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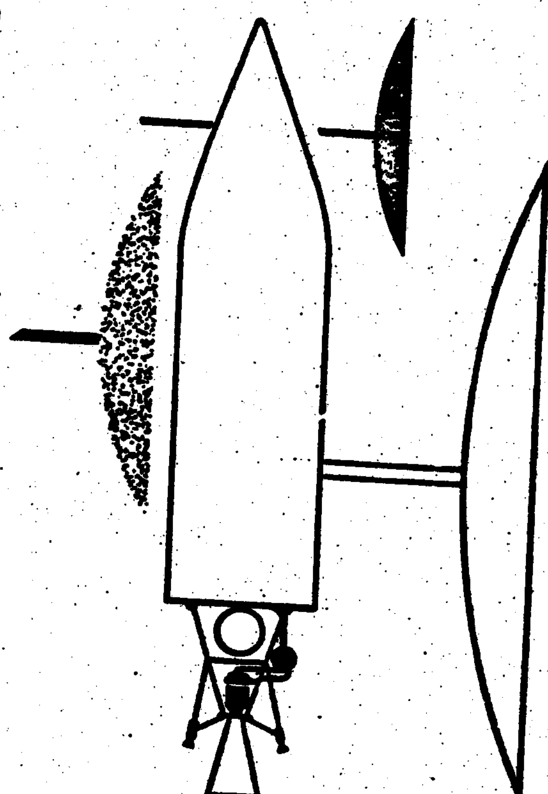

Pied Piper DEVELOPMENT PLAN

VOL. II SUB-SYSTEM PLAN

D. Guidance and Control

3-4-65

LOCKHEED AIRCRAFT CORPORATION
MISSILE SYSTEMS DIVISION
VAN NUYS, CALIFORNIA

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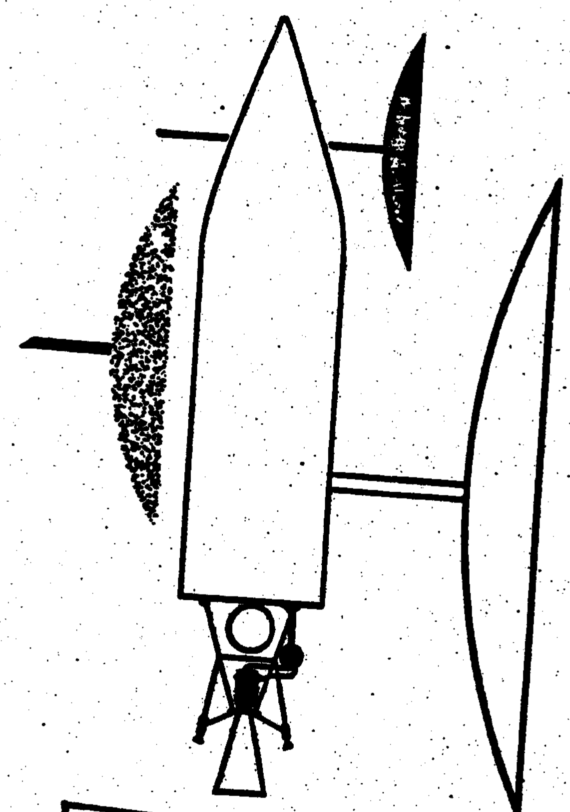

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VOL. II SUB-SYSTEM PLAN

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PIED PIPER DEVELOPMENT PLAN

VOLUME I. SYSTEM PLAN

VOLUME II. SUBSYSTEM PLAN

A. Airframe

B. Propulsion

C. Auxiliary Power

D. Guidance and Control

E. Visual Reconnaissance

F. Electronic Reconnaissance

G. Infrared Reconnaissance

H. Vehicle Electronics

I. Airborne Test Systems

J. Vehicle Intercept and Control Ground Station

K. Ground Data Processing

L. Vehicle Ground Support

CONTENTS

Subsystem D Guidance and Control

SECRET
SECURITY CLASSIFICATION

RDB PROJECT CARD

TYPE OF REPORT: New System-Development Plan		REPORTS CONTROL SYMBOL: DD-RDB(A)46
1. PROJECT TITLE: GUIDANCE AND CONTROL SUBSYSTEM FOR ADVANCED RECONNAISSANCE SYSTEM (UNCLASSIFIED) (PIED PIPER)	2. SECURITY: S.	3. PROJECT NUMBER: 1115
	4. INDEX NUMBER	5. REPORT DATE: 1 March 1956
6. BASIC FIELD OR SUBJECT	7. SUBFIELD OR SUBJECT SUBGROUP	7A. TECH. OBJ.
8. COGNIZANT AGENCY	12. CONTRACTOR AND/OR LABORATORY: Lockheed Missile Systems Division	CONTRACT/W.O. NO.: AF33(616)-3105
9. DIRECTING AGENCY		
OFFICE SYMBOL / TELEPHONE NO.		
10. REQUESTING AGENCY	13. RELATED PROJECTS	17. EST. COMPL. DATES: RES. / DEV. / TEST / OP. EVAL.
11. PARTICIPATION, COORDINATION, INTEREST	14. DATE APPROVED	18. FY / FISCAL ESTS. (IN $)
	15. PRIORITY: Maximum / 16.	
19.		

20. REQUIREMENT AND/OR JUSTIFICATION

20 a. The guidance and control subsystem is required to provide the following functions:-

1. Inertial guidance for the satellite from boost to a circular orbit at the prescribed altitude.
2. Correction signals to attitude control system and to orbital boost phase to obtain accurate speed and direction for a prescribed circular orbit.
3. Attitude control during non-powered flight, by use of inertia wheels, and control during orbital boost phase by use of autopilot and control motors.
4. Attitude control of vehicle orientation in orbit for maximum visual reconnaissance resulution.

22. RDP	SN	CN	IC & P	X.	L.	C.

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MISSILE SYSTEMS DIVISION

SECRET
SECURITY CLASSIFICATION

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R&D PROJECT CARD
CONTINUATION SHEET

SECRET
SECURITY CLASSIFICATION

1. PROJECT TITLE	2. SECURITY OF PROJECT	3. PROJECT NUMBER
GUIDANCE AND CONTROL SUBSYSTEM FOR ADVANCED RECONNAISSANCE SYSTEM (UNCLASSIFIED)	S.	1115
(PIED PIPER)	4.	5. REPORT DATE 1 March 1956

21 a. Brief and Operational Characteristics

This subsystem will provide the means for guidance and control of the orbiting vehicle so as to place it in a circular orbit at approximately 300 miles above the surface of the earth. In addition, the subsystem will operate in an orbit attitude control mode to stabilize the vehicle and to provide a platform suitable for mounting reconnaissance elements.

The attitude will be stabilized in order to prevent image motion from degrading resolution of visual data and also the attitude must be known with sufficient accuracy to permit the application of navigation location techniques to the data which are gathered.

b. Approach

Booster vehicles and guidance will be derived from the WS 107A program. The proposed subsystem does not require modification of the Atlas Boosters. It is designed to operate with the closed loop Radio Inertial Guidance System but the subsystem will also accommodate open loop operation.

Input data of altitude, velocity, and flight path angle will be derived from the first stage guidance system. These data will be used to compute differential corrections to a pre-calculated trajectory and are applied to the Orbit Stage Vehicle control system which is referenced to low drift gyros and an integrating accelerometer. It controls rocket engine thrust direction and burning time so that the impulse applied to the orbit stage vehicle is precisely that required to boost the vehicle into the orbit.

The attitude control is obtained from the interaction of the gyroscopic and differential gravity torques which act on a vehicle having elongated, or dumbell, shape and which contains an internal angular momentum directed parallel to the axis of maximum moment of inertia. This vehicle configuration has a single stable attitude. Attitude deviations and/or attitude deviation rates are sensed by gyros for the application of torques to counter-act disturbances and to apply damping torques as needed.

The effects of the environment on the sensing instruments will constitute the major problem in this development program. The effects of the dynamic loads on inertial instruments in ballistic rockets have not been fully assessed.

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21 c. Tasks of the Subsystem

1. a. Transition Computer

   b. Contractor: LAC MSD

   c. This computer employs linear theory to compute guidance corrections to a precalculated trajectory. These corrections are calculated from the deviation of the observed trajectory from a reference trajectory. They are applied to a program which was established prior to launching and are employed to modify the orbit stage vehicle control system settings to insure the attainment of orbiting conditions.

2. a. Thrust On/Off

   b. Contractor: LAC MSD

   c. The Thrust on/off control operates from a time signal (a clock) to initiate rocket engine burning. Engine shutdown is commanded when the integrating accelerometer indicates that the desired velocity increment has been added.

3. a. Attitude Reference Unit

   b. Sub-contract

   c. An accurate, low drift gyroscopic attitude references is required to ensure that the thrust applied to the vehicle during orbit stage boost does not contribute excessive vertical velocity to the vehicle at apogee and to provide a heading reference during the ascent.

   In addition, the attitude reference unit is employed in the instrumentation of an attitude control system for the coasting phase of the ascent and also for the orbital phase of the mission.

4. a. OSV Autopilot

   b. Sub-contract

   c. The orbit stage autopilot provides for dynamic control of thrust direction in order to maintain the vehicle in a

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controlled attitude during thrust accelerations. This unit, which is primarily control engine servos and amplifier, derives its commands from the attitude reference unit.

5. a. Attitude Control

b. Contractor: LAC MSD

c. The attitude control provides control during two distinct phases of the flight. During transition coast it removes any angular impulse due to separation of the orbit stage vehicle from the booster and stabilizes the vehicle into proper orientation for orbit stage boost; and during orbit flight it stabilizes the vehicle in proper orientation with respect to the earth and maintains stability of the vehicle in this attitude to reduce image motion and to provide a reference for direction finding.

Control is accomplished through the inertial reference unit, the damping computer and two rate controlled inertia wheels aligned along two of the vehicle axes.

6. a. Image Motion Compensation

b. Contractor: LAC MSD

c. Body rotations of the vehicle will cause blurring of the image formed by ground objects, if the attitude control is insufficient to permit maximum use of system resolution. These motions of the image may be compensated by counter motion of the optical carriage.

7. a. Attitude Indication

b. Contractor: LAC MSD

c. An indication of the instantaneous vehicle attitude is necessary in order to correlate reconnaissance date with geographical location. An indication of attitude is derived as three orthogonal angles from the attitude unit. These data are presented to the data transmission system.

21 d. Other Information

The guidance and control system described here is for an orbit stage vehicle only and, since it is to be launched at high altitude from an operational missile system, aerodynamic moments and forces are not of primary importance.

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Because the proposed vehicle is short it is possible that the structural deflection in flight will be small and, therefore, will not appreciably disturb the control system sensing elements. However, control disturbances due to fuel motions will be studied in detail.

The design described requires no modification of the Atlas Booster except those which are accounted for in changing the trajectory. With the exception of the horizon sensing elements, all components required are in production or development status at present. The horizon sensing element does not appear to require any significant advances in the state of the art.

21 e. Background History

Past work on guidance and control has been conducted at NAA, RAND and MIT in connection with Project Feedback.

Studies under Contract AF 33 (616)-3105 have shown feasibility of open-loop guidance of a satellite during ascent using a closed loop ATLAS C boost and the feasibility of attitude control by inertia wheels fixed in the satellite.

These studies have also shown the need for more detailed work concerning open and closed loop inertial guidance systems and attitude control obtained by gyroscopic forces associated with rotating inertia wheels in satellites.

The study of environment and its effect on initial guidance and attitude control components is being conducted under ICBM development studies.

21 f. Future Plans

It is planned to continue the studies of guidance systems and attitude control studies (including error analyses) already initiated and ramifications of these studies and systems leading to an optimization of the systems.

21 g. References

1. Appendix to Subsystem D. (S)
2. Monthly and Quarterly Reports of Project, Pied Piper (S)

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Pied Piper

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LOCKHEED AIRCRAFT CORPORATION
MISSILE SYSTEMS DIVISION

TABS

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SUBSYSTEM D - GUIDANCE AND CONTROL

Tab 1 - General Design Specification

1. GENERAL

The objective of the Guidance and Control Subsystem is to ensure that thrust is applied in such a way that the vehicle is placed in a circular orbit at an altitude of 300 n. miles. At this altitude the vehicle velocity must be in a horizontal plane and its magnitude must be approximately 25,500 ft/sec ($v = \sqrt{gR}$). When the vehicle enters the orbit the error in velocity must not exceed 30 ft/sec in magnitude and 1 milliradian in direction. If these conditions are met, a 300-mile orbit will have maximum and minimum altitudes of 320 and 280 miles respectively. After the orbiting condition has been obtained and the engines have been shut down, the guidance and control subsystem converts to an attitude control mode of operation. The vehicle attitude must be controlled so as to stabilize the line of sight with respect to a known reference frame to permit reconnaissance read-in and read-out.

The guidance and control system specifications presented here describe a system which is compatible with the Atlas C Boosters. The system is designed for operation with a closed-loop Atlas radio-inertial guidance system and will provide an eccentricity of less than 0.002 (i.e. orbit altitude variations of ± 10 miles). The system is compatible with Atlas open-loop operation however, and, when used in this way, the performance depends critically upon the accuracy obtained from the Atlas booster guidance.

FOREWORD

The Advanced Reconnaissance System (ARS) consists of a satellite vehicle containing equipment ~~to perform visual, ferret,~~ and infrared reconnaissance, together with the necessary system of ground stations and data processing centers.

This Development Plan for the accomplishment of the ARS was prepared by the Missile Systems Division, Lockheed Aircraft Corporation and its subcontractors, CBS Laboratories and Eastman Kodak Company. The specifications for the system were determined in the course of a one-year study now being conducted for the United States Air Force ~~under contract AF 33(616)-3105.~~ The plan is presented in two parts; Volume I, System Plan, and Volume II, Subsystem Plan. The subsystems are described in separate books, Volume II-A through II-L.

2. APPROACH

The Ascent Guidance and Control System has a three phase operating cycle:

Phase I:	Booster Power
Phase II:	Transition
Phase III:	Orbit Stage Vehicle Boost

I. The Booster Vehicles will be derived from the WS 107 (ICBM) program as GFE. Since it is assumed that these vehicles, in order to meet WS 107 requirements will have guidance and control capability compatible with the ARS requirements, the subsystem described below does not include the booster guidance characteristics.

II. Using input data of altitude, velocity, and flight path angle from the first stage guidance, differential corrections to a pre-calculated trajectory are computed. These corrections provide the following inputs to the OSV control system:

1. Velocity to be added at apogee (V_G).
2. Change in vehicle attitude required ($\delta\phi$).
3. Time to start engines ($t_a = t_b$).

III. The OSV flight control system, consisting of a highly stable autopilot, accepts these inputs to determine attitude errors, total impulse to be added at apogee, and approximate burning time. The control system, or autopilot, obtains its reference from low drift gyros, an integrating accelerometer mounted on the thrust axis of the vehicle, and a clock. It controls rocket engine starting and shut off, and steers the vehicle using two gimballed control engines.

The attitude control is obtained from the interaction of gyroscopic and differential gravity torques acting on the vehicle. These torques, although small, provide a stable attitude with the vehicle pitch axis (coincident with the vehicle internal angular momentum vector) parallel to the orbital angular velocity vector and with the vehicle long axis (axis of minimum moment of inertia) oriented parallel to the local gravity vector. A feedback control system is employed to sense attitude deviations and rates and to apply counter torques and damping torques as needed.

A block diagram of the control system is shown in Fig. 1.

Major problem areas identified with the guidance and control system arise in connection with environmental control of the sensing instruments and in obtaining reliable and precision operation of instrumentation over the complete range of dynamic operating loads. Precision gyros, rate gyros, and integrating accelerometers will probably establish the limit of system performance.

3. MAJOR TASKS

a. Transition Guidance Task

Transition guidance consists of a means to accept data from the booster stage guidance and to compute corrections to the precalculated trajectory so that the control program may be altered to account for deviations from the exit trajectory which was predicted before the launching.

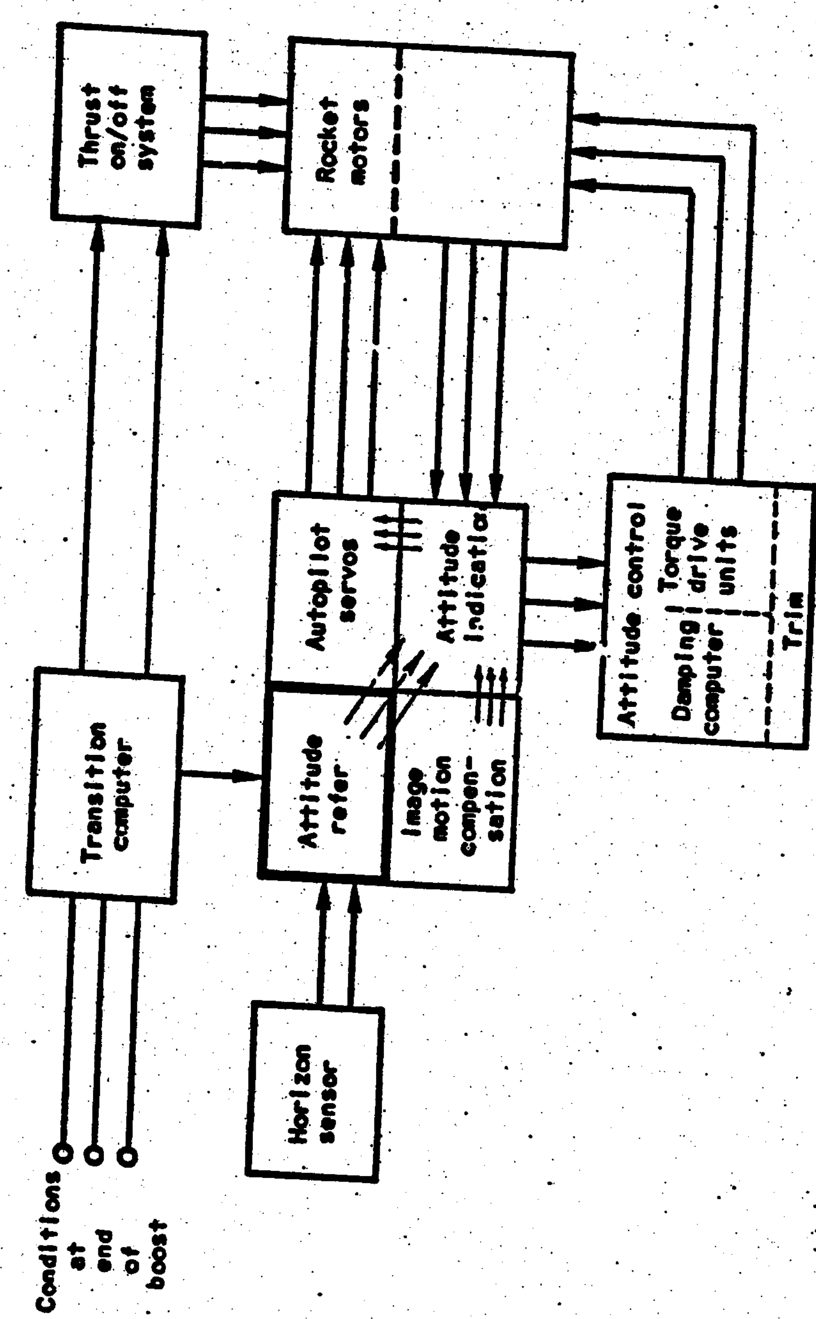

Fig. 1 Block Diagram of Guidance and Control System

D-Tab 1, p 4

A radio guidance data receiver is required if a closed-loop, radio-inertial guidance is employed. This receiver must be capable of accepting data regarding altitude, velocity, and flight path angle at booster burnout. These data, referred to a central inertial reference frame, may then be used in the transition computer to provide guidance commands to the vehicle for the orbit stage boost.

The transition computer accepts input data from the radio inertial guidance system and computes deviations from a precalculated trajectory. These are deviations of time to apogee, t_a, and of velocity, V_G, and thrust axis alignment, ϕ, at apogee using the deviations of the first-stage trajectory which are derived from a radio link as input data. In addition, the transition computer determines the time which must elapse before the rocket engines are started.

Corrections are computed from the linear expressions

$$\Delta t_a = \frac{\partial t_a}{\partial \gamma_o} \Delta \gamma_o - \frac{\partial t_a}{\partial V_o} \Delta V_o + \frac{\partial t_a}{\partial \lambda_o} \Delta \lambda_o$$

$$\Delta V_G = \frac{\partial V_G}{\partial \gamma_o} \Delta \gamma_o - \frac{\partial V_G}{\partial V_o} \Delta V_o + \frac{\partial V_G}{\partial \lambda_o} \Delta \lambda_o$$

$$\Delta \phi_T = \frac{\partial \phi_T}{\partial \gamma_o} \Delta \gamma_o + \frac{\partial \phi_T}{\partial V_o} \Delta V_o + \frac{\partial \phi_T}{\partial \lambda_o} \Delta \lambda_o$$

The corrected trajectory parameters are calculated from the design (programmed) values as follows

$$t_a = t_a^{\Delta} - \Delta t_a$$

$$V_G = V_G^{\Delta} + \Delta V_G$$

$$\phi_T = \phi_T^{\Delta} + \Delta \phi_T$$

The time to initiate burning is adjusted from the corrected time to apogee as follows:

$$t_a - t_f = t_a - \left(t_f^{\Delta} - \frac{V_G}{\delta} \right)$$

where δ represents the nominal acceleration due to thrust at the time of burnout.

The computed deviations are employed as differential corrections to a program which is preset into the OSV before launching. Since the transition computer is used to compute corrections, its precision is not critical; guidance error analysis indicates that an error of about 5 percent is acceptable.

b. Thrust On-Off Task

The Thrust On-Off system functions to initiate the OSV rocket engines, (7.5K main engine and two 150-pound control engines) and to shut them down again when the proper impulse has been added.

The thrust initiation signal is derived from a clock which is set to measure elapsed time from booster shutoff. This initiation cycle is set to initiate the control engines first before initiating the 7.5K main engines with a time delay of approximately two seconds in order to reduce the error in thrust axis misalignment before initiation of the large engines and to ensure against "pin-wheeling" of the OSV.

Engine shutdown commands are derived from an integrating accelerometer so that when a measured impulse has been given to the vehicle the engines are shutdown. The cycling is such that the control engines are shutdown after the main engine to permit stabilizing the vehicle during engine shutdown and the braking of rotary machinery in the propulsion unit.

The engine shutoff system incorporates an accelerometer-integrator combination for airborne velocity measurements. Since gravity accelerations have been accounted for in the reference program, it is not necessary to provide for gravity correction of the airborne accelerometer except as burning time and thrust application angle vary. Thrust application will be in the horizontal direction and no gravity correction need be applied to the system. A precision accelerometer is mounted to the missile frame with its sensitive axis aligned with the vehicle thrust axis. When the integrating accelerometer indicates that precisely the proper impulse has been added the accelerometer will provide the command for engine shutdown. An integrator-accelerometer unit having a precision of 1 part in 1000 of full scale will contribute about 8 feet per second to the velocity error.

No vernier engine cycle is contemplated since it appears that the 7.5K thrust engines may be shutdown with a small uncertainty in residual impulse. However, the engine shutdown cycle will provide for shutdown of control engines after decay of the thrust from the large engines.

c. Attitude Reference Unit Task

The CSV requires an accurate attitude reference system in order to ensure that the thrust applied to the vehicle during the burning of the orbit stage engine does not contribute an excessive vertical component to the velocity at apogee. The attitude reference also serves as a roll-yaw reference in order to decouple the pitch and yaw motions of the vehicle and to provide an azimuth, or heading, reference.

In addition, the attitude reference unit is employed in the instrumentation of an attitude control system for the coast phase and also the orbital phase of the mission.

The attitude reference unit will consist of a gyro stabilized platform and three rate gyros to sense vehicle body axis rates. This unit provides attitude and attitude rate information as required by the control system in its various modes of operation.

Guidance error calculations (see Appendix to this volume) indicate that if gyro drift errors may be taken as proportional to the acceleration applied, a 1/2 degree per hour gyro drift platform will be satisfactory for ascent guidance purposes. This quality of gyro is unsatisfactory, however, if drift rates are taken as proportional to the square of acceleration, i.e., drifts due to anisoelastic effects. In the latter case, a laboratory drift rate of about 0.1 degree per hour is required.

The attitude control system may employ the gyro stabilized platform; the exact configuration used will depend upon the results of analysis and study which are in progress. The long-term operation of an attitude control and indication device of the nature required will not permit drift rates of 0.5 degree per hour, or even 0.1 degree per hour, to go uncorrected.

Due to the natural torques on the pitch wheel, the roll and yaw axes of the vehicle are constrained to oscillate (or mutate) in a coupled motion about the desired stable attitude. This dynamic stability obviates the need for roll and yaw position references. A single rate gyro that senses vehicle roll rate is sufficient to provide damping torques about both the roll and yaw axes. A fore and aft looking horizon sensor may then be employed to correct drifts of the pitch gyro. In this manner a stable indication of attitude, free from troublesome drifts, may be established in the orbiting vehicle.

Preliminary proposals have been received from two vendors who propose to provide a stabilized platform of about 25-pounds weight having drifts of 0.5 degree per hour and less. These platforms have a major dimension of approximately 12 inches and are designed to meet environmental and life specifications which are compatible with the requirements of the Orbiting Test Vehicle.

d. OSV Autopilot Task

Thrust will be applied to the OSV in a direction parallel to the horizontal plane at the apogee. This thrust will be applied for about 30 seconds prior to apogee so that a measured increase is made to the vehicle horizontal velocity while no vertical velocity component is added. The vehicle heading is established to provide the proper value of the maximum latitude for the orbit.

The OSV autopilot provides the dynamic control of thrust direction through deflections of two gimballed 150-pound thrust control engines. This control is required to maintain a stable vehicle attitude during the OSV boost stage. Since this unit functions at very high