy.

Planetary atmospheres are by no means stable and constant. The shape and altitude of horizon intensity profiles vary as a direct consequence of atmospheric density profile variations, so atmospheric perturbations constitute an important error source for horizon measurements. Section 2.2 contains a discussion of phenomenon uncertainties and a brief description of instrument detection errors. The ultimate accuracy of a system depends on the type and the sophistication of the horizon detection technique. Section 2.3 describes several detection schemes in addition to the half-maximum technique described above, and explains some of their advantages and disadvantages.

Star horizon measurements can be divided into three classes by information content: star elevation and star-planet, which are considered Type 1, and planet diameter, Type 2 (Section 2.1.2). In this report, star elevation is defined as a measurement between a star and one limb of a planet. Star-planet is similar, except that both limb crossings on opposite edges of the planet in the measurement plane are used. The direction to the planet center is calculated from their average and the measurement is the angle from the star to the center of the planet. A planet center can be located more accurately by two narrow field of view measurements on opposite edges than by a wide field measurement of the radiation center. A simple explanation of the mathematical reasons for this is that the center is defined as half way between the edges, so measuring anything but the edges only adds noise to the measurement. In the first case the direction of information is in the measurement plane perpendicular to the line of sight to the limb, while in the latter case, the direction is perpendicular to the line to the planet center.

Another distinction between these two cases is that the star-planet measurement requires that both limbs be observed, which will require an infrared sensor since both sides of a planet are seldom sunlit, while the star elevation measurement can be made by a single observation with either a visible light or an infrared sensor.

The star planet measurement is theoretically the more accurate of the two because it involves the averaging of two measurements. Uncorrelated statistical errors will be reduced because of statistical averaging. Correlated errors such as uncertainties in planetary radius or in the altitude of the navigation phenomena above the geoid, and certain detection errors will cancel each other out when observations on opposite sides of the planet are averaged. Much of this advantage is lost in the practical situation because an infrared measurement is less accurate than a visible wavelength measurement.

2.1.2 Planet Diameter

The direction of navigation information from a planet diameter measurement is the distance along a line to the center of the planet, in contrast to the two previous limb measurements. To simplify navigation measurement instrument design and function, planet diameter measurements have been constructed from two star elevation measurements. It does not appear necessary to add a second limb scanner in order to eliminate the portion of the error caused by the intermediate star reference.

A planet diameter measurement made by the RSS design model uses the information obtained by subtracting angular measurements from opposite sides of the planet. The magnitude of navigation uncertainty from this case is worse than for the star elevation and star-planet cases because the two measurements required for one planet diameter measurement are statistically independent, and most correlated measurement errors add rather than cancel as in star planet measurements. The navigation accuracy is also degraded when the angle subtended by the planet is small. Despite these handicaps, the planet diameter measurement is useful because it provides information along a direction in which data is otherwise unobtainable

2-5

Operationally and mathematically, a star-planet and a planet diameter measurement which use the same two horizon measurements are equivalent to two star elevation measurements on opposite sides of the planet (which use the same star). However, the three types have been separated in the analysis for Chapter 3 to preserve the available informatic about the relative utility of the different directions of information content.

2.1.3 Star-Sun Measurement

This measurement is similar to the star-horizon measurement except that the sun is used as the local body rather than a planet. A separate sensor or attenuating technique would have to be used since the intensity levels of the sun are many magnitudes greater than the intensities of planets. As was the case with determining a planet's center, the best way to find the sun's center is to measure opposite edges and locate the midpoint between them.

Solar limb characteristics are very different from planetary limb characteristics because of the dynamics of the solar atmosphere and the active nature of the radiation source in contrast to the passive nature of scattered light from planets.

The sun generally provides less accurate navigation information than do planets for two reasons: The sun is much farther away from the spacecraft than the nearest planet, so angular errors represent much greater linear distances. Turbulence in the solar atmosphere is much more extensive in dimension because of the greater physical size of the sun and because there is energy available for much more violent phenomena than in the case for planetary atmospheres.

2.1.4 Star Occultation Measurements

A star occultation occurs when the relative motion of the spacecraft and a planet result in the planet passing in front of a star, thereby obscuring it. The navigation information available from this measurement is that at the instant of occultation the spacecraft, the planet, and the star all lie along a straight line.

This measurement is very accurate because only one instrument line of sight is involved, thereby errors due to detecting a reference line of sight, optical axis misalignment, and angle encoding are eliminated. However, two sources of error will be present in star occultation measurements. One is an error in the altitude of the phenomena on the planet which obscures the star. In most cases this will be the atmosphere and the discussion of phenomena uncertainty in Sections 2.3.1, 2.3.2, and 2.3.3 will apply to the uncertainty in this type of measurement.

The other error source is instrument detection-time error. A locator, such as the time when the star intensity diminishes to one half its intensity with a clear line of sight, will have to be chosen to define the measurement. Detection error is the inability of the instrument to precisely locate this point. Section 2, 2.4 contains an estimate of the magnitude of this error.

2.1.5 Known Landmarks

Known landmarks would have utility primarily for Earth sightings, where their locations are sufficiently well known. The navigational usage of Earth landmarks is presently being thoroughly investigated by MIT for application in the Apollo missions.

Known landmarks will not be considered further in this document since, at present, they appear impractical for the Mars and Venus mission discussed in Chapter 3.

2.1.6 Unknown Landmark Tracking

Navigation information can be obtained by measuring the relative rate of motion of planetary features or by tracking an unknown landmark on a planet surface. The location of the points of reference is unimportant, and their latitude and longitude need never be known. The parameter measured by the navigation instrument is the inertially-referenced angular rate at which the line of sight to a landmark changes with time due to the relative motion of the spacecraft and the planet. An image motion sensor would sense the rate of relative motion. A tracker would simply acquire some point on the planet, lock its line of sight onto that point and measure the change in that line of sight relative to an inertial reference over a given period of time.

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The navigation information derived from this measurement is the ratio of the component of velocity perpendicular to the line of sight to the radial distance from the spacecraft to the landmark. This type of measurement is often called a V/h measurement because of the nature of the parameter it measures. This measurement will be most effective when the spacecraft is near a planet so that the ratio V/h is high. Also, unknown landmark trackers are most useful when near the planet since less time is required to rotate the line of sight through a sufficiently large angle. Section 4.11 gives a description of several possible equipment designs which would accomplish this measurement and derive an applicable recursive navigation technique. A representative accuracy is quoted in Section 2.4. and a comparison of this technique with other navigation measurements is given in Chapter 3.

Several other techniques are feasible which would improve the accuracy of unknown landmark measurements. The most critical restriction on the system described in detail in this report is the short measurement time restriction. Part of this restriction is an artificial constraint imposed by small-angle approximations in the mathematical algorithm used to incorporate the measurement into the navigation update by the guidance computer. The technique can be extended to large angles by expanding the navigation state vector to include coordinates for landmark position. The line of sight at the beginning of the tracking period defines the landmark location and then subsequent line of sight directions can update the navigation estimate as though they were sightings to the known landmark position thereby defined. Using larger tracking angles would improve unknown landmark measurements by minimizing the effects of quantization errors and inertial platform drifts or star tracking uncertainties.

A fundamental limit for unknown landmark tracking is the drift rate of the line of sight across the terrain. Part of this drift is due to electronic noise and memory deterioration and can not easily be reduced. However, part of it is due to changes in the scene caused by aspect angle changes and optical zoom caused by spacecraft motion. The effect of these distortions can be minimized by using a video tracker and distorting the image electronically to compensate for the geometric changes.

A video display to an astronaut could be used as an accurate unknown landmark tracker. The astronaut could scan the planet surface with a manually aimed vidicon sensor for a distinguishing feature which he would like to use as a landmark. At some later time he could realign the vidicon on the landmark, using his judgement to compensate for geometric distortions and drifts.

Human judgement can contribute to the superiority of a vidicon system over a completely automatic system in several ways. First of all, a human operator can select an optimally distinguishable feature for the initial landmark. He can use his judgement to compensate for changing geometry and he can remember a landmark and reacquire it after breaking track. This would make it possible to track several unknown landmarks simultaneously by switching back and forth from one to another, and to use the same landmark on successive orbits if the spacecraft goes into orbit about a planet.

It is worth noting that in addition to natural features, a man-made unknown landmark could be provided by soft-landing a beacon on the planet surface. This beacon would be particularly useful if a manned landing is to be attempted. In this case the beacon might be part of an unmanned exploratory probe sent to examine possible landing sites such as Surveyor has done for Apollo. After finding a suitable site, the probe could help guide the manned spacecraft in for a precision landing. A surface probe is very useful to a flyby mission. Such a probe would probably be included as part of the scientific payload of any well-organized space mission. This probe could contribute to the navigation of the spacecraft in two ways. It could serve as a radar beacon and it could measure atmospheric properties, thereby reducing the contribution of phenomenon uncertainties in the optical star-horizon measurements.

2.1.7 Beacons and Probes

A number of interesting navigation measurements can be made by tracking probes or satellites. These could be located in orbit around the planet or they could accompany the spacecraft on flyby trajectories, and they could be tracked optically with the equipment described in this report. Although not required, supplemental data could be obtained by using range measurements provided by the onboard radar.

A satellite sent into orbit about a planet also could be very useful. If it were sent early enough before the arrival of the spacecraft, its orbit could be accurately determined and the spacecraft could navigate relative to it. Such a probe equipped with a horizon scanner could contribute immensely toward accurately specifying atmospheric profiles thereby reducing the phenomenon uncertainty in the navigation measurements made by the spacecraft. This information would be of great scientific value in understanding the atmosphere of the planet. The scientific and the navigation functions of the spacecraft could cooperate for their mutual benefit.

A more subtle navigation use for probes would be to track them as they follow their own separate trajectories which take them through a different part of the planet's gravitational field. A recursive navigation scheme which uses this source of information for orbital navigation has been proposed and analyzed by Control Data Corporation.⁽¹⁾ It should be noted that by tracking the motions. relative to the spacecraft, of a "near", free-falling object that it is possible to gain information on the spacecraft trajectory without also directly observing the location of the source of the gravitational force. If the spacecraft and the tracked object are in different regions of a gravity gradient field, they will have relative motions which will yield information on the position of the center of the gravitational force. This is a particularly good source of information near a planet where there is a relatively large gravity gradient. This source of information is at its best when the spacecraft and near-body are near to the center of a gravitational field for the maximum amount of time; such as when the spacecraft is in orbit. This source of information has not been included in the error sensitivity analysis discussed in Chapter 3, due to the emphasis on flyby missions. It

^{1.} Grosch, C. B. and Lillestrand, R. L., "Autonomous Navigation Based on Ejection of Artificial Test Probes", Presentation, 17th International Astronautical Conference, Madrid, 1966.

should be noted that the navigation and guidance of probes, used during a flyby encounter, will supply information which will reduce the spacecraft navigation uncertainties. For any future studies which also emphasize orbital phases of a mission, this technique should be included so as to be evaluated against other sources of navigation information.

Recursive navigation techniques such as this one and the unknown landmark tracking techniques discussed in Chapter 2 should prove very useful for orbital planetary exploration missions for which there is a minimum of initial information on the planets' geophysical phenomena and geography.

These are but a few uses or probes and beacons. Many more are possible. Time and space limitation prohibit further discussion of this topic in this report. However, future evaluations should keep these possibilities in mind. Particular emphasis should be placed on techniques which will both provide scientific data and reduce navigational uncertainties.

2.2 Horizon Reference Locators

The half-maximum intensity horizon locator has been assumed in the majority of the situations in this report. A number of other locators are feasible which provide greater accuracy or alleviate certain specific problems with this half-maximum technique but which are more complex.

The simplest modification to the half-maximum technique is to specify some fraction other than 1/2 to define the horizon location. The discussion on Mars in Section 2.3.3 pointed out a motivation for choosing an intensity ratio less than 1/2 which occurs at higher altitudes than the half maximum. A fraction greater than 1/2 may be useful for infrared detection to move the locator to a lower altitude which would be advantageous because the lower atmosphere is less sensitive to temperature variations than the upper atmosphere. Another motivation for shifting the infrared locator altitude might be to minimize the variation in source function temperature between the altitude of maximum intensity and the locator altitude.

Wavelength selection can provide altitude flexibility because both Rayleigh scattering coefficients and infrared absorption coefficients are sensitive functions of wavelength. Most of the flexibilities listed for intensity ratio variation can be obtained by wavelength selection.

Another flexibility of wavelength selection is the fact that some disturbing factors, such as certain types of clouds and atmospheric absorbers, are transparent at some wavelengths and opaque at others. Many disturbing factors can be eliminated by proper selection of wavelength. A good illustration of this is solar flares which can contribute thousands of kilometers of error to sun limb measurements at some wavelengths, but are invisible at other wavelengths. A video display of a limb monitored by an astronaut can be a valuable aid. Such a display could identify situations where clouds or dust storms would distort the horizon profile thereby preventing the automatic system from sensing misleading data.

Multiple color measurements can provide information for rejecting invalid horizon measurements. Dust, clouds, and haze all effect profiles nominally formed by Rayleigh scattering by different amounts at different wavelengths, so comparisons of profiles at several different wavelengths will indicate whether or not any of these factors are distorting the profiles.

A graphical display of intensity versus altitude of a profile observed by a scanner can greatly improve detection accuracy. Such a display could be generated by having an onboard computer generate a display on the cathode ray tube used for the video scanner display described elsewhere in this report. The most obvious improvement such a display would produce is that it would allow the entire profile to be matched to a nominal profile which the computer could generate. The entire profile could certainly be fitted more accurately to a locator than any one point, such as the half maximum, because more data would be used; thus, more of the measurement noise and atmospheric abnormalities would average out.

A more important improvement possible from the use of such a display stems from the fact that the shape of the profile reflects the meteorological conditions in the atmosphere. Specifically, the slope of intensity versus altitude is proportional to temperature, maximum intensity is a function of sun angle, albedo, and cloud cover and the shape also suggests the presence of dust and clouds. If several profiles at different wavelengths are displayed, each one gives information about a different altitude region and provides unique information about the relative proportion of Rayleigh scattering and scattering due to clouds or dust.

A large part of the meteorological condition of the atmosphere at the horizon can be determined from analysis of intensity profiles. This fact is of considerable utility to the scientific objectives of the mission and of great benefit to the navigation system. If the meteorology of the horizon can be determined, the contribution of phenomenon uncertainty to measurement error can be reduced. Navigation accuracy is thereby improved. This scheme would be most effective if a small general-purpose computer were available to generate theoretical profiles as a function of specified meteorological conditions which the astronaut could attempt to match to the measured profiles by selecting the proper conditions. The computer could then determine the navigation angle by a least-squares curve fit to match the altitude of the measured profiles to the theoretical profiles.

Some of the meteorological factors which could be determined by profile analysis are temperature profile, pressure, molecular weight, carbon dioxide content, ozone content, dust and aerosol content, cloud altitude, albedo and, on Mars, surface pressure and terrain elevation. Although the analysis necessary to determine these factors is rather complicated, it may be worthwhile when both the scientific and navigational benefits are considered.

A mission which involves either the spacecraft or a probe entering the planetary atmosphere for either landing or atmosphere breaking will be vitally interested in the atmospheric density profile. Optical measurements and particularly the profile analysis suggested above will give very accurate estimates of the position of the spacecraft relative to the density profile. In contrast, perfect navigation from Earth-based radar would be useless for navigating a spacecraft into an entry window if uncertainty in the altitude of the density profile exceeded the tolerance in the window. Moderate temperature uncertainties can cause high-altitude density uncertainties of an order of magnitude or more, so any precision entry vehicle will require information on the planetary atmosphere as well as on its own position. Optical navigation provides the required information directly.

2.2.1 Locator Detection Instrument Uncertainties

Several instrument design parameters also affect horizon locator detection accuracy. The optimum field of view size is a tradeoff between signal-tonoise ratio which is enhanced by a large field and slit integration error which is minimized by a small field. The optimum size is a function of distance from the planet so that several slits may be required. Still, compromises must be made. Slit width-to-height ratio is determined by similar constraints plus sensitivity to misorientation and horizon curvatures are added.

Scan rates, integration time, filter bandpass, and sensor type are all very important in the design optimizations. Theoretical limits on the performance of optimum systems can be made from considerations of basic measurement information content and theoretical noise limits. The detection accuracy figures given in this report generally reflect these considerations and may be taken as representative of well-designed systems even though many of the required suboptimizations have not been presented.

There are two limiting cases for the locator detection instrument uncertainties. When the spacecraft is a great distance from the planet, the horizon will subtend an angle below the diffraction limit of the horizon scanner photometer. Under these circumstances, the angle detection uncertainties considered in Section 4.5 will apply, and the phenomena locator uncertainties are insignificant. Conversely, near the planet the instrument locator detection angle uncertainties are completely dominated by the phenomena location position uncertainties discussed in this Chapter, and the instrument-produced uncertainties are insignificant. At intermediate ranges it is assumed, for the purposes of this study, that the two sources of error can be added as described in the next Chapter.

Based on detection calculations at several intermediate ranges, this model is felt to be sufficiently representative for the purposes of this study. For any future studies, one of the immediate and major tasks is analysis and determination of optimum locator-sensor combinations for the many horizon navigation sighting situations.

2.3 Planetary Horizon Uncertainty

2.3.1 Basic Horizon Equations

The most fundamental atmospheric physical parameter for specifying the optical intensity profile is density. The most versatile way to describe an atmosphere is to give its surface pressure and its temperature profile. Density follows from this through the gas laws and hydrostatic equilibrium.

An ideal atmosphere can be divided into two regions, a troposphere and a stratosphere. The dominant heat exchange mechanism in a troposphere is convection and in the stratosphere the dominant mechanism is radiation balance between absorption of solar and terrestrial radiation and the emission of thermal radiation. An ideal troposphere is characterized by an adiabatic lapse rate. The adiabatic lapse rate is a function of fundamental parameters such as atmospheric composition and planetary gravitation. This lapse rate specifies that the slope of the troposphere temperature profile will be a fixed number of degrees per kilometer altitude regardless of the density or the surface temperature. The entire troposphere temperature profile can thus be specified by only one number, surface temperature, if the lapse rate is followed. Temperature perturbations such as diurnal, seasonal, latitude, and even local variations will equally affect all parts of an ideal troposphere.

An ideal stratosphere is characterized by uniform temperature. This temperature is a function of composition, sun angle, ground temperature and albedo. As is the case with an ideal troposphere, the entire ideal stratosphere temperature profile can be specified by one number.

The density profile can be specified to a good approximation by four variables: 1) surface pressure, 2) surface temperature, 3) troposphere lapse rate, and 4) stratosphere temperature. Surface density can be calculated by Eq. 2.3.1-(1).

$$\rho_{0} = \frac{P_{0}m}{T_{0}} 1.218 \times 10^{4}$$
2.3.1-(1)

2 - 13

where

$$\rho_0$$
 = surface density - gm/m³

 P_0 = surface pressure - atm

 $T_o = surface temperature - ^K$

m = molecular weight - gm/mole

Hydrostatic equilibrium requires that Eq. 2.3.1-(2) be satisfied.

$$d\rho = -\rho \frac{dh}{s}$$
 2.3.1-(2)

where

h = altitude

 ρ = density

s = density scale height

The scale height referred to in Eq. 2.3.1-(2) is defined in Eq. 2.3.1-(3).

$$S = RT/mg$$
 2.3.1-(3)

where

R = universal gas constant

g = gravitational acceleration

Equation 2.3.1-(2) can be integrated for the case of uniform temperature and for constant lapse rate to yield Eqs. 2.3.1-(4) and (5).

$$\rho = \rho_0 e^{-h/s}$$
 2.3.1-(4)

$$\rho(h) = \rho_0 \left(\frac{T(h)}{T_0}\right)^{mg/R\ell}$$
 2.3.1-(5)

where

T = temperature at altitude h (T = $T_0 - \ell h$)

l = lapse rate

The adiabatic lapse rate is given in Eq. 2.3.1-(6). The actual lapse rate for a real atmosphere will be less than the adiabatic rate,

al
$$\frac{\gamma - 1}{(R/mg)}$$
 2.3.1-(6)

where

 $a\ell$ = adiabatic lapse rate

 γ = adiabatic gas constant

 $(\gamma = F(composition))$

A simplified radiation transfer equation is given below. The principle assumption in its derivation is that the source function is constant along the optical path.

$$I = P(1 - e^{-\tau})$$

2, 3. 1-(7)

2.3.1-(8)

where

I = intensity

P =source function

 τ = optical path length

For infrared radiation the source function in Eq. 2.3.1-(7) is Planck's black body radiation law and for visible scattered sunlight it is the Rayleigh scattering formula. The optical path length of a line of sight tangent to some altitude, h, is given in Eq. 2.3.1-(8). This is an approximation for a spherical planet with radius much greater than the scale height and approximately uniform temperature near the tangent altitude.

$$= \sigma P \sqrt{2\pi r S}$$

where

 σ = extinction coefficient

P = density at altitude h

r = radius of planet

S = scale height (Eq. 2.3.1-(3))

τ

The density required in Eq. 2.3.1-(8) can be found from Eqs. 2.3.1-(4) and (5). The extinction coefficient is a function of wavelength and composition. The Rayleigh scattering coefficient is plotted in Fig. 2-2 for Earth, Venus, and Mars. Since the dominant component of the Venusian and Martian atmospheres is carbon dioxide, the 15μ infrared band is of great interest. A plot of its extinction coefficient is given in Fig. 2-3. The intensities of the two major energy sources, the sun and Planck black body radiation, are given in Figs. 2-4 and 2-5.

A useful parameter for evaluating the effect of atmospheric perturbations on the intensity profile is the change in altitude of a constant density. Several useful equations can be derived by differentiation of the equations listed earlier in the chapter. For a variation in surface pressure,







Fig. 2-3 Carbon dioxide absorption coefficient at 234⁰K and 1 atmosphere pressure.



2-17

$$dh_{\rho=c} = S_h \frac{dP_o}{P_o}$$
 2.3.1-(9)

For a variation of surface temperature,

$$dh_{\rho=c} = (h - S_h) \frac{dT_o}{T_o}$$
 2.3.1-(10)

If Eq. 2.3.1-(10) is used for a point in the stratosphere and the stratosphere temperature is presumed not to change when T_0 changes, then the altitude of the tropopause should be used for h. For a variation of stratosphere temperature

$$dh_{\rho=c} = (h - h_{TP}) \frac{dT_s}{T_s}$$
 2.3.1-(11)

where

 $T_s = stratosphere temperature$

 h_{TP} = height of tropopause

The optical path length of a line of sight through a given density changes when the temperature changes due to the effect of scale height change in Eq. 2.3.1-(8). The change in altitude of a given profile intensity must be modified by Eq. 2.3.1-(12) after Eq. 2.3.1-(10) and (11) have been applied.

$$dh_{I=c} = \frac{S}{2} \frac{dT}{T}$$
 2.3.1-(12)

Another useful derivative is the slope of the intensity profile derived from Eq. 2.3.1-(7).

$$dI = Pe^{-\tau} \tau \frac{dh}{s}$$
 2.3.1-(13)

Another equation of interest to infrared analysis is the sensitivity of intensity to source function temperature. This is obtained by differentiating Planck's black body radiation law.

$$\frac{dI}{I} = \frac{(C_2/\lambda)}{\frac{-C_2}{\lambda T}} \frac{dT}{T^2}$$
2.3.1-(14)

where

$$C_2 = hC/k$$

= 1.44 × 10⁻² m °K

For the 15 μ carbon dioxide band this equation reduces to:

$$\frac{d L}{I} = \frac{960}{T^2} d T$$
2.3.1-(14a)

2.3.2 Venus Model

Any analysis of Venus is handicapped by the lack of accurate knowledge about its atmosphere. The best that can be done is to try to fit a profile to the Mariner and Russian probe data. Such a temperature profile is given in Fig. 1-3. This curve, which is used as the nominal case in this section is based on Russian composition and low-altitude pressure and temperature data and Mariner high-altitude data.

The tropopause in this model is nominally located at 65 km with a temperature of 243°K. There is some data to suggest that the cloud top may be near this point.

Intensity profiles computed from the nominal atmosphere and the optical coefficients in Figs. 2-2 and 2-3 have their half-maximum intensities at the altitudes shown in Figs. 2-6 and 2-7. A few representative infrared profiles are shown in Fig. 2-8 and a few visible profiles are shown in Fig. 2-9.

Infrared profiles are a very complicated function of pressure, temperature, density and composition and this function varies widely over the range of these parameters applicable to the Venusian atmosphere. The curves presented in this report are not intended to be accurate representations of Venus' infrared signature; their purpose is to illustrate the approximate altitude and shape of the infrared horizons.

Several fundamental problems in infrared profile prediction exist in the current state of the art, which handicap use of precision infrared profiles. Present knowledge of physical properties of the upper Venusian atmosphere does not allow sufficiently accurate specification of average values and deviations from average of critical parameters. Secondly, even if physical parameters are known, the mathematical formulas for calculating infrared profiles are very cumbersome and often do not give valid results except in idealized situations. The uncertainties used for the sensitivity analysis in Chapter 3 reflect those evolved from present state of knowledge.

Altitude and shape of infrared profiles are sensitive to the factors mentioned above and the relationships are very interdependent and non-linear. This makes precomputed empirical relationships unreliable. To use infrared measurements to their full potential, a space mission would have to undertake a real-time scientific analysis of the upper atmosphere. This effort would be very expensive in terms of personnel and computer workload; however, it may be profitable when the joint benefits to both the scientific and the navigation objectives of the mission are considered. Its great sensitivity to physical parameters makes infrared radiation a powerful scientific tool. Scientific

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Fig. 2-6 Altitude of half-maximum intensity of visible horizon on Venus.



Fig. 2-7 Altitude of half-maximum intensity of infrared horizon on Venus.

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Fig. 2-8 Typical infrared profiles on Venus.



1.1

المع المعند ليس analysis of a profile can determine many of the parameters necessary to make it a useful navigation measurement.

The half-maximums of the visible profiles fall very near the cloud-top altitude and since neither cloud altitude nor density is accurately known, these points may be obscured by clouds. If this is true, it will be possible to obtain a locator safety above the clouds by using 1/4 max up to 1/10 max as the criterion for defining the horizon. Even so, the cloud top will interfere slightly with determination of the maximum intensity reference level. Aerosols or dust concentrations in the air at high altitudes can also complicate the visual profiles. The error model presented in this chapter assumes that the visible horizon is located at 65 km and that it is not complicated by clouds or dust.

The infrared profiles depicted in Figs. 2-7 and 2-8 could be complicated by several factors. Almost no temperature measurements have been made at the altitudes where the infrared profile half-max. occurs. The density profile is uncertain because of this and, more seriously, the effective temperatures at the altitudes of profile half-max. and maximum may be different, thereby making the half-max. at constant altitude principle invalid. This latter effect is more likely to be a problem with infrared than with visible because the visible profiles go from max. to half-max. in about 10 km while the infrared requires up to 40 km.

The variation of the visible and infrared horizon with temperature and surface pressure can be found by applying Eqs. 2.3.1-(9) through 2.3.1-(12). The altitude of the visible horizon was taken to be 65 km and the infrared 110 km.

$$\Delta h_{vis} = 0.05 \Delta P_{o} + 0.091 \Delta T_{o} + 0.032 \Delta T_{s}$$

$$\Delta h_{IR} = 0.05 \Delta P_{o} + 0.091 \Delta T_{o} + 0.22 \Delta T_{s}$$
2.3.2-(2)

where

 Δh = change in horizon altitude - km

 ΔP_{o} = change in surface pressure - %

 ΔT_{o} = change in surface temperature - °K

 ΔT_{c} = change in stratosphere temperature - °K

The many sources of pressure and temperature abnormalities can be classified into several categories. Table 2-1 is a listing of possible classifications. This list is not meant to be completely comprehensive, but it should serve as a basis for identifying and evaluating horizon altitude sensitivity in terms of phenomena which have physical significance in the climate of the planet.

Source	A priori uncertainty			After recursive update		
	Amount	Δh_{vis}	Δh_{IR}	Amount	Δh_{vis}	Δh_{IR}
Climatological						
Surface pressure	10%	0.5km	0.5km	10%	0,5km	0.5km
Surface temperature	50°K	4.5	4.5	14°K	1.3	1.3
Stratosphere temp.	30	1.0	6.8	9	0.3	2.2
Diurnal variation	25	3.1	7.9	6	0.7	2.0
Latitude variation	40	4.9	12.6	5	0.6	1.6
Seasonal variation	15	1.8	4.7	5	0.6	1.6
Circulation	15	1.8	4.7	5	0.6	1.6
Local surface temp.	20	1.8	1.8	15	1.4	1.4
Local strat. temp.	20	0.6	4.5	10	0.3	2.3
Strat. temp. grad.	20°K	0	6.0	2°K	0	0.6
Total 1σ		8.0	19.8		2.4	5.1
Geodetic						
Planet radius	2km	1.0	1.0	1.3km	0.7	0.7
Gravitation field	$1/10^{5}$	1.8	1.8	$1/10^{5}$	1.8	1.8
Instrument misalign- ment	1 arc sec	5.0	5.0	0.7 arc sec	3.2	3.7
Total 10		5.5	5.5		4.2	4.2
Total 10 km error		9.7km	20. 5km		4.8	6.6

Table 2-1. Venus Phenomenon Uncertainties and Horizon Altitude Variations.

Since the mechanisms of Venusian climate and weather are little known and few reliable measurements of conditions exist, it is difficult to assign meaningful values to uncertainties in a detailed model. Despite this, a detailed model is of considerable value because it provides a framework on which to analyze the effects of the details we know or can conjure concerning Venus. A detailed model is essential to evaluate the information content of recursive updates of climatic parameters using a Kalman filter while navigating past the planet.

The first item in the list is surface pressure. A 10 percent uncertainty is presumed. This uncertainty could be caused by a 1-km uncertainty in the altitude above sea level of a soft-landed measurement probe. Inaccurate pressure transducers and local weather abnormalities at the time of the measurement can also influence the measurement accuracy. The Earth's barometric pressure fluctuates up to 10 percent and Venusian weather may be more severe and it will be difficult to make an accurate measurement of the surface pressure without landing several probes and operating them for a long time.

Surface temperature is difficult to estimate because many climatic effects such as seasonal, diurnal, and latitude variations superimpose changing patterns on every location on the planet. This effect makes it difficult to separate the average temperature from the perturbations.

High altitude temperatures and temperature gradients are even more difficult to measure than surface temperature because it is harder to keep a high altitude measuring device on station than it is to maintain a ground station. High altitude temperatures usually tend to be less sensitive to local fluctuations and tend to stay closer to their equilibrium values than do low altitude temperature, so they can be estimated more accurately.

The diurnal variation is a warm area in the sunlit hemisphere and a cold area in the dark. This effect should be more pronounced on Venus than it is on either the Earth or Mars because its rotation rate is very slow; on the same order as the revolution rate around the sun. The magnitude of the effect will depend on the efficiency of the convective cycle transporting heat to the dark side. The uncertainty in this is great because the circulation pattern is unknown.

The latitude variation is a warm-at-equator, cold-at-pole effect which is prominent on the Earth and Mars and should also exist on Venus, although it may be distorted by convection heat transfer if the planetary circulation is strongly oriented symmetrically about the diurnal variation axis.

The seasonal effect on Venus should be small because of the nearly vertical pole inclination and the slow rotation rate. Therefore the northern and southern hemispheres should have nearly identical temperatures.

The circulation pattern may strongly influence the temperature distribution because a large amount of heat must be transported from the sunlit side to the dark side. Several patterns have been suggested and one of the more interesting is a six-lobe system symmetrical about the sun line.

Local variation in both surface and high altitude temperature can be expected. These would correspond to the variation we call "weather" here on Earth.

The stratosphere temperature gradient listed in Table 2-1 is a difference in black-body temperature between the altitude of the infrared intensity maximum and altitude of the half-maximum. Its effect on measurements will be to bias the altitude of the infrared half-maximum intensity.

An arbitrary, a priori assignment of uncertainties to the components we have listed is the most stringent way to evaluate the performance of a navigation system. A great deal of information is lost by disregarding geometric symmetrics and component correlations through recursive updating of climatological parameters. The total information content of the navigation technique can best be evaluated by statistically simulating a navigation exercise using Kalman filtering to recursively update not only the spacecraft position and velocity but also all the items listed in Table 2-1 plus any other factors such as planet radius and gravitational constant which affect the locator reference uncertainty. The effect of this technique is to take advantage of all the information available and to share the total intrinsic information content of the situation equally among all the components. It is invalid to assign an arbitrary large uncertainty when there is enough information in the system to reduce the original uncertainty in that component.

The righthand three columns of Table 2-1 list the residual uncertainties after a representative Venus flyby using Kalman filtering on a 42-component system. The components used were: 6 for spacecraft position and velocity, 12 for local low altitude temperature variation, 12 for high altitude variation, 2 for the planet radius, 1 each for the remaining items in Table 2-1. In the computer simulation the navigation measurement was interpreted as a relative measurement of the spacecraft with respect to the atmosphere. The information was used to update both estimates of spacecraft position and atmospheric parameters. Key factors in the updating equations are weighting factors based on relative uncertainties and the utilization of cross correlations to give the processor "memory" of what factors are a function of which other factor.

An actual flight guidance computer would not have to carry along such a large number of components. The reason for the large number here is that we chose to investigate every possible significant effect so that we could determine which components were most important. An actual flight computer would carry along only those components which a preflight analysis suggests are major uncertainty contributors.

The flyby trajectory and the instrumentation used to prepare Table 2-1 are representative of typical missions, but the results should not be taken as limiting values or averages or as anything more than a representative case because the results can be expected to vary widely under different circumstances. Tentative results of this case suggest that almost all of the a priori estimates could be improved upon. The notable exceptions are the local temperature variations. A result not suggested by Table 2-1 was that most of the final navigation position and velocity error uncertainty was contributed by instrument axis misalignment and gravitational uncertainty. Diurnal, latitude, seasonal, and circulation variations contributed lesser amounts and the other contributions were comparatively insignificant.

2.3.3 Mars Error Model

The terrestrial surface of Mars will be visible at wavelengths longer than 3500Å, so it can be used as a horizon locator. However the surface roughness will limit the accuracy of the measurement. The anticipated Mars radius variations from Chapter 1 would contribute reference certainties of $\simeq 2$ km, 1 σ . This locator uncertainty is less than that expected from atmospheric phenomenon locators; however it may at times be obscured by dust, haze and clouds.

The Ames 1A atmosphere was chosen as the basis for evaluating atmospheric horizon measurements on Mars. The basic characteristics of that atmosphere are shown in Table 1-5. Using the optical coefficients given in Figs. 2-2 and 2-3, the half maximum intensity altitudes for visible and 15μ infrared horizon are shown in Figs. 2-10 and 2-11. Typical horizon density profiles are shown in Fig. 2-12.

The visible profiles lie very close to the planet's surface. This causes several problems in using them. The lower part of the curve intersects the ground before it reaches maximum intensity. Because of this, the scanner will have to look at the ground to get an estimate of the maximum intensity. This technique is very susceptible to errors introduced by non-uniform albedo.

Figure 2-12 illustrates the sensitivity of Martian visible profiles to albedo variations.

The Martian atmosphere is rich with intermittent low-altitude dust storms and haze layers, particularly in the polar regions and cloud formations which may range up to 100 km in altitude. All of these would either obscure or contaminate the Rayleigh scattering profiles.

The best hope for a visible atmosphere locator on Mars is to use the 1/8-max or some such fraction which will occur at 15 km above the surface, thereby escaping the low altitude dust storms. The operation of the scanner should be monitored so that a human operator can determine whether or not haze, clouds or dust is interfering with the profile. Either a television display of the limb being viewed or a graphic plot of intensity versus altitude would be required for this function.



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Fig. 2-10 Altitude of visible horizon on Mars.



Fig. 2-11 Altitude of half-maximum intensity infrared horizon on Mars.



Fig. 2-12 Martian visible horizon profiles

Further improvement in the scanner could be made by using several scans made simultaneously but at different wavelengths. The optical properties of Rayleigh scattering, ground albedo, dust and clouds are a function of wavelength. The function is different for each mechanism so the different effects can be separated by comparing several different wavelengths.

The 15μ infrared profiles are at a reasonably high altitude and should not be subject to interference from any other known atmospheric phenomenon. Many other infrared bands should be available from the ground or the clouds or dust and each presents its own advantages and disadvantages, which will not be discussed here. The 15μ band is presumed to be representative.

The equations for the temperature and pressure dependence of the Martian horizons are given below:

$$\Delta h_{vis} = 0.08 \Delta P_{o} + 0.003 \Delta T_{o} + 0.053 \Delta T_{s}$$

$$\Delta h_{IR} = 0.08 \Delta P_{o} + 0.003 \Delta T_{o} + 0.51 \Delta T_{s}$$
2.3.3-(1)
2.3.3-(2)

where

h = horizon altitude uncertainty - km

 ΔP_{o} = surface pressure (percent)

 ΔT_{o} = surface temperature - °K

 ΔT_{c} = stratosphere temperature - °K

Table 2-2 lists possible error sources for Martian horizon measurements. All the statements about recursive updating made in the section on Venus hold equally true for Mars. A very few updates would reduce the stratosphere temperature uncertainty and the bias temperature which are the largest contributors to infrared uncertainty.

2. 4 Representative Navigation Measurement Errors

A computer statistical simulation of navigation accuracy, based on the statistical errors summarized in this section, is described in Volume I of this report. Some of the results are presented in Chapter 3 of this volume. The measurement error model (Model #1) used in this analysis is presented here. Two factors make the preparation of planetary phenomenon uncertainties for this model difficult. The planetary phenomena are not accurately known, so specifications of uncertainties are themselves subject to uncertainty. A nominal value and a range are given for the phenomenon errors. This range indicates the bounds of the uncertainty based upon our present knowledge.

		Horizon Altitude Uncertainty	
Uncertainty Source	Amount of Uncertainty	Visible	Infrared
Surface Pressure	10%	0.8km	0. 8km
Surface Temperature	15°K	0	0
Stratosphere Temperature	20°K	1,0	10.2
Diurnal Variation	5	0.3	2.5
Latitude Variation	10	0.5	5.1
Seasonal Variation	10	0.5	5.1
Circulation Pattern	5	0.3	2.5
Local Weather	5	0.3	2.5
Infrared Bias Temperature	5	0	7.5
Non-uniform Albedo	$0.15(0.07^{*})$	4. 0(2 ^{**})	0
Total RMS		4.3(2.5*)	15(5**)

Table 2-2. Martian Atmospheric Uncertainties and Horizon Altitude Variation.

*Using multiple wavelengths

"Using recursive updating and profile analysis

The second difficulty in specifying uncertainties is that Model 1 accepts only uncorrelated, zero mean, lumped-together measurement errors. A second model was developed for horizon measurements which included 36 separate, correlated measurement errors in addition to the detection and instrument errors which can legitimately be considered zero mean, uncorrelated errors. The results of Model 1 simulations agree very well with the results of the more complex second model for Venus flyby navigation so the navigation sensitivity analysis is probably as valid as possible with the present state of planetary science knowledge. The phenomenon uncertainties used in Venus are in agreement with the complex model described in Section 2.3.2. The Martian uncertainties are typical of what could be expected using the terrestrial edge of the solid body as a navigation reference. The Type 2 atmospheric phenomena model is not applicable to the terrestrial limb, but there may be known geological correlations which could be profitably incorporated incorporated into the Martian terrestrial limb navigation model. A Type 2 model similar to that used for Venus could be applied to the atmospheric horizon of Mars, as discussed previously.

The navigation phenomena uncertainties used for the Type 1 model error sensitivity analysis of Chapter 3 are listed in Table 2-3.

Source	Nominal Error	Range of Error	
Instrument errors Scanning photometer Star occultometer Unknown landmark tracker	3 arc seconds negligible 5 arc sec per sec	1 arc sec to 10 arc sec none none	
Phenomenon errors Venus - Visible - Infrared Mars - Visible - Infrared Earth - Visible - Infrared Sun Unknown landmark	5 km 7.5 km 2.0 km 4.0 km 0.5 km 1.5 km 7500 km none	2.5 km-7.5 km 3 km-12 km 1 km-3 km 2 km-8 km 0.25 km-1 km 1 km-3 km 100 km-7500 km none	

Table 2-3. Summary of 10 Measurement Errors Used in Navigation Error Sensitivity Analysis.

The measurement errors were divided into two categories, phenomenon and instrument. Phenomenon uncertainties are expressed in linear measurement (kilometers) and are assumed to remain constant regardless of range. These phenomena errors are specified for one-half max horizon measurements. Star occultation locator uncertainties are assumed to be the same as the visible horizon uncertainty. The instrument measurement uncertainty for horizon measurements is expressed in angular units (arc sec) so the linear magnitude of this error is a function of range. A three arc second sextant was assumed to be the nominal design, and the range of accuracies from 1 to 10 arc sec was explored so that the sextant accuracy tradeoff could be evaluated. The anticipated uncertainties are discussed in more detail in Sections 4.3, 4.5 and 4.7. Table 2-3 summarizes this data, which was used for both the Type 1 and 2 error models studied in Chapter 3. It should be noted that the 30-arc second minimum uncertainty for IR detection was not included in the Type 1 error model used in the reported error sensitivity analysis discussed in Chapter 3, but was used in the Type 2 model. Actually, IR is of little value until the spacecraft is relatively close to a planet. When the spacecraft is in close proximity, the 30-arc second sensitivity limitation will be dominated by the phenomena uncertainty.

A number of physical equipment considerations resulted in the following geometric restrictions:

Scattered light - no star tracker or scanner line of sight within 20 degrees of the sun was allowed and no star within 10 degrees of a bright planet was used.

Maximum angle - the maximum allowable angle between a star and a navigation target for a valid measurement opportunity was 120 degrees. (This constraint was almost never exercised due to the general availability of stars much closer to the planet.)

Star occultation - the occultation must occur beyond 15, 25, or 5 degrees of latitude on the night side of the terminator for Earth, Venus, and Mars respectively and the star must approach the horizon at an angle of no less than 15 degrees to the horizontal.

Sun-lit planet - both visible wavelength horizon and unknown landmark sightings must occur on sunlit portions of a planet.

Slant angle to landmarks - The maximum angle of the line of sight to a landmark must be less than some designated angle in order to insure visibility through the atmosphere. The constraints are: Earth, ≤ 0 ; Mars, ≤ 0 ; Venus, (no landmarks visible).

Equation 2. 4-(1) was used to compute the linear measurement error for all horizon measurements.

 $\sigma^2 = (\sigma_{\text{inst}} \mathbf{r})^2 + \sigma_{\text{phen}}^2$

2.4-(1)

where

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 σ = measurement uncertainty

 σ_{inst} = instrument angular uncertainty

r = range from S/C to horizon

 $\sigma_{\rm phen}$ = the appropriate phenomenon uncertainty

The final measurement uncertainty for star elevation measurements was computed directly from Eq. 2.4-(1). Since star planet measurements require the averaging of two measurements made at opposite sides of the planet, an additional equation is required. Equation 2.4-(2), which is derived for the resulting uncertainty of two independent measurements averaged together, was used for this purpose.



2.4-(2)

where

σ_{sp} = uncertainty for star planet measurements

 $\sigma_{\rm A}$) horizon measurement uncertainties computed by Eq. 2.4-(2)

 $\sigma_{\rm B}$ for the appropriate phenomenon

Planet diameter measurements involve the subtraction of two measurements. Equation 2.4-(3) was derived to describe this case. The derivation assumes that two independent measurements are subtracted.

$$\sigma_{\rm PD}^2 = \sigma_{\rm A}^2 + \sigma_{\rm B}^2 \qquad 2.4-(3)$$

where

 σ_{PD} = uncertainties for planet diameter measurements

 σ_A , σ_B = (defined in Eq. 2.4-(2)

These equations do not completely represent the correlation which exists between the many sources of measurement error; however they do express the principle differences between the three types of horizon measurements. Star planet measurements are more accurate than simple star elevation measurements because averaging two observations should reduce uncorrelated errors by one half and many of the correlated errors, such as those in planet radius and symmetrical temperature variations cancel out when opposite limbs are averaged. Planet diameter measurements are degraded somewhat in accuracy because the RSS design model procedure used to measure planet diameter necessitated the inherent subtraction of two angular measurements and because phenomenon variations at opposite sides of the planet are usually symmetrically correlated.

The unknown landmark observation measures the rate of change of a line of sight. A simplified error model derived in Section 4.1.1 was used for the studies presented in Chapter 3.

Many additional refinements discussed elsewhere in this report which would improve the accuracy attainable from unknown landmarks were not represented in the model.

Star occultation measurements do not involve an angle measurement and thus do not have an associated angle measurement error. The speed and accuracy of occultation locator detection is the primary source of instrument error, but for the planetary mission occultation rates and for the star intensities assumed, the video tracker described in Section 4.5 will produce an insignificant amount of error in comparison to the phenomenon location uncertainty. Thus, only the phenomena linear unit uncertainties are applicable, so

$\sigma_{so} = \sigma_{phenomena}$

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so = star occultation

A good assumption for a model of this sophistication level is that the star occultation uncertainty will be equal to the visible horizon phenomena uncertainty for that planet.

3. NAVIGATION ERROR SENSITIVITY ANALYSIS

Chapters 1, 2, and 3 discuss the phenomena and hardware studies which form the basis for the study of navigation performance during simulated missions. Volume I of \hat{R} -600 has presented the concept and formulation used for these studies.

This chapter presents some of the results of a parametric study of several planetary mission phases. Some of the results are also presented in Volume I. The parameters investigated were: navigation instrument angular measurement accuracies, classes of navigation measurement types, phenomena uncertainties, and measurement schedule.

There are several probable mission phases which serve as motivation for different types of navigation system performance and navigation accuracies. Some of these are:

- 1. Interplanetary guidance
- 2. Aerodynamic braking
- 3. Insertion into planetary orbit
- 4. Probe navigation and guidance
- 5. Atmospheric entry
- 6. Vehicle landing

The missions studied are described in Section 1.1. Time permitted the investigation of only a few specific mission phases, but the analytic tools are operational and a minimum of additional effort is required to complete these studies or to investigate other missions and mission phases.

There are many combinations of optimization criterion which could be used in the analysis of the mission phases. It is necessary to balance the desire for the highest quality attainment of mission phase objectives against the desire for minimum velocity corrections (Δv), power consumption, system weight, constraints and requirements on other spacecraft systems and functions and for minimum probability of failure. An in-depth study of optimization criteria was not investigated during this study due to time limitations and due to limitations on the then existing state of definition of representative planetary missions. The optimization criterion used was the achievement of maximum navigation accuracy at critical points of a mission (such as at the planet and at exit from the planet sphere of influence), assuming systems and precedures which were considered to be practical. The results of two computer simulation models are presented and discussed. The first model is described in detail in Volume I, Section 1.2. It is a statistical simulation in which velocity correction, IMU measurements and guidance as well as navigation measurements are considered. The navigation model assumes that all instrument errors and phenomena uncertainties are zero mean. Gaussian distribution with the sigma values is listed in Section 2.4. Data is presented for five mission phases: An Earth-to-Mars leg, a Mars flyby leg of a Mars flyby mission, a Mars orbital phase, an Earth-to-Venus leg and a Venus flyby leg of a triple planet flyby mission.

The second computer simulation model emphasizes the individual components of phenomenon uncertainty for the Venus flyby case. The components and their initial individual uncertainties are described in Section 2.3.2. The objective of this program is to treat correlations in some phenomenon uncertainties, such as climatic temperature variations, and uncertainties in basic planetary parameters such as radius and surface pressure, more realistically than in the first model. An additional difference is that the second program measures and updates the estimate of several phenomenon parameters in addition to the spacecraft trajectory parameters.

The application of this model was limited to the first Venus flyby of the triple planet mission. Interplanetary navigation is analyzed in Section 3.1. Flyby results from model #1 are presented in Section 3.2 and results from Flyby model #2 in Section 3.3. Orbital navigation and unknown landmark tracking are discussed in Section 3.4. Recommendations for future efforts are summarized in Section 3.5.

3.1 Interplanetary Navigation

The two interplanetary trajectories studied are indicated in Section 1.1. For each trajectory a nominal case was prescribed and this and several perturbations of the navigation variables were run in the computer simulation to obtain parametric sensitivities of navigation accuracy. The error models, uncertainties, and physical constraints on navigation sighting availability listed in Section 2.4 were used in these simulations.

The nominal measurement schedule called for one measurement every day for the duration of the mission except when the spacecraft was within two days of the sphere of influence of either the Earth or the destination planet, in which case one measurement was made every 2.4 hours. At each measurement opportunity the measurement type and option was evaluated and the one which maximized an evaluation criterion was selected. The evaluation criterion nominally used was the ratio of improvement in position uncertainty projected to the end of the mission phase. The measurement options were star-planet, star elevation, planet diameter and sun-star. The simulation had the option of making each of these measurements in the plane containing any of the 37 navigation stars. Figure 3-1 illustrates the time history of the preferred type of measurements during the first interplanetary legs of both missions for the nominal cases. The nominal cases use a 3-arc second instrument, and nominal phenomena uncertainties.

It should be noted from the timeline graphs that the system prefers to skip around in its measurement situation. In addition to the variations in preferred planetary measurement types, shown in the figures, the system selected a wide variety of stars. Measurements using the same star were almost never repeated in sequence indicating that information in a variety of measurement planes is desirable.

Figure 3-2 illustrates the uncertainty in spacecraft location on the first Venus leg as a function of time for three sextant accuracies assuming nominal phenomena uncertainties. It should be noted that navigation uncertainty is less in the vicinity of Venus than in the vicinity of Earth, due to the manner of choice of navigation sightings. The program chooses the sightings which will minimize the uncertainty in spacecraft location upon arrival at Venus. If more accuracy is desired in the vicinity of Earth on the outbound leg, this can be readily achieved by proper modification of the sighting selection optimization criterion. Figure 3-3 illustrates the time history of the decreases in leg termination uncertainty as a function of time for three sextant accuracies. Figure 3-3 differs from Fig. 3-2 in that the variable plotted on the ordinate is not the position error at the current spacecraft location, but rather the projection to the end of the interplanetary leg of the position which would exist when the spacecraft reaches the planet if no more navigation measurements were made.

The times and magnitude of velocity corrections are also shown for the 3-arc sec sextant accuracy case. The criterion for making a velocity correction is that the ratio of the estimated required velocity correction to the uncertainty in that estimate exceed a given value, and that the ratio of required magnitude to the implementation uncertainty exceed a given value. This value was held the same throughout all three of the runs presented. This criterion is not necessarily optimum for minimizing the total velocity correction; however, it should illustrate the important trends. Figure 3-4 illustrates the uncertainty in spacecraft velocity during the first Venus interplanetary leg as a function of time for three sextant accuracies, assuming nominal phenomena uncertainties.

Figures 3-5, 3-6, and 3-7 respectively illustrate similar quantities for the first leg of the Mars interplanetary leg.



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Fig. 3-1 Preferred measurement timeline for two interplanetary trajectories.



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Fig. 3-6 Time history of Earth-to-Mars position uncertainty projected to Mars encounter.



For the Mars leg case the algorithm would like to obtain information from either the sun, or from Venus which lies in about the same direction. These alternatives are not reflected in Fig. 3-1 nor in the data shown in Figs. 3-5 through 3-7. The magnitude of the sextant and phenomenon uncertainties strongly influence whether or not these additional measurements are a significant contribution to the navigation. Since these bodies are very far away, a sextant accuracy of 3-arc sec or better is needed to make these measurements productive. For the sun to be useful, phenomenon uncertainty would have to be reduced below the maximum anticipated value of 75,000 km assumed for this run. The possibility of achieving 100 km phenomena locator location uncertainty by sensing selective wavelengths is indicated in Chapter 1.

The most obvious difference between the two cases is that the navigation at the end of the Venus leg is about seven times more accurate than that at the end of the Mars leg. The principle reason for this is the geometric difference between the two trajectories. At launch the Venus probe lies behind the Earth (relative to the sun), while the Mars probe moves ahead. The difference in these relative motions provides the Venus probe with more information through more diverse angular measurements than the Mars probe. This is illustrated by the closer spacing of velocity corrections at the beginning and the end of the Venusian trajectory than of the Martian. The geometric relationship on the approach to Venus is also more informative than the Martian approach, as illustrated by the steepness of the time history curve and the frequency of the velocity corrections.

A few conclusions on these interplanetary navigation cases are given below.

a. <u>Navigation Accuracy.</u> One feature to note in the data shown in Figs. 3-3 and 3-6 is the rapid fall in projected uncertainty during the period the spacecraft is near the two spheres of influence, which illustrates the fairly obvious fact that most of the navigation information is available when the spacecraft is near the planets. Also noteworthy is that accuracies produced at the planet by the onboard navigation measurements are essentially independent of ephemeris and astronomical unit (AU) uncertainties since the measurements are taken directly from the planet.

<u>Trajectory Geometry</u>. One of the two interplanetary legs considered in this study had terminal errors seven times smaller than another due to differences in trajectory geometry. It would be hazardous to extrapolate the results presented here to

different missions because each type involves unique geometric relationships.

c. <u>Sextant Accuracy</u>. During midcourse a nearly linear relationship exists between sextant accuracy and navigation accuracy for the cases studied thus far.

One thing to note is that one degree of freedom is largely constrained by sighting to a distant body other than the target planet and thus even when the spacecraft is near the sphere of influence, the navigation accuracy is still dependent upon the sextant accuracy. When the spacecraft is well within the sphere of influence (as discussed in the next section) other measurements to the target planet can constrain this third degree of freedom.

- d. <u>Phenomena Uncertainties</u> within the anticipated range discussed have an insignificant effect upon interplanetary navigation accuracies between the two spheres of influence for sextant errors greater than 1-arc second. The considered sextant accuracies thus dominate the navigation uncertainties during this portion of a mission.
- e. <u>Correlated Instrument Measurement Errors and Measurement</u> <u>Frequency</u>. If measurement errors are uncorrelated, an improvement in interplanetary navigation accuracy can be obtained by making more frequent measurements. There would be a limitation on the improvements that could be achieved by making more measurements if such things as angle encoder non-linearities, mechanical misalignments or detection biases, introduce correlated errors. The limitations due to these instrument associated error correlation factors have not been evaluated in detail. However, the present measurement schedule is thought to be conservative in this respect and it is felt that improvements in navigation accuracy are possible simply by increasing the number of measurements. (Another possible improvement is to optimize the measurement schedule without increasing the number of measurements.)
- f. <u>Velocity Corrections</u>. The Δv required to achieve a given interplanetary guidance accuracy (as allowed by the navigation accuracy) is approximately proportional to the sextant accuracy for the cases studied. More accurate sextant measurements

enable earlier and more accurate Δv 's and thus a considerably reduced total Δv .

3.2 Flyby Navigation

Two flyby situations were studied in detail. They were the Mars and the Venus flybys which form the next phases of the two interplanetary trajectories described earlier. The Venus flyby has its pericenter 292 miles above the planet surface while the Mars flyby altitude is 562 miles. On the Venus flyby the space-craft does not enter the shadow of the planet while the low-altitude portion of the Mars flyby trajectory does enter the planet's shadow. Parts of these trajectories are shown in Figs. 3-8 and 3-9.

Figures 3-10 through 3-18 illustrate some of the results of a parametric study. The parameters investigated were: sextant accuracy, phenomena uncertainty, infrared measurement utility, and measurement utility. The range of uncertainties used are those listed in Table 2-3. The nominal measurement frequency provided a measurement opportunity every 6 minutes while within 40 minutes of pericenter and every 20 minutes otherwise. The fast measurement schedule provided measurements every 2 minutes while near pericenter and the slow schedule, every 20 minutes throughout.

A timeline of the preferred measurement types for both flyby cases is shown in Fig. 3-10. Star elevation, planet-star, and planet diameter measurements are the three types which are most frequently chosen. Unknown landmark measurements were allowed to be an option only for Mars, and were chosen only when the spacecraft was very close to pericenter. Neither star occultation nor probe tracking measurements were included in the evaluation of these two flyby cases due to project time limitations, but it is anticipated that they would provide useful near-planet measurements and would be included in any future study.

It is worth commenting that a large number of navigation angle measurement planes were utilized for both interplanetary and flyby legs. This would cause the use of an Apollo type instrument with only 2 degrees of attitude isolation to be impractical due to the demands upon spacecraft attitude control.

Figures 3-11 and 3-15 illustrate the spacecraft position uncertainties during the Venus and Mars flybys respectively, time as a function of time for three levels of phenomena location uncertainty. These plots assume a nominal 3-arc second sextant accuracy and include IR measurements.

When analyzing the data in Figs. 3-10 through 3-18 it should be recognized that the navigation measurement selection was optimized to provide the



Fig. 3-8 Venus flyby trajectory.





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Fig. 3-10 Preferred measurement types during planetary flyby.



Fig. 3-11 Position uncertainty (1 RMS) vs. elapsed time from sphere of influence, triple planet flyby, Venus flyby #1.



Fig. 3-12 Phenomena survey - time history of Venus flyby navigation uncertainties projected to exit from sphere of influence.

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Fig. 3-13 Sextant survey - Time history of Venus flyby navigation uncertainties projected to exit from sphere of influence.



Fig. 3-14 Velocity uncertainty (1σ RMS) vs. elapsed time from sphere of influence, triple planet flyby, Venus flyby #1.





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Fig. 3-18 Velocity uncertainty (1s RMS) Mars flyby vs. elapsed time from sphere of influence.

maximum information on spacecraft position at flyby exit from the sphere of influence. This is a specialized objective, and the results are not directly applicable to the evaluation of navigation accuracy upon entering the sphere of influence, navigation accuracy at pericenter, probe navigation accuracy, Earth reentry navigation accuracy, etc. By appropriate optimization of the measurement selection and schedule it is possible to achieve better navigation accuracies for these other objectives.

Figures 3-12 and 3-16 illustrate, for the Venus and Mars cases respectively, the time history of the spacecraft position uncertainty projected forward to the time of spacecraft exit from the planetary sphere of influence. This is plotted for the two extremes of phenomena uncertainties and assumes a nominal 3-arc second sextant and includes IR measurements. (It should be noted that this uncertainty is a different variable than the one plotted in Figs. 3-11 and 3-15. This uncertainty is that which would exist when leaving the sphere of influence if no more measurements were taken out to the sphere of influence.) The slope of these curves illustrates where the most information is obtained. The divergence of these curves illustrates that the phenomena uncertainties are most significant when the spacecraft is near the planet.

Figures 3-13 and 3-17 illustrate the time history of the spacecraft position uncertainty extrapolated to the time the spacecraft leaves the sphere of influence. This is plotted for three sextant accuracies assuming nominal phenomena uncertainties and including IR measurements. One curve is shown on each plot which excludes IR measurements and assumes the nominal 3-arc second sextant.

Figures 3-14 and 3-18 illustrate for the Venus and Mars cases respectively, the uncertainty in spacecraft velocity as a function of time for three levels of phenomena location uncertainty, assuming a 3-arc second sextant and including IR measurements.

A few conclusions on these flyby navigation cases are listed below.

a. <u>Navigation Accuracy</u>. The nominal optical navigation system can navigate a flyby with an uncertainty of 3 to 5 miles in position, and an uncertainty of 0.33 ft/sec in velocity at exit from sphere of influence for both the Mars and Venus flyby cases. It should be noted that the ~7-times larger initial uncertainty at the Mars sphere of influence (vs. at the Venus sphere of influence) which remains after the interplanetary leg does not significantly affect the uncertainties near the planet.

- b. Infrared Measurements. The Mars flyby case demonstrated a particular sensitivity to whether or not infrared measurements are included. This is due to the fact that a significant portion of the last half of the flyby trajectory lies in the shadow of the planet thus preventing visible horizon measurements. The Venus flyby case was little affected by the exclusion of infrared measurements. This potential advantage of infrared measurements is questionable because it will be very difficult to design with available technology an infrared detector which will not significantly limit the navigation accuracy achievable with the 3-arc sec sextant. (A probable accuracy limit for an infrared detector is about 30-arc seconds which would somewhat affect its navigational utility.) In view of this it may be better to rely on star occultation for navigation measurements while in a planetary shadow, since star occultation measurements are potentially as accurate as visible horizon measurements and can be made by using the star tracker required for other navigation measurements.
- c. <u>Sextant Accuracy</u>. With the anticipated phenomena uncertainties, very little improvement in flyby navigation can be obtained by improving the sextant accuracy beyond 3-arc sec. The navigation accuracy produced by a 10-arc sec sextant is slightly worse than those produced by a 3-arc sec sextant. However, the sensitivity is much less than during the interplanetary mission phases in which there is a nearly linear relationship between sextant and navigation uncertainties.
- d. <u>Phenomenon Uncertainty</u>. Flyby trajectories pass sufficiently close to a planet that with the nominal 3-arc second sextant performance, phenomenon uncertainties are the limiting factors in determining navigation accuracy. During a flyby portion of a mission for a given schedule of measurement implementation, there is a nearly linear relationship between phenomena and navigation uncertainties. Figures 3-11 and 3-14 do not, however, illustrate the expected differential between the curves for nominal and minimum phenomena uncertainty since they do reflect different measurement selections and implementation schedules.
- e. <u>Correlated Phenomena Measurement Errors</u>. Great care should be taken when incorporating these measurements into the

navigation equations to account for correlated phenomenon variations, since many of the uncertainty sources are large-scale climatic atmosphere variations, or errors in basic planetary parameters such as surface pressure or radius. Section 3.3 contains a more comprehensive treatment of the effects of phenomenon uncertainties.

f. Measurement Frequency variations within the range considered have a significant effect upon navigation uncertainties for the assumed measurement error models. Data on measurement frequency effects upon navigation accuracies are presented in Volume I. For this model, which assumes Gaussian errors with zero means, unlimited improvements in navigation accuracy can be achieved by any increase in the measurement frequency. In reality this is not valid for very high measurement frequencies due to correlation effects. Within the range measurement frequencies considered in these runs, sighting correlation effects are not, however, considered to be a difficulty. The next section, 3.3, presents the results of a more realistic correlated phenomena error model which indicates a lesser effect upon navigation accuracy. It is worth commenting that the range of measurement frequencies considered at flyby would be very demanding upon an astronaut if a manual instrument was used. It is thus felt more practical to use an automatic instrument based upon electro-optical sensors.

3.3 Phenomenon Uncertainty Effects

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This section presents results of a study of the effects of specific measurement error uncertainties on navigation accuracy based upon a more realistic correlated measurement model. The computer program for this study performed a statistical simulation of the same Venus flyby trajectory described in previous sections. A 42-element state vector was updated by each measurement using Kalman filtering. The elements in the state vector were allocated to navigation variables in the following distribution: 6 for spacecraft position and velocity; 3 for planet geophysical properties, namely gravitational attraction, equatorial radius and polar radius; 2 for instrument axis misalignment and infrared detector bias; 3 for gross atmospheric parameters, surface pressure, surface temperature and stratosphere temperature; 4 for climatic temperature variations, namely diurnal, latitude, seasonal and circulation; 12 for local low-altitude temperature variations and 12 for high-altitude variations. (See Section 2. 2. 2 for a description of atmospheric phenomenon.) The nominal measurement schedule called for an observation every 5 minutes while within 2 hours of pericenter and every hour elsewhere. A 3-arc second sextant was used. (A 30-arc second angular measurement was assumed for the IR measurement as previously discussed. As with the first model, star elevation, planet-star and planet diameter measurements were used. Due to project time limitations, star occultation and unknown landmark measurement options were not included in the investigation.) Both visual and infrared horizontype measurements were utilized. Table 3-1 lists the nominal uncertainties in the navigation variables described above. To conserve space, only a representative selection of the local variations is listed. The first data column lists the initial navigation and phenomenon uncertainties. The spacecraft navigation uncertainties are representative of the "nominal" uncertainties that would exist at the end of the Earth-Venus leg of the mission (see Section 2.1). The values of the phenomena uncertainties are presented in Section 2.5 and are felt to be representative of the presently available, or soon to be available, data.

Table 3.1 also presents data describing improvements in the state of knowledge at pericenter and at exit from the sphere of influence, 612,000 km from the planet. (Note that distances in this section are denoted in kilometers and not miles, as in the preceding section.) Velocities are expressed in ft/sec and temperature in $^{\circ}$ K as has been the previous convention.

The second and fifth columns of Table 3-1 list the improved navigation and phenomena uncertainty estimates which can be made by processing the navigation measurements in this correlated manner. The position and velocity sensitivity columns list the error that would be introduced into the navigation estimate if any one of the true phenomenon parameters were different from the computed values by their 1 σ uncertainties. The numbers listed are the RMS totals of the uncertainties along three orthogonal axes.

In addition to this nominal case, several computer runs were made with a faster measurement schedule and without infrared measurements. Each of these cases showed less than 10-percent variation in navigation accuracy from the nominal case. All three cases were able to estimate phenomenon parameters to approximately the same accuracy, with the exception that the no infrared case was less effective in estimating high altitude temperatures.

It should be noted that the measurement selection criterion employed in this model was not the same as that used for the studies described in Section 3.2. Here the selected measurement was chosen so as to provide the maximum information content regarding the current spacecraft state vector. Consequently, the results cannot be directly correlated with those presented earlier. Important,

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		Pericenter			Sphere of Infrared Exit		
	Initial Uncertainty	Uncertainty of New Estimate	Position Sensitivity	Velocity Sensitivi <u>ty</u>	Uncertainty of New Estimate	Position Sensitivity	Velocity Sensitivity
Parameter							
Spacecraft position	150 km	1.0			9.8		a. /
Spacecraft velocity	1 ft/sec	2.5	km	ft/sec	0.35	km	ft/sec
Gravity	$1/10^{5}$	$1/10^{5}$	0	0.18	$1/10^{5}$	1.8	0.06
Equatorial radius	1 km	0.9	0	0.11	0.9	0.1	0.00
Polar radius	$2 \mathrm{km}$	1.5	0	0.24	1.3	0.5	0.02
Instrument alignment	1 arc sec	0.8	0.1	0.51	0.65	3.7	0.13
Infrared bias	20°K	0.5	0	0.18	0.4	0.4	0.01
Surface pressure	10%	10	0	0	10	0.0	0.00
Surface temperature	50°K	15	0.1	0.26	13	0.3	0.01
Stratosphere temperature	30°K	10	0	0.19	8	0.2	0.00
Diurnal variation	25°K	7	0	0.20	5	1.0	0.03
Latitude variation	40°K	5	0.1	0.35	4	1.0	0.03
Seasonal variation	15°K	6	0	0.24	4	1.0	0.03
Circulation variation	15°K	1	0.1	0.45	1	1.1	0.03
Low altitude temperature #1	20°K	17	0.1	0.49	15	0.7	0.02
Low altitude temperature #3	20°K	17	0	0.10	13	0.6	0.02
Low altitude temperature #6	20°K	16	0	0.10	13	0.6	0.02
Low altitude temperature #8	20°K	14	0.1	0.27	13	0.1	0.00
Low altitude temperature #10	20°K	16	0	0.22	14	0.4	0.01
Low altitude temperature #12	20°K	15	0	0.21	14	0.1	0.00
High altitude temperature #1	20°K	15	0.2	0.69	12	1.0	0.03
High altitude temperature #3	20°K	12	0.1	0.43	9	0.5	0.01
High altitude temperature #6	20°K	20	0	0.02	20	0.1	0.00
High altitude temperature #8	20°K	10	0.1	0.27	9	0.5	0.02
High altitude temperature #10	20°K	19	0	0.04	19	0.0	0.00
High altitude temperature #12	20°K	13	0.1	0.39	12	0.3	0.01
			1				

Table 3-1.Summary of Venus Flyby Navigation Errors Sensitivityto Measurement Errors.

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however, is the fact that this is an independent model and does indicate results which, though not optimized for terminal navigation accuracy, do compare reasonably with the previous model. The net differences in the indicated accuracies for the two models are actually attributable both to this difference in information optimization and to the difference in phenomena uncertainty models.

Some conclusions from this flyby navigation analysis are listed below.

- a. <u>Navigation Accuracy</u>. Position and velocity can be estimated with 10 uncertainties of 1 km and 2 ft/sec at pericenter and 10 km and 0.3 ft/sec at exit from sphere of influence using the nominal 3-inch sextant, and assuming the nominal phenomena uncertainty. No significant change is observed when infrared measurements are excluded or when the frequency of the measurement schedule is varied within the range considered.
- b. <u>Phenomenon Uncertainties</u>. The parameters which most seriously affect navigation accuracy are planet gravitational attraction and the climatic atmosphere variations (diurnal, latitude, seasonal, and circulation).
- c. <u>Probe and Entry Navigation</u>. Any precision entry vehicle, particularly one planning on atmospheric braking to establish an orbit, requires not only accurate navigation relative to a planet center, but also requires accurate data on atmospheric density and location. This data is necessary since the density of Venusian atmosphere at high altitudes for example can vary by a factor of three or more due to unpredictable temperature effects. Optical navigation meadurements can predict atmospheric density to a few percent, and provide atmospheric location data which can not be obtained by other means.
- d. <u>Scientific Data</u>. A great deal of data about the planet, its atmosphere and its climate can be obtained by analyzing the navigation data. Many scientific objectives, as well as the navigation objectives, can be achieved through optical navigation measurements.

3.4 Unknown Landmark Navigation

To evaluate the relative performance of the unknown landmark tracker by a direct comparison with other navigation measurement techniques, it was necessary to simulate various representative spacecraft trajectories. A computer simulation of each trajectory was then flown to enable a simultaneous evaluation of the usefulness of each method to be made. The decision of the effectiveness of each measurement technique was made on the basis of comparing the reduction each technique made in an assigned cost. Specifically, the cost reduction was defined to be the ratio of the trace of the six-by-six state error matrix after and before the measurement was made. For this calculation to be made, an uncertainty in the unknown landmark measurement is required. Based upon the specified accuracies of existing unknown landmark trackers, angle encoders, star trackers, and the proposed optical system, the uncertainty in the tracking rate for the worst case was estimated to be 5-arc sec/sec.

The selection of unknown landmark star combination was constrained as discussed in Section 2.4 by the following:

a) Requirement that the unknown landmark site lie on the sunlit side of the planet,

where \underline{i}_{sun} is the unit vector from the planet to the sun.

b) Requirement on sextant LOS

c) Star occultation due to planet

$$\underline{i}_{s} \cdot \underline{i}_{z} < \cos \left[\cos^{-1} \left(\frac{z^{2} - R^{2}}{z^{2}} \right)^{1/2} + 5^{\circ} \right] \qquad 3.4-(3)$$

The Apollo star catalogue of thirty-seven stars was selected for use in the unknown landmark measurement. Five landmarks were chosen to represent a variety of sites in order to select the best geometry. Landmark sites 0 through 3 (see Fig. 3-19) were located on a circle centered at the sub-orbital point such that the angle between the landmark line-of-sight and the local normal was 65 degrees. The fifth site was chosen to be the sub-orbital point. Altogether, there were 185 possible star landmark combinations to choose from. The computer tested the ones that did not violate the established constraints and selected the combination that yielded the best cost reduction. The following three simulations were flown in order to compare the performance of the unknown landmark tracker with other navigation measurement techniques.

The first simulation was a hyperbolic planet flyby mission, starting at the entrance to the sphere of influence, passing through perigee, and terminating at the exit from the sphere of influence. The total flight time for the mission was 1.26 days. Measurements were made every hour and one measurement was made





at approximately pericenter. The only unknown landmark measurement that made a significant reduction in the cost was the measurement at pericenter. Its cost reduction was 0.03 while the best of the other unknown landmark measurements had a reduction of 0.989 and occurred one hour before pericenter. The reason that there was no appreciable cost reduction except at pericenter was that the V/h (velocity/altitude) rate was very low due to the following two considerations. First, the component of spacecraft velocity perpendicular to the line of sight to the landmark was smaller than the component at pericenter; second, the altitude an hour before pericenter, 25,000 miles, was much greater than the altitude at pericenter, 562 miles. Hence, a problem with this simulation was that there were not enough measurement points around pericenter.

The second simulation was similar to the first, with the exception that the measurement schedule was changed to accommodate more measurements near pericenter, and the following measurement schemes were included in the schedule.

- 0) planet diameter
- 1) sun-planet
- 2) planet-star
- 3) star elevation
- 4) unknown landmark

The schedule near pericenter included a measurement every 0.00139 days. When compared to the effectiveness of the other measurements, unknown landmark tracking was never selected as one of the best three measurements at any measured point.

In the third set of simulations, the mission was changed to orbiting a planet. The first run was a 100-mile circular orbit and lasted for 0.339 days, which included four orbits. Measurements were taken every 0.0069 days. For this mission there were 27 opportunities to make unknown landmark measurements. Of these possibilities unknown landmark tracking was selected 26 times as the measurement which gave the best cost reduction.

The mission was now changed slightly to an elliptical orbit with a pericenter distance of 100 miles and an eccentricity of 0.02. Out of 28 opportunities to use the unknown landmark measurement, it was selected to give the best cost reduction 16 times.

For a 300 mile orbit, however, the unknown landmark measurement was selected once out of 15 possible opportunities.

The basic result derived from the computer simulations is that unknown landmark tracking, as defined in this paper, is a useful navigation technique for near planet missions. Its relative performance improves with increasing V/h rates. This result is supported by the three different simulated missions. In the 100-mile circular orbit, the technique provided the best cost reduction whenever a measurement opportunity was available. In the 300-mile orbit, the technique provided similar quality navigation information as compared to the other navigation measurements. However, for the hyperbolic flyby, with a 562-mile perigee, the technique appeared not to be useful when compared to the other navigation methods.

The problem in evaluating a particular technique, however, is the selection of a suitable criterion for comparison, i.e., a cost. The cost for the above simulations was selected to be the trace of the 6-by-6 error matrix evaluated at the time the mission was terminated. For example, in the case of the hyperbolic flyby, the cost was evaluated as the trajectory penetrated the sphere of influence. In the larger sense, however, it may be more suitable to evaluate the cost in terms of the overall mission requirements or at least at the point of reentry into the earth's atmosphere where the major considerations are the position uncertainties. These position errors depend to first order on the velocity errors upon leaving the Mars sphere of influence and to second order, on the position errors. It would therefore be more realistic to assign only the velocity uncertainties to the cost at the Mars sphere of influence. In view of these considerations, the unknown landmark tracking would be viewed in a better light for the hyperbolic flyby simulation since this technique is more sensitive to velocity error than to position errors at higher altitudes. Naturally, there are many other performance criteria for a specified mission that must be considered. Some of the important ones are fuel consumption, weight, power, and reliability. Of course, the weighting given these considerations depends upon other factors, such as whether the mission is manned or unmanned. In brief, the results presented here are by no means conclusive, but rather indicative of navigation performance based on present instrument capabilities and general mission considerations. The results of the simulations have, however, indicated that the technique of unknown landmark tracking is a useful measurement for low-orbit navigation.

3.5 Recommendations for Further Study

From these initial results it would be desirable to conduct further studies in the following areas:

a. Complete the parametric analysis of the two missions considered "back through an Earth landing.

- b. Perform a parametric study of the total ΔV required for complete missions and investigate ΔV schedule optimization.
- c. Incorporate additional mission phases and tasks into the analysis such as rendezvous, insertion into planetary orbit (possibly using aerodynamic braking), probe navigation and guidance, planet landings, and long-duration attitude control.
- d. Determine the sensitivity of navigational accuracy to phenomena uncertainties as related specifically to all potential navigation and navigation tasks.
- e. Investigate the desirability of increasing the dimensions of the state vector to include some phenomena uncertainties.
- f. Incorporate star occultation and probe tracking navigation measurement options into the simulation.
- g. Investigate error models of Surface Irregularity Tracking Systems and formulate large-angle recursive techniques.

4. RADIATION SENSOR SUBSYSTEM (RSS) DESIGN STUDIES

4.1 Introduction

The primary functions of the Radiation Sensor Subsystems considered in these studies were attitude alignment and navigation measurements. The RSS design model is motivated primarily by the navigation function since the attitude alignment function has the less demanding requirements. Yet the design model could still adequately perform the contemplated alignment functions, as discussed in Section 4.6.

Two other functions of the radiation sensor subsystem have been suggested which are within the capability required for the navigation function. One is the function of low-power, long-duration attitude determination and stabilization for those periods in which IMU power or life-time degradation are critical considerations.

Another function which might be achieved within the capabilities of the navigation system would be the navigation and guidance of associated vehicles or probes. The requirements for this information will become better defined as specific planetary missions are defined.

Thus, practically all of this chapter deals with topics associated with the navigation function. Sections 4.2 through 4.6 present the thinking associated with the development of the RSS design model in an evolutionary order starting with the Navigation Angle Measurement Philosophies in Section 4.2, and then to a discussion of Precision Angle Encoders, in Section 4.3. Next, Section 4.4 presents a discussion of the Space Sextant Design Considerations, followed by a discussion of Radiation Sensors, in Section 4.5. Finally a General Description of the RSS Design Model is presented in Section 4.6.

The majority of the navigation measurements reduce to a series of measurements of angles between points of known attitude in an inertial reference frame (such as stars in a celestial coordinate frame) and points of navigational utility, located within the solar system (such as planet horizons and landmarks). Section 4.7 discusses the Anticipated Navigation Angle Measurement Accuracies for three of the navigation instruments considered.

There are four primary sources of uncertainty which limit the accuracy of the angle measurements. The first is the phenomena location uncertainties which are discussed in Chapters 1 and 2. The second is the phenomena detection uncertainty which is determined by the signal strength and sensor capabilities in a particular mission situation. For the purposes of the Navigation Error Sensitivity analysis the part of the detection uncertainties which are in angular units are included with the total instrument angular measurement uncertainties discussed in Section 4.7. The part of the detection uncertainties which are in km units are discussed in Chapter 2.

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The third source of uncertainty in the navigation angle measurement is introduced by the required precision angle encoder. The accuracy limitation introduced by the most promising types are discussed in Section 4.2. The fourth source of uncertainty is due to the instrument non-rigidities between the two lines-of-sight. Non-rigidities can be caused by differential thermal expansions, structural distortions under loading, etc. This source of error is also included in the navigation angle measurement uncertainties discussed in Section 4.7.

The design model discussed in Section 4.6 is felt to contribute a minimum of angle measurement uncertainty and a maximum of versatility within the state-ofthe-art. Within the ground rules stated in Section 4.3, this design model is also felt to provide a maximum of reliability, which is also of primary concern for space missions.

Sections 4.8 through 4.13 discuss particular topics relevant to the design of a radiation sensor subsystem. Sections 4.8 and 4.9 discuss aspects of the optical system design for a RSS. An aspect of visibility research which is particularly important for rendezvous maneuvers is discussed in Section 4.10. Section 4.11 discusses several automatic landmark tracking devices and presents a recursive navigation scheme which is suited to their capabilities. (The utility of this scheme has been evaluated and discussed in Chapter 3.) Section 4.12 refers to work performed on bearing heat transfer which is reported in a separate document, E-2215. Bearing heat transfer is an important factor in the design of accurate RSS systems. Section 4.13 presents an analysis of the use of a ring laser as an inertially referenced angle encoder for sequential detection type angle measurements.

4.2 Navigation Angle Measurement Philosophies

There are two navigation angle measurement philosophies being considered. The first is the "sextant" concept which requires the use of two lines of sight which simultaneously sense the relative angle between two navigation features selected from stars and features of solar system bodies. The optical axes of these two lines of sight are related through precision angle encoders which measure the included angle. The majority of the hardware analysis and design performed during this study is centered around simultaneous detection techniques as discussed in Section 4. 4. This angle measurement philosophy has the advantages of being the more thoroughly investigated navigation angle measurement technique and of using conventional angle encoders which have been more thoroughly proven for precision angle measurement. This philosophy has the disadvantages of being more sensitive to structural flexure and of requiring simultaneous, high-performance attitude control of two lines of sight.

We believe that the design model presented in Section 4.6 is minimally susceptible to structural rigidity problems and that the attitude control requirements are also reasonable. Our conclusion is that acceptable accuracies are possible with such a sextant design. The trajectory accuracies possible using this type of design are certainly sufficient for the spectrum planetary missions. The expected navigation angle measurement uncertainties expected from the navigation system designs considered, are discussed in Section 4.7. The navigation error sensitivity analysis for several representative missions has been discussed in Chapter 3.

For specific missions, it may be desirable, as tradeoffs are considered, to have greater navigation accuracy than afforded by the best sextant designs. Additional accuracy might be desirable to further reduce the total ΔV required for a mission, to obtain the most accurate aerodynamic braking, or for experiment requirements, etc. If this additional accuracy is required then the second philosophy holds the most promise.

The second philosophy requires the use of one line of sight which is first pointed at one target and then is rotated until it is pointed at the second target at a later time. The angle subtended by the two lines of sight is then measured. To measure this angle, an inertially referenced angle encoder must be used.

The second philosophy has the advantages of not requiring an optical-axisto-axis angle calibration prior to the measurement, and of having the minimum alignment stability problems. It is worth commenting that a manned spacecraft may be required to maintain a significant rotation rate for artificial gravity and/or for solar heating control during the majority of a mission. The simultaneous detection design model discussed in Section 4.6 has an attitude isolation system designed to permit operation during such rotations. But this motion can be more advantageous for the sequential detection technique. It is possible to take advantage of this motion in order to improve the effectiveness of a scanner system since this motion permits the observation of sequential regions of space to be viewed from one location on the spacecraft and provides a natural scanner motion.

Sequential detection systems have thus far been used in conjunction with unmanned satellite systems. The predominant mechanizations thus far have used scanners strapped down to spinning satellite vehicles.⁽¹⁾ In effect, the constant angular momentum of the vehicle in conjunction with the precision time of target detection by the scanner constitutes an inertially referenced angle encoder.

One difficulty with using a spin-stabilized spacecraft as an inertially referenced angle encoder is the large number of different measurement planes that would be required (see Chapter 3) to obtain the maximum navigation accuracy. It is doubtful whether competitive navigation accuracies could be obtained with this technique if only one spin axis attitude was used, and it would be impractical to reposition the spin plane to correspond to the measurement plane of the massive and extended spacecraft contemplated for manned planetary missions. Restabilizing the spin axis of the vehicle prior to each measurement would probably be required in order to obtain spin-stabilized angle measurement accuracies superior to those possible with the design model "sextant" to be subsequently discussed.

(One possibility that has been considered but is not reflected in this study is that of a navigation satellite or even of several duplicate navigation satellites in close proximity to, and in close communication with, but not attached to, the spacecraft. If this were considered permissible, then the use of this satellite when spin stabilized as an inertially referenced angle encoder would definitely be feasible.)

Another difficulty with this type of "angle encoder" is that it can become less accurate as the size and complexity of the vehicle increases since the associated mass distribution rigidity decreases. Relative motions of any masses connected to the spacecraft body caused by any of the many complex systems will disturb the required constant rate of rotation as well as excite vibration modes in the structure, stored fuel, etc. which will also cause variations in the rotation rate. An additional problem for manned space vehicles is that man is not fixed to the spacecraft but can move about, thus causing additional rotation rate disturbances of a body-mounted scanner system.

For these reasons a different type of inertially referenced "angle encoder", such as a gyro, would probably be required in order to utilize sequential measurement techniques. This would permit the avoidance of the rotation rate variation problem and permit attitude isolation between the spacecraft and instrument attitude control requirements. In order that a sequential detection system be able to produce superior performance for a manned planetary mission to that possible with a simultaneous detection system, a more sensitive and accurate inertially-referenced angle encoder is required than is presently afforded by existing gyros.

A developmental inertially-referenced angle encoder which utilizes a ring laser and is potentially very accurate, simple, and rugged is discussed in Section 4.13. Ring lasers have been developed for gyro applications which would be suitable for application as an angle encoder, but small variations and simplifications in the designs would be desirable to optimize the ring laser for this function. The ring laser angle encoder promises to provide greater accuracy thus permitting the sensor to assume the major error in the navigation angle measurement error budget. This would permit improvement of the navigation accuracy, or alternatively permit a significant relaxation of the requirements on sensor performance.

The combination of a ring laser angle encoder mounted on a scanner system such as the unit discussed in Section 4.5, provides the basis for a conceptually elegant automatic alignment and navigation system. The alternative simultaneous detection angle measurement philosophy is emphasized in Section 4.2 since it is sufficiently accurate and further along in its development. It can be stated though, that the design model discussed in Section 4.6 would be changed only slightly by the addition of an inertially-referenced angle encoder in order to permit sequential detection. Sensors, optics, and the attitude isolation system would essentially be the same.

4.3 Precision Angle Encoders

The navigation angle measurement philosophies discussed previously require the use of an angle encoding device which can measure and convert the encoder mechanical angle into a precise numerical form to be communicated to the ACG & N computer. Angle encoder accuracy is one of the most significant limiting factors in determining the minimum navigation angle measurement uncertainty that can be achieved. Visually read optical discs, precision gears, incremental optical encoders and multispeed resolvers are angle encoder types which have been considered for sextant (simultaneous detection) type measurements.

Visually read encoders are not considered to be suitable for this application, primarily because of the difficulties associated with folding the precision disc image out through the required degrees of freedom. Other difficulties are the demands upon the astronaut and the additional step required to inject the information into the computer.

Precision gears are also not considered suitable for this application due to their accuracy limitations; that is:

- a. They have little averaging of "ruling" imperfections (few teeth engaged at one time)
- c. They are susceptible to torque loads since they must be used to drive the axis in contrast to the other encoding techniques

The precision angle encoders which are most suitable for this application are characterized by several features. They:

- a. Have the most numerous and precise quantization over the 2π angle
- b. Average the maximum number of the rulings during a reading to minimize the effect of quantization imperfections
- c. Simultaneously detect the displacement of the rulings on opposite sides of the "disc" in order to minimize the effects of bearing system "imperfections"
- d. Transmit negligible torque between the two members. (A separate angle drive motor is used)
- e. Transform the precision angle into digital electrical signals which are updated at a relatively high bandwidth

Two angle encoding systems are considered most appropriate for the navigation sextant precision angle measurements. Either could prove more suitable depending upon the specification of the missions and the spacecraft. Either could be incorporated into the design model discussed in Section 4.6. Both are presently felt to be capable of approximately the same accur γy (in the anticipated environment and when used in the designs to be discussed). They induce angle uncertainties just slightly larger than 1 arc second, 1σ . Adding in the effects due to the anticipated environment, bearing quality, mechanical rigidity, aging, signal encoding, quantization, bandwidth, etc. brings the anticipated error to just less than 2-arc seconds, 10 for a complete angle encoding system. The "system" implied here includes all of the mechanical and electrical components required to produce a digital number which represents the precision angle of articulation between the two "mounting surfaces" for the optics. One set of optics is used for each line of sight mounted on each of the two members. The peak error is expected to be approximately three times as large as these 1σ numbers. A gaussian distribution of these uncertainties is assumed for the error distribution used in the error sensitivity analysis of Chapter 3, which is felt to be sufficiently valid for that purpose. In-flight angle encoder calibration capability is provided by the design model, as discussed in Section 4.6, which would minimize the significance of calibration drifts.

One of these angle encoders is an incremental electro-optical encoder. It has the advantages over the other encoder of having the most precise ruling, and the smallest quantization. It has the disadvantage of requiring more complexity in the navigation head where it is most exposed to the environment and where the reliability requirements are most stringent.

The other angle encoder is a 128-speed pancake resolver with an electronic A to D converter. The resolver itself, which is the only part mounted on the optical head, has the advantage of being simple and rugged. Therefore only the most reliable parts are exposed to the space environment. It has the disadvantages of having much larger intrinsic quantization and of requiring a more precise and complex external unit to convert the signal information to digital angle numbers.

Both encoders require the use of error correction determined from calibration in order that their accuracy be limited only by their stability. Their repeatabilities are significantly better than their linearity. These connections, obtained either from pre-flight calibration and/or in-flight calibration can be expressed and stored as a few dominant Fourier coefficients, and used by the flight computer to calculate connections to the indicated angle of any measurement made.

Either of these two angle encoders can be incorporated into the space provided in the design model. The multispeed resolver is presently illustrated in the design model system shown in Figs. 4-13 and 4-14. Incorporating either of these angle encoders into the design model will permit accurate sextant-type navigation angle measurements and would be sufficiently accurate for interplanetary navigation as discussed in Chapter 3.

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An inertially-referenced angle encoder is discussed in Section 4.13 which would make sequential detection of navigation angles with the simultaneous detection technique used in the design model. The sequential detection accuracies permitted by the ring laser are superior to those permitted by the best simultaneous detection mechanizations. The ring laser also has the advantage of providing intrinsic precision digital quantization. These advantages are somewhat offset by the more limited versatility of the navigation system if only sequential detection is used. This limitation on versatility is due to the necessity of using scanner-type sensors in association with the ring laser as dictated by the requirement for a finite rotation rate. A navigation instrument which could perform both simultaneous and sequential measurements would provide the most versatility and accuracy as discussed in Section 4.6.

4.4 Space Sextant Design Considerations

There are several general ground rules used as the basis of the design studies that have been performed.

- a. The instrument should be designed so that the performance of the alignment and navigation measurement functions would be as independent as possible of spacecraft attitude environment and attitude control requirements. This ground rule resulted in two further guidelines:
 - 1. There should be complete attitude isolation between the alignment and navigation unit and the spacecraft. Also, any images displayed to the astronaut should be fixed in the spacecraft reference frame so that he would not have to move his head in order to eliminate the need for the line-of-sight through all of the required degrees of freedom.
 - 2. The attitude isolation should provide the maximum practical field of view from its location on the spacecraft
- b. The highest practical navigation angle measurement accuracy should be attained within the constraints of simplicity in the mechanization and function. Maximum simplicity is dictated by the concepts of good design and the severe reliability requirements for planetary missions. The sort of phenomena and angle measurement accuracies that will be required for various planetary mission navigation accuracy requirements have been discussed in Chapter 3.

Several alternative sextant mechanizations are presented in this section. In the sequence of this presentation the effects of several alternative design concepts are discussed, such as:

- a. The errors permitted by different techniques of sextant image superposition.
- b. The considerations in the physical placement of the optical elements in the sequence of mechanical members.
- c. The effects of several alternative techniques of information transfer through the attitude isolation mechanization.

The discussion leads in an evolutionary way to our conclusion that the design model presented in Section 4.6, which utilizes electrical transfer of information through the degrees of freedom, would provide the best all-around performance for a planetary mission.

4.4.1 Independent-Lines-of-Sight Sextant

The first sextant mechanization that was studied utilized one azimuth (shaft) and one elevation (trunnion) degree of freedom for each line-of-sight. The two collimated beams from the two lines-of-sight are combined with a beam splitter at the spacecraft and then focused by an optical system to an image which was viewed through an eyepiece by the astronaut as shown schematically in Fig. 4-1. This mechanization was motivated primarily by the desire to minimize the number of optical transfer elements required, and to minimize the associated mechanical complexity and size. Such a mechanization does provide a neat mechanical package (Fig. 4-2).

In describing the configuration shown, it is first appropriate to identify the specific ground rules under which the design was studied. These ground rules take the form of the following list:

- a. Fixed visual viewing axis
- b. Two line-of-sight, four degrees-of-freedom instrument(3 base motion and 1 precision)
- c. Minimum light loss
- d. Maximum scatter shielding
- e. Maximum spherical coverage
- f. Erect imaging (no inversions)

Figures 4-1 and 4-2 show schematically and by design layout a configuration for providing a dual line-of-sight, four degree-of-freedom visual reference sextant. The drawings limit themselves to depicting only the optical redirection elements involved in the motion isolation scheme. It will be noted that each line of sight has two degrees of freedom (i.e., trunnion axis and shaft axis). This mechanization allows each line of sight to be independently pointed in the desired direction.

In any altitude-azimuth configuration such as this, as the object being viewed approaches the zenith, the ability to track approaches a limit. This limit is established by the frequency response of the instrument in reacting to base motion inputs normal to the trunnion-shaft axis plane as the line-of-sight approaches coincidence with the shaft axis.

A further phenomenon that deserves clarification in this configuration is that the object being viewed will be rotated about its optical axis as a function of the trunnion angle motion. This is due to the skewing nature of the trunnion axis to the trunnion mirror normal. The undesirability of this facet must be





measured against the benefits derived. This particular mechanization allows scatter shielding to be optimized and further provides the ability to have a 1:1 relationship exist between the line of sight and the trunnion angle.

The primary difficulty with this mechanization is that since the superposition of the two lines of sight occurs after several degrees of freedom, the accuracy of the system is severely limited unless the two targets are viewed exactly on their respective optical axes and are superimposed at the center of the field of view. For certain combinations of gimbal angles the navigation angle measurement errors produced become first order relative to the deviations of the lines of sight from their optical axes. Thus, even with perfect superposition of target images, there will be significant errors produced unless the superposition occurs precisely in the center of the field of view. Such attitude positioning accuracy is not impossible to achieve, but it is difficult. A concise matrix formulation which is useful for analyzing this system has been presented by Polasek.⁽²⁾

One of the primary motivations for using a sextant type of instrument is that the superposition of the two images should be such that the errors induced in the precision angle measurement by sextant positioning deviations will be of second order. If an instrument has this characteristic, then the targets may be superimposed anywhere within a relatively large field of view and still not seriously degrade the angle measurement accuracy. For most sighting situations this instrument would not satisfy this requirement and excessive sextant line-of-sight positioning accuracy would be required.

In order to eliminate first order sensitivity to pointing deviations, the two lines of sight can be combined immediately before they are brought through the attitude isolation mechanization. This is usually done with the assistance of a mirror and a beam splitter as illustrated in Fig. 4-3. One of the primary questions is the manner of articulation of one of the elements (usually the mirror) in order to generate the precision separation angle between the two lines of sight.

One technique that was investigated utilized an articulation axis parallel to the direction of the line of sight after its reflection from the articulated mirror, as shown in Fig. 4-3. Hereafter this mechanization will be called a "tilting" mirror.

The motivation for using this articulation technique was that the increment in the angle of bending of the line of sight is equal to the increment in the angle of articulation. This permits a 1:1 utilization of the accuracy of the precision angle encoder which is mounted on the articulation axis. This 1:1 utilization is twice as good as that afforded by the "nodding" type of articulation shown in Fig. 4-4 which was used in the Apollo sextant.



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Fig. 4-4 Nodding mirror articulation.

The difficulty with using tilting articulation is that the advantage to be gained in the better utilization of the precision angle encoder accuracy is obscured by errors introduced by off-axis image superposition. A differential rotation is introduced between the images of the two lines of sight which is equal to the included angle between the lines of sight. As the included angle increases from zero, the error introduced by off-axis superposition becomes first order until at 90° the superposition error bears a 1:1 ratio to the deviations of the targets from their respective optical axes. This 1:1 ratio occurs because the two images are rotated 90° relative to each other. In this limiting case, incremental motions of the sextant body about the precision axis thus cause incremental motions of the two targets at the image plane in orthogonal directions. The magnitudes of the angles of most navigation sightings are large enough to make the superposition errors allowed by tilting mirror articulation not competitive with nodding mirror articulation. Thus, nodding mirror articulation is preferred for the majority of the navigation angle measurements required, even with its inherent 2:1 increase in the error produced by the precision angle encoder. All other schemes thus far proposed which conceptually might eliminate this 2:1 penalty have been unattractive because of their inherent complexity which has nullified any potential advantage.

4.4.2 Combined-Lines-of-Sight Sextant

The second sextant mechanization that was studied (Fig. 4-5) used the concepts shown combined in Fig. 4-4, of immediate superposition of the lines of sight and of nodding mirror articulation. Immediately after superposition of the lines of sight, the combined lines of sight were focused by a lens system and then folded through the degrees of freedom to an image in the spacecraft reference frame. The lens shown represents a simple bonded acromatic triplet which would easily be defraction-limited over a field of view of plus and minus one degree for a focal length of some 36 inches. (Thirty-six inches is a representative length for the optical distance between the inner member and the focal plane shown in the spacecraft.) The triplet is designed to minimize vacuum focal-length shift by proper choice of the curvatures of the inner and outer spherical surfaces. This is an easy lens to build and there is no reason to incorporate a more sophisticated lens system since this one will provide essentially perfect images.

The objective lens is shown close to the entrance aperture in Fig. 4-5 for several reasons. By focusing the column of light into a cone and down toward the spacecraft it is possible to progressively decrease the diameter of the cavity from the maximum diameter required at the entrance aperture down to a minimum cavity diameter at the focal plane. This reduction in cavity size permits





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the minimization of gimbal sizes where it will most reduce the total size of the system, i.e., in the outer gimbals. The instantaneous field of view will cause the column to be somewhat larger than a cone to the extent that the point of the cone is moved around to produce the full size of the focal plane. But in general, the image plane will be considerably smaller than the entrance aperture.

Placing the objective lens close to the entrance aperture permits more flexibility in selecting the size of the instantaneous field of view. This is true because it is possible to use various optical transfer elements in order to keep the size of the light bundle down to a manageable diameter, even if a large field of view were desired. Putting the objective lens further down the cavity from the entrance aperture would necessitate larger openings at the entrance aperture for a specified, effective light-collecting area.

If the complete focusing optics were placed on the spacecraft then the diameter of the bundle of light could be no smaller than the diameter of the objective lens no matter how small the instantaneous field of view. As stated before, 36 inches is a representative measure of the length of the folded path through the three degrees of freedom from the entrance aperture to the spacecraft. Over this representative path length, a practical field of view of $\pm 1.0^{\circ}$ would result in an addition of approximately 1.5 inches in the diameter of the entrance aperture above the diameter of the objective lens. Thus for minimization of size it is best to locate the objective lens near the entrance aperture.

There are no navigation errors that would be introduced by removing the lens elements from the inner member and distributing them on the other gimbal members, or even by mounting all of the lens elements on the spacecraft member. This is true because the superposition of the two lines of sight takes place immediately on the inner member and thus a displacement of the combined image at the focal plane will not significantly effect the sextant accuracy. One thing to be considered in a design is that any groups of highpower lens elements should be mounted rigidly in a single member so that the image-focus quality will not suffer due to any relative transitions between members. Having the objective lens mounted in the inner member and the focal plane in the spacecraft member can cause focus displacements due to the instabilities inherent in such a long, folded path; but this focus shift is such that an adjustment of the eyepiece will accommodate it without any deterioration in image quality. The reduction in the size of the attitude isolation system afforded by placing the objective lens on the inner member seems well worth the possible small difficulty of focus shifts.

Another question that needs to be answered is where to place the focal plane for any reticles that may be required. For the sextant shown in Fig. 4-5 the reticles are placed at the focal plane in the spacecraft. Because of this placement two types of effects might be troublesome: if the spacecraft is rotating during a measurement then gimbal motions will cause the image to rotate relative to the reticle. Since the main function of the reticle for both the navigation and alignment functions is to identify the center of the field of view, this rotation should not be a significant problem.

Lateral shifts in the effective location of the reticle is the other effect which can result from mounting the reticle to the spacecraft member. Lateral shifts in the effective location of the reticle relative to the optical axis on the inner member can be produced by optical path distortions which are more probable with an optical path folded through several degrees of freedom. Such a shift would be a primary concern during IMU alignment, but it is anticipated that the magnitude of this shift would probably be comparable to the error allowed by IMU alignment requirements.

There is probably very little alignment accuracy advantage to locating the reticle on the inner member. The improvements that could be realized by the placement of the reticle on the inner member are not worth the additional complexities produced by the required addition of more optical transfer elements.

The primary design task for this second sextant mechanization consisted of devising the best possible attitude isolation system which would transfer the combined line of sight to the spacecraft frame of reference where it could be viewed by an astronaut. Three degrees of freedom were required in addition to the one axis of precision mirror articulation. In order to somewhat simplify the signal transformations required to control the lines of sight, it could be desirable to make the first degree of freedom coincident with the axis of precision articulation. To do this would require three additional mirrors to fold the combined line of sight to this axis. The small simplification in the transformations to be derived by adding these three mirrors does not appear to justify the additional mechanical complexity required, so these mirrors are not included in the schematic diagram shown in Fig. 4-5.

Folding the line of sight through the degrees of freedom intrinsically introduces image rotation. An additional degree of freedom would be required to rotate the image into a specified orientation which might be desired by the astronaut. This could be done by placing an articulated double-dove prism at an appropriate point along the optical train. (The design model discussed in Section 4.6 has the intrinsic capability to provide this fifth degree of freedom by electrically rotating the image with the incorporated video tracker, without recourse to an additional mechanical degree of freedom.) Based on studies of the navigation phenomena that could be used for planetary navigation there does not appear to be any obvious benefit to be derived from including this additional degree of freedom which would warrant the additional complexity. It would be necessary though, to transform the line-of-sight servo signals so that motion of the image viewed by the astronaut would conform to that commanded by him in his frame of reference with his manual control stick.

One other difficulty which can occur when a line of sight is folded through the degrees of freedom with mirrors is that the image can be inverted (besides being rotated). The system shown schematically in Fig. 4-5 would require one more or one less mirror in order to produce a non-inverted image. It might be possible to use only two mirrors on the last gimbal, or an additional mirror may be required to bring the line of sight to the astronaut. Either alternative would alleviate this difficulty. There is not much more evidence for this question than there was for the image rotation question to indicate that image inversion is a major difficulty in planetary navigation. But, because of the relative ease with which this difficulty can be avoided, it could be eliminated.

This second sextant mechanization was pursued only slightly beyond the conceptual stage since it soon became obvious that this was not an optical solution for navigation requirements. One of the primary difficulties was the necessary size of the attitude isolation system for a meaningful size of optical collecting aperture. It was necessary to bring the line of sight out through the center of each attitude isolation axis thus requiring a relatively large inner diameter for most of the bearings, torquers, and angle encoders; thus increasing the size of the units and the associated size of the gimbals. There were three mirrors required on each gimbal member and a large clear cavity connecting the mirrors along the folded line of sight which made each of the gimbals additionally large. The connection of flex prints between members was made more difficult by the large inner diameters. Sliprings were even less satisfactory. The size of the system approximately doubled with the addition of each degree of freedom and so with the necessary three degrees of freedom it became particularly large. If the number of required degrees of freedom were reduced then optical transfer of the image would be more attractive.

Another difficulty resulted from the large number of folding mirrors required. With three mirrors required on each gimbal, the distortions introduced by the net effects of all of the mirrors can significantly degrade the image quality. When only a few mirrors are used, the distortions introduced by the mirrors are usually not significant. As alluded to in the previous paragraphs, one of the primary considerations in the design of a two-lines-of-sight instrument is the optimal transfer of navigation information through the attitude isolation system. As has just been stated, direct transfer of optical images by folding the line of sight through the gimbals is cumbersome. The size of such a system increases rapidly with the number of degrees of freedom required.

Fiber optics have also been considered for information transfer through the gimbals but are even less attractive than direct optical transfer. A highly folded optical system would be required on the inner member so as to produce an image to illuminate the termination of the fiber bundle. Such an optical system plus the bundle termination would appreciably enlarge the inner member. A fairly sizeable bundle of about one inch in diameter would be required and even that would degrade the resolution from that attainable with direct optical transfer. Such a bundle would be just as difficult to direct through the degrees of freedom as direct optical transfer. The torque required for twisting or flexing of the bundle at each degree of freedom would be troublesome, and there is the danger of breaking the fibers at these flex points. Another difficulty is that all fibers do not equally attenuate the light, and after three feet of transmission there would be disturbing non-uniformities introduced in the image intensity. Thus, for these several reasons fiber optical transfer has not been considered further.

4.4.3 Alternative Manual Sextant Design Concept

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In the course of studying potential candidates for visually operated sextant configurations, several schemes were considered. One of the configurations that deserves special mention involves an adaption of the current Apollo sextant head which would avoid much of the attitude isolation complexity. It must be noted that the Apollo sextant head, from the point of view of optical efficiency (i. e., number of optical redirection elements involved in the combining and measurement aspects of usage), approaches an optimum. It further must be noted that, inasmuch as all mirror normals involved in the mechanization lie in a common plane, superposition of sighted objects at a reasonable separation from the center of viewed field, say 1/4 of the total field, constitutes no degradation in accuracy of measurement, a fact not immediately apparent. Reasonable shielding potential plus experience with this configuration tend to emphasize its desirability. A schematic is shown in Fig. 4-6 and a design layout is shown in Fig. 4-7.

In order to bypass the operational constraints of base motion stability, inherent in the usage of the Apollo sextant, it was felt that we could allow limited freedom, say $\pm 15^{\circ}$, about two axes normal to the existing azimuth axis, without



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(Not available at time of publication)

Fig. 4-7 Design layout of alternative manual sextant $\frac{1}{2}$
additional folding of the optical path through these two degrees of freedom. Mechanization of this, however, would result in requiring the astronaut to move his head with respect to base structure. It was our understanding that this violation of ground rules, cited in the statement of work, was unacceptable; hence no serious study of this configuration was performed. It appears desirable, however, to represent this configuration as a possible alternative in the event that the ground rules may be reconsidered.

4.4.4 Design Model Sextant

The design model presented in Section 4.6 uses electrical transfer of information to the spacecraft. The relatively small additional complexity produced by the addition of electro-optical sensors is felt to be well worth the savings in the complexity of the optical system, and the appreciable savings in the total size of the system. Also, since photoelectric preprocessing or detection will probably be required to obtain sufficiently accurate horizon detection, it is preferable to put the photoelectric sensors on the inner member at the outset, thereby avoiding the need for optical transfer.

Another significant source of error resulting from the use of this optical superposition instrument was the difficulty in accurately detecting the superposition of targets. The navigation error sensitivity analysis results discussed in Chapter 3 showed that horizons were the solar system body phenomena which contained by far the best information for interplanetary navigation. Most of the information is contained in the gradual intensity gradients which are produced by atmospheric phenomena. In general, horizons cannot be detected very accurately by man using his unaided eye, unless there is some prior processing of the image information.

The preprocessing techniques needed to optimize horizon detection such as optical filtering, neutral density, selective spatial distribution attenuation, conversion of ultraviolet or infrared to the visual range, etc. are in general in conflict with the needs for optical star detection. For a planetary mission which has such a large and diverse set of navigation phenomena to be detected, it is nearly impossible to optimize the manual detection of all of the many starhorizon sightings required. The net result is that there is no accuracy advantage to the general concept of "sextant" superposition of the two raw optical images.

It is possible to make a more accurate and versatile instrument which has a separate set of optimal optics and electro-optical sensors for each line of sight. In this manner, information from each line of sight can be separately preprocessed, superimposed electrically, and then optimally displayed to the astronaut as required. It is also possible to devise automatic detection and information communication links with the computer and other systems.

The accuracy anticipated with the second sextant mechanization was not as good as is anticipated for the design model system. One reason was the 2:1 increase in the error of the navigation angle measurement over the error attainable with a precision angle encoder. As discussed previously, this two-for-one error was produced by the nodding mirror articulation necessitated by sextant image superposition accuracy considerations.

By utilizing an electrical instead of an optical superposition of target images it is possible to practically realize 1:1 error relative to the precision angle encoder error, as well as optimal navigation target detection. Thus, for the many reasons previously discussed, it is felt that the system presented in Section 4.6 is a more optimum system for planetary navigation.

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4.5 Radiation Sensors

The radiation sensor subsystem model discussed herein uses electrooptical sensors as the radiation transducers, in contrast to the astronaut's eyes as was done for Apollo navigation. This approach frees the astronaut to perform an executive function to the degree required by the sophistication of a manned planetary spacecraft.

Because it is less cumbersome, the electrical transfer of information through the three axes of gimbal attitude isolation between the alignment and navigation instrument and the spacecraft is preferable to optical transfer. The electrical transfer of information also facilitates deployment of the radiation sensor subsystem, since the transfer of information over long distances is more easily accomplished electrically than with direct optical transfer. In addition, a wider separation of the environmental needs of the RSS and astronauts is allowed. For example, should the astronauts require radiation storm shelters, electrical transfer would allow placing the shelter at a maximum distance from the hostile external environment.

The electrical transfer method will also simplify the deploying of several sensing systems on opposite sides of the spacecraft. Information can be conveyed from each of these subsystems to a common control station. This type of deployment scheme may become necessary to avoid imposing sighting field-of-view constraints or, conversely, attitude demands, upon the spacecraft. Through the redundancy inherent in a deployed, multiple-unit system, reliability too is enhanced.

Electro-optical sensors provide the additional advantage of facilitating image processing prior to receipt by the astronaut. Examples of this image preprocessing are: optimization of visual adaptation level; elimination of unwanted or unnecessary regions of the field of view (e. g. other bright objects); elimination of image motion and/or rotation of the image to a preferred orientation without including another mechanical degree of freedom. This preprocessing of the image also allows the flexibility to modify image magnification through electrical "zooming" and the conversion of UV and IR navigation target images to visible images.

Electro-optical sensors also permit the encoding of image data for transmission to Earth or to another spacecraft, capabilities which may be highly desirable during extended missions.

The performance characteristics of electro-optical sensors are particularly beneficial for star-horizon measurements. These measurements, as discussed in Chapter 3, provide the majority of useful navigation information. For this type of detection task, the electro-optical sensors permit resolution considerably below the optical resolution limit, are faster and more sensitive. These qualities allow optimal use of attitude servo capabilities and "high speed" target marking, thus facilitating more accurate measurements. Electro-optical sensors permit completely automatic operation of the RSS. This would allow the astronaut to concentrate his efforts around decision and investigative matters during time-critical phases of a mission, and also free him from the highly repetitive and time-consuming alignment and navigation tasks.

The navigation error sensitivity analysis has shown that a relatively high measurement frequency is desirable while within a planet's sphere of influence, and a high measurement frequency would be more practical with an automatic instrument. An automatic instrument would also allow unmanned missions or unmanned portions of mission.

In his executive role, the astronaut would primarily contribute his intelligence and versatility, monitoring on a display the performance of the alignment and navigation sightings.

The radiation spectrum from the ultraviolet through the microwave region has been considered for alignment and navigation sightings. The primary radiation phenomena which have thus far been considered are listed below.

Passive Radiation (as a result of natural phenomena)

- 1. Star attitude
- 2. Planet limb
- 3. Star occultation by a planet limb
- 4. Planet radiation center (for distant planet case)
- 5. Sun limb
- 6. Planet known landmarks
- 7. Planet unknown landmarks or other radiation spatial distribution variations
- 8. Asteriods, satellite, moon, and other solar system related bodies

Active Radiation (generated by man-made systems)

- 9. Microwave and laser radiation originating from the spacecraft and reflected off "near" targets
- Cooperative beacons located at locations of known or unknown location on planets, cooperative vehicles, satellites, or probes

There are a variety of alignment and navigation sightings which must be performed during a planetary mission. Several of the sensors studied appear to hold much promise for detecting and locating the above phenomena, and for incorporation into a guidance and navigation system. One primary goal for the design model was to have minimum complexity. To achieve this, a few versatile yet relatively simple sensors appear preferable to a large number of specialized instruments. The sensors discussed in this section and which are presently incorporated into the design model are felt to offer the best mix and total of versatility, accuracy and reliability.

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4.5.1 Video-Tracker

One sensor subsystem consists of a general-purpose video image dissector or vidicon system which is mated with a long effective focal length lens system which provides high resolution and a relatively wide field of view (diffraction limited resolution over a 1° field of view). This type of system could accurately track phenomena 1, 3, 4, 6, and 7 above. Where available and where necessary this instrument could track phenomena 8 and 10. This instrument could also be used to display a wide variety of targets to the astronauts with approximately "1000 lines" of video information.

This sensor system has the primary advantages of versatility and accuracy. One requirement for the use of this instrument is that the line of sight be relatively well stabilized because of the relatively large image size that is observed with a small instantaneous field of view.

A representative video tracker design is shown in Fig. 4-8. An image dissector is shown as the detector unit. This type of detector is presently being used in the star tracker of the MIT, X-15 Horizon Definition Experiment, and would be adequate for the task. "Next generation" tubes are becoming available which would permit improved performance within an even smaller volume.

An adaptation of the electronics developed at MIT for the X-15 system daytime tracker is certainly adequate for a planetary mission. The advanced detection and tracking techniques provided by this system would permit nearly optimum high speed star detection and very accurate tracking with a suitably high bandwidth. Star tracking uncertainties for phenomena 1 and 3 of less than one arc second, one σ are anticipated using this sensor-electronics package in conjunction with the optics to be subsequently discussed. This uncertainty is included in the navigation angle uncertainties listed in Section 4.7. (This number assumes an inflight calibration prior to a navigation measurement, as discussed in the section, in order to remove any long time constant drifts). These electronics could also provide the raster scan for search, and for image presentation to the astronaut.

Vidicon tubes have capabilities similar to those of image dissectors and should also be considered for this function whenever a development program is initiated. In the not too distant future, solid state image dissecting devices will probably be developed which will be competitive with tube type sensors, and would then be very attractive for this application.

Phenomenon number 4 (planet radiation center) is useful primarily beyond relative ranges at which the uncertainties introduced into the angle



measurement by the uncertainty in the definition of the terminator and by the noise introduced from the central region of the planet, are small in comparison to the uncertainties introduced by the navigation instrument. At ranges where phenomenon 4 would be navigationally useful, the video tracker could be used as a planet tracker.

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At intermediate ranges it is best to exclude the terminator and central region and utilize only the limb of the planet which contains the most useful information on the location of the planet. It is probably possible with some additions to the mechanization to track limb phenomena such as 2 and 7 with the video tracker but the second type of sensor, discussed in the next subsection, is better suited for the detection of limbs.

Also at these closer ranges, some detail on the planet will be visible (landmarks or even cloud variations) which could be used for navigation. On the Earth, phenomenon 7 would be available since there are identifiable landmarks at known locations determined within an accuracy sufficient for navigational utility. Known landmarks can be identified and tracked by viewing the landmark with the video tracker and displaying the image to the astronaut. By comparing the image with the superimposed electrically generated reticle the astronaut can track or mark the coincidence of the landmark and the reticle.

On Mars, identifiable features are not located to within a sufficient accuracy, and on Venus there are no fixed features available due to the obscuring clouds. Even on Earth it is difficult to find and identify known landmarks due to random weather obscurations. Thus even though features will become visible, their identity and location may not be sufficiently known so that they are of use as known landmarks.

As discussed in Section 2, and as evaluated in Section 3, tracking unknown landmarks can provide valuable navigation information when the spacecraft is near the planet. The visual display discussed above can be used for manual tracking of unknown landmarks.

For unknown landmarks the automatic tracking techniques become attractive since the sophisticated landmark identification problem is eliminated. Section 4.11 discusses the characteristics of several unknown landmark tracking instruments. The performance characteristics of these instruments were used as the basis for the unknown landmark navigation technique which was presented in Section 2 and evaluated in Section 3. All of the unknown landmark trackers considered had limitations on the aspect angle changes and zooming that could be tolerated without introducing significant errors. These restrictions were reflected in the limited tracking angle technique evaluated in Section 3. In Section 2 it was indicated that major improvements in the utility of unknown landmarks could be realized if a landmark could be tracked through a larger angle.

The manual technique of tracking landmarks using the video display described could be used to track through a large angle, but this would tie up an astronaut for prohibitively long periods, in contrast to the situation for known landmarks where only a single mark is required; a task of short duration.

The video tracker could be controlled to track unknown landmarks automatically through large angles. A modification of the Bolsey concept could be adapted to the video tracker. The fixed diameter circular scan and data processing used by Bolsey can detect the two axes of translation and the one axis of rotation of the image. The video tracker can be programmed to scan most any pattern, and the circle is one of the simplest for the tracker to provide.

By approximately modifying the "diameter" of the scan the video tracker can accommodate range changes. By appropriately distorting the circle into an ellipse the video tracker can also accommodate aspect angle changes. The information needed to permit the tracker scan to adapt to these two additional degrees of freedom can be obtained from a programmed set of cyclically varying scans which vary in size and ellipticity about the previous best ellipse. The orthogonal deflection axes of the tracker are ideally suited to provide this type of modulation, since variations in the amplitude and phase of two time related sinusoidal signals are all that are required to obtain these scan variations.

As with the Bolsey system the first circular scan would be recorded and each successive scan then compared to it to determine the quality of a correlation with each successive scan. By an appropriate selection of scan variations and proper interpretation of the associated correlations it is possible to determine error signals for all five of the degrees of freedom. The determination of the two degrees of freedom desired in addition to the three provided by the Bolsey instrument is allowed by the modulation of the scan with the additional size and ellipticity variations.

There is one variation in the pattern recorded by the initial scan cycle which cannot be accommodated by this relatively simple scan variation. The topographical altitude variations along the annular region observed will cause changes from the recorded pattern as the aspect angle changes which cannot be accommodated by size and ellipticity variations. In general this effect does not increase the errors of a recursive, unknown landmark navigation scheme since this uncertainty in topographic altitude is an inherent source of error. Also, the altitude averaging performed by the tracking instrument can reduce the errors introduced by this altitude uncertainty below those that would be obtained by tracking a single point. Thus, the inability to compensate for this sixth distortion is not considered to be a significant drawback.

Because of the landmark tracking capabilities of a video tracker, a separate unknown landmark tracking instrument is not included in the ACG&N design model (though the more severe constraints that would be imposed by the existing specialized instruments discussed in Section 4.11 have been assumed in formulating the error models and unknown landmark navigation schemes presented in Section 4.11 and analyzed in Section 3). Separate specialized image trackers or V/h sensors will probably be required for other functions as missions become defined.

4.5.2 Sensor Optics

The optical system shown in Fig. 4-8 is discussed in more detail in Section 4.8. This optical system has an aperture of 2-1/2 inches and an effective focal length of approximately 50 inches. Reflective optics have been chosen since under light shield volume constraints they are minimally susceptible to scattered light; are not susceptible to vacuum focus shifts; and though achromatic, will still pass wavelengths considerably beyond the visible range. The collecting aperture is large enough so that the system can track very dim stars or planets with a good signal to noise ratio. The long focal length of the system produces a diffraction limited image which approximately matches the aperture size of the image dissector tube. This permits very accurate tracking and a minimum susceptibility to background light.

The light baffling shown in Fig. 4-8 is a very important part of the optical design and is discussed in more detail in Section 4.9. An optical system design similar to the one shown would permit star tracking relatively close to the sun or to bright planets. The baffling shown has been developed using a combination of empirical knowledge and with the use of a scattered light evaluation computer program which calculates for various designs the amount of stray light which reaches the focal plane. This program is especially useful for indicating the sensitivity of the quality of entrained stray light to design variations, so that the baffle design may be analytically optimized.

4.5.3 <u>Scanner Photometer</u>

This same optical and baffle design would also be used for the second type of sensor system used in the design model. The sensor cavity of the sensor system shown in Fig. 4-8 would be filled instead with the sensors appropriate for characteristic scanning types of measurements. The low scattered light characteristics of the optics are even more important for this instrument since the instantaneous field of view will be larger than those of the tracker.

The second type of sensor system is a scanner photometer. This instrument would consist of rigid high resolution optics mechanically attached to a fixed field of view aperture (probably a series of rectangular slits). The aperture would be mounted or scribed on the backside of a quartz field flattener similar to the window of the image dissector shown in Fig. 4.8. The attitude of the target in the direction orthogonal to the slit is determined from the radiation passed by this aperture as the slit is scanned over the target image. This radiation is detected by photodetectors mounted behind the aperture plate. By having a couple of non-parallel slits or a "V" aperture it is possible to also determine the second degree of freedom. This type of relatively simple mechanical construction permits the maximum rigidity since there is no internal spatial modulation of the aperture position by mechanical or electrical modulators. The modulation of the target is provided by the scanning motion of the whole scanner instrument. This type of instrument is ideally suited for accurately determining the location of phenomena 1, 2, 3, 4, 5, 8 and 10. This sensor system has the primary advantages of versatility, high accuracy, reduced line of sight attitude control servo requirements, and simplicity.

A long focal length optical system similar to that required for the vidicon tracker is desirable for the scanning photometer in order to expand the image of the field of view on the focal plane for optimal aperture definition of the field of view. In general the successive targets will be detected at varying locations along the rectangular aperture. Apertures can be "machined" by various techniques with a precision of better than 0.0001"; so that with the 50" effective focal length chosen for the design model optics, the expected aperture imprecision would contribute less than one half of an arc second to the measurement uncertainty. This long focal length would also permit less dense placement of the sensors and/or easier "piping" of the energy to the respective sensors.

The design model reflective optics would transmit, and achromatically focus radiation from the UV through the IR, thus permitting the use of

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photomultipliers, photodiodes, infrared detectors, etc., with the same set of optics. From the UV through the near IR, sensitive high-bandwidth photodetectors can be used which permit very accurate determination of target location. (The use of these detectors will be discussed subsequently.) At wavelengths longer than approximately three microns, various IR detectors must be used which are slower and which permit lesser navigation angle measurement accuracies. IR phenomena are navigationally useful primarily near a planet where limb navigation phenomena subtend a large enough solid angle to be detected, and where the angle measurement uncertainty added by the IR detectors is less significant than the km uncertainty in the phenomena location. For practical navigation instrument designs, 30-arc seconds, 1σ is a representative number for the minimum navigation angle measurement uncertainty allowed by IR detectors. In general, IR phenomena would be relatively significant for navigation only in the few cases for which shorter wavelength phenomena are not available. In general there are sufficient phenomena available within the 0.2 μ to 2 μ range which can be located much more accurately using photo-emissive and photoconductive devices. These sensors will be the basis for the performance discussions in the remainder of this subsection.

A strong motivation for including a scanner-photometer instrument is its capability to accurately detect all sorts of limb phenomena (such as phenomena 2, 3, and 5), which have been found to provide the majority of the most useful navigational information on the location of solar system bodies. In general, more power is available from limb phenomena than from stars. Because of this stronger signal it should be theoretically possible to more accurately locate limb phenomena than stars. Much more analysis is required, though, in order to realize this potential accuracy, due to the greater complexity of the limb spatial energy distribution. The limb images may require mathematically simple profile descriptions, but even so their detection is not as simple to analyze as the detection of "point" images from stars. To develop an instrument which will realize the practically achievable capabilities of limb detection will require considerable additional analysis of phenomena, locators, and instrument design within the framework of general mission needs and priorities. As stated previously, it is believed that in general smaller angle related uncertainties are theoretically achievable for limb detection than can be achieved for star detection. With the necessary analysis and development of an appropriate instrument, it is felt that a practical instrument could realize limb detection accuracies approximately equal to those possible for stars.

[It should be noted that for limbs, an additional uncertainty term, in <u>km units</u>, must also be included in the navigation angle measurement error models used to describe limb measurements. This uncertainty term is due to the km uncertainty of the phenomena "locator" chosen (such as 1/2 maximum intensity) as detected by the instrument devised for this purpose. For star detection there is no equivalent "km" term. This additional uncertainty term is discussed in Section 2 and included in the error models used for the analysis in Section 3.]

Star detection uncertainties are the least difficult to analyze due to the elemental image spatial distribution produced by stars. Thus for the purposes of this study, the star location angle uncertainties allowed by the scannerphotometer have been assumed to also represent the angle related uncertainty term introduced into the limb location determinations.

Thus, one motivation for investigating the star detection uncertainties allowed by a scanner-photometer unit is to determine the uncertainties that can be expected for limb detection. It should be noted that phenomenon 10 and certain types of phenomenon 8 can be detected by the scanner and also produce elemental image spatial distribution so that the analysis of star detection uncertainties can be directly applied in the evaluation of the scanner capability to detect these phenomena. Another motivation for studying star detection uncertainties is that star detection capability would be desirable in the actual instrument in order to permit inflight sextant calibration prior to navigation measurements. Simultaneous detection of one star or of two stars would permit a calibration of any thermal or stress induced deformations between the two lines of sight which may have occurred during the mission subsequent to any calibrations performed on Earth. This type of calibration would significantly reduce the uncertainty that would have to be allocated to the instrument, and would provide a functional check prior to each set of measurements. Scanner-photometer star detection capability would also be desirable for the minority of navigation sightings which are more optimal if the tracker instrument is used to detect the solar system phenomena (such as for landmark tracking).

Figure 4-9 illustrates a typical change in the mean rate of expected photon arrival through an aperture opening as it is scanned across a star image imbedded in some background. The photon arrival is a nonstationary poisson process. But for very slow scan rates the randomness of photon arrival would cause negligible variances from the mean and the limitation



Fig. 4-10 Dynamic scan: typical photon arrival events vs. angle.

on star location determination accuracy would be the ability of the sensor and logic to phase detect the image location. For the optics image sizes discussed in Section 4.8 this would be a small fraction of an arc second and would not significantly limit the scanner accuracy for this application.

A typical "fast" scan from which there is a limited star signal is shown in Fig. 4-10 which illustrates basic information that is available from a scan, that is the association of photon arrivals with scan angles. For a "fast" scan as shown there is a limited expected number of photons from the star which will pass through the aperture opening. Since the information on star position is contained within the mean rate of arrival, the size of the statistical sample required in order to determine the mean within a given uncertainty is of primary importance in determining the location of a star to within a desired uncertainty. The variance in the "mean" rate of photon arrival evident in Fig. 4-10 from the mean rate expected is shown in Fig. 4-9.

It is also evident that the superposition of random background photon events constitutes noise which will increase the variance of the photon arrival rate above that obtained from the star alone, thus producing more uncertainty in the determination of star location. Thus the aperture field of view is usually reduced in width to just slightly larger than the star image, and minimized in length so as to reduce the background as much as possible. The optical system designed for this system should allow a minimum of this obscuring light through the aperture areas contemplated and except for stars very near to the sun this should not produce serious performance degradation. Even so, background will still be an important parameter in the determination of the optimum aperture geometry.

One thing to note is that there is no information on the position of the star relative to the aperture when the star is moving within the center of the aperture. This period is primarily useful for reducing the uncertainty in star detection (whether the star is somewhere within the aperture or not). Because of the low scan rates which would be used for this application to achieve minimal target location uncertainties there will be relatively little difficulty in achieving high detection probability. Information on the location of the star image relative to the aperture is provided only during the intervals during a scan when some portion of the star image is moving across either of the two edges of the aperture opening. The photons from the star which pass through the aperture when the "image" is completely within the opening constitute noise for star location determination if they are not excluded from the position estimation logic. There are several functional relationships which approximately relate the uncertainties of star location determination to some general parameters of scanner design and operation. The five functional dependences listed below all relate directly to the number of photons from the star which enter the aperture during the intervals in which some fraction of the image is interrupted by the edge of the aperture (when the star is not completely outside of, or completely within the aperture opening). It should be noted that all have the same inverse square root dependence due to the fact that they are all relative in the same way for a given image angular subtense and a given aperture angular width to the number of star photons passing through the aperture. These five functional dependences are:

location estimation ~ uncertainty

- 1. $1/\sqrt{\text{star intensity}}$
- 2. $1/\sqrt{\text{effective collection aperture}}$
- 3. $1/\sqrt{\text{time star is within aperture}}$
- 4. $1/\sqrt{\text{number of aperture slits (multiple properly spaced aperture openings may be cut in the aperture plate)}$
- 5. 1/√number of repeated scans (It is possible to repeatedly scan over the star in various ways and improve the accuracy. This functional dependence can be fully realized only if the scan can be repeated without uncertainties being added by the angular encoder reference.)

The uncertainties in estimation that can be achieved by an actual system are dependent upon the detection used to convert photons to an electrical signal and the particular circuitry and logic used to mark the estimated star location.

Figure 4-11 shows scanner uncertainties calculated by Control Data Corp. (CDC) for a scanner system using a single aperture, a photomultiplier, and a particular data filtering technique for a single scan.⁽³⁾ Negligible background and dark current noise has been assumed for these calculations. The uncertainties are plotted as a function of the rate of rotation. The parameter of the three curves is the angular width of the aperture. The data filtering technique used by CDC for these calculations does not exclude the counts produced when the star image is completely within the aperture opening. Thus, the effect of reducing this noise is evident since the estimation uncertainties are reduced as the width of the slit is reduced. An approximate



Fig. 4-11 Photomultiplier scanner estimation uncertainties.

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 $1/\sqrt{r}$ rotation rate dependence of the uncertainty upon the rotation rate is also evident from the curves. The accuracy limitations on particular types of photomultiplier star scanner systems have more recently been analyzed by Control Data Corporation.⁽⁴⁾ From the navigation error sensitivity simulation and analysis reported in Section 3, a third magnitude star is the dimmest target that need be considered which is the intensity of the star assumed for the CDC plot. The effective aperture of the design model optics is approximately 2 inches in diameter. Thus, since this is about half of the collection area assumed by CDC the uncertainties expected at a given rate using the design model optics would be $\sqrt{2}$ higher. But since the rotation rates to be used by the design model scanner are much lower than those used in the CDC calculations (between one and ten degrees per second), uncertainties considerably less than one arc second can be anticipated using the design model scanner under these conditions.

Solid state photodiode sensors are also attractive for use in a scanner system. They are very attractive for space applications because of their compact size, mechanical ruggedness and their wide spectral sensitivity. Silicon photodiodes are attractive for a couple of other reasons. It is possible to mount several miniature chips directly to the back side of the aperture plate, each directly behind its respective aperture, thus permitting multiple detectors in a rigid compact and reliable package. It is also much easier to sense the limb of the sun with photodiodes than with a photomultiplier, which is important for navigation and to permit in-flight calibration of the critical angle prism sun sensor, as discussed in the design model section.

In general, photodiode detector systems have higher quantum efficiencies and thus produce higher signals at low signal levels, but are noisier than photomultipliers. In order to minimize the noise it is necessary to use the smallest possible detector. A single miniature sensor is used as the basis for a scanner design developed by Baird Atomic Corporation. Figure 4-12 shows the estimation uncertainties calculated by W. Westell of Baird Atomic Corporation as a function of rotation rate for a scanner system based upon the use of the design model optics and a Baird Atomic sensor-data processing electronics design combination. Data is shown for several star-aperture situations. Two different sensor areas are analyzed as required by the different angular subtense of the two aperture slit lengths chosen, of 1 degree and 10-arc minutes. An indication of the sensor noise is that a "blip star" of +4 magnitude would give a signal equal to the detector noise produced by the detector for the 1° slit.

A 10-arc second wide slit was assumed in both cases so as to be just slightly larger than the star image size produced by the optics. This



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Fig. 4-12 Photodiode scanner estimation uncertainties.

corresponds to the results of the CDC analysis for photomultiplier systems which indicates the best accuracy for the smaller slit widths which are just slightly larger than the star image.

As an indication of background susceptibility for these two aperture lengths: a background of 0.5 foot-lamberts for the 1° length, and a background of 3 foot-lamberts for the 10-arc minute length would increase the uncertainties shown in Fig. 4-12 by a factor of $\sqrt{2}$.

Analysis still needs to be done on the photomultiplier scanner systems at the lower rate range anticipated for the navigation angle measurement application. At these lower rotation rates considerably better performance than that shown in Fig. 4-11 is anticipated. The $1/\sqrt{\text{rotation rate}}$ dependence should approximately hold in this region so that these curves can be extrapolated to these lower rotation rates to see what kinds of performance can be anticipated.

Thus, for both photomultiplier and silicon photodiode single aperture-single scan systems, estimation accuracies of a fraction of an arc second can be anticipated. Approximately 0.7 arc second, 1 σ estimation uncertainties are assumed for both star and planet limb detection which is felt to be sufficiently conservative. This number is incorporated into the "estimated navigation angle measurement uncertainties" of Section 4.7 which are used in the "navigation error sensitivity analysis" of Section 3. It should be noted, as discussed in Section 4.7, that this uncertainty number does not include the km unit uncertainty term related to locator detection variations, nor the km unit term determined by uncertainties in the location of the phenomena, nor does it include angle encoder and structural induced uncertainties.

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There are a large number of scanner design variations which are possible in order to optimize the design for specified objectives. Variations in aperture geometry have already been mentioned. Multiple apertures opening onto a single sensor would permit reduced location estimation uncertainties. Multiple detectors behind a single aperture could measure different wavelength bands by using a ladder of filter mirrors to direct the energy to its respective detector. (Multiple wavelengths would be useful during horizon scanning to reduce the phenomena uncertainty, as discussed in Section 2.) Multiple apertures combined with multiple detectors in innumerable ways are also possible. These combinations could permit detection of many different phenomena; could allow optimization for each of the many types of navigation sightings encountered during a planetary mission; and could permit duplication for maximum reliability.

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It is worth commenting that in the design model a single set of optics is mated to an aperture-sensor system containing several apertures and sensors, rather than using several separate special purpose instruments. This is done in order to minimize the total complexity; to permit the most accurate relative alignment accuracy between the individual aperture-sensor combinations; and to provide the most collected energy per sensor for a given limited system volume.

There is an important and extended development task which remains to be done. That is to optimize the versatility, accuracy, simplicity, and reliability of a practical scanner design based on the simultaneous analysis of mission requirements, phenomena uncertainties, and locator definitions. The analysis and design of an optimal horizon detection system is one of the primary development requirements for a planetary mission.

4.5.4 Sun Sensor

The third type of sensor which is incorporated into the design model is a critical angle prism sun sensor which would sense phenomenon 6. This sensor is very simple and small in comparison to other sensor systems considered. Since this device is used only to detect the sun, its primary utility for a planetary mission would be for the IMU alignment function discussed later. The reason that this sensor would not be as useful for navigation as the two previous sensors is not due to the accuracy capability of the sun sensor but is due instead to the limited navigational utility of the sun as discussed in Section 3. Actually the critical angle sun sensor is extremely accurate for its size.

4.5.5 Onboard Radar

Radar devices have not presently been included in the design model discussed in the next section since the primary utility of onboard radar is for close proximity navigation and guidance. The emphasis of this study has been on the primary task of navigation, guidance, and control of the spacecraft trajectory. Little effort has been expended thus far on considering the other mission CG&N functions.

4.6 Description of RSS Design Model

The configuration shown schematically by design layout in Figs. 4-13 and 4-14 depicts a dual-line-of-sight, automatic photometric sextant. Before describing the design facets of this instrument, let us identify the ground rules under which the design was developed. These ground rules take the form of the following list.

- a. A fully automatic photometric sextant providing vidicon or other appropriate executive monitoring.
- b. Two line-of-sight, four degree-of-freedom instrument (3 base motion and 1 precision)
- c. Maximum spherical coverage
- d. Maximum optical aperture with reflective optics
- e. Maximum scatter shielding and protection
- f. Inclusion study of IMU components on outer stabilization axis to provide calibration potential for inertial units.

The mechanization shows an outer azimuth isolation axis member in the form of a hemisphere to which the six IMU components are attached. This hemisphere constitutes the equivalent of a navigation base reference between the inertial and optical references. It further contains a heat exchanger for thermal control. The allowable motion of the azimuth isolation axis is $\pm 360^{\circ}$. Electrical lead throughs are supplied by a symmetric twelve-member flex-print preassembled canister.

The two remaining base-motion isolation axes lie in the ground plane of the azimuth axis. This configuration eliminates the possibility of any gimbal lock conditions due to base-motion inputs as a function of sighting directions. The cross-trunnion or middle axis isolation member has $\pm 30^{\circ}$ of angular freedom from the ground plane. This angular freedom is considered adequate to accommodate base-motion inputs during active sightings.

The trunnion or inner isolation axis has $\pm 180^{\circ}$ of angular freedom from the ground plane. This freedom is supplied for dynamic motion as well as for the purpose of allowing the detector heads to be totally stored during their non-operative periods. Co-axial with this trunnion or inner gimbal axis is the precision sighting angle axis, which allows for relative motions of the lines of sight with respect to each other. Each line of sight is capable of non-occulted viewing from -20° to +70° referenced to the azimuth ground plane. Inasmuch as the two lines of sight are assembled back-to-back, the total angular viewing capability on the precision axis is 2 (+70°) or 140° between the two objects being observed.

The design model configuration is governed primarily by the navigation angle measurement requirements but as discussed in Chapter 4, it can also readily





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perform the alignment functions. The basic navigation package consists of an image dissector or vidicon tracker which is related through one degree of freedom (containing a servo drive and a 128-speed, high-precision resolver) to a scanner photometer. This angle measuring unit serves the function of a space sextant. The two-sensor subsystems may be switched in function as is appropriate during different phases of the mission. Either instrument can be used to detect star or solar system body features, and depending upon the situation it may be best for one or the other to sense the star, etc. (One such situation would be near a planet where the video instrument might be used for landmark tracking, and the scanner would then be required to detect star locations.) It is also possible for both instruments to detect the same target or two similar targets (such as a pair of stars) thus permitting inflight calibration of the sextant. Both radiation sensors use diffraction-limited folded reflective optical systems (as discussed in Sections 4.5, 4.8, and 4.9). A sun sensor is attached to the "back" of the image dissector tracker housing so that we can use the tracking servos of the tracker line of sight, and for in-flight calibration using the sun limb detection capability of the scanner instrument. A video display is used to present images of phenomena such as landmarks and horizons from the tracker instrument to the astronaut. The video display would also be useful to display data on horizon profiles, star scans, etc. measured by the scanner instrument. Comparative data could be brought up from memory and simultaneously displayed for astronaut analysis.

It is evident that the many navigation situations encountered during a planetary navigation mission including near-earth, missourse, intermediate planet range, and near-planet observations situations will require a great deal of multiple capabilities and versatility. The design model described provides simultaneous detection of the two lines of sight. The sequential measurement philosophy which uses a scanner photometer mated to an inertially-referenced angle encoder such as a ring laser is conceptually elegant since it is capable of the highest potential accuracy, and allows reduced servo performance from the attitude isolation system. By the addition of a $3/4 \times 6$ inch ring laser pancake on the side of the scanner photometer, it would be possible to include also the sequential detection capability, without major changes in the mechanical design or in the attitude isolation system. To achieve the 360° of rotation required for ring laser operation would necessitate a relatively minor modification in the flex-print design. The measurements can be taken with the system rotating in either direction so that at the end of a measurement cycle the direction of rotation can be reversed and another measurement performed in the opposite direction as the flex-print is unwinding. When used for sequential detection the ring laser would provide the primary precision angle measurement. The scanner-ring laser combination would be able to provide the most accurate measurements of all

limb-star angles which appear at this time to constitute the probable majority of the navigation measurements.

On the other hand, the design model utilizes a conventional precision-angle encoder which requires less development than a ring laser angle encoder. The design model is also more versatile since it contains a vidicon instrument (in addition to the scanning instrument) which would be additionally useful for detecting near planet navigation references (such as landmarks), for unusually high-background star sensing situations, for auxiliary vehicle navigation and guidance, and for general astronaut viewing of targets and points of interest. The tracking instrument is not as useful as the scanner photometer instrument for sequential measurements because finite rates have to be maintained in order to realize the high-accuracy capability of the ring laser; thus considering sequential detection alone, there is little motivation to include a tracking instrument. But as discussed above, the additional instrument used in the design model to enable simultaneous detection is also a very valuable instrument because of its versatility. One advantage that the inclusion of a tracking instrument would have for sequential measurements is that it would provide a second _attitude reference signal in addition to the one axis of information provided by the ring laser, thereby relieving the computer and the Inertial Measurement Unit (IMU) of most of the task of gimbal attitude control during navigation measurements.

Besides enhancing accuracy and versatility, this duplicate capability would also increase total reliability by permitting various backup modes in the event of a failure in some part of the navigation package. Another advantage is that with the inclusion of this sequential detection capability, it would also be practical to perform both simultaneous and sequential measurements at the same time which would facilitate an accuracy cross check between the measurements.

Thus it appears desirable to consider the incorporation of both simultaneous and sequential navigation measurement capabilities into one navigation package. The development of this inertially-referenced pancake angle encoder could be carried on in parallel with the development of the engineering model of the "sextant" instrument and, if successful, could be incorporated into the system by the time the design of a planetary navigation system prototype is initiated.

4.6.1 Mechanization to Support the Navigation Package

In the design model, the navigation package is isolated from the spacecraft by three degrees of freedom, each with a servo drive and precision angle encoder. The navigation package is positioned in attitude during calibration, target search, sextant angle measurement, sequential angle measurement mapping, and IMU alignment modes using IMU attitude information. In the eventuality of a complete lack of this information, it is possible to obtain this information by using the navigation system in various ways, as discussed herein. The three degrees of freedom may also be controlled by signals from the radiation sensors completely, or by some mixture of IMU and radiation sensor information. These three degrees of freedom plus the precision degree of freedom can also be controlled manually.

The complexity of the attitude isolation mechanization and sophistication of the servos required depends to a large extent upon the spacecraft environment and constraints. Vehicle attitude constraints, large RSS field-of-view requirements, spacecraft rotation rates and accelerations, linear vibrations and accelerations all impede the objective of complete independence from spacecraft attitude and motion. Depending upon the severity of the spacecraft attitude isolation "environment", various factors will need to be considered, such as: rigidity of gimbals and bearings, gimbal restrictions on the field of view, gear drive vs. torquer servos, inertial rate sensors on the tracker housing for servo damping, angle encoder bandwidths, computer computation and communication rates, etc.

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The attitude isolation design required to isolate the navigation instrument is the portion of the development most dependent upon the spacecraft definition and thus has received a minimum of emphasis until such time as a spacecraft environment is specified. The attitude isolation systems used in the design model should afford the maximum independence of the spacecraft attitude environment, but once a spacecraft environment is specified, various modifications may be more optimal.

Presently, it appears that practically all communications to and from the radiation sensors subsystem can be handled through the common computer communications bus. The only exception considered thus far is the probable interface with a video display to the astronaut.

The radiation sensor subsystem interface with the computer will probably be defined at the communication format and timing logic which is tied to the communication bus. It may be desirable to define the interface even closer to the radiation sensors and control circuitry so that the A to D and D to A registers and other digital devices are included as part of the computer system responsibility.

The other major interfaces considered, in addition to the mechanical and communications interfaces, are the power, ground, and possibly the temperature control interfaces. Whether the RSS should include some power supplied or not would have to be decided after the spacecraft is defined. Except for peculiar power requirements, it would seem reasonable that most of the specified power types would be supplied by a central power system which would service the total guidance and navigation system. There are similar possibilities for the electrical ground and the temperature control systems. The system that is described above is an elemental radiation sensor subsystem. Due to the stringent reliability requirements and/or due to obscuration of navigationally useful regions of space from a single location on an unmaneuverable spacecraft, it may be desirable to utilize redundant Guidance & Navigation systems which are deployed at several locations on the spacecraft. It appears preferable at this time to use identical redundant systems instead of several different designs in order to simplify the design of the total system and, if any repair is required, to facilitate modular repair.

4.6.1 IMU Alignment Function and Design Concepts

The navigation function discussed previously requires accuracies in the arc-second range. IMU alignment accuracy from the radiation sensors subsystem in the arc-minute range is required for spacecraft guidance functions, and to a somewhat lesser accuracy for establishing the planes of the navigation angle measurements.

It appears that the fine alignment of the IMU (from the arc-degree range down to an arc-minute and less) could easily be performed by the navigation measurement system.

It seems reasonable at this time to assume that there will be suitable accurate coarse alignment information available during most of the mission due to spacecraft attitude requirements for communications antenna pointing and/or solar panel positioning. Any solar heating or artificial gravity rotations would also require attitude information which would restrict the attitude search field required for IMU fine alignment.

If, during some interval, coarse attitude information is not available, it is still possible to automatically acquire two axes of the IMU alignment with the use of the navigation instrument sun sensor. The design model scanner instrument could easily determine the attitude in the remaining degree of freedom. There is one situation in which the sun sensor could not be used; that is when the spacecraft is on the shadowed side of a planet. (It is anticipated that several identical systems will be deployed around the spacecraft so that shading by the spacecraft should not be a problem.) The scanner in association with the AC G & N IMU would have the ability to acquire all three alignment axes through star mapping and correlation techniques and could be used in such a situation to supply complete attitude information. The AC G & N design would utilize the IMU as the inertial memory required to develop the field of view from scans of the instantaneous field of view. The small instantaneous field of view and high power are desirable to minimize the scattered light and thereby to maximize the star visibility. A computational technique for obtaining inertial attitude which uses only the angle between stars, and which does not require the use of any star intensity or color information has been reported.⁽⁵⁾ Using the contemplated advanced computer, the required computations should not be at all prohibitive.

If an astronaut were free to assist with a coarse alignment, another alternative would be for him to perform an approximate determination of the spacecraft attitude. By viewing space through one of the windows which will probably be available, and with the assistance of some simple optical aids, it should be possible to fairly easily perform a satisfactory coarse alignment to within the field of view of the navigation sensors. The fine alignment could then be performed by one of the navigation instruments.

Because of these several alternatives, no effort has been expended in this study toward the development of a separate attitude-finding instrument, since the navigation instrument can perform all of the alignment functions considered thus far. As the manned mission requirements become defined, variations in these functions or additional tasks may indicate the need for a separate, steerable, wide-field-of-view-manned viewing instrument.

4.6.3 Other Radiation Sensor Subsystem Functions

Two other functions of the radiation sensor subsystem have been suggested which can be achieved by navigation instrument design. One is the function of low-power, long-duration attitude determination and stabilization for those periods in which IMU power or life-time degradation are critical considerations. The other is the navigation and guidance of associated vehicles or probes. The requirements for this information will become clarified as specific planetary missions are defined.

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4.7 Anticipated Navigation Angle Measurement Uncertainties

The sequential measurement technique utilizing a scanner photometer mated to a ring laser is expected to make navigation angle measurements accurate to 1-arc sec, 1σ . This is a technique which has been proven in parts, but the building and testing of a total package is yet to be done. It is the only system concept that we know of that could be available in a reasonable time and practically provide one arc second accuracy. The ring laser encoder and mechanical stability errors are expected to contribute negligible uncertainties to the total. The mechanical stability errors are believed to be negligible relative to the detection uncertainty since there are no bearings between the scanner instrument and the inertial angle encoder, and the mechanical path between the two lines of sight is "short". Also, the scanner discussed in Section 4.5 is a very rigid design, as is the ring laser, and the two are rigidly mated to each other. Only very short-term mechanical instabilities are of significance since the unit performs an intrinsic calibration during every measurement. The primary source of error for this system is due to detection uncertainties which are expected to contribute nearly all of the 1-arc sec, 1\sigma uncertainty.

The design model which uses electro-optical simultaneous detection of the two lines of sight is expected to be able to make navigation angle measurements accurate to 3-arc sec, 1σ . The detection uncertainties are assumed to be the same as for the sequential system, that is 1-arc sec, 1σ . The primary source of error is due to the precision angle encoder errors coupled with the minimum mechanical flexibilities allowed by the precision-axis bearing. It is anticipated that these two errors will contribute just less than 2-arc seconds, 1σ of uncertainty for a state-of-the-art angle encoder and bearings. The remainder of the error is due to the "longer" mechanical path between the two lines of sight and the associated additional small flexures.

Two types of sextant designs which use superposition of two lines of sight and visual detection (see Section 4.4), are expected to make navigation angle measurements accurate to 10-arc seconds, 1 σ . The inherent two-for-one multiplication in the articulation of the precision angle encoder produces an uncertainty of just under 4-arc seconds, 1 σ allocated to the angle encoder and bearing assembly. The main source of error is due to the difficulties associated with manual detection of the superposition of dissimilar targets, primarily stars and horizons. As stated previously, atmospheric horizons are a difficult phenomenon to detect manually but are of principle importance for planetary missions. Repeatibility of manual image superposition may be less than 5 seconds, 1 σ under favorable conditions but absolute accuracy is more difficult to achieve since the locator of a distributed phenomenon is difficult to uniquely define as required for navigation. The locator "points" marked by different individuals will vary as will the points marked by one individual on successive days due to variations in training, judgement, adaptation level, and physical condition. There is the compounding difficulty that even with high-power optics and high-performance servos small unavoidable image motions will somewhat degrade the manual detection accuracy. Thus we feel that 10-arc seconds, 1σ uncertainty in the angle measurement is a fairly accurate representation of this instrument's capabilities.

Under favorable conditions, somewhat smaller uncertainties than those quoted above would be obtained for all three instruments. These numbers could also be improved by the evolution of improved technology (although there is not much time for the development of new technology if it is to be operational in time for the first generation of manned planetary mission being considered). On the other hand, the planetary missions being considered are long and there would be a lengthy period without the close assistance of sophisticated ground support, and thus it is better to be somewhat conservative at this point. The navigation error sensitivity analysis has shown that the accuracy quoted for the design model is adequate. Reliability and simplicity of operation are criteria which are just as important as accuracy for the optimization of the RSS design. The sensitivity analysis results indicate that there is no immediate need to press for more accuracy and therefore it is better to seek maximum reliability and simplicity of operation.

These numbers are felt to be as representative as possible prior to actual development. Peak errors are expected to be approximately three times the 1σ values given. A Gaussian error distribution is assumed for the error sensitivity analysis in Chapter 3. This statistical description is felt to be sufficiently valid for the purposes of this study. It should be noted that these numbers assume the performance of in-flight calibrations to minimize uncertainties due to drifts. It should also be noted that these errors are not all inclusive. In addition to these angular measurement uncertainties, the total navigation measurement uncertainty must include a phenomena and detection uncertainty term, in km units. The anticipated magnitudes of this term are discussed in Chapters 1 and 2, and are included in the error sensitivity analysis.

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4.8 Folded Optics Design

The ability of the available optical design techniques to satisfy the presently anticipated needs of an advanced CG&N system has already been demonstrated both by work accomplished during the course of the Mainline Apollo Program and, more recently, by the design of a number of telephoto systems of very long effective focal lengths for star tracker systems, and also several UV camera lenses.

There are several computer-aided optical design and evaluation programs which are used to develop these designs which are being continually updated. Conic section, refractive and reflective elements can be handled by the optimal design program. A wide range of design constraints and a variety of optimization criteria can be put into the program in order to optimize specific design objectives. A versatile ray-trace evaluation program and a tolerance evaluation program are also available to support a design or to analyze existing systems.

The primary new program that is being developed is a scattered light evaluation program. This program (discussed in the next section) is the basis of a restriction which has impact on the optical design: In order to alleviate a potentially serious scattered light problem, no glass elements can appear in the direct path of sunlight, i.e. in the forward section of the optical system. This criterion plus the desire for conserving physical space leads to the use of a predominantly reflective design of a Cassegrainian type and eliminates the use of the family of Schmidt-type objectives because of their employment of a large correcting lens up front.

The following paragraphs discuss the MIT Instrumentation Laboratory investigation and experimentation in optical system design for advanced CG&N systems. Our project objectives are discussed along with our progress to date, including our suggestions for future effort.

4.8.1 Project Objectives

- a. To update computer programs to accomplish general optical designing and testing.
- b. To develop practical, diffraction-limited, very long effectivefocal-length optical systems which are miniaturized so that they can be contained in a very small volume.
- c. To determine the optimal optical designs for the navigational measurements considered.
- d. To design optical systems which are optimum for the requirements of the space sextant designs considered.

The basic considerations in the design of the optics for an automatic sensing device are:

- a. The resolving power of the sensor (e.g. lines per inch)
- b. The angular accuracies required to perform the necessary tasks
- c. The mechanical constraints to be anticipated, especially on the basis of previous experience with Apollo designs

The resolving power of the sensor will determine the best F-number (focal length/collecting aperture) for the optics because:

- a. The linear size of the diffraction pattern is a function of F-number
- b. The size of images produced by a point source should be made about equal to that of the smallest discs distinguishable by the sensor when placed with their radii apart from one another. For the space sextant presently under design, we have assumed the use of a vidicon or image dissector capable of resolving 0.001 inch diameter images. Consequently, an F/20 system is being considered.

The collecting aperture is determined by the trade-off between the desire for high signal strength and the need for minimum system size. A 2- to 2 1/2-inch aperture system is planned for each half of a space sextant. Image sizes of less than 4-seconds of arc over the field result from this aperture size.

The mechanical considerations for the severe environment and limited space available require the use of as few optical components as possible and a highly-folded, telephoto-type design. Whenever feasible, optical components should be hard-mounted to a single main housing. (In the Apollo Alignment Optical Telescope (AOT), all the lenses and lens assemblies had to be epoxied in place.) The main shift to be eliminated is an axial change in focus since small x-y-type displacements can be calibrated out by the two-line-of-sight instrument. A very careful choice must be made of the material for the primary and secondary mirrors in order to minimize optical power changes caused by temperature variations.

Two types of designs have been investigated extensively, each type being developed for overall lengths (first element to focal plane) from 5 to 12 inches, 50 inches effective-focal-length, one degree field of view, and 2to 3-inches aperture. The first type is a simple Cassegrain with conic primary and secondary. Figures 4-15 and 4-16 show examples of a short and long version of this type. The long effective focal length is produced by placing the





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Fig. 4-16 Reflective Cassagranian for design model package.

secondary very close to the focal point of the primary. If the separation between the primary and secondary is at least 4 inches (for 2- to $2 \frac{1}{2}$ -inch apertures), geometric images smaller than the diffraction limit can be maintained over no more than a $2/3^{\circ}$ field of view in a single plane. Attempts have been made to develop a field flattening lens to be positioned just in front of the focal plane, as shown in Fig. 4-16. The use of a single element has the following complications: A higher order asphere is required (a sinusoid shows the most promise). Also the glass thickness required at the edge of the field of view in order to fully flatten the entire field introduces lateral color which reduces the effectiveness of the lens near the edges of the field. Thus a less than full flatness-correcting element was tried in order to avoid lateral color problems, and has thus far improved the diffraction limited focal plane up to 0.85° with negligible lateral color effects. The other lantenative, which was selected as preferable for the design model, was to flatten the field so that it would be diffraction limited over the full field of view and suffer the small lateral color effects at the edge of the field of view. For all of the situations considered this shift would be insignificant since its only effect would be a change in scale factor at the edge of the field. Each of the navigation phenomena considered lies within a sufficiently restricted spectral range that there would be no significant blurring of the images. To correct the color and flatten the field simultaneously would require three lenses and would result in a design essentially comparable to the second type investigated. One difficulty with the three element system is that some of the glasses required would only transmit light near the visible region of this application. The single element corrector selected for the design model can be made of quartz to allow use of the optics from the UV through the IR.

The second type of system consists of a Cassegrain of lower power (approximately 25-inch EFL) and a Barlow lens (a negative lens of about 2 power). The most compact version of this type is shown in Fig. 4-17. This kind of system balances off the sensitivity between the reflective and refractive parts, and produces diffraction-limited images over the whole field in a single plane. The Barlow consists of a cemented triplet placed between the primary and the focal plane and color corrected over the visible spectrum (a larger range than originally anticipated). The same glass types have proved usable over the entire range of systems including the most recent: a 3-inch aperture, 86-inch EFL, 11 1/2-inch long system (Fig. 4-18). The disadvantages of the reflective Barlow design are the additional lens mounting required and the poor transmission of the glass below 4,000Å. The vacuum shift of focus has been found to be negligible.



Fig. 4-17 Reflective Barlow short system.


In the case of the space sextant, the shortest optical system does not result in the most compact packaging. Because of the large size of the vidicons (approximately 2 1/2- by 4-onches), it is most efficient to place the sensor parallel to the Cassegrain and use redirection mirrors to transfer the optical axis (see Fig. 4-8). This packaging calls for about a 10-inch long optical system before folding, for a 2 1/2-inch aperture.

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In general, it appears that the reflective Barlow design will give the best performance with a sensor which has sensitivity limited to the visible region. The all-reflective design will give the best all-around performance over the full spectral range from the UV through the IR and the 1° diffractionlimited, color-corrected FOV is felt to be adequate for the requirements of the space sextant. The amount of lateral color shift beyond 0.85° does not cause a serious degradation for the applications considered. If larger fields of view are necessitated by modifications in the concepts assumed herein, then a reflective-Barlow system could be incorporated into an instrument design. Thus, because of its versatility, simplicity, and suitable performance, the all reflective system shown in Fig. 4-16 and 4-8 has been chosen for incorporation into the design model.

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4.9 Stray Light Evaluation Program

One of the critical characteristics of an optical system design for space navigation and alignment is its susceptibility to stray light from outside the field of view. Reference phenomena (such as stars, distant planets, limbs, beacons, etc.) in general provide very low power, relative to the intensity of the sun, or in comparison to the light received from a near sun illuminated planet.

A program has been developed for the purpose of supporting the design of optical systems with low susceptibility to stray light. The availability of this type of evaluation program early in the optical design stages eliminated much of the "cut and try" design iterations. With this type of analysis tool it is possible to achieve a more optimum design within given time and budgetary constraints.

The function of this program is to determine that fraction of stray light incident upon the telescope which reaches the focal plane. An elemental case is represented by a plane wave of stray light incident upon the entrance aperture and outside the field of view, at a given attitude relative to the optical axis. In general, some fraction of this power per unit area will reach the focal plane, producing an effective background illumination some power per unit area on the focal plane. The attitude of this beam of light constitutes an important parameter in determining the magnitude of this fraction. It is possible to plot, as a function of this attitude angle, the ratio of the incident stray light which reaches the focal plane. The stray light which would reach the focal plane from a more complex "hemispheric" distribution of stray light can be calculated by linearly superimposing a combination of the elemental plane wave cases discussed above.

There are two objectives of the program. One is to indicate sensitivities to optical system design parameter changes in order to determine which design changes will most improve the design. These sensitivities can be determined at whatever incident stray light attitude and intensity distribution is considered to be the most indicative of a given application.

A second objective is to quantitatively determine the numerical ratio described above in order to evaluate actual sensor performance (trackers, scanners, etc.). This is the more difficult of the two since it requires the most accurate quantitative simulation of the optical system design. Accurate data on the specular and nonspecular reflectivity of the surfaces, and the scattering characteristics of any transmissive elements, are required in order to obtain accurate data on the effective background illumination.

A future objective is to develop an optimization program which will use the sensitivities derived by this program and iteratively redesign portions of the stray light baffling toward a selected optimization criterion. This would be a more complex program but it would eliminate much of the human optimization that would need to be done if only the sensitivity analysis program were ϵ vailable.

The present scattered light evaluation program is formulated to include specular reflections of non-optical surfaces. Both specular and non-specular reflections are included for optical surfaces. Scattering on or within transmissive elements has not thus far been included since the most promising optical designs avoid transmissive elements near the entrance aperture.

The present program is essentially a ray-trace program which traces each of a group of rays until each ray is reflected out of the telescope or is reflected a sufficient number of times to be effectively extinguished, or reaches the focal plane. A sufficient number of appropriately distributed rays are required to allow a statistically meaningful sample to reach the focal plane. Naturally, the better the telescope scattered light performance, the more incident rays will be necessary in order to obtain a large enough sample. The large sample is needed in order to correlate the fraction of rays which reach the focal plane (including attenuations from each scattered event before reaching the focal plane) to the fraction of total power.

The distribution of rays is also important so as not to bias the fraction toward one stray light effect, which would produce an inexact power fraction calculation. To determine the power fraction from the fraction of rays reaching the focal plane, a ray is effectively correlated with an area whose scale is approximately equal to the separation between adjacent rays.

One difficulty in using a purely ray-trace technique is the cumbersome evaluation of baffle designs which use diffuse black surfaces; especially in a good baffle design where many ray bounces are produced before a ray reaches the focal plane. At each reflection, each ray becomes an infinite number of new rays which distribute in direction and magnitude as determined by the character of the diffuse reflector. In practice, only a relatively small finite number of rays can be generated and analyzed at a ray reflection so that again, care must be taken to assure that this set of generated rays also represents a statistically meaningful sample.

For the evaluation of diffuse surfaces a program formulation using continuous distribution reradiation patterms is probably more appropriate than a purely ray-trace formulation. A continuous distribution type of formulation only becomes comparatively practical when high-performance baffle systems with diffuse scattering elements are to be analyzed. To utilize this type of formulation it would be necessary to first determine the reradiation character of each type of baffle element. The necessary functions for a general configuration of this baffle element could be obtained by use of the ray-trace program, and/or by photometric experiments. Only a few general types of baffle elements are usually used in a design and they are usually repeated many times so that a function derived in this manner would then apply for many of the baffle elements to be analyzed. This formulation would thus relieve the stray light program used to evaluate the total optical system from having to effectively develop this function for each baffle element and for each geometry change.

4.9.1 <u>Cassegrainian Telescope Baffle Design</u>

The Cassegrainian telescope shown in Fig. 4-16 was used as the test system for developing the scattered light evaluation program. The analysis capability of the program to determine the stray light sensitivity to variations in baffle geometry was used to develop the baffle design used in the design model as illustrated in Fig. 4-8. There are some general design features that can be indicated. For the optical system shown in Fig. 4-8 no unreflected stray light can reach the focal plane due to the pipes inserted between the primary and secondary mirrors. Thus, in order for stray light to reach the focal plane, the focal plane must be able to "see" some surface or scattering element which is itself illuminated by stray light.

One way that stray light can reach the focal plane is through nonspecular reflections from the primary mirror which can be seen by the focal plane. For stray light incidence angles greater than approximately 20°, no direct stray light can reach the primary mirror since it is considerably recessed from the entrance aperture. For stray light incidence angles of less than 20°, direct stray light can reach portions of the primary mirror. The amount reaching the primary mirror will usually increase as the angle is decreased toward the optical axis. In general, for a mirror in good condition and relatively free of dust, the amount of the non-specular reflection which would reach the focal plane is still relatively small but can still be the predominant source of stray light.

For stray light incidence angles larger than 20°, the primary mirror must be scattered from the more forward surfaces of the housing. The outer diameter (OD) and the inner diameter (ID) stops of the system are formed by the extremes of the exposed annulus of the primary mirror. This prevents the focal plane from "seeing" these forward internal surfaces of the housing by direct specular reflection, and only non-specular scattering will allow the focal plane to receive any of this radiation

The forward internal surfaces of the housing and the external surface of the pipes are broken up with baffles so that only a small portion of the radiation incident upon the forward surfaces can reach the primary mirror. The edges of these baffles are made as sharp as possible so that the effective scattering area will be minimal. It appears at this time that the edge contributes less stray light if it is very sharp, even if it has a relatively high reflectivity, than it would when covered with a low reflectivity coating which would increase its scattering area.

The back surfaces of the baffles are slanted so that no direct stray light can illuminate the back surfaces which could then scatter some of this radiation to the primary mirror. Thus, multiple reflection would have to occur before any stray light could reach the primary mirror where nonspecular reflection could then scatter some of this radiation toward the focal plane.

The baffles are made as deep as possible within the mechanical design volume constraints so that as many reflections as possible will occur within each baffle cavity before it leaves the cavity. For a given limitation on depth it is desirable also to put the baffles close together, so as to further increase the number of reflections prior to the ray leaving the baffle cavity. There is a disadvantage to spacing the baffles too closely since this effectively increases the total area of baffle edges which can scatter stray light; thus a tradeoff must be made.

It is preferable to have the opposite surfaces of a baffle cavity somewhat close to parallel, in such a way that as many internal reflections as possible will occur within the cavity before the ray can escape, thus effectively trapping the ray. The actual optimum angular relationship between the two surfaces depends upon all of the geometry of the situation and the baffle design. The variation in tilt of the baffle allowed by the constraint which prevents the back side of the baffle from being directly illuminated by stray light, allows the opposite surfaces of a baffle cavity to be more nearly parallel than would be structurally feasible if they all had the same slant.

As stated previously, the OD and ID aperture stops are placed at the primary mirror. This placement is preferable to placing the stops near the entrance paerture since the edges of the system aperture stop can be "seen" through mirror specular reflections by the focal plane. The further the edges are from the entrance paerture, the less likely the edges of the stop will be illuminated.

There is one aperture stop which is presently near the entrance aperture. This is the "spider" support for the secondary mirror. To minimize the amount of scattered light which reaches the primary mirror, the sides of the three rectangular supports (Fig. 4-8), which are nominally parallel to the optical axis, are slanted toward the front of the optics so that the sides of the supports cannot be seen by the primary mirror. Also, the rearward edges of these supports are made as sharp as possible so as to minimize the effective reflective edge area that can be seen by the primary mirror. Calculations have shown that these edges will contribute much less stray light than would a glass window support of the secondary mirror. It is also possible to place masks, somewhat larger than the projection of the spider supports, at the primary mirror thus forming the stop at the mirror.

Other concepts reflected in the design shown in Fig. 4-8 are that it is easiest to shield a small field of view optical system and/or a longeffective-focal-length optical system from stray light.

Another way that stray light can reach the focal plane is if the secondary mirror or portions of the inside of the pipes which can be seen by the focal plane are illuminated by stray light, but this is a secondary effect. There are various other paths of lesser significance to the focal plane which will not be enumerated.

Fortunately, the program developed does not require the foreknowledge of what all of these paths are, if the system is modeled in a proper manner. The program effectively searches out these paths and indicates the sensitivity of this path to design changes.

4.10 Flashing Light Visibility

4.10.1 Introduction

During planetary missions, it is expected that many rendezvous and clocking maneuvers will be performed. The visibility of a flashing beacon has important implications in the design of rendezvous guidance systems to aid in these precise maneuvers. This section discusses the MIT Instrumentation Laboratory investigation into the visibility of a flashing beacon.

Flashing beacons are motivated by the desire to maximize target visibility while minimizing average electrical power consumption, a chronic concern in spacecraft designs. Previous experiments have determined that greater target visibility can be obtained by storing energy and periodically releasing it in a high-intensity flash than can be achieved by a continuous source using the same average power. New data are presented herein on the improvements that can be achieved in beacon power consumption by means of the flashing light technique. One of the most significant results of this work is the development of a simple and accurate technique for obtaining several types of flashing light visibility data.

4.10.2 Flashing Light Visibility Experiment

The flashing light visibility experiment, originally planned as a simple test, initially sought to determine the ratio of the luminous power from a steady point source to the average luminous power from a flashing light source; both near the visual threshold.

A translation of the classic paper on this subject by Blondel and Rey (Journal de Physique, Ser, 5, Vol. 1, p. 530, 1911) by Dr. Arthur C. Hardy. (MIT Physics Department) in collaboration with Instrumentation Laboratory personnel, matured into IL Report E-1942, <u>On the Perception of Flashes of Light at the Limit of Their Perceptibility</u>, January 1964. Briefly, when Blondel and Rey attempted to compare the conspicuity of a steady source with that of a flashing source, they encountered difficulties which they avoided by causing both sources to flash, one with a constant time-duration of 3 seconds, and the other with a variable time-duration down to 0.001 second. However, it can be questioned whether a flash of light that persists for 3 seconds is the equivalent of a steady source.

Further provocation for this study was prompted by a one-page reference to an assembly called a Rendezvous Evaluation Pod (REP). It was there estimated that the flashing lights on the REP at 20 nautical miles would be equivalent to the intensity of a third magnitude star. Since the literature would not enable such an estimate to be made with confidence, it was proposed that the Blondel-Rey ratio be determined experimentally by a new procedure that eliminates virtually all photometric uncertainties.

4.10.2.1 The Flashing Mode

The source of light used in this experiment was an incandescent lamp with a horizontal tungsten filament approximately 2 mm wide. The effective length of the filament was reduced to 2 mm by an aperture adjacent to the envelope. These 4 square millimeters of hot tungsten were reduced to a "point source" by a 10× microscope objective turned end for end. As a matter of convenience, the observer sat beside the source, which was in a ventilated, light-tight box, and he observed the point target by reflection from a firstsurvace mirror on the opposite wall of the room; the total distance from the target to the observer being some 25 feet.

In the plane of the target was an opaque disc rotating at 30 rps. A radial slot in this disc produced 30 flashes of light per second, each with a duration of approximately 0.001 second. Another disc rotated about the same axis at only 1 rps. This latter disc occulted 29 of the 30 flashes per second; so that, in the flashing mode, the observer saw one brief flash every second.

When making observations, the observer looked with one eye into a tube which allowed him a 10° field of view. This tube contained a sheet of glass which served as a beam-splitter; and the luminance level of his field of view was provided by an auxiliary diffuse source attenuated by appropriate neutral gelatin filters.

In the space between the lamp and the microscope objective was a trough with slots to support neutral gelatin filters for coarse adjustment of the intensity of the point source. The fine adjustment was provided by a neutral density wedge and compensator, the wedge being moved longitudinally by a micrometer screw. In the days when it was believed that a visual threshold represented a discontinuity between "seeing" and "not seeing," the pro-cedure would have been to set the wedge near its midpoint, to find a neutral filter that reduced the intensity of the target to a value close to the visual threshold, and then to determine the "end point" by adjusting the neutral/ wedge. This procedure was again used with the addition of a special feature, an occulting filter, whose optical density was ordinarily 0.3 (transmittance = 50%). The observer could introduce this occulting filter into the target beam at will by depressing a foot switch. His end point was the setting of the wedge that enabled him to see the flashes without difficulty before introducing the occulting filter, but to miss seeing the first flash after the occulting filter had been introduced.

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4.10.2.2 Precision

In view of the problems normally encountered in threshold determinations, the ease and precision of this occulting-filter method were found to be surprisingly satisfactory. As a sample of intra-day and day-to-day reproducibility of the wedge settings, the following table of raw data is included for one observer on three consecutive days. On these three days, the 10° field was matched in color temperature to that of the service lamp in a Macbeth illuminometer, and the luminance level was adjusted to 10 foot-lamberts. A neutral gelatin having an optical density of 3.0 was then introduced to reduce the luminance of the field to 0.01 foot-lambert, a little more than ten times the luminance level at which a point target is seen more readily by averted vision. The observer was allowed to adapt to this luminance level; and thereafter was encouraged to pretend that he was looking out through a window of a spacecraft or through an onboard instrument. The wedge settings given in the three columns represent the end points (in millimeters) which caused the observer to miss seeing the flashing target immediately on introducing the occulting filter. After each reading of the wedge position the wedge was backed off (toward the thinner end) by an arbitrary amount, and the observer thereafter had to make some six or more adjustments of the wedge before reaching a new end point.

Sept. 12	Sept. 13	Sept. 14
<u>54.8 mm</u>	55.8 mm	52.8 mm
58.3	55.7	54.6
57.8	54.2	54.4
58.7	56.7	54.3
58.1	54.7	54.3
56.7	55.9	56.0
56.1	57,5	54.3
54.8	56.6	55.8
55.2	56.0	56.4
Mean = 56.1 mm		Mean = 54.8 mm
mean = 50.7 mm a.d. = 1.3 mm	a.d. = 0.8 mm	a.d. = 0.9 mm

The mean value of the wedge settings on the three consecutive days is shown at the bottom of each column, as is also the average deviation (a.d.) of the individual settings. It will be seen that, on the average, a single setting differed from the mean value by about one millimeter on the wedge. The nominal wedge constant (units of optical density per millimeter) being 0.02, a displacement of the wedge by one millimeter corresponds to a change in the

effective intensity of the target of less than 5 percent. This implies that the mean of 10 settings should have an average deviation of some 1.5 percent.

4.10.2.3 The Steady State

In principle, the threshold of this source in the steady mode could be obtained merely by opening the motor circuit at such an instant in time that, in the final stationary position, the apertures in the two discs would be open. Obviously, the observer would then receive approximately 1000 times as much luminous energy per second as when both discs were rotating. In the flashingmode tests reported above, it has been necessary to insert a neutral gelatin filter having an optical density of 2.0 between the lamp and the microscope objective to bring the threshold settings near the midportion of the neutral wedge. If the human eye were capable of integrating luminous power over an interval as long as one second, an additional neutral density of 3.0 would be required to do this in the case of the steady mode. Some preliminary experiments showed that the required density would be in the neighborhood of only 2.3; but even an optical density of 2.3 (transmittance = 0.5 percent) cannot easily be measured with the accuracy warranted.

In the procedure actually employed in these tests, the steady mode was simulated by stopping the disc that rotated at 1 rps and allowing the other to continue to rotate at 30 rps. Then, if the frequency of the power supply is a constant 60 cps, the observer will receive 30 times as much energy per second as he did from a single flash per second. Under the conditions of these tests, no 30-cycle flicker was observed; indicating that Talbot's law (1834) applies. As a consequence, the visual effect of this pseudo-steady mode is the equivalent of a steady mode having the same average luminous power content. It bears noting that although the nominal duration of the individual flashes was 0.001 second, the actual duration did not need to be known because 0.001 second is well within the region of brief flashes for which Bloch's law (1885) applies.

It was predicted from some preliminary tests that, if the threshold settings for the pseudo-steady mode were to fall near the midportion of the neutral wedge, an auxiliary neutral filter with a density of 0.8 would be needed between the lamp and the microscope objective. With such a filter in place on the three consecutive days mentioned previously, the same observer made 10 wedge settings for the pseudo-steady mode, using the occulting-filter method. The mean values of these 10 settings (in millimeters) were 56.5, 55.9, and 54.1, respectively, and the average deviations of a single setting (from the mean of 10 settings) were, respectively, 0.8, 1.0, and 0.5 millimeters. The identical tests were repeated on the three days mentioned with a neutral filter having an optical density of 0.7 instead of 0.8. The mean wedge settings in this case were, respectively, 59.8, 57.9, and 57.9, and the average deviations of a single setting from the mean of 10 were 1.1, 0.7, and 0.7 millimeters. respectively.

It will be noticed that the mean wedge settings obtained with the two auxiliary filters (0.8 and 0.7) bracket the settings obtained in the flashingmode tests. The wedge constant (units of density per millimeter) can be calculated in each case from the wedge displacement corresponding to a density difference of 0.1. Then, by a linear interpolation, the effective densities (auxiliary filter plus wedge) were found to be 0.80, 0.79, and 0.78, respectively. These values of optical density correspond to transmittances of 0.160, 0.162, and 0.166.

As mentioned at the outset, the purpose of these tests was to determine the ratio of the luminous power from a steady source to the average luminous power from a flashing source, both being at or near the visual threshold. It was shown above that the average luminous power was increased 30-fold when the switch was made from the flashing mode to the pseudosteady mode. However, as indicated in the preceding paragraphs, the average luminous power received by the observer in the pseudo-steady mode was attenuated by 0.160, 0.162, and 0.166, respectively. Multiplying these numbers by 30 gives 4.80, 4.86, and 4.98, the mean value of which is 4.88. It is therefore concluded that this observer, under the above-recited conditions, required 4.88 times as much average luminous power when the stimulus was effectively steady as he did when the stimulus consisted of brief flashes at intervals of one second.

4.10.2.4 Effect of Adaptation Level

On a single day, another observer determined the above ratio by the identical procedure except that he made only a single careful adjustment of the wedge, this time varying the luminance to which his eye was allowed to adapt. His results are shown in the table below, where B_0 is the adaptation luminance in foot-lamberts and R is the ratio of the average luminous power received by the observer from the pseudo-steady source to that received from a source that flashed briefly only every second, both sources being near the threshold in each instance.

It will be noted that the ratio R appears to be nearly independent of the observer's adaptation level. The relatively high value of R obtained by the observer at 10 foot-lambers was found by him on several subsequent occasions.

R
5.68
4.84
4.78
4.72
4.73

He complained that when he looked into a 10° field at this luminance level, it seemed so uncomfortably bright that he had little confidence in his wedge settings. Possibly the optical density of the occulting filters should have been increased at the higher adaptation levels, or it may be that the value of R is actually higher above 1 foot-lambert. In any event, the performance of the eye at high adaptation levels, although perhaps of physiological significance, would not be important in rendezvous maneuvers because a flashing source that would be visible against an intensely lighted background would require inordinate amounts of average power.

4.10.2.5 Color of the Target

It should be mentioned that the current through the ribbon filament lamp was maintained with the utmost care at its rated 18 amperes. Obviously, the ratios obtained in these tests, although valid for this source, might not be applicable if the source had a different spectral composition, as it would, for example, in the case of a gaseous discharge tube. Later, another observer investigated the ratio R at $B_0 = 0.01$ foot-lambert with a Wratten #29 filter in the target beam. This is a deep red filter which is virtually monochromatic visually, and is as unlike the previously used unfiltered tungsten radiation as could easily be arranged. Nonetheless, the ratio R for this deep red target was found to be 4.62, surprisingly close to the value found by this observer for the unfiltered radiation.

4.10.3 Summary and Conclusions

1. The indications of the tests reported above are that, at the visual threshold, an observer requires almost five times as much average luminous power from a steady point source as he does from a point source of the same spectral quality flashing briefly at a rate of one flash per second. Or - inversely - a source that flashes briefly once every second requires only about 20 percent of the average power that is required to maintain a steady source of the same spectral quality - both sources being at or near the visual threshold.

- 2. The precision of the occulting-filter method used in these threshold determinations was surprisingly high. The indications are that the method can be used to compare sources of radically different spectral compositions. Presumably, the results would then depend on the luminosity function of the observer. Conversely, this method might be of scientific interest in examining subjects with abnormal luminosity functions.
- 3. Thresholds determined by this occulting-filter method are undoubtedly higher than those determined statistically by, for example, a forced choice type of experiment. On the other hand, thresholds determined by the latter method are customarily multiplied by a field factor suitable for the task at hand. In a sense, the occulting-filter method has a "built-in" field factor.
- 4. The procedure devised for these tests should greatly simplify the determination of the number of candle-seconds of luminous energy emitted by a discharge tube that flashes periodically. A microscope objective turned end for end (or equivalent) could be made to form an effective point image of such a source; and the threshold of that point image could be compared with a similar point image of another flashing source. If desired, the comparison source could be a tungsten lamp and a pair of rotating sectored discs. Then, by removing the discs, the intensity of the tungsten source could be determined for the steady state by conventional photometric techniques.

4.11 Unknown Landmark Tracking

4.11.1 Devices

The instruments applicable for automatic unknown landmark tracking can be grouped into three major categories:

1. area correlators

- 2. scanners
- 3. parallel-slit reticles

The ensuing sections will treat each device separately and specify the advertised accuracies of the devices involved.

4, 11.2 Area Correlators

Area correlations are generally used for known landmark identification and tracking. It is also possible to use them for unknown landmark tracking through suitable aspect angle changes.

Area correlators perform an optical correlation between a prepared map, sketch or photograph and the terrain containing the landmark. Bomzer and Gansler of Kearfott Systems Division (Wayne, N. J.) reported at the Fourth Space Congress in April 1967 that Kearfott has been working on such a device that "requires no optical lens, no scanning, no data processing, no electrical memory and no active sources". They report that their system can be used when the aspect angle is within 15° of the vertical with no correction. When the angle is greater than 15° rectification can be used to allow tracking. The system uses a four-quadrant solid state device to detect the correlation spot and "is a completely autonomous and automatic acquisition device".

From flight test programs, it is estimated that with this system angular accuracies corresponding to a few hundred feet can be achieved for a low orbit.

The device could be used with a remote electronic display if the sensor is a vidicon camera. To obtain the optimum correlation, the scale of the reference should correspond to the height of the vehicle above the terrain. This can be open loop controlled, but is one of the sources of error in the device.

In the situation involving a planet flyby, where no prepared maps or photographs are possible and the time of tracking is limited to a few seconds, any implementation would have to be done by an electronic display, perhaps using memory tubes or similar instruments.

4.11.3 Scanners

The Bolsey Associates, Inc. (Glen Brook, Conn.) Image Motion Detector is a proven space-tested device that assisted the Lunar Orbitor spacecraft in producing the excellent photographs of the lunar surface. The device, as shown in Fig. 4-19, consists of a rotating aperture that scans the field of view. The scan radiance is fed to a photodetector (photomultiplier or solid state detector) and is recorded during one revolution on a magnetic drum that is part of the rotating slit assembly. During subsequent scans the incoming signal is compared to the prerecorded signal and a correlation is performed between the two signals to extract the tracking information. Due to the high scanning rate, 6,000 to 12,000 rpm, only the low-frequency information is recorded and compared. Image plane motion as low as 1 or 2 microns can be detected and used to correct the errors in a control loop. The angular error depends on the actual optical system used. For example, with a focal length of one-half meter, 2.5 microns at the image plane corresponds to 1 second of angular motion. This motion will only be seen if the diffraction limit of the objective lens is less than 1 second of arc, which requires an objective of 5.35 inches diameter.

In addition to the normal x-y tracking, the scanning device senses roll attitude changes as well. The electronic circuits can perform the correlation as the signal is being received and the processing is simple and fast; 20- to 100milliseconds. A variation in scale from 0.7 to 1.5, and in aspect angle, $\pm 45^{\circ}$, can be tolerated. At higher light levels, the device will track as diffuse a surface as a sheet of bond paper. The contrast level necessary for tracking at low levels will be as high as 1.5 to 1°.

Principal defects for a long space application appear to be the rotating member and the high rate of rotation. For application with low V/h rates, as in a planet tracker, lower rpm's may be possible while retaining the high track-ing accuracy.

4.11.4 Parallel Slit Reticles

If an optical device, as in Fig. 4-20, has a parallel-slit reticle in the image plane and this device is pointed normal to the direction of travel, then the light intensity passing through the reticle will be modulated. The frequency is directly proportional to the line density of the reticle and to the velocity of the vehicle and is inversely proportional to the height above the terrain; hence, the name V/h rate measuring device. This is the type of device made by Chicago Aerial Industries (Chicago, Ill.), and is principally used to control the film motion in aerial strip film cameras. It is possible to obtain an analogue signal that will give the V/h rate to one percent, but reducing this error is difficult. The signal also gives no indication of phase, so there is ambiguity if the velocity goes through zero.

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Ball Brothers' Research Corp. (Boulder, Colo.) has developed a variation on the reticle scanning technique which employs a rotating reticle to determine phase. The signal passing through the reticle has a frequency that depends upon the scan rate and the angular velocity of the target. By means of a frequency discriminator, the magnitude and direction of V/h can be determined. An angular velocity component perpendicular to the scanning direction will not be detected, although it will increase the noise output of the detector. Hultquist and Bartoe⁽⁶⁾ have reported that this device will measure angular velocities in the range of 0.6 to 29.5 milliradians/sec.

Baird-Atomic, Inc. (Cambridge, Mass.) has carried the rotating scanner idea a step further and uses the device as a tracker. The signal chopped by the reticle is compared in frequency with the signal resulting from the same reticle chopping the light from a lamp. If the frequency increases or decreases a signal is fed to a serve system, correcting the positioning so that the frequency difference returns to zero. Although there is no long-term memory with this device, William Westell of Baird-Atomic reports that accuracies of 2-seconds per second can be obtained with their optimal signal processing techniques.

Seward, Croopnick, and Rosenbloom of the MIT Instrumentation Laboratory have reported on a device that uses no memory, but does use the spatial phase in order to develop a tracking device. Two reticles are used (see Fig. 4-21) that are cemented to glass prisms in such a way that the image falling on one set of lines is displaced by half a line width from the image falling on the other reticle. If the image now moves over each reticle, similar frequencies will be developed but, more important, they will carry phase information. An x-y recorder made the lines shown in Fig. 4-22 as the image from an aerial photograph was passed over the two reticles. The higher the contrast levels, the larger the radiance of the trace. The rate of rotation of the trace depends on the V/h rate, the reticle spacing, and the particular optical system used. A zero crossing, point A in Fig. 4-22, for example, indicates that the signal could be used as an error signal in a feedback control system. It is estimated that the noise in the system corresponded to about a 5-micron tracking accuracy in the image plane. This compares favorably with the 2.5 micron accuracy obtained by Bolsey Inc. Sensitivity of this system to changing aspect has not been tested, nor have the effects of cross motion. In a typical tracking system, two of these devices would be required to perform x-y tracking, thus cross motion will be zero; however, rotation can still occur and if that poses a problem, another tracker could be placed with its optical axis at an angle to the x-y tracker so that rotation would produce another signal that could be used for error control. However, in the typical navigation system, other devices such as a star tracker

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or an inertial system will detect rotation of the line of sight of the tracker. The MIT device seems best suited to track unknown landmarks that are not changing aspect more than a few degrees.

In general, it is possible to obtain tracking accuracies of a few arc seconds using off-the-shelf hardware with a typical optical system such as a 75-mm objective and a 1-m focal length. Selection of a particular device to accomplish unknown landmark tracking for a specific mission will depend upon constraints such as reliability, power, weight, size, and compatibility with the optical system. In addition, the performance of each device should be evaluated for flight conditions subject to the particular mission, such as errors due to change in aspect angle, low level illumination, low contrast, off-axis tilt, and long term tracking.

For the purpose of the error sensitivity analysis of Chapter 3, it is required that some uncertainty values be placed on the probable errors to be expected from the tracking device. The specifications given for the Image Motion Detector made by Bolsey Associates, Inc. will be used, since the device has been flight tested.

4.11.5 Use of Unknown Landmark Tracking as a Measurement in Recursive Navigation Systems

Based upon the capabilities of the devices just discussed, this section develops an analytical expression for the technique of using unknown landmark tracking as a measurement scheme for recursive navigation.

Space navigation is the problem of determining the two vector quantities, position and velocity, throughout a mission. The state-of-the-art technique for estimating these vectors is recursive navigation. This basically consists of specifying a nominal trajectory and calculating the deviations from this trajectory at various times using navigation measurements.

For example, assume that at a specified time along the nominal trajectory the measured angle between a star and a planet should be A degrees. However, when the measurement is actually made, there is a difference of dA degrees between the estimated angle and the measured angle. The measured difference, dA, is then used to update the position and velocity estimates and thereby establish a better estimation of the trajectory. The current best estimate of the spacecraft's position and velocity may then be used to predict the same information at a future time in the mission. If necessary, corrections may be applied to the velocity vector at suitable times to fulfill mission trajectory requirements. There exist, however, certain constraints on the number and magnitude of these velocity corrections due to the fuel budget. Obviously, the better the navigation information, the less the fuel requirements. Hence, it is desirable to obtain high-accuracy navigation fixes utilizing instruments with low weight, low power, and high reliability.

The unknown landmark measurement discussed herein is defined as the measurement of the rate of change of the angle between an unknown landmark on a planet's surface and a known star (see Fig. 4-23). This measurement will then be compared with the predicted rate, calculated from the current best estimate of the spacecraft's state, and the difference will be used to improve the estimation of the spacecraft's position and velocity. The instrument that would perform this measurement is an Unknown Landmark Tracker (ULT).

The purpose of the following section is to derive the analytical expressions necessary to incorporate this unknown landmark measurement into a recursive navigation scheme.

A "rate" formulation is used instead of a large angle tracking formulation for two reasons: First, the tracking angle is restricted to small angles so as to remain within the known capabilities of the instruments just discussed. Second, we minimize the complexity of the analytical expressions. It is anticipated that a large angle-tracking formulation would permit better navigation accuracies than those reported in Chapter 3, and such a general formulation would be incorporated into any future studies. Section 4.5 discusses the techniques for accomplishing manual and automatic large-angle tracking with the video tracker incorporated into the design model.

4.11.6 Theory

This section will be devoted to obtaining the relationships necessary to incorporate an unknown landmark measurement in a recursive navigation scheme. What is required is a six component state vector that describes respectively spacecraft position and velocity as a function of time. Output of an unknown landmark measurement will be used as an input to a Kalman filter routine, whereupon the information may then be used to improve the current estimate of the state. Essentially, the measurement consists of comparing the angular rate of an unknown landmark at a specified time with the calculated nominal rate. To first order, the linear relationship between the measurement deviation and state deviation from their nominal values is:

$$\delta \Omega(\mathbf{t}) = \underline{\mathbf{h}}_{\mathbf{m}}^{\mathrm{T}}(\mathbf{t}) \ \delta \underline{\mathbf{X}}(\mathbf{t}) \qquad 4.11.6^{-}(1)$$

where $\delta\Omega(t)$ is the measurement deviation, $\underline{h}_{m}(t)$ is the measurement vector whose components are sensitivities in state deviation to measurement deviation, and $\delta \underline{X}(t)$ is the dimensional state deviation vector, i.e.,



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$$\delta \underline{X}(t) = \frac{\delta \underline{r}(t)}{\delta \underline{v}(t)}$$

For readability it will be convenient at this point to exclude the time dependence when we write \underline{x} , \underline{r} , \underline{v} , Ω and \underline{h}_{m} . The measurement vector, \underline{h}_{m} , may be derived from the star-landmark measurement equation (7.7) which is given in <u>Astronautical Guidance</u> by Battin, as

$$\delta \mathbf{A} = \frac{\rho \cdot \delta \mathbf{r}}{|\mathbf{z} + \mathbf{p}|} \qquad 4.11.6-(3)$$

where \underline{z} is the vector from the spacecraft to the center of the attracting body, <u>p</u> is the vector from this center to the unknown landmark and <u>p</u> is a unit vector in the star-landmark plane perpendicular to the line of sight to the landmark (see Fig. 4-24). Define

$$\underline{h} = \underline{z} + \underline{p}$$
 4.11.6-(4)

so that \underline{h} is the vector from the spacecraft, hence,

$$\delta \mathbf{A} = \frac{\underline{\rho} \cdot \delta \mathbf{r}}{|\mathbf{h}|} \qquad 4.11.6-(5)$$

Since δA is the angular difference between the nominal star-landmark angle and the actual star-landmark angle, the angular rate difference $\delta \Omega$ may be obtained by differentiating expression 4.11.6-(5). Where,

$$\delta A = A_N - A_t$$
 4.11.6-(6)

so that,

$$\delta\Omega = \frac{d}{dt} (\delta A) = \frac{d}{dt} (A_N) - \frac{d}{dt} (A_t) \qquad 4.11.6-(7)$$

hence, $d/dt(A_t)$ is the quantity that is actually measured and $d/dt(\delta A)$ is the quantity that is calculated from Eq. 4.11.6-(7). Performing the indicated differentiation,

$$\frac{\mathrm{d}}{\mathrm{d}t} \left(\delta \mathbf{A} \right) = \frac{\mathrm{d}}{\mathrm{d}t} \left(\mathbf{h}^{-1} \underline{\rho} \cdot \delta \underline{\mathbf{r}} \right) \qquad 4.11.6^{-}(8)$$

yields:

$$\frac{\mathrm{d}}{\mathrm{d}t}(\delta A) = h^{-2} \frac{\mathrm{d}h}{\mathrm{d}t}(\underline{\rho} \cdot \delta \underline{\mathbf{r}}) + h^{-1} \left[\underline{\rho} \cdot \frac{\mathrm{d}(\delta \underline{\mathbf{r}})}{\mathrm{d}t} + \frac{\mathrm{d}\underline{\rho}}{\mathrm{d}t} \cdot \delta \underline{\mathbf{r}}\right] \qquad 4.11.6-(9)$$

4.11.6-(2)



Fig. 4-24 Geometry of unknown landmark measurement.

Evaluating the derivative terms as they occur yields:

$$\frac{\mathrm{dh}}{\mathrm{dt}} = -\frac{\underline{V} \cdot \underline{h}}{h} \qquad 4.11.6-(10)$$

Define:

 $\frac{i}{h} = \frac{h}{h}$ 4.11.6-(11)

so that

$$\frac{dh}{dt} = -\underline{V} \cdot \underline{i}_h \qquad 4.11.6-(12)$$

where \underline{V} is the spacecraft's velocity. Therefore,

$$\frac{d(\delta \underline{r})}{dt} = \delta \underline{V} \qquad 4.11.6-(13)$$

To facilitate the definition of $\underline{\rho}$ and $\underline{d\rho}/dt$, define the direction of the star to be \underline{i}_s . Define also the normal to the measurement plane, \underline{i}_n :

$$\underline{\mathbf{i}}_{n} = \frac{\underline{\mathbf{h}} \times \underline{\mathbf{i}}_{s}}{|\underline{\mathbf{h}} \times \underline{\mathbf{i}}_{s}|}$$
4.11.6-(14)

Hence,

$$\underline{\rho} = h^{-1}(\underline{h} \times \underline{i}_n)$$
 4.11.6-(15)

differentiating to find $d\underline{\rho}/dt$;

$$\frac{\mathrm{d}\rho}{\mathrm{d}t} = \frac{1}{\mathrm{h}} \left(\frac{\mathrm{d}\underline{\mathrm{h}}}{\mathrm{d}t} \times \underline{\mathrm{i}}_{\mathrm{n}} \right) + \frac{1}{\mathrm{h}} \left(\underline{\mathrm{h}} \times \frac{\mathrm{d}\underline{\mathrm{i}}_{\mathrm{n}}}{\mathrm{d}t} \right) - \frac{1}{\mathrm{h}^2} (\mathrm{h} \times \underline{\mathrm{i}}_{\mathrm{n}}) \frac{\mathrm{d}\mathrm{h}}{\mathrm{d}t}$$
4.11.6-(16)

It may be shown that

$$\frac{dh}{dt} = -\underline{V} \qquad 4.11.6-(17a)$$

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$$\frac{\underline{d}\underline{i}_{n}}{dt} = \frac{(\underline{i}_{n} \times \underline{i}_{s})(\underline{i}_{s} \cdot \underline{i}_{n})}{h|\underline{i}_{n} \times \underline{i}_{s}|^{3}} \left[(\underline{V} \cdot \underline{i}_{n})(\underline{i}_{s} \cdot \underline{i}_{n}) - (\underline{V} \cdot \underline{i}_{s}) \right]$$

$$\frac{1}{\mathbf{h}|\underline{\mathbf{i}}_{n}\times\underline{\mathbf{i}}_{s}|}\left[(\underline{\mathbf{V}}\cdot\underline{\mathbf{i}}_{n})(\underline{\mathbf{i}}_{n}\times\underline{\mathbf{i}}_{s})-(\underline{\mathbf{V}}\times\underline{\mathbf{i}}_{s})\right]$$

4.11.6-(17b)

Substituting in Eq. 4.11.6-(16),

$$\frac{d\rho}{dt} = -\frac{\underline{\mathbf{V}} \times (\underline{\mathbf{i}}_{h} \times \underline{\mathbf{i}}_{s})}{h|\underline{\mathbf{i}}_{h} \times \mathbf{\mathbf{i}}_{s}|} + \underline{\mathbf{i}}_{h} \times (\underline{\mathbf{i}}_{h} \times \underline{\mathbf{i}}_{s}) \left[\frac{(\underline{\mathbf{i}}_{s} \cdot \underline{\mathbf{i}}_{h})^{2} (\underline{\mathbf{V}} \cdot \underline{\mathbf{i}}_{h})}{h|\underline{\mathbf{i}}_{h} \times \underline{\mathbf{i}}_{s}|^{3}} + \frac{\underline{\mathbf{V}} \cdot \underline{\mathbf{i}}_{h}}{h|\underline{\mathbf{i}}_{h} \times \underline{\mathbf{i}}_{s}|} - \frac{(\underline{\mathbf{V}} \cdot \underline{\mathbf{i}}_{s})(\underline{\mathbf{i}}_{s} \cdot \underline{\mathbf{i}}_{h})}{h|\underline{\mathbf{i}}_{h} \times \underline{\mathbf{i}}_{s}|^{3}} - \frac{\underline{\mathbf{i}}_{h} \times (\underline{\mathbf{V}} \times \underline{\mathbf{i}}_{s})}{h|\underline{\mathbf{i}}_{n} \times \underline{\mathbf{i}}_{s}|} + \frac{(\underline{\mathbf{V}} \cdot \underline{\mathbf{i}}_{h})^{2} (\underline{\mathbf{V}} \cdot \underline{\mathbf{i}}_{h})}{h|\underline{\mathbf{i}}_{h} \times \underline{\mathbf{i}}_{s}|} + 4.11.6 - (18)$$

To simplify the final results, let \underline{h}_m be represented by two vectors $\underline{\ell}$ and \underline{m} as indicated

$$\underline{\mathbf{h}}_{\mathrm{m}} = \begin{pmatrix} \underline{\ell} \\ \underline{\mathrm{m}} \end{pmatrix} \qquad 4.11.6-(19)$$

The result, upon substitution of the appropriate terms into Eq. 4.11.6-(19) yields

$$\underline{\ell} = \frac{\rho}{h^2} \left[\frac{\left(\underline{i}_{s} \cdot \underline{i}_{h}\right)^2 (\underline{V} \cdot \underline{i}_{h}) - (\underline{V} \cdot \underline{i}_{s})(\underline{i}_{s} \cdot \underline{i}_{h})}{\left|\underline{i}_{h} \times \underline{i}_{s}\right|^2} \right] + 3(\underline{V} \cdot \underline{i}_{h}) - \frac{\underline{V} \times (\underline{i}_{h} \times \underline{i}_{s}) + \underline{i}_{h} \times (\underline{V} \times \underline{i}_{s})}{h^2 |\underline{i}_{h} \times \underline{i}_{s}|}$$

$$4.11.6 - (20a)$$

and, accordingly:

$$\underline{\mathbf{m}} = \frac{\rho}{\mathbf{h}} \qquad 4.11.6-(20b)$$

where

$$\underline{\rho} = \frac{\underline{\mathbf{i}}_{\mathbf{h}} \times (\underline{\mathbf{i}}_{\mathbf{h}} \times \underline{\mathbf{i}}_{\mathbf{S}})}{\left|\underline{\mathbf{i}}_{\mathbf{n}} \times \underline{\mathbf{i}}_{\mathbf{S}}\right|}$$
4.11.6-(21)

(These equations are used for the navigation error sensitivity analysis reported in Chapter 3.)

4.12 Bearing Heat Transfer

During the period April to November, 1967, bearing heat transfer studies were conducted at the MIT Instrumentation Laboratory Thermal Laboratory, Bedford Flight Facility, Bedford, Massachusetts. The studies conducted were analytical and experimental investigations into the thermal resistance of a typical angularcontact instrument bearing. The results of these studies are contained in MIT Instrumentation Laboratory Report E-2215, <u>Analytical and Experimental Investigation on</u> the Thermal Resistance of Angular Contact Instrument Bearings.

The analytical investigation consists of two distinct sections: the elastic contact analysis between a ball and race under typical loading conditions, and the thermal resistance analysis.

The elastic contact study shows that the contact area is always elliptical in shape having very small linear dimensions relative to those of the ball or race. E-2215 includes a detailed description which will enable engineers to quickly calculate the size of the contact area knowing only the geometry and physical properties of the ball, both races and the load.

The thermal study shows that the thermal resistance at the ball/race interface is a function of the elliptical contact area and the ratio of the semi-major axis to the semi-minor axis. For symmetrical loading and heating conditions the bearing resistance is inversely proportional to the number of balls.

E-2215 also discusses the correlation of heat transfer data with thermal resistance theory and includes an analysis of the influence of thermal strain upon thermal resistance when bearing loads are very light.

4.13 Ring Laser Angle Encoder

The difficulties of realizing a practical digital angle encoder which has the attributes of high sensitivity and high accuracy and also simplicity and reliability has led to the conception and investigation of another angle-encoding technique. The gas ring laser has characteristics which permit its use as high performance digital angle encoder. The intrinsic inertial characteristic of the ring laser is advantageous for certain applications, but causes limitations on its applicability for other applications.

ĩ This section presents a discussion of the concepts involved in the use of a ring laser as an angle encoder. The first subsection, 4.13.1, discusses the properties of the ring laser which make it attractive for angle encoding applications. The second subsection, 4.13.2, discusses a typical angle measurement cycle and the intrinsic self-calibration performed on the ring laser, and on the navigation target sensor. The third subsection, 4.13.3, discusses the ring laser error mechanisms which become either insignificant or which are readily avoided in the following ring laser angle encoder design. The fourth subsection, 4.13.4, discusses the primary source of error. The fifth subsection, 4.13.5, discusses one way that the effect of this error mechanism can be minimized using the attitude isolation torquer and presents analysis on the anticipated angle measurement uncertainties as a function of scan rate. At the nominal scan rate assumed for the navigation angle measurement application of 1/10-radian/second, the calculated maximum angle measurement uncertainty contributed by the ring laser angle encoder is 1/300 of an arc second, as plotted in Fig. 4-33. Thus the conclusion drawn from this analysis is that the ring laser would contribute an insignificant amount of error under the conditions of this application.

Persons familiar with the concepts of the ring laser may wish to skip various subsections. More detailed discussions of pertinent ring laser phenomena are available in the literature. Most of that work was motivated by gyro applications. The purpose of this section is to discuss the properties of the ring laser from the viewpoint of the angle encoder application.

4.13.1 Angle Encoder Properties of the Ring Laser

A typical ring laser is shown schematically in Fig. 4-25. Ideally, the a-c (beat) output from the ring laser photodetector is a sinusoidal signal whose frequency is directly proportional to the inertial rotation rate of the ring laser about the sensitive axis.⁽⁷⁾ This signal is the beat frequency between the two contratraveling optical wave frequencies within the ring laser as measured in the rotating reference frame. An alternate interpretation is based upon the fact that the two contratraveling electromagnetic waves in the equivalent



EQUIVALENT CIRCULAR RING LASER RADIUS = $2A/\ell = \ell/8$, AND N_R = $\pi\ell/2\lambda$, E.G., $\ell = 32$ CM, $\lambda = 6328$ Å, R = 4CM, N_R = 7.94 X 10⁵/ BEATS COUNTED PER REV; OR 1.634 ARC-SECS/BEAT.



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circular ring laser have the same frequency in a non-rotating inertial frame and combine to produce a standing optical wave within the ring laser cavity. The detector in the rotating frame then passes peaks and nodes of the standing wave at a rate which is proportional to the rate of inertial rotation^(8, 9, 10) (Fig. 4-26).

4.13.1.1 Quantization Size

The standing optical wave and detector, shown schematically in Fig. 4-26, provide high resolution, and would be intrinsically digitized. As an example, a 6 inch diameter pancake, Helium-Neon laser unit would have approximately 10⁶ nodes per revolution, or, approximately one node (or beat) per arc second. For the application to be considered here, rotation rates on the order of one milliradian per second up to one radian per second occur so that no phase detection would be required for resolution, even considerably below one arc second. Interpolation in time is accomplished by counting clock frequency pulses between antinode (or beat) "marks", and dividing this count into a count of the number of clock pulses between the first beat mark and the mark from the sensor at target detection. At the maximum angular accelerations expected and within the angular velocity range considered, this ratio would yield digital encoding resolution accurate to a small fraction of an arc second. The signal-to-noise ratio available from the ring laser is sufficiently large that statistical variations in the time of detection of the antinode "marks" would introduce less than one-thousandth of a beat of error into the time interpolation digital quantization.

The lower rotation rate would be important as a limit to time interpolation accuracy if there were large variations of angular velocity of the ring laser block, which would cause varying "mark" intervals. For the applications considered here, the possible errors induced into a linear time interpolation by the maximum angular accelerations anticipated and the minimum rate considered (one milliradian per second) are insignificant for this application.

At a high rotation rate, such as 10 radians per second, when the "marks" occur with an interval of less than a microsecond, a primary limitation on time interpolation occurs because the mark interval approaches the time constants of the counting and timing logic. However, this is not a primary limitation to attitude sensing since the time constants of any associated attitude sensor component, such as a scanner or tracker, would contribute larger errors at these high rotation rates, as discussed in Section 4.5, so that the 1-arc second intrinsic quantization would be sufficient and beat or "mark" interpolation would not have to be considered.



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Fig. 4-26 Gedanken construction of ideal circular cavity.

At the nominal rate considered for this navigation angle measurement application of 1/10-radian per second, the technique of counting clock pulses between beat marks would allow digital quantization precise to approximately one-thousandth of a beat. This would result in a digital quantization of approximately one part in 10^9 of a revolution which is much more than is required for the navigation angle measurement application.

4.13.1.2 Quantization Precision

The precision of the equivalent circular ring laser "ruling" would be orders of magnitude better than those of incremental optical encoders and the 128-speed resolver. The wavelength of the standing light wave can readily be maintained uniform to one part in 10^9 during the time the ring laser is rotated through an inertial angle of 360°, within the previously considered angular velocity range. The linearity of the equivalent ring laser ruling is thus an insignificant source of error. This inherent linearity can be contrasted with the difficulties of producing a precision "ruling" on a disc accurate to much better than one part in 10^6 of a revolution. In order for the previously discussed angle encoders to achieve measurement linearities of the order of one part in 10^6 , various averaging techniques are utilized which tend to minimize the effects of the "ruling" variations. Several Fourier compensation components, determined from calibrations, must also be applied to the 128-speed resolver in order to achieve the above linearity.

4.13.1.3 Angle Measurement Sensitivity to Bearing Imperfections and Loading

Another feature of the ring laser is that it is insensitive to bearing runout and rigidity problems. Conventional angle encoders are required to have components on both members and are sensitive to relative translation motion between the two members. By contrast, the ring laser shown in Fig. 4-25 has the beat detector and the navigation phenomena sensor rigidly mounted to the ring laser block and the whole unit resides on one member, thus avoiding dependences on relative translations of two members. Calculations have shown that accelerations up to several hundreds of g's would cause negligible errors in the detection of the precision angle due to deformations in the ring laser.

Calculations have shown that it is relatively more difficult to obtain negligible flexure of the navigation phenomena sensor optics, which would be the limiting source of flexure error under high loadings.

4.13.1.4 Inertial Reference Property

The primary considerations for the use of the ring laser as an angle encoder is the relationship of the standing wave to a useful reference frame. Conventional angle encoders measure an angle in relation to the reference frame of the other member. The ring laser, on the other hand, is referenced to inertial space. For the purposes of space navigation and alignment, this is a very useful property, while on the other hand the ring laser is less readily adapted to measuring attitude isolation, gimbal angles, etc. in which the angle encoder must ultimately be referenced to the other member and not to an inertial reference frame.

If there were some way to tack the standing wave to the other member (as well as the ring laser block) then the ring laser could be used as would a conventional angle encoder. As it is, the ring laser's reference frame is physically tied to inertial space which prevents its use as a conventional angle encoder for applications in which neither member is inertially referenced. But there remain many areas where this inertial property permits exceptional performance using sequential detection techniques.

One of the primary advantages of being referenced to inertial space lies in the fact that sequential detection of two navigation targets with a single instrument is possible. Single sensor detection of both navigation phenomena produces an intrinsic calibration during every angle measurement, thus eliminating the error caused by distortions of the system prior to the measurement. Also, the "shorter" mechanical path between the two lines of sight provided by using a single scanner instrument, minimizes the amount of distortion error which can build up during a measurement. Also, since no tracking instrument is involved, the pointing servo requirements may be reduced to those required by a scanner photometer instrument.

Any sources of error in using the ring laser as an inertially referenced angle encoder are limited to those mechanisms which cause variations in the attitude of the equivalent circular standing wave relative to inertial space within the period of the angular rotation rate.

4.13.1.5 Angle Encoder Self-Calibration

Ring lasers have thus far been developed primarily considering gyro applications. The difficulties encountered in using the ring laser for gyro applications are certainly not sensitivity (quantization) or precision of the "ruling", but rather drifts in the attitude of the standing wave inertial reference. For a gyro application, the important questions are how much the drift rate of the standing wave will vary from the time the unit is calibrated (probably on the ground) until and during usage in a mission weeks, months, or even a year later?

By contrast, the angle encoder application provides an intrinsic calibration during every angle measurement which is repeated at an interval of less than a minute down to an interval of a few seconds. Thus, only short-term variations in drift rate of the standing wave are significant for the angle encoder applications. Fortunately, the error mechanisms which have paced the development of the ring laser gyro have predominately long time constants, such as temperature drift and dc power supply drift.⁽⁸⁾ Also, much larger variations in the standing wave drift rate can be tolerated for angle encoder applications than for gyro applications since there is much less concern about the time propagation of the errors due to the repeated calibration during each measurement cycle. The net result is that the intrinsic calibration of the angle encoder application eliminates the need for practically all of the error mechanism control loops and devices which are necessary for gyro applications.

4.13.2 Typical Measurement Cycle

The description of a star-limb angle measurement cycle would serve to illustrate the self-calibration character of the device. A ring laser unit (Fig. 4-25) is mounted rigidly to a scanner photometer instrument (such as described in Section 4.5). This combination is isolated from spacecraft attitude by bearings and a torquer servo motor as required for spacecraft attitude independence. The relevant components are shown schematically in Fig. 4-27 along with a functional block diagram.

A typical star-limb measurement cycle would consist of first establishing the plane of measurement which contains the lines of sight to the star and to the limb [as would be done for a simultaneous detection (sextant) measurement] and then positioning the bearing axis so as to be perpendicular to the plane of measurement. Secondly, the torquer motor accelerates the unit up to a selected inertial rotation rate, which is indicated by a comparison of the ring laser beat frequency with a preselected control frequency. As the scanner photometer rotates it detects the star, then detects the limb, and continues rotating until it again detects the star. Either direction of rotation may be used, thus the unit may reverse its direction at the end of any number of measurement cycles and perform the next measurement rotating in the opposite direction. During this 360° measurement cycle the ring laser beat antinode "marks" have been counted such that each of the scanner detections are correlated to a beat count (total unit beats plus fractions of a beat if accuracies of better than 1-arc second are required for an application, as determined by time interpolation between the "marks"). The number of beats in a 360° rotation of the ring laser in inertial space are found by counting from the first detection of the star until the second detection of the same star. This provides the scale factor relating beat counts to angles which is used to convert the


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Fig. 4-27 Ring laser angle encoder

beats counted in the interval from the star detection to the limb detection into the navigation angle. These two digital numbers can be directly communicated to the computer where they are combined to form a ratio which will give the desired star-limb angle.

The self-calibrating nature of the measurement cycle is equally advantageous for the calibration of the navigation phenomena sensor. As discussed for the ring laser, only the error mechanisms with time constants short enough to cause drifts during the measurement cycle are significant. The optics must only maintain a fixed, but arbitrary attitude relative to the laser block during a measurement cycle. Thermal, shock, vibration, and creep induced deformations between the optical axis and the ring laser block which occur prior to, or after the measurement, do not effect the accuracy of the measurement. Electrically induced drifts would be minimal since the scanner design utilizes a field-of-view aperture rigidly mounted to the scanner block (in contrast to a vidicon sensor in which the aperture is positioned by electrical signals). For these reasons, errors induced into the measurements due to drifts in the attitude of the optical axis relative to the ring laser would be small relative to those introduced by mechanical deformations during a simultaneous detection angle measurement.

4.13.3 Ring Laser Angle Encoder Design

Ring lasers have been built for gyro applications which could be used for angle encoder applications. A suitable quartz ring laser design for an angle encoder application has been shown in Fig. 4-25. There are several features that are important for the implementation of a ring laser for use as an angle encoder. First, a self-sustaining standing wave must be established within the laser cavity at a single ${\rm TEM}_{\rm oom}$ mode. For this to occur, only the TEM mode of the optical cavity is allowed within a spectral region in which the laser will supply gain by suitable choice of optical path length and excitation.⁽⁸⁾ To maintain a specified mode requires that the temperature of the cavity not vary more than $\pm 3^{\circ}$ F for the quartz ring laser shown in Fig. 4-25. Thermal expansions caused by temperatures outside this range would move the cavity resonance frequency associated with this mode outside of the gain region. For this ring laser design, the temperature range for sustaining the neighboring modes will be very close to, or just overlap the specified mode temperature range, and so, if the temperature deviates too mucn another mode would be established. The neighbor modes will have one more or one less mode around the ring laser optical circuit than the "specified" mode, depending upon whether the ring has expanded or contracted.

For the angle encoder application it is not important that a specified mode be established (in contrast to gyro applications). A prespecification of the number of modes is not necessary since a calibration for any mode established is determined during each measurement cycle. Thus, any one of the series of modes which would allow the use of the instrument at its nominal, momentary temperature would be suitable.

Once a mode is established, the cavity resonant frequency is brought near the peak of the gain curve (of a $\text{He}^3 - \text{Ne}^{20} - \text{Ne}^{22}$ ring laser) by small adjustments in the cavity optical path length. A small amplitude, low-frequency $(10^2 - 10^3 \text{ cps})$ dither variation of optical path length will produce small amplitude intensity variations of the laser output at harmonics of the dither frequency.⁽¹¹⁾ Information in the first and second harmonics can be converted into an error signal controlling the cavity mean optical path length. For this application, a simple temperature adjustment, which uses a heat transfer control would be suitable. A mechanism for heat transfer control of the cavity circumference would most likely be available since there would probably be some sort of temperature control for parts of any navigation instrument exposed to a space environment. Another technique could be used which would eliminate the need to heat or cool the instrument. The piezoelectric crystal used to dither the cavity dimension can be d-c biased to change the mean dimension.

Drift in the Langmuir flow rate (12, 13, 14) of the excited neon gas, induces a varying drift rate of the "standing" optical wave. The velocity of the gas oppositely doppler shifts the dispersion curves associated with the two directions around the optical circuit of the ring laser. The gas velocity is primarily a function of the d-c current exciting the laser. The small amount of drift occurring in the best power supplies can be fatal to the highest-accuracy ring laser gyro application, but produces a negligible effect in the short selfcalibrating interval of the angle encoder application.

Since just the fact of rotation of the ring laser also produces an oppositely directed doppler shift of the dispersion curves⁽⁸⁾ one effect which can be very significant for gyro applications is the variations in the drift rate of the standing wave in inertial space when the slope of the dispersion curve varies in time. In essence, the phase velocity of the resonant modes of the contratravelling light beams depends on the laser gas excitation and upon the frequency of detuning of the modes from the neon isotopes spectral line centers. A variation in the drift rate of the standing wave can be produced when the difference in phase velocities of the two waves varies. Variations in plasma excitation produce variations in the slope of the whole dispersion function. Variations in detuning shift the modes to different portions of the dispersion curve. For the more demanding gyro applications, this must be controlled by active feedback of the excitation and detuning.⁽⁸⁾ Fortunately, it turns out that with the use of a reasonably stable power supply, as mentioned in the previous paragraph, the variations in dispersion slope have long time constants relative to the angle encoder measurement cycle. Variation in detuning occurs mainly with temperature variations which also have long time constants in the solid quartz laser cavity. Thus these effects turn out to be insignificant in the angle encoder application.

4.13.4 Scattered Radiation Mode Coupling Nonlinearity

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An effect which is the primary source of ring laser error for the angle encoder application and which has paced the development of the ring laser for gyro applications is the sobject of this subsection. For the angle encoder application, the requirements on the control of this effect are considerably reduced. Because of these lesser requirements it is possible with a relatively simple technique (which uses the required attitude isolation servo motor) to reduce the uncertainties introduced into an angle measurement to an extremely small number. The expected uncertainties are shown in Fig. 4-33 at the end of this subsection.

In a ring laser, that portion of the light scattered by the moving m_{iir} rors from the direction of one travelling wave into the direction of the other travelling wave produces a fairly strong steady-state signal at a different frequency (doppler shifted by the moving mirror) from the primary wave oscillation as well as causing a slight shift in the primary frequencies. The combined optical oscillator frequencies associated with each travelling wave direction appear to cause the previously mentioned standing wave of the equivalent circular cavity to rotate slowly in inertial space in the direction of rotation of the ring laser.⁽¹⁵⁾ As the ring laser rotation rate decreases, the scattered-wave frequencies more closely approach the primary travelling wave frequencies and the steady-state amplitudes of the primary wave and scatter-induced waves approach each other. Thus, as seen in inertial space, the standing wave rotates more rapidly as the ring laser rotates less rapidly. At some point the amplitudes reach equality, the two rotation rates also become equal and the detector passes no modes and therefore senses no beat frequency. This is the rotation rate at which mode locking occurs and below which no beat frequency or indication of rotation is detectable. The mathematical treatment of this interaction reveals additional features such as the distortion from sinusoidal of the "standing" wave as the rotation rate is decreased toward the mode locking rate⁽¹⁵⁾ due

to the four waves of differing frequencies. A typical ring laser beat vs rotation rate curve is shown in Fig. 4-28.

For the above-described nonlinearity of beat frequency vs rotation rate, the lower lock-in rotation rate (which is indicative of the nonlinearities encountered), can be decreased by as much as a factor of 10 by such techniques as introducing an a-c Faraday element into the cavity,⁽¹⁴⁾ utilizing the external mirrors to feedback light in such a phase and amplitude as to cancel the internally scattered light⁽¹⁴⁾ and by mechanically vibrating the whole ring laser.⁽¹⁶⁾ Although the nonlinearity is reduced, the accuracy is sacrificed because an artificial rotation rate bias is introduced by these techniques which is limited in stability by the electrical and mechanical means employed. In effect, such a bias uncertainty would shift the curves shown in Fig. 4-18 to the right or to the left in an unknown manner.

At MIT, two techniques have been devised for reducing this nonlinearity which do not introduce an undesirable bias uncertainty. One technique oscillates the mirror surfaces and any other material which backscatters significant amounts of radiation in such a manner that the backscattered light experiences full doppler shift and the primary waves experience zero doppler shift. This oscillation causes varying amplitude, with a high rate of phase shift variations between the backscattered radiation and the primary travelling wave. This technique promises to reduce the lock-in frequency by more than a factor of 1000. This technique shows great promise for strapdown gyro applications, as well as yielding great linearity over a large dynamic range of rotation rate for the inertial angle encoder applications.

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All of the above technique's have the effect of reducing the lower lock-in rotation rate and also the associated nonlinearities occurring at rotation rates above this lock-in rate. The effect of introducing these techniques is to expand the abscissa scale of the typical case shown in Fig. 4-28 so that the lock-in occurs at a lower rate, but the shape of the curve remains essentially unchanged. Improvements in the state-of-the-art of mirror fabrication also will effectively shift the abscissa.

Considering this physical characteristic of functional similarity, a primary question for the use of a ring laser in a gyro or angle encoder application is what range of, and variation in, rotation rate inputs will occur. The manner of operation of the ring laser over the nonlinear beat-rotation rate relationship will determine the attitude drift uncertainties which will be produced. Fortunately for the angle encoder application, there is no need for the rate of rotation to be near the lock-in region. And in fact, when used in



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conjunction with a scanner photometer a finite rate of rotation is required for scanner operation. Also, there need be no reversal in the direction of rotation during a measurement cycle (in contrast to strapdown gyro applications in which the rate of rotation may frequently pass through zero).

The larger the rotation rate is above the lock-in region, the larger are the variation of rates which can be allowed during a measurement cycle and still achieve a specified precision of the angle measurement. The accuracy provided when the ring laser is operated within a given range of rotation rates, or alternatively, the range of rotation rates allowed for a required accuracy is strongly dependent upon the ratio of the nominal rate of rotation to the lock-in rate of rotation. At the lowest rotation rates, the nominal rotation rate is constrained by the time allowed to make a measurement. If a very short time is allowed for a measurement cycle and a very high accuracy is still desired then the best manner of operation would be to slowly scan through a target in order to maximize detection accuracy and then to rotate at a much higher rate to the next detection point where the rotation rate would again be slowed down to permit the most accurate detection, etc. The speeding up of the rotation rate between detections will add little error since the operation at higher rotation rates is even more linear. A small amount of error could be introduced by this rate change if the slower rate was pushed closer to the lock-in rate. Adding the MIT shear mirror technique to the ring laser design would greatly reduce the lock-in rate, and would eliminate even the small error which would be introduced by the larger range of rates required. This mode of operation can certainly be provided, but at this time there does not appear to be any need to go through the additional conceptual analysis.

A measurement cycle using a single uniform inertial rotation rate of the ring laser which would produce less than an arc second of angle measurement uncertainty (including both ring laser and scanner uncertainties) would take less than a minute. The mechanization for performing this type of measurement cycle and the mode of operation would be that which has been described in Section 4.13.2. This technique of rotating the ring laser at a constant <u>inertial</u> rate is the second scheme referred to previously, which <u>does not</u> introduce a bias which must be subtracted out of the relationship between the beat frequency vs. rotation rate. Drift rates of the standing wave are only very weakly coupled to variations in the ring laser body <u>inertial</u> rotation rate. Thus the rotation rate does not have to be "subtracted out" of the beat frequency in order to convert the beat frequency into a rotation rate. The following section analyzes the errors produced by the ring laser during this type of measurement cycle.

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4.13.5 <u>Ring Laser Angle Measurement Error Caused by Angular</u> Velocity Variations

The ring laser can be used for extremely precise angle measurement providing suitable steps are taken to stabilize the processes taking place in the laser. As just discussed, the primary source of error for this application is due to the partial coupling of the standing wave to the ring laser reference frame. Backscattering phenomena within the laser cause the "beat frequency" output of the ring laser to deviate from proportionality to ring laser angular velocity according to the following relationship (which is the typical nonlinear relationship shown in Fig. 4-28):

$$f_{B} = 2 \frac{R}{\lambda} \left[\omega_{r}^{2} - \omega_{c}^{2} \right]^{1/2}$$
 4.13.5-(1)

where

R = the radius of a circle inscribed in the ring laser polygon

 λ = the wavelength of the sustained mode

 ω_r = the instantaneous inertial angular velocity of the ring laser

 ω_{c} = the greatest laser inertial angular velocity at which the standing wave in the ring laser is completely coupled to the ring laser cavity, and no beat frequency can be observed

It can be shown⁽¹⁵⁾ that the error in the angle measured by the ring laser is given by the following expression to second order

$$\theta_{\epsilon} = \theta_{I} - \theta_{r} = \theta_{0} \left(\frac{\omega_{c}}{\omega_{r0}}\right)^{2} \left(\frac{\overline{(\delta\omega/\omega_{r0})}}{1 - (\omega_{c}/\omega_{r0})^{2}} - \frac{\overline{(\delta\omega/\omega_{r0})^{2}}}{2[1 - (\omega_{c}/\omega_{r0})^{2}]^{2}}\right)^{2} \left(\frac{4.13.5 - (2)}{4.13.5 - (2)}\right)^{2}$$

where

 $\theta_{\rm I}$ = the indicated angle measured by the ring laser

 $\theta_r = \int_{-\infty}^{T} \omega_r dt$ = actual angle through which the ring laser has rotated

 $\theta_0 = \omega_{ro} T$ = the nominal angle through which the ring laser would have rotated in T sec if $\omega_r = \omega_{ro}$ throughout that time interval

 ω_{ro} = the nominal constant angular velocity of the ring laser

 $\delta \omega = \omega_r - \omega_r = the variation of the instantaneous angular velocity from the nominal$

 $\overline{(\delta \omega / \omega_{ro})}$ = the temporal mean of the ratio enclosed in parentheses $\overline{(\delta \omega / \omega_{ro})^2}$ = the temporal mean square value of the ratio enclosed in parentheses

To control the angular velocity of the ring laser so that it is approximately constant, the ring laser is mounted on a rotating shaft and torqued at a servo controlled angular velocity.

To calculate the error from Eq. 4.13.5-(2), it is necessary to have an estimate of $\delta \omega$. This is expedited by designing an angular velocity servo loop mathematical model. Use of this model will then permit calculations of $\delta \omega$ if a disturbance input is given. A block diagram of the angular velocity servo loop mathematical model used for this study is shown in Fig. 4-29. The error calculations done with this loop assume the use of a 20 oz.-in. d-c torquer having 31 commutator segments (hence 31 ripple torque cycles/revolution). Although an open loop gain of 260,000 is theoretically realizable with the mathematical model selected here, a nominal loop gain of 52,000 was selected to be conservative and to allow for the use of less sophisticated hardware,

There are expected to be a number of sources of disturbance inputs seen by the ring laser angular velocity servo that will tend to cause the ring laser angular velocity to vary. Disturbance torques are expected to come primarily from three sources: flexi-pinrt or wire twist torques; bearing, seal, slipring and brush friction; and torquer ripple torque.

The steady state or average friction torque numbers used in this study were generated by adding the torquer brush friction to the expected bearing friction, and then doubling that sum. The doubling is expected to take care of any flexi-print or wire twist torque, or alternatively any slipring torque, which is added by the mechanization required for transmitting power and signals through the rotational degree of freedom. At the relatively low rates considered here, it is expected that the average friction torques produced by the bearings and torquer brushes (any sliprings that may be used) will be essentially Coulomb in nature without any significant viscous effects. It turns out that it is not the average torques, but the variations in torques occurring during a measurement cycle which are the main source of difficulty in maintaining a uniform rotation rate.

Over the range of the spacecraft attitude motion environments anticipated, it is reasonable to assume that the relative velocity of the ring laser with respect to its neighboring member will not change sign during a measurement cycle. Thus, no large transients in torque which are caused by friction need be considered. Also the torque variations contributed by any flex-prints or



Fig. 4-29 Conventional rate servo stabilization loop.

wire twists through the degree of freedom are not expected to significantly deteriorate angular velocity stabilization performance, in comparison to the torque variations contributed by the bearings and torquer.

Based upon the data gathered to date, calculations have shown that bearing and torquer ripple torques are the limiting factors in determining the angle encoding accuracy of the ring laser. It is possible to use an a-c torquer in place of a d-c torquer to reduce the ripple torque, but this is not necessary because the accuracies attainable using a d-c torquer with a ring laser are still quite exceptional. From Instrumentation Laboratory experience with IMU gimbal servos, it is expected that at the rotational speed considered here, that the bearing ripple torque will be about the same order of magnitude as the torquer ripple torque.

Direct-current torquers have been shown to exhibit two periodic components of ripple torque, one sinusoidal and one sawtoothed. The error generated in ring laser output by each component has been calculated separately and the resulting error (time) functions have been added to yield the data shown in Figs. 4-30 and 4-31. To allow a conservative margin for the effects of bearing ripple torque, as well as any torque variations caused by flex-prints or sliprings, the amplitudes of the torquer ripple torque have been doubled to yield the data in Figs. 4-30 through 4-33. Bearing ripple torque is expected to be somewhat random in nature and hence its effect would be more likely to add statistically to that of torquer ripple torque. Figures 4-30 and 4-31 show the ring laser angle encoding error caused by doubled torquer ripple disturbance at every point in one ripple cycle.

Figure 4-30 is plotted to a linear scale so that the relative magnitudes of one curve during the cycle may be appreciated. Figure 4-31 plotted the log of the absolute value of error versus θ_0 (or time, since $\theta_0 = \omega_{ro}$ t). From Fig 4-32 the effect of changing the operating point of the angular velocity servo, ω_{ro} on the magnitude of ring laser output error is more readily observed. This is considered "worst case" data because the phasing of the two components of ripple torque is adjusted to produce the maximum mean angular velocity error. The actual phase relationship between these two ripple torque components is not known. "Worst case" was used to place an upper boundary on what could be expected from this error source. Figure 4-33 shows the ring laser output error remaining at the end of one cycle of ripple torque plotted as a function of ω_{ro} . Since the torquer induced errors are largely deterministic and hence repeatable, much of the residual errors could be removed by calibration, but this is not necessary since the errors introduced, even if the unit is assumed to be linear, are still miniscule.



Fig. 4-30 Instantaneous "worst case" angle encoder error $\omega_{c} \leq 10 \text{ ERU}$











Fig. 4-33 Maximum "worst case" angle encoder error.

The maximum value of error occurring during one cycle of ripple torque (at $\theta_0 = 0.10134$ radians) is plotted against nominal angular velocity, ω_{ro} in Fig. 4-33. This is the maximum error to be expected (from ripple torque disturbance) in any measurement.

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Figures 4-30 through 4-33 have been normalized with respect to loop gain, K, and the mode locking angular velocity, ω_{c} . To obtain the accuracies obtainable at loop gains other than 52,000 and less than 260,000, and for values of ω_{c} other than 10 Earth Rate Units (ERU) (but within the constraints noted on each figure), use the following relationship for Figs. 4-30, 4-32 and 4-33:

$$\theta_{\epsilon} = \left(\frac{\omega_{c}}{10}\right)^{2} \frac{52,000}{K} \theta_{\epsilon \text{ nom}}$$

where θ_{enom} is given in Figs. 4-30 through 4-33, ω_{C} is in "earth rate units", and K is loop gain. For Fig. 4-33, use

$$\theta_{\epsilon} = \left(\frac{\omega_{c}}{10}\right)^{2} \frac{52,000}{K}^{2} \theta_{\epsilon \text{ nom}}$$

From these plotted results, which assume all ripple torque disturbance factors adding up in such a way as to produce worst case errors, it is evident that very high angle measurement accuracies are possible by using a ring laser in this manner. Under the constraints previously assumed, which are believed to be reasonable for the spacecraft designs now being considered, the disturbance inputs studies (torque and bearing ripple torque) would be expected to be the most significant cause of variation in ring laser encoder angular velocity and the errors resulting ther from. Since the error estimates presented here represent a worst case estimate with respect to this error source, these estimates should provide a conservative upper bound on the total angle measurement uncertainties for the most stringent interpretation of the beat output which assumes perfect linearity. For other applications and user situations, these estimates provide an indication of the order of magnitude of the uncertainties in ring laser encoder angle measurement to be expected.

For the nominal rate of 1/10-radian per second, which is for the navigation angle measurement application, an error of less than 1/300 arc second would be produced by the ring laser. (See Fig. 4-33). The minimum sequential angle measurement uncertainty provided by a ring laser-scanner combination occurs near the rotation rate at which each of the two units separately contribute the same magnitude of uncertainty. Assuming_the simplest scanner mechanization which uses one slit 10 minutes long and one pass, and a star intensity of 2.5 magnitude in a background of 3 foot-lamberts, this rate would occur at approximately 3/100 of a factor per second (see Figs. 4-12 and 4-33). The corresponding combined angle measurement error from these two sources would be approximately 2/10-arc sec, 1 σ . Thus we feel that the combination of the ring laser angle encoder described, coupled with a scanner instrument designed for this purpose, has the potential to provide more than an order of magnitude improvement in the accuracy of angle measurements over that possible within the present state-of-the-art of angle encoders. Based on the studies we have performed, it is our belief that this concept can provide the next major step in angle encoder technology.

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