CR-86011

07398-6016-R000

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

FINAL REPORT

VOLUME I—SUMMARY

30 JUNE 1967

Contract No. NAS 12-141
Prepared for
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
ELECTRONICS RESEARCH CENTER
Cambridge, Massachusetts

TRW
SYSTEMS GROUP
ONE SPACE PARK · REDONDO BEACH, CALIFORNIA

07398-6016-R000

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

FINAL REPORT

VOLUME I—SUMMARY

30 JUNE 1967

Contract No. NAS 12-141

Prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
ELECTRONICS RESEARCH CENTER
Cambridge, Massachusetts

T.W. Layton

T.W. Layton
Study Manager

C.W. Sarture

Approved: C.W. Sarture
Manager
Guidance and
Navigation Laboratory

TRW
SYSTEMS GROUP
ONE SPACE PARK · REDONDO BEACH, CALIFORNIA

FOREWORD

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

The broad objectives of this study were to:

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

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

CONTENTS

CONTENTS (Continued)

CONTENTS (Continued)

ILLUSTRATIONS

TABLES

TABLES (Continued)

1. INTRODUCTION

This report presents the results of a nine-month study of "Radio/Optical/Strapdown Inertial Guidance Systems" for future NASA unmanned space missions, conducted by TRW Systems for NASA/Electronics Research Center. This volume presents a summary of the study results, conclusions, and recommendations.

Section 1 of this volume discusses the study objectives and constraints, the method of approach used in conducting the study, and the various assumptions made with regard to the mission definitions and vehicle characteristics related to the guidance system functions.

The principal conclusions and recommendations of the study are summarized in Section 2. Section 2 also summarizes the principal design features of the recommended onboard optically aided strapdown inertial guidance system.

A summary of the mission characteristics, guidance system operational sequences, performance requirements, and guidance system configuration and operating characteristics is presented in Section 3.

A summary of the results of the various performance analyses carried out to demonstrate the feasibility and to establish guidance system performance requirements is presented in Section 4.

1.1 STUDY OBJECTIVES

The broad objectives of this study were to:

- Establish the guidance requirements for a selected group of future NASA unmanned space missions.
- Investigate possible guidance concepts based on the appropriate use of radio, strapdown inertial, and optical techniques, with the further objective of establishing the proper functional role, the capabilities, limitations, and constraints of each of these elements in the overall guidance system concept.
- Define feasible radio/optical/strapdown inertial guidance system design concepts and equipment configurations.

- Perform analyses to establish the feasibility (performance) of the selected design concepts.
- Indicate areas of technology where state-of-the-art advances are necessary.

The study constraints and scope of the work are defined as follows:

1) The representative missions to be studied were:
   - Synchronous earth orbit
   - Mars orbiter mission
   - Lunar orbiter mission
   - Solar probe mission using Jupiter assist.
2) The choice of inertial systems was limited to strapdown systems.
3) Only existing NASA and DOD radio tracking systems were considered, i.e., no new equipment development or facilities were considered.
4) The inertial instrument and optical sensor technology to be considered was limited to reasonable projections of the current state-of-the-art (to the early 1970's).
5) Initially, the study was limited to upper-stage guidance only. At ERC's request, the study was expanded to include boost phase guidance.

1.2 SUMMARY OF STUDY ELEMENTS AND STUDY PLAN

In accordance with the stated objectives, the study was carried out in the six major steps listed below.

1) Functional and performance requirements for the strapdown inertial guidance subsystem and the electro-optical sensors were defined by mission phase for each of the four generic missions studied.
2) A survey was accomplished of state-of-the-art electro-optical sensors and strapdown inertial components (gyros and accelerometers) that potentially could be used.

3) Based upon the results of 1) and 2), appropriate candidate sensors were selected and performance (error) models were developed for them.

4) A study of possible radio guidance concepts and the capabilities of existing NASA and DOD tracking systems was conducted to define candidate systems, their applicability, limitations, and performance capabilities for the four missions.

5) An overall radio/optical/strapdown inertial guidance system concept, equipment configurations, and operating sequences were developed for each of the four mission categories.

6) Performance analysis studies were conducted both to investigate the performance capabilities of the candidate radio/optical/strapdown inertial guidance configurations and to demonstrate their adequacy for the four missions.

Figure 1-1 shows the manner in which the study was conducted in the form of a study task flow plan.

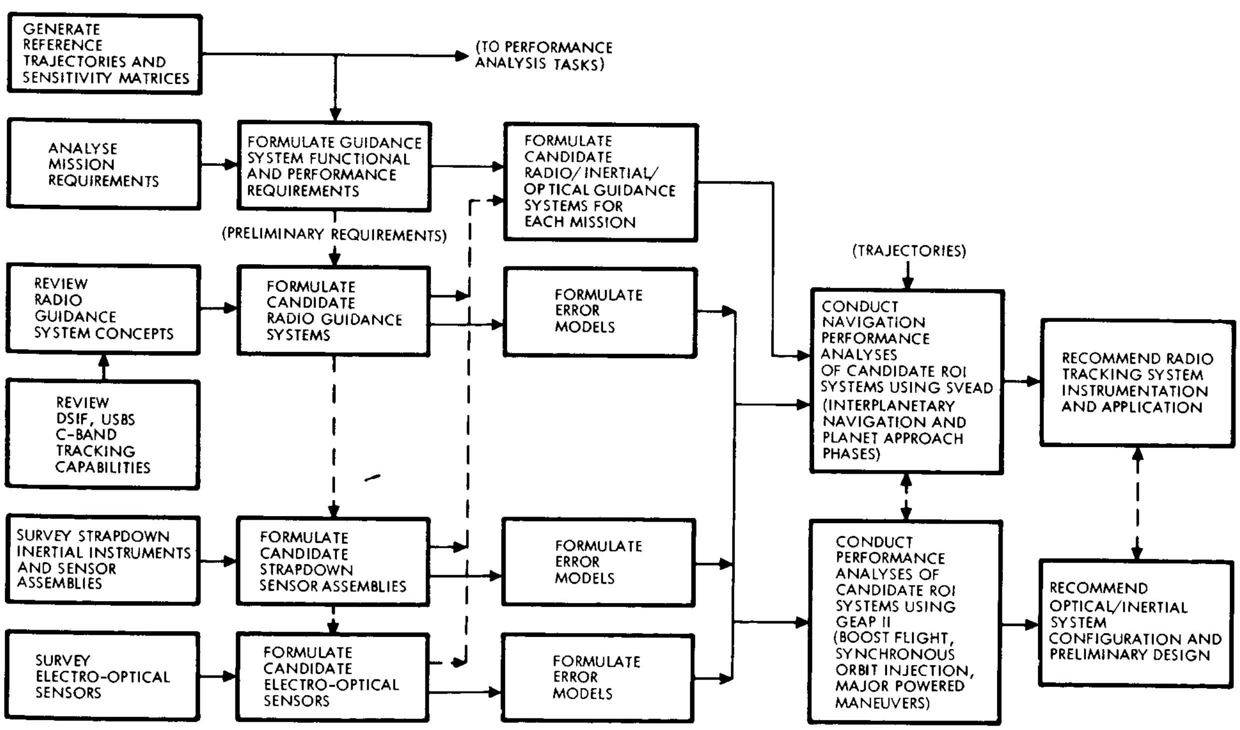


Figure 1-1. Study Task Flow Diagram

The major emphasis in this study was placed on defining the configuration of the onboard optically aided strapdown inertial subsystem, and the definition and analysis of the functional and performance requirements for this equipment. Although the digital computer, the tracking transponder and data link, and the vehicle control system are essential elements of the total guidance and control system, they have been considered only in a functional sense in defining the guidance concept and configuring the system; detailed consideration of these elements is outside the scope of this study.

1.3 STUDY ASSUMPTIONS

1.3.1 Mission Definitions and Requirements

It is assumed that the guidance requirements for the missions studied are representative of at least a major portion of the total requirements for NASA unmanned missions in the next decade. However, mission objectives are not precisely defined at the present time and definitive payload characteristics are not available. Also, launch vehicle selections for the missions have not been firmly made, and definitive design data are not available on vehicle upper stage concepts currently in the planning and development stages. For these reasons, it was necessary to postulate, somewhat arbitrarily, a set of specific mission performance requirements, launch vehicle selections, and vehicle and payload characteristics. These assumptions are detailed below.

For the same reasons as given above, it is not possible to present complete and definitive performance (accuracy) requirements for the guidance system. Consequently, some of the accuracy requirements presented in this report are based on mission requirements determined from past studies. Others are presented in parametric form. As more definitive trajectory data and mission objectives become available, these requirements can be updated.

The formulation of functional requirements and generic-candidate guidance system configurations is also dependent on mission analysis, although not to the extent that the formulation of accuracy requirements is. The functional requirements and candidate configurations can, therefore, be discussed in terms general enough to be applicable to any reasonable contemplated mission plans.

1.3.2 Postulated Vehicle/Payload Combinations

For purposes of this study, specific launch vehicle/payload combinations were postulated for the four missions. Table 1-I shows these assumed combinations along with the interpretation of what constitutes the "kick stage" for each of the missions. For the purposes of this study, the "kick stage" has been defined to be the final propulsive stage. In some cases, this may be an actual upper stage such as Centaur or HEUS (High Energy Upper Stage), and in other cases it may be a spacecraft such as Voyager. The portion of the mission to which the kick stage guidance is applicable also varies from mission to mission. The specific assumptions made as to the "guidance regime" are shown in Table 1-I.

1.3.3 Upper Stage Characteristics

Widely accepted quantitative values do not yet exist for the kick stage weights, mass ratios, propulsion capabilities (thrust, specific impulse), and ΔV (velocity increment) capabilities. Without these, it is impossible to define with certainty the accuracy requirements for any mission phase or midcourse correction velocity limits. Lacking these data, it has been decided (1) to draw on results from other related studies as much as possible, or (2) to present the requirements in parametric form.

For the thrusting and ΔV capabilities, it will be assumed, for the lunar and interplanetary missions, that the kick stage has two (or more) discrete thrust levels, the lowest thrust level suitable for corrective ΔV applications ranging from a few meters per second up to 100 m/sec. The highest thrust levels would be used for major orbital changes with ΔV values up to several thousand meters per second. It is also assumed that the kick stage has complete three-axis control capability.

The postulated separation of functions between the primary booster guidance and control (G and C) and the kick stage G and C are described in Section 3.

1.3.4 Separation of Guidance Functions Between Kick Stage and the Mission Payload

It was necessary for some of the missions to postulate the existence of capabilities within the mission payload such as the capability for independent attitude control, propulsive maneuvers for small orbital corrections,

Table 1-I. Vehicle Payload Combinations Postulated for Study Purposes

Generic Mission	Example Mission	Launch Vehicle	Kick Stage	Guidance Regime
Synchronous Earth Satellite	Advanced earth orbiting missions such as advanced meteorological or communication satellites	Atlas/Centaur	Centaur or HEUS†	Launch through snychronous orbit insertion (S/C payload assumes orbit trim and station keeping functions after insertion)
Mars Orbiter	Voyager	Saturn V	Voyager Interplanetary Spacecraft	Interplanetary orbit insertion through Mars orbit insertion and orbit trim
Lunar Orbiter	Advanced lunar orbiter photographic mission	Atlas/Centaur/HEUS†	Lunar Orbiter Spacecraft	Launch through lunar orbit insertion including orbital change maneuvers
Jupiter Flyby/ Swingby	• Solar probe with Jupiter assist • Jupiter flyby for scientific observation of planet	Atlas/Centaur/HEUS††	Interplanetary Spacecraft	Launch through injection into the interplanetary trajectory. Midcourse corrections and cruise attitude control provided by the spacecraft.†††

†A High Energy Upper Stage (HEUS) may or may not be required depending on the payload weight. It is assumed that the guidance equipment is located in the final stage but may provide guidance to the lower stages.

††Other upper stages such as Burner II may be used depending on the mission requirements and payload weight.

†††It may be possible to locate the guidance equipment in the spacecraft and utilize it to provide the midcourse guidance and cruise attitude control. The utility of this approach depends on the spacecraft concept (e.g., spinning or nonspinning spacecraft). The guidance systems studied here will provide the required capabilities for a fully attitude stabilized spacecraft.

communications, etc. At the time of separation of the payload spacecraft from the kick stage, these functions are activated and the kick stage functions terminated.

Since the payloads for the missions studied have not been defined in detail, reasonable assumptions have been made based on current spacecraft design trends. For example, for the synchronous orbit mission, the the payload spacecraft is assumed to have a capability for orbit trim and station keeping functions, as well as attitude stabilization and control, after insertion into orbit by the kick stage. Specific assumptions made for other missions are discussed in Section 3 of this volume.

1.4 DEFINITION OF TERMS

Certain of the definitions pertaining to the missions, the launch vehicle, mission events, and trajectories used throughout this report are summarized below.

1.4.1 Missions

In general, the term "mission" is used in this report to encompass and describe the events which are associated with directing the launch vehicle or the spacecraft from the earth and which terminate with the accomplishment of the mission objectives. In the analysis of the various missions throughout this study, the following terms are used:

Synchronous Earth Orbit Mission	In the synchronous earth orbit mission, the launch vehicle is used to place the satellite payload into an earth synchronous (24-hr period) equatorial orbit at a desired longitude. The injected payload (satellite) is assumed to have orbit trim and stationkeeping capability.
Orbiter Missions	In an orbiter mission, approximately at the time when the spacecraft is closest to the target body (moon or planet), its trajectory is deliberately altered by a propulsive maneuver so that it remains in an orbit about the target body as a satellite.

Solar Probe Mission	In a solar probe mission the spacecraft is injected into a heliocentric orbit that passes within a specified distance of the sun. This is an untargeted mission requiring no trajectory alterations subsequent to injection.
Flyby Mission	In a flyby mission, the spacecraft passes close to the target planet. No propulsion forces are employed to alter the trajectory so as to remain in the vicinity of the target planet. The spacecraft departs from the region of the target planet, although its trajectory will have been perturbed.
Solar Probe with Planetary Swingby	In this type of mission the spacecraft passes close to a planet with the purpose of significantly altering the spacecraft trajectory. After departure from the target planet, the spacecraft continues on a heliocentric trajectory to within a prescribed distance from the sun. No propulsive forces are employed to alter the trajectory in the vicinity of the target planet. For a given distance of closest approach to the sun, this technique may be used to significantly reduce the launch vehicle ΔV requirements, usually at the expense of considerably longer mission durations.

1.4.2 Vehicle Terms

Launch Vehicle	The launch vehicle includes the multistage boost vehicle which injects the spacecraft into the desired trajectory and includes all hardware up to the field joint where the spacecraft is mated and the payload shroud attaches which protects the spacecraft. Generically, the launch vehicle system also includes all appropriate ground support and test equipment.
Kick Stage	For the purposes of this study, "kick stage" refers to the final powered stage of the launch vehicle (the payload spacecraft is assumed

	to have only limited velocity capability for incremental orbit corrections). The kick stage is assumed to provide complete three-axis guidance, navigation and control capability for all launch vehicle stages except for the Saturn V (Mars orbiter mission).
High Energy Upper Stage (HEUS)	This is a particular kick stage concept using an advanced propulsion system burning high energy propellants such as H_2/F_2. Typical gross weight is 3200 kg. The thrust to weight ratio is approximately 1.
Spacecraft	The spacecraft system encompasses the payload itself and all its component subsystems, the science payload, the adapter which is mounted to the kick stage, and limited propulsion capability for orbital corrections.
Launch Operations System	The launch operations system does not include any flight hardware, but constitutes the operational responsibility for supporting and conducting the launch of the combined launch vehicle and spacecraft through the separation of the spacecraft from the launch vehicle.
Mission Operations Systems	Operational responsibility for supporting and conducting the mission after the spacecraft is separated from the launch vehicle is borne by the mission operations system.

1.4.3 Mission Events

In the analysis of the various missions throughout the study, the following terms are used:

Prelaunch	Collectively, all events before liftoff.
Launch	Collectively, all events from liftoff to injection.
Liftoff and Ascent	Departure of the combined launch vehicle-spacecraft from the ground and ascent to a parking orbit of specified altitude (typically 185 km (100 nmi)).

Injection (synchronous earth orbit mission)	Thrust termination of the kick stage, placing the kick stage/payload into a transfer trajectory to synchronous altitude from the parking orbit or, alternately, into the final synchronous earth orbit.
Injection (lunar or interplanetary mission)	Thrust termination of the lower stages of the launch vehicle, placing the kick stage/payload into an interplanetary or translunar trajectory, from the parking orbit.
Separation (shroud)	Detachment of the nose fairing from the launch vehicle during ascent.
Separation (spacecraft)	Detachment of the spacecraft from the spacecraft kick stage adapter after injection.
Orientation Maneuver	A programmed alteration of the kick stage attitude to cause it to assume a desired orientation.
Reorientation Maneuver	A programmed alteration of the kick stage attitude to cause it to return to the cruise orientation.
Midcourse Trajectory Correction Maneuver	A propulsive maneuver performed to compensate for inaccuracies or perturbations so as to redirect the kick stage toward the intended aiming point. Generally, it requires orientation to a specific attitude, operation of the rocket engine, and reorientation to the cruise attitude. The time of this maneuver is during the interplanetary or translunar flight, but not necessarily at the midpoint.
Encounter	Generally, encounter encompasses events occurring when the spacecraft is near the target planet. Specifically, it refers to the time when the kick stage is at its point of closest approach (periapsis).
Orbit Insertion	The propulsive braking maneuver by which the (orbiter) spacecraft trajectory at the target planet is changed from approach (hyperbolic) to orbital (elliptical).

1.4.4 Trajectory Terms

In discussing the trajectories possible for the various missions studied, the following terms are used:

Direct Trajectory	An interplanetary trajectory from the earth to a target planet, in which no intermediate planets (or satellites) are approached closely enough to significantly influence the trajectory.
Swingby Trajectory	An interplanetary trajectory from the earth to a target planet, in which an intermediate planet is passed sufficiently closely to exploit the effect of its gravitational attraction. This exploitation may provide reduced mission duration, reduced launch energy, or an opportunity for scientific observations of the intermediate planet.
Launch Opportunity	The time during which trajectories to a target planet may be initiated from the earth, with reasonable launch energies. A launch opportunity is usually identified by the year in which it occurs, and the target planet.
Launch Period	The space in arrival date-launch date coordinates in which earth-planet trajectories are possible in a given launch opportunity; specifically, the number of days from the earliest possible launch date to the latest.
Launch Window	The time in hours during which a launch is possible on a particular day.
Geocentric (heliocentric, planetocentric)	Described or measured with respect to inertial coordinates centered with the earth (sun, planet). Pertaining to the portion of the flight in which the trajectory is dominated by the gravitation of the earth (sun, planet).

C3, Launch Energy, Injection Energy	Twice the geocentric energy-per-unit mass, of the injected spacecraft. This is equivalent to the square of the geocentric asymptotic departure velocity.
Asymptote	The line that is the limiting position which the tangent to a hyperbolic (escape) trajectory approaches at large distances from the attracting center.
DLA	Declination of the outgoing geocentric launch asymptote.
ZAL	Angle between the outgoing geocentric asymptote and the sun-earth vector.
ZAP	Angle between the incoming planetocentric asymptote (at the target planet) and the planet-sun vector.
ZAE	Angle between the incoming planetocentric asymptote (at the target planet) and the planet-earth vector.
V_∞ or V_{HP}	Planetocentric asymptotic approach velocity.
Parking Orbit	An unpowered, geocentric, approximately circular orbit, separating the powered portions of the launch and injection sequence.

1.4.5 Coordinate Systems

The various coordinate systems used in specifying performance requirements and powered flight performance analysis results are defined here.

ECI (Earth-Centered-Inertial)	This is a right-handed coordinate system, in which Z lies along the earth's polar axis and X and Y lie in the earth's equatorial plane. The X-axis passes through the Greenwich meridian or in the direction of the Vernal Equinox at the time of launch, (specified in text).

RTN (Radial-Tangential-Normal)	A right-handed orthogonal coordinate system in which R lies in the direction of the nominal position vector from the center of the earth, and N lies in the direction of the orbital angular momentum. T forms a right-handed orthogonal set with R and N.
(X, Y, Z) Selenographic	Moon-Centered Inertial Coordinates. This is a right-handed orthogonal coordinate system in which Z lies along lunar polar axis, and X, Y lie in the lunar equatorial plane with X passing through zero lunar longitude (Sinus Medii).

2. SUMMARY OF MAJOR STUDY CONCLUSIONS AND RECOMMENDATIONS

The principal conclusions and recommendations resulting from the study are summarized in this section. Subsection 2.1 presents our conclusions relative to the overall guidance system concept and configuration for each of the missions. Subsection 2.2 discusses the principal conclusions of the portion of the study dealing with the application, limitations, constraints, and performance capabilities of radio guidance. Subsection 2.3 presents a summary of the principal design features of the onboard optically aided strapdown inertial guidance system.

2.1 GUIDANCE SYSTEM CONCEPT

2.1.1 General Conclusions

In this study, the applicability of state-of-the-art guidance concepts utilizing appropriate combinations of ground-based radio tracking and onboard inertial and optical sensors has been evaluated for four representative missions (see Section 3 for a description of the mission characteristics). The following general conclusions relative to the system concept were reached:

1) The guidance functions for the four missions can feasibly be accomplished in an efficient manner by utilizing appropriate combinations of navigation sensors consisting of ground-based radio tracking and onboard inertial and optical sensors. The concept of the radio/optical/inertial guidance system evolved during this study consists of a "core" strapdown inertial subsystem (inertial reference unit and computer) with the capability of adding appropriate electro-optical sensors (star trackers, horizon sensors, sun sensors, etc.) to tailor the system for a particular mission application. The onboard system also includes a transponder and data link working in conjunction with the ground-based tracking systems.

2) For the synchronous earth orbit mission, the guidance functions (launch-through-final-orbit insertion) can be performed efficiently by the onboard inertial system, supplemented by optical aids for attitude and position updating during long coast periods. Radio tracking may also be used as an alternate method of position updating, however, severe operational and mission constraints are encountered, which make its use as the primary guidance system very unattractive.

3) For the interplanetary missions, radio guidance (i.e., ground based radio tracking and orbit determination) is essential to meet the mission objectives and is the only reasonable method† of meeting the demanding mission performance requirements. The concept recommended here uses the existing NASA Deep Space Instrumentation Facility (DSIF) as the primary means of orbit determination during the interplanetary trajectory phases. Powered maneuvers for trajectory correction and insertion into orbit around the target body are performed under control of the onboard optical/inertial system.

4) For the lunar mission, it is concluded that the most reasonable approach is to use the NASA Unified S-Band (USBS) tracking system as the primary navigation sensor, and to use the onboard optical/inertial system for controlling the powered maneuvers in a manner similar to that for the interplanetary missions. Although the performance requirements for an autonomous system would be considerably less severe than those for interplanetary missions, it would be very difficult to achieve the navigation accuracies attainable with the presently existing radio-tracking systems.

5) Radio guidance is of very limited utility for boost-phase guidance for the launch vehicles considered, and is not recommended except under special circumstances. (See Subsection 2.2)

6) Boost-phase guidance (launch-through-orbit insertion including the parking-orbit coast phase) may be performed with sufficient accuracy, using only the onboard inertial system. However, for all the missions considered, onboard optical sensors are required in the extended coast phases (earth-orbit, translunar, or interplanetary coast) for correcting the attitude drift rate of the onboard inertial sensors (gyros), or as the primary attitude reference.

---

†Although it is theoretically possible to perform the interplanetary missions using a completely autonomous, onboard, optically aided, inertial system (no ground-based radio tracking), this approach places extremely severe accuracy requirements on the onboard system (particularly the optical sensors) and requires significantly greater fuel allowances for performing trajectory-correction maneuvers.

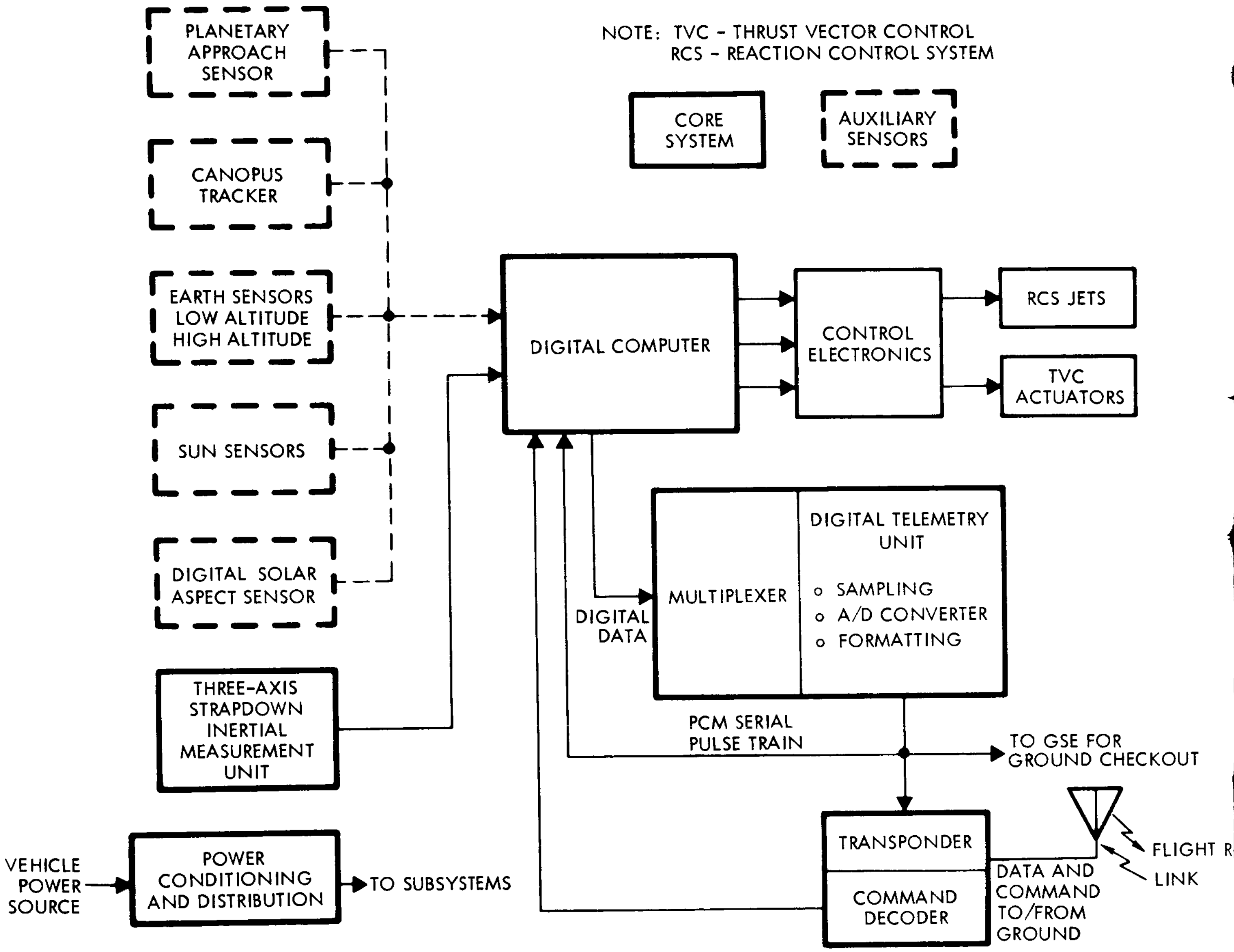
7) If the onboard system is located in the final stage of the vehicle or in the payload spacecraft, it is feasible from a functional and performance point-of-view to use it for guidance of the lower stages starting at liftoff, provided that the interface between the guidance system and the vehicle control system is properly configured. The evaluation of the desirability of using a single guidance system or, alternately, a separate system for lower-stage guidance and control, depends on vehicle and program considerations not considered in this study.

8) The guidance system concept recommended in this study and the specific configurations recommended for each of the missions represent a minimum assemblage of hardware necessary to meet the functional and performance requirements established by the mission. There are, therefore, few backup or alternate operating modes available for increasing the overall mission/vehicle reliability. Although quantitative reliability estimates have not been established, it is our opinion that the reliability of the guidance system (nonredundant configuration) will be sufficient to meet all of the reliability requirements, except those requiring extended operation of the equipment (the missions that fall in this category are the Mars Orbiter and Jupiter flyby mission). For the extended duration missions, one method of increasing the overall system reliability is to incorporate redundant subsystems (e.g., dual Canopus trackers) into the guidance system. The modular system configuration recommended permits this to be accomplished with relative ease, provided that this requirement is anticipated and properly reflected in the interface design.

2.1.2 Equipment Configuration by Mission

A block diagram of the total guidance system suitable for any of the missions is shown in Figure 2-1, together with a matrix showing the specific equipment utilization by mission. The recommended configuration is that of a basic "core" system used for all the missions, with auxiliary sensors added in a modular or building-block fashion to configure the system to a particular mission. The auxiliary sensors interface with the core system through the digital computer. If the computer input/output design is such as to accommodate any set of auxiliary sensors without any required redesign, then the mission-dependent changes can be accomplished with a minimum of effort by suitably changing the stored computer programs (software).

EQUIPMENT CONFIGURATION



COMMAND LINK USED FOR FLIGHT AND GROUND SEQUENCING, MODE CONTROL, ETC.
TELEMETRY LINK (HARD WIRE AND/OR RF) USED FOR GROUND CHECKOUT

EQUIPMENT UTILIZATION

EQUIPMENT		EARTH SYNCHRONOUS ORBIT	LUNAR ORBIT	SOLAR PROBE (JUPITER SWING-BY)	MARS ORBITER
CORE SYSTEM	3-AXIS STRAPDOWN INERTIAL MEASUREMENT UNIT	▲	▲	▲	▲
	DIGITAL COMPUTER	▲	▲	▲	▲
	S-BAND TRACKING TRANSPONDER & COMMAND LINK	▲ NOTE 1	▲	▲	▲
	AUXILIARY EQUIP. POWER CONDITIONING & DISTRIBUTION, TELEMETRY, ETC.	▲	▲	▲	▲
AUXILIARY SENSORS	STAR (CANOPUS) TRACKER (ATTITUDE REFERENCE)		▲	▲	▲
	EARTH SENSOR (HORIZON SCANNER) (LOCAL VERTICAL REFERENCE) • LOW ALTITUDE • HIGH ALTITUDE	▲			
	SUN SENSOR (CRUISE ATTITUDE REFERENCE)		▲	▲	▲
	SUN SENSOR SOLAR ASPECT SENSOR FOR ATTITUDE REFERENCE & NAVIGATION FIX (OPTIONAL)	▲ NOTE 1			
	PLANETARY APPROACH SENSOR				▲ NOTE 2

NOTE 1: THE NAVIGATION UPDATE REQUIRED PRIOR TO THE FINAL ORBIT INSERTION MAY BE MADE EITHER BY GROUND-BASED RADIO TRACKING OR BY AN ON-BOARD TECHNIQUE UTILIZING THE SUN SENSOR (SEE SECTION 3.2.4).

NOTE 2: THIS SENSOR MAY BE ADDED IF IMPROVED APPROACH GUIDANCE AND ORBIT INSERTION IS REQUIRED (THIS REQUIREMENT IS NOT FIRMLY ESTABLISHED FOR THE MARS MISSION CONSIDERED IN THIS STUDY). (SEE SECTION 3.3.3.) THIS SENSOR, TOGETHER WITH A PRECISION CANOPUS TRACKER AND SUN SENSOR, CAN BE USED TO REDUCE THE UNCERTAINTY IN THE SPACECRAFT TRAJECTORY WITH RESPECT TO MARS DURING THE APPROACH PHASE (SEE SECTION 4.3).

Figure 2-1. Equipment Configuration and Utilization

While the implementation of the "core" inertial guidance system is identical in each of the missions, its role varies significantly from mission to mission. For example, in the synchronous earth orbit mission, the strapdown subsystem (supplemented by appropriate electro-optical sensors) can essentially provide complete autonomous guidance and navigation. In the lunar orbit mission, it provides precise guidance during midcourse and orbit-insertion maneuvers with primary translunar navigation provided by ground tracking during the coasting phases. The inertial subsystem provides primary attitude reference information for the synchronous earth orbit mission; in the other missions, primary attitude-reference information during heliocentric orbit phases is provided by the sun and star sensors.

The inertial measurement unit, shown in Figure 2-1, is a strapdown configuration. Outputs of the three orthogonal body-mounted gyros are in the form of pulses, each quantized pulse representing an incremental attitude change about the gyro's sensitive axis. The computer accepts this information and can generate body angular rate information and/or total body attitude information. The output pulses of the three body-mounted accelerometers represent velocity increment information, which is combined with the gyro data to provide total velocity change information in some chosen set of inertial reference axes. A detailed description of the strapdown inertial subsystem is presented in Volume II, Section 4, and is summarized in Paragraph 2.3.1 following.

The auxiliary sensors in this study have been limited to electro-optical sensors, used primarily for attitude referencing. These sensors include earth horizon scanners, sun sensors, star trackers, and planetary approach sensors. The application of these sensors by missions and by mission phases is explained in detail in Section 3. Detailed discussions of individual sensors are presented in Volume II, Section 5, and are summarized in Paragraph 2.3.2 following.

The interface areas involving the control subsystem, the power subsystem, and the tracking, telemetry and control subsystems are not discussed in this report.

2.1.3 Strapdown Inertial Guidance Subsystem

In this study the use of the strapdown mechanization of the inertial guidance subsystem was postulated.† Two Inertial Reference Unit (IRU) configurations were selected for this study and detailed error (performance) models were developed for them. The design characteristics of these two units are summarized in Subsection 4.1. Although a particular choice of instruments was made for the purpose of this study, it is not intended that this choice constitute a recommendation for development of IRU's based on these instruments. The major motivation for choosing these particular instruments was 1) to span the range of currently available performance capabilities and 2) to select instruments on which a reasonable amount of test data was available to TRW for the purpose of constructing error models.

It was determined that the four specified missions could be accomplished utilizing either of the IRU's postulated. Use of optical sensors is required to correct for the attitude drift rate of the strapdown IRU over extended coast periods in all of the missions. The use of the auxiliary sensors is discussed in Section 3.

Performance studies were carried out using TRW's GEAP II Inertial Guidance Digital Computer Program for all significant mission phases using the error models developed for the two strapdown IRU's. The results of these performance analyses are summarized in Subsection 4.1 and are presented in detail in Section 7 of Volume II. On the basis of these performance studies, it is concluded that:

1) Boost phase guidance (launch through initial parking orbit insertion) may be performed satisfactorily by a strapdown system. Although it may be possible to

---

† An alternate approach utilizing a three- or four-gimbal gyro stabilized platform was not specifically studied here except for comparative evaluation of the performance capabilities of the Centaur guidance system with a strapdown system for the launch phase guidance of the lunar mission (See Subsection 7.4, Volume II).

achieve higher accuracies by using a gyro stabilized platform, these higher accuracies are not required for the missions considered.†

2) The strapdown IRU may be used as a short-term vehicle attitude reference during coast phases. For short (less than one orbit) parking orbit coasts, no auxiliary sensors are required. For longer parking orbit coast times and for translunar and interplanetary cruise phases, the inertial attitude reference provided by an auxiliary set of optical sensors is required. For the missions considered in this study, a sun sensor/star (Canopus) tracker is recommended.

3) Since the accelerometers are not required during long coast phases, they may be either turned off if this leads to a significant reliability improvement or power saving,†† or their outputs may be ignored.

4) The strapdown IRU together with the digital computer and the vehicle control system provides a precision capability for performing powered maneuvers for the midcourse trajectory correction, orbit insertion, and orbit trim maneuvers required by the missions. This system is capable of providing preburn attitude maneuvers, closed loop steering during the propulsive phase, and accurate thrust cutoff based on the velocity (ΔV) accumulated during the burn.

2.1.4 Electro-Optical Sensors

In this study the applicability of state-of-the art electro-optical attitude sensors has been evaluated for use in the radio/optical/strapdown inertial guidance system. It was determined that the four specified missions could be accomplished by utilizing various combinations of sun sensors, earth sensors, a Canopus sensor, and a planetary approach sensor. Only in the case of the Mars orbiter mission was it determined that state-of-the-art equipment was not applicable. In this case it was determined that in

---

†If ground-based radio tracking is used for navigation in subsequent mission phases, then guidance accuracy may be traded off against the weight of propellants necessary to correct, at a later point in the mission, for the trajectory dispersions resulting from the guidance inaccuracies.

††This depends on the particular choice of sensors and detailed IRU mechanization considerations. In some cases the reliability may be degraded by turning the instruments off and on. No general recommendation can be given on the basis of this study.

order to obtain a higher degree of accuracy, higher precision would be required for both the Canopus sensor and the planetary approach sensor. A preliminary design concept of instruments for both of these applications is defined in Section 5, Volume II.

The degree of accuracy required in the Canopus sensor and the planetary approach sensor was determined to be beyond the present state-of-the-art of development of these types of instruments. However, in the case of the Canopus sensor, an approach has been defined by which increased accuracy may be obtained, primarily by utilization of a high-resolution refractive optical system in conjunction with the recently developed high-resolution image dissector (vidissector). The extent of development of planetary approach sensors which has been conducted to date is extremely limited. The approach defined in this report – utilization of high resolution optics and a high resolution image dissector in conjunction with digital scanning and signal processing techniques – appears attractive in improving instrument accuracy beyond that which has been obtained to date.

In both cases, the limitations in accuracy which may be obtained are dependent upon the following:

1) The precision with which nonlinearities in the image tube deflection components may be measured and calibrated.

2) The effects of component aging and environmental effects upon the stability of the instrument bias level over the duration of the mission.

3) The effects of spacecraft thermal stress throughout the mission.

4) The precision with which the instruments may be initially aligned to the spacecraft coordinate system as determined by the angular precision of both mechanical alignment and optical simulation equipment.

Determination of the feasibility of obtaining the required accuracy may be estimated analytically, but can be proven only by developing experimental equipment and subjecting the equipment to calibration, life, and environmental testing.

If the advantage of utilizing optical aids for advanced Mars missions appears sufficiently attractive, it is recommended that additional effort be conducted in the development and evaluation of engineering models of both the proposed Canopus sensor and the planetary approach sensor.

Additional studies of subsystem reliability are also recommended to determine the degree of redundancy which should be incorporated into the sensor subsystems. The results of this effort would permit extension and refinement of the subsystem error models which have been developed in this study and would result in a more definitive estimate of system performance. In addition, promising techniques for the identification and correction of failures at the instrument level may be developed.

In the system conceptual design area, the time updating scheme using a sun sensor suggested (Volume II, Section 2.3) for the earth synchronous satellite mission should be further analyzed. It appears to offer a simple solution requiring a minimum of onboard computational complexity. The use of this scheme would obviate the need for ground-based radio tracking for position updating prior to the final injection maneuver.

2.2 RADIO GUIDANCE UTILIZATION

2.2.1 Summary and Conclusions

One objective of the Radio/Optical/Strapdown Inertial Guidance Study is to determine the role of radio command in the guidance of unmanned launch vehicles employing the advanced kick stage. The principal results and conclusions of this study, presented in Section 6, Volume II, are summarized here.

The approach to this portion of the study has been to:

a) Identify where (by mission phase) radio guidance can be most usefully employed for the four mission categories defined in Section 3.

b) Identify the limitations and constraints on the vehicle and mission profile (trajectory and sequence of operations).

c) Identify candidate radio guidance systems.

d) Develop error models for the candidate systems and analyze their performance capabilities for selected mission phases.

On the basis of these results, recommendations are made as to the utilization of the radio tracking instrumentation and its interface with the onboard guidance system.

The assumptions and ground rules used in conducting this study are as follows:

a) Only existing NASA and DOD radio tracking systems are considered, i.e., no new systems are postulated nor has relocation of existing equipment been considered. The tracking systems considered are those shown in Table 2-I.

b) Those tracking systems that cannot be used for near real-time trajectory or orbit determination without major additions of equipment such as ground links, ground computational facilities, ground/vehicle data links, etc., are not considered. Generally, this eliminates the range instrumentation systems such as MISTRAM,† AZUZA, UDOP, GLOTRAC, etc.

Table 2-I. Radio Tracking Systems Considered in Study

System	Location
DOD Systems • GE Mod III • BTL	Eastern and Western Test Ranges (Cape Kennedy and Vandenberg AFB)
NASA Systems • STADAN • C-Band Radars† • Unified S-Band System (USBS) • DSIF	World-wide deployment. See Tables 6-IV and 6-V, Volume II, for station locations

† DOD C-Band radars are included.

†MISTRAM has a limited real-time capability (it is used for range safety). Its uncertain future makes it questionable for this application.

The following general conclusions result from this study:

a) Use of the C-Band radars is limited to low-earth orbit tracking only. Station locations, coverage, data communication constraints, and system accuracy limitations are such as to eliminate these systems from consideration as useful radio guidance systems for the missions considered. However, tracking and orbit determination of spacecraft in low-altitude earth parking orbits is possible to reasonable accuracies (as was done on the Gemini program).

b) The GE Mod III and BTL radio/inertial systems may be used for accurate guidance during the launch phase from both ETR and WTR. These systems are currently in use for Atlas/Agena and Thor/Delta launches. A limitation is reached when the elevation angle of the vehicle, as seen from the radar site, drops below 5 deg. This condition is reached prior to orbit insertion for most vehicles employing upper stages such as Centaur, Agena, and Delta (final stage). Nevertheless, it is possible to use three systems to guide the lower stages of certain multistage vehicles and to "turn over" the guidance to the onboard systems at the appropriate time during the mission.

c) The use of the NASA STADAN net is useful for long-term tracking of spacecraft in earth orbit. Its use is suggested for the synchronous earth orbit mission (after final orbit insertion) for long-term orbit determination and station-keeping. The vehicle equipment required is normally associated with the mission payload and not considered to be part of the launch vehicle guidance.

d) The NASA Unified S-Band and DSIF nets provide excellent coverage and orbit determination capabilities for the lunar and interplanetary missions. These systems require extensive ground communications and computational facilities. The USBS is generally limited to near-earth and lunar missions. The DSIF net extends this capability to interplanetary distances.

e) The use of the DSIF for tracking and orbit determination is virtually a necessity for the interplanetary missions. Although completely autonomous onboard optical/inertial systems may be conceived for these missions, the required performance is considerably beyond the present state-of-the-art for most missions. An accurate onboard system is required, in any case, for controlling accurately powered maneuvers such as midcourse corrections and orbit insertion maneuvers.

2.2.2 Radio Guidance System Concepts and Tradeoffs

The methods of implementing radio command guidance which were considered in this study are:

a) A ground-based computer, receiving information from a radar or radar net during powered flight, computes engine on-off commands and transmits turning rate commands to an onboard attitude control system.

An example of this type of system is the radio-guided Atlas (GE Mod III System). It requires a minimum of onboard inertial equipment but is satisfactory only for near-earth operations because of transit time delays. It also has the disadvantage of constraining the maneuver times because of incomplete coverage. A second example of such a system is the BTL radio/inertial system used for Thor/Delta and other vehicles. In both systems a radar is used to track during powered flight and a filter is used to estimate the position, velocity, and acceleration components. Because the acceleration components are estimated by the filter, only a minimum of inertial equipment (an autopilot) is required. The system errors are the result of an optimum weighting between the radar noise and the vehicle uncertainties (thrust, I_{sp}, mass) and autopilot gyro drifts.

b) A ground-based computer, receiving information from a radar net during free flight, computes the time of initiation, direction, and magnitude of a desired velocity increment. The required onboard equipment includes a sequencer to start and control the burn, an attitude reference system including optical alignment devices, and an integrating accelerometer. This type of system was used for the Ranger/Mariner midcourse corrections and is satisfactory mainly for small burns.

The errors in this type of system are in determining the desired velocity increment and in the execution of the burn. The errors in determining the desired velocity increment are the result of errors introduced during free flight tracking by radar noise and biases. Execution errors are the result of inertial and optical instrument errors and vehicle dispersions. The vehicle dispersions cause errors in three ways:

1) Thrust misalignments and center-of-gravity offsets introduce directional errors. It is possible to use accelerometers to sense and correct these errors.

2) Thrust, weight, and I_{sp} dispersions cause the burnout position to deviate from nominal. Without onboard computing capability, the velocity increment cannot be modified to compensate for these errors.

3) Thrust tailoff impulse dispersions cannot be corrected for by guidance unless vernier engines are provided.

c) Ground tracking during free flight is used to provide a position and velocity estimate which is used to update a complete inertial guidance system onboard the spacecraft. The Apollo mission will utilize this type of guidance.

   The errors in this system are caused by radar noise and biases during free flight tracking and inertial and optical instrument errors as well as thrust tailoff impulse.

d) Inertial guidance is used without radio aid. Although this type of guidance is conceivable for a synchronous satellite mission, it is totally unfeasible for a lunar or interplanetary mission unless some sort of terminal navigation sensor is used. Depending on the mission requirements, this may be beyond the current state-of-the-art.

The candidate radio/inertial systems considered in this study are shown in Table 2-II and include systems of all four types.

2.2.3 Descriptions and Error Models for the Candidate Tracking Systems

Descriptions of the BTL and GE Mod III radio/inertial guidance systems, the NASA STADAN system, the C-Band and S-Band (USBS) trackers, and the Deep Space Instrumentation Facility (DSIF) are given in Subsection 6.3 of Volume II. Further information on the NASA systems is given in Ref 8 through 15. Error models for these systems (with the exception of STADAN) are given in Volume II, Subsection 6.4. Only the error model for the DSIF tracking system, which plays an essential role in the interplanetary missions, is summarized here.

Ref 15 gives the present and projected (1970's) DSIF capabilities. The primary errors are range-rate measurement errors as shown in Table 2-III for two-way doppler tracking (nondestructive T-count).

Table 2-II. Candidate Systems - Radio/Inertial Guidance System Combinations

Type	Ground Based System	Vehicle Subsystems	Tracking System	Limitations
1	Tracking radar computer, data link to vehicle Cutoff and steering commands generated in ground computer on the basis of tracking information and transmitted to vehicle	Transponder/Decoder Autopilot Gyros torqued by turning rate commands	GE-Mod III (Atlas) BTL (Titan I, Thor Delta)	Existing systems limited to launch guidance only Minimum elevation angle 5 deg
2	Tracking radar(s), ground communication net, computer, data link to vehicle	Partial inertial system (e. g. Ranger/Mariner) • Attitude ref. (optically aided) • Single accel.	C-Band radars	• USBS limited to near earth and lunar missions • Appreciable time required for gathering data and computing trajectory
3	Tracking data used to compute orbit. Orbital data transmitted to vehicle over data link	Complete inertial system • Attitude ref. (optically aided) • 3-axis IRU • Computer	S-Band radars (USBS) DSIF	• C-Band systems limited to low altitude earth orbits; coverage limited
4	None	Complete inertial system (IRU and computer) with optical aids (as required)	None	Adequate for synchronous orbit injection, lunar/planetary orbit injection

Table 2-III. DSIF Error Model (From Ref 15)

	Range Rate Measurement Errors	
	Present (Mariner Mars)	1970's
Guaranteed accuracy at 1 AU	0. 5 Hz (0. 030 m/sec)	0. 015 Hz 0. 001 m/sec
Probable accuracy under same conditions	0. 010 Hz 0. 0006 m/sec	0. 003 Hz 0. 0002 m/sec

For purposes of this study, a conservative value intermediate between the guaranteed and probable accuracies for the 1970 time period has been selected (essentially equivalent to the present probable accuracy). In addition, a range-rate bias error is assumed, uncorrelated from station to station. These errors are shown in Table 2-IV.

Table 2-IV. DSIF Range Rate Errors Assumed for Analysis Purposes

Error Source	RMS Error
Uncorrelated noise on doppler rate	0. 732 x 10^{-2} m/sec (equivalent to 0. 12 ft/sec per 1 sec sample, 25 measurements averaged) (also equivalent to 0. 0006 m/sec uncorrelated rms error a 1 sample/min)
Range-rate bias	10^{-2} m/sec (0. 0328 ft/sec)

2.2.4 Limitations, Constraints, and Performance Capabilities

Radio guidance performance capabilities, limitations, and constraints for the earth orbit, lunar, and interplanetary missions are summarized below. The requirements will be discussed for each significant mission phase as indicated in Tables 2-V and 2-VI.

Table 2-V. Mission Phases for Lunar and Interplanetary Missions

Mission	Launch Through Parking Orbit Insertion	Parking Orbit Coast	Translunar or Interplanetary Orbit Insertion Burn	Midcourse Correction	Terminal Burns for Orbit Insertion
Lunar	X	X	X	X	X
Mars	X	X	X	X	X
Jupiter	X	X	X	X	

Table 2-VI. Mission Phases for Synchronous Earth Orbit Mission

Launch Through Parking Orbit Injection	Coast in Low Altitude Parking Orbit	Transfer Burn	Transfer Orbit Coast	Final Injection Burn at Sync. Altitude

2.2.4.1 Use of Radio Guidance - Launch Through Parking Orbit or Interplanetary Orbit Insertion

Radio guidance is currently in use for several NASA launch vehicles (Atlas/Agena, Thor/Delta, Titan II/Gemini) and AF launch vehicles, Titan III, Atlas/Agena, Thor/Delta). Launch phase radio guidance is provided for these vehicles using either the GE Mod III or BTL radio/inertial guidance systems. In all cases the tracking radar is located in the vicinity of the launch site and tracks the vehicle to the lower elevation angle limit (5 to 10 deg, depending on the mission accuracy requirements). By suitably shaping the launch trajectory to maintain acceptable elevation and vehicle antenna look-angles, accurate guidance can be provided through the first two, and portions of the third, stages of powered flight. For Atlas/Agena, guidance is assumed by a simple onboard inertial system (attitude reference, programmer, and a single axially mounted accelerometer for thrust cutoff) during the Agena burn. The radio guidance serves to initialize the inertial system.

A number of difficulties are encountered in extending the use of radio guidance to vehicles employing high performance upper stages (Atlas/Centaur) or requiring additional stages to meet the requirements of higher energy missions. As indicated below and in Section 6, Volume II, the best available tracking radars suitably located at downrange sites will meet the launch phase guidance requirements for many lunar and interplanetary missions. However, there are severe siting and related problems such as acquisition and vehicle antenna coverage. Some payload (weight) penalties and launch azimuth (and consequently launch window) constraints are incurred due to tracking system geometrical constraints. Trajectories are, in general, limited to direct ascent types. The whole approach of using radio guidance with parking orbit trajectories appears impractical. (See Paragraph 6.5.1.1, Volume II, for further discussion).

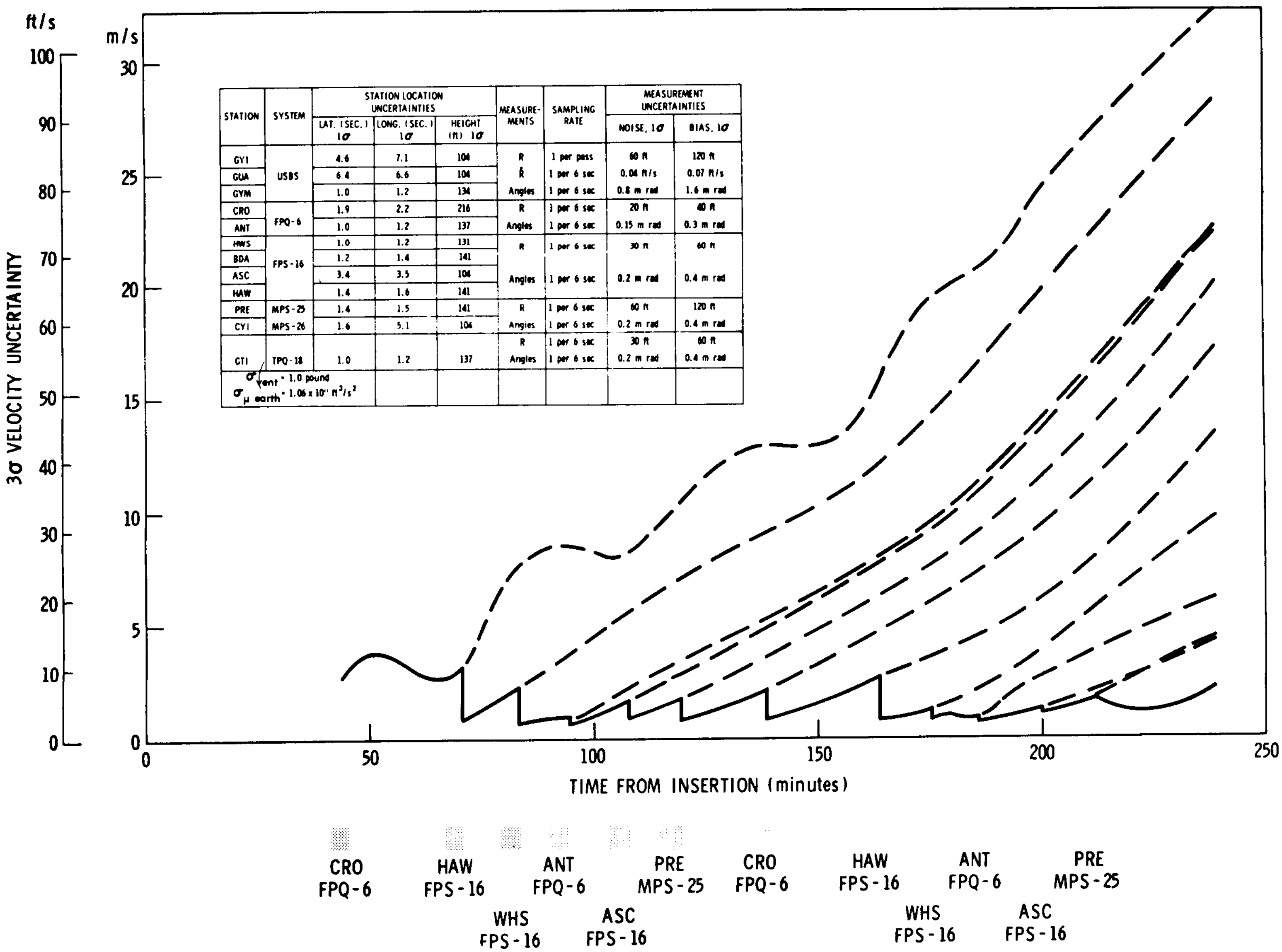
The analysis of radio guidance feasibility and performance during the launch through injection phases has been based on a lunar mission, and an Atlas/Centaur trajectory has been assumed. Performance results are presented in Volume II, Paragraph 6.5.1.2. The performance criterion used is the midcourse ΔV correction required to correct the miss and time of flight errors at the moon due to the launch guidance errors. Typical Figure of Merit (FOM) values for this mission are 10 m/sec (1σ).

2.2.4.2 Orbit Determination Accuracy During Earth Orbit Coast

Numerous studies have been made of orbit determination accuracies for spacecraft in low and high altitude earth orbits in support of Mercury, Gemini, Apollo, and other NASA and DOD space programs. Some results from these studies that are particularly pertinent to the present study are summarized here and in Volume II, Paragraph 6.5.2. Use of all available NASA C-Band and USBS tracking stations is assumed, as is the availability of appropriate computing facilities for near-real-time orbit computation.

Figure 2-2 presents some typical results from Ref 11 showing the orbit determination accuracies for a vehicle in a low altitude (185 km) earth orbit. The C- and S-Band stations and their tracking periods are shown along the bottom of the figures. The dashed lines show the degradation in the vehicle velocity uncertainties if tracking is terminated at the points indicated. The need for multiple stations is evident.

For lunar and planetary missions utilizing relatively short parking orbit ascent trajectories and for extended duration low-altitude earth orbits, the conclusion is drawn that the use of radio guidance is not



STATION	SYSTEM	STATION LOCATION UNCERTAINTIES LAT. (SEC.) 1σ	LONG. (SEC.) 1σ	HEIGHT (ft) 1σ	MEASURE-MENTS	SAMPLING RATE	MEASUREMENT UNCERTAINTIES NOISE, 1σ	BIAS, 1σ
GY1	USBS	4.6	7.1	104	R	1 per pass	60 ft	120 ft
GUA		6.4	6.6	104	$\dot{R}$	1 per 6 sec	0.04 ft/s	0.07 ft/s
GYM		1.0	1.2	134	Angles	1 per 6 sec	0.8 m rad	1.6 m rad
CRO	FPQ-6	1.9	2.2	216	R	1 per 6 sec	20 ft	40 ft
ANT		1.0	1.2	137	Angles	1 per 6 sec	0.15 m rad	0.3 m rad
HWS	FPS-16	1.0	1.2	131	R	1 per 6 sec	30 ft	60 ft
BDA		1.2	1.4	141				
ASC		3.4	3.5	104	Angles	1 per 6 sec	0.2 m rad	0.4 m rad
HAW		1.4	1.6	141				
PRE	MPS-25	1.4	1.5	141	R	1 per 6 sec	60 ft	120 ft
CY1	MPS-26	1.6	5.1	104	Angles	1 per 6 sec	0.2 m rad	0.4 m rad
GTI	TPQ-18	1.0	1.2	137	R	1 per 6 sec	30 ft	60 ft
					Angles	1 per 6 sec	0.2 m rad	0.4 m rad

σ_{vent} = 1.0 pound
$\sigma_{\mu\ earth}$ = 1.06×10^{11} ft^3/s^2

Figure 2-2. 3σ Velocity Uncertainties for a 90° Launch Azimuth Earth Parking Orbit (from Reference 11) practical for the missions and vehicles covered in this study. This is due to a combination tracking system coverage limitations, tracking system performance limitations and time delays inherent in gathering the data, transmitting it to a central computing facility, reducing the data, computing vehicle commands, and transmitting these commands via data link to the orbiting vehicle.

For the synchronous orbit mission, it is shown in Section 7, Volume II, that a navigation update† is required prior to synchronous orbit injection for missions that involve long parking orbit coast times. This correction can be made by either of two methods:

a) Radio tracking during the transfer orbit coast to determine the position error. The major part of the error can be removed by proper adjustment of the time of initiation of the final orbit insertion burn.

b) Use of an onboard electro-optical sensor (e.g., a sun sensor to establish a "line-of-position" fix at some point during the transfer orbit coast. The position error is removed as in a) above.

The feasibility of method a) depends on the availability of suitably located tracking stations. The desired location depends on the longitude of the satellite desired after injection into the final synchronous orbit. Although it may be possible to select suitable tracking stations for most final longitudes of interest, some operational and trajectory constraints are evident.†† The use of the second method, which can be implemented entirely within the onboard system, appears very attractive. Further study of this technique is recommended.

---

†This is in addition to the attitude updates required prior to the transfer burn and final orbit insertion. The errors to be corrected are primarily the accumulated position errors.

††It is also possible to use different modes of ascent from the one studied here. One commonly used technique is to inject the satellite into an equatorial orbit whose period is substantially different from 24 hr and let the satellite "drift"to the required longitude, at which point the orbital period is corrected.

For tracking a spacecraft after injection into the final synchronous orbit, ground based tracking is somewhat more useful. Such a capability is useful for long-time stationkeeping which requires periodic orbit prediction and adjustment that may be easily implemented with either the S-Band systems or the NASA STADAN net. The latter system is recommended for this purpose.

2.2.4.3 Orbit Determination Accuracy During Translunar Trajectory Phases Using the S-Band Tracking Systems

Exhaustive studies have been made of orbit determination accuracies for lunar missions in support of the Apollo (Reference 16), Lunar Orbiter, and other programs. Similar but less exhaustive studies have been made for various interplanetary missions (References 4 and 5). Some results from these studies are summarized here that are pertinent to the present study. Additional study results for the Mars mission are presented in Section 9, Volume II and are summarized in Subsection 4.3 of this volume.

The results of tracking accuracy studies are normally computed in the form of state-vector uncertainties as a function of time from injection. The quantities used to represent the uncertainties are the square root of the sum of the variances of the three position and velocity components.

The results presented in Reference 16 and summarized in Section 6, Volume II, indicate that launch azimuth, earth orbital coast type, flight time, and launch data have effects on DSIF tracking during the early portion of the flight due to their effects on coverage. In the latter portion of the trajectory, the accumulated accuracy of DSIF tracking is nearly independent of the trajectory. Flight time is the only trajectory parameter with a noticeable effect on the latter portion of the trajectory. C-Band radar is found to be useful in reducing uncertainties in the early part of the flight, but it is limited to tracking the first 1.5 hr of the trajectory. The addition of range information to this network gives a marked improvement in tracking accuracy.

Table 2-VII presents some typical results of position and velocity uncertainties at encounter for various tracking system configurations with and without the simulation of midcourse correction effects. The trajectory used is described in Paragraph 6.5.3.4, Volume II.

Table 2-VII. Translunar Trajectory Determination Accuracies

Data Type	Midcourse Correction Effects Not Included		Midcourse Correction Effects Included	
	1σ Position Uncertainty (km)	1σ Velocity Uncertainty (m/sec)	1σ Position Uncertainty (km)	1σ Velocity Uncertainty (m/sec)
DSIF (range, range rate, angle data)	0.1	0.06	0.8	0.46
DSIF (no range)	2.1	1.5	3.7	2.9
C-Band radar	1.9	0.37	---	---

The USBS/DSIF network assumed to be tracking the spacecraft during the translunar trajectory consists of Goldstone, Canberra, and Madrid. Table 6-VII, Volume II, lists the locations of these stations.

Additional results indicating the tracking capability during the translunar trajectory with earth-based radar are presented in paragraph 6.5.3.4, Volume II. Certain generalizations can be made, keeping in mind the assumptions of this study.

The position and velocity uncertainties associated with radar tracking only may be characterized by the following properties:

a) Sensitivity over the early portion of the trajectory to launch azimuth, type of coast, flight time, and date of launch, due to changes in tracking coverage

b) Insensitivity to the trajectory parameters over the latter portion of the trajectory

c) Large uncertainties in the downrange direction (measured in orbit plane coordinates)

d) Sudden drops in the overall uncertainties at the start of periods of simultaneous or near simultaneous tracking by two stations when range data are used.

In general, it can be said that DSIF tracking is greatly improved by the addition of range information, particularly if simultaneous or near simultaneous tracking by two stations is possible.

2.2.4.4 Interplanetary Orbit Determination Accuracy Using DSIF

The use of DSIF for tracking and orbit determination is virtually a necessity for the interplanetary missions considered in this study. The results presented in Subsection 4.3 for the Mars orbiter mission show that a completely autonomous onboard optical/inertial system cannot meet the desired mission accuracy requirements within the present (or near future) state-of-the-art. However, use of the onboard optical/inertial system in conjunction with DSIF is extremely attractive both in terms of accuracy and operational utility. In this mode of operation, DSIF is used as the primary source of accurate position and velocity data (with respect to the earth) and the onboard system is used to accurately control the mid-course, orbit insertion, and orbit trim maneuvers. Use of onboard sensors is also helpful in determining the spacecraft orbit relative to a planet whose position with respect to the earth is uncertain to a significant degree. See Section 8 of Volume II for a more detailed discussion.

The orbit determination accuracies attainable with DSIF depend on the mission trajectory and will change significantly throughout the mission. Detailed results are presented in Subsection 4.3 for the Mars orbiter mission using the trajectory described in Section 3. Table 2-VIII presents some approximate present and future capabilities for the Mars mission. A comparison is also made with the expected errors at encounter in the absence of tracking data for a typical launch injection guidance error of 10 m/sec.

Table 2-VIII. Approximate Trajectory Determination Accuracies for a Mars Mission

	Error At Encounter
Launch Injection Guidance Only	
10 m/sec	90,000 - 200,000 km
Earth Based Tracking Using DSIF	
Present (Mariner 4 Results)	
• 5 days after injection	2400 km
• All data including post encounter tracking	500 km
Future (1971)	
• Injection - 5 days	1000 km
• 5 - 120 days	150 km
• After 120 days	100 km

2.3 DESIGN CHARACTERISTICS OF THE ONBOARD OPTICAL/INERTIAL GUIDANCE SYSTEM

2.3.1 Strapdown Inertial Guidance Subsystem

2.3.1.1 Subsystem Design Characteristics and Instrument Selection

Based upon the inertial equipment survey presented in Volume III (Part II), two representative strapdown Inertial Reference Units (IRU's) were configured for purposes of this study. These IRU mechanizations, denoted by TG-166 and TG-266, are based on presently available inertial instruments and represent a range of readily achievable performance capabilities. Characteristics of the selected IRUs are shown in Table 2-IX.

Table 2-IX. Inertial Instrument Selection and Physical Characteristics of the TG-166 and TG-266 Inertial Reference Units

IRU Model No.	Gyros	Accelerometers	Volume (cm^3)	Weight (kg)	Power (w)
TG-166	Nortronics GIK7	Kearfott Model C 702401-005	8,200	8.7	72
TG-266	Honeywell GG334	See Volume IV for selected accelerometer†	11,000	13.0	83

Although a particular choice of instruments was made for purposes of this study, it is not intended that this choice constitutes a recommendation for development of IRUs based on these instruments. The major motivation for choosing these particular instruments was 1) to span the range of currently available performance capabilities and 2) to select instruments on which a reasonable amount of test data was available to TRW for the purpose of constructing error models.

The TG-166 is an IRU with moderate performance (accuracy) and is available at moderate cost. The TG-266 represents a higher performance IRU subsystem and is available at a higher cost.

The strapdown configuration for both candidate IRUs consists of three single-degree-of-freedom gyros and three accelerometers mounted in an orthogonal triad. A functional block diagram of the TG-166 and TG-266 IRU mechanizations is shown in Figure 2-3. Both mechanizations employ pulse torqued†† gyros and analog rebalanced accelerometers with analog-to-digital converters providing an interface with the digital computer.

†In order to permit an unclassified presentation of performance data in this section, the identification of the TG-266 accelerometer is made in the Classified Annex, Volume IV.

††Pulse torquing techniques are discussed in detail in Appendix A to Volume II.

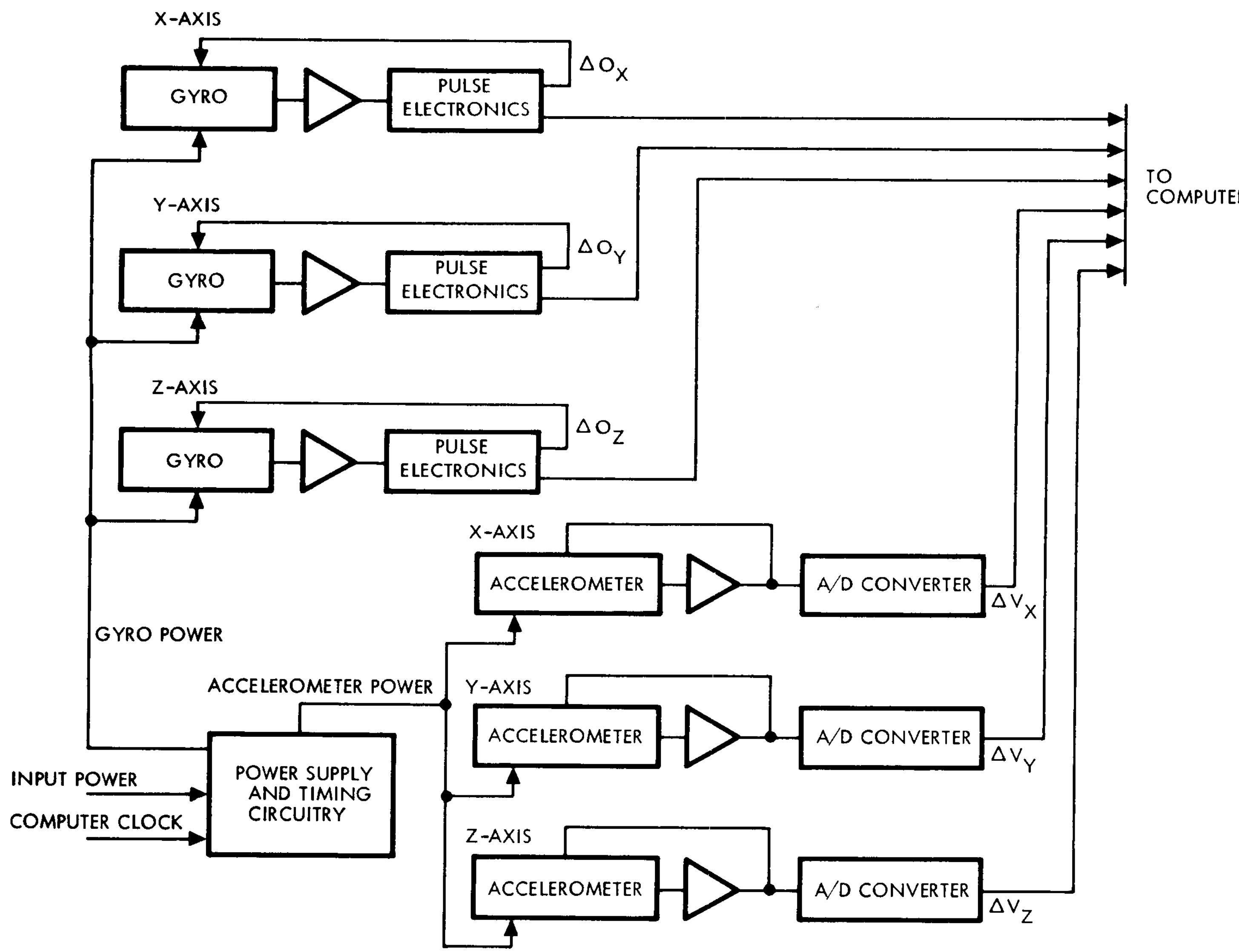


Figure 2-3. TG-166 and TG-266 System Block Diagram

The actual system and loop configurations of the two IRUs are the same except that the TG-266 accelerometer loop utilizes a servo position amplifier instead of a force-to-balance loop.

2.3.1.2 Subsystem Performance Characteristics

Error models for the two IRU configurations are summarized in Table 2-X. A detailed discussion and derivation of the error models is given in Section 4 of Volume II. Figure 2-4 shows the instrument axis orientations assumed.

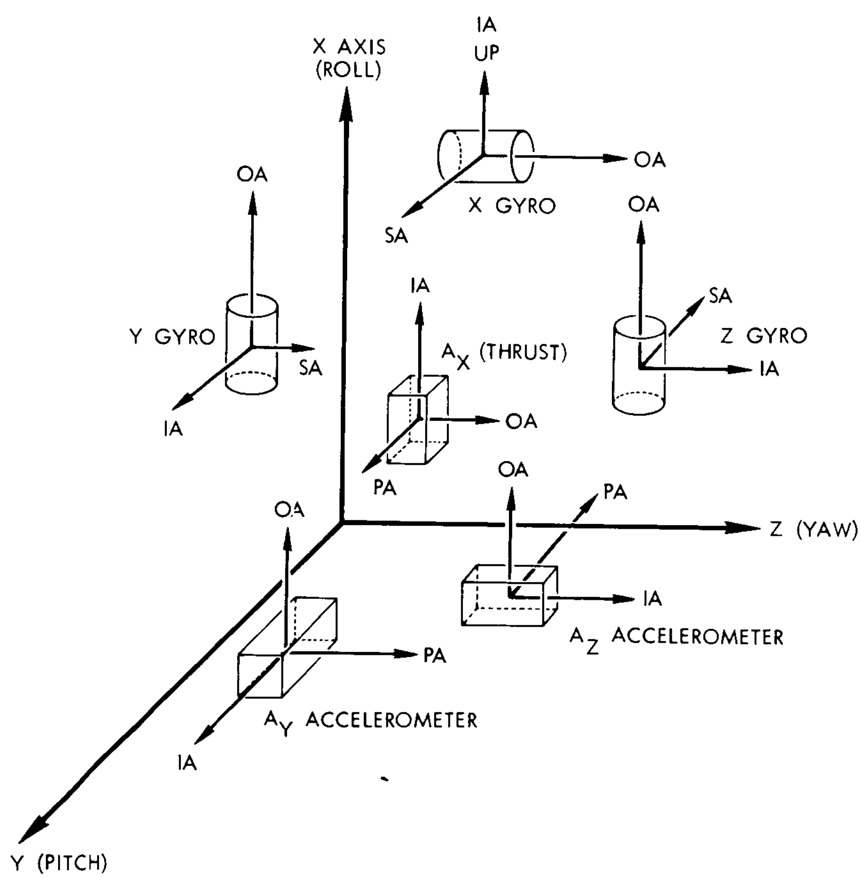


Figure 2-4. Strapdown Coordinate Axes (Prelaunch Orientation)

Table 2-X. Summary of Performance Characteristics for Strapdown Inertial Reference Units TG-166 and TG-266

Description	TG-166	TG-266	Units
Accelerometer			
Bias	21	14	μg
Scale factor	75	24	$\mu g/g$
x acc. input axis rotation toward y axis	12	10	arc sec
x acc. input axis rotation toward z axis	12	10	arc sec
y acc. input axis rotation toward z axis	12	10	arc sec
Pendulous axis g sensitivity	15	10	$\mu g/g$
Output axis g sensitivity	1	1	$\mu g/g$
Input-pendulous g product sensitivity	50	30	$\mu g/g^2$
Input-output g product sensitivity	0. 5	0. 5	$\mu g/g^2$
Gyro			
Bias drift	0.187	0. 09	deg/hr
Input axis g sensitive drift	0. 627	0. 16	deg/hr/g
Spin axis g sensitive drift	0. 627	0. 16	deg/hr/g
Output axis g sensitive drift	0. 02	0	deg/hr/g
Anisoelastic drift	0. 04	0. 04	$deg/hr/g^2$
Scale factor	57	26	ppm
Gyro input axis rotations toward each of other two axes	10	10	arc sec
Gyro input axis rotations toward each of other two axes	10	10	arc sec

The error model coefficients were derived from hardware sensitivities presented in Subsection 4. 3, Volume II. These sensitivities were derived from actual test data, information obtained from the instrument manufacturers, and TRW circuit design studies. In those cases where data were not available, an attempt was made to estimate the error sensitivity terms in a conservative fashion. Although several terms of the error model had to be obtained by this method, the sensitivities which were estimated are generally insignificant in practice.

Two additional error models are presented in Volume II for each configuration, one in which a calibration update is performed just prior to launch and one without an update. This correction or updating would be made to the thrust axis accelerometer bias and scale factor and the roll axis gyro fixed drift and mass unbalance along the spin axis within a few hours of launch. The calibration update is derived from a measurement of the output of the thrust accelerometer and roll gyro immediately before or after the system is installed in the launch vehicle and a second measurement just prior to flight. It is shown in Subsection 4. 7 (Volume II) that the system statistical figure of merit can thereby be improved.

2. 3. 2 Electro-Optical Sensors

2. 3. 2. 1 Sensor Selection and Utilization

The method of implementation which has been considered in this study is that of a strapdown inertial guidance system in which the electro-optical sensors are used to update system alignment and bound the errors due to gyro drift. In addition, the electro-optical sensors may be used for regaining control of spacecraft attitude after a complete power shutdown during an interplanetary coast phase or after recovery from a complete power failure.

The candidate electro-optical sensors which have been selected are based upon those defined in a state-of-the-art survey presented in Volume III (Part I) of this report. Information in this survey was obtained either directly from manufacturers and research laboratories or was extracted from applicable data compiled under the USAF Standardized Space Guidance System Study (Reference 7). Both the current state-of-the-art and projected advancements were defined in the survey and the following types of optical sensors were included:

- Sun sensors, including both nulling devices and solar aspect sensors
- Earth sensors, including both horizon sensors for use in earth orbit and long-range earth sensors for use in interplanetary flight
- Star trackers, including both gimbaled and strapdown subsystems using both mechanical and electronic scanning, and photoelectric or solid-state optical radiation detectors
- Star field sensors, using photoelectric and solid-state detectors with either mechanical or electronic scanning techniques
- Planet sensors for terminal approach or planetary orbit, employing both mechanical and electronic scanning.

It was determined that the four specified missions could be accomplished utilizing various combinations of sun sensors, earth sensors, a Canopus sensor, and a planetary approach sensor. Only in the case of the Mars orbiter mission was it determined that state-of-the-art equipment was not applicable. In this case it was determined that, in order to obtain a higher degree of accuracy, † higher precision would be required for both the Canopus sensor and the planetary approach sensor.

---

† This type of mission can be performed with reasonable accuracy without the use of an approach guidance sensor. More specifically, the early Voyager missions can be accomplished using a combination of an onboard optical inertial system (without the approach sensor) with precision earth-based tracking if the projected improvements in the DSIF can be achieved (see Subsection 2.2). Nevertheless, the accuracy improvement due to use of the approach guidance sensor may be useful for advanced orbiter missions.

In the following paragraphs and in Section 3 the operational sequence of utilization of the selected electro-optical sensors is discussed for several phases of the specified missions. The sensors which have been selected for the various missions are defined in Table 2-XI. The performance and design characteristics of the various sensors and a preliminary design concept for the high-accuracy Canopus and planetary approach sensors are given in Subsection 5.3, Volume II, and are summarized in Paragraphs 2.3.2.2 and 2.3.2.3 following.

Earth Synchronous Orbit Injection

The primary attitude reference will be the inertial elements of the guidance system during launch, injection into the parking orbit, coast-in-parking orbit, injection into the transfer orbit, and injection into synchronous orbit.

Launch and injection into the parking orbit will be accomplished using only the strapdown inertial guidance system. The duration of the parking orbit will vary between 15 min and 12 hr, depending upon the longitude desired for boost into the transfer orbit. If the duration of the parking orbit is long enough to require correction of the inertial reference system prior to boost into the transfer orbit, optical sightings will be utilized at this time. A low-altitude, earth-horizon sensor will be used to obtain a measurement of the vertical, in conjunction with sun sensors to obtain yaw alignment. Two choices of sun sensor configurations are apparent. Using a combination of coarse and fine sun sensors, vehicle maneuvers will be required in order to obtain a solar sighting, after which the vehicle will be returned to the earth-referenced attitude. Alternately, the use of a digital solar aspect sensor will permit a solar sighting to be obtained simultaneously with measurement of the vertical by the earth horizon sensor without requiring vehicle maneuvers. The latter choice is recommended.

After approximately 5 hr in the transfer orbit, correction of the inertial reference system alignment will again be required prior to injection into the earth-synchronous orbit. Again, the sun will be used as a reference for correcting the vehicle attitude in yaw, and the earth will be used as a reference for correction of the vertical. The same

Table 2-XI. Recommended Electro-Optical Sensors for Various Missions

		MISSION AND PHASE								
		EARTH SYNCHRONOUS ORBITER		LUNAR ORBITER		MARS ORBITER			SOLAR PROBE WITH JUPITER ASSIST	
TYPE OF SENSOR	RECOMMENDED SENSOR	Low Altitude Parking Orbit	Transfer Orbit	Lunar Transfer Orbit	Midcourse Correction Maneuver	Interplanetary Transfer Orbit	Midcourse Correction Maneuver	Planetary Approach	Interplanetary Transfer Orbit	Midcourse Correction Maneuver
SUN SENSORS										
Coarse	TRW Systems – VASP Sun Sensors (Modified)	▲	▲	●	●	●	●	●	●	●
Fine	Ball Bros. Research Corp. Fine Eye No. FE-5A	▲	▲	●	●	●	●	●	●	●
Digital Aspect	Adcole Corporation Aspect Sensor No. 1402	●	●							
EARTH SENSORS										
Low Altitude	Advanced Technology Division American Standard Advanced OGO Earth Sensor	●								
High Altitude	Advanced Technology Division American Standard Modified OGO		●							
CANOPUS TRACKER										
Contemporary	ITT Federal Laboratories Lunar Orbiter Canopus Tracker			●	●		●		●	●
High Accuracy	Vidissector Tracker (New Design)							●		
PLANETARY APPROACH SENSOR										
High Accuracy	Digitally Scanned Image Tube (New Design)							●		

● - Primary Choice
▲ Alternate Choice

choice of sun sensors pertains, and the digital solar aspect sensor is again recommended. Due to the reduced angular subtense of the earth at synchronous altitude, a high-altitude earth sensor must be used.

Lunar Orbiter

As in the case of the earth-synchronous orbiter, the inertial elements of the guidance system will be utilized as the primary attitude reference for launch, injection into parking orbit, and for coast-in-parking orbit. The duration of the parking orbit may vary from 0 to 20 min. For this short coast phase, no attitude update of the inertial system is required.[†]

After injection into the lunar transfer orbit, the primary attitude references will be the Sun and Canopus. The coarse and fine sun sensors will be used in conjunction with a single-axis Canopus tracker during this phase of the mission. The precision available from these sensors, together with the onboard inertial system and ground radio tracking aids (see Subsection 2.2), are adequate to perform the midcourse correction maneuver and deboost into lunar orbit without the use of additional electro-optical sensors for approach guidance.

Mars Orbiter

Injection into interplanetary transfer orbit will normally require parking orbit coasts not exceeding 30 min. Thus, no optical sensors are required during this phase of the mission.[†] After injection into the interplanetary transfer orbit, coarse and fine sun sensors will be used in conjunction with a Canopus tracker. However, the Canopus tracker is also used for the approach guidance to Mars. To achieve any significant improvement in the approach trajectory estimation over that available

[†] The attitude error accumulated over short parking orbits of 20-30 min duration is at most a few tenths of a degree. (See Section 7, Volume II.) This error creates a lateral velocity error during the orbit injection burn approximately equal to the attitude error multiplied by the velocity accumulated during the burn. The resulting velocity errors (together with other accumulated errors) are well within reasonable correction capabilities for the midcourse maneuver.

with earth-based tracking (Subsection 2.2), very high precision is required during this phase of the mission. Therefore, a Canopus sensor with higher tracking accuracy than that available in state-of-the-art equipment is required for this mission. A preliminary design concept for such a sensor is presented in Paragraph 5.3.4, Volume II, and is summarized in Paragraph 2.3.2.3 following.

Precise sensing of the line-of-sight to the sun is required during approach guidance and may be performed with an available fine sun sensor. With the reference frame established by lines-of-sight to the Sun and Canopus, a planetary approach sensor will be used to define spacecraft position data with respect to the planet by a) defining the direction to the center of Mars, and b) establishing the apparent angular subtense of the planet to permit stadimetric ranging. A preliminary design concept of a high-precision planet sensor, utilizing a high-resolution electronically scanned image tube for this phase of the Mars mission, is presented in Paragraph 5.3.5, Volume II, and is summarized in Paragraph 2.3.2.3 following.

Solar Probe with Jupiter Assist

Again, no electro-optical sensors are required prior to injection into the interplanetary transfer orbit. For the interplanetary cruise phase and for midcourse corrections, the sensors used will be identical to those used for the Mars mission. The use of a planetary approach sensor for approach to Jupiter is not required since no trajectory corrections are made subsequent to the midcourse maneuver.

2.3.2.2 Summary of Sensor Performance Characteristics

A summary of the electro-optical sensors chosen to meet the requirements of the four missions is shown in Table 2-XII. The instrument error values in the table are derived in Volume II, Subsection 5.4.

2.3.2.3 Summary of Sensor Design Characteristics

The design characteristics of the sensors selected for the various missions are described in detail in Section 5, Volume II, and are summarized in the following paragraphs.

Table 2-XII. Summary of Electro-Optical Sensor Errors

Sensor	Type	Null Accuracy			Offset Accuracy (1)			
		Noise (rms) (deg)	Bias Instability (deg)	Alignment Error (deg)	Offset (deg)	Noise (deg)	Bias Instability (deg)	Alignment Error (deg)
Sun Sensors								
Coarse	TRW VASP (Modified)	(2)	±1.0	±1.0	20	(2)	±1.0	±1.0
Fine	BBRC/FE 5A	(2)	±0.022	±0.004	5	(2)	±0.022	±0.004
Digital Aspect	ADCOLE 1402	0.006	±0.04	±0.004	32	0.006	±0.04	±0.004
Earth Sensors								
Low Altitude	ATD Advanced OGC	0.03	±0.09 (3)	±0.016	10	0.03	±0.14	±0.016
High Altitude	ATD Modified OGO	0.04	±0.10	±0.016	2.5	0.04	±0.10	±0.016
Canopus Sensor								
Lunar Mission	ITT Lunar Orbiter Canopus Tracker	0.02	±0.014	±0.016	4	0.02	±0.05	±0.016
Interplanetary (5)	Vidissector Tracker (New Design)	0.002	±0.008	±0.004	4	0.002	±0.008	±0.004
Planet Approach								
Sensor (4) (5)	Digitally Scanned Image Tube (New Design)	0.003	±0.008	±0.004	6.75	0.003	±0.008	±0.004

NOTES:

(1) For two-axis measurement system, Figures given include sum of error in both axes.
(2) Negligible, i.e., < 1 arc sec.
(3) Oblateness error and horizon altitude error included as sources of bias error.
(4) 1.5 deg planet subtense.
(5) Proposed design specifications for new sensors.

Whenever feasible, existing state-of-the-art equipment was selected. Some redesign would probably be required in most cases to implement the interfaces with the digital computer in the manner required by the modular concept described in Subsection 2.1. The design of these interfaces is beyond the scope of this study.

Sun Sensors

The coarse/fine sun sensor combination is utilized in all four missions as an attitude reference during orbit coast and cruise phases and as a means of vehicle reorientation during recovery from power shutdown or failure. The coarse sun sensor design proposed by TRW for the Voyager program is based on a sun sensor developed for the VASP spacecraft. The fine sun sensor is the FE-5A "fine eye" developed by Ball Brothers Research Corporation.

The operation of these sensors is described in detail in Subsection 5.3, Volume II. Figure 2-5 shows the electronic design configuration of the sensors. The specifications for these units are shown in Table 2-XIII.

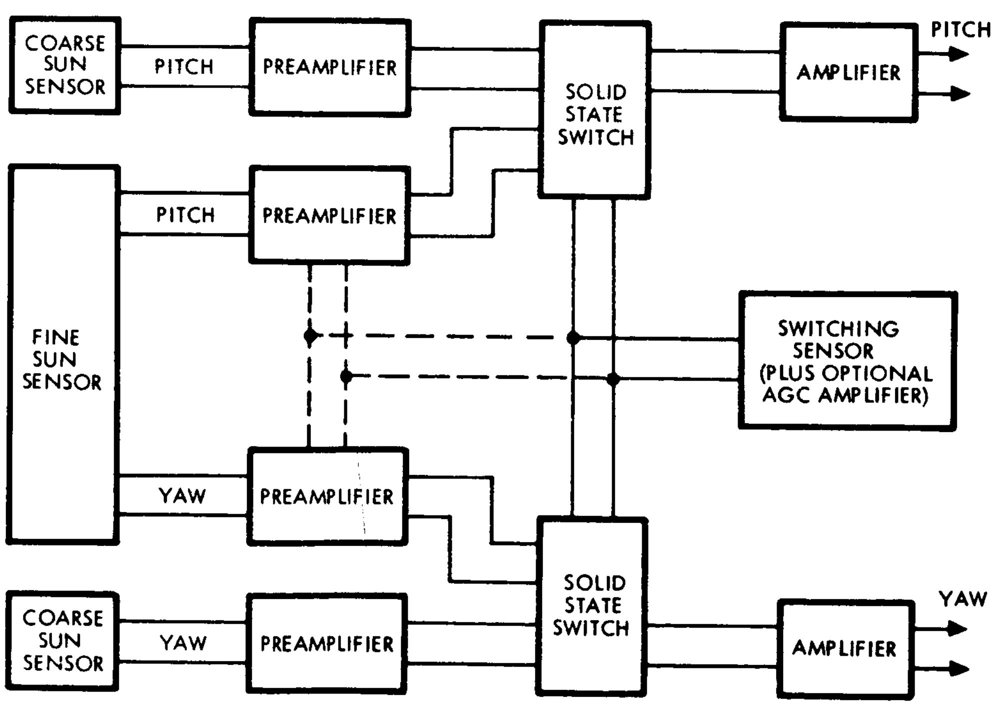


Figure 2-5. Sun Sensor Electronic Configuration

Table 2-XIII. Coarse and Fine Sun Sensor Specifications

Fine Sensor Assembly[†]	
Accuracy (fine eye pair at null)	± arc min
Peak output (short-circuit current in direct sunlight)	1.5 ma nominal
Angular range (fine eye pair)	±15 deg nominal
Angular sensitivity (front edge)	5 μamp/arc min
Spectral response	0.70 to 1.1 μ; 0.81 μ peak
Response time (rise time from 10 to 90 percent of peak value)	20 μsec or less
Resolution	Virtually infinite
Temperature operating range	-20°C to + 85°C
Weight: FE-5A fine eye	6.0 gm
Eye assembly retainer	3.5 gm
Coarse Sensor Assembly	
Field of view	4π ster
Null accuracy (each axis)	±1 deg
Linearity (over ±20 deg each axis)	±10 percent
Physical Characteristics (Includes Electronics)	
Size	330 cm^3
Weight	0.34 kg
Power	700 mw

[†]Manufacturer's specifications.

The digital aspect sensor recommended for the earth-synchronous orbit mission is a device designed and manufactured by the Adcole Corporation. This sensor measures two orthogonal components of the sun's offset from the instrument reference axis and presents the data in digital form. The performance characteristics of this unit are summarized in Table 2-XIV. The operation of the sensor is described in Subsection 5.3, Volume II.

Earth Sensors (Horizon Scanners)

In the parking orbit of the earth-synchronous mission, the half-angle subtended by the earth is approximately 75 deg; at synchronous altitude this half-angle is 8.7 deg. The present status of earth sensor technology precludes precise determination of the vertical over this wide

Table 2-XIV. Digital Solar Aspect Sensor Specifications[†]

Model	Adcole type 1402
Field-of-view	64 x 64 deg
Resolution	1/64 deg
Accuracy	2 arc min
Output	Two 12-bit words
Operating temperature range	-70 to 100°C
Size	1.3 x 1.3 x 2.1 cm
Weight	0.15 kg
Power	None required

[†]Manufacturer's specifications

angular range with one instrument. Therefore, two earth horizon sensors are recommended, one for use in the earth parking orbit and the other for use at synchronous altitude.

The earth sensor recommended for the low-altitude mission is the advanced OGO horizon tracker developed by the Advanced Technology Division of American Standard (Figure 2-6). The instrument consists of four sensors arranged at 90 deg intervals in yaw, utilizing linear scanning and edge tracking of the horizon as shown in Figure 2-7. The method of interconnection of the four sensors (one is redundant) is shown in Figure 2-8. The electronic configuration for a single channel is shown in Figure 2-9. Performance requirements for the instrument and other pertinent operating characteristics are shown in Table 2-XV. The operating principles are described in detail in Paragraph 5.3.3, Volume II.

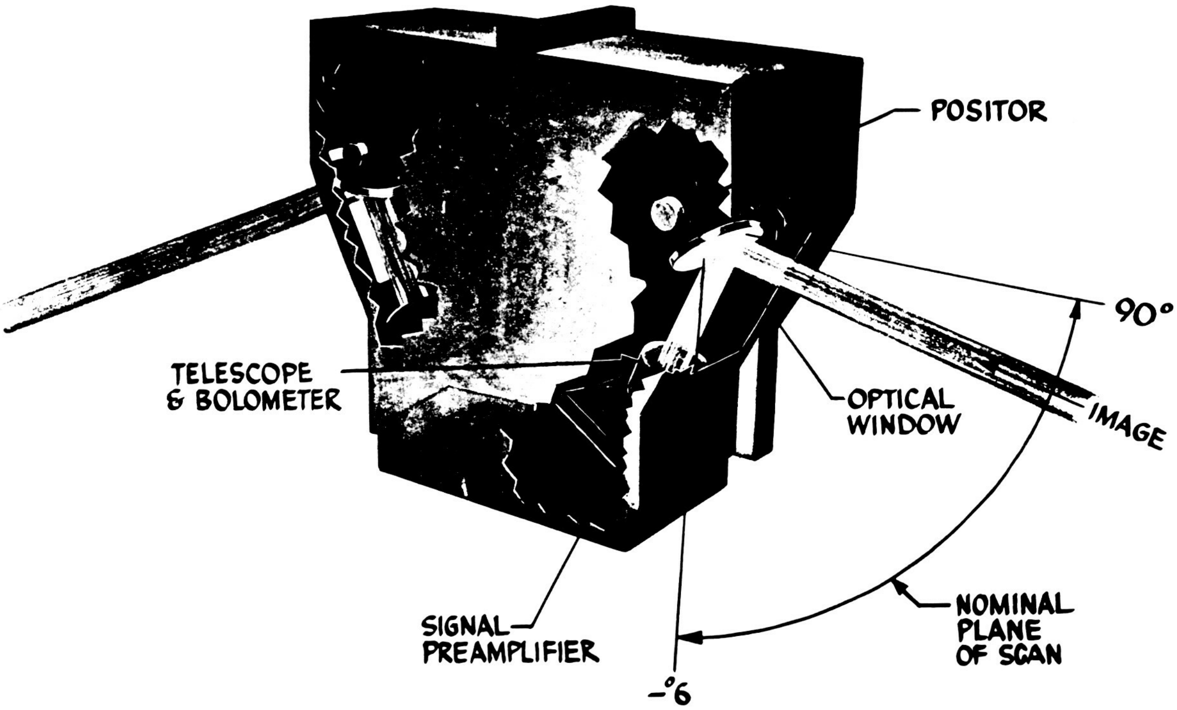


Figure 2-6. A-OGO Horizon Sensor

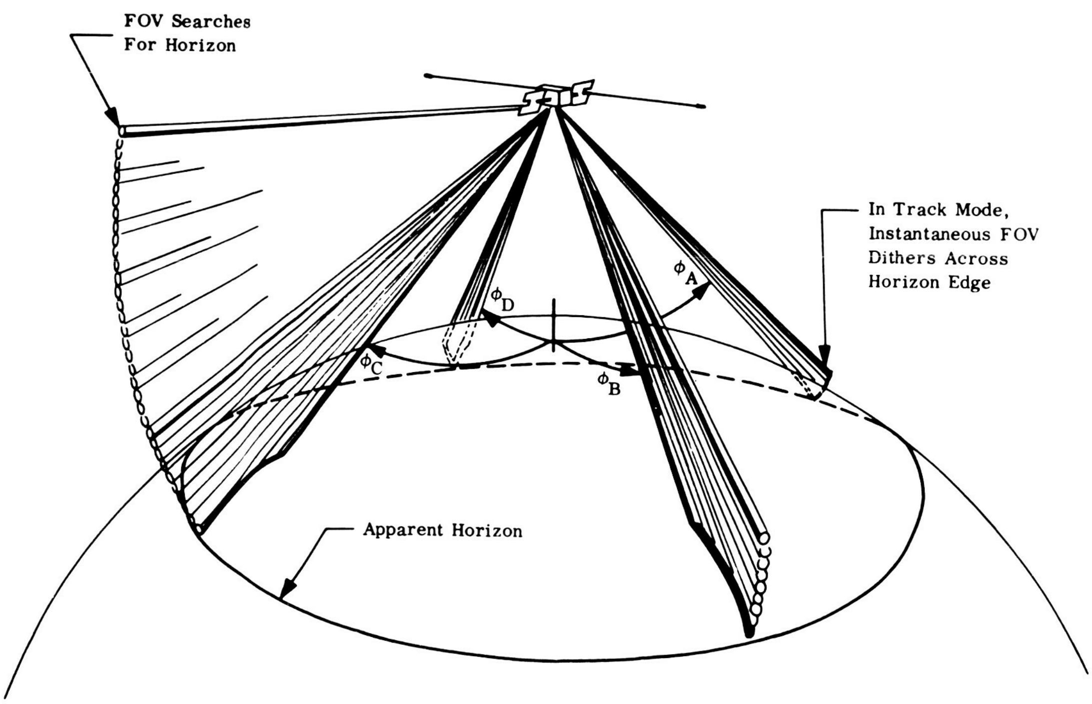


Figure 2-7. A-OGO Horizon Edge Tracking Technique

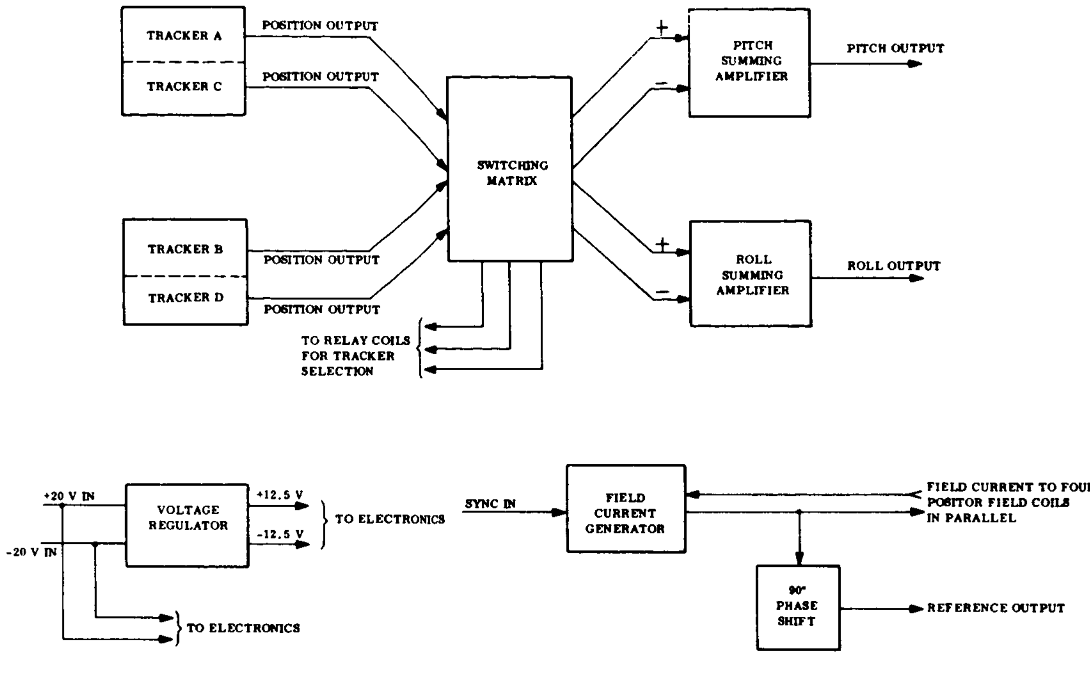


Figure 2-8. A-OGO Horizon Sensor System Block Diagram

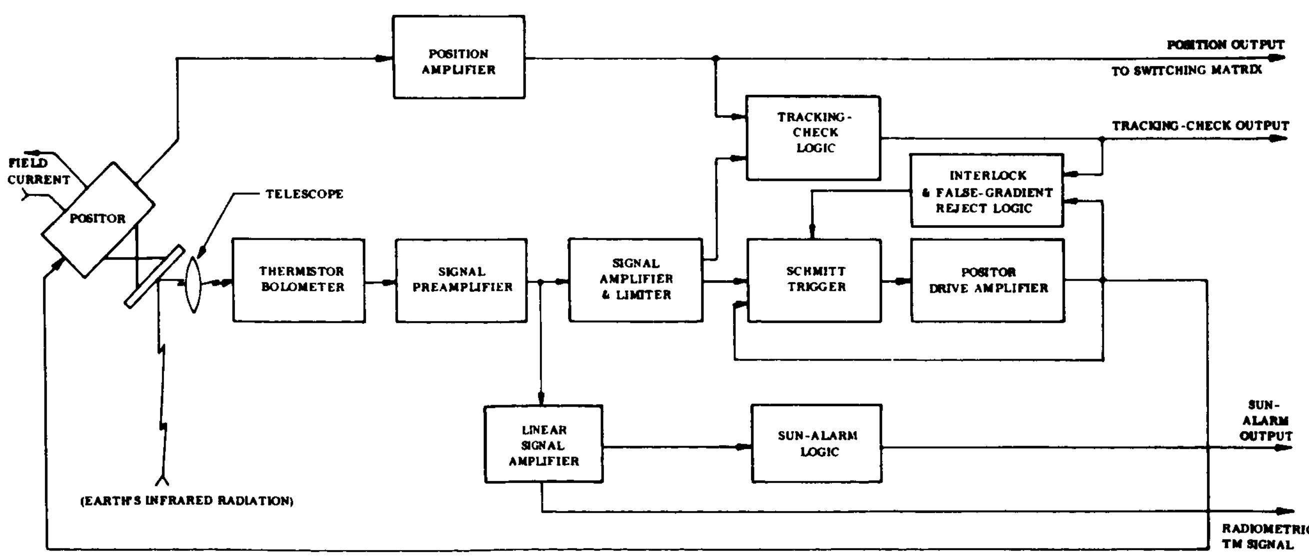


Figure 2-9. A-OGO Tracker Block Diagram (Single Channel)

Table 2-XV. A-OGO Horizon Sensor System Specifications‡

<u>Optical Characteristics</u>	
a) IR detector	Immersed thermistor bolometer
b) IR spectral bandpass	14.0 to 16.0 μ
c) Telescope field-of-view	1.2 deg at half-response contour
<u>Sensor Outputs</u>	
a) Pitch/roll	2461-cps signal with amplitude proportional to roll and/or pitch attitude error
<u>Physical Characteristics</u>	
a) Size	5000 cm^3 (maximum)
b) Weight	7.6 kg (including electronics)
c) Power required	10 w (nominal), 12 w (maximum)
<u>Performance</u>	
a) Tracking range (each of 4 trackers)	-2 to +85 deg (min)
b) Tracking rate	>15 deg/sec
c) Operational range	±30 deg (±45 deg from nominal)
d) Altitude range	220 to 150,000 km or 90 to 110,000 km
e) Accuracy‡‡	$\leq$0.05 deg (3σ)
Null	
±10 deg‡‡‡	$\leq$0.10 deg (bias) + 0.05 deg (3σ)
f) Reliability (for 3 or 4 trackers operating)	0.95 for 1 year (present parts) 0.98 for 1 year (highest reliability parts available)
g) Operational life	$\geq$1 year
h) Storage life	3 years
i) Pitch and roll scale factors	0.4 v rms/deg
j) Position-output scale factor	0.1 v rms/deg
k) Noise	±0.02 deg peak-to-peak at 0.6 Hz bandwidth
<u>Environmental Levels</u>	
See Section 5.0, Volume II	

‡Manufacturer's specifications.

‡‡Excluding geometric cross-coupling errors (which can be calibrated out) and errors due to horizon anomalies and earth oblateness.

‡‡‡Simultaneous roll and pitch.

The earth sensor recommended for high-altitude operation is a modification of the OGO sensor. It utilizes the same scanning mechanism but has only two sensors, arranged at 90-deg intervals in yaw, with linear scanning across the earth's disc in two orthogonal planes. The design characteristics and operating principles of this instrument are described in detail in Paragraph 5.3.3, Volume II.

The selection of these instruments was based on the following considerations.

a) For use in a nonspinning vehicle, edge-tracking, radiance-balance, horizon-sector, and conical scanners may be considered. The radiance balance technique was rejected due to low accuracy. The latter two were rejected from the standpoint of reliability, as rotating mechanisms are required for scan generation.

b) The edge-tracking sensor has the advantage of having a scanning mechanism which utilizes flexural pivots of high reliability.

c) The edge-tracking technique inherently has a higher signal-to-noise ratio than the conical scanning method.

d) The spectral bandpass utilizing the 14- to 16-μ CO_2 absorption band provides improved definition of the infrared horizon of the earth in comparison to previous sensors utilizing infrared wavelengths shorter than 14μ, in which inaccuracies resulted due to discontinuities in the infrared horizon.

Canopus Trackers for Lunar Orbiter and Jupiter Missions

For the lunar orbiter and Jupiter missions, the ITT Federal Laboratories lunar orbiter Canopus tracker is recommended.

The ITT tracker is contained in a single package, consisting of optics, an ITT FW 143 multiplier phototube detector, and electronics. The electronics comprise signal detection circuits, scanning logic, deflection circuits, and power supplies. The tracker provides an analog output signal which is proportional to the angular displacement of the line-of-sight to Canopus about one axis.

A simplified block diagram is illustrated in Figure 2-10. Specifications and physical characteristics of the ITT instrument are given in Table 2-XVI.

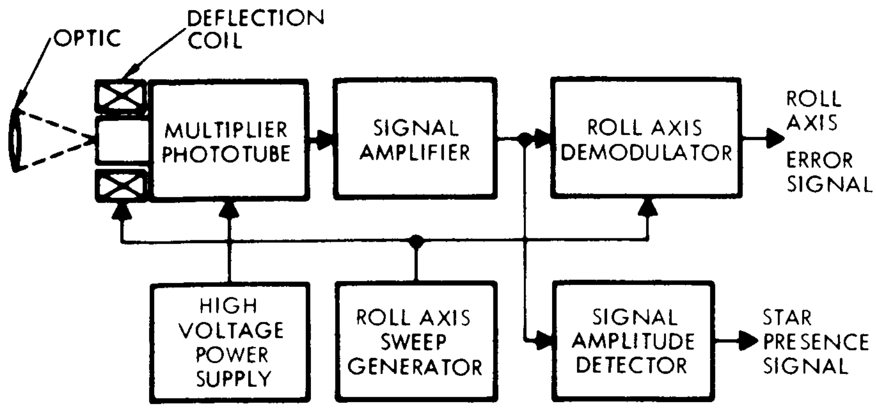


Figure 2-10. Simplified Functional Diagram for ITT Federal Laboratories Canopus Sensor

Table 2-XVI. Specifications and Physical Characteristics of Canopus Sensors

ITEM	LUNAR ORBITER CANOPUS SENSOR	HIGH ACCURACY CANOPUS SENSOR (NEW DESIGN)
1. Manufacturer	ITT Federal Laboratories	
2. Application	Lunar Orbiter	Mars Orbiter (Voyager)
3. Optical System	Refractive – 7 elements + corrector 20 mm f/1.0	Refractive 25 mm f/1.0
4. Detector	ITT Type FW 143 photomultiplier	ITT FW 4012 Vidissector
Spectral Response	S-20	S-11
Aperture Dimensions	0.024 x 0.435 cm	0.044 x 0.64 cm
5. Instrument Field of View		
Roll (total)	8.2° ($\pm 6^\circ$ after acquisition)	8°
Pitch (total)	18°	30°
Instantaneous	1° (roll x 18° (cone)	1° (roll) x 16° (cone)
6. Gimbaling	Mechanical adjustment for launch window	Electronic
7. Scanning		
Roll (search)	$\pm 4.1^\circ$ triangular at 14 Hz	$\pm 4^\circ$ at 10 to 20 Hz
Roll (track)	$\pm 1.5^\circ$ triangular at 800 Hz + dc bias	$\pm 1.5^\circ$ triangular at 1 kHz
Pitch	None	Six (maximum) programmed increments
8. Stellar Sensitivity (threshold)	-1.92 to + 0.08 m	Between 1/4 and 4x Canopus Mag.
9. Linear Range	$\pm 6^\circ$	$\pm 1^\circ$
10. Electronic Bandwidth	0.75 Hz	1.0 Hz
11. Time Constant	0.2 sec (roll axis)	<0.5 sec (roll axis)
12. Acquisition Rate	Not spec	0.1°/sec
13. Error Gradient (at null)	1 v/deg	8 v dc/deg
14. Signal to Noise	24 at 0.1 deg	
15. Accuracy		
At Null		
Noise	15 arc sec rms	0.003° (10 arc-sec) rms 1-sigma
Bias	50 arc sec rms (stability)	$\pm 0.008^\circ$ ($\pm$ 29 arc-sec)
Alignment	Not spec	$\pm 0.004^\circ$ ($\pm$ 15 arc-sec)
Off-Axis		(4° off-axis)
Noise	Not spec	0.003° (10 arc-sec) rms 1-sigma
Bias	Not spec	$\pm 0.008^\circ$ ($\pm$ 29 arc-sec)
Alignment	Not spec	$\pm 0.004^\circ$ ($\pm$ 15 arc-sec)
Major Error Source	I.D. and aperture alignment and edge irregularities	Bias calibration and stability, alignment
16. Sun Protection	CDS sensor – 100 FT-C threshold	Required – 1000 FT-C threshold
17. Weight	3.2 kg (7 lb)	2.7 kg (6 lb)
18. Volume	10.2 x 14.0 x 30.5 cm (4 x 5.5 x 12 in)	10.2 x 12.7 x 30.5 in (4 x 5 x 12 in)
19. Power	3.1 w at 21 vdc, 4.95 w at 31 vdc	6 w plus 2 w for sun shutter

Two other instruments were also considered, based upon the survey presented in Volume III. The first instrument considered had a specified accuracy of $+0.1$ deg and developed by NASA-JPL and the Barnes Engineering Co. for use in the Mariner programs. Although the performance of this instrument has been proven in the Mariner program, the ITT instrument is preferred from the standpoint of higher accuracy.

The second instrument considered was the Canopus tracker developed by Hughes Aircraft Co. (Santa Barbara Research Corporation) for the Surveyor program. Although this instrument has been successfully employed in the Surveyor program, the ITT Federal Laboratories tracker is preferred, primarily from the standpoint of reliability and secondly from the standpoint of higher accuracy.

Canopus Tracker for Mars Mission

Results of computer simulation for advanced Mars orbiter missions (See Section 9, Volume II) have determined that the precise approach guidance may be obtained through utilization of an approach guidance sensor in conjunction with a fine sun sensor and Canopus tracker. A high degree of accuracy is required in all three sensors. The accuracy of the Canopus tracker is beyond that of state-of-the-art equipment. The composite error due to bias calibration, bias stability, and alignment must be in the order of 0.5 arc min.

In considering improvements which can be made to increase accuracy as well as performance and producibility, the following are apparent (although other instrumentation approaches may prove superior).

a) The use of a refractive optical system (in comparison to the wide-angle Cassegrain used in the Mariner instrument) will give improved optical image resolution.

b) The use of the recently developed high-resolution ITT FR012 vidissector is superior to the image dissectors used in contemporary instruments.

A complete preliminary design is not defined since the electronic circuit design approach is not unique and would be similar to that of the

Mariner and Lunar orbiter instruments. However, particular features which are recommended in the design are:

a) Utilization of a low-frequency search scan over a field of ±4 deg in roll and a high-frequency scan of ±1.5 deg superimposed for star acquisition.

b) Maintenance of tracking over a field of ±4 deg after star acquisition.

c) Programmed increments of cone angle adjustment during the course of the mission, accomplished by application of a dc bias to the imaging section of the vidissector.

A proposed specification for the high-accuracy Canopus sensor is defined in Table 2-XV. The accuracy requirements are particularly stringent and will require particular attention to the following:

a) Measurement and calibration of nonlinearities in the image tube angular deflection versus the deflection current transfer function.

b) Consideration of the effects of component aging and the resultant changes in bias level throughout the course of the mission.

c) Consideration of the effects of spacecraft thermal stress, which may result in changes in bias level throughout the course of the mission.

d) Precision in initial alignment of the sensor reference axis with respect to the spacecraft coordinate system, requiring very small angular tolerances in both mechanical alignment and optical simulation equipment.

Planet Approach Sensor

Results of computer simulation of the Mars mission (See Section 9, Volume II and Subsection 4.3 in this volume) have determined that the accuracy of approach guidance may be improved over that obtainable with radio-inertial guidance by the use of an optical planetary approach sensor, providing that the approach sensor has the capability of determining the relative angular position of the geometrical centroid of the planet to an accuracy of 1 arc min. This accuracy will be determined primarily by bias errors and long-term stability of the sensor.

The preliminary design concept of a planetary approach sensor is defined in Paragraph 5.3.5, Volume II and summarized here. In addition to having the capability of determining the clock and cone angle to the geometrical centroid of the planet, it also has the capability of measuring the apparent angular subtense of the planet, permitting stadimetric ranging.

A high-resolution, electronically scanned image tube is chosen as the radiation sensor, primarily from the standpoint of reliability. This type of design permits electronic gimbaling in clock and cone angle during planetary approach and is preferred to the more conventional alternative approach that uses a single-element point detector with mechanical scanning and gimbaling. The rationale for selecting this design approach is discussed in detail in Paragraph 5.3.5.

The configuration selected utilizes both deflection voltages and error signals in digital form to compute clock angle, cone angle, and apparent planetary angular subtense with high precision. (See Figure 2-11.) This computation can either be computed onboard the spacecraft by the guidance system computer, or may be telemetered to earth for use in ground-data processing.

Performance specifications and physical characteristics for the recommended sensor design are given in Table 2-XVI. Accuracy requirements are particularly stringent and will require that particular attention be given to the same four items listed above for the high accuracy Canopus sensor.

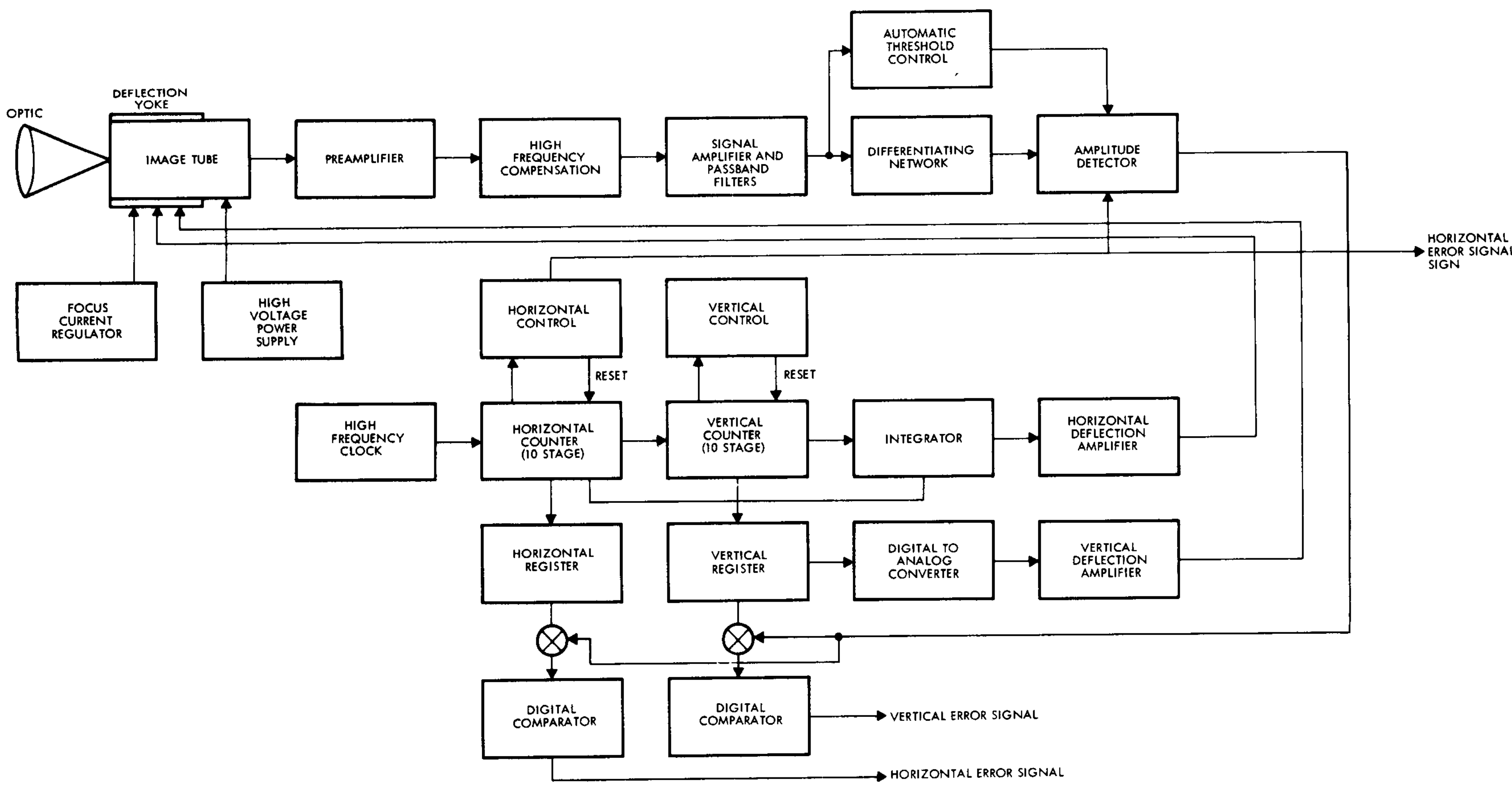


Figure 2-11. Functional Diagram — Approach Guidance Sensor

3. SUMMARY OF MISSION CHARACTERISTICS, GUIDANCE SYSTEM OPERATIONAL SEQUENCES, PERFORMANCE REQUIREMENTS, AND EQUIPMENT CONFIGURATIONS

3.1 INTRODUCTION

This section presents a summary of the mission characteristics and requirements, and the guidance system functional and performance requirements derived from these.

The mission and launch vehicle characteristics, trajectories, and mission performance (accuracy) requirements are summarized under each mission heading. Guidance system functional requirements and operating sequences derived from the mission requirements, vehicle characteristics, and guidance equipment capabilities are specified. Equipment configurations and functional interconnections are presented for each of the missions.

Sections 2 and 3 of Volume II present additional supporting data and analyses made in defining the guidance requirements.

3.2 SYNCHRONOUS EARTH SATELLITE MISSION

For this mission, a typical launch vehicle can be assumed to be the Atlas SLV3X-Centaur[†] with a variety of communication and meteorological satellites as the payload. It is assumed that the satellite payload itself has the capability of providing a ΔV for final orbit trim and station-keeping. The ultimate functional and performance requirements imposed on the kick stage for this mission are to place a payload into a near synchronous earth orbit, at the desired longitude, with sufficient precision that final orbit trim corrections can be performed utilizing the limited propulsion capability of the payload. The kick stage guidance system accuracy requirements may be conveniently stated in terms of the payload ΔV required to correct the residual errors after final injection. Reasonable values lie in the range of 15-30 m/sec.

[†] The payload and coast duration capabilities of this vehicle are severely limited for this mission using the existing Centaur vehicle. For the purposes of this study, these problems are ignored. It is assumed that the Centaur vehicle may be modified to increase the payload capability, to extend the permissible coast duration, and to permit three-burn operation. Another alternative, providing a large increase in payload capability, is to add an upper stage (such as HEUS) to the vehicle. The guidance requirements are not expected to be significantly different for either vehicle concept.

For purposes of this study, it is assumed that the kick-stage guidance system provides the complete guidance and control of the launch vehicle from liftoff through parking orbit insertion, transfer injection, and synchronous orbit injection. Two extremes of ascent trajectories have been considered. In the first, the kick stage is injected into the transfer trajectory to synchronous altitude from a 185-km "parking orbit" at the first equatorial crossing from launch. In the second, the kick stage/payload may remain in the 185-km parking orbit for as long as 12 hr before transfer ignition. These are the extremes of the parking orbit coast period required to reach any desired final longitude for this mode of ascent.

3.2.1 Mission Characteristics

For the purposes of this study, the major events of the synchronous mission developed for the Atlas/Centaur (AC-8 configuration) have been adopted and modified. Following liftoff from the Atlantic Missile Range, a roll is introduced in the launch vehicle to obtain a launch azimuth of 90 deg. The Atlas booster is then controlled up to its cutoff (BECO) by a predetermined booster pitch program.

Injection into the parking orbit is accomplished by using two constant pitch rates selected to achieve the altitude and flight path angle for injection into the 185-km parking orbit. The first pitch rate occurs during the Atlas sustainer flight, lasting for 10 sec after initiation of that phase, while the second rate occurs during the Centaur powered phase. After injection into the parking orbit, the Centaur coasts to the vicinity of the equator (first crossing) at which time the second burn (approximately 1.5 min) injects it into a Hohmann transfer ellipse. This burn is performed with a pitch rate that keeps the Centaur in a fixed attitude relative to the radius vector, and terminated on a predicted apogee altitude equal to that of the required synchronous circular orbit. During the coast in the Hohmann transfer, approximately 5 hr, the Centaur maintains a fixed inertial attitude.

Optimally, minimum energy requirements suggest dividing the orbit inclination plane change between perigee and apogee. For launch from AMR at 90-deg launch azimuth, the orbit inclination is 28.5 deg; approximately 2 deg should be removed at perigee and the remaining 26.5 deg at apogee. For this study, the gains from pursuing this approach do not overcome the complexities introduced. Therefore, the method adopted for the third Centaur burn at apogee was to perform the total orbit plane change simultaneously with injection into the synchronous orbit.

Just prior to reaching apogee, instantaneous yaw and pitch attitude maneuvers were performed to establish an initial attitude for the final burn (approximately 39 sec) such that the Centaur would achieve the correct synchronous orbit. Characteristics of the actual synchronous orbit obtained are:

• Altitude	35,850 km (19,326.5 nmi)
• Longitude	102.7 deg
• Velocity magnitude	3.08 km/sec (10,087.3 ft/sec)
• Eccentricity	0
• Inclination	0 deg
• Period	1436.1 min

After injection into the circular synchronous orbit, the payload separates from the Centaur. Any errors in the resulting spacecraft orbit are then corrected by the spacecraft itself.

Developing the nominal trajectory presented above was contingent upon making the following simplifying assumptions:

a) A mission of this type requires a three-burn capability from the Centaur. Since presently only a two-burn capability is available,† the detailed sequence of events of the second burn was duplicated for a third burn.

b) Payload maximization could be obtained by optimizing several trajectory parameters such as launch azimuth, plane change philosophy, parking orbit altitude, vehicle attitude history, etc. However, for this guidance study, the exact maximum payload weight is irrelevant to the guidance scheme adopted. Hence, no payload maximization analysis was performed.

† A two-burn (Centaur Stage) mission profile is also possible, using the technique as discussed under c). Although the three-burn capability and the extended coast capability required for either mission profile is not in the present Centaur design, these capabilities could be provided by an improved Centaur stage or an alternate stage having similar capabilities. It is beyond the scope of this study to assess the technical feasibility of these design changes.

c) Positioning a 24-hour synchronous spacecraft above a specified longitude may also be accomplished by injecting into an orbit offset from the required circular synchronous orbit. A drift rate results which allows the spacecraft to change its longitude. This drift rate is then removed, and the final orbital corrections are made when the required longitude is reached. Since these corrections would be executed by the spacecraft and not the launch vehicle, guidance techniques for the Centaur would not be affected if such considerations were incorporated into this analysis. Consequently, the spacecraft was targeted directly into the 24-hr synchronous equatorial orbit, thus neglecting offset drift orbit considerations.

d) An eight-orbit phasing coast in a 185-km parking orbit is simulated for certain runs by the analytical propagation of errors in the error analysis program (See Section 7, Volume II). The remarks in b) above concerning Centaur capabilities apply here as well. The event times for the synchronous orbit missions are given in Volume II, Tables 2-II and 2-III, for cases without and with an eight-orbit phasing coast, respectively.

3.2.2 Guidance System Operational Sequence

The guidance system operational sequence during each of the mission phases is summarized below:

a) Launch and boost to ~ 185-km parking orbit:† The strapdown inertial guidance subsystem is presumed to be providing the guidance function for this phase.††

b) Coast in parking orbit for a period t, with t depending on desired longitudinal positioning of satellite ($15 \text{ min} < t < 12 \text{ hr}$): During the coast period, the inertial guidance subsystem is required only to provide vehicle attitude control reference. The exact attitude profile to be followed during the coast phase will depend on the mechanization concept developed; however, at transfer ignition (at equatorial crossing) the kick stage attitude must be at that thrusting attitude required to place the kick stage/payload into the desired transfer orbit. The

† This is a typical value assumed for this study and represents a reasonable lower limit for this type of mission. The parking orbit altitude is chosen as low as possible so as to maximize the injected payload weight. However, below about 185 km, drag effects limit the orbital lifetime of the vehicle.

†† Under certain conditions, radio guidance could also be used. See Subsection 2.2 for a discussion of this possibility.

attitude control during the period immediately prior to transfer ignition might be inertial only or optically aided inertial using earth (horizon) and sun sensors. Ground tracking might possibly be used, particularly for the multiorbit cases, for nagivation updating of the transfer ignition burn time and of the required vector velocity increment. See Paragraph 2.2.4.2 for a discussion of this possibility.

c) Transfer burn to apogee: This phase will be controlled autonomously by the strapdown inertial guidance subsystem.

d) Transfer coast: During the approximately 5-1/4 hr coast in the Hohmann transfer to the apogee at nominal synchronous altitude, the inertial guidance subsystem can again be relegated to the role of an attitude reference set. The kick stage/payload can possibly be ground tracked to determine the apogee burn requirements (See Paragraph 2.2.4.2.) Five to fifteen minutes prior to apogee burn, an absolute inertial attitude update can be provided by celestial sightings.

e) Apogee burn: The apogee burn is designed to circularize the orbit at synchronous altitude and is controlled by the strapdown inertial guidance subsystem. The use of the kick stage is presumably terminated at this time and the payload is separated from the kick stage.

3.2.3 Guidance System Performance Requirements

Because of (1) imperfect tracking or navigation during the transfer coast and (2) thrusting attitude and ΔV errors of the kick stage at apogee burn, the payload orbit will be imperfect in several respects:

a) The orbit is in general elliptical.

b) The orbital inclination is, in general, not zero.

c) The longitude of the subsatellite point is, in general, not the desired longitude.

The capability of the payload propulsion to correct for these errors dictates the final accuracy requirements of the kick stage apogee burn. Subsection 2.3 of Volume II analyzes the relationship of trajectory errors to payload ΔV requirements.

The results of this analysis are a set of nonlinear expressions relating position and velocity errors at injection to the ΔV required to correct the errors.[†] If ΔV_A represents the available payload propulsion capability, then the performance requirement for this mission may be stated as

$$\Delta V_{Total}\,(95\%) < \Delta V_A$$

where ΔV_{Total} is the value of ΔV required for 95 percent probability of successfully performing the correction. Reasonable values for ΔV_A lie in the range 15-30 m/sec.

Results of a detailed performance analysis for this mission are presented in Section 7, Volume II, and are summarized in Subsection 4.1 following.

3.2.4 Guidance System Configuration and Operational Features

A functional block diagram of the recommended system configuration for the earth synchronous orbit mission is given in Figure 3-1. The core configuration for this mission consists of the three-axis strapdown inertial guidance subsystem with the three single-degree-of-freedom pulse-torqued gyros, three accelerometers body-mounted in an orthogonal triad, and a digital computer. This core configuration has been analyzed[††] for the minimum parking orbit case using the two different sets of strapdown inertial components (see Paragraph 2.3.1). Under the conditions of the criteria defined in Paragraph 2.3.2 of Volume II, this configuration gives marginal performance, requiring 35 to 73 m/sec ΔV capability from the payload for orbit error correction.

The primary error source for this poor performance is the effects of gyro bias drift acting over a period of approximately 5-1/2 hr and resulting in misapplication of the thrust direction of the apogee burn. (See Section 4.1, Table 4-III.)

[†] These expressions are given in Paragraph 2.3.2 of Volume II.

[††] See Subsection 7.3 of Volume II; and Subsection 4.1 of this volume.

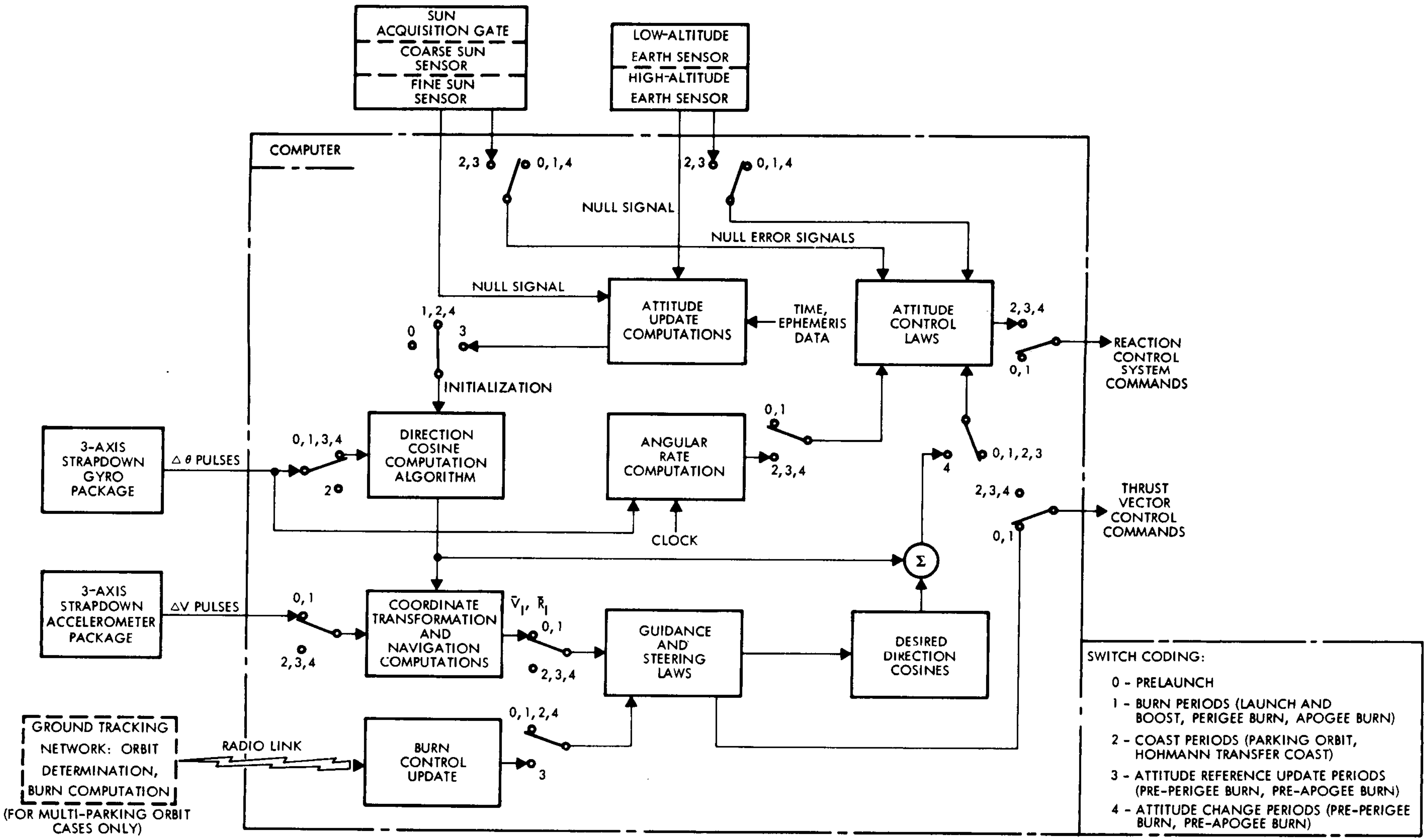


Figure 3-1. Guidance System Functional Block Diagram for Earth Synchronous Mission

The obvious solution to this problem is to provide a means for attitude updating a short time prior to apogee burn. Several alternative techniques for providing this function of attitude updating (or more precisely, of attitude referencing) during the long transfer coast period are discussed below:

a) Use an earth-horizon sensor to provide kick-stage attitude reference about two vehicle axes and "gyrocompassing" about the third axis. This scheme has the natural disadvantage that attitude control about the third axis is still imprecise because of the gyro-drift effects. This technique has not been investigated.

b) Use both an earth-horizon sensor and a sun sensor for complete three-axis attitude referencing. The sensors may be body-fixed or gimbaled relative to the body. In the first case, the kick-stage attitude will be fixed relative to the local vertical/sun line frame and will have to be altered prior to the apogee burn. In the second, the complexities of mechanical gimbaling and angular readout requirements are added and no distinct total system advantage is expected.

   In either case, it would be preferable if the launch time/mission profile were constrained so that the sun is at a zenith angle between 45 and 135 deg for a period of 5 to 15 min before apogee burn time. The zenith angle constraint is introduced for pointing accuracy considerations (note that for a zenith angle of 0 deg, the combination of sun and earth sightings does not give complete three-axis attitude information). The time constraint is introduced to minimize the time over which attitude must be remembered and maintained inertially.

   It should be noted that greater flexibility in the launch time/mission profile constraint can be obtained by replacing the sun sensor with a star tracker. For this equipment configuration, a suitable star can be selected prior to launch.

   One possible problem with this equipment configuration is that associated with the inflight identification and acquisition of the chosen star.

The performance of the system using earth and sun sightings to improve attitude information prior to apogee burn was investigated. The results indicate a six- to eight-fold improvement in performance over the core configuration (Runs 2, 4, and 6 in Table 4-III) at least for the missions in which transfer ignition occurs at the first equatorial crossing. For missions in which the kick stage remains in several parking orbits before transfer ignition, it is apparent that attitude updating would be required prior to the transfer ignition to minimize the effects of gyro drift during the long parking orbit coast periods. This attitude updating, or referencing, can be provided by the same earth horizon sensor/sun sensor combination. The launch time/mission profile constraint imposed by this technique is that the sun should be at a zenith angle of 45 to 90 deg for a period of 5 to 15 min prior to the transfer, or perigee, burn. The lower zenith limit is imposed for accuracy considerations and the 90 deg limit is imposed to minimize the effects of atmospheric refraction. The zenith angle constraint can be met for both perigee and apogee burns with proper selection of the launch time (see Volume II, Figure 3-3).

Even with attitude updating prior to both the perigee and the apogee burns, the system performance is poor for the multiparking orbit case (Runs 7, 9, and 11 in Table 4-III). This poor performance is attributable to parking orbit injection errors which ultimately appear as errors in transfer burn ignition time. The effects of this error source can be minimized by properly updating the ignition time.

The significant improvement (five- to eight-fold) in system performance obtainable with the addition of this updating procedure can be seen in Table 4-III (Runs 8, 10, and 12). Several techniques for ignition time update are possible:

a) Complete parking orbit determination via ground tracking.

b) Complete parking orbit determination via earth, sun, and star sensor readings.

c) A combination of a) and b).

d) Simple prediction of equatorial crossing time (and hence proper time for transfer burn ignition) via earth and sun sensor readings.

Parking orbit determination by ground tracking is a proven technique. However, there may be several shortcomings to this technique for this particular application. The primary disadvantage is that the kick stage may be required to remain at the parking altitude for fewer than two complete orbits, depending on mission objectives. In this situation, the accuracy of the orbit determination may be degraded to the point that no performance advantage is gained. (Section 6 in Volume II discusses the problems of tracking in low-altitude earth orbit.)

Parking orbit determination by a series of multiple celestial sightings has the same limitations as above. Further complications arise from the fact that the sun will be eclipsed for half of each parking orbit. Also, to ensure sufficient sun sightings while the sun is visible, the kick stage may have to be maneuvered continually during the daylight portion of the orbit to achieve favorable sun line-of-sight angles relative to the kick stage.

Technique d) is a "one-shot" open loop prediction technique, and even under the best of conditions is the least accurate of those listed though it may prove to be adequate. It is attractive from several standpoints: It does not require ground tracking and can be implemented with a body-fixed earth horizon tracker and a body-fixed sun aspect sensor. Detailed analyses of this technique have not been conducted as yet but are recommended as an area for further study. The basic concepts for this technique are outlined in Volume II, Paragraph 3.3.1.

3.3 MARS ORBITER MISSION

For this mission, it is assumed that the kick-stage guidance system† takes over after the Saturn V booster places the spacecraft†† into the interplanetary transfer trajectory. The guidance system then is required for attitude referencing and control during the long coast phases, applying the midcourse ΔV corrections (based on ground-tracking data) and final insertion of the payload into the desired orbit about Mars.

The specification of the guidance system accuracy requirements, and hence the system configuration, is tentative due to the fact that definitive propulsion capabilities data and information regarding the desired final orbit characteristics and accuracy requirements are not available. Present studies on the planned Mars Voyager missions indicate that a sophisticated strapdown inertial guidance system for performing the midcourse and orbit insertion maneuvers may not be required although such a system has much to offer in the way of flexibility and growth potential to meet more demanding future Voyager mission requirements.†††

For purposes of this study the use of a complete three-axis inertial reference unit is postulated. A comparison of the capabilities of this type of system with simpler systems is made in Section 8 of Volume II. Use

---

†Although the kick stage guidance system could conceivably be used to guide the Saturn V booster, that possibility is not considered in this study. It is assumed that ST-124 Inertial Guidance System located in the Saturn V Instrument Unit (IU) is used for the boost phase (launch through SIVB stage burnout including intermediate coast phases). A review of the performance capabilities of this guidance system show that it is completely satisfactory for this mission phase.

††The interplanetary spacecraft (Voyager concept) includes the necessary propulsion capabilities for the midcourse corrections, Mars orbit insertion, and orbital trim maneuvers. For this mission, the "kick stage" and the interplanetary spacecraft are synonomous. The "payload" consists of a Mars landing capsule deployed from the spacecraft after insertion plus the scientific instrumentation aboard the main spacecraft.

†††These requirements are currently undefined. However, it is reasonable to expect, based on past experience, that as the missions become more sophisticated the mission accuracy requirements will become more demanding. Some possible advanced Voyager missions (1975 through 1984) are described in Reference 20.

of an electro-optical approach guidance sensor to improve trajectory prediction accuracy near the target planet is also postulated although use of such a sensor appears to be necessary only if terminal mission accuracy requirements† are considerably more stringent than those imposed by the present Voyager mission.

3.3.1 Mission Characteristics

It is beyond the scope of this report to furnish effectively even a general comprehension of the complex mission analysis†† which must be performed before the selection of representative Mars trajectories for guidance system evaluation is possible. For the purposes of this study, trajectories available from previous studies conducted at TRW were used.

For the purpose of generating guidance system functional and performance requirements, the current Voyager mission requirements were used as the starting point. Mission requirements and constraints were established from "Performance and Design Requirements for the 1973 Voyager Mission" (Ref 3). The principal mission requirements and constraints that directly concern the launch vehicle performance capabilities and guidance system accuracy are briefly discussed in the following paragraphs.

- Ascent Mode: The parking orbit ascent mode will be utilized for the 1973 mission. Parking orbit coast times will vary between 10 and 90 min.
- Transfer Trajectory: Type I transfer trajectories will be used for the 1973 mission.
- Launch Energy: A maximum injection energy C_3 of 32.5 km^2/sec^2 will be assumed for the 1973 mission.[3]
- Launch Window: A nominal launch period of 45 to 60 days and a minimum daily firing window of 2 hr will be provided.

†The major source of error in the knowledge of the approach trajectory with respect to Mars is the uncertainty in the orbital position of Mars relative to the Earth (see Section 9, Volume II, for further discussion).

††Some of the more basic mission analysis concepts, particularly those relating to launch vehicle and guidance system requirements, are discussed in Ref 1 and 2.

- Declination of Launch Asymptote: The absolute value of the declination of the launch asymptote (DLA) will not exceed the following values: 5 deg $\leq |DLA| \leq$ 36 deg. Declination of the launch asymptote is dependent on both liftoff and coast time. The injection errors attributable to the guidance hardware are also a strong function of this coast time.

- $V_{\infty Mars}$: To achieve a reasonable vehicle mass fraction into Mars orbit, the approach $V_{\infty Mars}$ magnitude must not exceed a maximum of 3.25 km/sec.

- Injection Accuracy: The miss plus time-of-flight trajectory dispersons, due to random errors arising from launch vehicle injection errors, will be correctable with a 1-sigma midcourse trajectory correction velocity increment of 5 m/sec or less applied 2 days after injection.

- Mars Orbit Characteristics: The periapsis altitudes of the desired Mars satellite orbits lie between 500 and 1,500 km. The apoapsis altitudes lie between 10,000 and 20,000 km. These ranges apply both before and after orbit trim. No requirements have been set on either orbital inclination or the orientation of the line of apsides. Thus, the deboost velocity is assumed to be applied tangentially at the periares common to the hyperbolic approach trajectory and the elliptic orbit.

Since the launch phase guidance functions will be performed by the Saturn V guidance system, only the interplanetary trajectory is of interest here. The Earth-Mars trajectory chosen[†] has a launch date of 11 May 1971 and a trip time of 177 days. Characteristics of interest are shown in Figures 3-2, 3-3, and Table 3-I (from Ref 4). Guidance system performance requirements based on the Voyager mission requirements are discussed in the following section. As a guideline for determining final approach and orbit accuracy requirements, equations for parametric evaluation of velocity requirements are also presented in Volume II, Paragraph 2.4.3.

[†]Present Voyager planning calls for use of the Saturn V launch vehicle with the first mission scheduled for 1973. Trajectories for the 1973 launch opportunity were not available during this study. Since only post injection guidance is considered, the differences in boost phase trajectories are insignificant for the purposes of this study. The interplanetary trajectory used adheres to the mission constraints discussed above.

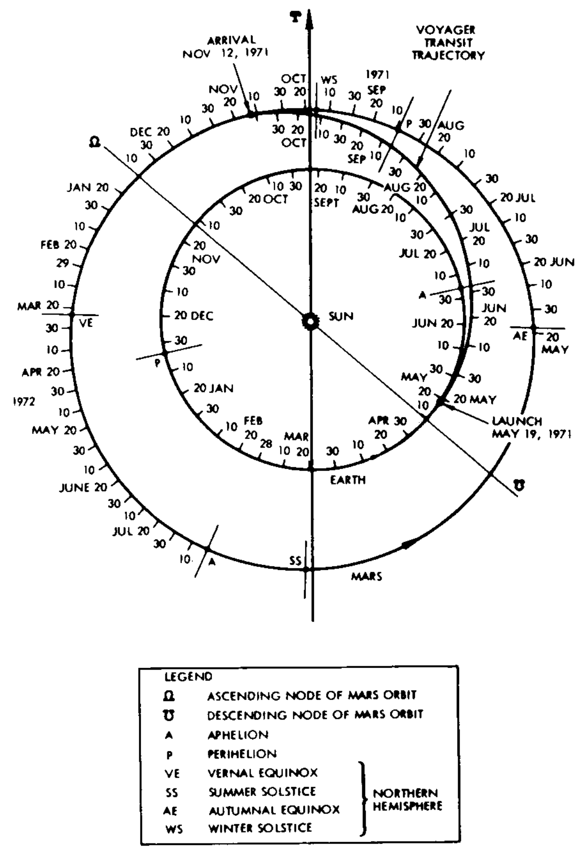


Figure 3-2. Voyager Sample Trajectory (from Reference 4)

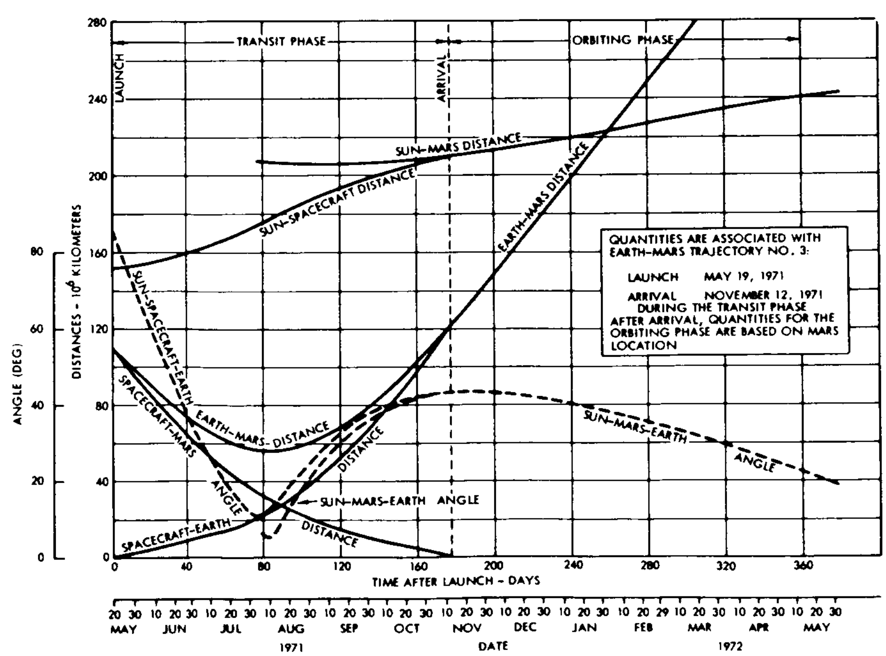


Figure 3-3. Geometrical Quantities Versus Time

Table 3-I. Characteristics of 1971 Earth-Mars Trajectory (from Reference 4)

Launch date	19 May 1971
Arrival date	12 November 1971
Time of flight	177 days
Departure asymptote (from Earth)	
V_∞	2.92 km/sec
C_3	8.53 km^2/sec^2
Angle to ecliptic	-16 deg
Angle to Sun-Earth line	88 deg
Approach asymptote (to Mars)	
V_∞	3.25 km/sec
Angle to plane of Mars' orbit	-3 deg
Angle to Mars-Sun line	119 deg
Interplanetary orbit	
True anomaly at arrival	142.5 deg
True anomaly at launch	4.5 deg
Heliocentric central angle	138 deg
Inclination to ecliptic	1.5 deg
Perihelion distance from Sun	151.2×10^6 km
Aphelion distance from Sun	220.5×10^6 km
Eccentricity	0.1853

3.3.2 Guidance System Operational Sequence

The operation sequences for the Mars orbiter mission are assumed to be as outlined below:

a) Launch, parking orbit, and injection into interplanetary trajectory: These phases are presumed to be under the control of the primary Saturn guidance system. The kick stage (spacecraft) guidance is in a semidormant state.

b) Separation from booster and first cruise phase: The kick stage strapdown inertial subsystem is used to provide rate damping signals to stabilize the separation-induced tumbling transients. After the rate stabilization is accomplished, a celestial reference acquisition (Sun and Canopus) sequence is initiated. The Sun and Canopus trackers will be body fixed and will serve as the primary long term inertial attitude references. After Sun/Canopus lock-on is achieved, the gyros may be turned off (except for heaters) until required for the midcourse reorientation maneuver.

Deep Space Network (DSIF) tracking will be used during this cruise phase for orbit determination and to compute the first midcourse velocity correction required to reduce the effects of injection errors. The midcourse thrust vector pointing and magnitude commands and time of execution will be transmitted to the onboard guidance system for execution.

c) Midcourse execution: If the gyros were shutdown in the previous cruise phase, wheel power would be applied sufficiently early to ensure proper gyro operation during the following sequence. Ten to thirty min (the time will be dependent on available spacecraft slew rates and maximum required turn-through angles) prior to the time of execution of the midcourse correction burn, vehicle rotations will be commanded to orient the thrust vector in the required inertial direction. The attitude change sequence will depend on the type of inertial configuration chosen. Three basic inertial configurations can be considered.

1a) A simple inertial configuration with a three-axis gyro package and one longitudinal axis integrating accelerometer.

1b) The above with the addition of two orthogonal lateral accelerometers (of the nonintegrating type).

2) A full strapdown inertial system, with a three-axis gyro package and a three-axis integrating accelerometer package, with full direction cosine computation capability.

For Configurations 1a and 1b, the orientation maneuver will be entirely open loop, using precomputed precision torquing signals applied sequentially to two out of the three gyros. For Configuration 2, closed-loop control over final attitude can be exercised and simultaneous biaxial attitude changes can be commanded.

When the proper attitude is achieved, and at the correct time, midcourse burn is initiated. Attitude control during burn is again dependent on the inertial configuration chosen:

1a) For Configuration 1a, attitude will be maintained under control of the three rate integrating gyros. Engine misalignments will be the largest source of burnout error in this case.

1b) For Configuration 1b, attitude will again be maintained under control of the three rate integrating gyros with the important modification that signals processed from the nonintegrating accelerometers can be used to provide attitude correction signals to compensate largely for engine misalignments.

2) For Configuration 2, full three-axis ΔV control capability is provided and engine misalignments become unimportant. Further discussions will be limited to this configuration.

d) Subsequent cruise and midcourse correction phases: After completion of the first midcourse correction, the spacecraft will be "unwound" to the original Sun/Canopus reference attitude and continue in a cruise phase identical to the first. One or more further midcourse corrections will be made in a manner similar to the first. After the last midcourse correction, the trajectory will have been corrected such that terminal approach conditions meet the requirements (see Paragraph 3.3.2).

e) Approach phase: On the premise that a mission with terminal approach requirements is more stringent than those imposed on Mariner 1969 or Voyager, it is postulated that some form of approach navigation will be required that is more accurate than that available with DSIF tracking. To cover this possibility, this study includes considerations of an approach sensor. The sensor to be considered will be a planet tracker with 2 deg of electronic scan freedom relative to the kick stage. The sensor can provide:

1) Stadimetric ranging data

2) Clock and cone angles relative to the Sun/Canopus frame of reference. (See Figure 4-3.)

On the basis of approach measurements provided either via DSIF alone or in combination with the approach sensor, the approach trajectory will be determined and the following will be computed:

1) Magnitude and inertial direction of the deboost velocity to achieve the desired orbit about Mars

2) Time of initiation of the deboost thrust

DSIF orbit determination accuracy requirements are discussed in Paragraph 2.2.4.

f) Deboost velocity application: The sequence of events and operations of this phase is the same as for the midcourse correction phase.

3.3.3 Guidance System Performance Requirements

3.3.3.1 Terminal Accuracy Requirements

Because of (1) midcourse errors, (2) imperfect approach trajectory estimation, and (3) execution errors at deboost burn, the final orbit will differ from the desired orbit. The Voyager mission requirements[†] shown in Table 3-II have been translated into system accuracy requirements for these three errors and are summarized in the following paragraphs.

3.3.3.2 Orbit Correction Requirements

The required precision with which the spacecraft must execute the arrival date separation maneuver[††] and interplanetary trajectory corrections is a complicated function involving a number of considerations such

[†] The overall mission accuracy requirements have not been firmly established for the Voyager mission. The requirements stated in Table 3-II are based on preliminary requirements for the early missions.

[††] This maneuver is intended to separate the arrival time of the two spacecraft launched by the Saturn V booster by at least 8 days. The maneuver is made in a similar manner, but prior to the first midcourse correction.

Table 3-II. Voyager Mission Terminal Accuracy Requirements

Parameter	Allowable Dispersion (3σ)
Impact parameter $\overline{B}$	500 km
Time of encounter	3 min
Periapsis altitude of initial orbit (prior to orbit trim maneuver)	30%
Orbital inclination of initial orbit	5 deg

as the final mission accuracy requirements, the orbit determination uncertainty as a function of time, the number of maneuvers to be performed, the amount of trajectory biasing necessary to satisfy the probability-of-impact constraint, the orbit trim philosophy, and several other considerations, all of which are vitally interwoven in the mission formulation. In order to achieve a rational balance among the various mission accuracy requirements, Table 3-III from Ref 3 specified the maximum allowable maneuver errors during different phases of the flight. The maximum allowable maneuver errors are defined by an error ellipsoid with an axis of symmetry parallel to the specified velocity increment.

3.3.3.3 Orbit Determination Accuracy Requirements

To meet the overall mission accuracy requirements given above, specification must also be placed on the orbit determination accuracy using the earth based DSIF. Due to the trajectory geometry, tracking system characteristics, and the presence of trajectory disturbances, the orbit determination uncertainties vary throughout the mission. The allowable uncertainties (from Ref 3) are shown in Table 3-IV.

Table 3-III. Maximum Velocity Error Ellipsoids (from Reference 3)

Maneuver	Minimum Velocity Increments† (m/sec)	Maximum Velocity Increments (m/sec)	3σ Error component parallel to specified velocity increment (m/sec or % of specified velocity increment)	3σ Error component normal to specified velocity increment along any two orthogonal axes (m/sec or % of specified velocity increment)
Interplanetary trajectory corrections	1.0 (0.3)	100 (200)	larger of 0.1 m/sec or 3.0% (larger of 0.03 m/sec or 2.0%)	larger of 0.1 m/sec or 3.0% (larger of 0.03 m/sec or 2.0%)
Mars orbit insertion	1.0 km/sec	2.0 km/sec	3.0% (1.5%)	5.0% (3.0%)
Mars orbit trim	5.0 (1.5)	150	larger of 0.5 m/sec or 5.0% (larger of 0.2 m/sec or 3.0%)	larger of 0.5 m/sec or 5.0% (larger of 0.2 m/sec or 3.0%)

Note: Numbers not in parentheses are maximum values.

Numbers in parentheses are design goals.

†For purpose of error calculations only.

Table 3-IV. Maximum Allowable 3σ Orbit Determination Uncertainties (from Ref 3)

Time at which orbit estimate is calculated	Magnitude and time of prior orbit corruptions†	Uncertainty in magnitude of impact parameter vector (km)	Uncertainty in aiming plane normal to nominal impact parameter vector (km)	Uncertainty in time of encounter (min)
Injection +2 days	Planetary vehicle injection	2000 (1000)	2000 (1000)	15 (7)
Injection +30 days	150 m/sec arrival date adjustment and interplanetary trajectory correction at I +5 days	1000 (500)	1500 (750)	10 (5)
Encounter -30 days	5 m/sec interplanetary trajectory correction at I +30 days	500 (300)	750 (500)	4 (3)
Encounter - 2 days	1 m/sec interplanetary trajectory correction at E -30 days	400 (150)	500 (200)	3 (1)
Encounter - 4 hr	Same as above	300 (100)	500 (150)	2 (0.5)

†For purposes of error analysis only. Numbers not in parentheses are maximum allowable uncertainties. Numbers in parentheses are design goals.

Table 3-IV. Maximum Allowable 3σ Orbit Determination Uncertainties (Continued) (from Ref 3)

Time at which orbit estimate is calculated	Magnitude and time of prior orbit corruptions†	Uncertainty in orbit semi-major axis (km)	Uncertainty in orbit eccentricity	Uncertainty in time of periapsis passage (sec)
Orbit insertion +4 returning to orbit periapsis	2.2 km/sec orbit insertion maneuver	10 (1)	10^{-4} (10^{-5})	5 (0.1)
Orbit trim +3 returns to orbit periapsis	Orbit trim maneuver of 100 m/sec	10 (1)	10^{-4} (10^{-5})	5 (0.1)

†For purposes of error analysis only. Numbers not in parentheses are maximum allowable uncertainties. Numbers in parentheses are design goals.

3.3.4 Guidance System Configuration and Operational Features

The onboard core guidance configuration includes the three-axis strapdown inertial guidance subsystem, a body-fixed sun sensor assembly, a body-fixed Canopus tracker assembly, and a digital computer (Refer to Figure 3-4 for the matrix of subsystem elements). Midcourse and orbit insertion $\Delta\overline{V}$ commands will be generated on the basis of DSIF tracking data and executed under control of the strapdown inertial system. For improved approach guidance and Mars orbit insertion, a planetary approach sensor can be added. The functional block diagram for this configuration appears in Figure 3-5.

In this configuration, the primary attitude reference is established by the spacecraft-Sun line and the spacecraft-Canopus line. The Sun/Canopus acquisition sequence will be completely automatic. The acquisition sequence is functionally the same as that used on past lunar and planetary space shots:

a) Null error signals first from the course and then the fine sun sensor will be used to control the appropriate reaction control jet thrusters until the sun sensor optical axis is aligned to the sun.

b) Programmed body rate signals are then used to rotate the spacecraft about the just established sun line to locate Canopus. (The Canopus tracker (1 x 16 deg instantaneous field of view) is mounted such that 1) its long view dimension and the sun sensor optical axis are coplanar and 2) its chosen null axis is such that Canopus will be in the field of view throughout the interplanetary trajectory).

   Canopus discrimination can be accomplished by utilizing both minimum and maximum signal threshold detection, although spectral class discrimination could increase assurance of correct acquisition.

c) When the Canopus image is nulled in the narrow view dimension, complete three axis inertial attitude reference for the spacecraft will have been established. Initial values for the direction cosines relating spacecraft orientation to a reference inertial coordinate frame are thus established.

G AND C MODES OF OPERATION / EQUIPMENT USAGE	INJECTION			CRUISE AND MIDCOURSE				
	LAUNCH THROUGH PARKING ORBIT INJECTION	PARKING ORBIT	INTERPLANETARY ORBIT INJECTION BURN	CRUISE			MIDCOURSE COR	
							REORIENT	BURN
	LAUNCH GUIDANCE	INERTIAL ATTITUDE HOLD RE-ORIENTATION	POWERED FLIGHT GUIDANCE	SUN ACQ	STAR ACQ	CRUISE ATTITUDE CONTROL	REORIENT-ATION MANEUVER	POWERED PHASE GUIDAN
MAJOR FUNCTIONAL REQUIREMENTS/ CONSTRAINTS	GUIDANCE AND NAVIGATION THROUGH BOOST PHASE ACCOMPLISHED BY SATURN V GUIDANCE	MAX-PARKING ORBIT 90 MIN	INJECTION ACCURACY REQT 5 M/SEC (1σ) (MID-COURSE ΔV)	ACQUIRE SUN AND CANOPUS 3-AXIS ATTITUDE CONTROL			M/C CORRECTION BY ON BOARD S) S DSIF/DSN NAVIGA CORRECTION ACC 100 KM, 1 MIN (R̄, 2 CORRECTIONS A 30 DAYS AFTER INJ ΔV AVAILABLE 100	
STRAPDOWN SENSOR PACKAGE	▲	▲	▲	▲	▲	▲	▲	▲
DIGITAL COMPUTER	▲	▲	▲	▲	▲	▲	▲	▲
BOOSTER THRUST VECTOR CONTROL (TVC)	▲		▲					
BOOSTER ATTITUDE CONTROL (RCS)		▲						
SPACECRAFT TVC								▲
SPACECRAFT RCS HIGH T LOW T				▲	▲	▲	(TWO LEVEL SYST ▲	▲
SUN SENSOR(S) COARSE FINE				▲		▲		
STAR TRACKER (CANOPUS)					▲	▲		
PLANET APPROACH TRACKER								
USBS TRACKING	▲	▲	▲					
DSIF/DSN TRACKING AND ORBIT PREDICTION						▲		
EARTH-SPACECRAFT COMMAND/DATA LINK					▲	▲	▲	▲

				APPROACH AND ORBITAL OPNS								
CTION(S)		CRUISE	MARS APPROACH	MARS ORBIT INJECTION		ORBITAL ATTITUDE CONTROL		ORBIT TRIM MANEUVER	ORBIT ATTITUDE CONTROL	CAPSULE DEPLOY		ORBITAL OPNS
	REORIENT			REORIENT	BURN	REORIENT	HOLD			REORIENT	DEPLOY	
E	REACQ SUN AND CANOPUS	CRUISE ATTITUDE CONTROL										
XECUTED EM USING ION RACY (3σ) PLANE) 5 DAYS AND CTION /SEC		3-AXIS ATTITUDE CONTROL	• APPROACH ORBIT DETERMINATION USING DSIF, DSN 500 KM (3σ) 3 MIN ($\bar{R}$, $\bar{T}$ PLANE) • ON-BOARD APPROACH SENSOR (OPTIONAL) (MEASURE SUN/MARS CANOPUS, MARS ∠'S)	NORMAL ORBIT 500-1500 KM PERIAPSIS 10,000-20,000 KM APOAPSIS V_∞ < 3.25 KM/SEC TOLERANCES (3σ) ±30% PERIAPSIS ALT ±5 DEG INCLINATION		3-AXIS ATTITUDE CONTROL		ΔV = 100 M/SEC	3-AXIS ATTITUDE CONTROL			ANTENNA POINTING FOR DATA RELAY SCIENTIFIC EXPERIMENTS
	▲	▲	▲	▲	▲	▲	▲	▲	▲	▲	▲	▲
	▲	▲	▲	▲	▲	▲	▲	▲	▲	▲	▲	▲
					▲			▲				
A ASSUMED)	▲			▲	▲	▲		▲		▲	▲	
		▲	▲				▲		▲			▲
	▲	▲	▲				▲		▲			▲
	▲	▲	▲				▲		▲			▲
			(START 5 HRS PRIOR TO ENCOUNTER) ▲									
		▲	▲				▲		▲			▲
	▲	▲	▲	▲	▲	▲	▲	▲	▲	▲	▲	▲

Figure 3-4. Mars Orbiter Mission-System Elements by Mission Phase

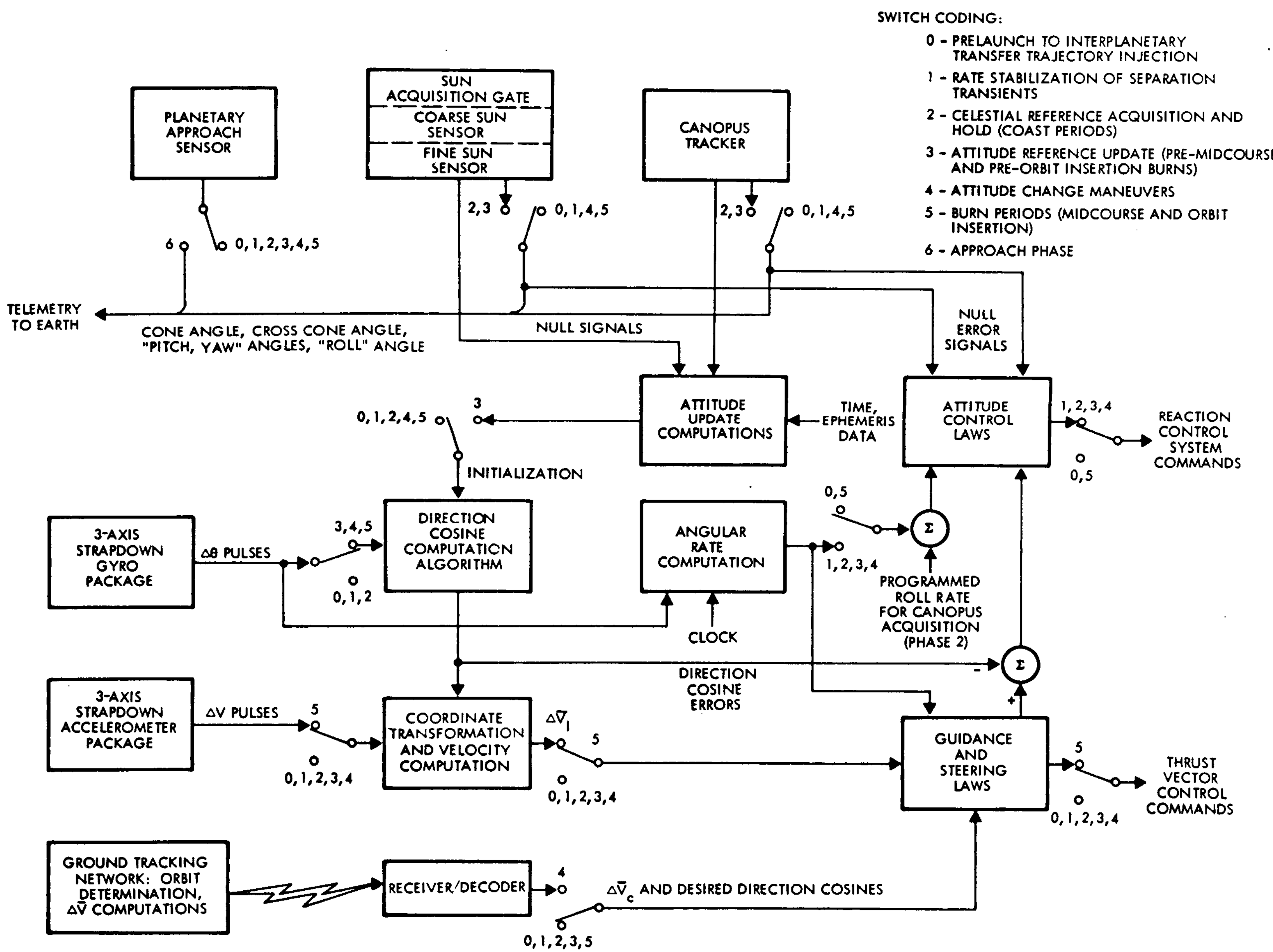


Figure 3-5. Guidance System Functional Block Diagram for Mars Orbiter Mission

The desired spacecraft attitude for all periods of thrust application (i.e., for the midcourse corrections and for the Mars orbit injection) can be expressed in terms of desired direction cosine values. The attitude change maneuvers required to achieve the proper thrusting directions can be controlled by differencing the proper three elements of the desired and computed direction cosine matrices. The burn time will be controlled on the basis of the desired ΔV magnitudes and outputs of the accelerometers, which can be turned on only for the thrusting periods.

The body-fixed approach sensor will be designed to have a 15 x 15 deg total field-of-view and will have its null axis prealigned such that the target planet will be in its field-of-view from 6-1/2 days to 1 day before encounter. The geometrical relationships between the three electro-optical sensors and their field-of-view are illustrated in Figure 3-6.

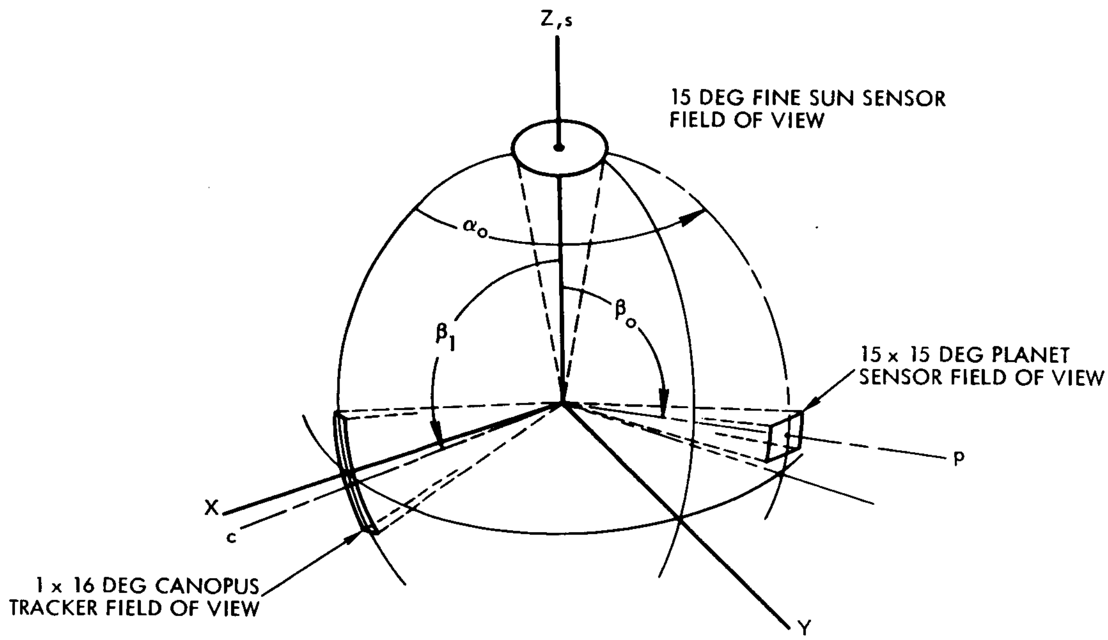
The approach sensor is sensitive in the visible light spectrum and, as a consequence, the planet image may be gibbous or crescent depending on approach conditions. Several algorithms suitable for determining apparent planet diameter and centroid location within the total field-of-view have been tested. One algorithm uses a simple three-point fix scheme and requires several complete scans of the image to minimize the effects of instrument errors. The second uses all avaliable data points obtained in one scan.

3.4 LUNAR ORBITER MISSION

The booster/payload combination for this mission is assumed to be the Atlas SLV3X/Centaur/HEUS/lunar orbiter. Based on past experience, e.g., Surveyor and Lunar Orbiter, the need for a sophisticated inertial guidance system on the spacecraft is questionable for the translunar and lunar operations phases of this mission. However, for this study, it is assumed that the kick stage guidance system is to be used not only for these phases but also for primary guidance and control of the lower booster stages. The operational sequences and functional requirements are summarized below.

3.4.1 Mission Characteristics

This mission is operationally very similar to the Mars Orbiter Mission discussed in Subsection 3.3. The primary difference to be noted is that USBS/DSIF tracking and orbit estimation accuracy will probably be sufficient to obviate the need for an approach sensor.



X, Y, Z - VEHICLE FIXED REFERENCE COORDINATE SET

s - SUN SENSOR OPTICAL AXIS

c - CANOPUS TRACKER OPTICAL AXIS

p - PLANET SENSOR OPTICAL AXIS

α_o, β_o - AVERAGE CLOCK AND CONE ANGLES RESPECTIVELY FOR GIVEN TRAJECTORY

β_1 - AVERAGE SUN - CANOPUS CONE ANGLE FOR GIVEN TRAJECTORY

Figure 3-6. Electro-Optical Sensor Field-of-View Geometry

A parking orbit ascent trajectory with a coast time of approximately 14 min was selected for this study. The rationale for this selection was based on the fact that the largest figure-of-merit is obtained for parking orbit missions having coast times in this range.

The lunar mission reference trajectory used for error analysis purposes was a closed loop targeted trajectory for the Atlas Centaur (AC-12 Configuration) launch vehicle. The trajectory profile is shaped by a predetermined pitch steering program from launch to booster cutoff (BECO). After BECO the sustainer is ignited and closed loop guidance is initiated. The guidance system continues to steer the vehicle through sustainer cutoff (SECO) and Centaur first-burn ignition until parking orbit is reached. The first-burn duration (launch to parking orbit injection) is approximately 585 sec and injects the vehicle into a 167 km perigee, 173 km apogee orbit. The Centaur stage coasts in this orbit for 845 sec, whereupon it reignites and burns for another 106 sec, injecting the payload into a highly elliptical ($e = 0.97167$) transfer orbit. The transfer time is approximately 65 hours.

Two midcourse corrections are assumed for this mission, the first at 15 to 20 hr after injection and the second a few hours prior to lunar intercept.

Deboost is made into an intermediate orbit with approximate apsis distances of 3590 and 1990 km. The deboost velocity increment required is 745 m/sec. After accurate determination of the orbit has been made, a final orbit adjust maneuver is made to place the vehicle into a 3589 by 1784 km orbit.

3.4.2 Guidance System Operational Sequence

The guidance system operational sequence for the various phases of the lunar orbiter mission is described below:

a) Launch and boost to ~ 167 km parking orbit: The kick stage strapdown inertial guidance subsystem will provide the guidance function for this phase.

b) Coast in parking orbit: The kick stage and payload will coast in the parking orbit until translunar injection, which occurs approximately 14 min after entering the parking orbit. The inertial guidance subsystem will be relegated to the role of an attitude reference during this phase.

c) Translunar injection: The kick stage will be ignited to inject the kick stage/payload into the translunar trajectory. Attitude and burn control will be provided by the strapdown inertial guidance subsystem.

d) Coast until first midcourse correction: Following the injection burn, a celestial reference acquisition sequence is initiated and the kick stage/payload will be attitude fixed to the sun and the star Canopus via body-fixed sun and star sensors. The strapdown accelerometers can be turned off (except for heaters), and the flight computer algorithm for updating the direction cosines can be placed in a standby mode.

Deep-Space Network (DSIF) tracking will be used during this coast phase for orbit determination and to compute the midcourse velocity correction required to reduce the effects of injection errors. The midcourse thrust vector pointing and magnitude commands and time of execution command will be transmitted to the kick stage system.

e) First midcourse correction: Approximately 15 to 20 hr from translunar injection, the first midcourse correction will be executed. Ten to 30 min prior to the time of execution, the accelerometers will be turned on, the direction cosine solution algorithm will be initialized, and the vehicle rotations will be commanded to orient the thrust vector in the required inertial direction. When the proper attitude is achieved, and at the correct time, the midcourse burn is initiated.

f) Second coast phase and second midcourse correction: After completion of the first midcourse correction, the kick stage/payload will be "unwound" to the original Sun/Canopus reference attitude and continue in a cruise phase identical to the first. The second midcourse burn will occur a few hours prior to lunar injection and is designed to null selected miss components at lunar intercept.

g) Coast until deboost maneuver into intermediate lunar orbit: This phase will be identical to the other coast phases.

h) Deboost into intermediate lunar orbit: Based on the tracking data obtained, the kick stage/payload will be injected into an intermediate orbit with approximate apsis distances of 3590 and 1990 km. The deboost velocity increment required is approximately 745 m/sec.

i) Coast in intermediate orbit: The amount of coast time in the intermediate orbit will be chosen such that the orbit is properly phased with respect to the preselected target. The kick stage/payload will be tracked by DSIF stations to determine orbital parameters and the retro-maneuver required to place the kick stage/payload into the final orbit.

j) Retro into final orbit: Based upon the orbital estimates obtained from DSIF tracking data, and controlled by the strapdown inertial guidance system, the spacecraft will be injected into the final orbit. The desired final orbit will nominally have an apocynthion and pericynthion of 3589 km and 1784 km, respectively.

3.4.3 Guidance System Performance Requirements

3.4.3.1 Translunar Injection

The kick stage/payload must be injected into a translunar trajectory such that the desired lunar orbit can be achieved by the kick stage propulsive capability. A set of deviations† of the kick stage/payload position and velocity from the nominal trajectory which will permit the meeting of the requirements of the final orbit is listed in Table 3-IX.

3.4.3.2 Translunar Coast Phases

Prior to the first (second) midcourse corrections, the deviations of position and velocity from the nominal trajectory must be within certain limits. These limits are determined by the correction capability of the midcourse correction system. A set of injection deviations from the nominal trajectory propagated to the point of the first midcourse correction which satsify the midcourse correction capability are listed in Table 3-V. Prior to the second midcourse maneuver, the deviations must be such that the correction of miss components at the target are within the capability of the second midcourse maneuver. A set which satisfies these requirements is shown in Table 3-V.

---

†The position and velocity errors are stated either in an earth-centered inertial (ECI) coordinate system (X-axis in the direction of the vernal equinox) or in selenographic coordinates. Note that these errors are stated as deviations from the a priori nominal trajectory. See Subsection 1.4.5 for definition of coordinate Systems.

Table 3-V. Guidance Requirements for Translunar Injection and Translunar Coast Phases (1σ Values)

Parameter	Translunar Injection	Coast until First Midcourse Correction	Coast until Second Midcourse Correction	Coast until Deboost into Intermediate Lunar Orbit
Δx	14.9 km	685 km	160 km	81 km
Δy	18.4 km	996 km	358 km	47 km
Δz	19.5 km	680 km	262 km	16.7 km
$\Delta \dot{x}$	21.4 m/sec	18.8 m/sec	2.1 m/sec	15.3 m/sec
$\Delta \dot{y}$	19.5 m/sec	10.8 m/sec	4.6 m/sec	30 m/sec
$\Delta \dot{z}$	32.4 m/sec	12.0 m/sec	3.0 m/sec	14.1 m/sec
Coordinate System	ECI	ECI	ECI	Selenographic

During each phase, the position and velocity of the kick stage/payload will be determined by earth-based tracking stations. At the end of the final coast phase, as a result of the midcourse corrections, the position and velocity of the kick stage/payload must be within the limits shown in the last column of Table 3-V.

3.4.3.3 Midcourse Correction Maneuvers

Approximately 15 hr after translunar injection, the first midcourse correction will be commanded. The requirements on the maneuver execution errors are shown in Table 3-VI. The guidance law assumed is directed to nulling the errors in the impact plane and error in the time of flight or the impact plane error only.† Hence, these controlled quantities will be reduced by the midcourse maneuver.

The second midcourse maneuver will be executed a few hours prior to translunar injection. The requirements on the maneuver execution errors are in Table 3-VI.

† The performance analysis results are presented in Paragraph 4.1.2 for both guidance laws. Detailed mission payload requirements dictate the choice for a given mission.

Table 3-VI. Guidance and Control Requirements for First and Second Midcourse Corrections

Parameter	First Midcourse Correction	Second Midcourse Correction
Pointing error	0.4 deg (1σ)	0.4 deg (1σ)
Error proportional to ΔV	0.04% (1σ)	0.04% (1σ)
Velocity cutoff Resolution error	0.02 m/sec (1σ)	0.02 m/sec (1σ)
Velocity increment required (not to be exceeded more than 1% of the time)	64 m/sec	3 m/sec

3.4.3.4 Deboost into Lunar Orbit

Based upon tracking data, the following quantities will be determined for injection into the intermediate orbit:

a) Thrust initiation time
b) Body attitude
c) Velocity increment

These quantities will be computed to null the deviations from nominal of the apocynthion, inclination, longitude of the ascending mode, and the argument of pericynthion. A set of required accuracies of position and velocity at the end of this phase which will meet the orbital requirements is given in Table 3-VII.

3.4.3.5 Coast in Intermediate Orbit and Final Orbit Insertion

There are no active guidance requirements during the intermediate orbiting phase. However, the position and velocity must be within certain limits at the end of this phase. A set of position and velocity accuracies which (in combination with the expected execution errors) will not violate the orbit accuracies required is indicated in Table 3-VIII

Table 3-VII. Guidance and Control Requirements for Deboost Into Intermediate and Final Lunar Orbit

Parameter	Deboost Into Intermediate Orbit (1σ values)†	Deboost Into Final Orbit (1σ values)†
Δx	80.4 km	14.2 km
Δy	24.2 km	18.9 km
Δz	9.7 km	13 km
$\Delta \dot{x}$	2.7 m/sec	13 m/sec
$\Delta \dot{y}$	47 m/sec	3.3 m/sec
$\Delta \dot{z}$	11.5 m/sec	0.7 m/sec
Pointing error	0.4 deg	0.4 deg
Error proportional to ΔV	0.04%	0.04%
Velocity cutoff	0.02 m/sec	0.02 m/sec
Velocity increment required (not be exceeded more than 1% of the time)	758 m/sec (2485 ft/sec)	33 m/sec (110 ft/sec)
Coordinate System	Selenographic	Selenographic

†All values are 1σ except the required velocity increment.

Table 3-VIII. Guidance Requirements for Coast in Intermediate Orbit

Parameter	Specification (1σ Values)
Δx	14.4 km
Δy	19 km
Δz	13 km
$\Delta \dot{x}$	0.6 m/sec
$\Delta \dot{y}$	3.6 m/sec
$\Delta \dot{z}$	1.5 m/sec
Coordinate System	Selenographic

The required maneuver for final adjustment of the orbit will be calculated using previous estimates of position and velocity. The maneuver (pitch attitude, yaw attitude, velocity magnitude) will be calculated so as to null the deviations at the pericynthion after retrothrusting at apocynthion to give a specified pericynthion inclination and argument of pericynthion.

At the completion of the maneuver, the position and velocity must be within prescribed limits so that the desired lunar orbit can be achieved. The final lunar orbital requirements are given in Table 3-IX.

Table 3-IX. Lunar Orbital Phase Accuracy Requirements

Parameter	Specification (1σ Values)
Error in semimajor axis	7.24 km
Error in pericynthion altitude	0.2 km
Inclination error	0.01 deg
Error in ascending node at first target pass	
• Selenographic latitude	0.1 deg
• Selenographic longitude	0.1 deg
Error in argument of periapsis at first target pass	0.01 deg

3.4.4 Guidance System Configuration and Operational Features

Inasmuch as it is anticipated that the guidance system for the Lunar orbiter mission is to be used to control the launch and ascent phases as well as the translunar and lunar phases, the complete three-axis strap-down inertial guidance subsystem will be required. The system block diagram and operational features are similar to that for the Mars orbiter mission (less the approach sensor) and will not be repeated.

3.5 SOLAR PROBE WITH JUPITER ASSIST

Prior to planetary encounter, this mission is similar to the Jupiter flyby mission studied in Reference 5. For the solar probe mission, the spacecraft trajectory passes in close proximity to Jupiter (a few radii is typical). The trajectory is altered by the gravitational field of Jupiter so that after encounter the spacecraft is in a heliocentric trajectory passing close to the Sun (not necessarily in the ecliptic plane). The flyby mission is similar in that the encounter is made again at a few radii with the purpose of making scientific measurements during the encounter phase.

The Atlas SLV3X/Centaur/HEUS can be considered as a typical booster for this mission. The kick stage (HEUS) guidance system is assumed to provide the primary guidance and control functions for all booster stages and until completion of the final midcourse correction maneuver. After the midcourse correction is completed, the system will be used only to control vehicle attitude. No further trajectory corrections are required.

3.5.1 Mission Characteristics

3.5.1.1 Post Encounter Trajectories

It has been indicated (Reference 6) that spacecraft passing close to the planet Jupiter can make use of the gravitational energy added to or subtracted from the orbital energy (expressed in heliocentric coordinates) to achieve a number of missions subsequent to encounter. Among these missions are those leading to the far reaches of the solar system (exploration of the outer planets and interplanetary space beyond Jupiter, even to the extent of achieving trajectories which escape from the solar system), those in which heliocentric energy is decreased and the spacecraft returns to the Earth or even substantially closer to the Sun, and those employing orbits highly inclined to the ecliptic. Not all of these options are possible from the Earth-Jupiter trajectories associated with the relatively low launch energies and 20- or 30-day launch periods. Some of the options which are available include trajectories which lead to trans-Jupiter regions of the solar system (achievable by eastward equatorial passages), trajectories which return closer to the Sun (perihelion distances 4 to 0.6 AU, achievable by westward equatorial passages), and 20- to 40-deg inclinations of subsequent heliocentric orbits to the ecliptic (achievable by polar passages). Some of the general characteristics of these post encounter trajectories are tabulated in Table 3-X.

3.5.1.2 Preencounter Trajectory

The particular preencounter trajectory selected for this study is a sample from the 1972 Jupiter launch opportunity. The general nature of the trajectory is illustrated in Figure 3-7. Some significant mission parameters for the particular trajectory selected are given in Table 3-XI.

3.5.1.3 Encounter Geometry

The geometrical characteristics of the spacecraft trajectory in the near vicinity of Jupiter are determined by 1) the large gravitational influence of the planet, 2) the choice of interplanetary trajectory, and 3) the choice of the input parameter $\overline{B}$. While the first influence is not subject to control, the other two are.

For a given interplanetary trajectory, the choice of the impact parameter vector $\overline{B}$ specifies in which direction from Jupiter and at what distance the approach asymptote lies. $\overline{B}$ is commonly expressed in components $\overline{B} \cdot \overline{R}$ and $\overline{B} \cdot \overline{T}$, where $\overline{R}$, $\overline{S}$, $\overline{T}$ are a right-hand set of mutually orthogonal unit vectors aligned as follows: $\overline{S}$ is parallel to the planetocentric approach asymptote, $\overline{T}$ is parallel to the plane of the ecliptic and positive eastward, and R completes the set and has a positive southerly component. The magnitude of $\overline{B}$ determines the distance of closest approach to Jupiter, and the angle

$$\theta = \tan^{-1} \frac{\overline{B} \cdot \overline{R}}{\overline{B} \cdot \overline{T}}$$

Table 3-X. General Characteristics of Trajectory Subsequent to Jupiter Encounter (from Ref 5)

Launch Energy Required (C_3) (km^2/sec^2)	Flight Time (days)	V_{HP} ‡ (km/sec)	ZAP‡ (deg)	Characteristics of Heliocentric Trajectory after Jupiter Encounter†		
				Eastward (Equatorial)	Westward (Equatorial)	Polar
100 (Type I)	500	11	150	In ecliptic, approximately tangential to Jupiter's orbit, velocity 1. 3 times solar escape velocity	In ecliptic, returning inward to perihelion as close as 0. 6 AU	Inclined up to 43 deg to ecliptic
82 (minimum, Type I)	750	7	120	In ecliptic, tangential or outward from Jupiter's orbit, velocity 1. 1 times solar escape velocity	In ecliptic, inward to perihelion 0. 9 to 4 AU depending on closeness of approach to Jupiter	Inclined up to 31 deg to ecliptic
95 (Type I or II)	950	6	90	In ecliptic, outward from tangential, velocity up to 0. 95 times solar escape velocity	In ecliptic, inward from tangential, velocity up to 0. 95 times solar escape	Inclined up to 27 deg to ecliptic
90 (Type II)	1250	7	60	In ecliptic, outward from tangential, velocity up to 0. 9 times solar escape velocity	In ecliptic, tangential or inward from Jupiter's orbit, velocity up to 1. 05 times solar escape velocity	Inclined up to 31 deg to ecliptic

†All heliocentric orbits subsequent to encounter are posigrade.

‡See glossary of terms, Paragraph 1.4.

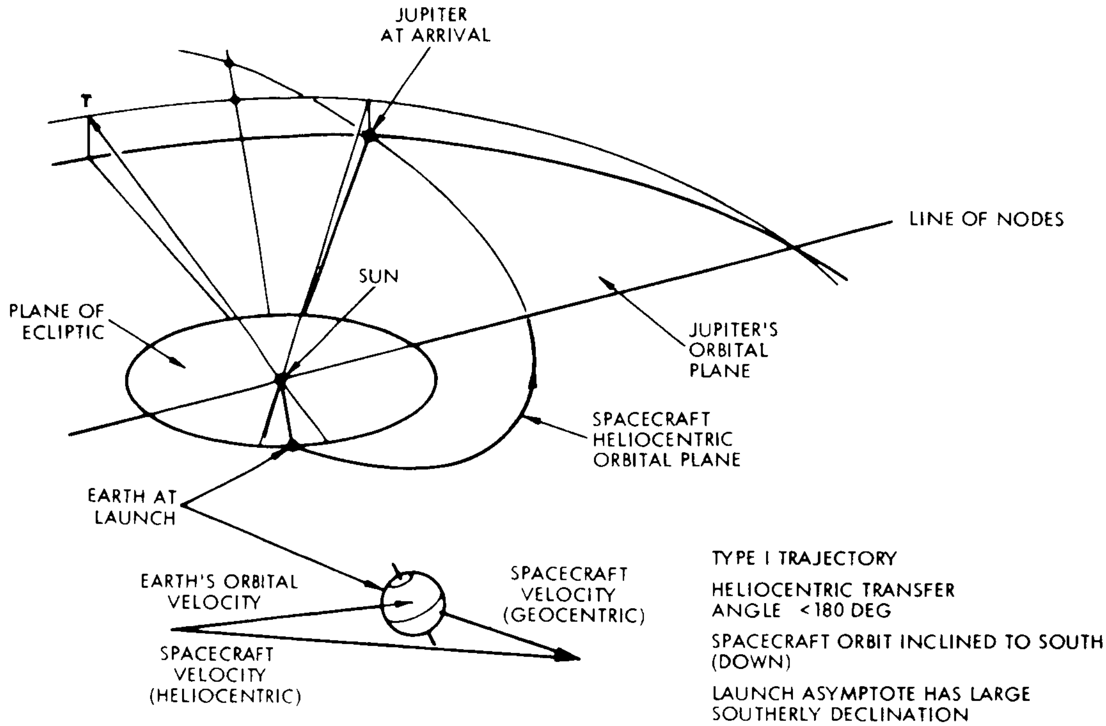


Figure 3-7. Typical Type I Earth-Jupiter Trajectory

Table 3-XI. Characteristics of 1972 Earth-Jupiter Trajectory (from Ref 5)

Launch date	14 March 1972
†C_3, km^2/sec^2	86.24
Arrival date	26 March 1974
Flight time, days	742
Heliocentric transfer angle, deg	156.63
†DLA, deg	-30.2
†V_{HP}, km/sec	7.00
†ZAP, deg	122
†ZAE, deg	116
†ZAL, deg	70
Jupiter-Earth distance at encounter, AU	5.83
Inclination of spacecraft orbit plane to ecliptic, deg	2.30

†See glossary of terms, Section 1.4, for definition of symbols.

specifies the orientation of the Jupiter-centered orbit plane as a rotation about the $\overline{S}$ axis. These definitions are illustrated in Figure 3-8.

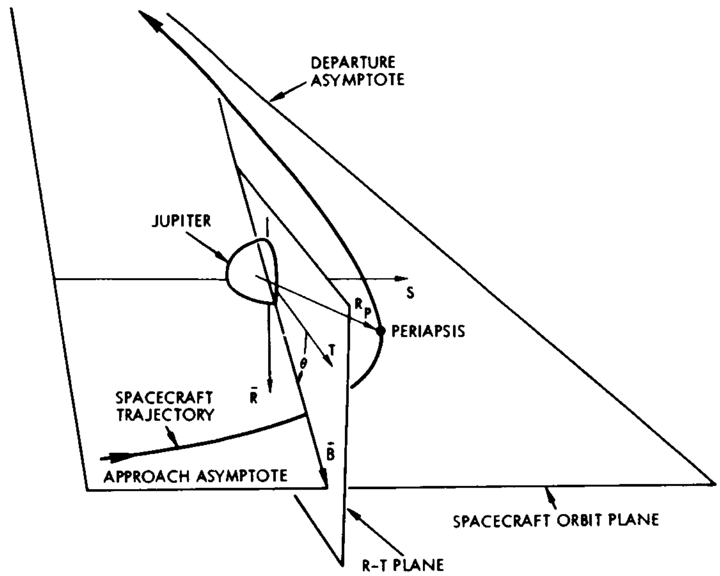


Figure 3-8. Encounter Geometry

The effects of Jupiter's gravitational field will increase the spacecraft velocity to a maximum, at the point of closest approach to the planet, and focus and bend the trajectory. The closer the spacecraft is to Jupiter at periapsis, the greater these effects are. Figure 3-9 indicates how some of these quantities vary with R_P, the periapsis distance from the planet's center, for two different values of the asymptotic approach velocity. All quantities in the figure are expressed in planet-centered coordinates.

Sample encounter trajectories have been generated and plotted for eastward equatorial passages following the interplanetary trajectory discussed above. The planet-centered trajectory is not exactly in the plane of the equator, but can be within about 5 deg of the equatorial plane. Values of R_P, the periapsis distance from the planet's center, of 1.5 and 3 R_J, (Jupiter radii) are used. Figure 3-10 shows the path within 100 R_J of Jupiter, with R_P equal to 1.5 R_J. Figure 3-11 shows the same trajectory at an enlarged scale, within 6 R_J of the planet.

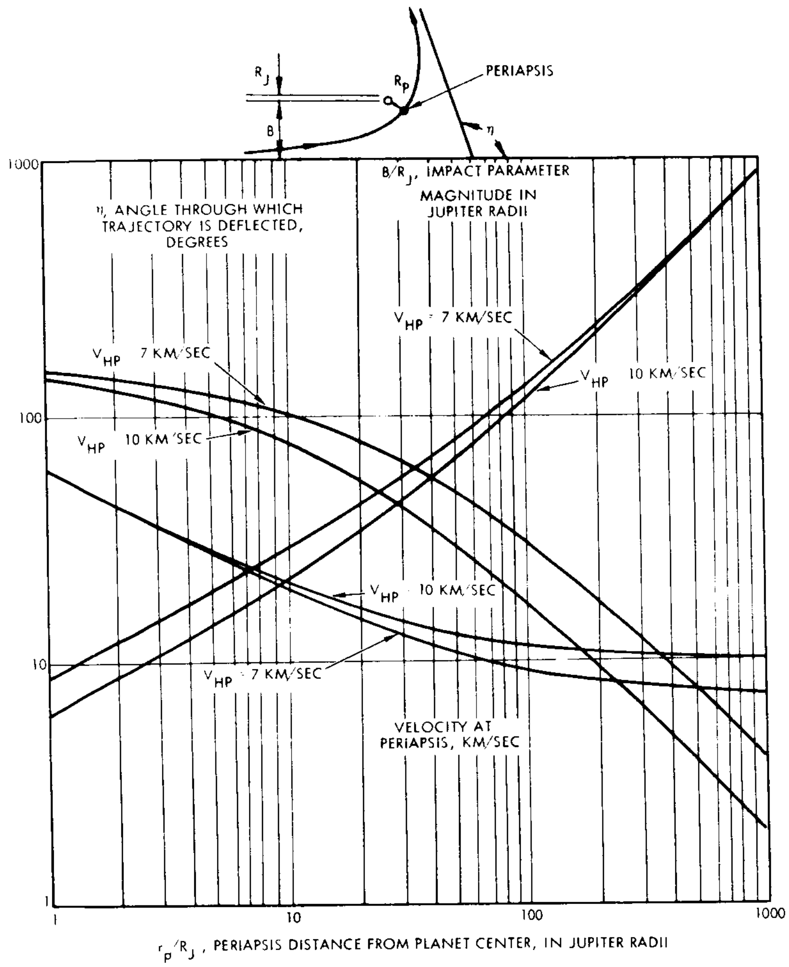


Figure 3-9. Jupiter Gravitational Effects Versus Periapsis Distance (from Ref 5)

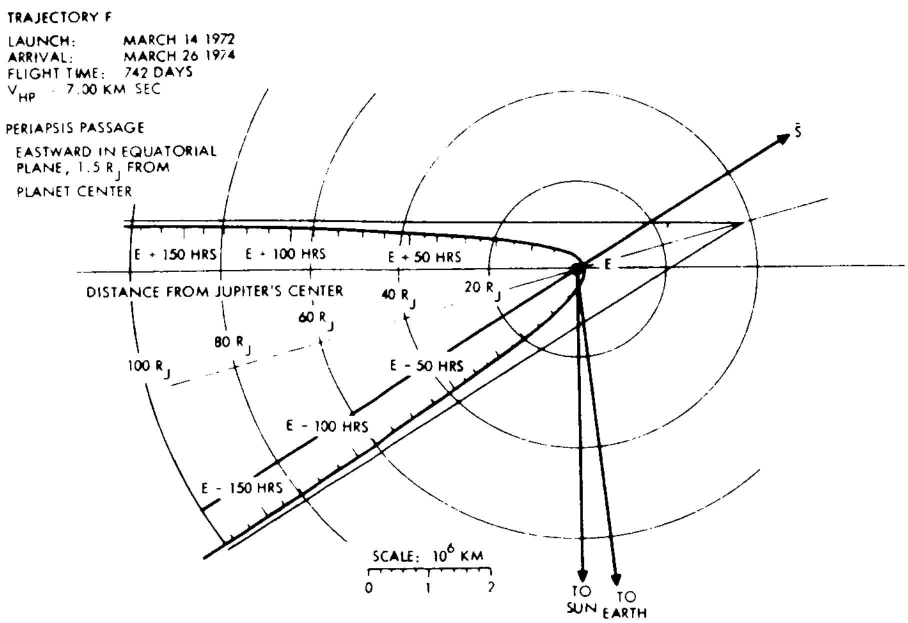


Figure 3-10. Encounter Geometry at Jupiter, $R_P = 1.5\ R_J$ (from Ref 5)

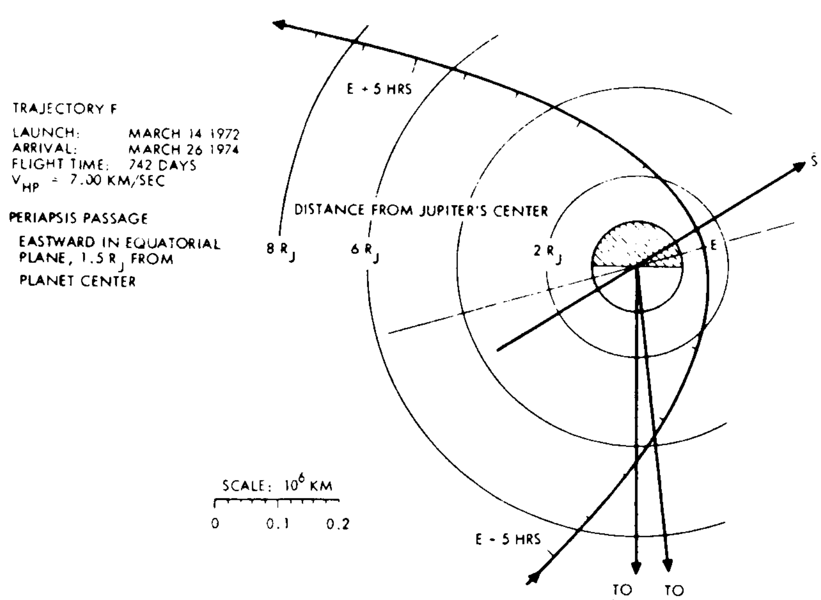


Figure 3-11. Encounter Geometry at Jupiter, $R_P = 1.5\ R_J$, Enlarged Scale (from Ref 5)

3.5.2 Guidance System Operational Sequence

Independent of the specific mission trajectory chosen, the various mission profiles do not differ significantly from one another in the mission phases and required guidance system functions. The typical trajectory will contain the following phases with the indicated guidance system functions required:

a) Launch and boost to ~ 185 km parking orbit: The kick stage strapdown inertial guidance system will provide the guidance function for this phase.

b) Coast in parking orbit: Following injection into the parking orbit, the kick stage/payload will coast until the interplanetary orbit injection maneuver. The inertial guidance system will perform attitude reference and control functions during this phase.

c) Heliocentric orbit injection: For the injection energy assumed, a velocity increment of approximately 7.0 km/sec is needed. This will be divided between the Centaur second burn and the kick stage. The kick stage inertial guidance system will provide the attitude and burn control for both stages.

d) Coast in heliocentric transfer ellipse and midcourse correction: These phases are similar to the corresponding phases for the Mars and Lunar Orbiter missions. The midcourse correction will occur 5 to 20 days from injection.

e) Coast to Jupiter encounter: The strapdown inertial guidance system will perform only attitude control functions during this phase, with the primary attitude reference being obtained from the body-fixed sun and Canopus sensors.

3.5.3 Guidance System Performance Requirements

3.5.3.1 Overall Mission Accuracy Requirements

For both the solar probe mission with Jupiter swingby and Jupiter flyby mission to observe the planet requires that the vehicle pass the planet at a prescribed point defined by the impact vector $\overline{B}$. Another major mission requirement is the midcourse correction capability of the spacecraft. The tolerances shown in Table 3-XII are typical values and have been used as requirements in this study.

Table 3-XII. Assumed Jupiter Mission Requirements

Parameter	Value or Tolerance
Encounter distance	142,800 km ±40,800 km (3σ) $2R_j^{\dagger}$ ±0.6R_j (3σ)
Maximum allowable ΔV for midcourse corrections (not to be exceeded more than 1 percent of the time)	100 m/sec
†R_j denotes the radius of Jupiter.	

3.5.3.2 Interplanetary Trajectory Injection

The ascent guidance phase will include the atmospheric and exoatmospheric ascent, the injection into a parking orbit, and the final injection into the heliocentric elliptic transfer orbit. The accuracy of the injection conditions can be traded off with the midcourse correction requirements. The requirements shown in Table 3-XIII are based on typical midcourse correction capabilities.

Table 3-XIII. Injection Guidance Requirements for the Jupiter Mission

Parameter	Specification (1σ values)
Error in velocity magnitude at injection	9.5 m/sec
Total velocity error perpendicular to the velocity direction	34.7 m/sec

3.5.3.3 Midcourse Corrections

Midcourse corrections are required to remove the terminal errors resulting from injection inaccuracies. The number and timing of these corrections are functions of the correction philosophy, the tracking system accuracy, and the trajectory or spacecraft constraints on the maneuver. For the purposes of this study, a particular correction philosophy, trajectory, spacecraft configuration, and single midcourse correction are assumed (see Subsection 4.2). The midcourse correction removes either the time-of-flight error and terminal errors in two mutually perpendicular directions or terminal errors only.

The requirements for execution of the midcourse maneuver are presented in Table 3-XIV.

Table 3-XIV. Guidance Requirements for Midcourse Correction

Parameter	Specification (1σ Values)
Proportional error	0.75 percent
Pointing error	2/3 deg
Velocity cutoff resolution error	0.0188 m/sec

3.5.4 Guidance System Configuration and Operational Features

Inasmuch as it is anticipated that the guidance system for the Jupiter mission is to be used to control the launch and ascent phases as well as the interplanetary phase, the complete three-axis strapdown inertial guidance system will be required. The system block diagram and operational features will be similar for the Mars orbiter mission (less the approach sensor) and will not be repeated here.

4. SUMMARY OF GUIDANCE SYSTEM PERFORMANCE ANALYSES RESULTS

4.1 PERFORMANCE ANALYSES OF CANDIDATE OPTICAL/INERTIAL SYSTEMS FOR BOOST FLIGHT AND SYNCHRONOUS ORBIT INSERTION

This subsection summarizes the performance analyses of the candidate strapdown inertial guidance systems (described in Section 2.3) for boost and other major powered phases. Primary emphasis has been placed on analysis of the complete synchronous orbit mission and on the lunar mission (launch through translunar orbit insertion).

The navigational errors of the inertial guidance subsystems, as augmented by the optical sensor subsystem, were determined by means of an inertial guidance error analysis program which calculates the effect by integrating the first order perturbation equations along a nominal trajectory. The error analysis computer program is described in Paragraph 7.1.2, Volume II, and in Reference 17.

In the synchronous orbit mission, the errors at injection into synchronous orbit were first calculated. The delta-velocity required to achieve the desired orbit was then determined by Monte Carlo techniques. Twelve different runs were made with different candidate systems. In the translunar orbit-injection mission, the errors at injection into translunar orbit were first calculated. The delta-velocity required to perform the midcourse corrections was then determined. Four different runs were made — three with different candidate strapdown systems and, for comparison, one with the Centaur gimbaled platform.

In each case a nominal trajectory (launch through injection) was generated which was representative of the mission desired, and an error analysis tape containing the position, acceleration, and attitude history for the powered flight phase was produced for input to the error analysis program. The general characteristics of these powered flight trajectories are given in Section 2, Volume II.

4.1.1 Error Analysis for Synchronous Orbit Insertion

The synchronous orbit mission involves extended flight times so that a pure inertial system can cause unacceptable injection errors. Both optical attitude updates using onboard sensors and a time of perigee burn update using ground tracking or an autonomous navigator are considered as solutions to the problem.

The inertial subsystem error models used are for the TG-166 and TG-266 strapdown inertial reference units. (See Section 4, Volume II for additional details.)

The initial velocity errors were taken as zero relative to the earth. The initial orientation errors include, in addition to the 20 arc sec shown in Table 4-I, the effects of accelerometer errors. These effects are introduced because it is assumed that the accelerometers are used in a leveling mode to initialize the direction cosine matrix. The initialization of the direction cosine matrix in azimuth is assumed to be accomplished optically.

Table 4-I. Error Model Used for Strapdown Inertial Guidance Performance Analysis (See Table 2-IX for Strapdown IRU Error Model)

Description	Type	Value
Vertical position	Initial condition	3.0 m
East, north position	Initial condition	15.3 m
Orientation	Initial condition	20 arc sec
Roll axis at apogee (earth)	Optical sensor	9 arc min
Yaw, pitch axes at apogee (sun)	Optical sensor	2 arc min
Roll, pitch axes at perigee (earth)	Optical sensor	18 arc min
Yaw axis at perigee (sun)	Optical sensor	2 arc min
Update time	Ground tracking or onboard sun sensor	0.238 sec

It is assumed that optical attitude update measurements may be made in the 185 km coasting orbit 10 min before perigee burn and in the Hohmann transfer orbit 10 min before apogee burn. The earth sensor errors are assumed to be 18 arc min per axis in 185 km orbit and 9 arc min per axis in synchronous orbit. The sun sensor errors are assumed to be 2 arc min per axis. It is assumed that the sun lies approximately in the direction of the vehicle roll axis during the apogee measurement and fairly near the horizontal plane in the perigee measurement. The sun sensor is used for pitch and yaw angles in the apogee measurement and for the yaw angle in the perigee measurement, with the earth sensor being used for the remaining angles.

Summary of Results and Conclusions

Table 4-II identifies the 12 runs made and Table 4-III presents the results of these runs. One sigma position, velocity, and orientation errors at injection into synchronous orbit are presented in RTN (radial tangential, normal) coordinates along with the ΔV required for 95 percent probability of successful synchronization. The same results are presented in ECI (earth centered inertial) coordinates in Table 4-IIB.

An identification of the largest instrument error sources contributing to the position, velocity, and orientation errors is given in Section 7, Volume II.

The following conclusions were reached for the synchronous orbit mission:

- Prelaunch calibration is desirable.
- Apogee attitude update is necessary for all missions.
- Perigee attitude update and time update are necessary for missions with a 185 km altitude coast period of long duration.
- The performance of the TG-166 system for long coasts and of the more accurate TG-266 system for both short and long coasts is limited by the horizon tracker errors.
- Time update errors of the magnitude used are not significant compared to other error sources.
- A full position and velocity update would not provide significant improvement unless the attitude update errors were reduced.

Table 4-II. Synchronous Mission Runs

Run No.	Coast Orbits	System No.	Prelaunch Calibration	Time Update	Attitude Update	
					Perigee	Apogee
1	0	166	No	No	No	No
2	0	166	No	No	No	Yes
3	0	166	Yes	No	No	No
4	0	166	Yes	No	No	Yes
5	0	266	No	No	No	No
6	0	266	No	No	No	Yes
7	8	166	No	No	Yes	Yes
8	8	166	No	Yes	Yes	Yes
9	8	166	Yes	No	Yes	Yes
10	8	166	Yes	Yes	Yes	Yes
11	8	266	No	No	Yes	Yes
12	8	266	No	Yes	Yes	Yes

Table 4-IIIA. Error Analysis Results for the Synchronous Mission (RTN Coordinates)

Run No.	Position (km)			Velocity (m/sec)			Orientation (arc sec)			95% ΔV (m/sec)
	R	T	N	R	T	N	Yaw	Roll	Pitch	
1	56.5	41.8	35.7	26.7	11.2	23.4	2900	3110	3670	73
2	56.7	41.8	35.7	7.4	1.9	1.9	176	505	308	13
3	49.7	20.8	19.8	26.4	11.0	23.6	2930	3090	3670	75
4	50.0	20.8	19.9	6.2	1.8	1.6	176	490	307	9
5	30.2	20.3	14.0	13.0	5.2	11.1	1380	1500	1760	35
6	30.3	20.3	14.1	5.2	1.1	1.2	136	482	285	8
7	513	793	430	83.5	14.5	8.1	176	505	308	163
8	59.1	128	73.8	9.9	2.0	2.1	176	505	308	22
9	354	534	290	56.7	10.2	5.8	176	490	307	109
10	52.8	122	68	9.1	1.9	2.1	176	490	307	19
11	259	408	222	42.7	7.4	4.4	136	482	285	83
12	33.2	121	66	8.0	1.3	1.8	136	482	285	19

Table 4-IIIB. Error Analysis Results for the Synchronous Mission (ECI Coordinates)

Run No.	Position (km)			Velocity (m/sec)			Orientation (arc sec)			95% ΔV (m/sec)
	X	Y	Z	X	Y	Z	X	Y	Z	
1	60.1	36.0	35.4	26.3	11.6	23.3	2900	3100	3670	73
2	60.4	36.0	35.7	7.5	1.3	1.9	187	501	308	13
3	51.2	18.4	19.8	26.1	11.6	27.5	2920	3090	3670	75
4	51.5	18.4	19.9	6.3	1.3	1.6	186	487	307	9
5	32.3	16.8	14.1	12.7	5.5	11.0	1380	1490	1760	35
6	32.3	16.8	14.1	5.2	0.9	1.2	148	479	285	8
7	634	701	430	84.8	1.6	8.1	187	501	308	163
8	71.3	121	73.8	10.0	1.5	2.1	187	501	308	22
9	436	472	289	57.6	1.6	5.8	186	487	307	109
10	64.0	117	68.0	9.1	1.5	2.1	186	487	307	19
11	314	357	218	42.4	1.3	4.3	148	479	285	83
12	46.3	116	65.9	8.0	1.3	1.8	148	479	285	19

4.1.2 Error Analysis for Translunar Orbit Insertion

The lunar mission was analyzed from liftoff to injection into the translunar orbit. The ΔV required for a 95 percent probability of successfully performing the midcourse correction is taken as a figure of merit.

The performance of the TG-166 and TG-266 systems was compared with that of a Centaur gimbaled inertial guidance system. Table 4-IV presents the error model for the Centaur A/C-10 gimbaled IMU as obtained from Reference 18. Figure 4-1 shows the Centaur gyro and accelerometer orientation at launch.

Summary of Results and Conclusions

Table 4-V identifies the four runs made and summarizes the one-sigma position, velocity, and orientation errors at injection into earth-moon transfer orbit, and the ΔV required for 95-percent probability of successfully performing the midcourse correction. The errors are presented in both ECI (Earth Centered Inertial)† and RTN (radial, tangential,

† See Paragraph 1.4.5 for coordinate system definitions.

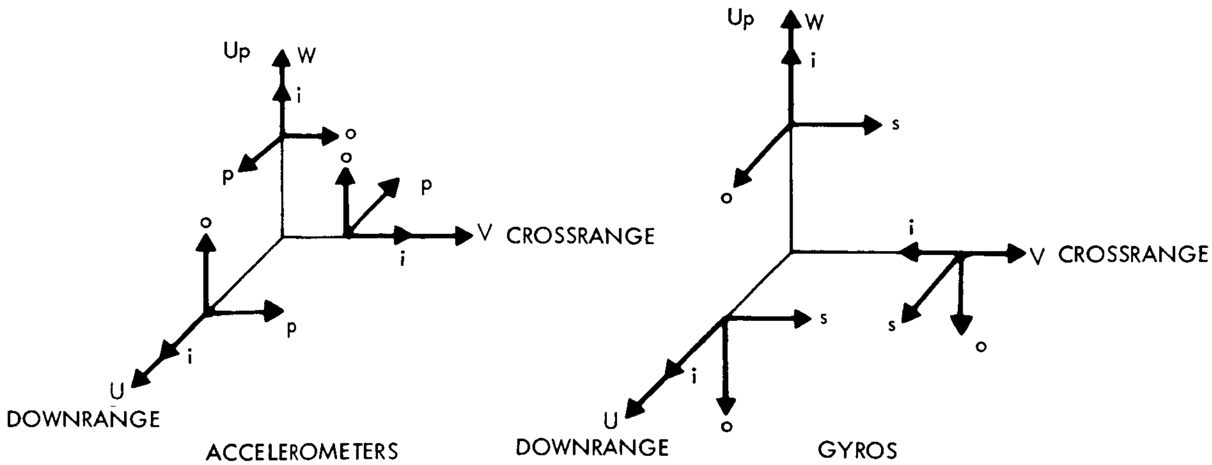


Figure 4-1. Centaur Sensor Orientation

Table 4-IV. Error Model for the Centaur IMU (from Ref 18)

Type	Description	Value	Units
Initial	Vertical position	3.0	m
Initial	East, north position	15.3	m
Initial	Azimuth error	18.6	arc sec
Initial	Level errors	11.1	arc sec
Accelerometer	Bias	42	μg
Accelerometer	U accelerometer inflight bias	24	μg
Accelerometer	V accelerometer inflight bias	26	μg
Accelerometer	W accelerometer inflight bias	29	μg
Accelerometer	Scale factor	51	μg/g
Accelerometer	V accelerometer input axis rotation toward U axis	10.3	arc sec
Accelerometer	W accelerometer input axis rotation toward U axis	10.3	arc sec
Accelerometer	W accelerometer input axis rotation toward V axis	11.3	arc sec
Accelerometer	Scale factor g proportional nonlinearity	9.4	$\mu g/g^2$
Accelerometer	Output axis g^2 sensitivity	9	$\mu g/g^2$
Accelerometer	Input-pend. g-product sensitivity	13	$\mu g/g^2$
Accelerometer	Input-output g-product sensitivity	12	$\mu g/g^2$
Accelerometer	Pend.-output g-product sensitivity	8	$\mu g/g^2$
Gyro	U gyro bias drift	0.084	deg/hr
Gyro	W gyro bias drift	0.094	deg/hr
Gyro	V gyro bias drift	0.093	deg/hr
Gyro	U gyro input g-sensitive drift	0.106	deg/hr/g
Gyro	W gyro input g-sensitive drift	0.114	deg/hr/g
Gyro	V gyro input g-sensitive drift	0.101	deg/hr/g
Gyro	U gyro spin g-sensitive drfit	0.173	deg/hr/g
Gyro	W gyro spin g-sensitive drift	0.177	deg/hr/g
Gyro	V gyro spin g-sensitive drift	0.190	deg/hr/g
Gyro	Input-spin g-product drift	0.009	$deg/hr/g^2$

normal) coordinates.[†] The ΔV requirement is given for the two cases of variable time of arrival guidance and fixed time of arrival guidance. Additional detailed results are presented in Section 7, Volume II.

The following conclusions were reached for the translunar orbit injection mission.

- Prelaunch calibration is desirable.
- The most significant error sources are pitch gyro bias and roll gyro mass unbalance for the strapdown systems and y-gyro mass unbalance for the gimbaled system.
- All resulting errors are well within the requirements summarized in Tables 3-V and 3-VI of Section 3.

Table 4-VA. Error Analysis Results for the Lunar Mission (RTN Coordinates)

Run No.	System No.	Pre-launch Cal.	Position (km)			Velocity (m/sec)			Orientation (arc sec)			95% ΔV (m/sec)	
			R	T	N	R	T	N	Yaw	Roll	Pitch	Variable Time	Fixed Time
1	TG-166	No	1.8	3.5	5.6	7.5	2.9	9.0	568	233	304	50	62
2	TG-166	Yes	1.4	3.4	3.3	7.2	2.8	4.8	291	233	304	49	60
3	TG-266	No	0.8	1.9	1.8	3.6	1.4	2.8	175	114	147	25	31
4	A/C-10	No	1.5	3.4	1.5	5.0	2.4	2.4	161	152	150	37	48

Table 4-VB. Error Analysis Results for the Lunar Mission (ECI Coordinates)

Run No.	System No.	Pre-launch Cal.	Position (km)			Velocity (m/sec)			Orientation (arc sec)		
			X	Y	Z	X	Y	Z	X	Y	Z
1	TG-166	No	2.7	3.9	5.0	8.1	4.8	7.7	550	253	319
2	TG-166	Yes	2.1	3.2	3.0	7.5	3.0	4.1	299	249	283
3	TG-266	No	1.2	1.8	1.7	3.8	1.7	2.3	172	123	143
4	A/C-10	No	2.0	2.9	1.8	5.2	2.1	2.1	160	152	152

[†] See Paragraph 1.4.5 for coordinate system definitions. In this section the X-axis of the ECI coordinate system lies along the Greenwich meridian at launch.

4.2 PERFORMANCE ANALYSES FOR THE MIDCOURSE PHASE

Midcourse trajectory corrections are required, in general, to meet the terminal accuracy requirements of lunar and interplanetary missions because for many missions the injection errors, propagated to the target planet or to the moon, exceed the desired errors at encounter. The injection errors depend somewhat on the launch vehicle characteristics, but primarily on the accuracy of the booster guidance system. The state-of-the-art of boost phase guidance is quite advanced; however, even for the best available guidance systems, the errors at injection considerably exceed those desired for most targeted interplanetary or lunar mission.

The capabilities of ground based radio tracking and orbit determination techniques (See Subsection 2.2) have advanced to the point where midcourse trajectory corrections can be made with sufficient accuracy to meet the mission terminal objectives with a reasonably small expenditure of spacecraft propellants.

The midcourse correction problem is briefly discussed in this section primarily for the Jupiter mission.† See Volume II, Section 8 for a more detailed discussion. The analysis applies either to a flyby mission or to solar probe with a Jupiter swingby. A fully attitude stabilized spacecraft with suitable propulsion for making the necessary maneuvers is assumed. The analysis relies heavily on the results contained in References 4 and 5.

The guidance concept is similar to that employed in Ranger, Mariner, Surveyor, Lunar Orbiter missions, and other missions:

- The DSIF (S-Band) tracking systems and ground computational facilities are assumed for orbit determination from injection through encounter with the target planet (see Subsection 2.2).

†Analysis of the Lunar mission or Mars mission is similar and will not be presented here.

- Based on this determination of the spacecraft position and velocity, corrective maneuvers are computed and transmitted to the spacecraft on-board guidance equipment for execution.

The midcourse maneuver is defined by the impulsive velocity correction, ΔV, necessary to correct the target errors and (optionally) the time of flight.

There are many tradeoffs associated with:

- Single versus multiple midcourse maneuvers and the points at which the corrections are applied
- Allowable spacecraft ΔV capability (this ultimately becomes a tradeoff with payload weight)
- Ranges of possible injection guidance errors (these depend on the booster guidance system and on the launch through injection trajectory)
- Tracking system accuracies attainable (these are a function of the trajectory geometry, tracking radar capabilities and utilization, and ground data reduction capabilities)
- Midcourse maneuver execution errors (these depend on the sophistication of the on-board optical/inertial system)

Analysis of these tradeoffs is beyond the scope of this study.

4.2.1 Midcourse Guidance Techniques

Midcourse guidance is performed by pointing the spacecraft thrust in a direction so that a single velocity increment removes the target errors. This technique, called "arbitrary pointing," was used with Ranger, Mariner, and Surveyor, and allows a single correction to remove all target errors or to remove two components of miss at the target (critical plane correction) and ignore time-of-flight errors.

Target errors are conveniently specified in terms of the components of the impact parameter vector $\overline{B}$ in the R-T plane and the time of flight t_f (see Figure 4-2).

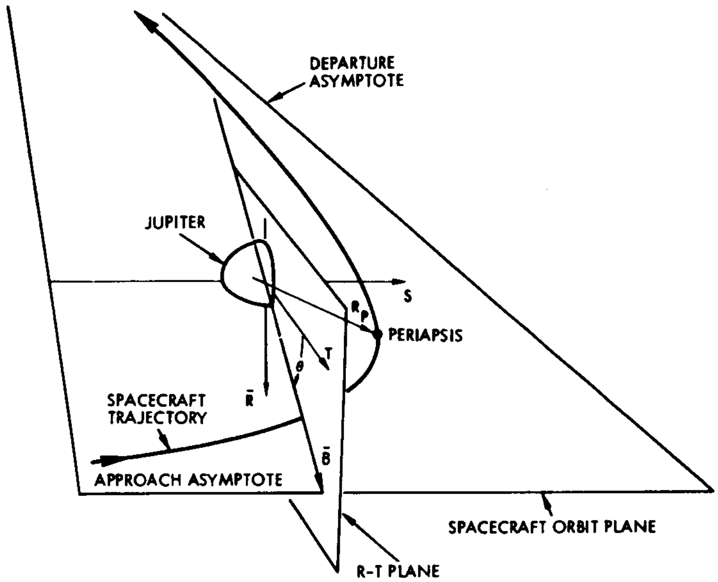


Figure 4-2. Encounter Geometry

For a given interplanetary trajectory, the impact parameter vector $\overline{B}$ specifies in which direction from the planet and what distance the approach asymptote lies. $\overline{B}$ is commonly expressed in components $\overline{B} \cdot \overline{R}$ and $\overline{B} \cdot \overline{T}$, where $\overline{R}$, $\overline{S}$, $\overline{T}$ are a right-hand set of mutually orthogonal unit vectors aligned as follows: $\overline{S}$ is parallel to the planet centered approach asymptote, $\overline{T}$ is parallel to the plane of the ecliptic and positive eastward, and $\overline{R}$ completes the set and has a positive southerly component. The magnitude of $\overline{B}$ determines the distance of closest approach to the planet and the angle

$$\theta = \tan^{-1} \frac{\overline{B} \cdot \overline{R}}{\overline{B} \cdot \overline{T}}$$

specifies the orientation of the planet-centered orbit plane as a rotation about the $\overline{S}$ axis. These definitions are illustrated in Figure 4-2.

4.2.2 Post Midcourse Trajectory Accuracy Analysis

Estimates for the uncertainty of control of the interplanetary trajectory subsequent to the midcourse correction maneuver are presented in the following paragraphs. The contributions to this uncertainty are the error in execution of the midcourse trajectory correction, the uncertainty in tracking the spacecraft from injection to midcourse correction, ephemeris and astronomical unit errors, and certain identifiable but unpredictable trajectory perturbations acting after the midcourse correction. The midcourse guidance technique described in Subsection 8.2 is assumed for this analysis. It consists of a single midcourse correction about 10 days after launch, with the thrust vector directed essentially parallel to the critical plane to reduce $\overline{B} \cdot \overline{T}$ and $\overline{B} \cdot \overline{R}$ errors.

The root-mean-square and percentage contributions to the target coordinates $\overline{B} \cdot \overline{T}$ and $\overline{B} \cdot \overline{R}$ are listed in Table 4-VI. The percentage contribution of the total deviation in $\overline{B} \cdot \overline{T}$ and $\overline{B} \cdot \overline{R}$ are computed by assuming that the mean square error contributions are additive.

The midcourse execution errors are calculated for a Mariner-type midcourse guidance system (Configuration Ia described in Paragraph 2.4.1.3, Volume II) and represent the largest error contribution, as might be expected.

More accurate control of the trajectory, if required, could be obtained by improving the precision of the midcourse maneuver either by using a full strapdown guidance system or by increasing the number of maneuvers. Of the remaining errors, the greatest is the pre-midcourse tracking uncertainty† which causes the estimated position of the spacecraft to be in error. This error is based on present state-of-the-art tracking accuracies attainable by the DSIF (see Subsection 2.2). Presumably, by 1972 greater accuracy can be attained. Likewise, ephemeris errors and uncertainty in the astronomical unit are based on present state-of-the-art and by 1972 will be appreciably reduced.

---

†Paragraph 8.3.1 of Volume II describes the results of an analysis of pre-midcourse tracking performed to calculate the state vector uncertainties due to radar tracking and the associated dispersion ellipse at Jupiter. The reader is referred to this paragraph for the detailed results.

Table 4-VI. Post-Midcourse Trajectory Errors
(from Ref 5)

Error Source	RMS $\overline{B} \cdot \overline{T}$ Error (km)	RMS $\overline{B} \cdot \overline{R}$ Error (km)	Percent of Total $\overline{B} \cdot \overline{T}$ Variance	Percent of Total $\overline{B} \cdot \overline{R}$ Variance
Injection errors	951,000	388,000	†	†
Midcourse execution errors ††	8,850	10,600	93.0	99.4
Pre-midcourse tracking errors	2,050	625	5.0	0.3
Nongravitation perturbations (unpredictable portions)	1,067	217	1.4	-
Ephemeris errors	500	500	0.3	0.2
Astronomical Unit conversion factor uncertainty	303	303	0.2	0.1
Total rss	9,150	10,650	100.0	100.0

Total 99 percent miss ellipse:

Semimajor axis = 26,300 km

Semiminor axis = 17,400 km

† Does not directly affect post-midcourse target errors.

†† Arbitrary pointing critical plane correction at 10 days past injection.

4.2.3 Midcourse Execution Errors

Orientation and execution errors introduced by the midcourse correction subsystem have been evaluated for a Mariner-type strapdown guidance system and the TG-166 strapdown inertial guidance system. The results appear in Table 4-VII. It is evident that at least an order of magnitude improvement is available by using the more sophisticated strapdown

Table 4-VII. Comparison of Midcourse Execution Errors for Two Types of Inertial Guidance Subsystem Mechanizations

	Mariner-Type Simplified Strapdown Guidance System	TG-166 Full Strapdown Guidance System†
Proportional velocity error	0.75 percent (1σ)	0.043 percent (1σ)
Pointing error	0.67 deg (1σ) (11.6×10^{-3} rad)	0.06 deg (1σ) (10^{-3} rad)
Resolution error	0.0188 m/sec	(Negligible)
ΔV error in performing a maximum 100 m/sec maneuver	0.75 m/sec (1σ) (parallel component)	0.04 m/sec (1σ)
	1.2 m/sec (1σ) (lateral component)	0.1 m/sec (1σ)

†See Paragraph 3.3.1.2 for error model.

inertial system. Optical sensor accuracies are comparable in the two systems (3 arc min inertial attitude accuracy in each axis is assumed for the latter system).

The errors presented in Table 4-VII for the two types of optical/inertial systems may be applied directly to the analysis of the midcourse correction requirements for other missions and to other maneuvers such as orbit insertion. The resultant mission errors will, of course, be different than those given above for the Jupiter mission.

The TG-166 performance satisfies all of the midcourse correction and orbit insertion ΔV requirements summarized in Paragraph 3.3.2.3 (Table 3-III). The TG-266 system, which has better accelerometer performance, also satisfies these requirements.

4.3 NAVIGATION PERFORMANCE ANALYSES FOR INTERPLANETARY AND PLANET APPROACH PHASES

The radio/optical/inertial tracking and navigation error analyses were conducted using the SVEAD† computer program. The results of the study, presented in Section 9, Volume II, are summarized here. Briefly, this error analysis was concerned with the comparative performance of DSIF tracking (earth-based doppler) and onboard optical navigation. Optical instruments considered were: star Canopus sensor, planet (Mars) sensor, and sun sensor. The planet sensor is used in conjunction with the other sensors to make measurements of the cone and clock angles (defined below) and to make an angular subtense (range measurement) of Mars. Major error sources considered were: slowly drifting biases in the optical equipment, uncertainty in the diameter of Mars, Mars ephemeris errors, doppler bias error (slowly drifting), and uncertainty in the dynamic model of the solar system (i.e., errors in solar radiation forces on the spacecraft, gravitational constants, planet oblateness, etc.).

†SVEAD is a state variable estimation and accuracy determination program. See Ref 19. The equations for the error analysis program are discussed in detail in Appendix D, Volume II.

The principal purpose of the optical measurements is to locate the position of the planet (Mars) relative to the spacecraft. The lines of sight to two known stars may be used to provide a known coordinate system, in which Mars may be located. For this study, one star was taken to be Canopus, and the other was taken to be the sun. Mars is then located by a cone angle ψ and a clock angle θ, as shown in Figure 4-3. The angle ϕ, shown in Figure 4-3, is the Sun-Canopus angle. The subtense angle α, not shown, is an angular diameter measurement which can be used to determine the distance to Mars. Useful optical measurements, for the trajectory considered in this study, could actually be made only over the period from 10 days to 0.5 day prior to Mars encounter (Mars

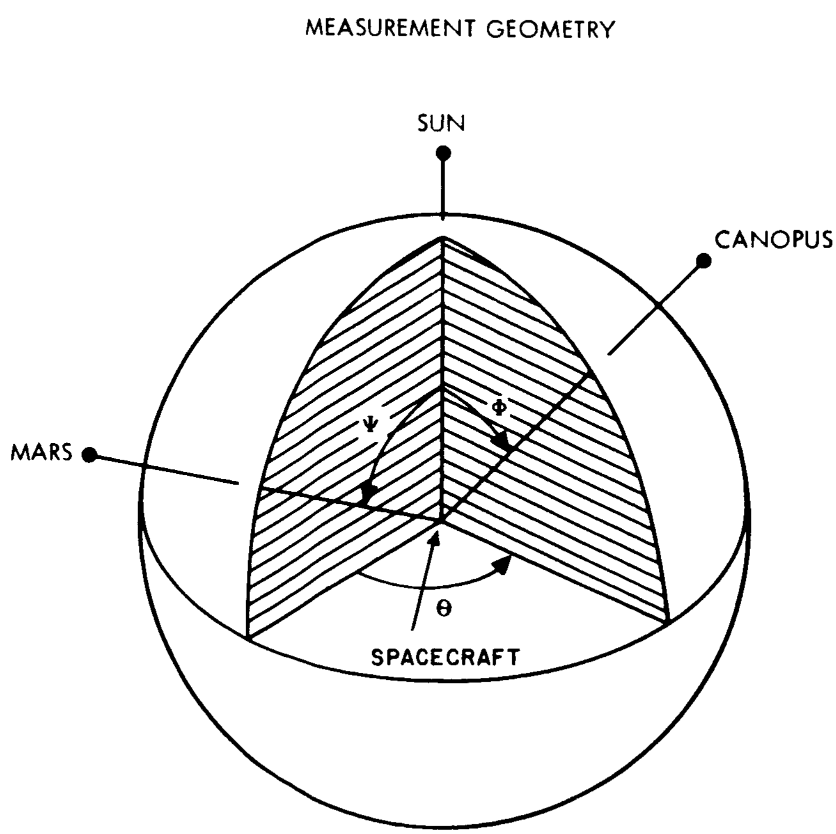


Figure 4-3. Optical Angle Measurements

perifocus). Some of the limitations† are due to fixed-axis trackers (no gimbals), and the limited field of view (about 15 deg). Also, the small image size of Mars several days from encounter reduces the accuracy of the subtense angle measurements, and near encounter the image size may exceed the available field of view. However, for the error analysis runs, the optical trackers were assumed to work all the way from the Earth to Mars perifocus.

4.3.1 Trajectory Used for Navigation Error Analysis

The interplanetary trajectory used in this study is described in Paragraph 3.3.1. The total time of flight from Earth to Mars was $15.228 \cdot 10^6$ sec or 176.25 days. The tracking and navigation error analysis started at 3 hr out from Earth. The X, Y, Z coordinates mentioned below are Earth Centered Inertial coordinates with the Z axis parallel to the earth's axis of rotation, and the X axis directed toward the vernal equinox (See Paragraph 1.4.5).

4.3.2 Error Model Summary

All "biases" in the error models were assumed to be slowly drifting random variables, exponentially correlated in time. Thus each bias error has a standard deviation and a time constant associated with it; the larger the time constant, the more nearly constant is the bias.

A list of pertinent inputs is given in Tables 4-VIII, 4-IX, and 4-X.

Two different error models for the electro-optical sensors were used in order to investigate the possible improvements in orbit determination accuracies possible by using the Mars approach sensor.

---

†Some of these limitations may be removed at the expense of a more complex sensor. For example, dual range optics and gimbaling of the sensor might be employed. These additional complexities are undesirable from a reliability point of view.

Table 4-VIII. Radio/Optical/Inertial Error Model for Mars Mission

Error	$(\text{Variance})^{1/2}$	Time Constant
Random Acceleration Acting on Spacecraft (Models uncertainty in the dynamic model of the solar system, i.e., errors in solar pressure forces, gravitational constants, etc.)	0.531×10^{-8} m/sec^2 (0.174×10^{-7} ft/sec^2) (causes a 200 km position error in 176 days)	1 week
Tracking System Errors		
• Range rate bias	10^{-2} m/sec (0.0328 ft/sec)	1/3 day
• Uncorrelated noise on doppler rate	0.732×10^{-2} m/sec (0.024 ft/sec) (equivalent to 0.12 ft/sec per 1 sec sample, 25 meas. averaged)	
Vehicle Errors at Injection (3 hr)		
• Position	2 km (6560 ft)	
• Velocity	2 m/sec (6.56 ft/sec)	
Mars Ephemeris Error (Relative to Earth)		
• Position	220 km (7.22×10^5 ft)	
• Velocity	0.05 m/sec (0.164 ft/sec)	
Radius of Mars	20 km (6.56×10^4 ft)	1 day

Table 4-IX. Optical Error Model A

Error	$(\text{Variance})^{1/2}$		Time Constant
Sun Sensor Bias	1.746×10^{-3} rad	(6 arc min)	1/2 week
Sun Sensor Uncorrelated Noise	0.349×10^{-4} rad* (0.1746×10^{-3} rad)†	(0.12 arc min)	
Mars Sensor Bias	1.92×10^{-3} rad	(6.6 arc min)	1/2 week
Canopus Sensor Bias	0.873×10^{-3} rad	(3 arc min)	1/2 week
Mars Sensor Uncorrelated Noise	0.349×10^{-4} rad* (0.1746×10^{-3} rad)†		
Canopus Sensor Uncorrelated Noise	0.1746×10^{-4} rad* (0.873×10^{-4} rad)†		
Mars Subtense Measurement			
• Lower limit on $(\text{variance})^{1/2}$ of bias	0.873×10^{-3} rad	(3 arc min)	1/2 week
• Error proportional to subtense angle††	1%		1/2 week
• Uncorrelated noise	0.1746×10^{-4} rad* (0.873×10^{-4} rad)†	0.06 arc min (0.3 arc min) †	

*Equivalent error of 25 measurements averaged. This was the value used in the error analysis
†Single measurement error.
††Percent of subtense angle (α) contributing to the standard deviation of the bias error adding to subtense angle, i.e., $\sigma_\alpha^2 = (0.873 \times 10^{-3})^2 + (0.01\alpha)^2$

Table 4-X. Optical Error Model B (Improved Optics)

Error	$(\text{Variance})^{1/2}$		Time Constant
Sun Sensor Bias	0.407×10^{-3} rad	(1.4 arc min)	1/2 week
Sun Sensor Uncorrelated Noise	0.349×10^{-4} rad* $(0.1746 \times 10^{-3}$ rad)†	(0.12 arc min)	
Mars Sensor Bias	0.391×10^{-3} rad	(1.35 arc min)	1/2 week
Canopus Sensor Bias	0.391×10^{-3} rad	(1.35 arc min)	1/2 week
Mars Sensor Uncorrelated Noise	0.349×10^{-4} rad* $(0.1746 \times 10^{-3}$ rad)†		
Canopus Sensor Uncorrelated Noise	0.1746×10^{-4} rad* $(0.873 \times 10^{-4}$ rad)†		
Mars Subtense Measurement			
• Lower limit on $(\text{variance})^{1/2}$ of bias	0.485×10^{-4} rad	(0.17 arc min)	1/2 week
• Error proportional to subtense angle††	0%		
• Uncorrelated noise	0.1745×10^{-4} rad* $(0.873 \times 10^{-4}$ rad)†	0.06 arc min (0.3 arc min)†	

*Equivalent error of 25 measurements averaged. This value was used in the error analysis
†Single measurement error.
††See note on previous table

4.3.3 Summary of Results and Conclusions

Five different doppler tracking and optical navigation cases were considered. They were:

a) Doppler only (Table 4-VIII)
b) Doppler plus optical (Table 4-IX)
c) Doppler plus improved optics (Table 4-X)
d) Optical only (Table 4-IX)
e) Improved optics only (Table 4-X)

As mentioned previously, the optical equipment can actually work only in the range of 10 days from Mars perifocus to 1/2 day from perifocus. However, for the error analysis study, all equipment was assumed to be operating 3 hr out from Earth to Mars encounter.

A summary of the results of the error analysis study is shown in Table 4-XI. It appears from this table that the value of optical measurements is marginal if doppler information is available up to perifocus.† Had the optical measurements been cut off at 1/2 day to encounter, then the optical performance would have been worse than shown.

Table 4-XI. Standard Deviations of Position Errors

Case	2.5 Days to Encounter			At Mars Perifocus		
	σ_x km	σ_y km	σ_z km	σ_x km	σ_y km	σ_z km
1	570	500	450	4.3	7.5	1.4
2	570	500	440	4.0	7.0	1.35
3	500	450	375	2.4	4.1	0.9
4	32,000	28,000	15,500	24	21	15.5
5	4200	3750	2150	5.4	5.4	4.0

†No consideration was given to the problem of Mars eclipsing the vehicle. This problem would be very mission-dependent.

The results given in Table 4-XI are shown graphically and in more detail in Figures 9-4 through 9-8 of Volume II. Based on these limited results, it appears that optical tracking may offer a significant improvement in tracking accuracy only during the brief time range of 1.5 days to 0.05 days from encounter, the maximum improvement occurring at 0.1 days. The optical instruments under consideration in this study, however, can be used no closer than 0.5 days to encounter.

Position and velocity error curves for the complete Earth-to-Mars mission are shown in Figures 9-10 through 9-15 of Volume II. Some conclusions that may be drawn from these limited results are: one may deduce, with the aid of these curves, that the principal error source, prior to 6 days from encounter, is due to the Mars ephemeris error when doppler or doppler plus optical tracking is employed; the principal error source in optical tracking alone, prior to 6 days from encounter, is the initial position and velocity error of the spacecraft with respect to the Earth. It is also seen from these curves that a good position fix with respect to Mars occurs some time after 1 day to encounter, irrespective of the measurements employed. In fact, doppler-only tracking does not obtain a good position and velocity fix until some time after 2000 sec (0.02 day) to encounter. However, this might be improved by increasing the doppler sampling rate. It is of interest to note here that it takes about 370 sec for light to travel between Earth and Mars at the time of encounter. Thus, position and velocity updates, radioed from the Earth to the vehicle, would be based on data that was 740 sec old. This problem could be eliminated if the vehicle carried its own ultrastable oscillator and processed the doppler data onboard. However, in this case the range-rate bias errors would probably be larger than those considered here.

REFERENCES

1. "Design Parameters for Ballistic Interplanetary Trajectories, Part I. One-Way Transfer to Mars and Venus," JPL Technical Report No. 32-77, 16 January 1963

2. "Trajectory Selection Considerations for Voyager Missions to Mars During the 1971-1977 Time Period," JPL Document EPD-281, dated 15 September 1965

3. "Performance and Design Requirements for the 1973 Voyager Mission, General Specifications for," JPL Document SE 002BB-001-1 B21, (Draft) dated 1 January 1967

4. "Phase 1A Study Report, Voyager Spacecraft," TRW Systems Report 5410-0004-RU-001, dated 30 July 1965

5. "Advanced Planetary Probe Study, Final Technical Report," TRW Systems Report 6547-6004-R000, dated 27 July 1966

6. "Utilizing Large Planetary Perturbations for the Design of Deep Space Solar Probe and Out-of-Ecliptic Trajectories," JPL Report TR 32-849, dated 15 December 1965

7. "Standardized Space Guidance System, Final Report for Phase 1a," TRW Space Technology Laboratories Report SSD-TDR-64-132, Vol. III, dated 29 May 1964

8. "Space Tracking and Data Acquisition Network Facilities Report (STADAN)," Goddard Space Flight Center, Greenbelt Maryland, Report No. X-530-66-33, December 1965

9. Harold L. Hoff, "Tracking, Command, Control and Data-Acquisition of NASA Flight Programs," Annals New York Academy of Sciences, Vol. 134, 22 November 1965

10. "Apollo Mission and Navigation System Characteristics," MSC-GSFC Apollo Navigation Working Group Technical Report No. 66-AN-1.1, 1966

11. "Apollo Navigation, Ground and On-board Capabilities," MSC-GSFC Apollo Navigation Working Group Technical Report No. 65-AN-2.0, 1 September 1965

12. J. H. Painter and G. Hondros, "Unified S-Band Telecommunications Techniques for Apollo," Volume 1 — Functional Description, NASA Technical Note TN D-2208, March 1965

13. "Description of Deep Space Network Operational Capabilities as of 1 January 1966," Jet Propulsion Laboratory, NASA Technical Memorandum 33-255, 1 July 1966

14. "Planned Capabilities of the DSN for Voyager 1973," Jet Propulsion Laboratory, Engineering Planning Document No. 283, Revision 2, 1 January 1967

15. N. A. Renzetti, M. S. Glenn, D. L. Gordon and J. R. Hall, "Projected NASA/JPL Deep Space Network Capabilities in the 1970's," Jet Propulsion Laboratory, March 1966, presented at AIAA/AAS Stepping Stones to Mars Meeting, Baltimore, Maryland, March 21-30, 1966

16. "Analysis of Apollo Orbit Determination Accuracy with Random Errors in Ground Based Radar and Onboard Optical Observations," Volume 3, The Translunar Trajectory, TRW Systems Document 8408-6042-RC-000, 4 May 1964 (Confidential)

17. D. F. McAllister and J. C. Wilcox, "Digital Computer Program for a Generalized Inertial Guidance System Error Analysis, Version II (GEAP II)," TRW Document No. 08768-6009-T000, 11 May 1967

18. D. Strawther, "Centaur Guidance System Accuracy Report, AC-10 Accuracy Analysis," GD/C-BTD 64-013-11, 1 April 1966

19. W. M. Lear, "SVEAD User's Manual," TRW Systems Report 7221.11-10, 28 April 1967

20. "Voyager Support Study, Advanced Mission Definition Final Report," TRW Document No. 04480-6001-R000, November 1966.