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1536
VOL. II
PART D
C-3-P (C.E-5)

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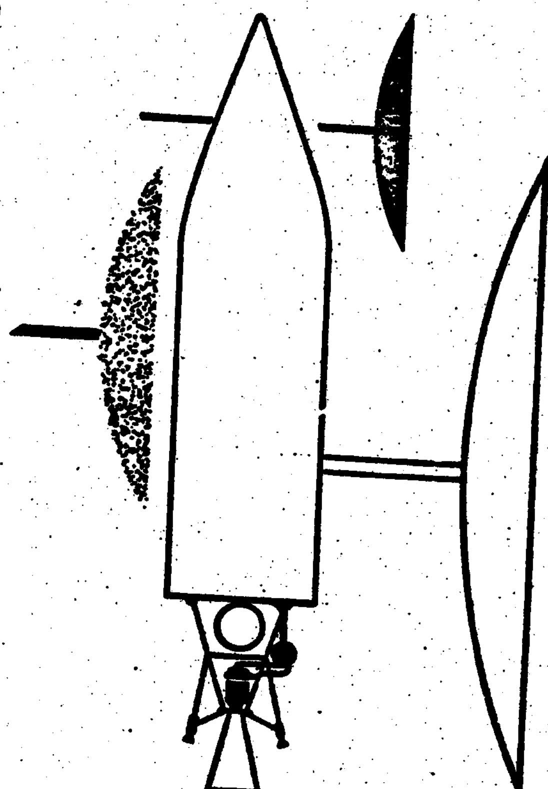

Pied Piper DEVELOPMENT PLAN

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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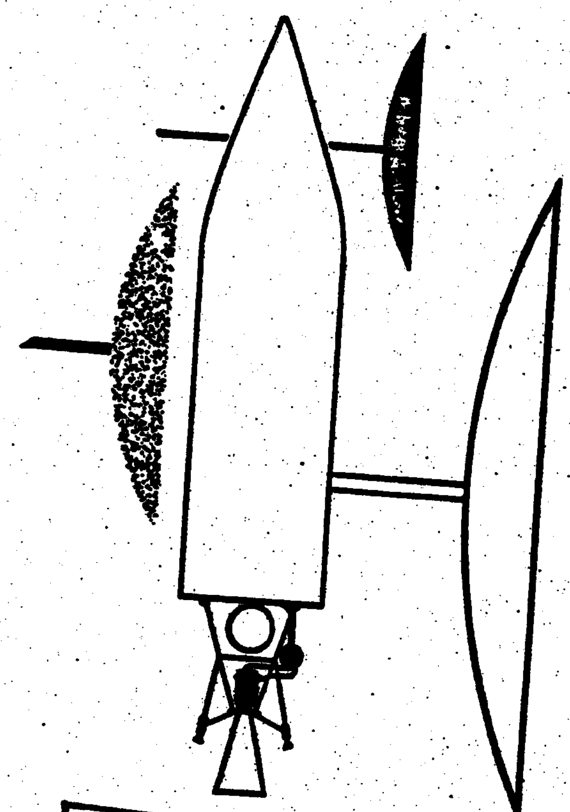

Pied Piper DEVELOPMENT PLAN

VOL. II SUB-SYSTEM PLAN

D. Guidance and Control

LOCKHEED AIRCRAFT CORPORATION
MISSILE SYSTEMS DIVISION
VAN NUYS, CALIFORNIA

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)48
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	SM	CN	IC & P	X.	L.	C.

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) (PIED PIPER)	S.	1115
	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.

1. PROJECT TITLE	2. SECURITY OF PROJECT	3. PROJECT NUMBER
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	4.	5. REPORT DATE 1 March 1956

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 or 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

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.	111.
(PIED PIPER)	4.	5. REPORT DATE 1 March 195

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.

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) (PIED PIPER)	S.	1115
	4.	5. REPORT DATE 1 March 1956

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

MSD 1536

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 ($t_s = t_b$).

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.

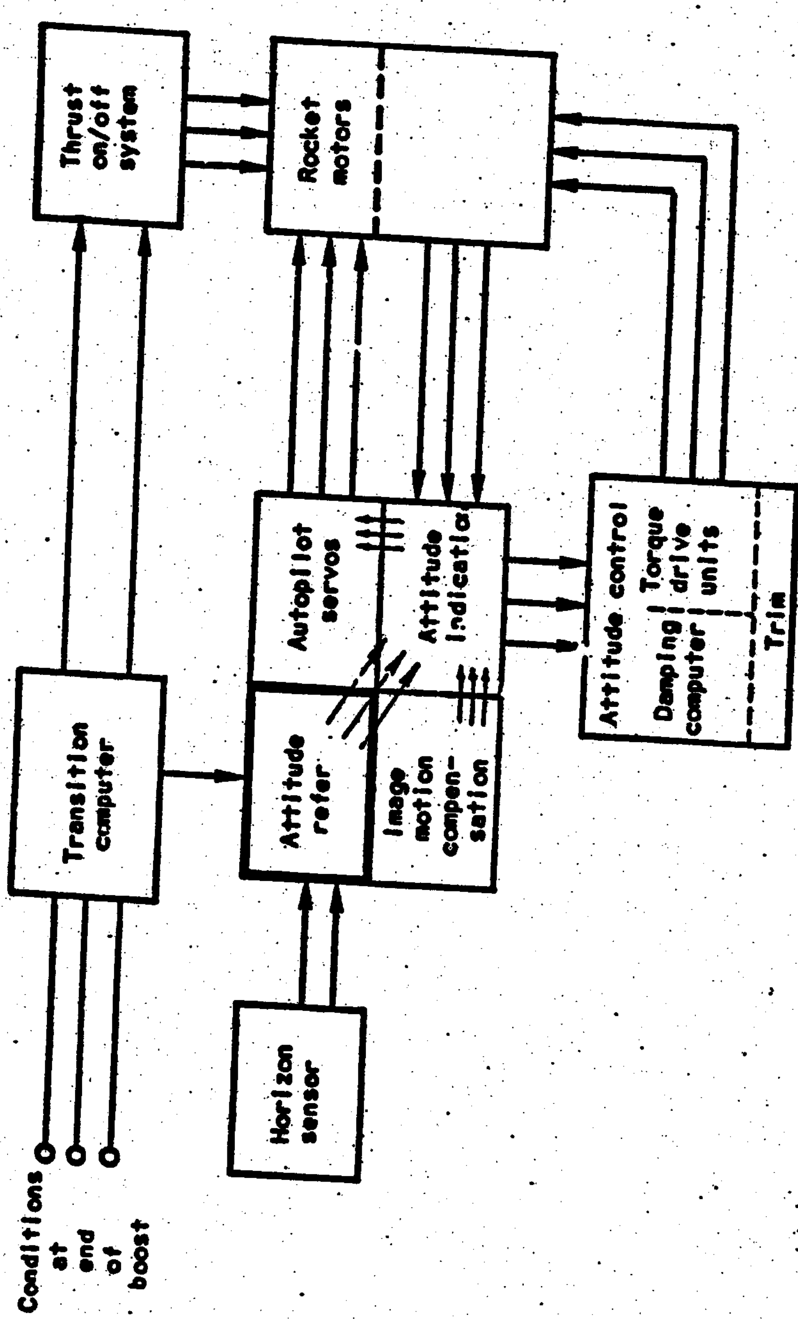


Fig. 1 Block Diagram of Guidance and Control System

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

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

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

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

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 altitude and after the OSV has already gained a high velocity, the primary requirement on it is that it be capable of providing stable flight control. Through reference to the attitude reference unit, the autopilot receives error signals required to correct initial errors in a short time, and to ensure that thrust is applied in the proper direction to avoid large residual vertical velocity components at the end of the boost stage.

An analysis of this unit is given in the appendix.

e. Attitude Control System

Attitude control of the OSV must be exercised whenever the engines are not operating. During the coast, or transition, phase of flight the residual angular impulse due to separation must be removed and the attitude must be stabilized for proper thrust orientation prior to initiation of the OSV boost.

During orbiting flight the vehicle attitude must be controlled so that payload elements will be aligned properly for reconnaissance purposes. The directions of lines of sight, antenna axes, etc., must be controllable and, in some cases, they must be known within accurate limits.

There are many requirements placed against the attitude control system; the most stringent ones arise because of the image motion stabilization requirements. These are tabulated below for an image blurring of about 30 feet.

Exposure Time (sec)	Pitch (rad/sec)	Roll (rad/sec)	Yaw (2-in. film width) (rad/sec)
0.1	1.6×10^{-4}	1.6×10^{-4}	1.9×10^{-3}
0.01	1.6×10^{-3}	1.6×10^{-3}	1.9×10^{-2}
0.001	1.6×10^{-2}	1.6×10^{-2}	1.9×10^{-1}

The attitude control during coast phase of flight will be accomplished by torques applied to inertia wheels. These torques will be in response to attitude and attitude rate signals from the attitude reference unit. The initial settling into the stable attitude for orbiting will be accomplished by the same means.

The orbital attitude control system depends upon the stable attitude of an elongated vehicle which arises from the torques due to differential gravity and the interaction of the vehicle orbital angular momentum with a bias angular momentum which is oriented along the vehicle pitch axis.

While the torques described provide the orbiting vehicle with a unique stable attitude the effects of external and internal disturbing torques must be eliminated through a damping system. This damping system will employ torques applied through inertia wheels. Since the use of inertia wheels amounts to transferring the angular momentum of the vehicle to the wheels, this system is subject to saturation as a result of long-term application of a bias torque, e.g., friction in bearings of rotating machinery on the vehicle. Accordingly, provisions will be made for the application of damping and control torques to the vehicle from another source periodically, e.g., the exhaust from a chemical APU, while the inertia wheels are trimmed.

f. Image Motion Compensation Task

The requirements for image motion compensation have been cited in part e above. The stability of the attitude control system directly affects the need for image motion compensation and, at present, analysis of this system indicates that all IMC requirements can be accomplished in the camera mounts.

The visual reconnaissance camera is a strip or continuous film type and, therefore, image motion compensation in the velocity direction is equivalent to adjusting the rate of film motion past the camera slit.

Body rotations may cause blurring of the image formed by ground objects. The camera optics can be designed to include a mirror which is gimballed and tilted from signals derived from the attitude reference unit in such a way as to eliminate image motions due to body oscillations.

Since the vehicle yaw axis and the line of sight are very nearly coincident, the effects of yaw oscillations may be removed by a rotation of the camera about its optic axis.

C. Attitude Indication

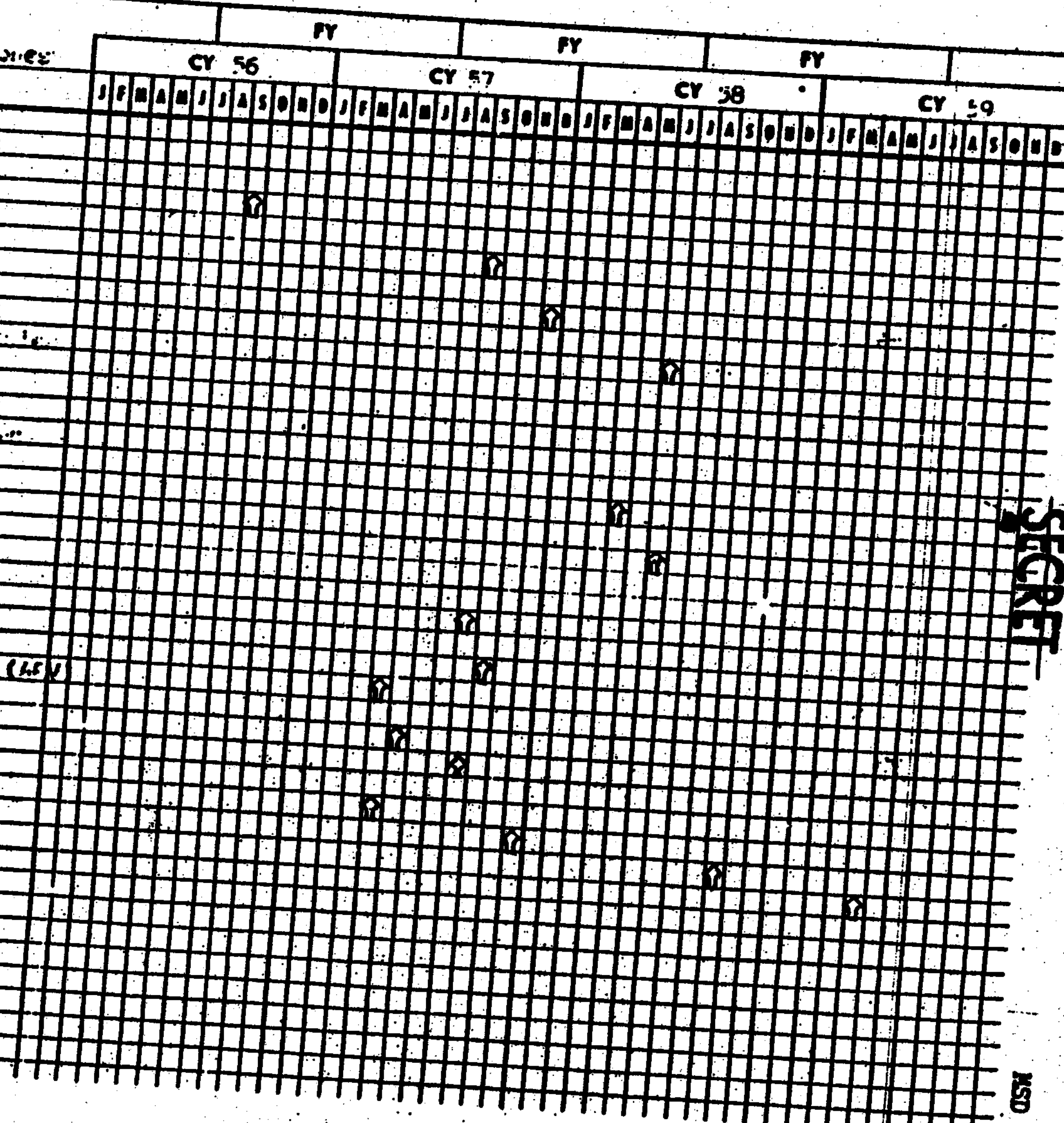
An indication of the instantaneous attitude of the vehicle is necessary in order to correlate reconnaissance data with geographical location. Direction finders etc. will be referenced to the vehicle and, since the geographical orientation of the vehicle is known as a function of time, these data can be converted to geographical position.

An attitude indication is derived from the attitude reference unit. The gyro platform represents a satisfactory attitude reference except that long term drifts degrade its accuracy. These drifts may be eliminated, or maintained within bounds by the application of the dynamic constraints on the vehicle and a horizon sensing element. The long term drifts of the gyro and the short term, noise, character of the output of a horizon senser are used to complement the operation of each.

Subsystem D - GUIDANCE & CONTROL

Tab 2 Summary - Subsystem Milestones

1 Flight Test Schedules
2 (For Reference)
3
4 System Test Vehicles
5
6 Orbit Stage Test Vehicles
7
8 Non-Orbiting Test Vehicles
9
10 Orbiting and Pioneer Test Vehicles
11
12 Advanced Recon. Vehicle
13
14 Guidance & Control System Schedules
15
16 Transition Computer
17
18 Orbit Correction
19
20 Thrust On-Off System
21 STV
22 OSV
23 Attitude Reference Unit (OSV)
24 Attitude Reference Unit (STV) & (ARV)
25 Autopilot
26 STV
27 OSV & ARV
28 Attitude Control System
29 STV
30 OSV
31 INC STV
32 (if needed) OSV
33 ARV
34
35
36
37
38
39
40



Subsystem D - GUIDANCE & CONTROL

Tab 2 Summary - Subsystem Milestones (Continued)

FY | FY | FY

CY 60 | CY 61 | CY 62 | CY 63

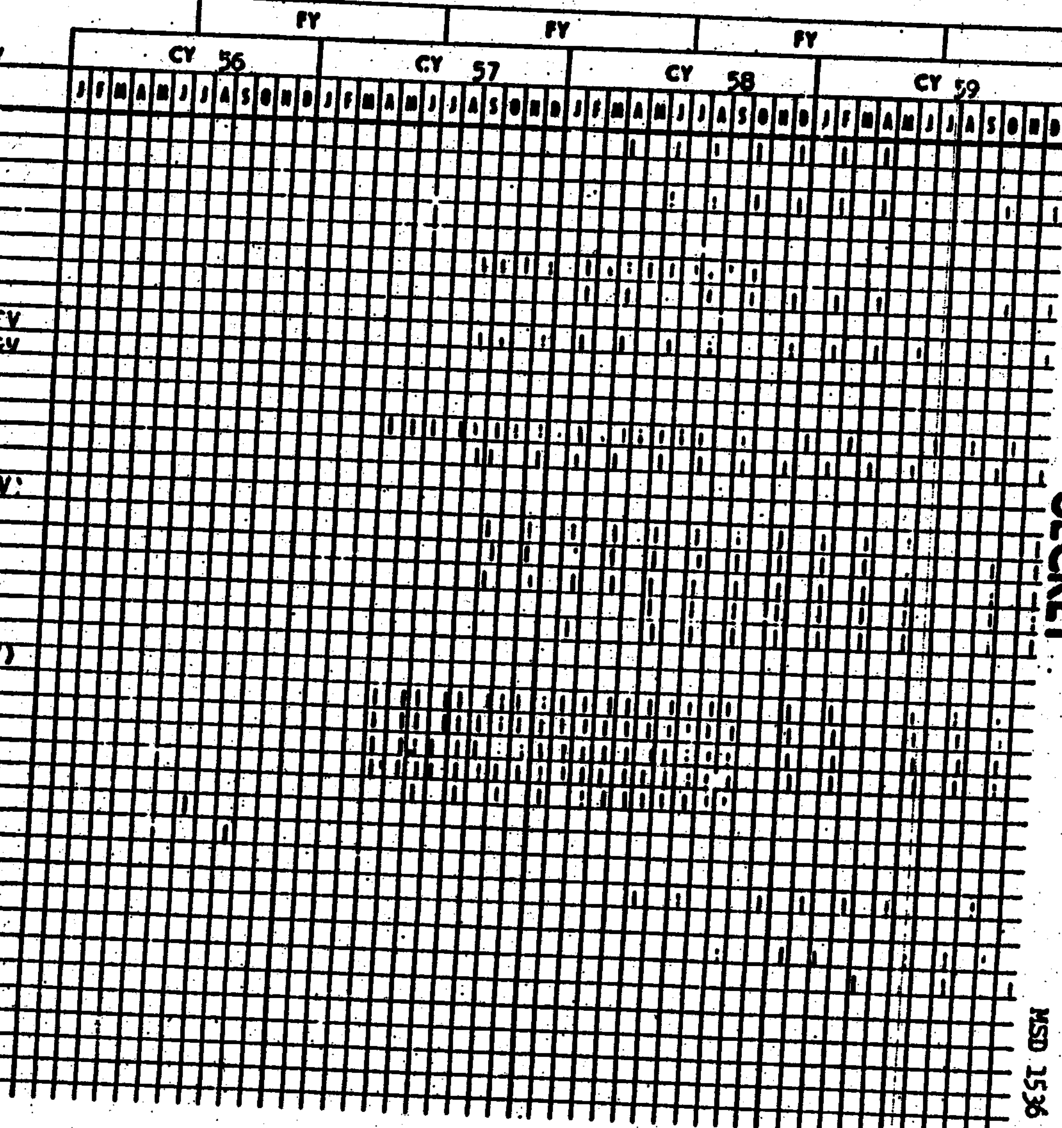
No.	Item
1	Flight Test Schedules
2	
3	System Test Vehicle
4	
5	Orbit Stage Test Vehicle
6	
7	Non-Orbiting Test Vehicles
8	
9	Orbiting and Pioneer Test Vehicle
10	
11	Advanced Recon. Vehicle (ARV)
12	
13	
14	
15	Guidance & Control Schedules
16	
17	Transition Computer
18	
19	Orbit Correction
20	
21	Thrust On-Off
22	STV
23	OSV & ARV
24	Attitude Reference Unit (OSV)
25	Attitude Reference Unit (ARV)
26	(STV)
27	Autopilot
28	STV
29	OSV & ARV
30	Attitude Control System
31	STV
32	OSV
33	ARV
34	Complete System
35	Image Motion Compensation STV
36	OSV
37	ARV
38	
39	
40	

Revised Form 103

Subsystem C - GUIDANCE & CONTROL

Tab 2 Summary - Hardware Delivery

No.	Item	
1	Transition Computer	
2		
3	Orbit Correction	
4		
5	Thrust on/off System	
6	STV	
7	CSV	
8		
9	Attitude Ref. Unit	CSV
10		CSV
11		[illegible]
12	Autopilot	
13	STV	
14	OSV	
15		
16	Attitude Control System	([illegible])
17	Pitch Torque Drive Unit	
18	Yaw Torque Drive Unit	
19	Damping Computer	
20	Torque Wheel Trim System	
21	Horizon Sensor	
22		
23	Attitude Control System	([illegible])
24	Gyro Platform	
25	Pitch Torque Drive Unit	
26	Yaw Torque Drive Unit	
27	Damping Computer	
28	Horizon Sensor	
29	Experimental Systems	
30	Bird-Cage Experiment	
31		
32	Complete System	
33		
34	Image Motion Compensation	([illegible])
35		([illegible])
36		([illegible])
37		
38		
39		
40		



Revised Form 103

Subsystem D - GUIDANCE & CONTROL

Tab 2 Summary - Hardware Delivery
(Continued)

Line	Item	Vehicle	CY 60 (J F M A M J J A S O N D)	CY 61 (J F M A M J J A S O N D)	CY 62 (J F M A M J J A S O N D)	CY 63 (J F M A M J J A S O N D)
1	Transition Computer					
2		STV				
3		OSV	[illegible]			
4		ARV	[illegible]	[illegible]	[illegible]	
5	Thrust on/off		[illegible]	[illegible]	[illegible]	[illegible]
6	STV					
7	OSV		[illegible]	[illegible]	[illegible]	
8	ARV		[illegible]	[illegible]	[illegible]	
9			[illegible]	[illegible]	[illegible]	
10	Attitude Ref. Unit		[illegible]	[illegible]	[illegible]	[illegible]
11		OSV	[illegible]			
12		ARV	[illegible]	[illegible]	[illegible]	
13	Autopilot		[illegible]	[illegible]	[illegible]	[illegible]
14	STV					
15	OSV		[illegible]	[illegible]	[illegible]	
16	ARV		[illegible]	[illegible]	[illegible]	
17			[illegible]	[illegible]	[illegible]	
18	Attitude Control System	STV	[illegible]	[illegible]	[illegible]	[illegible]
19	Gyro Platform					
20	Pitch Torque Drive Unit		[illegible]	[illegible]		
21	Yaw Torque Drive Unit		[illegible]	[illegible]	[illegible]	
22	Damping Computer		[illegible]	[illegible]	[illegible]	
23	Horizon Sensor		[illegible]	[illegible]	[illegible]	
24	Experimental Systems			[illegible]	[illegible]	
25	Bird Cage Experiment					
26						
27	Attitude Control System (CSV & ARV)					
28	Pitch Torque Drive Unit					
29	Yaw Torque Drive Unit		[illegible]	[illegible]	[illegible]	
30	Damping Computer		[illegible]	[illegible]	[illegible]	
31	Torque Wheel Trim System		[illegible]	[illegible]	[illegible]	
32	Horizon Sensor		[illegible]	[illegible]	[illegible]	
33			[illegible]	[illegible]	[illegible]	
34	Complete System		[illegible]	[illegible]	[illegible]	
35			[illegible]	[illegible]	[illegible]	
36	Image Motion Compensation		[illegible]	[illegible]	[illegible]	
37		STV	[illegible]	[illegible]	[illegible]	
38		CSV	[illegible]	[illegible]	[illegible]	
39		ARV	[illegible]	[illegible]	[illegible]	
40			[illegible]	[illegible]	[illegible]	

Revised Form 103

Subsystem 9 - GUIDANCE & CONTROL

Tab 2 Summary - Subsystem Test Schedule

No.	Item	Vehicle
1	Transition Computer	
2		
3	Orbit Correction	
4		
5	Thrust on/off System	
6	STV	
7	OSV	
8	ARV	
9		
10	Attitude Ref. Unit	
11		OSV
12		ARV
13	Autopilot	
14		
15		STV
16		OSV
17	Attitude Control System	STV
18	Gyro Platform	
19	Pitch Torque Drive Unit	
20	Yaw Torque Drive Unit	
21	Damping Computer	
22	Horizon Sensor	
23	Experimental Systems	
24	Bird Cage Experiment	
25		
26	Attitude Control System	(OSV &
27		ARV)
28	Pitch Torque Drive Unit	
29	Yaw Torque Drive Unit	
30	Damping Computer	
31	Torque Wheel Trim System	
32	Horizon Sensor	
33		
34	Complete System	
35		
36	Image Motion Compensation	STV
37		OSV
38		ARV
39		
40		

FY		FY		FY	
CY 56	CY 57	CY 58	CY 59		
J F M A M J J A S O N D	J F M A M J J A S O N D	J F M A M J J A S O N D	J F M A M J J A S O N D		

Revised Form 103

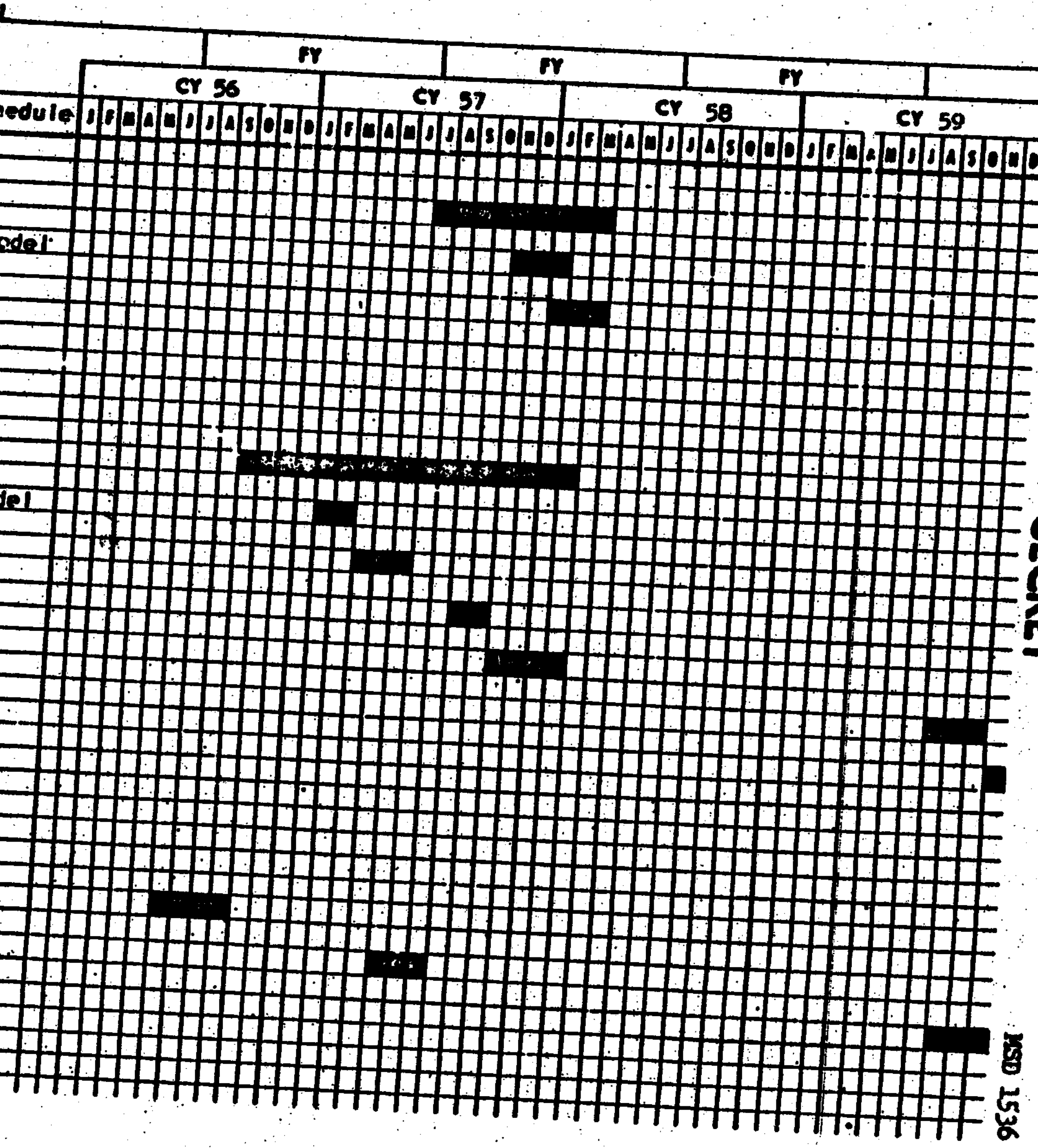
SECRET

MISSILE SYSTEMS DIVISION

Subsystem D - GUIDANCE & CONTROL

Tab 2 Summary - Research and Development Schedule

1
2 Transition Computer-Task
3 A. Research and Development
4
5 B. Fabrication of Develop. Model
6
7 C. Tests of Dev. Mod.
8
9
10
11
12
13 Thrust on/off Control-Task
14 A. Research and Development
15
16 B. Fabrication of Develop. Model
17
18 C. Tests of Dev. Mod.
19
20 D. Modification for OSV
21
22 E. Tests of OSV Mod.
23
24 F. Modification for ARV
25
26 G. Tests of ARV Mod.
27
28
29
30
31
32 Attitude Reference-Task
33 A. Requirement Study
34
35 B. Compliance Tests (OSV)
36
37 C. Compliance Tests (ARV)
38
39
40



Revised Form 103

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MSD 1536

D - T b 2, p 6
(original)

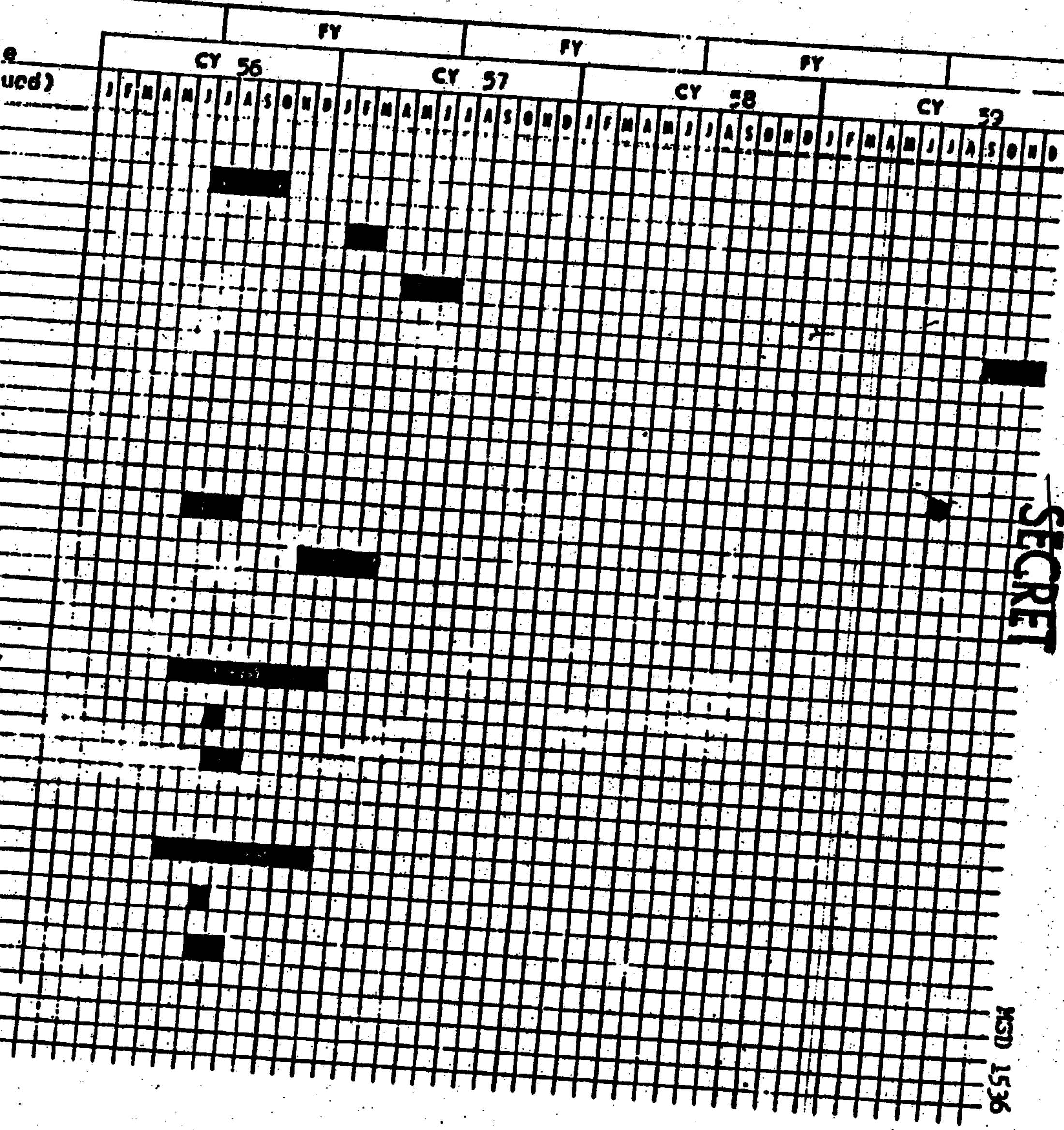
LOCKHEED AIRCRAFT CORPORATION

Subsystem D - GUIDANCE & CONTROL

Tab 2 Summary - R & D Schedule
(Continued)

Autopilot-Task
A. Requirement Study
B. Compliance Tests (STV)
C. Compliance Tests (OSV)
D. Compliance Tests (ARV)

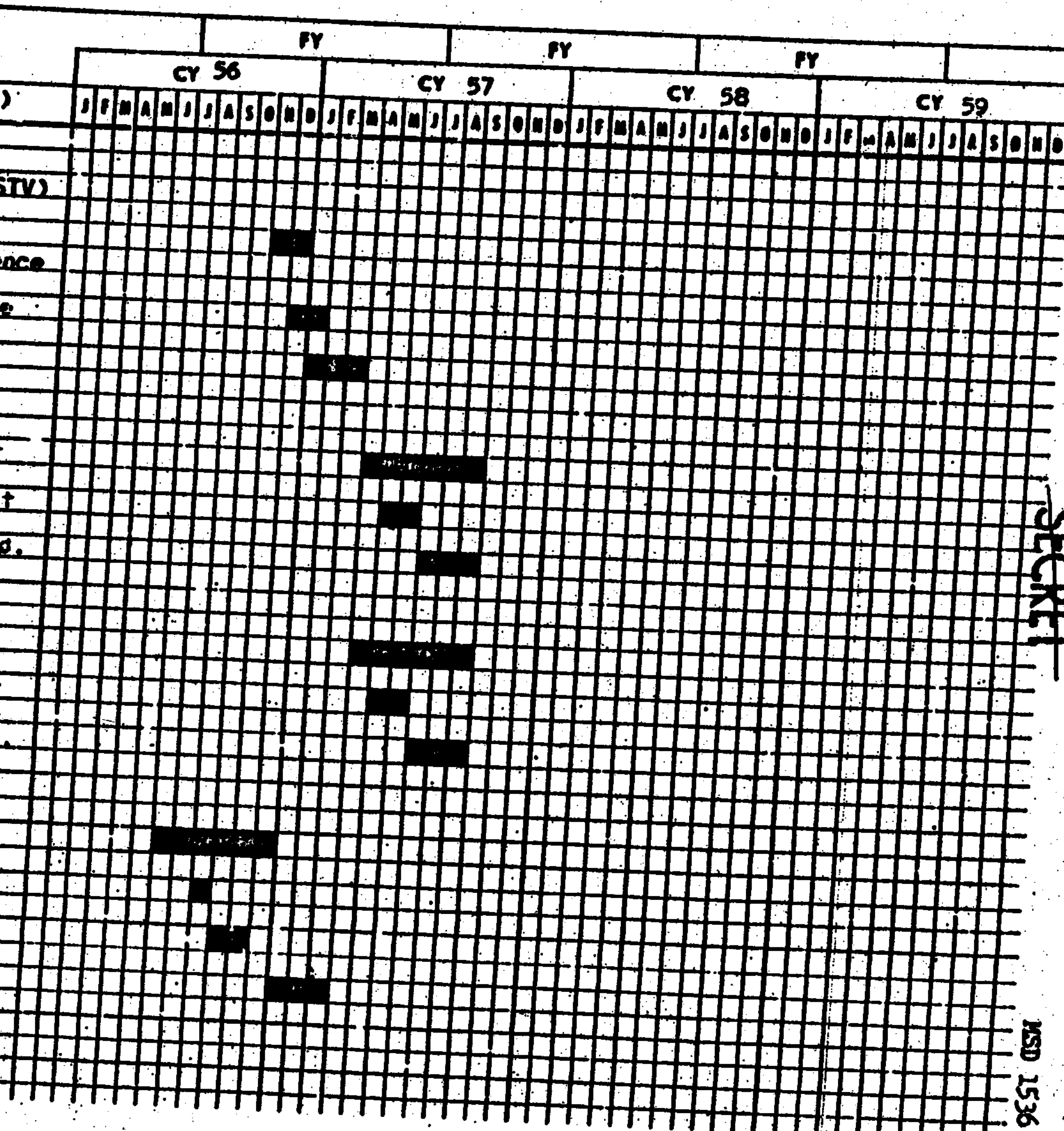
Attitude Control-Task (STV)
A. Gyro Platform
1. Requirement Study
2. Compliance Tests
B. Torque Drive Units (Pitch and Yaw) (STV)
1. Research and Development
2. Fabrication of Dev. Mod.
3. Tests of Dev. Mod.
C. Damping Computer
1. Research and Development
2. Fabrication of Dev. Mod.
3. Tests of Dev. Mod.



Revised Form 103

Subsystem D - GUIDANCE & CONTROL

Tab 2 Summary - R & D Schedule (Continued)



D. Complete Attitude Control System (STV)

1. Functional Tests without Attitude Reference

2. With Attitude Reference

3. Environmental Tests

E. Torque Drive Units (OSV)

1. Research and Development

2. Modifications to STV Unit

3. Tests of Modification Mod.

F. Damping Computer (OSV)

1. Research and Development

2. Modifications to STV Unit

3. Tests of Modification Mod.

G. Horizon Sensor (STV)

1. Research and Development

2. Fabrication of Dev. Mod.

3. Tests of Dev. Mod.

4. Environmental Tests

Revised Form 103

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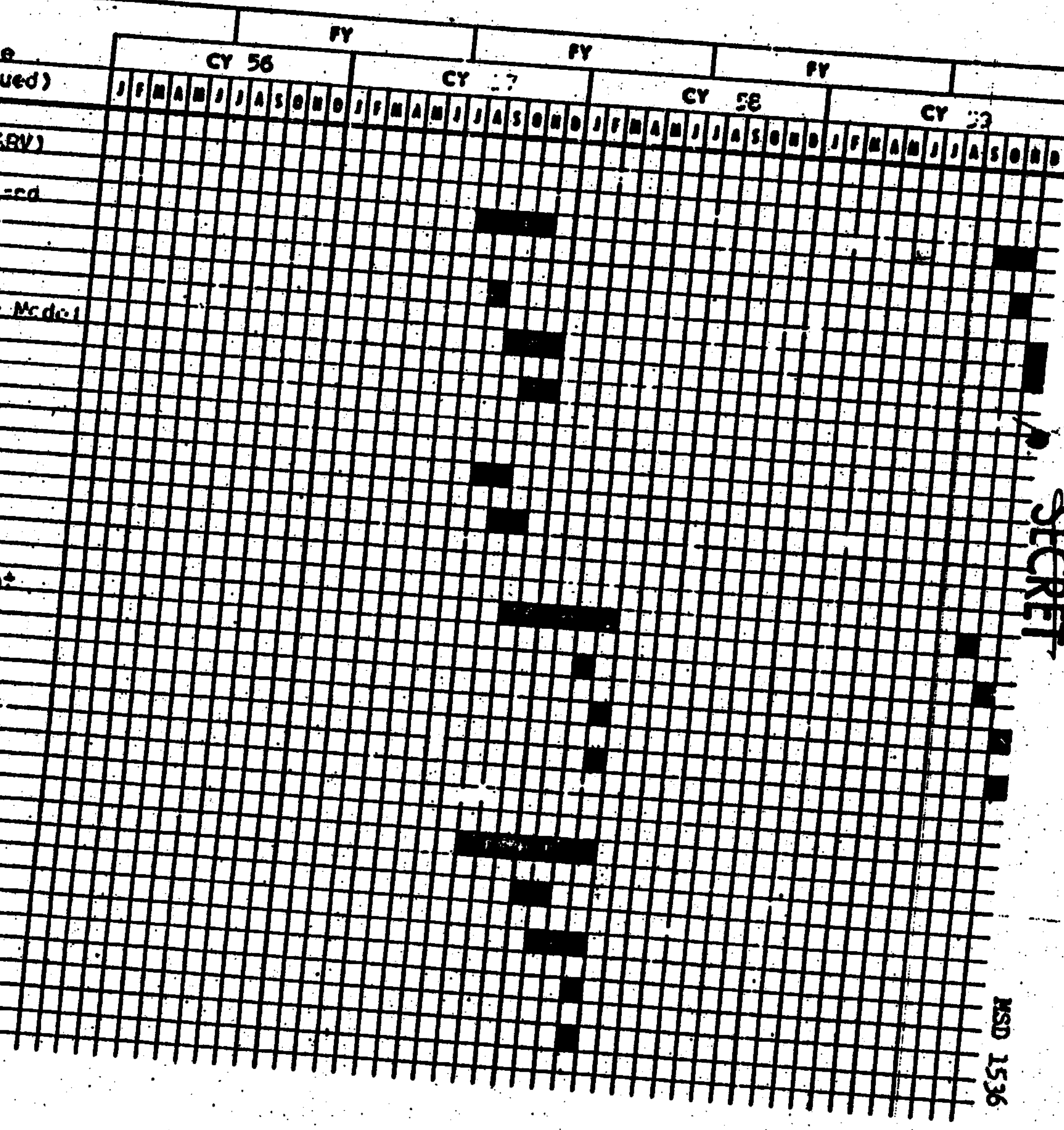
Subsystem D - GUIDANCE & CONTROL

Tab 2 Summary - R & D Schedule
(Continued)

1
2 H. Horizon Sensor (OSV & ARV)
3
4 1. Additional Research and
5 Development
6
7 2. Modifications
8
9 3. Tests of Modification Model
10
11 4. Environmental Tests
12
13 I. Complete System (OSV)
14
15 1. Functional Tests
16
17 2. Environmental Tests
18
19 J. Torque Wheel Trim System
20
21 1. Research and Development
22
23 2. Tests of Dev. Mod.
24
25 3. Functional Tests
26
27 4. Environmental Tests
28
29 Image Motion Compensation-Task
30
31 A. Research and Development
32
33 B. Fabrication of Dev. Mod.
34
35 C. Tests of Dev. Mod.
36
37 D. Functional Tests
38
39 E. Environmental Tests
40



Revised Form 103

Subsystem D - GUIDANCE & CONTROL

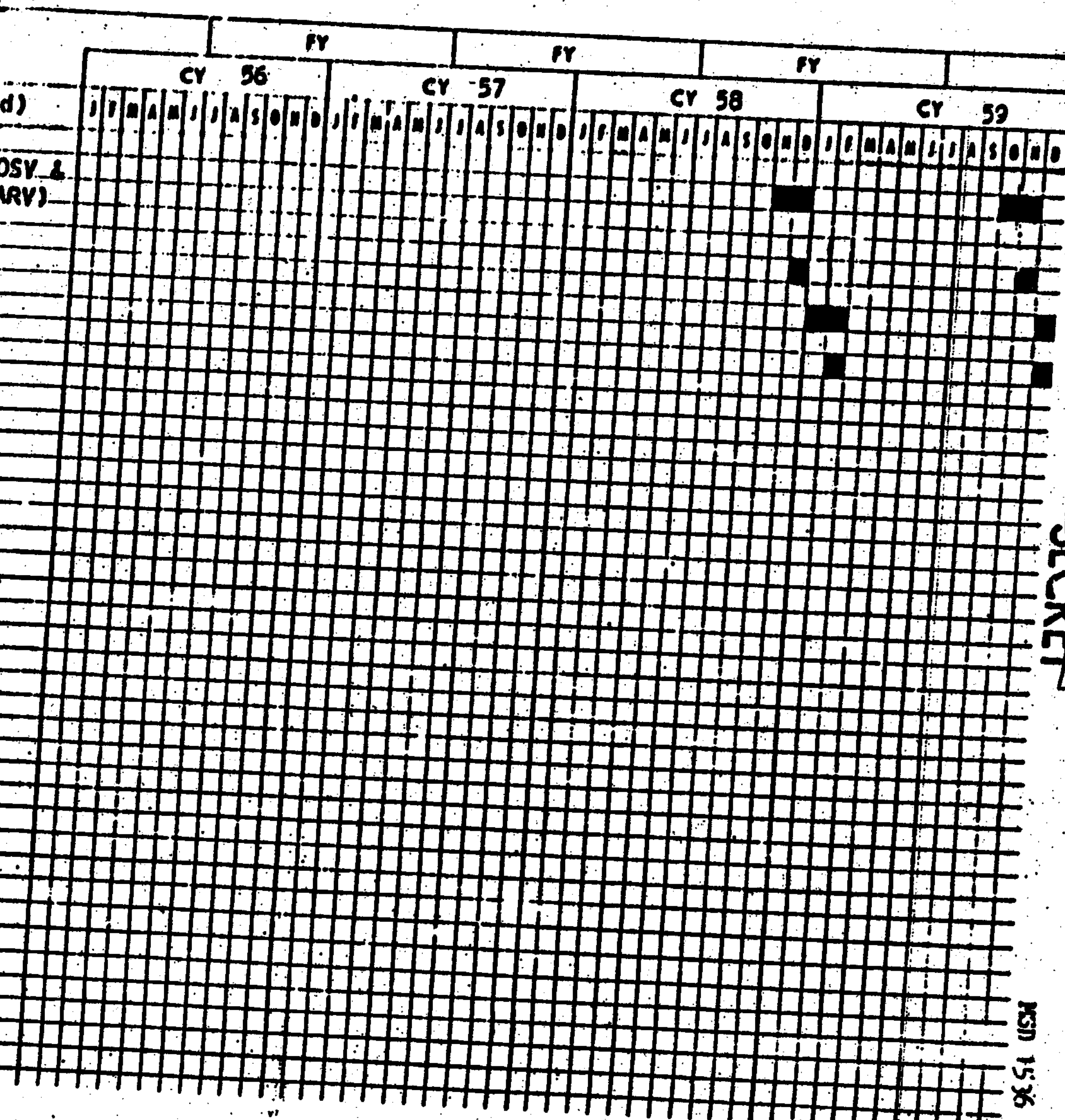
Tab 2. Summary - R & D Schedule (Continued)

F. Research and Development (OSV & ARV)

G. Modifications to STV Unit

H. Tests of Modif. Unit

I. Functional Tests



SECRET

1.

R & D TEST ANNEX

☐ SYSTEM ☐ PROJECT ☒ TASK ☐ OTHER

2. REPORTS CONTROL SYMBOL	
PAGE 1 OF 1 PAGES	
3. DATE	1 F[illegible] 1[illegible]
6. NUMBER	

4. TITLE: TRANSITION COMPUTER (DEVELOPMENT TESTS)

5. INITIAL ☐ CHANGE

7. RESP CENTER	8. PROJECT OFFICER	9. SUPPORTS (Sys or Proj)	10. CONTRACTOR	11. CONTR NR	12. PRIORITY AND PREC	13. SECURITY
		ARS	LOCK[illegible] MSD			[illegible]

14. ITEM NUMBER	15. TEST ITEM	16. TEST DESCRIPTION	17. TEST AGENCY AND SITE	18. TEST ITEM AVAILABLE	19. RDO TEST COMPL DATE
1.	Computer (Complete)	Functional tests with specified inputs	MSD Research Lab	Jun. '58	Feb. '58
2.	Computer (Complete)	Functional tests with simulated vehicle flight	MSD Research Lab	Jun. '58	Feb. '58
3.	Computer (Complete)	Compatibility tests with thrust on/off system	MSD Research Lab	Feb. '58	Mar. '58
4.	Computer (Complete)	Flight Test*	AFMTC	Mar. '58	Jul. '59

* About 8 units are required.

TEST CENTER APPROVAL

20. NAME	ORGANIZATION	
21. NAME	ORGANIZATION	DATE

RESPONSIBLE CENTER APPROVAL

22. NAME	ORGANIZATION	DATE
		DATE

ARDC FORM 105, 1 JUL 55 PREVIOUS EDITIONS OF THIS FORM ARE OBSOLETE.

MISSILE SYSTEMS DIVISION

SECRET

LOCKHEED AIRCRAFT CORPORATION

SECRET

R & D TEST ANNEX

☐ SYSTEM ☐ PROJECT ☒ TASK ☐ OTHER

Field	Entry
1.	
2. REPORTS CONTROL SYMBOL	
PAGE	1 OF 1 PAGES
3. DATE	1 [illegible] 195[illegible]
4. TITLE	TRANSITION COMPUTER (ENVIRONMENTAL TESTS)
5. INITIAL ☒ CHANGE	
6. NUMBER	
7. RESP CENTER	
8. PROJECT OFFICER	
9. SUPPORTS (Sys or Proj)	ARS
10. CONTRACTOR	LOCKH[illegible]
11. CONTR NR	
12. PRIORITY AND PREC	
13. SECURITY	[illegible]

14. ITEM NUMBER	15. TEST ITEM	16. TEST DESCRIPTION	17. TEST AGENCY AND SITE	18. TEST ITEM AVAILABLE	19. RQD TEST COMPL DATE
1.	Computer	Shock & vibration tests temperature tests	[illegible] Research L[illegible]	F[illegible]	

TEST CENTER APPROVAL

20. NAME	ORGANIZATION	
21. NAME	ORGANIZATION	DATE

RESPONSIBLE CENTER APPROVAL

22. NAME	ORGANIZATION	DATE

ARDC FORM 1 JUL 55 105 PREVIOUS EDITIONS OF THIS FORM ARE OBSOLETE.

SECRET

R & D TEST ANNEX

☐ SYSTEM ☐ PROJECT ☒ TASK ☐ OTHER

1.

2. REPORTS CONTROL SYMBOL

PAGE 1 OF 1 PAGES

3. DATE 1 February 1956

4. TITLE THRUST ON/OFF SYSTEM (DEVELOPMENT TESTS)

5. INITIAL ☐ CHANGE

6. NUMBER

7. RESP CENTER

8. PROJECT OFFICER

9. SUPPORTS (Sys or Proj) ARS

10. CONTRACTOR LOCKHEED MSD

11. CONTR NR

12. PRIORITY AND PREC

13. SECURITY ~~SECRET~~

14. T.I.L. NUMBER	15. TEST ITEM	16. TEST DESCRIPTION	17. TEST AGENCY AND SITE	18. TEST ITEM AVAILABLE	19. RDD TEST COMPL DATE
1.	System (Complete) (STV)	Function Tests with specified inputs	MSD Research Lab	Apr. '57	May '57
2.	System (Complete) (STV)	Functional Tests with transition computer and simulated vehicle flights.	MSD Research Lab	May '57	Jun. '57
3.	System (Complete) (STV)	Flight Tests *	AFMTC	Jun. '57	Oct. '58
4.	System (Complete) (OSV)	Functional Tests with modification from STV	MSD Research Lab	Nov. '57	Nov. '57
5.	System (Complete) (OSV)	Flight Tests **	MSD Research Lab	Feb. '58	Jun. '59

* About 14 units are required (STV)

** About 9 units are required (OSV)

TEST CENTER APPROVAL

20. NAME | ORGANIZATION

21. NAME | ORGANIZATION | DATE

RESPONSIBLE CENTER APPROVAL

22. NAME | ORGANIZATION | DATE

DATE

ARDC FORM 1 JUL 56 105 PREVIOUS EDITIONS OF THIS FORM ARE OBSOLETE.

~~SECRET~~

MSD 1536

1.

R & D TEST ANNEX

☐ SYSTEM ☐ PROJECT ☒ TASK ☐ OTHER

2. REPORTS CONTROL SYMBOL

PAGE 1 OF 1 PAGES

3. DATE: 1 February 1956

4. TITLE: THRUST ON/OFF SYSTEM (ENVIRONMENTAL TESTS)

5. INITIAL ☒ CHANGE

6. NUMBER

7. RESP CENTER

8. PROJECT OFFICER

9. SUPPORTS (Sys or Proj): ARS

10. CONTRACTOR: LOCKHEED MSD

11. CONTR NR

12. PRIORITY AND PREC

13. SECURITY: ~~SECRET~~

14. ITEM NUMBER	15. TEST ITEM	16. TEST DESCRIPTION	17. TEST AGENCY AND SITE	18. TEST ITEM AVAILABLE	19. RDD TEST COMPL DATE
1.	System Complete (STV)	Shock & Vibration Tests and Temperature Tests	MSD Research Lab	May '57	June '57
2.	System Complete (OSV)	Shock & Vibration Tests and Temperature Tests	MSD Research Lab	Jan. '58	Feb. '58

TEST CENTER APPROVAL

20. NAME	ORGANIZATION	DATE
21. NAME	ORGANIZATION	DATE

RESPONSIBLE CENTER APPROVAL

22. NAME	ORGANIZATION	DATE

ARDC FORM 1 JUL 55 105 PREVIOUS EDITIONS OF THIS FORM ARE OBSOLETE.

R & D TEST ANNEX

☐ SYSTEM ☐ PROJECT ☑ TASK ☐ OTHER

Field	Entry
1.	
2. Reports Control Symbol	
3. Page 1 of 1 Pages	
4. Title	ATTITUDE REFERENCE UNIT (STV) (ACCEPTANCE TESTS)
5. Date	1 February 195[illegible]
6. Number	
7. Resp Center	
8. Project Officer	
9. Supports (Sys or Proj)	ARS
10. Contractor	LOCKHEED MSD
11. Contr Nr	
12. Priority and Prec	
13. Security	[illegible]
Initial ☑ / Change	

14. Item Number	15. Test Item	16. Test Description	17. Test Agency and Site	18. Test Item Available	19. RDD Test Compl Date
1.	Reference Unit (STV)	Response Tests* to specified input signals	MSD Research Lab.	Jan. '57	Feb. '57
2.	Reference Unit (STV)	Vibration Tests and temperature tests.	MSD Research Lab.	Jan. '57	Feb. '57
3.	Reference Unit (STV)	Functional Tests with complete attitude control system. (Bird cage)	MSD Research Lab.	Feb. '57	Mar. '57
4.	Reference Unit (STV)	Functional tests with autopilot system (simulated vehicle)	MSD Research Lab.	Feb. '57	Mar. '57
5.	Reference Unit (STV)	Flight Tests**	AFMTC	Apr. '57	Oct. '58

* About 2 required for acceptance tests (Lab.)
** About 18 required for flight tests.

TEST CENTER APPROVAL

Name	Organization	Date
20. Name	Organization	
21. Name	Organization	Date

RESPONSIBLE CENTER APPROVAL

Name	Organization	Date
22. Name	Organization	Date

ARDC Form 105, 1 Jul 55 — PREVIOUS EDITIONS OF THIS FORM ARE OBSOLETE.

MSD 1536

R & D TEST ANNEX

☐ SYSTEM ☐ PROJECT ☒ TASK ☐ OTHER

1.

2. REPORTS CONTROL SYMBOL:

PAGE 1 OF 1 PAGES

3. DATE: 1 February 19[illegible]

4. TITLE: ATTITUDE REFERENCE UNIT (OSV) (ACCEPTANCE TESTS)

5. INITIAL ☒ CHANGE

6. NUMBER:

7. RESP CENTER:

8. PROJECT OFFICER:

9. SUPPORTS (Sys or Proj): ARS

10. CONTRACTOR: LOCKHEED MSD

11. CONTR NR:

12. PRIORITY AND PREC:

13. SECURITY: ~~SECRET~~

14. ITEM NUMBER	15. TEST ITEM	16. TEST DESCRIPTION	17. TEST AGENCY AND SITE	18. TEST ITEM AVAILABLE	19. RQD TEST COMPL DATE
1.	Reference Unit (OSV)	Response tests to specified input signals	MSD Research Lab.	June '57	July '57
2.	Reference Unit (OSV)	Vibration & temperature tests	MSD Research Lab.	June '57	July '57
3.	Reference Unit (OSV)	Functional Tests with complete attitude control system (Bird Cage)	MSD Research Lab.	July '57	Aug. '57
4.	Reference Unit (OSV)	Functional Tests with autopilot & IMC (Simulated Vehicle)	MSD Research Lab.	July '57	Aug. '57
5.	Reference Unit (OSV)	Flight Test*	AFMTC	Aug. '57	Jun. '59

* About 11 units required.

TEST CENTER APPROVAL

20. NAME: ORGANIZATION:

21. NAME: ORGANIZATION: DATE:

RESPONSIBLE CENTER APPROVAL

22. NAME: ORGANIZATION: DATE:

ARDC FORM 105, 1 JUL 55 — PREVIOUS EDITIONS OF THIS FORM ARE OBSOLETE.

R & D TEST ANNEX

☐ SYSTEM ☐ PROJECT ☒ TASK ☐ OTHER

Field	Entry
1.	
2. Reports Control Symbol	
Page	1 of 1 pages
3. Date	[illegible]
4. Title	AUTOPILOT (STV) (ACCEPTANCE TESTS)
5. Initial / Change	Initial ☒
6. Number	
7. Resp Center	
8. Project Officer	
9. Supports (Sys or Proj)	ARS
10. Contractor	LOCKHEED MSD
11. Contr Nr	
12. Priority and Prec	
13. Security	

14. Item Number	15. Test Item	16. Test Description	17. Test Agency and Site	18. Test Item Available	19. Rqd Test Compl Date
1.	Autopilot (STV)	Response Tests* (Simulated Load)	MSD Research Lab.	Feb. '57	Mar. '57
2.	Autopilot (STV)	Vibration and temperature tests. (Simulated load)	MSD Research Lab.	Feb. '57	Mar. '57
3.	Autopilot (STV)	Simulation Tests (Simulated Vehicle)	MSD Research Lab.	Feb. '57	Mar. '57
4.	Autopilot (STV)	Flight Test**	AFMTC	Apr. '57	Oct. '58

* About 2 units required for acceptance tests.
** About 18 units required for (STV)

TEST CENTER APPROVAL

20. Name	Organization	
21. Name	Organization	Date
		Date

RESPONSIBLE CENTER APPROVAL

22. Name	Organization	Date

ARDC FORM 1 JUL 55 105 PREVIOUS EDITIONS OF THIS FORM ARE OBSOLETE.

1.

R & D TEST ANNEX

☐ SYSTEM ☐ PROJECT ☒ TASK ☐ OTHER

2. REPORTS CONTROL SYMBOL

PAGE 1 OF 1 PAGES

3. DATE 1 February 1...

4. TITLE AUTOPILOT (OSV) (ACCEPTANCE TESTS)

5. INITIAL ☒ CHANGE

6. NUMBER

7. RESP CENTER	8. PROJECT OFFICER	9. SUPPORTS (Sys or Proj)	10. CONTRACTOR	11. CONTR NR	12. PRIORITY AND PREC	13. SECURITY
		ARC	LOCKHEED MSD			

14. ITEM NUMBER	15. TEST ITEM	16. TEST DESCRIPTION	17. TEST AGENCY AND SITE	18. TEST ITEM AVAILABLE	19. RQD TEST COMPL DATE
1.	Autopilot (OSV)	Response Test* (Simulated Load)	MSD Research Lab	May'57	June'57
2.	Autopilot (OSV)	Vibration & temperature tests (Simulated Load)	MSD Research Lab	May'57	June'57
3.	Autopilot (OSV)	Simulation Tests (Simulated Vehicle)	MSD Research Lab	June'57	July'57
4.	Autopilot (OSV)	Flight Tests**	AFMTC	July'57	June'59

* About 2 units required for acceptance tests (Lab.)
** About 12 units required for OSV tests.

TEST CENTER APPROVAL

20. NAME	ORGANIZATION	DATE
21. NAME	ORGANIZATION	DATE

RESPONSIBLE CENTER APPROVAL

22. NAME	ORGANIZATION	DATE

ARDC FORM 1 JUL 55 105 PREVIOUS EDITIONS OF THIS FORM ARE OBSOLETE.

1.

R & D TEST ANNEX

☐ SYSTEM ☐ PROJECT ☒ TASK ☐ OTHER

2. REPORTS CONTROL SYMBOL

PAGE 1 OF 4 PAGES

3. DATE: 1 February 1956

4. TITLE: ATTITUDE CONTROL (DEVELOPMENTAL TESTS)

5. INITIAL ☒ CHANGE

6. NUMBER

7. RESP CENTER

8. PROJECT OFFICER

9. SUPPORTS (Sys or Proj): AR'

10. CONTRACTOR: LOCKHEED MSD

11. CONTR NR

12. PRIORITY AND PREC

13. SECURITY: [illegible]

14. ITEM NUMBER	15. TEST ITEM	16. TEST DESCRIPTION	17. TEST AGENCY AND SITE	18. TEST ITEM AVAILABLE	19. RQD TEST COMPL DATE
1.	Torque Drive Units - Pitch & Yaw (STV)	Response to specified input signal	MSD Research Lab.	Sept.'56	Oct.'56
2.	Damping Computer (STV)	Frequency response tests for specified input signals (Units in conjunction with mathematical vehicle)	MSD Research Lab.	Sept.'56	Oct.'56
3.	TDU Plus Damping Computer (STV)	"Bird Cage" Test - 3 dimension response to specified input signal	MSD Research Lab.	Oct.'56	Nov.'56
4.	Complete attitude control system (STV)	With availability of reference platform -- "Bird Cage" response tests to specified input signals. Simulated moments of inertia.	MSD Research Lab.	Dec.'56	Jan.'57

TEST CENTER APPROVAL

20. NAME	ORGANIZATION	DATE
21. NAME	ORGANIZATION	DATE

RESPONSIBLE CENTER APPROVAL

22. NAME	ORGANIZATION	DATE

ARDC FORM 1 JUL 55 105 PREVIOUS EDITIONS OF THIS FORM ARE OBSOLETE.

R & D TEST ANNEX

☐ SYSTEM ☐ PROJECT ☒ TASK ☐ OTHER

1.	2. Reports Control Symbol
	Page 2 of 4 Pages

4. Title: ATTITUDE CONTROL (DEVELOPMENTAL TESTS)

5. Initial ☒ Change

6. Number:

7. Resp Center:

8. Project Officer:

9. Supports (Sys or Proj): ARS

10. Contractor: LOCKHEED MSD

11. Contr Nr:

12. Priority and Prec:

13. Security:

14. Item Number	15. Test Item	16. Test Description	17. Test Agency and Site	18. Test Item Available	19. Rqd Test Compl Date
5.	Horizon Sensor	Response tests to varying inputs	MSD Research Lab.	Sept. '56	Oct. '56
6.	Horizon Sensor	"Bird Cage" Tests. With simulated horizon.	MSD Research Lab.	Sept. '56	Nov. '56
7.	Complete System (STV) (Attitude Control)	Flight Tests* - No Horizon Sensor	AFMTC	May '57	Oct. '58

* Approximately 18 attitude control systems required STV.

20. Name	Test Center Approval — Organization	
21. Name	Organization	Date
22. Name	Responsible Center Approval — Organization	Date

ARDC Form 105, 1 Jul 55 — Previous editions of this form are obsolete.

MSD 1536

R & D TEST ANNEX

☐ SYSTEM ☐ PROJECT ☒ TASK ☐ OTHER

1.

3. REPORTS CONTROL SYMBOL

PAGE 3 OF 4 PAGES

4. TITLE: ATTITUDE CONTROL (DEVELOPMENTAL TESTS)

5. INITIAL ☒ CHANGE

5. DATE: 1 February 1956

6. NUMBER

7. RESP CENTER

8. PROJECT OFFICER

9. SUPPORTS (Sys or Proj): ARS

10. CONTRACTOR: LOCKHEED MSD

11. CONTR NR

12. PRIORITY AND PREC

13. SECURITY: ~~SECRET~~

14. ITEM NUMBER	15. TEST ITEM	16. TEST DESCRIPTION	17. TEST AGENCY AND SITE	18. TEST ITEM AVAILABLE	19. RQD TEST COMPL DATE
8.	Torque Drive Units (OSV)	Response Tests of modified TDU. with and without horizon sensor.	MSD Research Lab.	July '57	Aug. '57
9.	Attitude System for (OSV)	Response Tests for STV system modified for OSV ("Bird C ge") -- Simulated moment of inertia.	MSD Research Lab.	Sept. '57	Oct. '57
10.	Complete OSV System	Flight Test* -- NOTV No Horizon Sensor	AFMTC	Oct. '57	Apr. '58
11.	Complete OSV System	Flight Test** -- CTV	AFMTC	Apr. '58	July '59

* About 3 for NOTV Flights
** About 7 for OTV Flights

TEST CENTER APPROVAL

20. NAME	ORGANIZATION	
21. NAME	ORGANIZATION	DATE

RESPONSIBLE CENTER APPROVAL

22. NAME	ORGANIZATION	DATE

ARDC FORM 1 JUL 55 105 PREVIOUS EDITIONS OF THIS FORM ARE OBSOLETE.

SECRET

MSD 1536

R & D TEST ANNEX

☐ SYSTEM ☐ PROJECT ☒ TASK ☐ OTHER

1.

2. REPORTS CONTROL SYMBOL

PAGE 4 OF 4 PAGES

3. DATE: 1 February 195[illegible]

4. TITLE: ATTITUDE CONTROL (DEVELOPMENTAL TESTS)

5. INITIAL ☒ CHANGE

6. NUMBER -

7. RESP CENTER

8. PROJECT OFFICER

9. SUPPORTS (Sys or Proj): ARC

10. CONTRACTOR: LOCKHEED MSD

11. CONTR NR

12. PRIORITY AND PREC

13. SECURITY

14. ITEM NUMBER	15. TEST ITEM	16. TEST DESCRIPTION	17. TEST AGENCY AND SITE	18. TEST ITEM AVAILABLE	19. REQ TEST COMPL DATE
12.	TDU Trim System* (OSV)	Response Tests to specific input signals using simulated vehicle flight.	MSD Research Lab.	Mar.-'58	Mar.-'58
13.	TDU Trim System (OSV)	"Bird Cage" Tests with attitude Control system.	MSD Research Lab.	Apr.-'58	Apr.-'58

* About 7 Units required for flight test.

TEST CENTER APPROVAL

20. NAME	ORGANIZATION	DATE
21. NAME	ORGANIZATION	DATE

RESPONSIBLE CENTER APPROVAL

22. NAME	ORGANIZATION	DATE

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R & D TEST ANNEX

☐ SYSTEM ☐ PROJECT ☒ TASK ☐ OTHER

1.

2. REPORTS CONTROL SYMBOL

PAGE 1 OF 2 PAGES

3. DATE: 1 February 1956

4. TITLE: ATTITUDE CONTROL (ENVIRONMENTAL TESTS)

5. INITIAL ☒ CHANGE

6. NUMBER

7. RESP CENTER

8. PROJECT OFFICER

9. SUPPORTS (Sys or Proj): ARS

10. CONTRACTOR: LOCKHEED MSD

11. CONTR NR

12. PRIORITY AND PREC

13. SECURITY: SECRET

14. ITEM NUMBER	15. TEST ITEM	16. TEST DESCRIPTION	17. TEST AGENCY AND SITE	18. TEST ITEM AVAILABLE	19. MOD TEST COMPL DATE
1.	Torque Drive Units (STV)	Shock, Vibration, and temperature tests & (Humidity)	MSD Research Lab.	Jan. '57	Feb. '57
2.	Damping (Computer STV)	Shock, Vibration, and temperature tests & (Humidity)	MSD Research Lab.	Jan. '57	Feb. '57
3.	Horizon Sensor (STV)	Shock, Vibration, and temperature tests & (Humidity)	MSD Research Lab.	Jan. '57	Mar. '57
4.	Torque Drive Units (OSV)	Shock, & Vibration, and temperature tests.	MSD Research Lab.	Aug. '57	Sept. '57
5.	Damping Computer	With modification from STV shock & vibration and temperature tests	MSD Research Lab.	Aug. '57	Sept. '57
6.	Complete Attitude Control System STV	Tie down checks with OSV Rocket firing	STF	Feb. '57	Mar. '57

TEST CENTER APPROVAL

20. NAME — ORGANIZATION

21. NAME — ORGANIZATION — DATE

DATE

RESPONSIBLE CENTER APPROVAL

22. NAME — ORGANIZATION — DATE

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R & D TEST ANNEX

☐ SYSTEM ☐ PROJECT ☒ TASK ☐ OTHER

2. REPORTS CONTROL SYMBOL

PAGE 2 OF 2 PAGES

3. DATE: 1 February 1958

4. TITLE: ATTITUDE CONTROL (ENVIRONMENTAL TESTS)

5. INITIAL ☒ CHANGE

6. NUMBER

7. RESP CENTER

8. PROJECT OFFICER

9. SUPPORTS (Sys or Proj): ARS

10. CONTRACTOR: LOCKHEED MSD

11. CONTR NR

12. PRIORITY AND PREC

13. SECURITY: ~~SECRET~~

14. ITEM NUMBER	15. TEST ITEM	16. TEST DESCRIPTION	17. TEST AGENCY AND SITE	18. TEST ITEM AVAILABLE	19. REQ TEST COMPL DATE
7.	Horizon Sensor (OSV)	Shock, vibration & temperature tests.	MSD Research Lab.	Oct. '57	Dec. '57
8.	Complete Attitude Control System OSV (Less Horizon Sensor)	Tie Down checks with OSV Rocket Firing	STF	Oct. '57	Nov. '57
9.	TDU Trim System	Shock & Vibration Tests and temperature tests	MSD Research Lab.	Apr. '58	Apr. '58

TEST CENTER APPROVAL

20. NAME | ORGANIZATION

21. NAME | ORGANIZATION | DATE

RESPONSIBLE CENTER APPROVAL

22. NAME | ORGANIZATION | DATE

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R & D TEST ANNEX

☐ SYSTEM ☐ PROJECT ☒ TASK ☐ OTHER

1.

2. REPORTS CONTROL SYMBOL

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3. DATE: 1 February 1958

4. TITLE: IMAGE MOTION COMPENSATION* (DEVELOPMENT TESTS)

5. INITIAL ☒ CHANGE ☐

6. NUMBER

7. RESP CENTER

8. PROJECT OFFICER

9. SUPPORTS (Sys or Proj): ARS

10. CONTRACTOR: LOCKHEED MSD

11. CONTR NR

12. PRIORITY AND PREC

13. SECURITY: SECRET

14. ITEM NUMBER	15. TEST ITEM	16. TEST DESCRIPTION	17. TEST AGENCY AND SITE	18. TEST ITEM AVAILABLE	19. RQD TEST COMPL DATE
1.	Film Speed Compensator (Pitch mode)	Response Tests to Specified Input Signals	MSD Research Lab.	Feb. '58	Mar. '58
2.	Roll-Yaw Mode Compensator	Response Tests to Specified Input Signals	MSD Research Lab.	Feb. '58	Mar. '58
3.	Complete Compensator	"Bird Cage" tests with Attitude Control System in Operation	MSD Research Lab.	Mar. '58	Apr. '58
4.	Complete Compensator	Flight Tests** (OSV)	AFMTC	Apr. '58	June '58

* If needed in photographic system.
** About 7 Units would be required.

TEST CENTER APPROVAL

20. NAME	ORGANIZATION	DATE

21. NAME	ORGANIZATION	DATE

RESPONSIBLE CENTER APPROVAL

22. NAME	ORGANIZATION	DATE

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R & D TEST ANNEX

☐ SYSTEM ☐ PROJECT ☒ TASK ☐ OTHER

2. REPORTS CONTROL SYMBOL

PAGE 1 OF 1 PAGES

3. DATE: 1 February 19[illegible]

4. TITLE: IMAGE MOTION COMPENSATION* (ENVIRONMENTAL TESTS)

5. INITIAL ☒ CHANGE

6. NUMBER:

7. RESP CENTER	8. PROJECT OFFICER	9. SUPPORTS (Sys or Proj)	10. CONTRACTOR	11. CONTR NR	12. PRIORITY AND PREC	13. SECURITY
		ARS	LOCKHEED MSD			SECRET

14. ITEM NUMBER	15. TEST ITEM	16. TEST DESCRIPTION	17. TEST AGENCY AND SITE	18. TEST ITEM AVAILABLE	19. RQD TEST COMPL DATE
1.	Complete Compensator	Vibration & Temperature Tests (Environmental Tests)	MSD Research L[illegible]	M r. '58	Apr. '58

* If needed in photographic system.

TEST CENTER APPROVAL

20. NAME	ORGANIZATION	DATE

21. NAME	ORGANIZATION	DATE

RESPONSIBLE CENTER APPROVAL

22. NAME	ORGANIZATION	DATE

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MSD 1536

1. R & D TEST AND TEST SUPPORT AIRCRAFT ANNEX

☐ SYSTEM ☒ PROJECT ☐ TASK ☐ OTHER

2. REPORTS CONTROL SYMBOL

PAGE 1 OF 1 PAGES

3. DATE 1 March 1956

4. TITLE Subsystem D - GUIDANCE AND CONTROL

5. INITIAL ☐ CHANGE

6. NUMBER

7. ITEM NUMBER	8. AIRCRAFT REQUIRED QTY	TYPE, MODEL AND SERIES	SERIAL NUMBER	9. ASG CODE	10. MOD REQD	11. DATE REQD AND LOCATION	12. ESTIMATED RELEASE DATE	13. RECOMMENDED DISPOSITION	14. F HRS R EST	15. EST COST

AIRCRAFT WILL NOT BE REQUIRED FOR TESTS OF GUIDANCE AND CONTROL SUBSYSTEM

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R & D MATERIEL ANNEX

☐ SYSTEM ☐ PROJECT ☐ TASK ☐ OTHER

1.

2. REPORTS CONTROL SYMBOL

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3. DATE February [illegible]

4. TITLE

G[illegible]ANCE AND CONTROL [illegible] T[illegible]M

5. INITIAL ☒ CHANGE

6. NUMBER

7. MATERIEL REQUIREMENTS (Indicate Items in Columnar Form using Columns as cited in Examples)

MATERIEL	ESTIMATED COST (APPROX)	NEED DATE
Medium Precision Analog Computer		
60 Operational Amplifiers		
8 Function Generators		1957
6 Multipliers	16,000	1957
2 Recorders (4 Channel)	5,000	1957
2 Signal Generators (Audio Oscillators)	7,000	1957
Intercommunication System to Controls Lab.	1,000	1957
Attitude Control Simulation Test Stand (Bird C go)		1956
Horizon Sensor Test Stand	20,000	1956
Autopilot load simulation test stand	4,000	1957
	3,000	1957

*Such a linear system is presently available in MSD Research Laboratories. Additional nonlinear equipment is needed.

Required to support, controls development and simulation at MSD Research Lab. This equipment will not be part of the MSD computer facility.

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MISSILE SYSTEMS DIVISION

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MSD 1536

R & D MATERIEL ANNEX

☐ SYSTEM ☒ PROJECT ☐ TASK ☐ OTHER

1. REPORTS CONTROL SYMBOL

PAGE 2 OF 3 PAGES

3. DATE: 1 February 1...

4. TITLE: GUIDANCE AND CONTROL SUBSYSTEM

5. INITIAL ☒ CHANGE

6. NUMBER

7. MATERIEL REQUIREMENTS (Indicate items in sequence of need using columns as cited in Examples)

MATERIEL	ESTIMATED COST	NEED DATE
Gyro Rate Table (.01° per sec to 1200° per sec)	50,000	195[illegible]
Servo Drive, 25-50 pound capacity, built in strc ...		
Tilt Table (0.01%)	(5,000)	1957
Rate Table (0.01%)	9,000	1956
Equatorial drift test stand (0.01%)	(5,000)	1957
Scorsby Test Table (0.01%)	2,400	1957

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MSD 1536

1\.

R & D MATERIEL ANNEX

☐ SYSTEM ☐ PROJECT ☐ TASK ☐ OTHER

2\. REPORTS CONTROL SYMBOL

PAGE 3 OF 3 PAGES

3\. DATE: 1 February 1956

4\. TITLE: COMPUTER FACILITY

5\. INITIAL ☒ CHANGE ☐

6\. NUMBER

7\. MATERIEL REQUIREMENTS (Indicate Items in Columnar Form using Columns as cited in Examples)

MATERIEL	ESTIMATED COST	NEED DATE
1. High Precision Analog Computer*		
96 Operational Amplifiers		
8 Servo Resolvers (Precision)		
20 Multipliers (Servos)		
10 Electronic Multipliers		
2 X-Y Plotters (Small)		
1 X-Y Plotter (Large)		
10 Function generators		
40 Amplifiers*		
8 Servo Multipliers		
5 Diode Function Generators		
2 - 4 channel recorders		
2. Digital Computer Facility* already planned for MSD RESEARCH LAB. (Fall 1956)		

**Now in use at MSD Research Lab.

This requirement is subject to correlation with other programs and equipment.

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MSD 1536

SUBSYSTEM D- GUIDANCE AND CONTROL

DATE 1 March 1956

ITEM: MSD - "INPLANT" TEST FACILITY *
SYSTEMS TEST FACILITY *

USING AGENCY:

LOCATION:

BUDGET CONTROL ESTIMATE:

NEED DATE:

SCHEDULE:	1956						1957												1958												1959											
	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D

DESCRIPTION AND UTILIZATION:

* Complete description of the facilities are given in TAB 6 - Subsystem L - Vehicle Ground Support.

REMARKS:

Revised Form 108

Tab 7

R & D Contract Funds

Subsystem D - Guidance and Control

Subsystem D. GUIDANCE AND CONTROL

Tab 7. R & D Contract Funds (in thousands of dollars)

	FY 57				FY 58				FY 59				FY 60	
	1	2	3	4	5	6	7	8	9	10	11	12	13	14
LAC														
(1) Research and Development	81	137	104	273	336	360	403	428	416	377	329	334	199	9
(a) Sub Contracts	88	82	70	97	186	168	168	168	168	112	96	96	-0-	-0-
(2) Fabrication	106	106	109	186	186	215	279	318	318	240	235	235	183	266
(a) Purchased Components	288	288	288	495	495	525	655	726	728	520	520	520	355	486
Sub Total	505	575	663	1,052	1,144	1,277	1,506	1,648	1,631	1,250	1,182	1,149	738	960
Fee	50	57	66	105	114	127	150	164	163	125	118	114	73	96
TOTALS	556	633	730	1,158	1,258	1,405	1,657	1,808	1,795	1,376	1,301	1,260	812	1,059
Total Fiscal Year			3,077				6,127				5,732			

* Differences in totals due to rounding

MSD 1536

Subsystem D. GUIDANCE AND CONTROL

Tab 7. R & D Contract Funds (in thousands of dollars)

LAC	FY 61					FY 62				FY 63					Totals*
	15	16	17	18	19	20	21	22	23	24	25	26	27		
(1) Research and Development	234	254	269	259	249	239	204	194	202	206	206	206	127		6,948
(a) Sub Contracts	-0-	-0-	-0-	-0-	-0-	-0-	-0-	-0-	-0-	-0-	-0-	-0-	-0-		1,300
(2) Fabrication	238	385	430	368	384	419	318	346	317	453	575	136	-0-		7,357
(a) Purchased Components	399	644	780	600	630	698	551	618	549	791	1,003	239	-0-		14,388
Sub Total	867	1,285	1,421	1,228	1,264	1,351	1,068	1,194	1,067	1,451	1,787	581	127		29,934
Fee	86	128	148	122	126	135	106	115	106	145	170	58	18		2,993
TOTALS*	954	1,414	1,569	1,351	1,391	1,487	1,176	1,270	1,175	1,597	1,966	640	140		32,928
Total Fiscal Year		4,238			5,798				5,817			2,745			32,928

*Differences in totals due to rounding

D-Tab 7, p 3

Tab 8

Estimate of Manpower Requirements

Subsystem D - Guidance and Control

Subsystem D. GUIDANCE CONTROL

Tab 8. Estimate of Manpower Requirements

WORK ITEM	Type of Manpower	QUARTERS 1	2	3	4	5	6	7	8	9	10	11	12	13	14
LAC Research and Development	1-2-3*	16	27	37	52	64	70	79	84	82	74	66	67	40	42
LAC Fabrication and Assembly	4	30	30	30	51	51	59	79	90	90	68	68	68	53	77
TOTAL		46	57	67	103	115	129	158	174	172	142	134	135	93	119

* Average

40% Type 1 Scientific & Technical

50% Type 2 Engineering Support

10% Type 3 Management & Administration

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Subsystem D. GUIDANCE CONTROL

Tab 8. Estimate of Manpower Requirements (Cont'd)

WORK ITEM	Type of Manpower	QUARTERS														Total Man Quarters
		15	16	17	18	19	20	21	22	23	24	25	26	27	28	
LAC Research and Development	1-2-3*	47	51	54	52	50	48	41	39	40	41	41	41	25		1370
LAC Fabrication and Assembly	4	67	111	124	106	111	121	90	100	91	139	166	39			2100
TOTAL		114	162	178	158	161	169	131	139	131	171	207	80	25		3470

* Average

40% Type 1 Scientific & Technical

50% Type 2 Engineering Support

10% Type 3 Management & Administration

~~SECRET~~

Pied Piper

MSD 1536

LOCKHEED AIRCRAFT CORPORATION
MISSILE SYSTEMS DIVISION

APPENDIX

~~SECRET~~

Subsystem D, GUIDANCE AND CONTROL

APPENDIX

1. INTRODUCTION

This appendix presents the general analytical results of the guidance and control subsystem study. It has been prepared in support of the General Design Specifications (Tab 1) appearing in this volume. As the study has been conducted on a parallel basis, it has not been possible in every case to feed the results back and to perform a second iteration of the system. Similarly, further coordination with the other subsystems, particularly vehicle, payload, and propulsion, is required now that the several subsystem studies are being completed.

In particular, the requirements of image motion compensation, attitude control and altitude indication are closely interrelated. The results of the attitude control study have been satisfactory to the extent that it appears that attitude indication and image motion compensation may be superfluous. If this is indeed the situation, then considerable simplification of the attitude control system will result. But if examination of the imperfect, or nonideal, character of the instruments and mechanizations prove that these compensations and corrections are required, it is to be expected that they will require second-order corrections only.

The introduction of an Advanced Reconnaissance System (ARS) in the 1960's may produce a requirement for a more advanced guidance system than the one described here. In this event, it is presumed that a guidance suitable for IRBM application might be used. The IRBM requirements are such that guidance for a satellite is included in a system capable of the IRBM guidance.

The greatest element of uncertainty in the design of a guidance and control system of the kind described here is introduced by the environment in which it must perform. Fortunately, the Atlas, Redstone, and Corporal programs are providing considerable information in this area. The non-linear dynamic problems, such as the elastic airframe and fuel sloshing, must be examined with respect to the specific vehicle design, but it appears that the small Orbital Stage Vehicle (OSV) which is operated outside of the atmosphere will be relatively free of disturbances from these causes.

2. ERROR ANALYSIS OF ASCENT GUIDANCE

The ascent of the booster-vehicle combination is composed of three phases:

1. The Atlas "C" boost phase
2. The transition coast phase
3. The orbital boost phase

The Orbital Stage Vehicle is boosted and guided into a specific climb trajectory by the Atlas "C".

At the instant of cutoff the first stage, referred to as "boost", possesses errors in position, velocity, and time. These are due to variations in such parameters as thrust, vehicle mass, winds, and drifts in the accelerometer and autopilot.

The magnitudes of these errors depend upon whether the Atlas "C" vehicle is guided by open-loop or closed-loop radio-inertial system and by the state of development of that system. Hence, consideration has been given to mounting an orbital test vehicle without relying upon the closed-loop, radio-inertial guidance system.

At the end of the boost stage, the OSV continues to coast in an ellipse until its apogee is approached. This is called the transition coast stage. A transition computer, Fig. 2-1 is used to apply corrections to the guidance according to the active Atlas burnout conditions. The errors arising in this stage are given by the errors propagated from the boost stage, the inaccuracy in computed time and attitude at apogee, and the gyro reference drifts.

At the time the satellite is computed to reach its apogee, a rocket adds a measured impulse whose magnitude is measured on the basis of the calculations from the cutoff position and velocity. Errors at this point result from a calculation of the time to reach apogee which is based in imperfect knowledge of cutoff conditions. The magnitude of the thrust impulse is incorrect for the same reason. Further, the accelerometer may measure the thrust inaccurately. Drift in the gyroscopic attitude reference will cause the thrust to be applied in an incorrect direction. The thrust on-off system is shown in Figure 2-2.

The accumulated errors in the trajectory at apogee have been studied in terms of errors in the vertical and horizontal components of the velocity. At apogee, the vertical velocity should vanish, but if there is an error in the system there will be a vertical component of the velocity. For all practical purposes, the error in the horizontal velocity is the error in the orbital velocity. Accumulated errors are summed up in this study as root-mean-square values..

The following equations have been used to compute the vertical and horizontal errors at apogee in terms of the errors developed during transition coast phase (TR) and orbital boost phase (OBV).

$$\left(\varepsilon V_{v_{TOT}}\right)^2 = \left(V_o \varepsilon \gamma_o\right)^2 + \left(\varepsilon V_o \gamma_o\right)^2 + V_c^2 \left[\left(\varepsilon \gamma_o\right)^2 + \left(\varepsilon \gamma_{TR}\right)^2 + \left(\varepsilon \gamma_{OBV}\right)^2\right] \qquad (1)$$

$$\left(\mathcal{E}V_{H_{TOT}}\right)^2 = \left[\frac{\partial V}{\partial V_o}\mathcal{E}V_o + \frac{\partial V}{\partial \gamma_o}\mathcal{E}\gamma_o + \frac{\partial V}{\partial h_o}\mathcal{E}h_o\right] + \left(\mathcal{E}V_{OSV}\right)^2 \qquad (2)$$

where

V_o = velocity, γ_o = flight path angle, h_o = distance from center of earth: conditions at end of Atlas "C" boost.

$\mathcal{E}V_o$, $\mathcal{E}\gamma_o$, $\mathcal{E}h_o$ = errors at end of Atlas "C" boost.

V_G = velocity gained in orbital boost stage, fps

$\mathcal{E}V_{V_{TOT}}$ = total vertical velocity error, fps

$\mathcal{E}V_{H_{TOT}}$ = total horizontal velocity error, fps

$\mathcal{E}\gamma_{TR}$ = accumulated flight path error during transition coast, radians

$\mathcal{E}\gamma_{OSV}$ = accumulated flight path error during orbital boost stage, radians

$\left[\frac{\partial V}{\partial V_o}\mathcal{E}V_o + \frac{\partial V}{\partial \gamma_o}\mathcal{E}\gamma_o + \frac{\partial V}{\partial h_o}\mathcal{E}h_o\right] - \mathcal{E}V_{TR}$ = accumulated error in velocity during transition as determined by Ref.6 fps

$\mathcal{E}V_{OSV}$ = accumulated error in velocity during orbital stage boost, fps

The solution of these equations (see Fig. 2-3) illustrates the range of errors obtainable for a given probable range of errors existing at end of Atlas "C" boost (whether guided by an open-loop or closed-loop guidance system.) Assume, for the moment, that the error in velocity, $\mathcal{E}V_o$, at the end of Atlas boost is 30 feet per second and the error in flight path angle, $\mathcal{E}\gamma_o$, is 2 milliradians. Then from Fig. 2-3 the error in vertical velocity at apogee is 47 feet per second and the error in horizontal velocity, 19 feet per second.

From Fig. 2-4, which depicts the effect of vertical and horizontal velocity errors on the change in distance from the center of the earth (the ellipticity of the orbit), a value of 0.0038 is obtained. This indicates the satellite will travel about 15 n. miles above and below the desired altitude of 300 n. miles at a frequency approximately equal to the orbital frequency. The error in the horizontal velocity will give an amplitude oscillation of about 10 n. miles approximately 90 degrees out of phase with the vertical velocity error.

These calculations provide an indication of the acceptable degradation of the Atlas performance if a 15-n. mile orbital tolerance is to be maintained. The table below shows the permissible error in Atlas guidance and the corresponding effects on Atlas CEP and the satellite orbit condition.

ERROR AT BURNOUT	ERROR COEFFICIENT	RESULTANT ATLAS RANGE ERROR	RESULTANT AMP. OF SATELLITE ORBIT
$\mathcal{E}V_o$ = 45 ft.per sec	1 n. mile per ft per sec	45 n.mi	15 n.mi
$\mathcal{E}\gamma_o$ = 2 mils	4 n. mile per mil	8 n.mi	15 n.mi

From these, it is clear that flight path angle control is the most critical parameter in the Atlas boost of the ARS. If the performance required is not obtained, then the ARS guidance requires the use of either a flight path computer in the OSV or some means for orbit correction.

Calculations presented in Ref. 2 show that if a 300-n. mile ($\pm$100 n. miles)orbit can be achieved that orbit correction can subsequently be applied to reduce this error. If this technique is used, an Atlas burn-out error of about 10 mils appears tolerable. This represents a degradation to a 40-mile error in Atlas at impact. Despite this degradation due to errors in flight path angle and a similar degradation due to errors in velocity cutoff, vehicle performance is still acceptable.

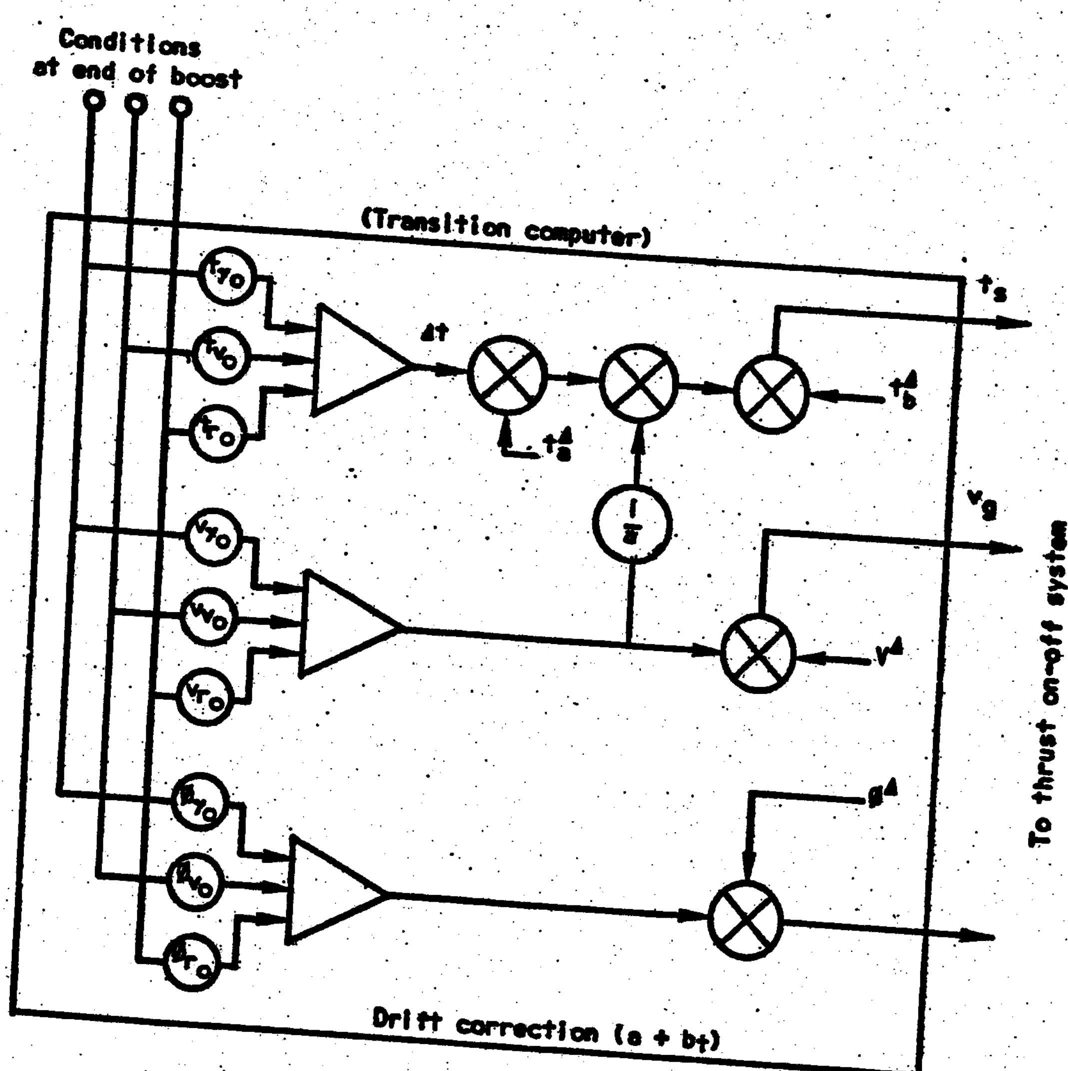


Fig. 2-1 Block Diagram of Transition Computer

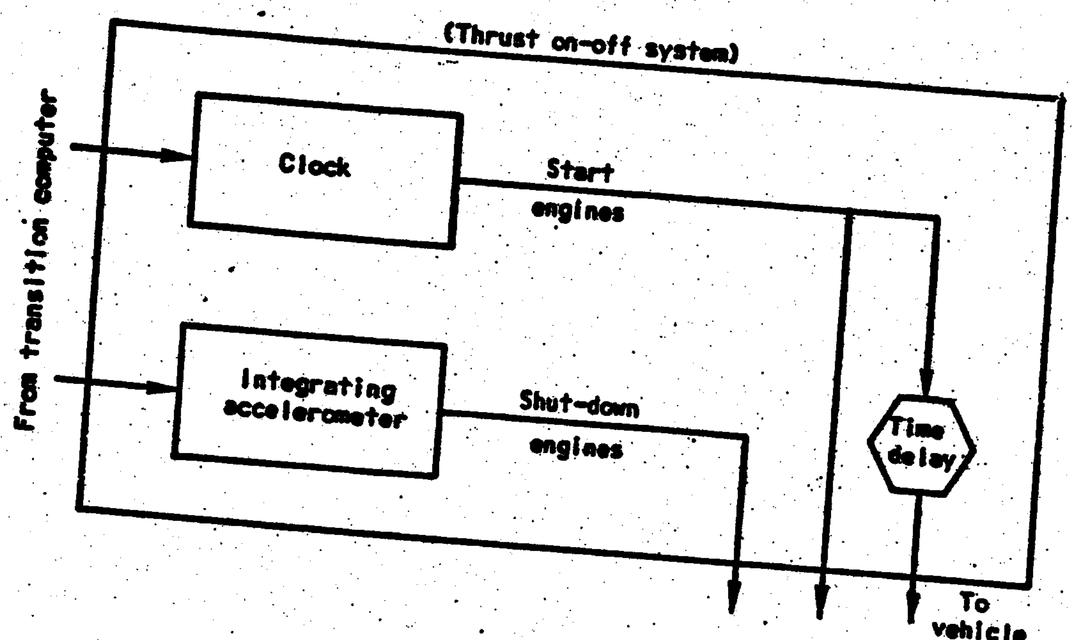


Fig. 2-2 Thrust On/Off Control

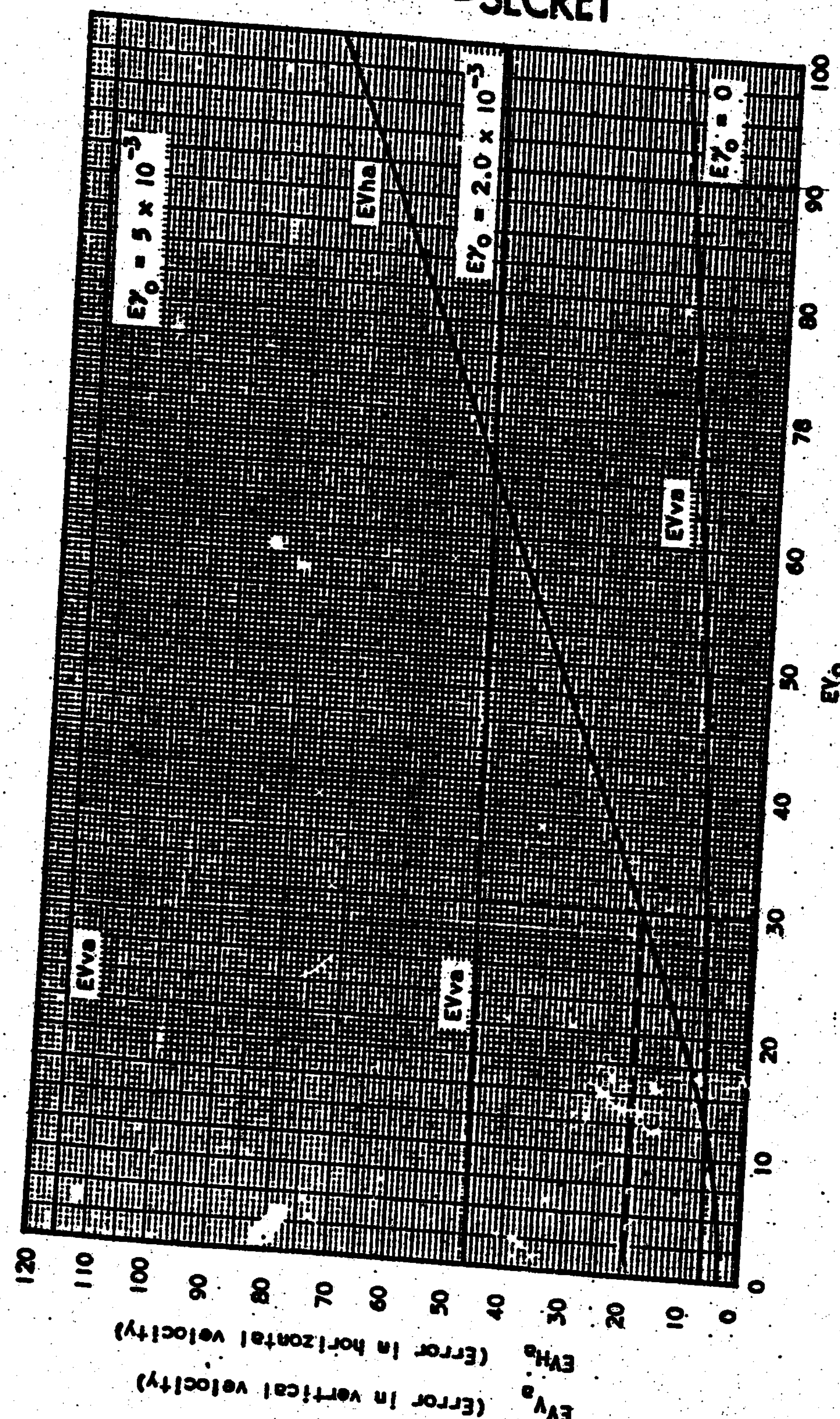


Fig. 2-3 Errors in Horizontal and Vertical Velocities at Apogee as a Function of Errors in Speed and Direction at end of Launch Boost

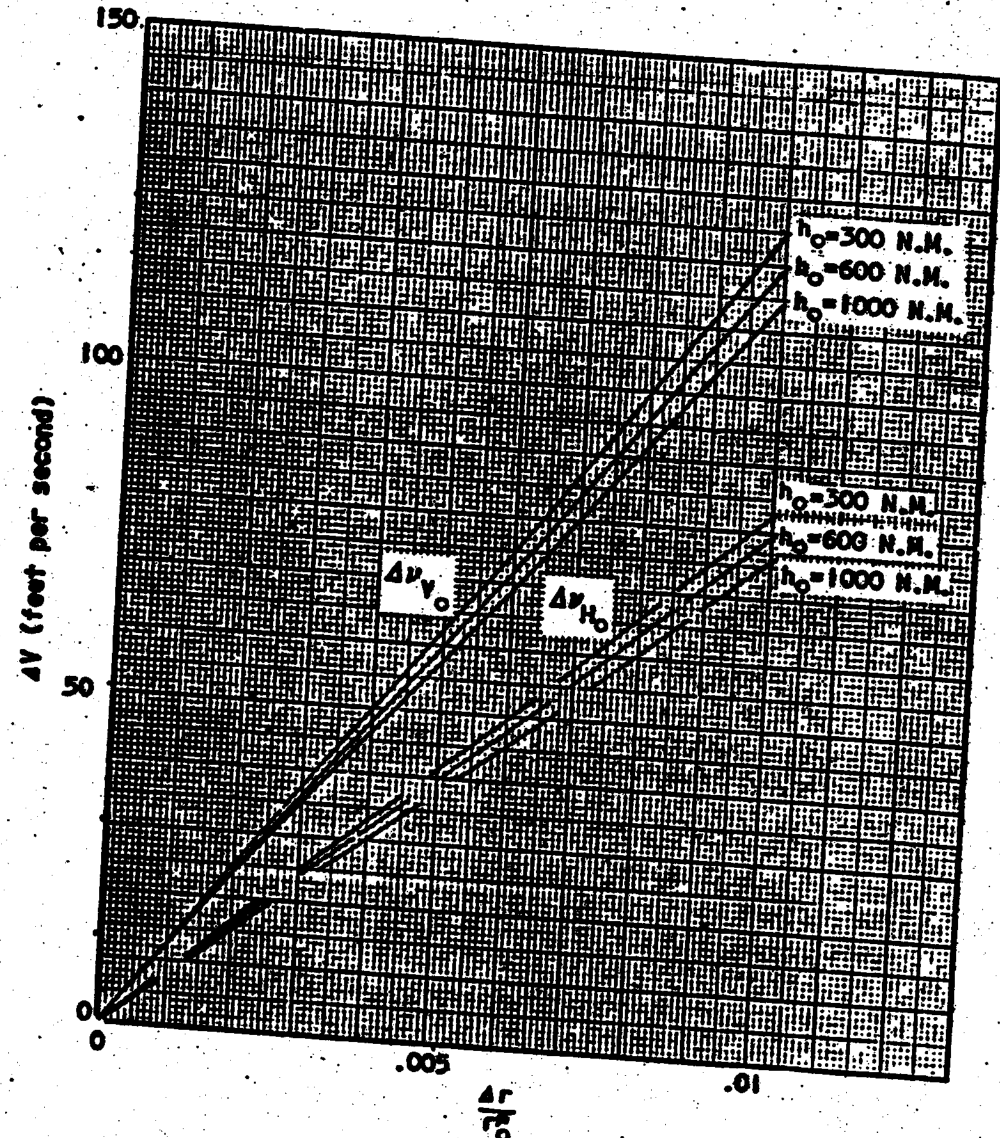


Fig. 2-4 Amplitude of Orbit Radial Oscillations Arising from Errors in Initial Orbital Velocity

3. CONTROL DURING ORBITAL POWERED PHASE

At the apogee thrust will be applied to the Orbit Stage Vehicle in a direction parallel to the horizontal plane. This thrust will be applied for about 40 seconds prior to apogee so that a measured increase in the vehicle horizontal velocity of about 3500 feet per second is made while no vertical velocity component is added. The vehicle heading is established so as to provide the proper value of maximum latitude for the orbit. Thrust on-off commands will come from the thrust on-off unit, and attitude reference for the autopilot will be derived from the Central Attitude Reference Unit.

It is envisioned that only attitude will be controlled in this phase so that a simplification in the necessary instrumentation can be made. The attitude obtained at the end of the transition phase will, except for a small error, be the horizontal corresponding to the apogee. This is the reference attitude that will persist in the ensuing phase. To achieve this reference, a precomputed bias will be added to the indication of the primary gyro reference. The control engines are started first to reduce initial errors from transition and to avoid a strong out-of-trim moment about the center of mass which would result from the possible thrust misalignment of the main engine.

If one assumes, for the moment, that the control engines may be moved in their gimbals instantaneously to a position, η , then the perturbation equations which serve to define stability for simple attitude and attitude rate feed back are:

$$I s^2 \phi = T_c \ell \eta$$

$$\eta_c + (K_1 + K_2 s) \phi = \eta$$

where

s is the Laplace operator

I is the moment of inertia of vehicle about the pertinent axis

η_c is the commanded engine attitude and Fig. 3-1 applies:

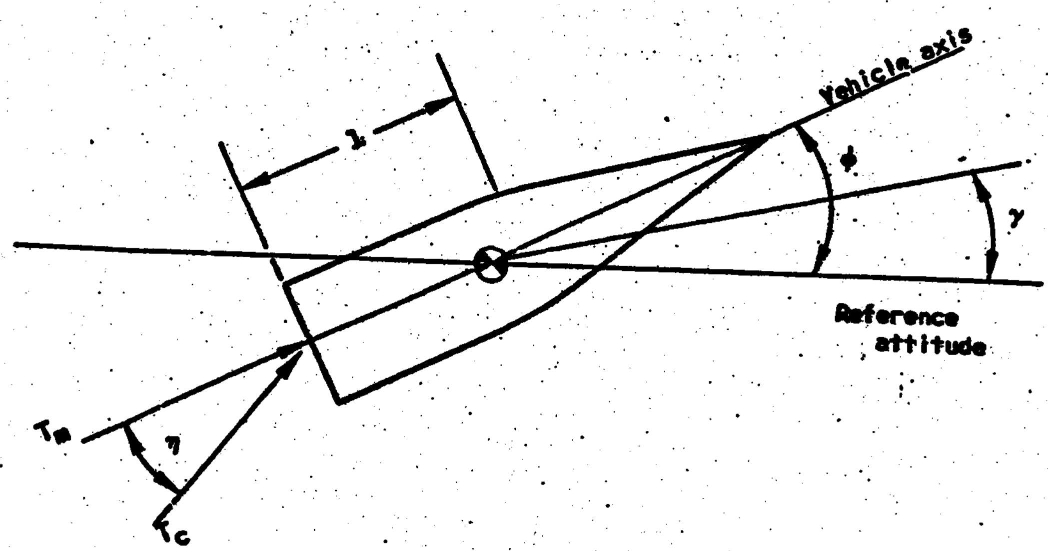


Fig. 3-1 System of Notation for Autopilot Study

Solving the above set of equations,

$$\frac{T_c \ell}{I} \eta_c = \phi \left[s^2 + \left(K_2 \frac{T_c \ell}{I} \right) s + K_1 \frac{T_c \ell}{I} \right]$$

and the relative damping,

$$\zeta = \frac{K_2}{2\sqrt{K_1}} \sqrt{\frac{T_c \ell}{I}}$$

circular frequency, $\omega = K_1 T_c \ell / I$

In this simple case, adjustment of K_1 and K_2 will give any stability desired. Thus, representative values of the parameters are

	Fueled	Empty
T_c	300 lbs.	--
ℓ	6 ft.	8 ft.
I (Pitch & Yaw)	2000 slug-ft^2	1600 slug-ft^2

and for $K_1 = 4, \quad K_2 = 3.6$

ω_e = 2.2 radians per sec

ζ = near critical damping

and the effective time constant is approximately 1 second.

One could then assign to the servo the following characterisites:

ω_s = 6 radians per second

Amplitude = ± 1/2 radian

Max. Velocity = 3 radians per second

Max. Acceleration = 10 radians per sec^2

A servo with the above characteristics would approximate the original assumption of an instantaneous servo and is easily obtainable. A more refined analysis, embodying the essential nonlinearities of a valve-controlled hydraulic motor, should not appreciably change this picture. The relationship between servos and sensing instruments is shown in Fig. 3-2.

The two 150-pound control motors are sufficient for the trimming of the main 7500-pound motor. The anticipated eccentricity is no larger than 1 degree at an offset of about 1 inch from the center of mass. The out-of-trim moment is then $7500 \times 1 \times \frac{1}{60} = 125$ inch-pounds. To balance this, a control deflection of $\eta = \frac{125}{300}\left(\frac{l_m}{l_c}\right)$ is required. If $\frac{l_m}{l_c}$ is 0.9 or less a control deflection of $\pm 1/2$ radian will be adequate. The design value of $\frac{l_m}{l_c}$ is approximately 0.5.

An estimate can be made of the errors in flight path angle, γ, suffered as a result of controlling ϕ rather than γ. The equation defining γ is

$$MV\dot{\gamma} + g = T_c(\phi - \gamma + \eta) + T_M(\phi - \gamma)$$

As a conservative estimate, assume ϕ and γ are zero and ignore g.

Then, $MV\dot{\gamma} = -T\gamma$

and

$$\gamma = \gamma_0 e^{-\frac{Tt}{MV}}$$

During the first 2 seconds when only the control motors are thrusting,

T = 300 pounds

M = 700/32 = 220 slugs

V = 20,000 feet per second

t = 2 seconds

$$\gamma = \gamma_o e^{-\frac{600}{220\,(20,000)}} = \gamma_o\left(1 - \frac{1}{7000}\right)$$

When the main engines are thrusting,

T = 7800 pounds

M = 156 slugs (fuel near exhaustion)

V = 22,000 feet per second

t = 30 seconds

$$\gamma = \gamma_o\ (1-0.09)$$

For an uncertainty in T of 500 pounds,

T = 500 pounds

M = 156 slugs

V = 22,000 feet per second

t = 30 seconds

$$\gamma = \gamma_o\ (1-0.0044)$$

Thus, the uncertainty in T will produce a rotation of the momentum vector of 0.004 γ_o, and the main thrust will produce a rotation which may be computed and used as a factor in setting the burning time of 0.09 γ_o. If γ_o is approximately 0.02 radians and is known within 5 milliradians, then the maximum error in flight path angle is (0.09) (0.005) + (0.004) (0.02) ≈ 0.0005 radians, which is still 1/2 the allowable error. If analysis proves it necessary, this error may be reduced by measuring the thrust components and applying the transverse nongravitational acceleration to the autopilot as a steering correction.

A requirement, which arises from the guidance error analysis, is that the autopilot shall be capable of holding the attitude with 0.5 degree of the reference attitude.

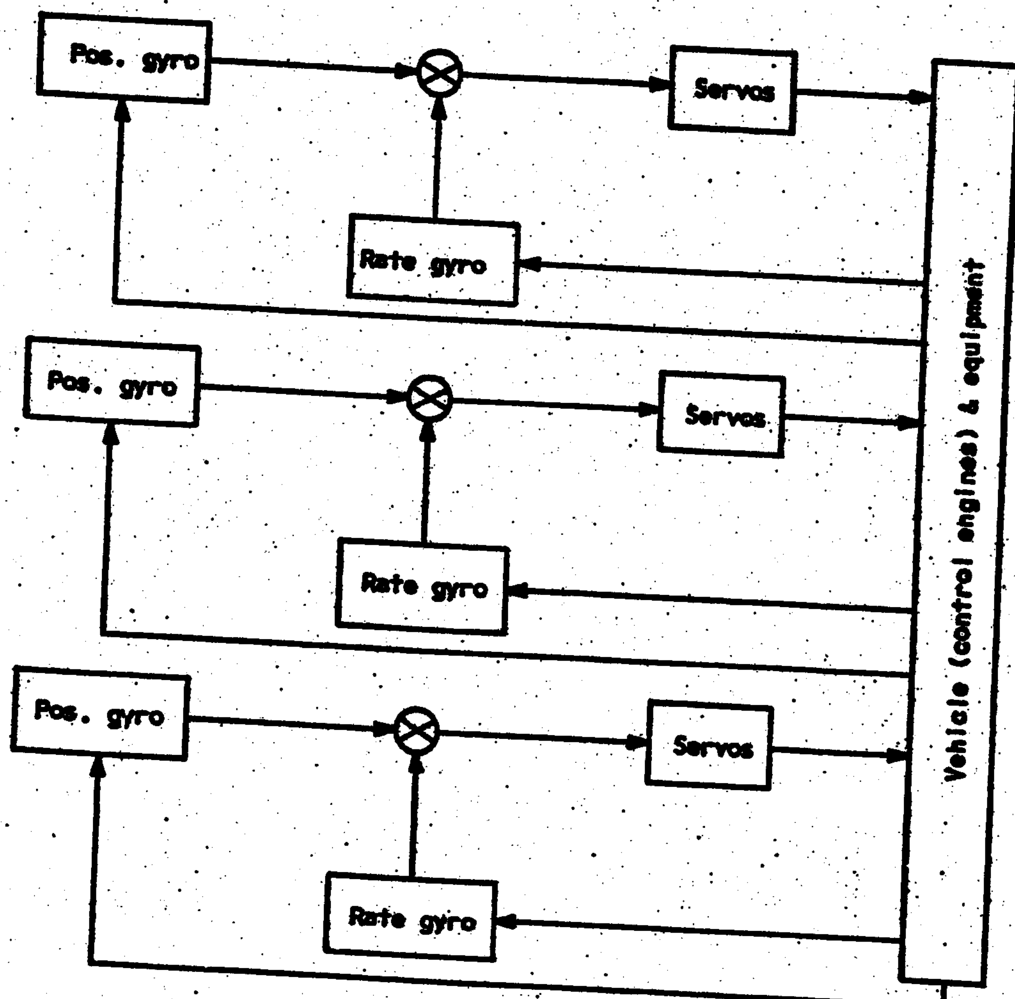


Fig. 3-2 Block Diagram of OSV Autopilot

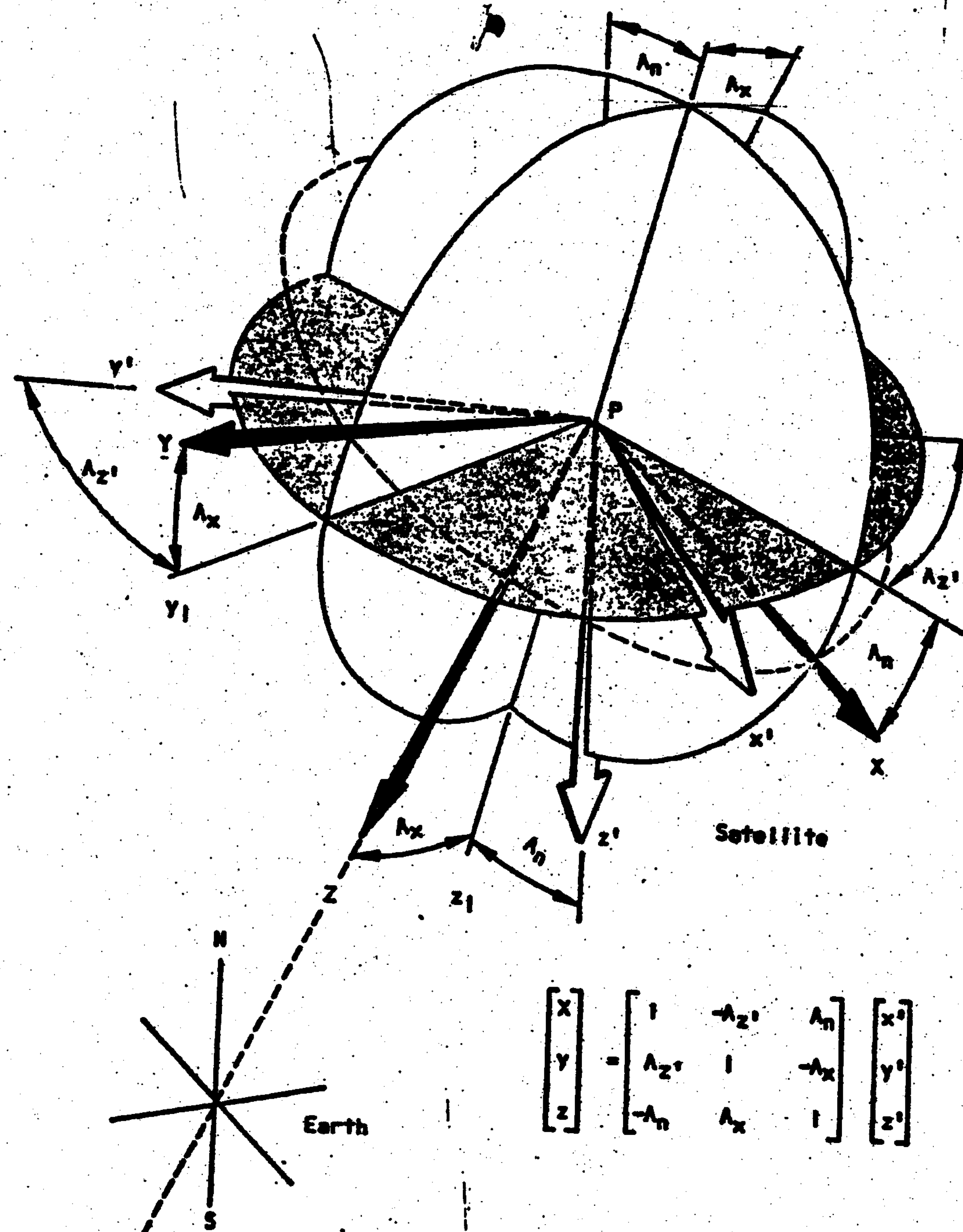
4. ATTITUDE CONTROL

4.1 Introduction

Attitude control of the proposed vehicle will maintain orientation with respect to the earth so as to allow the highest degree of visual reconnaissance and to supply sufficient damping against perturbing torques arising from within and without the vehicle. Attitude control will be supplied during portions of the flight where no rocket power is used, i.e., during the transition coast and in the orbit.

The proposed attitude control system adds or subtracts energy from the vehicle through two torque drive units composed of torque motors and flywheels. One is oriented with its axis of rotation parallel to the pitch axis and the other with its axis of rotation parallel to the yaw axis. This relation is defined by Figure 4-1 and the list of symbols. Through a selected damping computer receiving signals of attitude and/or rate of change of attitude, the roll and yaw modes are cross-coupled in such a way as to extract the energy from one mode while dissipating it in the other, thereby providing a means of sharing the damping. The pitch mode on the other hand is uncoupled from the yaw and roll modes.

A preliminary study has been made of the dynamic motion for several damping systems in order to ascertain which system would be the most feasible from the standpoint of damping random disturbances to angular rates small enough to allow good photographic resolution. (A



$$\begin{bmatrix} x \\ y \\ z \end{bmatrix} = \begin{bmatrix} 1 & -A_{z'} & A_n \\ A_{z'} & 1 & -A_x \\ -A_n & A_x & 1 \end{bmatrix} \begin{bmatrix} x' \\ y' \\ z' \end{bmatrix}$$

Fig. A-1 Coordinate System for Study of Attitude Control of OSV standard blurred image corresponding to 30 feet was taken as the tolerance.) Minimizing power requirements was also considered a design requirement.

Use of damping torques supplied by inputs of attitude rates from each of the body axes would be a satisfactory and obvious method. On the other hand it is possible if resulting motions proved to be small that only attitude sensing would be sufficient. But, by a careful examination of the vehicle dynamics it appears that control can be accomplished by sensing only the pitch attitude and yaw attitude rates as inputs to the damping computer. Previous work done at MIT utilizing integrated rate feedback (Ref. 3) is considered here as a source of possible ramifications of the foregoing systems.

Another method of control used side jets or rockets exhausting gases in such directions as to create torques about the center of gravity of the vehicle. The damping computer would provide signals to control the amount of thrust. This system suffers in that the ability to make small adjustments in angular rate or position would be difficult.

Still another method is the use of heat dissipation by radiation to supply the controlling torques. Signals from a damping computer would select the radiator in the proper location and allow proper time duration to create damping torques. This system has some merit but is not considered since it is possible that the weight to control power ratio may be large.

To facilitate a study of the dynamics of the proposed attitude control system a number of basic assumptions are necessary which, for all practical purposes, do not impare the physical significance of the analysis. These assumptions are:

1. The vehicle is traveling in a circular orbit about the earth at an altitude of 300 n. miles.
2. The earth may be represented as a homogeneous sphere of radius 21×10^6 ft.
3. Amplitudes of motion are small (less than 10°).

4.2 Symbols

$$\left.\begin{matrix} A_x \\ A_n \\ A_{z'} \end{matrix}\right\} \text{ Eulerian angles; } \left.\begin{matrix} x' \\ y' \\ z' \end{matrix}\right\} \text{ with respect to } \left.\begin{matrix} X \\ Y \\ Z \end{matrix}\right\} \text{ (radians)}$$

Vehicle parameters

$$\left.\begin{matrix} I_{x'} \\ I_{y'} \\ I_{z'} \end{matrix}\right\} \text{ moments of inertia about } \left.\begin{matrix} x' \\ y' \\ z' \end{matrix}\right\} \text{ axes, (slug ft.}^2\text{)}$$

$$F = \frac{I_{y'} - I_{z'}}{I_{x'}} \quad \text{(dimensionless)}$$

$$G = \frac{I_{x'} - I_{z'}}{I_{y'}} \quad \text{(dimensionless)}$$

$$J = \frac{I_{y'} - I_{x'}}{I_{z'}} \quad \text{(dimensionless)}$$

ω_c orbiting angular rate of axes system $\left.\begin{matrix} X \\ Y \\ Z \end{matrix}\right\}$ (rads/sec.)

Attitude control parameters

$I_{cx'}$ moment of inertia of roll torque wheel, slug-ft.^{2}

$I_{cy'}$ moment of inertia of pitch torque wheel, slug-ft.^{2}

$I_{cz'}$ moment of inertia of yaw torque wheel, slug-ft.^{2}

$\omega_{cx'}$ angular frequency of roll torque wheel, rads/sec.

$\omega_{cy'}$ angular frequency of pitch torque wheel, rads/sec.

$\omega_{cz'}$ angular frequency of yaw torque wheel, rads/sec.

H_o total constant angular momentum of rotating components aligned with the pitch axis, lb·ft·sec.

$H_{cz'}$ total constant angular momentum of rotating components aligned with the yaw axis, lb·ft·sec.

$\dot{H}_p = I_{cy'} \dot{\omega}_p$, torque applied by the torque drive unit in the pitch mode, ft·lbs.

$\dot{H}_{cz'} = I_{cy'} \dot{\omega}_{cz'}$, torque applied by the torque drive unit in the yaw mode, ft·lbs.

Note: Dots over parameters imply the derivative with respect to time τ non-dimensional time in $\omega_c t$.

4.3 Equations of Motion

The equations of motion as used in this study are presented here for convenience. No derivation will be shown since this is obtainable in references 4 and 5. The following set of equations are of a non-dimensional nature; the time scale has been changed such that $\tau = \omega_c t$. These equations represent the variation of an Eulerian set of angles with reference to the system of axes fixed in the orbit (see Fig. 4-1).

$$\ddot{A}_x + \left[4F - \frac{I_{cy}'\omega_{cy}'}{I_x'\omega_c}\right]A_x = \left[(1-F) + \frac{I_{cy}'\omega_{cy}'}{I_x'\omega_c}\right]\dot{A}_z' - \frac{I_{cx}'\dot{\omega}_{cx}'}{I_x'\omega_c} - \frac{I_{cz}'\omega_{cz}'}{I_x'\omega_c}\left(\dot{A}_n - 1\right) \qquad (1)$$

$$\ddot{A}_n + 3G A_n = -\frac{I_{cy}'\dot{\omega}_{cy}'}{I_y'\omega_c} - \frac{I_{cx}'\omega_{cx}'}{I_y'\omega_c}\left(\dot{A}_z' + A_x\right) + \frac{I_{cz}'\omega_{cz}'}{I_y'\omega_c}\left(\dot{A}_x - A_z'\right) \qquad (2)$$

$$\ddot{A}_z' + \left[J - \frac{I_{cy}'\omega_{cy}'}{I_z'\omega_c}\right]A_z' = \left[(J-1) - \frac{I_{cy}'\omega_{cy}'}{I_z'\omega_c}\right]\dot{A}_x - \frac{I_{cz}'\dot{\omega}_{cz}'}{I_z'\omega_c} - \frac{I_{cx}'\omega_{cx}'}{I_z'\omega_c}\left(\dot{A}_n - 1\right) \qquad (3)$$

The study is confined to that of a two torque wheel control system in stipulating the following conditions.

$$\omega_{cy'} = -\omega_0 + \omega_p$$
$$\omega_{cx'} = 0 \qquad (4)$$

where ω_p is a small change in $\omega_{cy'}$ and, in order to preserve near-liniarity, $\omega_p << \omega_0$. Substitution of equations (4) into equations (1), (2) and (3) and describing in terms of angular moments gives the following set of linearized equations:

$$\ddot{A}_x + \left[4F + \frac{H_o}{I_x'\omega_c}\right]A_x = \left[(1-F) - \frac{H_o}{I_x'\omega_c}\right]\dot{A}_z' - \frac{H_{cz}'}{I_x'\omega_c}\left(\dot{A}_n - 1\right) \tag{5}$$

$$\ddot{A}_n + 3G\,A_n = -\frac{\dot{H}_o}{I_y'\omega_c} + \frac{H_{cz}'}{I_y'\omega_c}\left(\dot{A}_x - A_z'\right) \tag{6}$$

$$\ddot{A}_z' + \left[J + \frac{H_o}{I_z'\omega_c}\right]A_z' = \left[(J-1) + \frac{H_o}{I_z'\omega_c}\right]\dot{A}_x - \frac{\dot{H}_{cz}'}{I_z'\omega_c} \tag{7}$$

The damping to the system is provided by rate signals fed back through the damping computer. The following systems of equations are representative of a possible damping computer.

$$\left.\begin{aligned} \dot{\mu}_p &= K_1 \dot{A}_x + (2\zeta_n \tau_n I_y' \omega_c)\dot{A}_n \\ \dot{\mu}_{cz}' &= -K_2 \dot{A}_x + (2\zeta_z' \tau_z' I_z' \omega_c)\dot{A}_z' \end{aligned}\right\} \tag{8}$$

$$\left.\begin{aligned} I_{cy}'\dot{\omega}_p &= K_1 \int \dot{A}_x \, d\tau + \int (2\zeta_n \tau_n I_y' \omega_c)\dot{A}_n \, d\tau \\ I_{cz}'\dot{\omega}_{cz}' &= -K_2 \int \dot{A}_x + \int (2\zeta_z' \tau_z' I_z' \omega_c)\dot{A}_z' \, d\tau \end{aligned}\right\} \tag{9}$$

$$\left.\begin{aligned} I_{cy}'\dot{\omega}_p &= (2\zeta_n \tau_n I_y' \omega_c)\dot{A}_n \\ I_{cz}'\dot{\omega}_{cz}' &= (2\zeta_z' \tau_z' I_z' \omega_c)\dot{A}_z' \end{aligned}\right\} \tag{10}$$

$$\left.\begin{aligned} I_{cy}'\dot{\omega}_p &= (2\zeta_n \tau_n I_y' \omega_c)\int \dot{A}_n \, d\tau \\ I_{cz}'\dot{\omega}_{cz}' &= (2\zeta_z' \tau_z' I_z' \omega_c)\int \dot{A}_z' \, d\tau \end{aligned}\right\} \tag{11}$$

where:

K_1 and K_2 are follow-up ratios and their selection is not necessarily motivated by the desire to reduce coupling as much as their effect upon the overall performance of the system.

ζ_z' is a selected damping ratio.

τ_z' undamped natural frequency about the yaw axis.

These equations represent the energy added to the dynamic system by changes in the pitch and yaw torque drive units as a function of vehicle angular rate.

4.4 Perturbing Torques

To evaluate the dynamic response (frequency and amplitude characteristics) of the attitude controlled vehicle system, the nature and possible magnitude of perturbing torques should be estimated. Torques acting on an unsymmetrical body in space are derived from two discrete sources:

1. Energy is applied to the system from outside the vehicle.
2. Energy is applied to the system from various components and parts within the vehicle.

4.4.1 External Torques

The external torques which are considered to be acting on the vehicle have been studied and energy levels have been estimated. They are as follows, (excluding the gravitational, or "dumb bell", torque which is accounted for in the equations of motion):

1. Torques developed from the sun-radiation pressure acting on an unsymmetrical body.
2. Unsymmetrical heat radiation torques arising from heat dissipation of that absorbed from the sun.
3. Torques arising from the oblateness of the earth.
4. Aerodynamic torques due to the drag on the unsymmetrical vehicle oriented on the gravitational vector.
5. Other external torques such as meteor collisions, celestial bodies, magnetic and electric field

effects of the earth are small with respect to the major torques, and are therefore neglected. The estimated magnitudes of the external torques are presented in Figure 4-2.

Theoretical derivations for the external torques may be found in Ref. 6.

4.4.2 Internal Torques

Internal torques acting on the vehicle develop from the following sources:

1. Acceleration and decelleration of moving parts, such as propellant pumps for main engine, moving data link antennae, image motion compensation, etc.
2. Unsymmetrical shutdown of control motors.
3. Exhaust of APU or any depressurization within the vehicle system.
4. Unsymmetrical separation of Atlas C booster and the OSV during transition coast.

By proper arrangement and control, these internal torques can be utilized within the limits of practicability to assist in the attitude control of the vehicle, and therefore for the purpose of dynamic study will not be considered as perturbation torques. The estimated magnitudes of the internal torques are presented in Fig. 4-3.

4.4.3 Radiation Torque

Estimated variation of the radiation torque with time which is representative of one revolution around the world is shown in

Fig. 4-4. During the daylight portion of the orbiting, the torque due to the radiation pressure from the sun varies from the north pole to the south pole, becoming zero and changing sign as the vehicle passes through the plane of the ecliptic. This is because the vehicle maintains orientation with the gravitational vector at each point along the surface of the earth. On the dark side of the earth, the torque is assumed to vary linearly from the south pole to the north pole. This torque is the effect of heat dissipation from the vehicle (heat that is absorbed from the sun during the flight in the orbit over the southern hemisphere) and acts in the same direction during the flight on the dark side of the earth.

Some of the above torques will be applied periodically but in general the torques will be somewhat random and therefore the response of the vehicle must be studied on this basis.

4.5 Method of Solution of Linearized Equations

A general block diagram of the attitude control system is shown in Fig. 4-5.

Dynamic motions of the satellite for a two-torque wheel control system were obtained through an analog solution of equations (5), (6), and (7) and for each of the damping computer equations (8), (9), (10), and (11).

Numerical data substituted in these equations for the vehicle are listed in Table I, and for the attitude control system in Table II, for the configuration shown in Fig. 4-6. In the smaller vehicle the mass distribution is assumed to be symmetrical about the long (z') axis of the body. This, in effect, reduces the gravitational torque in the yaw mode to zero and the stiffness of the vehicle becomes a function of the angular momentum of the pitch torque drive unit only.

To simplify the study and provide a basis for comparison, motions of the vehicle were determined in response to a standard one inch-ounce torque, for the most part applied as a step function. Exceptions to this are the sinusoidal variations of the radiation pressure torques applied about the pitch axis and the sinusoidal variation of the data link antenna torques applied about the pitch and roll axes and the application of torques at random intervals about the pitch and roll axes Once the vehicle is oriented in the orbit, the actual torques acting on the vehicle would be considerably less than the one inch-ounce. However, the amplitudes of motion for any other torque can be approximated by linear extrapolation from the one inch-ounce torque results and still not jeopardize the significance of the results inasmuch as the non-linearities are small for the system studies.

Sec. 4. Table I

PHYSICAL CHARACTERISTICS OF VEHICLE AND ATTITUDE CONTROL COMPONENTS

	Payload* A (520 pounds)	Payload B (2520 pounds)
$I_{x'}$	1400 slug ft.^{2}	2275 slug ft.^{2}
$I_{y'}$	1400 slug ft.^{2}	2510 slug ft.^{2}
$I_{z'}$	130 slug ft.^{2}	271 slug ft.^{2}
F	1.0	0.9859
G	0.81	0.7968
J	1.0	0.9546
c.g	Sta. 180	Sta. 110

$\omega_c = 1.16 \times 10^{-3}$ rad/sec

* Payload is that of the satellite only.

Sec. 4. Table II CHARACTERISTICS OF ATTITUDE CONTROL SYSTEM

Assumed parameters:

$I_{cy'} = 0.05$ slug ft^2

$I_{cz'} = 0.005$ slug ft^2

$\zeta = 0.7$

Varied parameters:

[illegible]	0.5	to	1.5
$H_{cz'}$	0.075		(constant)
K_1	0.5	to	1.5
K_2	0.5	to	1.5

Time constants:

Integration feedback constants

$\tau_n' = 0.02 \approx 17$ secs.

$\tau_z' = 0.02 \approx 17$ secs.

$\omega_c = 1.16 \times 10^{-3}$ rads/sec.

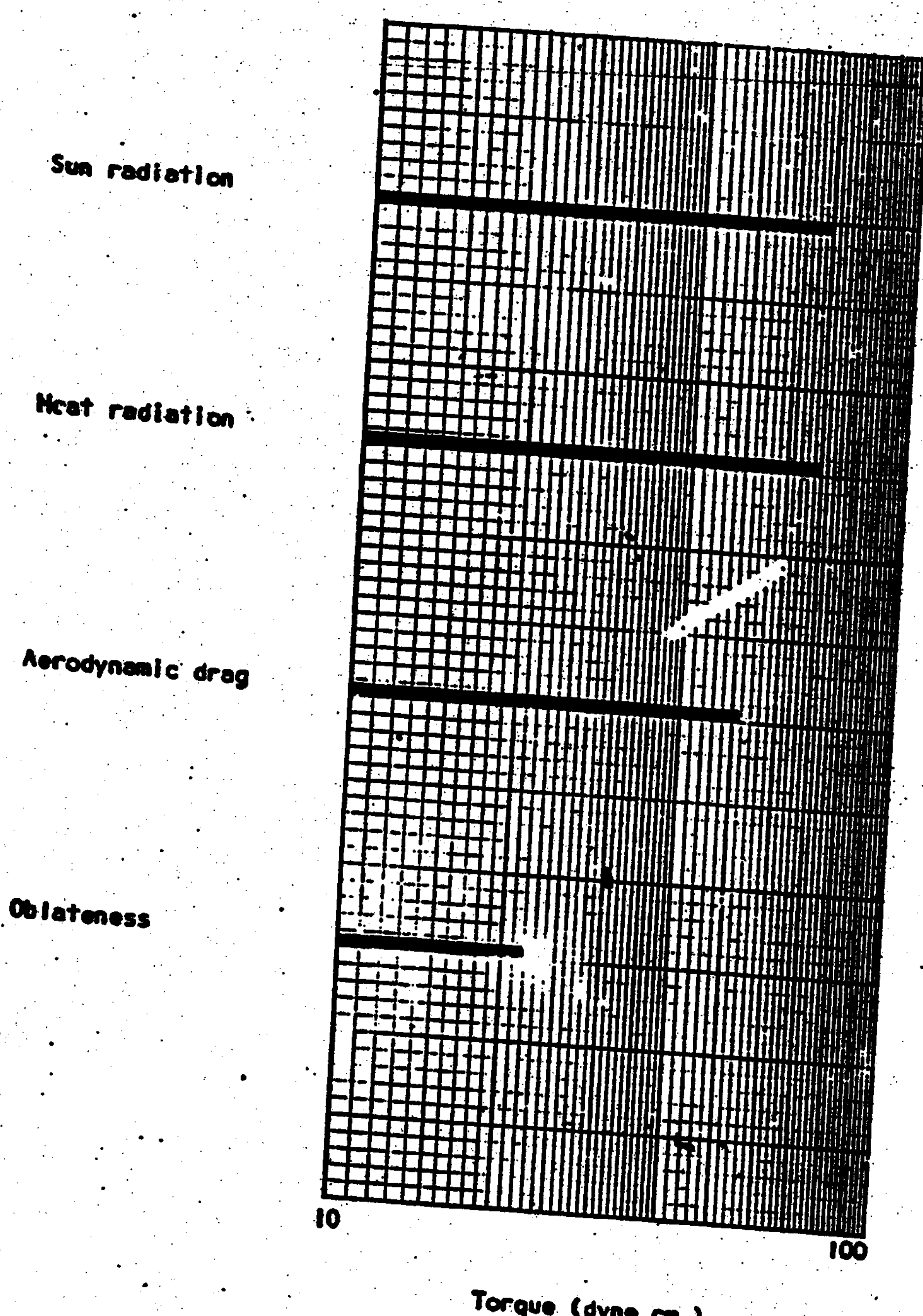


Fig. 4-2 Estimated Magnitudes of External Torques

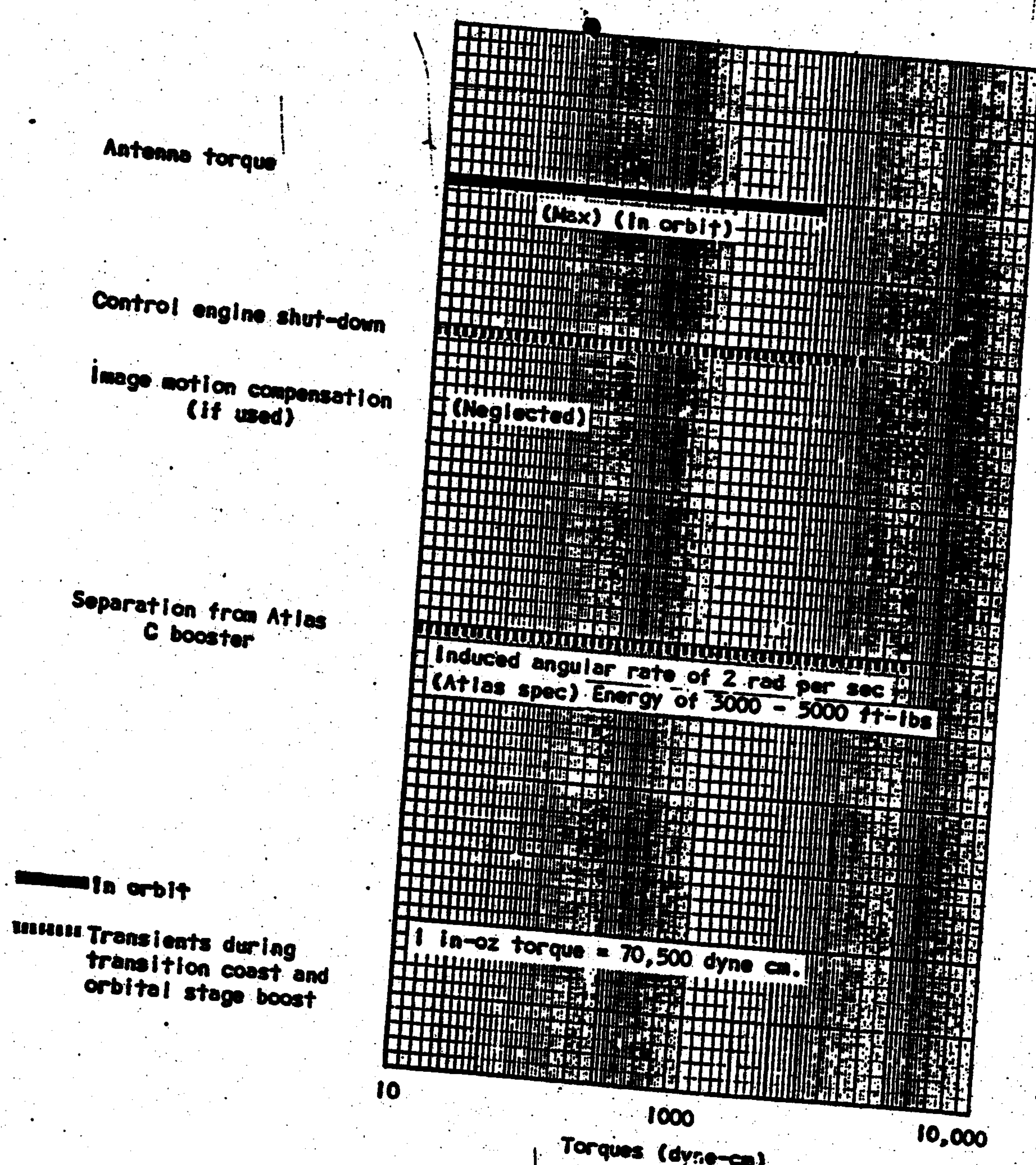


Fig. 4-3 Estimated Internal Torques to be Accounted for by the Attitude Control System

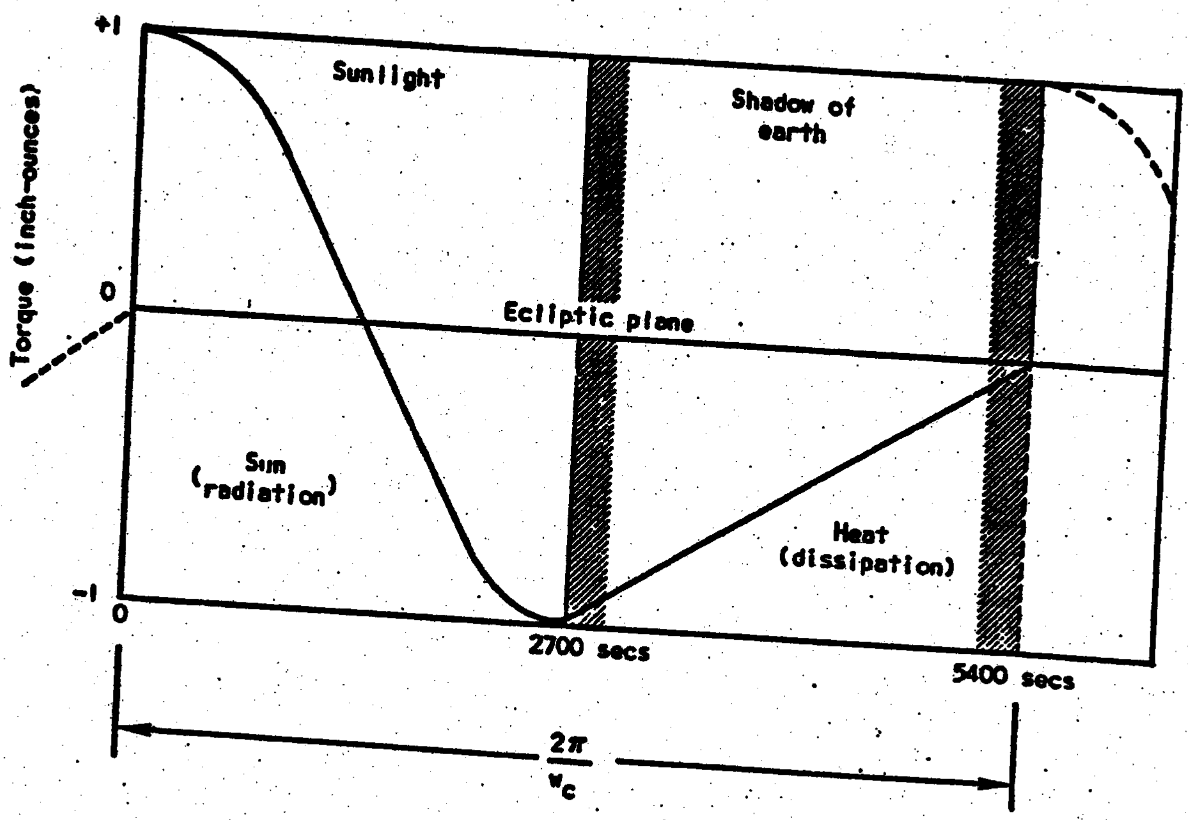


Fig. 4-4 Input Torque Variation for Simulator Study

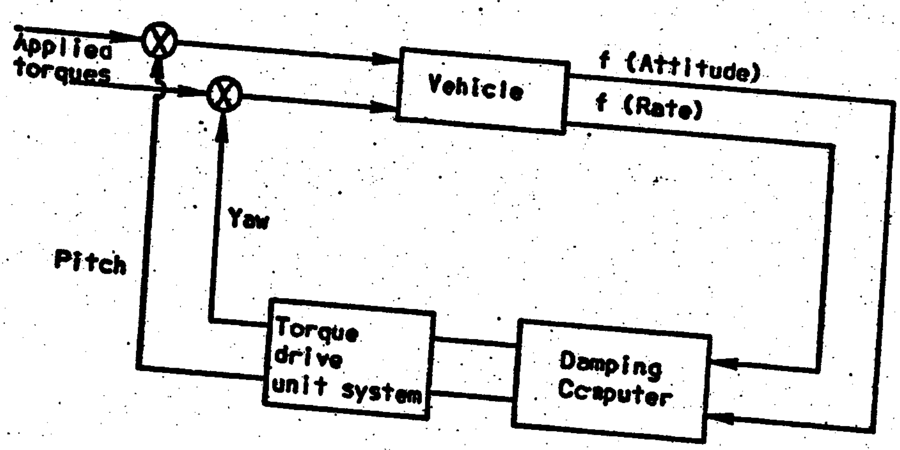


Fig. 4-5 Block Diagram of Attitude
Control-Vehicle System as Studied

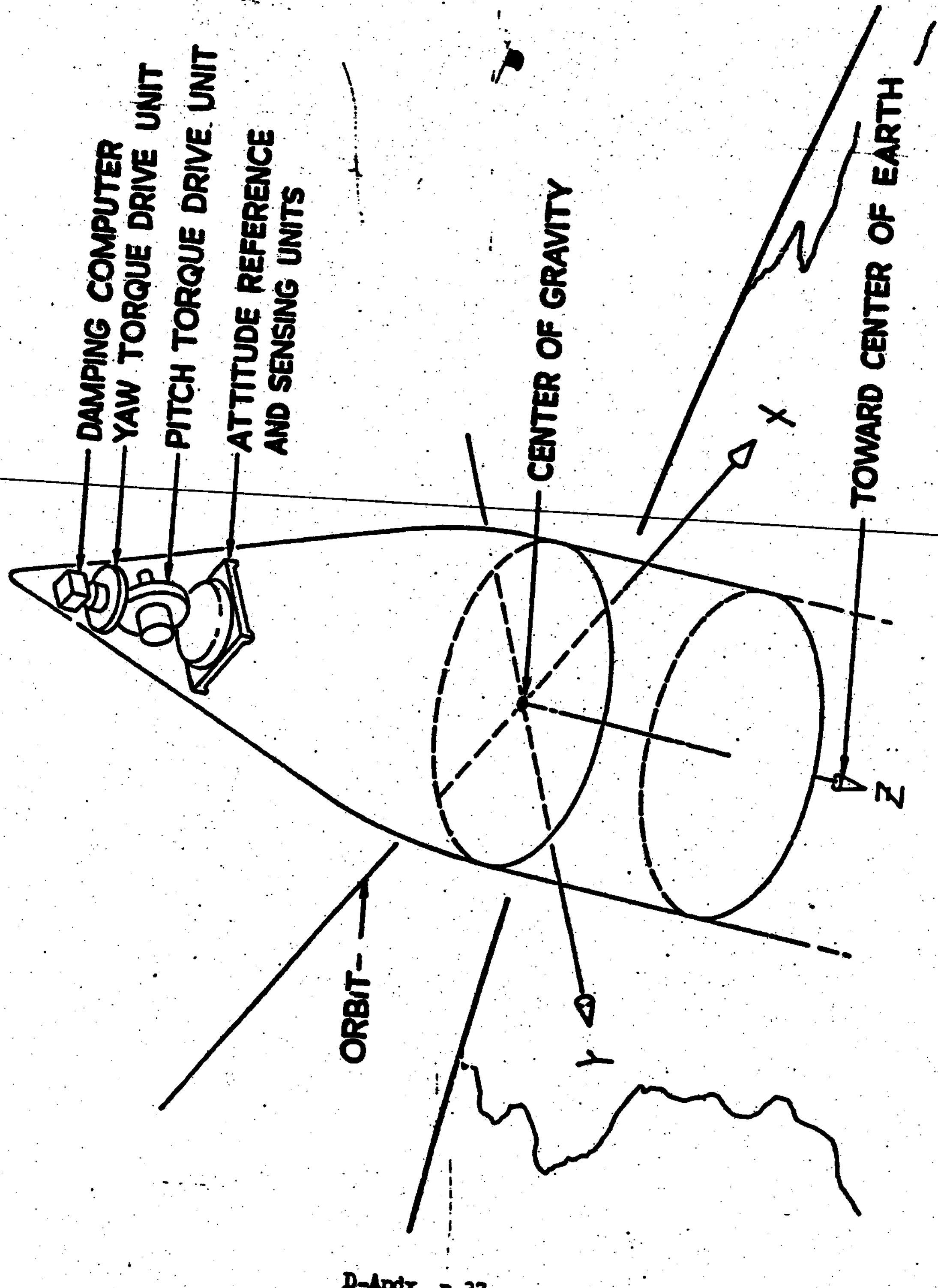


Fig. 4-6 Sketch of Configuration as Used in Attitude Control Study

4.6 System Dynamics

The damping computer has been assumed to be a "perfect" unit (this includes the torque drive units) in order to study the system without the uncertainties of sensing instruments and their effect on the damping of the oribiting vehicle.

The solution of the linearized system of equations for an idealized all rate damping computer (Figs. 4-7, 4-8 and 4-9) illustrates the damping obtainable when $K_1 = H_{cz'}$, and $K_2 = I_{z'}\omega_c$. This system senses rate about each of the three body axes. It should be pointed out that varying K_1 and K_2 as shown in Table II does not appreciably effect the degree of damping.

The dynamic characteristics of the system were obtained from the transient oscillations by observing the changes in the period (coupled and uncoupled) the time constant associated with the motion. The time constant $(\frac{1}{\zeta\omega})$ in this report is represented by the time to reach 37 per cent steady state amplitude. The dynamic characteristics obtained from this solution are shown in Figs. 4-10 and 4-11. These data show the effect of various values of the constant momentum (H_o) in the pitch mode on the dynamics of the complete system for a given yaw total momentum ($H_{cz'}$) and indicates a value of total momentum about the pitch axis that gives a small roll-yaw time constant. Varying the yaw momentum ($H_{cz'}$) from .075 to 7.5 and maintaining a constant pitch momentum (H_o) does not appreciably affect the period and the damping of the resultant motions.

If the pitch oscillation were left undamped, the vehicle would oscillate two cycles for one revolution around the earth. For constant pitch and yaw momentum values corresponding to data shown on Figs. 4-8 and 4-9, (coupled yaw and roll motion), the vehicle oscillates 2 to 3 cycles for one revolution around the earth. This long period motion has the advantage of being accompanied by small rate changes of angular motion which is good from an image motion compensation point of view but not so good from the point of view of the sensing elements involved in the control system.

Application of the assumed periodic torque shown in Fig. 4-2 to the pitch mode of the equation of motion results in the motion illustrated in Fig. 4-12. The initial transient oscillation in the roll-yaw mode is due to a steady state error in roll which is a function of the yaw torque drive momentum. Energy of the yaw torque drive unit is dissipated in a steady roll angle error of 6×10^{-3} radians per inch-ounce of torque. Maximum pitch mode angular amplitudes and rates are about $\pm 2.5 \times 10^{-3}$ radian and $\pm 2.8 \times 10^{-6}$ rads. per sec. respectively. The maximum yaw angular rates due to the steady state roll error are about $\pm 15 \times 10^{-6}$ rads. per sec. whereas the roll angular rates are about $\pm 5 \times 10^{-6}$ radians per second.

Simplification of the damping computer so as to sense only pitch and yaw rates (see block diagram Fig. 4-13) produces the motions illustrated in the left hand portion of Figs. 4-14 and 4-15.

roll-yaw damping characteristics are slightly reduced but not beyond the point of not being acceptable; the time constant decreases approximately 90 seconds, (see Fig. 4-11.) Elimination of one gyro provides a reduction of the power required to operate the attitude control system.

If rate feedback is to be supplied from rate gyros, it is quite apparent that the very small angular velocity makes it extremely difficult to obtain adequate sensing. Existing floated rate gyros can readily detect angular velocities as low as 2×10^{-5} radians/sec. (4 sec./sec). Maximum angular displacements of 2.5 milliradians per inch-ounce of torque are indicated.

4.7 "Dead Zone"

As the sensitivity of existing gyro units is on the order of 20×10^{-6} radians per second and the expected vehicle "idealized" angular velocities less than this value, the combined dynamic system will respond as a "dead zone" system over the range $\pm 20 \times 10^{-6}$ radians per second. Such a system, if disturbed with sufficiently small inputs, will oscillate (output rates are less than 20×10^{-6} rads per second) as a spring mass system at amplitudes proportional to the constant input. When the input step is applied at random intervals, the amplitude builds up with time until the dead zone amplitudes are reached and the damping of the system becomes effective. Then the oscillation is maintained at some small amplitude slightly greater than the dead zone.

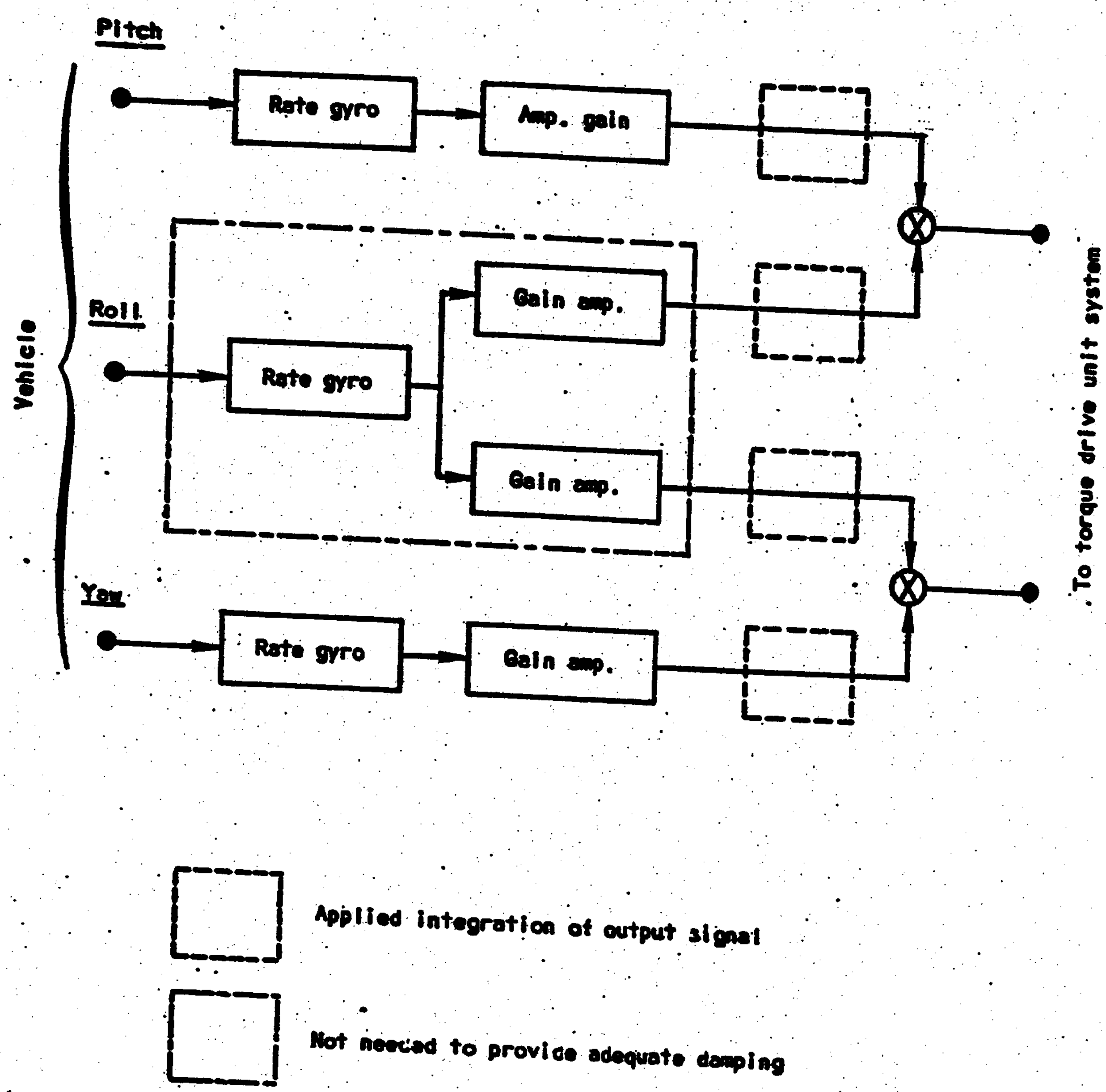


Fig. 4-7 Damping Computer with 3-rate Input (or Integrated Rate)

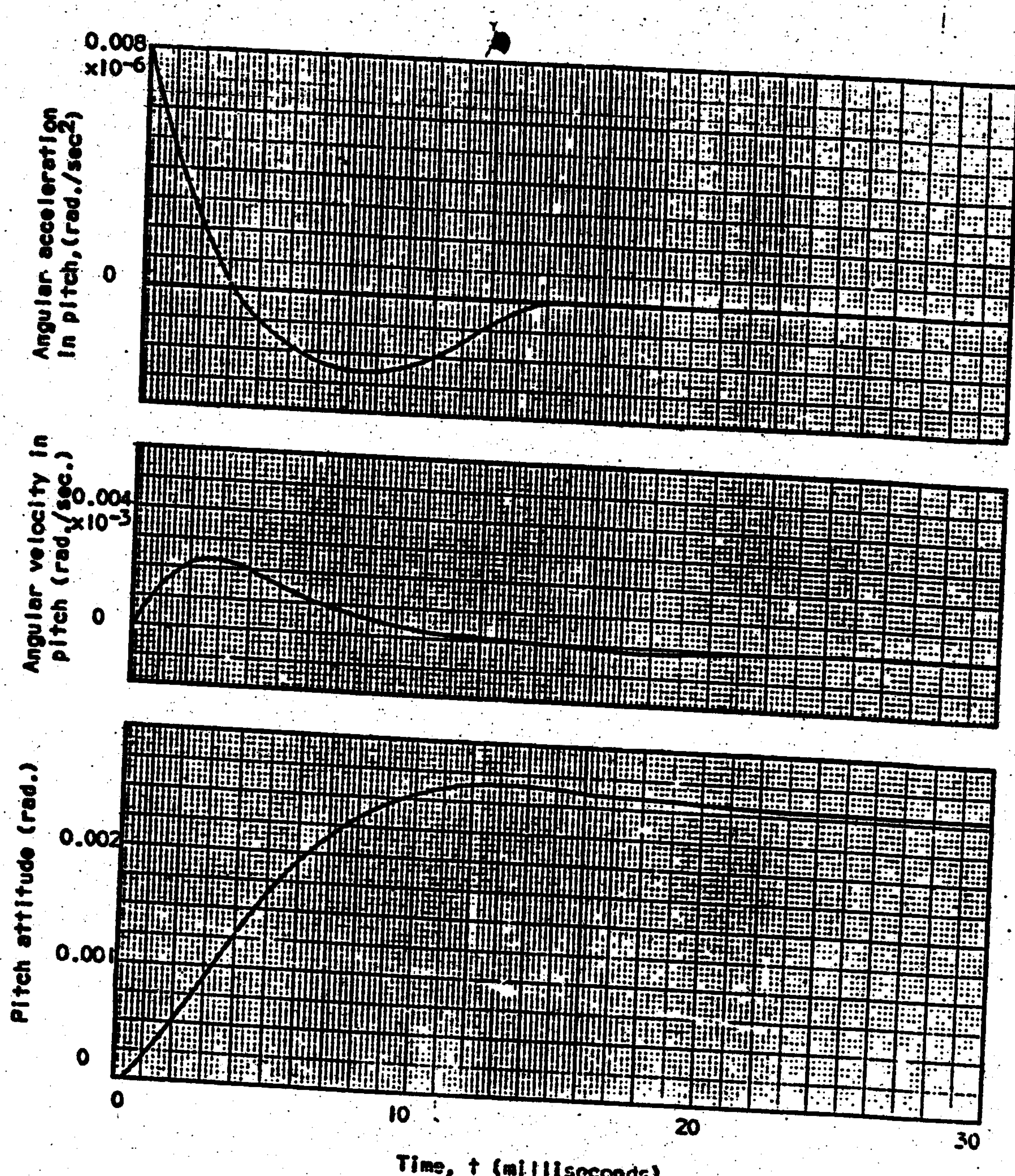


Fig. 4-8 Calculated Time History of Pitch Mode Rotation for OSV (H_o = 2.3) in Response to 1 inch-ounce Torque

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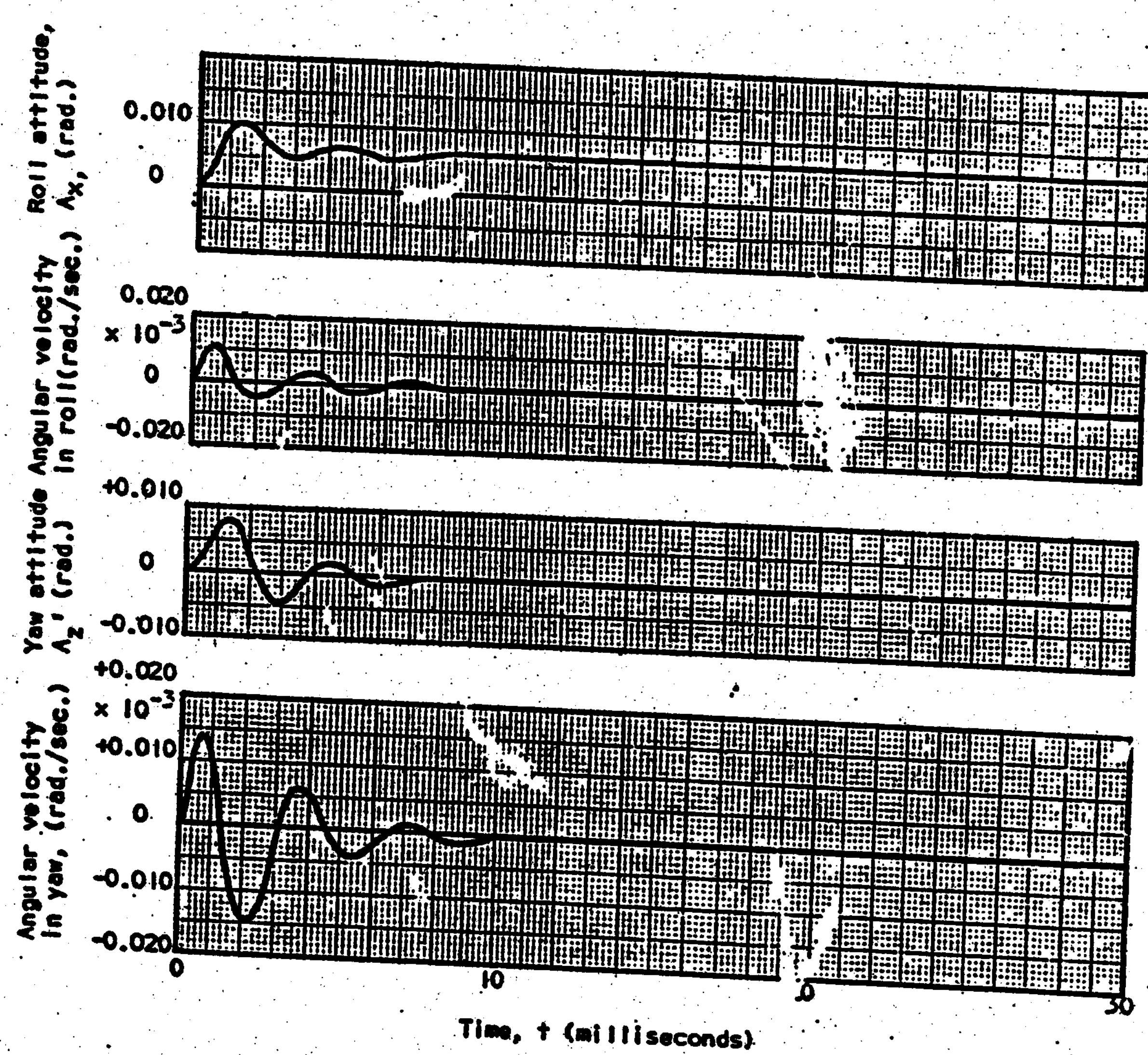


Fig. 4-9 Calculated Time History of Roll-Yaw Mode Rotation for OSV (Icy' ω_o = 2.3) in Response to 1 inch-ounce Torque Applied in Pitch Mode

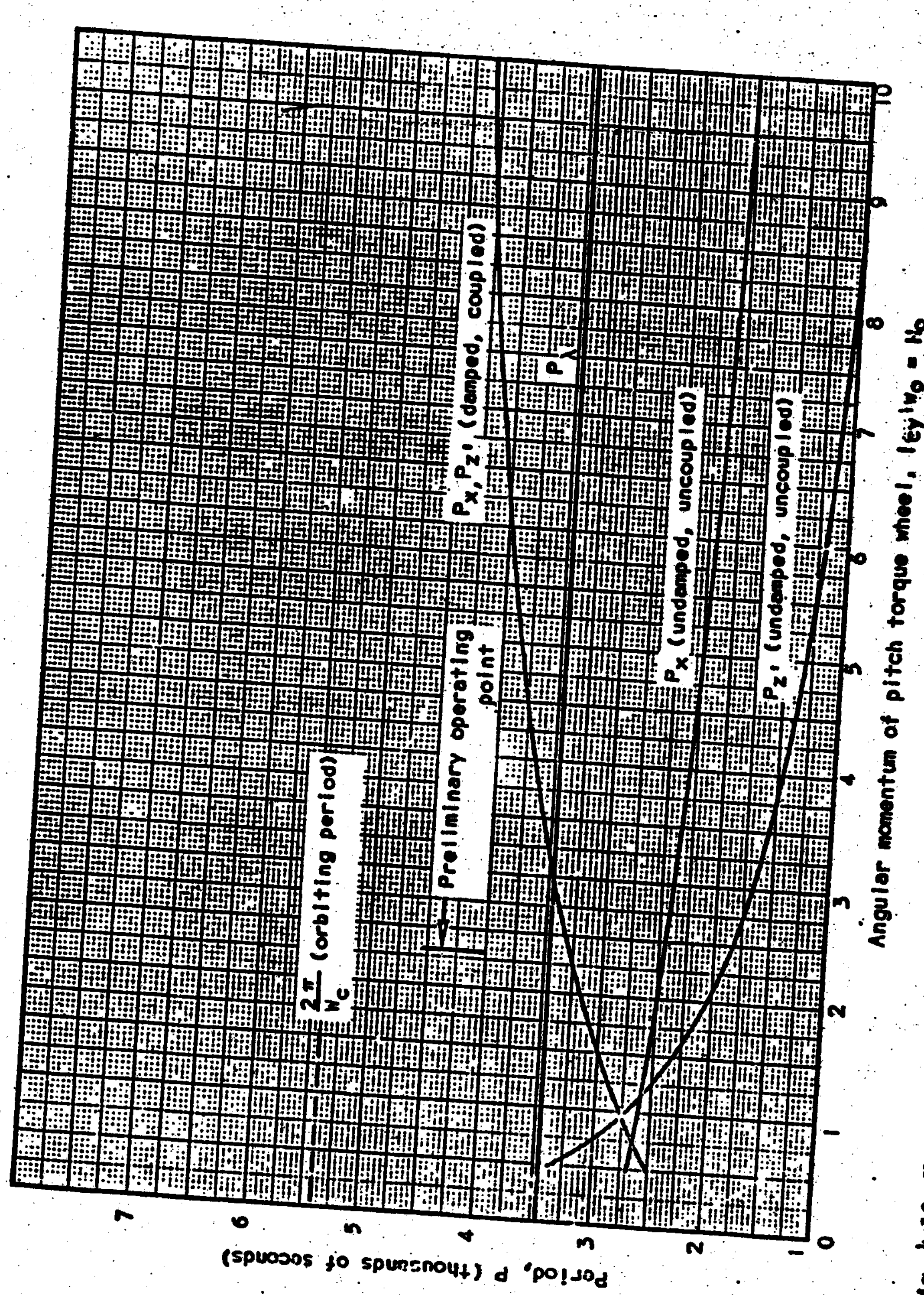


Fig. 4-10 Variation of Oscillation Periods in the Three Vibration Modes (Coupled and Uncoupled Modes)

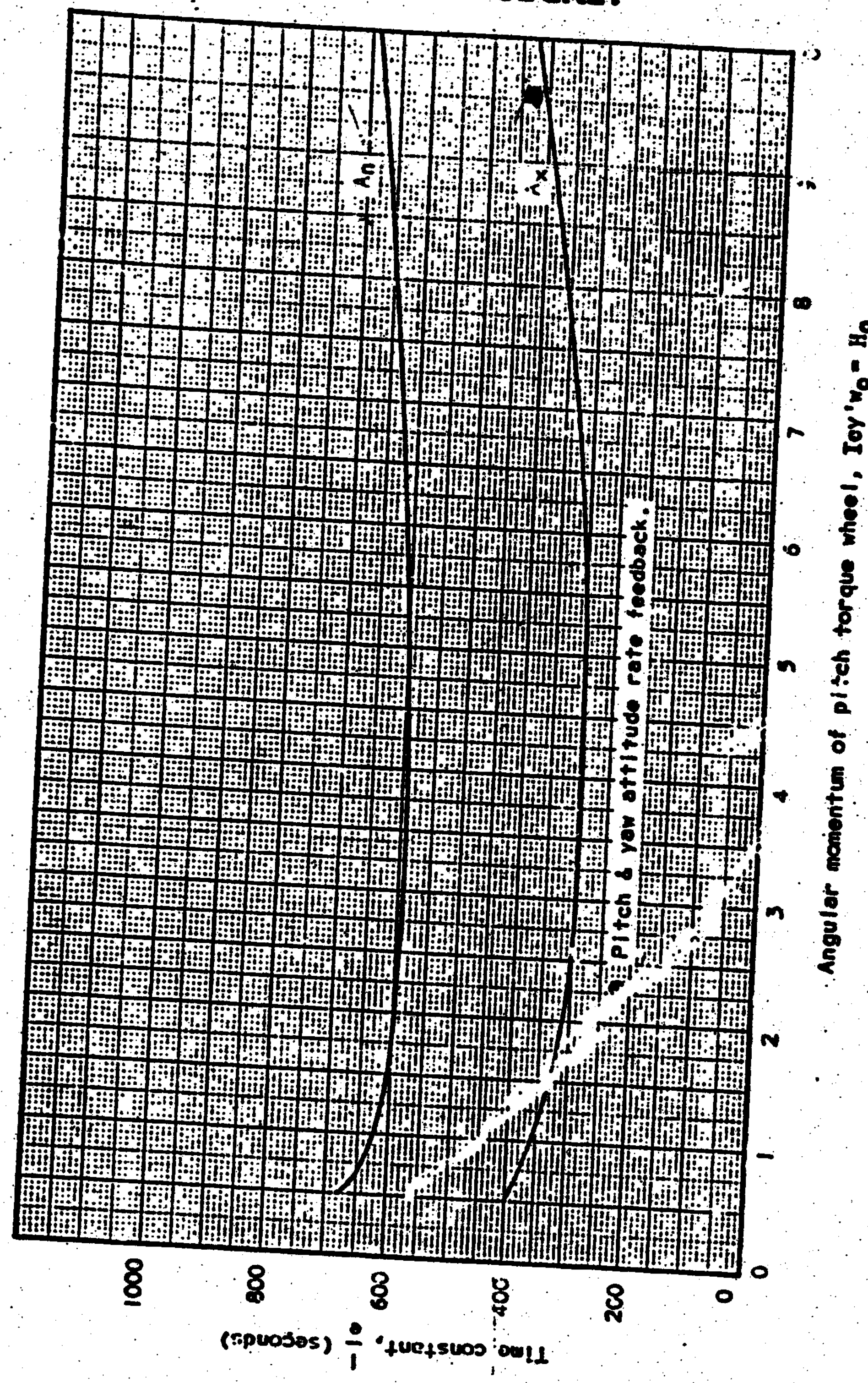


Fig. 4-11 Variation of the Time Constant to 37 per cent Steady State Amplitude in Pitch and Roll Modes.

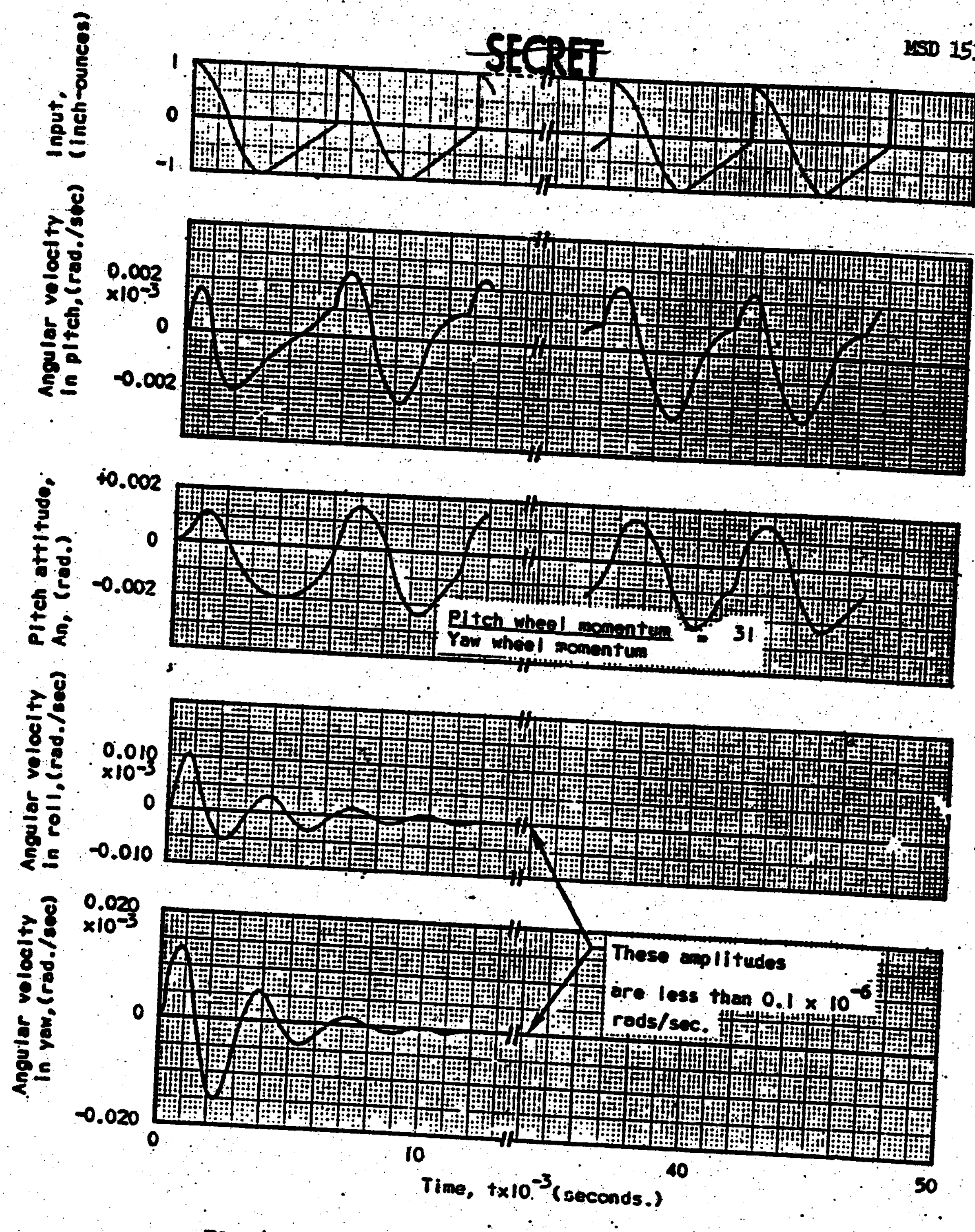


Fig. 4-12 Calculated Time History of Angular Motion of OSV (H_y' W_0 = 2.3) in Response to a Periodic Torque Input

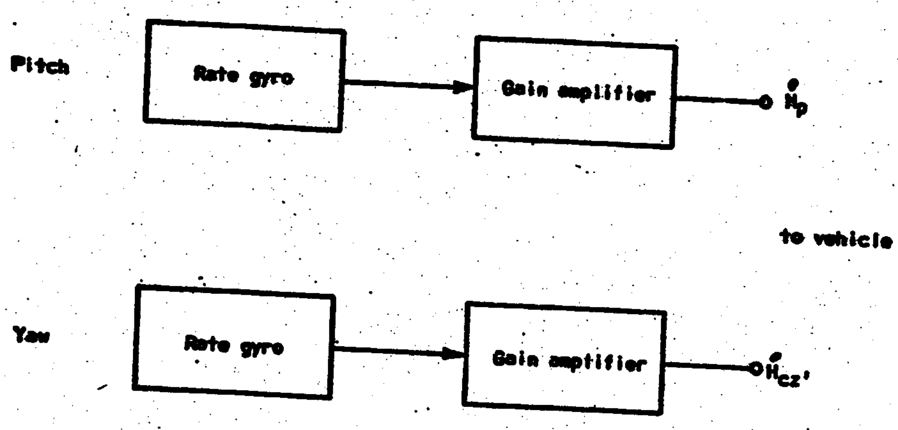


Fig. 4-13 Damping Computer with 2-rate input

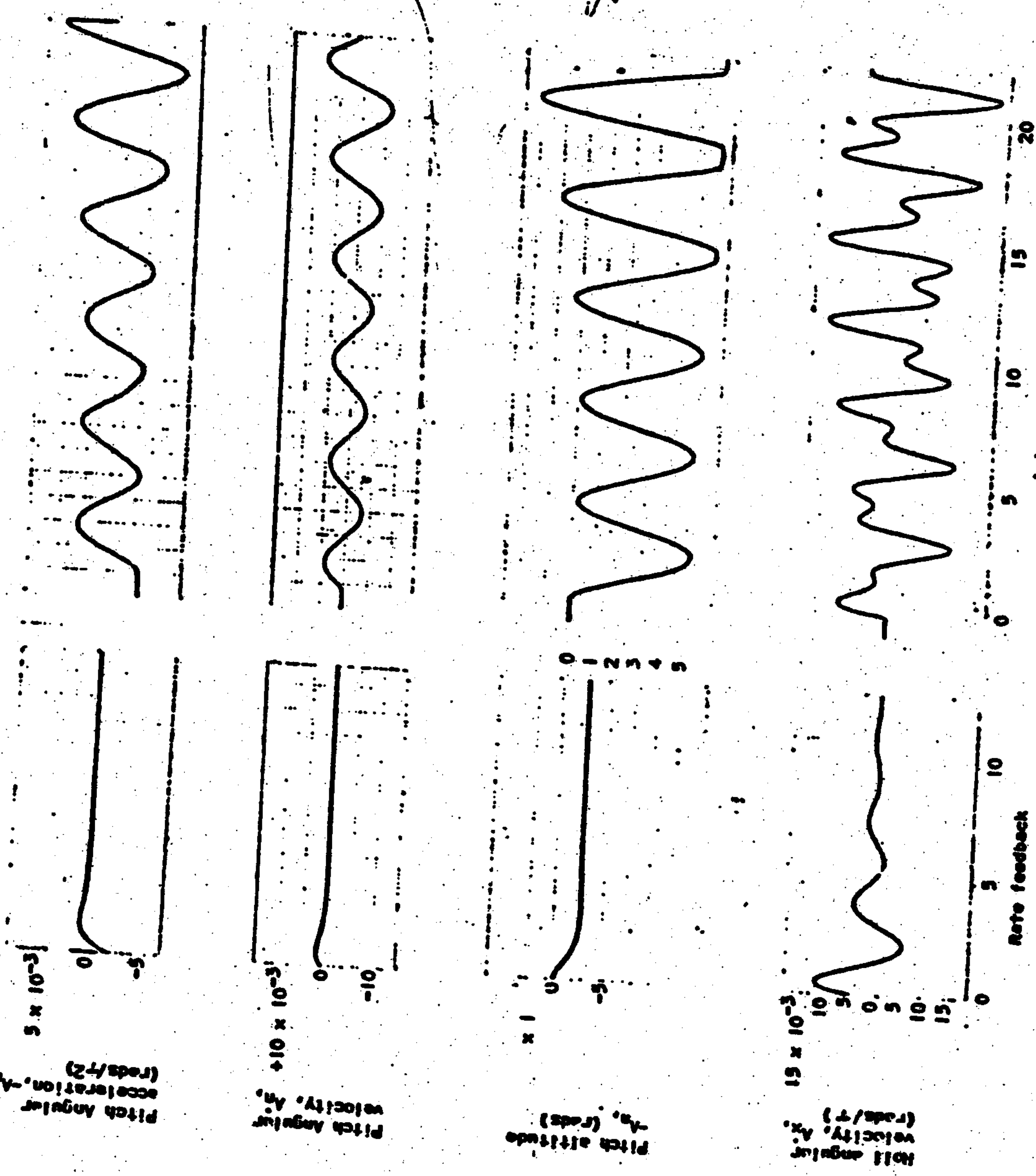


Fig. 4-14 Solution of Attitude Control with Rate Feedback and Integrated Rate Feedback (Sensing $\dot{A}_n$ and $\dot{A}_x$.)

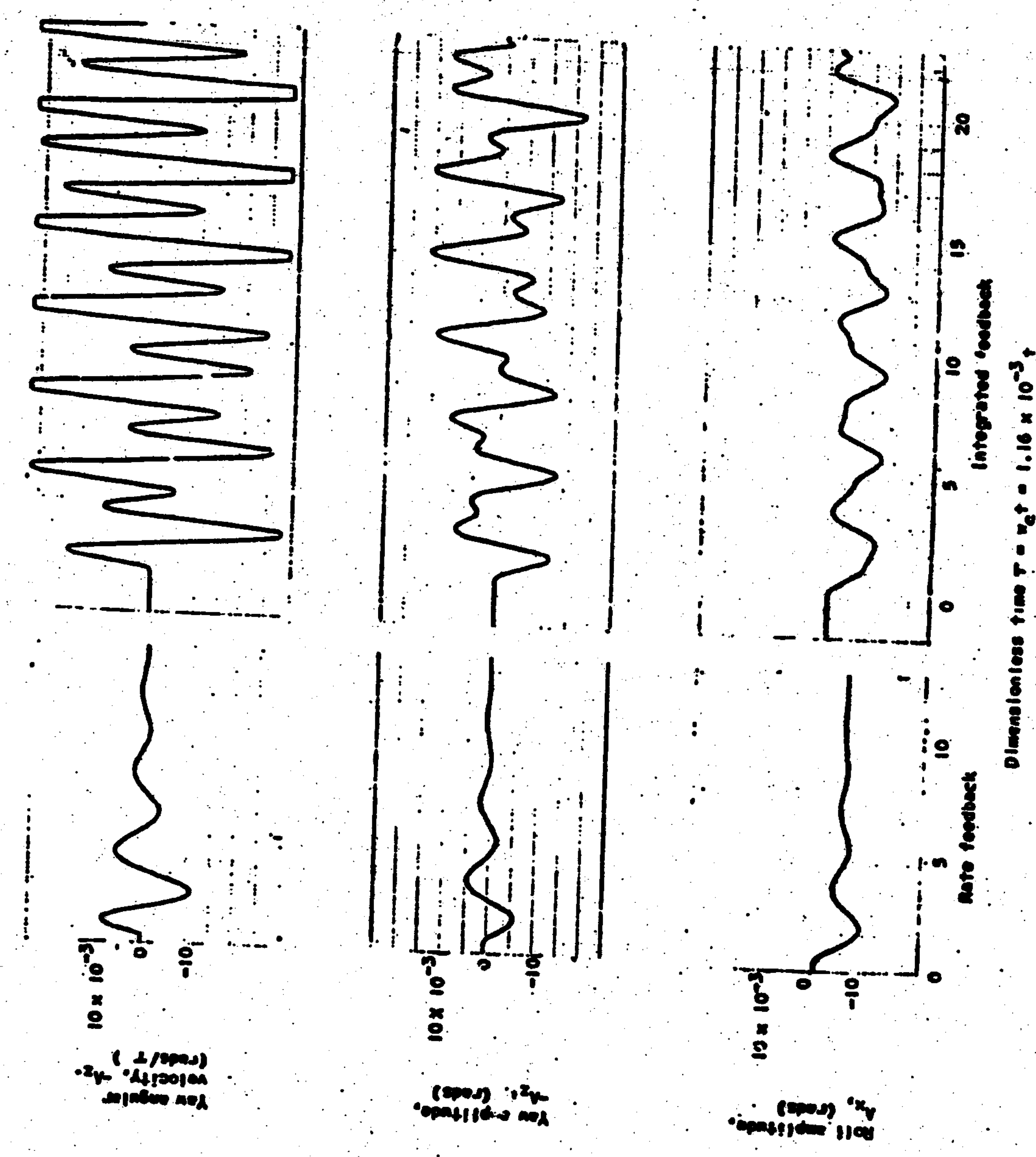


Fig 4-15 Solution of Attitude Control with Rate Feedback and Integrated Rate Feedback (sensing $\dot{A}_n$ and $\dot{A}_z$)

The solution of vehicle equations (5), (6), and (7) and damping computer equation (10) with dead zone (see Fig. 4-16) illustrates the effect of dead zone. Initially the input is insufficient to excite the vehicle to an angular rate of greater than $\pm 2.8 \times 10^{-6}$ rad. per second. However, with the application of random inputs the amplitude builds up beyond the dead zone after which the system damps to angular rates slightly greater than the dead zone. Two facts are obtained from this; first, the amplitude in excess of the dead zone is reduced to almost the dead zone, and second, for a spring mass system (zero damping) the amplitude of response increases without limit for a random type input.

4.8 Other Torque Drive Unit Computers

Other methods of stabilizing the vehicle in orbit were studied as idealized systems.

A study of the equations (1), (2) and (3) reveals that if only position feedback (see block diagram Fig. 4-17), is used in any mode the resultant motion will be one of constant amplitude for a step input. This is not undesirable if the input remains constant. However, if the input is random (and it may well be) the amplitude of the oscillation diverges as a function of time. This more or less obviates the desire of using position feedback and requires that the damping computer be based on angular rates about two or three axes.

The solution of the equations of motion using integrated rate feedback (see Equations (11) and Figs. 14 and 15 - right hand side) illustrates the expected motion of the system at an integration time constant of $\tau' = 0.02$ ($\frac{\tau'}{\omega_c}$ = 17 seconds). Larger time constants aggravate the motion. This response, like the position gyro and dead zone response, was observed to diverge with increasing time when subjected to random inputs.

4.9 Torque Drive Units

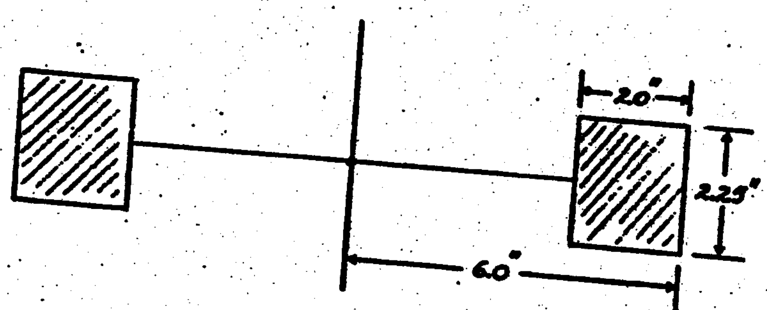
The torque drive units, Fig 4-18, consist primarily of a drive motor accelerating or decelerating a flywheel of specified dimensions. Initially the torque drive unit characteristics were described as angular momenta and parametrically studied with specific damping computer schemes to determine magnitudes that would give desirable vehicle damping. Results of the study thus far conducted were used to approximate the size of the torque drive flywheel and the angular rate required for satisfactory attitude control. The constant angular momentum in pitch (H_o) consists of a summation of all the momenta of all rotating parts aligned with the pitch body axes plus the torque wheel momentum. If the number of rotating parts is small then the difference must be made up by the torque drive wheel.

Initially the vehicle will have a few large momentum components and therefore in order to effect a higher frequency in roll-yaw mode the pitch torque drive unit must provide most of the required momentum.

The total momentum of the torque drive unit is the sum of the motor momentum (which is constant) and the momentum of the flywheel.

For all practical purposes the momentum of the motor will be assumed to be small. To satisfy an H_o of 2.3 lb·ft·secs. assume a torque wheel with a moment of inertia of 0.05 slug-ft.^{2} Such a wheel would then be rotating at 440 rpm.

A possible size of pitch torque wheel is shown in the following diagram:



Material: Aluminum

Moment of inertia: 0.05 slug-ft^2

The yaw torque wheel would be smaller because the moment of inertia is assumed to be one-tenth the pitch mode moment of inertia. With the presence of additional rotating components the size of the torque wheel can be reduced but it cannot be eliminated since variation of its angular rate provides torques for damping the vehicle perturbed motions.

To prevent the possibility of the torque drive motor from saturating, a speed regulator device would be used to measure the difference in angular rate from a reference angular rate and supply a correction command signal to the motor. At the same time a signal would command an independent torque (eg., gas jet or some similar device) equal and opposite to the torque produced by adjustment of the torque wheel. This operation would occur at widely spaced intervals so as not to interfere with the normal operation of the attitude control system.

4.10 Response to Random Inputs

The study reported treats the response of several control system configurations to step inputs. In reality the magnitude, duration and time of application of the torques will be random in nature. The response for systems which are mathematically similar to the attitude control system has been studied exhaustively in two RAND Reports (Ref. 7 and 8). These reports show how, in the absence of damping, the expected dispersion of the oscillatory amplitudes grow with the square root of time. When damping is introduced into the system the response to a random (stochastic) input is bounded with a bound established by the rms level of the disturbance, the correllation time of the disturbance, and the natural frequency of the vehicle.

Accordingly, after examinations of the referenced analytical results as well as several experiments with torques applied at random intervals, it appears that it is essential that the attitude control system be based on a rate instrumentation in order to introduce damping.

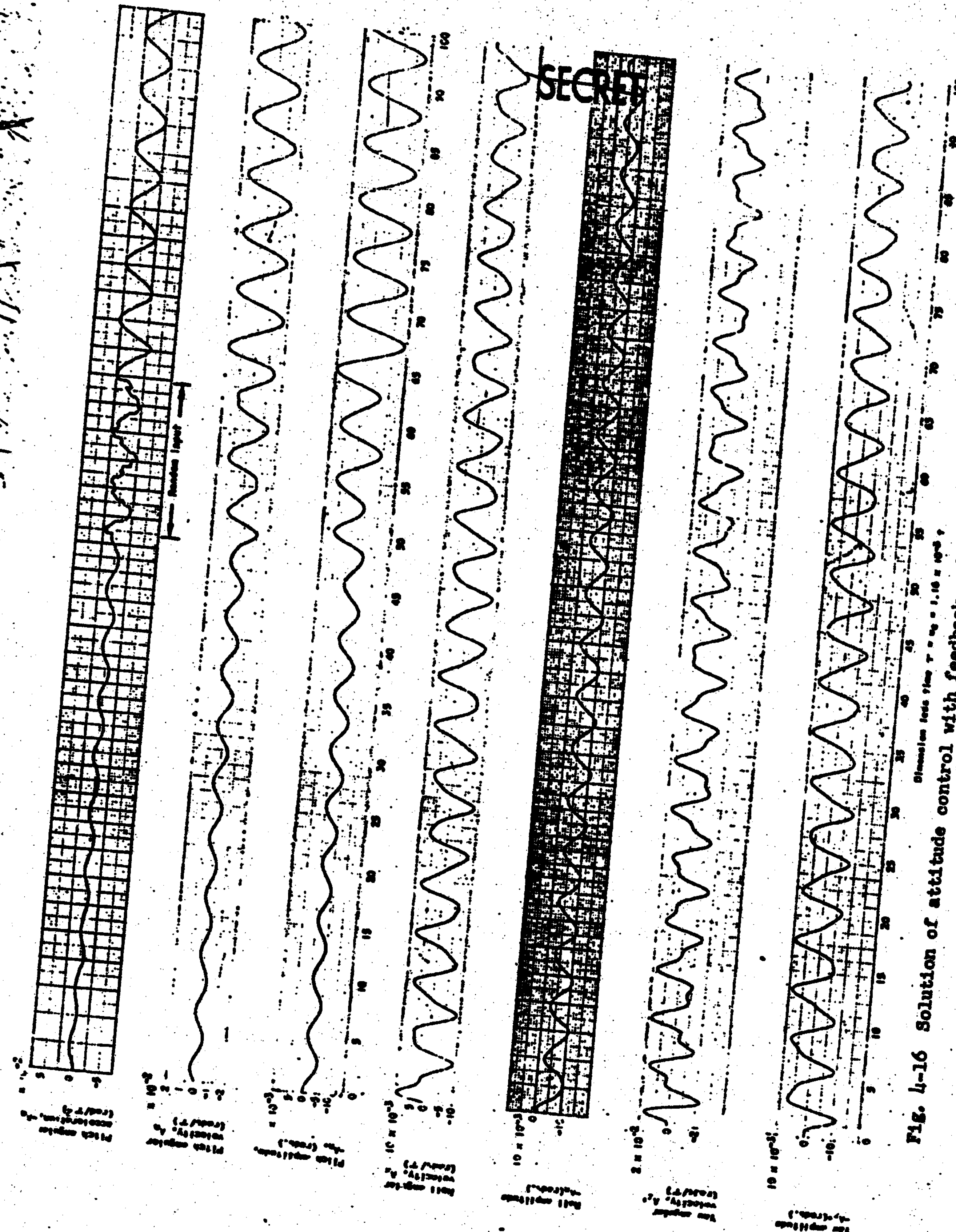


Fig. 4-16 Solution of attitude control with feedback and rate gyro deadzone of ± 18 x 10^{-6} rads/sec.

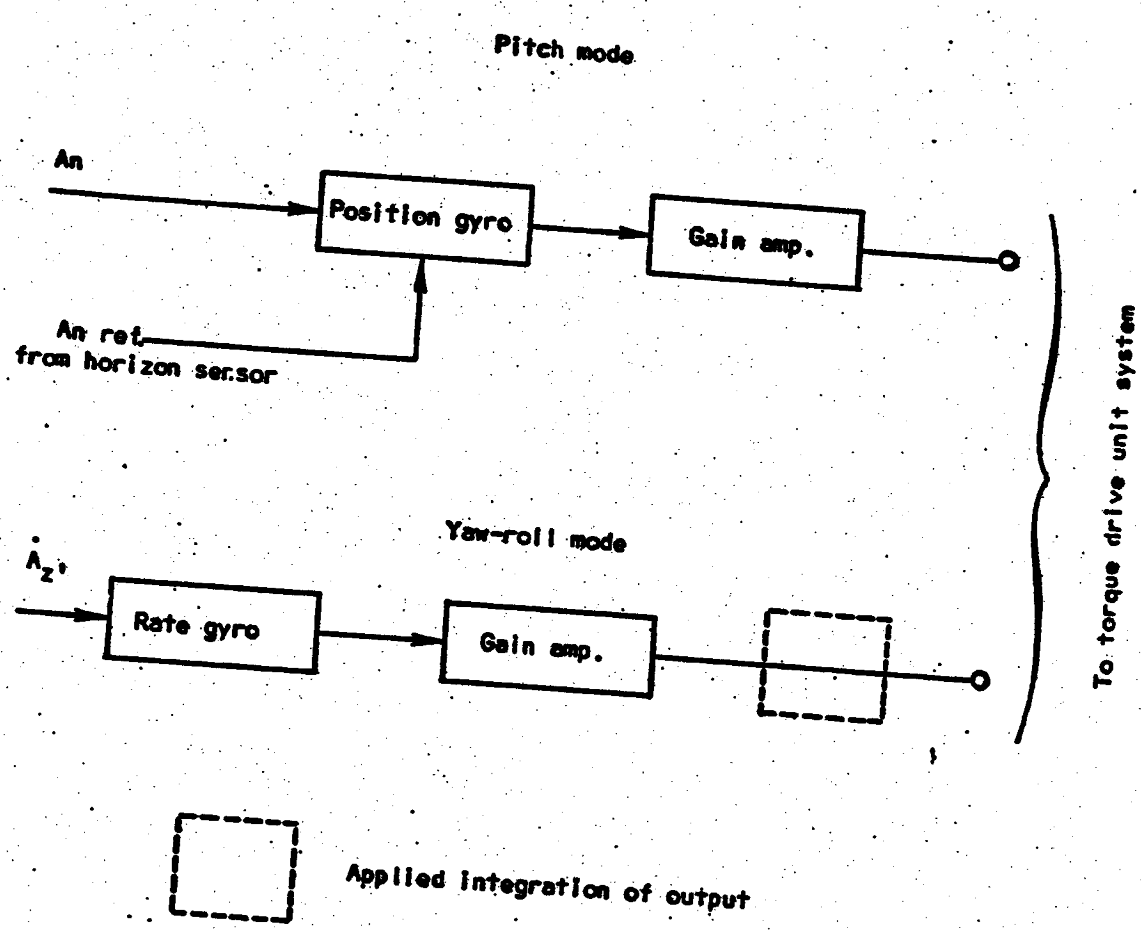


Fig. 4-17 Position and Rate Damping Computer
(or Position and Integrated Rate)

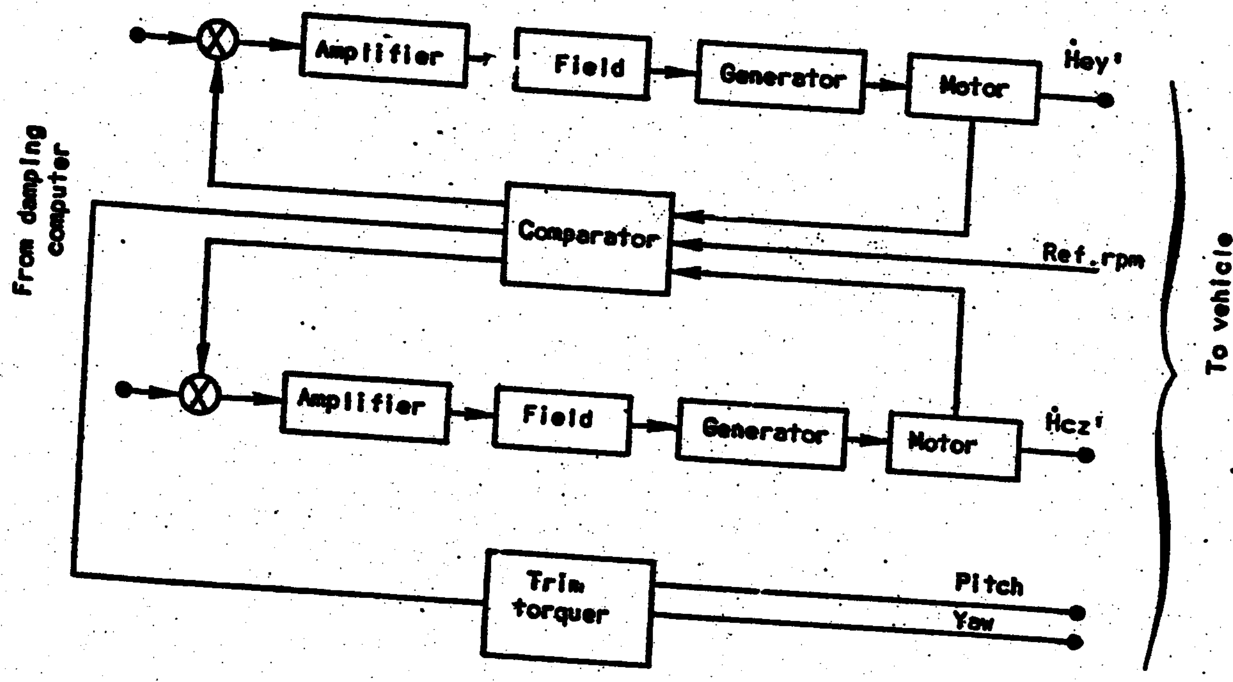


Fig. 4-18 Block Diagram of Torque Drive Units and Speed Governor

4.11 Regression of the Orbit Plane

The regression rate of the orbital plane acts to disturb the roll and yaw modes of the vehicle. The resolution of the regression rate vector and the pitch angular momentum vector of the vehicle produces an angular torque in the ecliptic plane and pointing away from the sun, i.e., it is radial. For the orbiting conditions of the proposed vehicle the torque applied is 1.1×10^{-4} inch ounces, (approximately 8 dyne-cm). The body axis system rotates with respect to the torque vector which therefore appears in the roll and yaw modes cyclically at amplitudes of 8 dyne-cm.

However, the induced dynamics would in effect be averaged out inasmuch as the uncertainties in the sensing instruments ($\pm$ 20×10^{-6} radians per second for available rate gyros) are such as to cause the vehicle to oscillate continually at the very low noise level of the instruments. Perturbations above the noise level of the instruments are damped out by the attitude control system.

4.12 Special Points of Consideration

From the foregoing study a number of general remarks can be made concerning requirements and conditions that satisfy desired stability and damping.

1. The major portion of the rotating components should be aligned such that their axes of rotation are parallel to the pitch axis. Rotating parts aligned parallel to the yaw axes should be kept to a minimum. Components

should be so aligned as to result in the desired constant angular momentum for satisfactory stability. Some of the components may be required to cancel the effects of other components.

2. Torques from angular acceleration of rotating components (e.g., data link antennas and servos) should not exceed a noise level of 2 inch-ounces. The attitude control system serves to balance out the torques that are applied, but the visual reconnaissance requirements for high photographic resolution during orbiting place limitations on the allowable torques. During transition and orientation these visual reconnaissance requirements do not exist.

3. At widely spaced intervals the torque drive units should be restabilized to a specified energy level. The right combination of torques over a period of time will cause the torque drive units to saturate. Readjustment of the torque drive wheel rate must be counter-balanced by an independent torque (e.g., gas jets from pressurized tanks) so as not to cause a disturbance to the vehicle.

4. The guidance and control equipment should be capable of satisfactory operation for a long period of time at an environmental temperature condition of 5 degrees centigrade. This environmental condition is discussed fully in Appendix of subsystem A.

5. HORIZON SENSOR

5.1 Introduction

In view of the function of a satellite station, it is all important to specify and control its orientation relative to the earth. In particular, two specified perpendicular axes of the satellite must be aligned with the velocity vector and the earth's local vertical. Previous work indicates how the vehicle can be controlled so that a specified axis of it will always remain perpendicular to the plane of the orbit, within an accuracy of a few milliradians. But this leaves open the possibility of pitching motion in the plane of the orbit. Since the initial orientation of the vehicle is achieved with the use of a gyrostabilized table, the pitching motion can be controlled by a "pitch gyro". This method incorporates a 90-minute precessing device, plus a gyro containing inherent drifts. Therefore, it is desirable to have an auxiliary device to monitor the system and correct for any drifts that may occur during the lifetime of the vehicle.

Assuming that the vehicle is sufficiently controlled in roll and yaw, a kind of vertical can be obtained by observing the fore-and-aft horizon. This vertical information can then be used to monitor the gyro pitch control system and reduce its drifts to an acceptable amount.

The system proposed is shown in Fig. 5-1. The fore-and-aft horizon is reflected from a mirrored prism through separate lens systems onto infrared detectors. If the center line of the fore-and-aft lens systems is not aligned with the bisector of the angle formed by the fore-and-aft line of sight to the horizon, then the signal received by each detector will be different. This difference is a measure in sign and magnitude of the deviation from the vertical. To permit the use of ac amplification, a chopper is inserted in the lens system.

Fig. 5-2 shows a block diagram of the system. The horizon signals are projected onto the detectors through the chopper. The signals obtained from the detectors are first amplified and then subtracted. In order to restore the dc level of the signal, a synchronous detector using a signal obtained from the optical chopper, is employed. The average value of the resulting wave form is then proportional to the deviation from the local vertical, α.

5.2 Detector

For a missile at an altitude of 300 n. miles, the slant range to the horizon is 1570 miles. If the atmosphere is assumed to extend to an altitude of 100 miles, then the transition from earth to sky at the horizon subtends an angle of 3.5 degrees. If a field of view is chosen which is a 1/6 degree wide and 4.5 degrees high as in Fig. 5-3, the central portion will be taken up by the atmosphere; this leaves 1/2 degree in height for the sky and 1/2 degree for land. The field of view is chosen to be narrow in order to minimize the effect of the curvature of the earth and tilting.

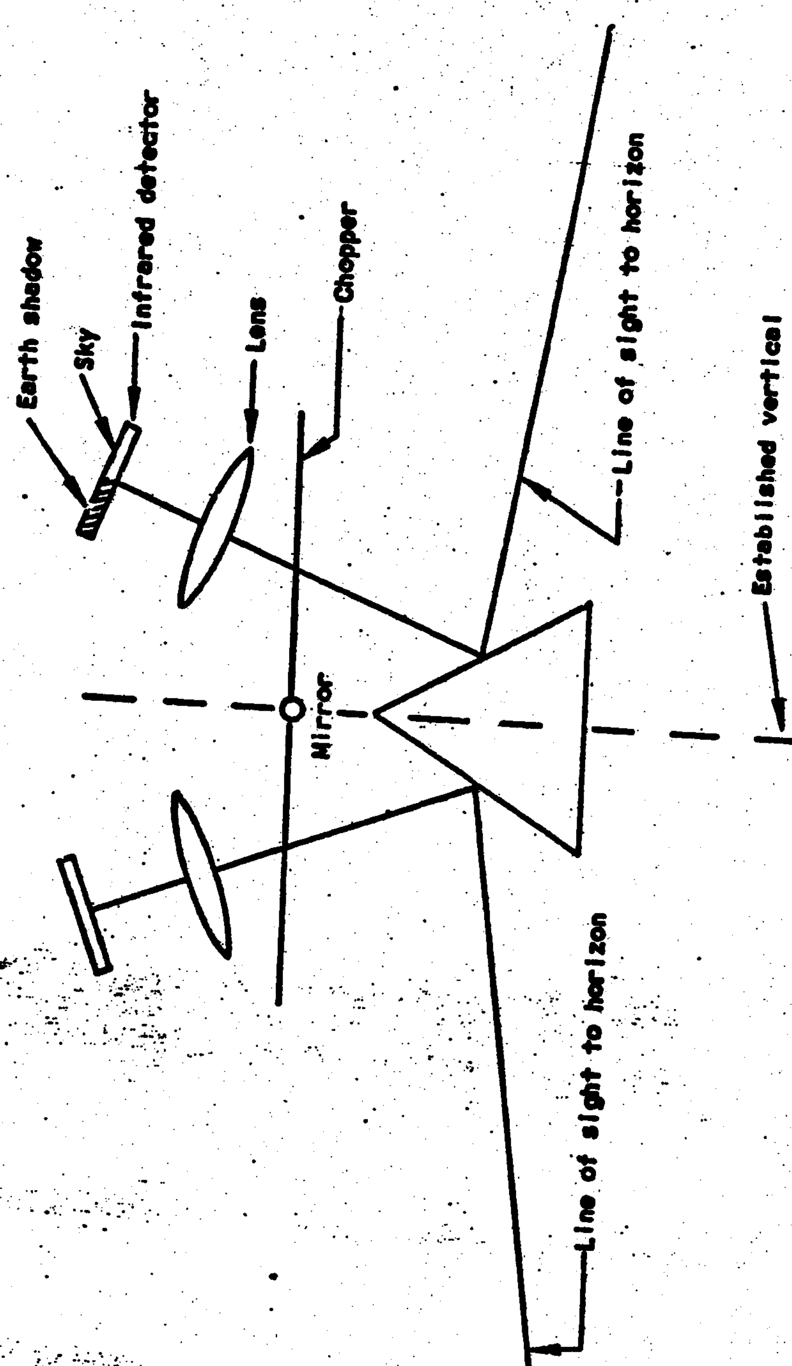


Fig. 5-1 Schematic of Horizon Sensor

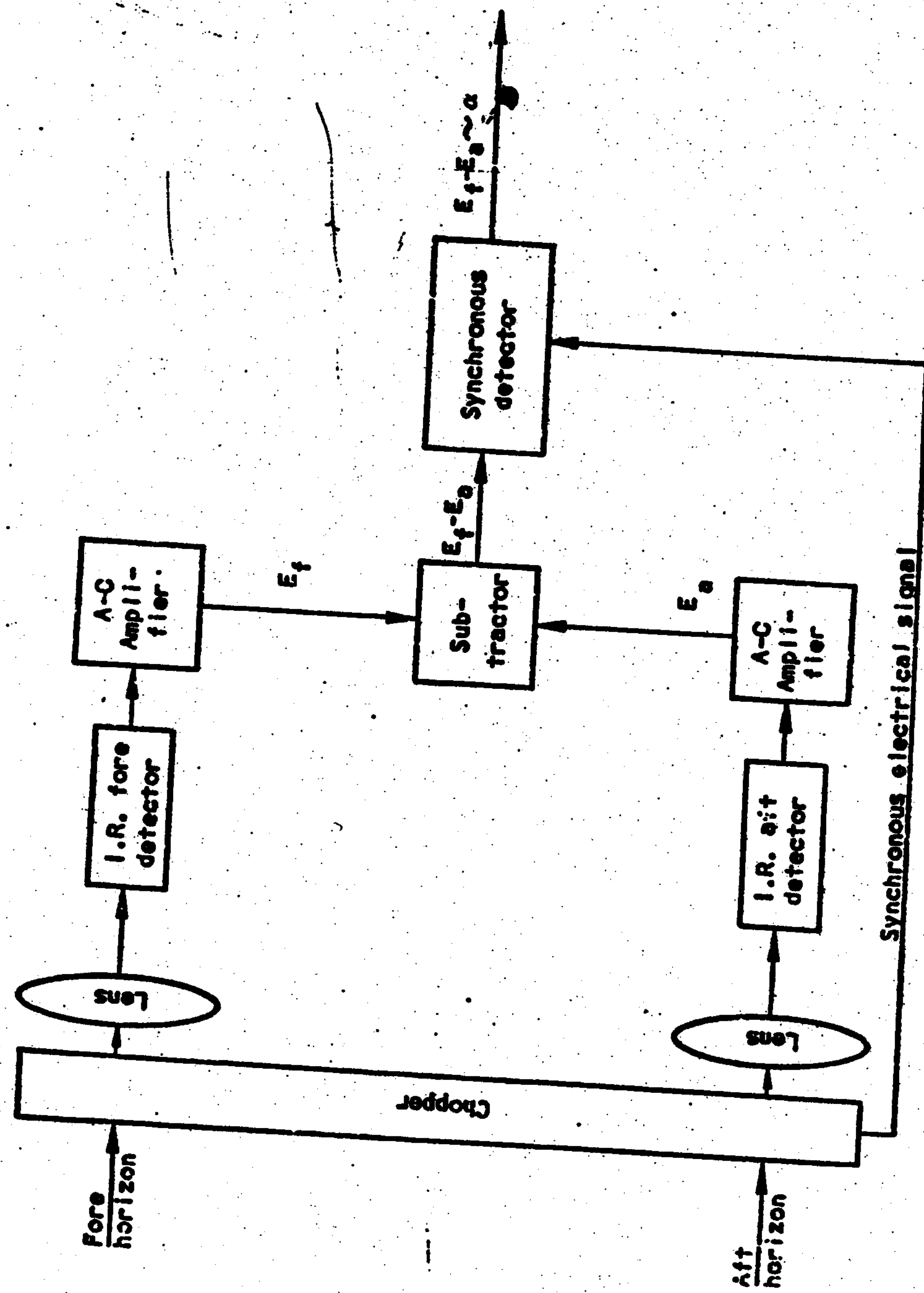


Fig. 5-2 Block Diagram of Horizon Sensor

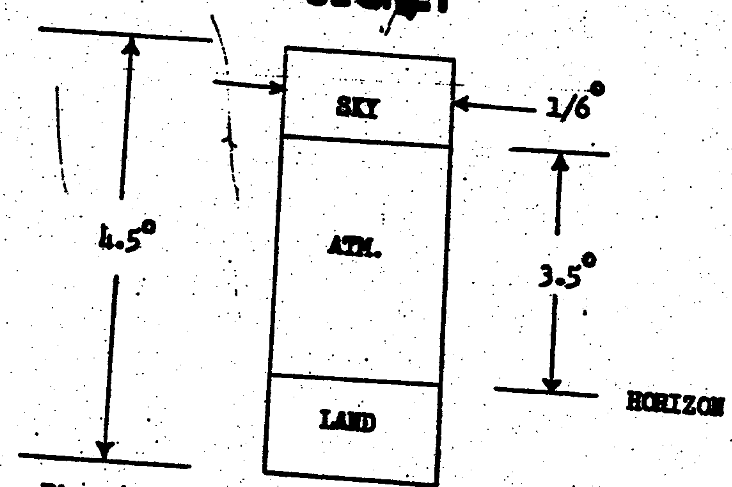


Fig. 5-3. Field of View

For a range of field of view perpendicular to the horizon, ±1/2 degree, the contribution to the signal from the atmosphere is constant. It is intended to have two viewers viewing the horizon at points 180 degrees apart. The viewers will be rigidly attached so that as one tilts up the other tilts down. The signal of interest is the difference between the signals from the two viewers. Such a signal has an amplitude proportional to the tilt and an algebraic sign indicating the direction of tilt.

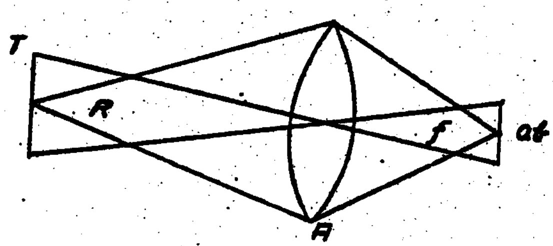
Since the missile is stabilized for yaw and, because of coupling, also for roll, it appears that only one set of viewers is needed to supply stabilizing information for pitch. Should further study indicate the necessity for extra stabilizing information for roll, another set of viewers can be incorporated to supply this data.

From what little work that has been done on radiation from the horizon (Refs. 9 - 11), it appears that the spectral region to use for the detection of the horizon should be in the infrared. If the near infrared region (1-3 microns) is used, then the various absorption bands of CO_2, H_2O, etc. may not be quite as serious as for a choice of medium infrared (3-8 microns). Also, the necessity for cooling the detector is avoided by using the near infrared region.

From the work of Bieber and Clark (Ref. 11) it appears that the land temperature may be taken as 250 degrees K. The sky as seen by a missile at an altitude of 300 n. miles should be the temperature of interstellar space about 4 degrees K and, for the purposes of this report, may be taken as 0 degrees K. Estimates made of the temperature of the missile where the viewers most likely will be placed indicate a temperature of 250 degrees K. Accordingly, that portion of the detector which views the land will experience no net radiation change; the portion viewing the atmosphere above the horizon will suffer radiation loss which increases towards that portion viewing the sky losing the most radiation. In the range about equilibrium, as has been noted, the radiation loss to the atmosphere will be constant, and the difference between the two viewer signals will cancel this contribution. However, the change in radiation which results from changing the amount of sky being viewed in one viewer will be multiplied by two (because one viewer increases by the same amount as the other viewer decreases) to get the result of the signal from the pair. Planck's radiation formula gives $W = \sigma(T_1^4 - T_2^4)$, where T_1 = 250 degrees K, and T_2 = 0 degrees K, as

W = 0.15 watts per square inch as the radiation power leaving the detector and going to the sky. It is assumed that the emissivities of the detector and earth are each unity.

If the area being viewed is T, the flux P intercepted by the lens area A, is (see Fig. 5-4).



$$P = \frac{TW}{\pi}\frac{A}{R^2}$$

Now

$$\frac{T}{R^2} = \frac{ab}{f^2}$$

Fig. 5-4. Scanner Lens

where a = the height of the detector, $b(\theta)$ = the width of the detector, and f is the focal length of the lens. Therefore, the flux which falls on the detector is

$$P = \frac{W}{\pi} A \frac{ab}{f^2}$$

If D is the diameter of the lens, then

$$P = \frac{WD^2 ab}{4f^2}$$

The entire detector does not receive or emit radiation, and as has been discussed for the difference signal, a height Δa is of consequence. If α is the angle of tilt, then

$$\frac{\Delta a}{f} = \alpha$$

The signal flux, however, is determined by 2 α and so

$$P_S = \frac{WD^2 b\alpha}{2f}$$

with b/f = 2.91 x 10^{-3} radians according to Fig. 5-1. Substituting also W = 0.15 watts per square inch

$$P_S = 218 \times 10^{-4} D^2 \alpha$$

Suppose now that the detector chosen is PbS. The responsivity of PbS is spectrally dependent, and the spectral distribution of a black body varies with the temperature. Therefore, information on the "Noise equivalent input power, P_N" (Ref. 9) of PbS for a 500 degree black body is decidedly erroneous when applied to a 250 degree black body. What is more, in the case under discussion, radiation is not falling on the detector but rather radiation is leaving. Nevertheless, for want of more applicable data, the information about PbS given by R. Clark Jones (Ref. 9) shall be used. R. Clark Jones defines the "noise-equivalent power in reference condition C" as

$$\mathcal{S} = \frac{P_N}{\left(ab \log_e \frac{f_2}{f_1}\right)^{1/2}}$$

Experimentally, it has been found that for PbS,

$$\mathcal{S}\tau_\rho = \text{constant},$$

τ_ρ = the time constant. The median value of the constant is 15 x 10^{-12} watt sec. cm^{-1} or 3.81 x 10^{-11} watt sec. in^{-1}. If a detector is chosen with τ_ρ = 10^{-3} sec., then $\mathcal{S}$ = 3.81 x 10^{-8} watt in^{-1}.

For a single detector

$$P_N = 5\left(ab \log_e \frac{f_2}{f_1}\right)^{\frac{1}{2}}$$

The ratio of the upper frequency to the lower frequency of the bandwidth associated with this device will be arbitrarily made to be e. Therefore

$$P_N = 3.81 \times 10^{-8} (ab)^{\frac{1}{2}} \text{ watts}$$

From Fig. 5-1

$$\frac{a}{f} = \frac{4.5\pi}{180} = 7.85 \times 10^{-2} \quad \text{or} \quad a = 7.85 \times 10^{-2} f$$

and

$$\frac{b}{f} = \frac{1/6\,\pi}{180} = 2.91 \times 10^{-3} \qquad b = 2.91 \times 10^{-3} f$$

Therefore

$$P_N = 5.76 \times 10^{-10} f \text{ watts}$$

Each detector should have a signal which is at least $5P_N$ to ensure that what is obtained is definitely not noise. Here W is only acting over a portion of the detector say $\frac{1}{4.5}(1.5 \times 10^{-1}) = W = 1.67 \times 10^{-2}$ watts per square inch

$$5P_N = \frac{W\ D^2 ab}{4f^2}$$

and, therefore, the minimum size for D is given by

$$D^2 = 3.03 \times 10^{-3}\ f$$

Thus, if $f = 4$ inches, $D > 0.11$ inch.

The difference signal must be larger than the $\sqrt{2} \times 5\rho_N$ so that

$$5\sqrt{2}\,\rho_N = \frac{W D^2 t \alpha}{2f}$$

and accordingly

$$\alpha = 1.87 \times 10^{-5} \frac{f}{D^2}$$

If $D = \frac{1}{2}$ inch, and $f = 4$ inches, then the minimum angle of tilt which may be detected is

$$\alpha = 3 \times 10^{-4} \text{ radian} = 1 \text{ minute of arc.}$$

5.3 Errors

Three distinct types of errors will arise in the aforementioned device. The differences arise due to the source of the error. Specifically, they are terrestrial errors, construction errors, and orbital errors.

Fig. 5-5 shows the geometry of the problem. It is assumed that the earth is spherical and that the lens system is a simple one. In addition, it is assumed that all pitching motion takes place about the intersection of the center line of the two lens systems. The errors introduced by the non-spherical earth can be studied by examining the errors introduced by the angle θ defined below. The error introduced by assuming a simple lens system and that all pitching motion takes place about the intersection of the center line of the two lens systems is of the order of f/R, where f is the focal length and R is the slant range. Since f is the order of inches and R is approximately 1500 miles, this error is truly negligible.

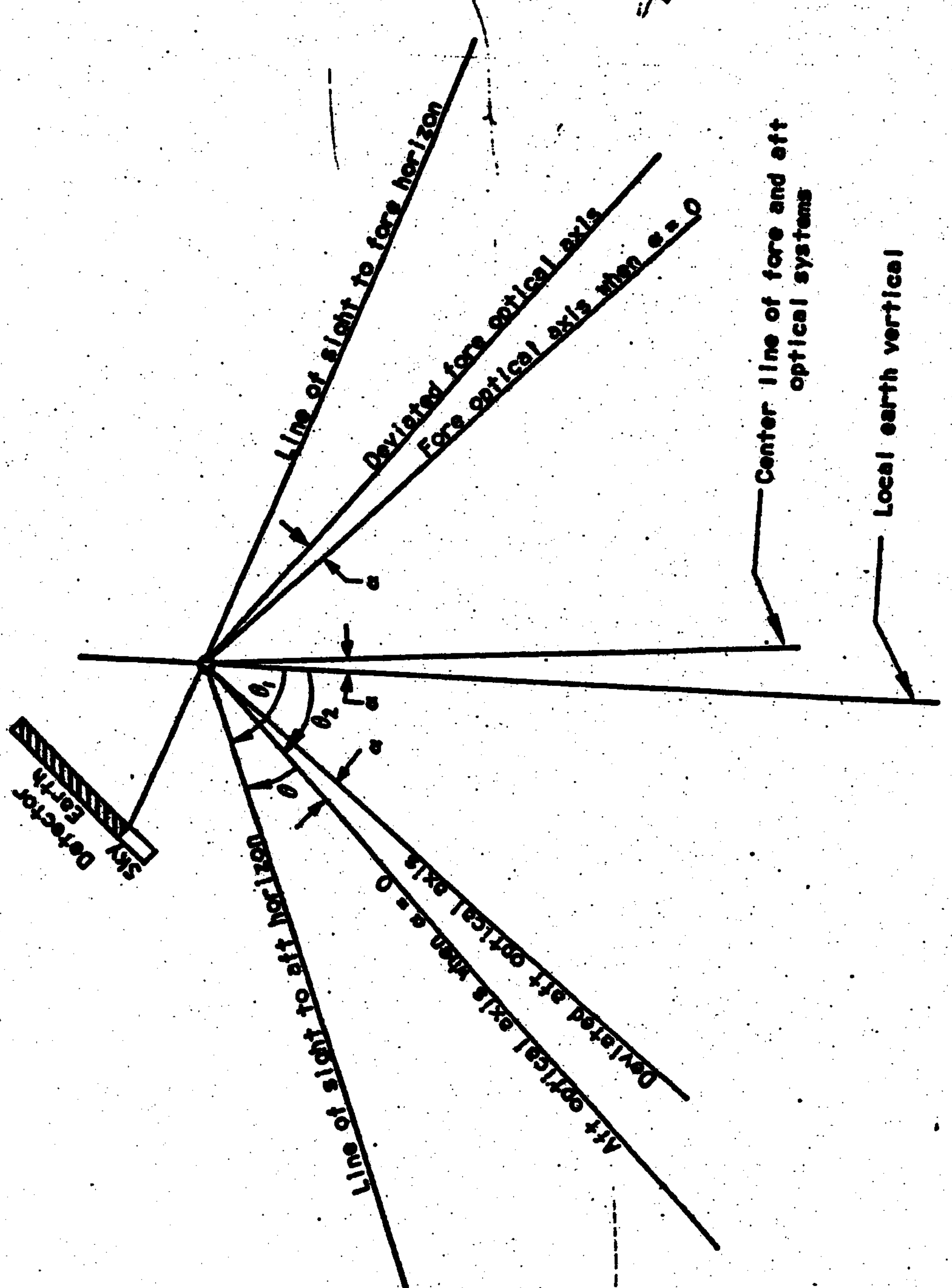


Fig. 5-5 System of Axes Associated with the Horizon Sensor

In the analysis that follows and in Fig. 5-5 the following symbols are used:

θ_1 = Angle between local vertical and tangent (line of sight) to earth

θ_2 = Angle between the center line of fore and aft optical systems and the center line of either the fore or the aft optical system

$$\theta = \theta_1 - \theta_2$$

a_f, a_a = Length of fore and aft detector

b_f, b_a = Width of fore and aft detector

f_f, f_a = Focal length of fore and aft detector

$$\phi_a = \tan^{-1} \frac{a_a}{2f_a}$$

$$\phi_f = \tan^{-1} \frac{a_f}{2f_f}$$

E_f, E_a = Electrical signal from fore and aft detector

α = Angle vertical and center line of fore and aft optical systems.

Assuming that the electrical signal obtained from the detector is proportional to the area not masked by the earth, we have

$$E_a - A_a b_a f_a \left[\tan \phi_a - \tan(\theta + \alpha)\right]$$

and

$$E_f - A_f b_f f_f \left[\tan \phi_f - \tan(\theta - \alpha)\right]$$

where A_a and A_f are the proportionality factors. If both fore and aft systems are exactly alike, then

$$E_f - E_a = K\left[\tan(\theta+\alpha) - \tan(\theta-\alpha)\right]$$

where $K = A_a\, t_a f_a = A_f\, t_f f_f$. If this difference signal is used in proportional control, then the final steady state conditions achieved will be $E_f - E_a = 0$, which means $\alpha_{ss} = 0$.

In the event that both fore and aft systems are not exactly alike, then

$$E_f - E_a = \frac{1}{2}\left(A_f\, t_f\, a_f - A_a\, t_a\, a_a\right) + A_a\, t_a f_a \tan(\theta+\alpha) - A_f\, t_f f_f \tan(\theta-\alpha).$$

Replacing the tangent by the angle and setting the difference equal to zero, we have

$$\alpha_{ss} = \frac{A_f\, t_f f_f - A_a\, t_a f_a}{A_f\, t_f f_f + A_a\, t_a f_a}\,\theta + \frac{1}{2}\,\frac{A_a\, t_a\, a_a - A_f\, t_f\, a_f}{A_f\, t_f f_f + A_a\, t_a f_a}$$

The term containing θ represents essentially a second-order error term because it results from two errors and is zero if either error vanishes.

The angle θ is attributable to the errors in construction and the non-spherical earth. A good estimate of its magnitude is one-half degree. If the terms $A_f\, t_f f_f$ and $A_a\, t_a f_a$ are within 10 per cent of each other, the first term will contribute an error of $0.05\,\theta$, or approximately 0.[illegible] degree. Since this is less than a half of a milliradian, [illegible] is a second-order effect.

The second term represents the error introduced solely by dissimilarities in the two lens systems, detectors, and amplifiers. In order to estimate its magnitude, it can be written in the following form:

$$\frac{1}{2}\,\frac{A_a t_a a_a - A_f t_f a_f}{A_f t_f f_f + A_a t_a f_a} = \frac{A_a t_a f_a - A_f t_f f_f}{A_a t_a f_a + A_f t_f f_f}\,\phi$$

where $\phi = \phi_a$ or ϕ_f and second-order effects have been neglected. With the assumption of 10 per cent matching in the optical systems and ϕ = 0.04 radian, the error due to this term is 0.002 radian or approximately 0.12 degree. A reduction of this error can be achieved by finer matching of the lens systems.

If the altitude, h , of the vehicle varies as it progresses through its orbit, the angle θ_1 will vary. A change in θ_1 is directly reflected into a change in θ . To estimate this effect on α_{ss} , $d\theta/dh$ can be calculated, and the previous formula used.

$$\sin\theta_1 = \frac{r}{r+h}$$

$$\frac{d\theta}{dh} = \frac{d\theta_1}{dh} = -\frac{r}{r+h}\,\frac{1}{\sqrt{2rh+h^2}}$$

Assuming the radius of the earth r = 4000 miles and the altitude h = 300 miles, we have

$$d\theta/dh = 0.6 \text{ milliradian per mile}$$

This also represents a second-order effect because the error in α_{ss} for 10 per cent matched optical systems would be only 0.03 milliradian per mile.

The errors introduced by mountains and clouds are more direct. The vertical resulting will differ from the geocentric vertical by half the increase in the angle θ_1. Assuming a false horizon appearing at 30,000 feet (see Fig. 5-6), the error in α_{sy} will be of the order of 0.12 degree. This

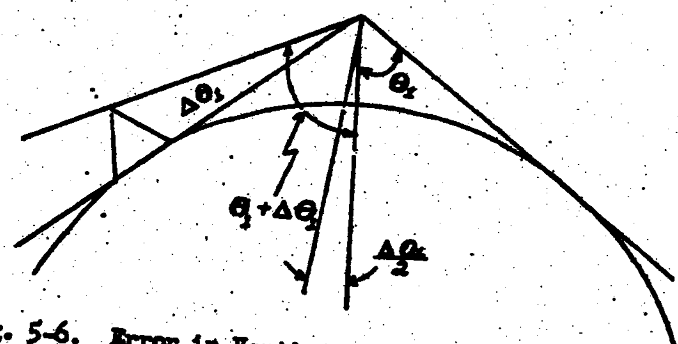


Fig. 5-6. Error in Vertical Introduced by Mountains and Clouds

represents a large error that is not easily eliminated. However, its effect will be reduced by the averaging process that takes place in the synchronous detector. The averaging process will weigh the terrestrial defects according to the time they are in the field of view.

6. IMAGE MOTION COMPENSATION

The degree of control of the vehicle attitude is dictated in part by the photographic resolution desired. The angular rates which are permissible about the respective missile's axes to reduce image blur to 30 feet are presented in Table 6-1. An analysis of attitude control shows the maximum angular velocity obtained per inch-ounce of torque to be about 0.028×10^{-4} radian per second in yaw. Using these values and assuming an exposure time of 0.1 seconds for the proposed attitude control system, an input torque of approximately 70 inch-ounces in pitch and of approximately 100 inch-ounces in yaw can be tolerated and still hold to a 30-foot image motion during exposure time. These calculations indicate that a film speed of 0.08 inch per second and an exposure time of 0.1 second is feasible from the standpoint of attitude control.

More realistic values of angular rates are obtainable with the presence of a gyro "dead spot" in the system. The maximum angular rate becomes $\pm 0.21 \times 10^{-4}$ radian per second and is still less than the tolerated angular rate for the 30-foot image motion. This reduces the allowable input torque by an order of magnitude. From these results image motion compensation is not necessary unless an image motion of less than 30 feet is required.

Table 6-1

PERMISSIBLE ANGULAR RATES

Image Motion During Exposure 30 Feet

Exposure Time (sec)	Pitch (rad/sec)	Roll (rad/sec)	Yaw (for 2-inch wide film) (rad/sec)
01.	1.6×10^{-4}	1.6×10^{-4}	1.9×10^{-3}
0.01	1.6×10^{-3}	1.6×10^{-3}	1.9×10^{-2}
0.001	1.6×10^{-2}	1.6×10^{-2}	1.9×10^{-1}

7. POWER REQUIREMENTS FOR OSV GUIDANCE AND CONTROL

For the purposes of planning the development of an auxiliary power unit (APU), estimates have been made of the power required for guidance and control of the OSV. Two distinct estimates were necessary. The first covers the OSV autopilot and guidance system, while the second estimate is for the attitude control system.

7.1 OSV Guidance and Control

Power is used for the controls in two ways. It is dissipated in the controls system instrumentation and feedback loop and is also used in the application of forces, torques, etc. to the vehicle. A typical autopilot loop is shown in Fig. 7-1.

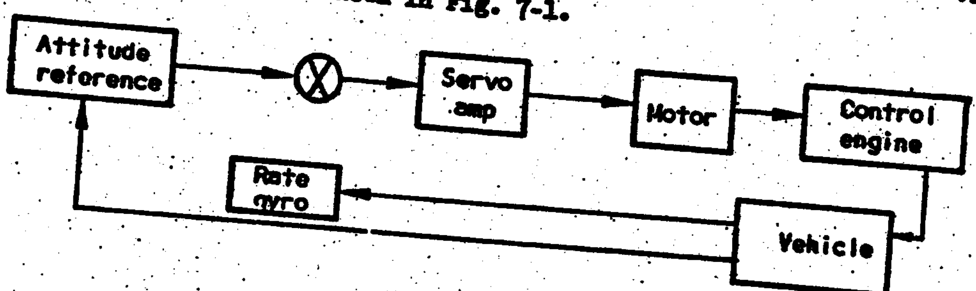


Fig. 7-1 Typical Autopilot Loop

The power consumption estimated for the loop, exclusive of the motor and primary torques, is shown in Table 7-1.

Table 7-2

1. Integrating Rate Gyro (HIG-6)	
Operating Power	
Heater Power	15 watts
2. Rate Gyros	20 watts
3. Amplifier	10 watts
Power dissipated per channel	15 watts
	60
Total Power dissipated (3 channels)	180

In addition to the power consumption tabulated above, the controls autopilot is required to provide the means to remove energy from the vehicle or change its course slightly by directed applications of power from the control engines. Since the motor power for gimballing the control engines is derived from an APU which uses the same fuel as the control engines, the power required for this application will be estimated as a whole rather than separately.

The energy of the vehicle in rotation is given by

$$E = \frac{1}{2} I (\dot{\phi})^2$$

where, if the vehicle is in a simple harmonic oscillation, $\dot{\phi} = \phi_0 (2\pi f) \cos (2\pi f) t$. The power required to counter this oscillation is given by

$$P = \frac{dE}{dt} = I \dot{\phi} \ddot{\phi} = I \phi_0 (2\pi f) \phi_0 (2\pi f)^2 \sin 2 (2\pi f) t$$

$$P_{max} = I \phi_0^2 (2\pi f)^3$$

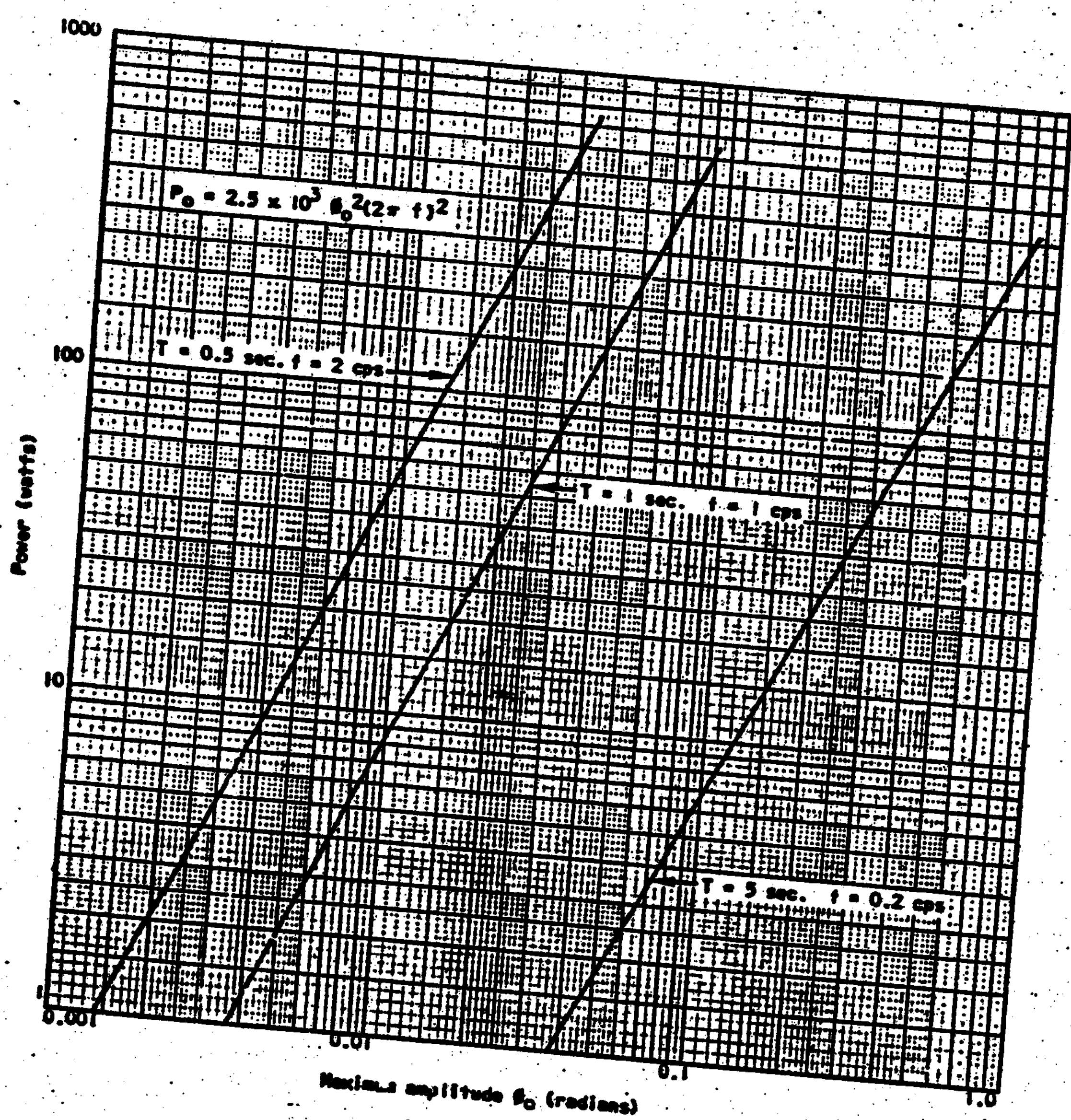


Fig. 7-2 Power Derived from Control Engines to Stabilize Vehicle

Using the moment of inertia listed in Sec. 4, Table I, as a typical example

$I = 11.6 \times 10^6$ lb-in.^{2} = 2.5×10^3 slug-ft.^{2}

$$P_{max} = 2.5 \times 10^3 \phi_o^2 (2\pi f)^3$$

The power required to absorb these oscillations at several frequencies and amplitudes are shown graphically in Fig. 7-2.

7.2 Altitude Control

The attitude control power is used in two ways; power is dissipated in sensing instrumentation and friction of the torque wheels and also in the feedback loop for the application of control torques to the vehicle. A typical attitude control system loop is shown in Fig. 7-3.

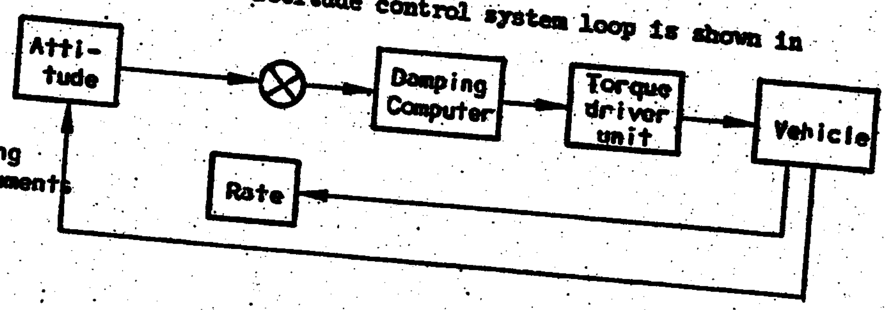


Fig. 7-3 Typical Attitude Control System Loop

The sensing instruments are the same as those listed in Table 7-1, but the power required depends on the exact configuration. If two rate gyros and two amplifiers are used, this will be about 40 or 50 watts.

Additional power is required to operate the torque drive units. Maximum power requirements occur at the separation of the OSV from the Atlas "C" booster. It is assumed that the separation torque energy develops an angular velocity on the OSV of 2 radians per second.

This rate must be reduced to zero in about 30 seconds. Such damping requires a 10 watt motor and is more than enough to control any other expected disturbance.

For the realistic damping and frequency conditions during orbiting the power required to damp the complete vehicle for a maximum pitch rate of 0.2 radian per second is of the order of 1/2 watt. However, if one assumes that the power efficiency is of the order of 10 percent, then it would be safe to estimate that the power required would be of the order of 5 watts.

The power requirements of the pitch wheel to overcome the applied external torques (rms value) are quite small when compared to the power requirements to damp the complete vehicle because of the very small amplitudes and rates. This power requirement is about 8×10^{-6} watt (the rate signal to the damping computer is small). A more refined analysis is required but will not increase the power requirements materially.

Investigation of the various power requirements associated with the components of the attitude control system shows that more power may be required to keep the elements at a constant temperature than to actually operate the equipment. However, part of this problem can be alleviated by extracting heat from other sources, e.g., the exhaust gases of an APU, temperatures of which have been calculated to about 1000 degrees Rankine.

D-APPENDIX REFERENCES

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