FOREWORD

This report was prepared by the Research Laboratories Division of The Bendix Corporation as part of the research and study program performed under Air Force Contract No. AF 33(657)-10998.

The project engineer for the Air Force is Mr. James Morris.

The work described in this report was performed by the Attitude Controls Group of the Dynamic Controls Department. Mr. J. G. Rivard is the group supervisor and Mr. L. B. Taplin is the department head. Supervision and technical direction of the program was the responsibility of Mr. R. H. Larson.

Several individuals contributed to this research and study program. Mr. J. H. Tarter analyzed the re-entry trajectory and defined the attitude maneuvers required. Mr. J. T. Kasselmann analyzed the control requirements of the re-entry maneuver and selected and designed an attitude control loop to meet these requirements. Mr. L. L. Evans investigated the feasibility of using momentum exchange devices for generating control moments. Mr. J. C. Walberer was responsible for the design of the attitude control engines and associated components. Messrs. R. W. Presley and E. Rappaport conducted the analogue computer simulations of control system dynamics. Mr. R. H. Larson analyzed the orbital transfer and rendezvous maneuvers and performed the work on secondary injection thrust vector control.

ABSTRACT

This study is concerned with the application of high moment-producing techniques for the attitude control of manned space vehicles which perform orbital transfer, rendezvous, and lifting re-entry maneuvers. The specific accomplishments were:

(a) Establishment of control system, subsystem and component requirements for critical vehicle flight phases.

(b) Analysis of promising techniques for performing the flight control functions required during the various maneuvers.

A hypothetical vehicle was selected and its space mission was analyzed to determine the attitude control requirements. The rendezvous and re-entry maneuvers were found to be especially demanding on the attitude control system.

The production of control moments by means of reaction wheels, gyros, mass expulsion, secondary injection thrust vector control, and aerodynamic surfaces was investigated. Reaction engines, secondary injection thrust vector control, and aerodynamic flaps were selected for use in the required attitude control systems, and specific designs were made. The control loops were analyzed with regard to response, accuracy, and fuel consumption. A substantial amount of this analysis was accomplished by means of analog computer simulation.

This technical documentary report has been reviewed and is approved.

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SECTION 1

INTRODUCTION AND SUMMARY

1.1 OBJECTIVE

The objective of this study is to provide methods and techniques for achieving control and stabilization of large manned vehicles during orbital rendezvous and re-entry maneuvers. The following general tasks have been accomplished:

(a) Establishment of the control system, subsystem, and component requirements for critical vehicle flight phases.

(b) Detailed analysis of certain promising techniques for performing the flight control functions required during the various phases and maneuvers.

1.2 PROBLEM DEFINITION

1.2.1 Hypothetical Vehicle and Requirements

The vehicle and associated requirements assumed for this study are defined below.

(a) Vehicle initial weight - 200,000 lbs

(b) Vehicle initial orbit altitude - 150 n.m.

(c) Maximum orbit altitude - 1000 n.m.

(d) Vehicle length - approximately 200 ft.

(e) Capability for rendezvous

(f) Vehicle is manned

(g) Aerodynamic control during re-entry

1.2.2 Vehicle Maneuvers

The manned space vehicle performs several maneuvers during its mission. These are described briefly and in sequence as follows:

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(a) Boost - The main rocket engine or a booster engine launches the vehicle from the earth and puts it into a parking orbit considerably below that of the target with which it is to rendezvous, and possibly in a different plane. Ideally, both the vehicle and target are in circular orbits at this point.

(b) Change of Plane - At the intersection of the vehicle and target orbital planes, the velocity vector of the vehicle is rotated so that the orbits of the vehicle and target become coplanar. The altitude of the vehicle will not be changed. This maneuver is preceded by rotating the vehicle about its yaw axis to properly position the thrust vector.

(c) Orbital Transfer (ascent) - When the vehicle and target are at the proper relative position, a thrusting period is introduced to put the vehicle into an elliptical orbit, with the apogee at the same altitude and position as the target. At the apogee, an additional thrust pulse may be used to increase the vehicle velocity nearer to that of the target.

(d) Rendezvous - The rendezvous phase begins as soon as the vehicle reaches the target altitude. The vehicle is initially ahead of the target and with a lower velocity, therefore the distance between the bodies decreases with time. Rendezvous is achieved by applying a series of thrust pulses to the vehicle to bring the vehicle velocity up to that of the target as the distance between them decreases. During this phase, the attitude control system maintains proper thrust vector alignment with the line-of-sight between the two bodies.

(e) Docking - Docking is the terminal phase of rendezvous and is one of the most critical maneuvers because non-destructive physical contact must be achieved. The docking maneuver begins when the relative velocity and range are very small. It involves orienting the vehicle and making vernier thrust corrections. At physical contact a latching scheme absorbs the impact energy and locks the bodies together.

(f) Separation - Separation is somewhat the reverse of docking because vernier control is used to separate the bodies until larger engines can be used.

(g) Change of Plane - If it is necessary to put the vehicle into a different orbital plane in preparation for re-entry, it would be accomplished before reducing orbital altitude. It is most efficient to change the orbital plane at a high altitude because the velocity is lower and requires less correction.

(h) Orbital Transfer (descent) - The descent orbital transfer is the reverse of that used in ascent. The velocity is decreased to generate an elliptical orbit, and at the perigee a velocity decrement circularizes the orbit.

(i) Re-entry - The re-entry phase begins by decreasing the velocity and allowing the vehicle to enter the atmosphere. Aerodynamic surfaces control the velocity-altitude relationships within allowable limits for safe re-entry.

Insofar as performance of the attitude control system is concerned, the most critical maneuvers are rendezvous and re-entry. This report will be concerned with the initial portions of rendezvous and re-entry where fully automatic control of the vehicle is imperative.

1.3 SUMMARY

The vehicle mission was analyzed with regard to the attitude control requirements imposed by orbital transfer, rendezvous, and re-entry. For orbital transfer, the attitude errors during thrusting periods cannot exceed one degree, while during rendezvous, the most critical requirement restricts the attitude error to a maximum value of one-half degree. The complexity of the re-entry maneuver does not lend itself to definition of attitude control requirements in terms of only accuracy. In essence, the attitude control system must rapidly and accurately respond to the attitude commands of the guidance system while under varying load conditions imposed by changes in velocity and atmospheric density.

Various methods for producing large control moments on the hypothetical vehicle have been studied. While control moment gyros are preferable to reaction wheels from the standpoint of torque and power requirements, no momentum exchange system of any kind was proposed for the vehicle because adequate attitude control can be maintained using reaction engines. Secondary injection thrust vector control is a convenient means for compensating for engine misalignment and maintaining vehicle attitude during main engine firing. Moveable aerodynamic surfaces are mandatory for controlling the attitude of a lifting re-entry vehicle.

The optimum attitude control system for orbital transfer and rendezvous uses mass expulsion moment producers incorporated in a control loop which functions in an on-off mode with rate feedback. When the control moments are small, reaction engines are used, but when the main engine fires, secondary injection thrust vector control must be employed to compensate for engine misalignment. The valve-combustor system proposed for the reaction engines uses a two-stage solenoid-operated poppet valve which feeds the propellants into a vortex doublet combustor. An all-fluid vortex valve system is defined for controlling the secondary injection flow rate.

The attitude control of a lifting re-entry vehicle requires the use of both mass expulsion and aerodynamic surfaces. At high altitudes where the dynamic pressure is low, aerodynamic control is insufficient and requires augmentation by control moments created by mass expulsion. At a later time, however, the dynamic pressure reaches a sufficient level to allow complete aerodynamic control of attitude. The actuator selected for driving the aerodynamic flaps is an expansion vane motor controlled by a spool-type servo valve. A piston-cylinder lacks the performance and fuel economy of an expansion vane motor. Reaction jets are completely out of the question for flap control because of the enormous quantities of fuel required.

SECTION 2

MISSION ANALYSIS

2.1 ASSUMPTIONS AND CONSTANTS

The following assumptions are made with regard to orbital transfer:

(a) The target is in a circular orbit

(b) The manned vehicle is initially in a circular parking orbit below that of the target

(c) The orbital planes of the target and vehicle do not coincide

(d) The thrust magnitude of the main engine is high enough so that the use of impulsive velocity increments is a good approximation.

For rendezvous, it is assumed that:

(a) The vehicle and target are at the same altitude

(b) The vehicle is initially ahead of the target and traveling at a lower velocity

(c) The relative motion is coplanar

(d) All maneuvers are performed by the vehicle

(e) All guidance engines have constant thrust.

The re-entry maneuver was analyzed on the basis that:

(a) The earth is a uniformly homogeneous, nonrotating sphere with the mass concentrated at the center

(b) All trajectories lie in a plane containing the earth's center

(c) Drag due to the atmosphere is negligible above 400,000 ft

(d) Vehicle altitude is negligible compared to the radius of earth

(e) Atmospheric density below 400,000 feet is given by:

$$\rho = \rho_B e^{-\beta h}$$

where

$$\rho_B = \rho_{SL} = 0.00238 \text{ slugs/ft}^3$$

and

$$\beta = (1/24100) \text{ ft}^{-1}$$

(f) Automatic re-entry control ceases at a vehicle velocity of 5000 fps and the pilot takes over to land the vehicle.

The over-all vehicle mission, involving orbital transfer, rendezvous, and re-entry, is predicted on the basis of the following parameters.

(a) Altitude of initial parking orbit

h_i = 150 n.m.

(b) Altitude of target orbit

h_t = 300 n.m.

h_t = 600 n.m.

h_t = 1000 n.m.

(c) Differential inclination of initial parking orbit and target orbit

$\psi = 10°$ max

(d) Altitude of final parking orbit at initiation of re-entry

h_f = 66 n.m.

(e) Average acceleration of gravity during re-entry

$$g = 31.4 \frac{\text{ft}}{\text{sec}^2}$$

(f) Velocity at initiation of re-entry

V_i = 26,000 fps

(g) Average distance from vehicle to earth's center during re-entry

$R = 21.1 \times 10^6$ ft

(h) Weight of vehicle in initial parking orbit

$$W_i = 200{,}000 \text{ lb}$$

(i) Thrust of main engine

$$T_{ri} = 400{,}000 \text{ lb}$$

(j) Specific impulse of all engines

$$I_{sp} = 400 \text{ sec}$$

(k) Earth's radius

$$R_e = 20.9 \times 10^6 \text{ ft}$$

(l) One nautical mile = 6076 ft

2.2 ORBITAL TRANSFER AND RENDEZVOUS

2.2.1 Description

Simple orbital transfer maneuvers are used to bring the vehicle to the proximity of the target. Initially, the orbital planes may have a differential inclination. To make the planes coincide, the velocity vector of the vehicle must be rotated through the angle of inclination when the vehicle is at one of the nodes. This is accomplished by applying a velocity increment with the proper magnitude and direction. The transfer maneuver to the target altitude is accomplished by increasing the vehicle velocity such that it follows a Hohmann transfer ellipse. When the vehicle reaches the apogee of this ellipse, the rendezvous maneuver is initiated.

The rendezvous maneuver requires the use of three control systems. A range rate control system acts to attenuate the closing rate as the range between the vehicle and target decreases, the goal being to prevent damage at impact. An angular rate control system establishes the vehicle on a collision course with the target by nulling the angular rate of the line-of-sight between the two bodies. An attitude control system properly orients the thrust vectors of the control engines with respect to the line-of-sight.

The geometry of the rendezvous problem is shown in Figure 1. The vehicle is ahead of the target but traveling at a lower velocity so that $\dot{r}$ is negative. $\dot{\gamma}$ is the angular rate of the line-of-sight between the two bodies. The longitudinal engine is a part of the range rate control system and acts along the line-of-sight to produce the acceleration a_r. A transverse-mounted engine acts normal to the line-of-sight through the vehicle center of gravity. It is the prime mover of the angular rate control system and produces a_n.

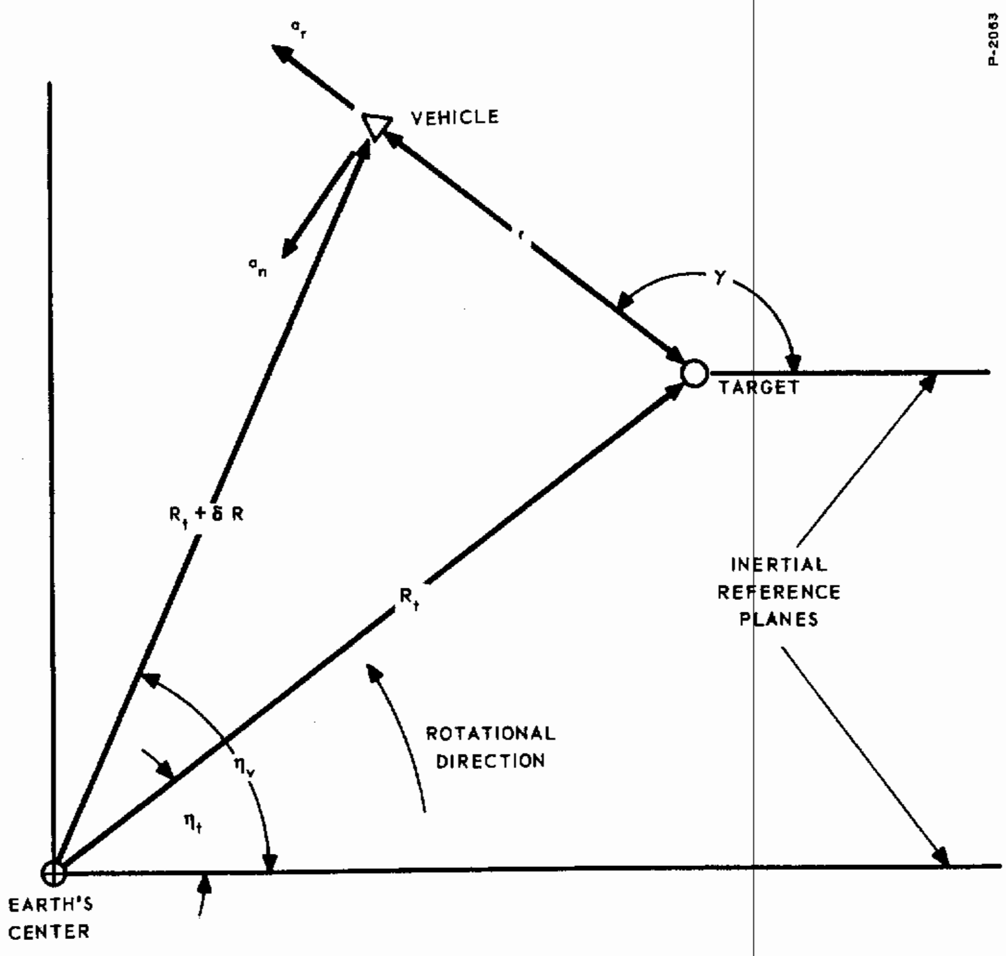

Figure 1 - Rendezvous Geometry

The rendezvous maneuver involves the use of multiple corrections for bringing the two bodies together. The angular rate and range rate control systems operate at somewhat regular intervals to correct for errors introduced by each other as well as errors which grow as functions of time. The sequence is such that the time between corrections allows adequate data filtering and smoothing by the guidance computer. The attitude control system can be either continuous or on-off in nature, so long as the vehicle is in proper orientation during the application of the normal and longitudinal corrections.

The rendezvous maneuver begins when the vehicle's radar picks up and begins to track the target. The range at this point is approximately 60 miles and the range rate is -1242 fps for the case where the target altitude is 1000 nautical miles. Immediately, the attitude control system puts the longitudinal axis on the line-of-sight, and the angular rate control system applies the necessary lateral pulse to reduce the angular rate of the line-of-sight to near zero. This logic continues until the range is reduced to a few miles. From this point on, the time-to-go parameter,

$$\tau = \frac{r}{-\dot{r}}$$

plays a major role in that the logic used with the range rate control system is to maintain τ within specified limits. When τ reaches the lower limit of 40 seconds, the longitudinal engine is turned on to increase the vehicle velocity a sufficient amount to bring τ to the upper limit of 60 seconds. This continues until the range is reduced to one hundred feet or less and the range rate is 1 to 2 fps. At this point the docking system takes over and no more velocity changes are made except to maintain a collision course. When the bodies are a few feet apart, a mechanical or magnetic system controls the impact and hook-up.

2.2.2 Maneuver Characteristics

Orbital transfer and rendezvous maneuvers were analyzed for three target altitudes, 300, 600, and 1000 nautical miles. The results for the 1000 nautical mile altitude are presented here. During the ascent orbital transfer maneuvers, a net velocity change of 5740 ft/sec is required, resulting in a total fuel consumption of 72,000 pounds. At the initiation of rendezvous, the vehicle weighs 128,000 pounds, is approximately 370,000 feet ahead of the target, and is traveling at a speed 1242 ft/sec slower than that of the target.

The rendezvous maneuver was simulated on a digital computer. The results obtained for the ideal case where the attitude error is zero are shown in Table 1. Definition of the symbols used in the table are as follows:

a_n = acceleration created by transverse engine, ft/sec^2

a_r = acceleration created by longitudinal engine, ft/sec^2

r_o = range between vehicle and target at the initiation of a correction, ft

r_1 = range at the termination of a correction, ft

t_o = total elapsed time at the initiation of a correction, sec

t_1 = total elapsed time at the termination of a correction, sec

Δt_c = time duration between corrections, sec

Δt_n = thrusting time duration of transverse (normal) engine, sec

Δt_r = thrusting time duration of longitudinal engine, sec

$\dot{\gamma}_o$ = angular rate of line-of-sight at the initiation of a correction, rad/sec

$\dot{\gamma}_1$ = angular rate of line-of-sight at the termination of a correction, rad/sec

τ_o = range (r_o) divided by minus the range rate ($\dot{r}_o$) at correction initiation, sec

τ_1 = $r_1/-\dot{r}_1$ at correction termination, sec

From Table 1, it is seen that the total maneuver takes less than 10 minutes and requires the use of three sizes each of transverse and longitudinal engines.

Table 1 - Rendezvous Maneuver at 1000 Nautical Mile Altitude with Zero Attitude Error

CYCLE	Δt_c sec.	t_o sec.	r_o ft.	τ_o sec.	$\dot{\gamma}_o$ rad/sec.	Δt_r sec.	a_r ft/sec^2	Δt_n sec.	a_n ft/sec^2	r_1 ft.	τ_1 sec.	$\dot{\gamma}_1$ rad/sec.	t_1 sec.
1	110.7	111	236,000	191.8	0.00100	0	100	12.69	-13	220,000	179.4	0.00037	123
2	70.3	194	134,000	109.6	0.00100	0	100	7.21	-13	125,000	102.4	0.00038	200
3	39.6	240	76,800	62.9	0.00100	0.39	100	4.14	-13	71,900	60.8	0.00038	245
4	20.8	265	47,300	40	0.00088	4.55	100	2.10	-13	42,900	59.0	0.00038	270
5	19.1	289	29,100	40	0.00082	2.80	100	1.17	-13	27,400	61.3	0.00035	292
6	21.3	313	17,900	40	0.00082	1.72	100	0.72	-13	17,300	62.7	0.00032	315
7	22.8	337	11,000	40	0.00079	8.14	13	2.69	-2	9,200	54.1	0.00046	346
8	14.1	360	5,000	40	0.00085	5.03	13	1.86	-2	6,100	58.0	0.00040	365
9	18.0	383	4,200	40	0.00084	3.11	13	1.13	-2	3,900	60.9	0.00036	386
10	20.9	407	2,600	40	0.00083	1.92	13	0.69	-2	2,500	62.1	0.00034	409
11	22.1	431	1,600	40	0.00082	1.19	13	0.41	-2	1,500	63.0	0.00033	432
12	23.1	455	990	40	0.00081	4.78	2	3.38	-0.15	900	58.7	0.00039	460
13	18.8	479	610	40	0.00084	2.94	2	2.18	-0.15	580	61.0	0.00036	482
14	21.0	503	377	40	0.00085	1.81	2	1.37	-0.15	363	62.5	0.00033	502
15	22.5	527	232	40	0.00080	1.12	2	0.78	-0.15	227	63.3	0.00031	528
16	23.3	551	143	40	0.00099	0.69	2	0.47	-0.15	141	64.1	0.00031	552
17	24.1	576	90	40									

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2.2.3 Attitude Control Requirements

The required accuracy of the attitude control system depends upon the particular flight maneuver being executed. When the vehicle is following a parking orbit, there may be no reason at all for maintaining a certain attitude orientation, particularly if a large number of rotations about the earth are required to properly phase the vehicle with the target. Even when it is necessary to direct antenna toward the earth for communications or ephemaris determination, the attitude accuracy tolerance would be no less than 30 degrees. The major problem during periods of coasting, then, is that of maintaining attitude stability for the sake of personnel comfort.

The maneuver which accomplishes the rotation of the vehicle orbital plane to coincide with the target orbit requires moderately accurate positioning of the engine thrust vector. It was determined that an angular error of one degree will create a lateral error of approximately one mile at the inception of rendezvous. This would be acceptable and does not appear to be difficult to achieve.

Angular errors at the initiation of the ascent maneuver affect the angular rate of the line-of-sight at the inception of rendezvous. It was found that an angular error of one degree will create line-of-sight angular rates of 0.0011 rad/sec at a 1000 nautical mile attitude and 0.0057 rad/sec at 300 nautical miles. An error of one degree is certainly the maximum permissible, since the resulting angular rates are sufficiently large to require the use of the main engine for making the first angular rate correction.

Attitude accuracy requirements during descent and deboost are not as critical as those associated with ascent, since the re-entry maneuver can tolerate moderate deviations of initial conditions. It is concluded that attitude accuracies of within one degree are adequate.

The accuracy requirements of the attitude control system during rendezvous were determined on the basis of a constant error persisting during a given correction. This study revealed that average pitch errors of +0.010 radian or -0.010 radian during each correction would permit a successful rendezvous. It was also found that yaw accuracies within ±0.010 radian and roll accuracies within ±0.020 radian would be acceptable.

2.3 RE-ENTRY

2.3.1 Description

A detailed study to arrive at an optimum re-entry shape is outside the scope of this project which is, after all, directed toward studying moment-producing techniques. It was thus decided to assume the use of a delta winged glider configuration, because considerable data on the hypersonic aerodynamic characteristics of such configurations have been accumulated. As a result, not only is much of the basic information required to evaluate various moment producing techniques readily available, but also confidence may be expressed that if the configuration selected is not the absolute optimum, it is at least quite practical and reasonable.

The basic equations of motion for a re-entry vehicle were modified according to the initial assumptions and used in conjunction with a control law to compute various trajectories. The control law was defined in terms of vehicle acceleration whereby the maximum acceleration along the flight path is not allowed to exceed 2 g's.

The geometry of the re-entry motion is described by Figure 2. The symbols are defined as:

D = drag force on vehicle, lb

L = lift force on vehicle, lb

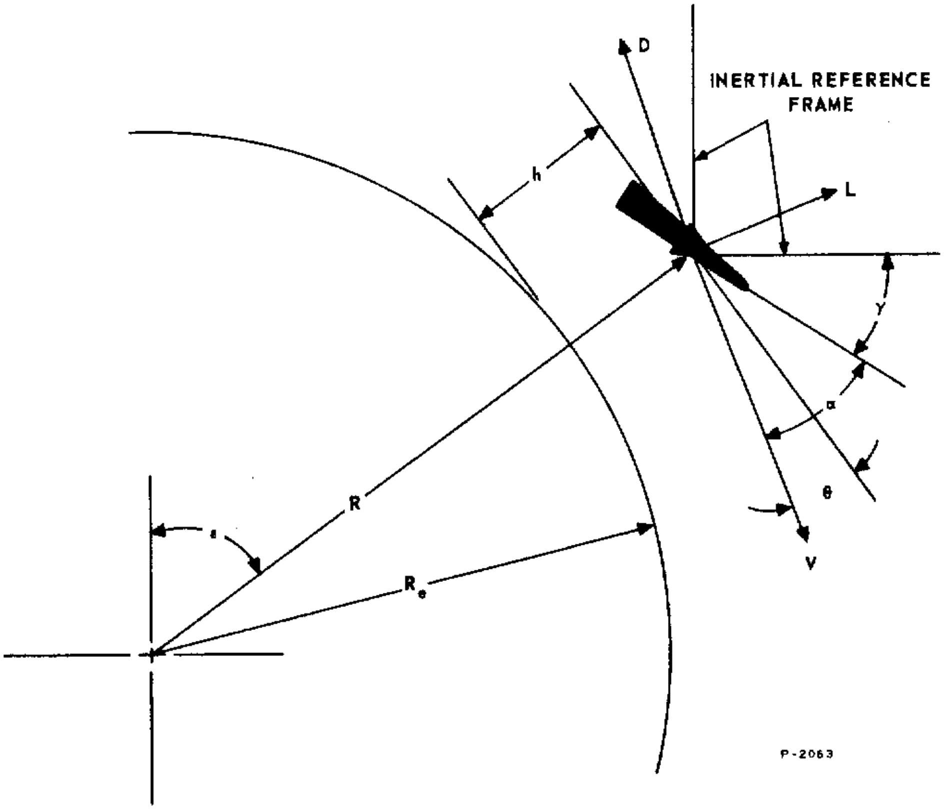

Figure 2 - Geometry of Re-Entry Motion in Pitch Plane

h = vehicle altitude above earth, ft

R = radius from earth's center to vehicle, ft

R_e = earth's radius, ft

V = velocity of vehicle, ft/sec

α = angle-of-attack, rad

γ = inertial reference angle, rad

θ = angle between horizontal and velocity vector, rad

At initial entry, although the velocity is high, the atmospheric density is very small. The drag force D cannot be built up to a large enough value to bring path deceleration immediately to its desired value, however, the attitude control system will pitch the vehicle up to the maximum allowable angle of attack in order to obtain the greatest possible drag. This will also give a minimum value of lift-to-drag ratio L/D. As the vehicle descends, the drag force will build up as atmospheric density increases, and the increased drag will gradually bring the path deceleration up to the desired value. When the limiting deceleration value is reached, the vehicle will begin to pitch down in order to maintain a drag which will give the desired value of path deceleration. With the aerodynamic characteristics assumed here, a decrease in drag causes an increase in L/D which makes the vehicle pull up and start to climb. As atmospheric density decreases, the vehicle will commence to pitch up to maintain the deceleration at its desired value. This will cause a decrease in L/D such that the vehicle will descend again, and the whole process will start over.

2.3.2 Trajectory Selection

Three factors of major importance during re-entry are acceleration, dynamic pressure, and aerodynamic heating. These must be maintained within proper limits if the vehicle and payload are to survive re-entry. Accelerations experienced by the vehicle and its contents are shown in Figure 3. From these curves it is seen that acceptably low accelerations are imposed when the initial path angle θ is 2.5 degrees. It was also determined that the heating rates and dynamic pressures experienced by the vehicle are not prohibitive for this trajectory.

2.3.3 Attitude Control Requirements

The requirements of the attitude control system during re-entry cannot be explicitly defined because they are a function of the vehicle velocity and atmospheric density as well as the guidance law. For example, when the vehicle is at a high altitude where the atmospheric density is low, greater errors in angle of attack can be tolerated than when the density and accompanying aerodynamic effects become large. In essence, during re-entry, attitude control requirements are not adequately defined in terms of only position error because of the complete integration of the attitude control system into the guidance system.

2.4 VEHICLE CHARACTERISTICS

The mission analysis was based upon a total vehicle weight of 200,000 pounds in an initial parking orbit at 150 nautical miles.

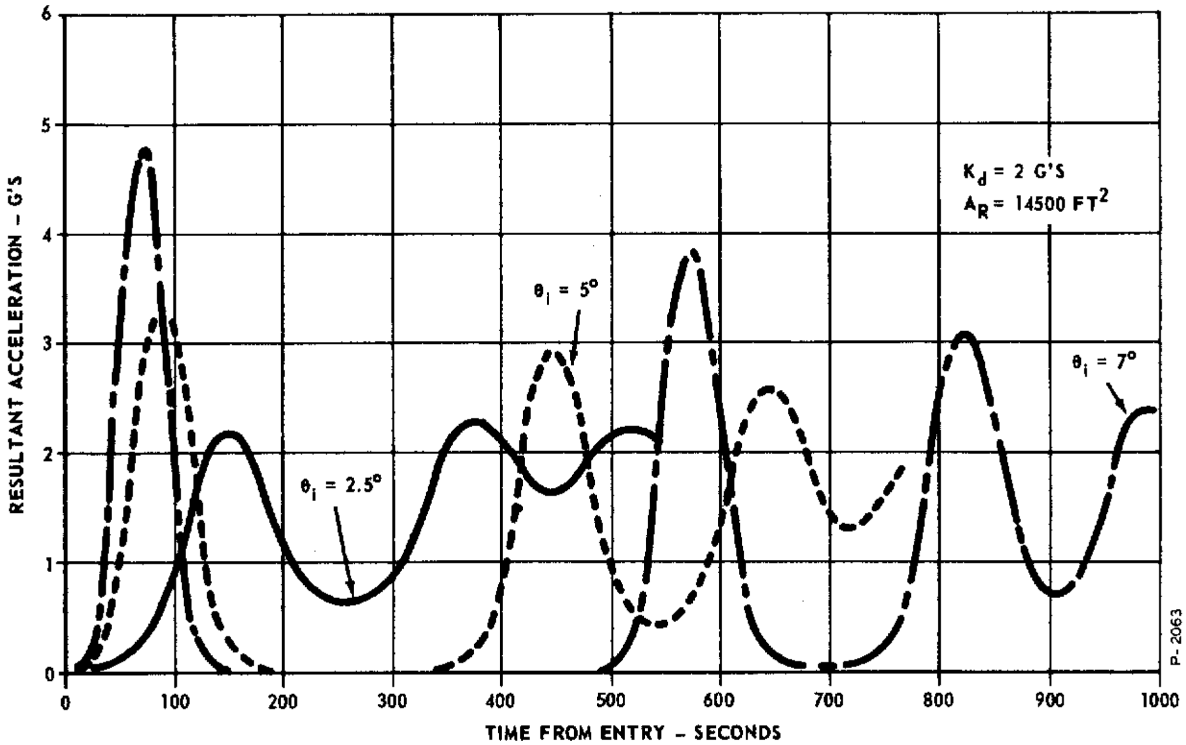

Figure 3 - Re-Entry Trajectory Characteristics; Resultant Acceleration Versus Time

Assumptions made concerning the space maneuvers required to reach the orbit of the target resulted in a vehicle weight of approximately 128,000 pounds during rendezvous. The weight during re-entry was estimated to be 100,000 pounds.

A study of the re-entry maneuver has defined the vehicle shape which in turn has allowed the principal moments of inertia to be estimated. Figure 4 shows a plan view of the vehicle. The ranges of the principal moments of inertia as determined by full and empty fuel tanks are:

$$J_p = 4.94 \times 10^6 \text{ lb-sec}^2\text{-ft to } 4.13 \times 10^6 \text{ lb-sec}^2\text{-ft}$$

$$J_y = 5.78 \times 10^6 \text{ lb-sec}^2\text{-ft to } 5.00 \times 10^6 \text{ lb-sec}^2\text{-ft}$$

$$J_r = 1.06 \times 10^6 \text{ lb-sec}^2\text{-ft to } 0.95 \times 10^6 \text{ lb-sec}^2\text{-ft}$$

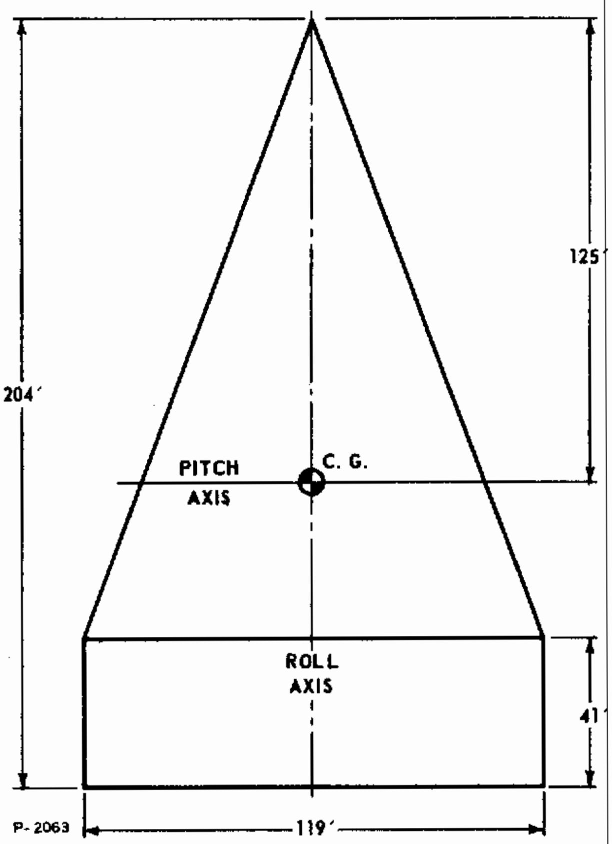

Figure 4 - Vehicle Plan View

where the subscripts p, y, and r refer to pitch, yaw, and roll axes respectively. The center of gravity does not move along either the pitch or roll axis, but does move a total of 1.10 feet along the yaw axis as the fuel is consumed.

The thrust levels required for vehicle guidance at various times during the flight are:

(a) 400,000 lbs

(b) 50,000 lbs

(c) 8,000 lbs

(d) 600 lbs

The 400,000 pound thrust is used for plane change, orbital transfer and re-entry de-boost, while all thrust levels are required during the rendezvous maneuver. During rendezvous the following thrust combinations are required:

(a)	Longitudinal thrust:	400,000 lbs
	Normal thrust:	50,000 lbs
(b)	Longitudinal thrust:	50,000 lbs
	Normal thrust:	8,000 lbs
(c)	Longitudinal thrust:	8,000 lbs
	Normal thrust:	600 lbs

It is highly unlikely that the guidance engines and vehicle structure can be so mated that the thrust vectors always pass through the vehicle center of gravity. The firing of the guidance engines, therefore, produces disturbance torques which must be counteracted by the attitude control system. Table 2 contains the estimated maximum moment arms which could exist and the resulting disturbance torques exerted on the vehicle for each of the various engines.

Table 2 - Engine Misalignment Estimates

ENGINE SIZE, LB	400,000	50,000	50,000	8,000	8,000	600
AXIS LOCATION	ROLL	ROLL	YAW	ROLL	YAW	YAW
MAX MOMENT ARM ABOUT ROLL, FT	0	0	0.5	0	0.5	0.5
MAX ROLL MOMENT FT-LB	0	0	25,000	0	4,000	300
MAX MOMENT ARM ABOUT PITCH, FT	1	1	1	1	1	1
MAX PITCH MOMENT, FT-LB	400,000	50,000	50,000	8,000	8,000	600
MAX MOMENT ARM ABOUT YAW, FT	0.5	0.5	0	0.5	0	0
MAX YAW MOMENT, FT-LB	200,000	25,000	0	4,000	0	0

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Secondary injection thrust vector control will be incorporated into a separate attitude control loop to handle the large disturbance torques produced during the 400,000 pound thrust. The smaller engines will not use thrust vector control, consequently their disturbance torques will be imposed upon the normal attitude control system. The maximum total disturbance torques exerted by the smaller guidance engines are estimated to be:

Maximum pitch moment, ft-lb	25,000
Maximum yaw moment, ft-lb	58,000
Maximum roll moment, ft-lb	25,000

SECTION 3

MOMENT-PRODUCING DEVICES

3.1 MOMENTUM EXCHANGE SYSTEMS

3.1.1 Definition

A momentum exchange system employs a rotating device which generates a control moment by virtue of an exchange of angular momentum between the vehicle and the device. The most effective systems utilize either reaction wheels or single-degree-of-freedom gyros. Rotating spheres and fluid gyros have been studied, but their development has not progressed far enough for them to be considered practical. This study is concerned with the application of gyros and reaction wheels as moment-producing devices in the hypothetical vehicle.

3.1.2 Control-Moment Gyro

Figure 5 shows a schematic diagram of a single-degree-of-freedom gyro where the axes $\bar{e}_1$, $\bar{e}_2$, and $\bar{e}_3$ are locked in the gyro, and the axes $\bar{a}_1$, $\bar{a}_2$, and $\bar{a}_3$ represent the vehicle. The angular momentum

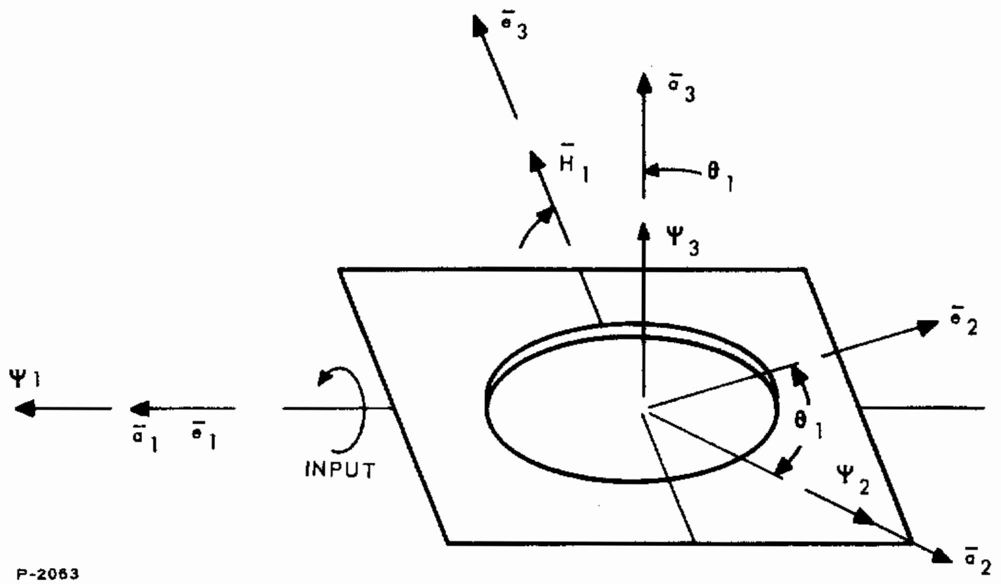


Figure 5 - Single-Degree-of-Freedom Gyro Schematic

vector $\overline{H}_1$ is directly along the $\overline{e}_3$ axis, and gyro precession is restrained to rotate only about the common $\overline{e}_1$, $\overline{a}_1$ axis. The ψ_1, ψ_2, ψ_3 terms represent angular velocities of the vehicle about the respective vehicle axes.

The $\overline{e}_2$ and $\overline{e}_3$ axes are initially aligned with the $\overline{a}_2$ and $\overline{a}_3$ axes. By precessing the gyro at a rate $\dot{\theta}_1$, a moment with a magnitude $H_1 \dot{\theta}_1$ is exerted on the vehicle along the $\overline{e}_2$ axis. As θ_1 goes from zero to $\pi/2$ radians, the moment vector rotates from the $\overline{a}_2$ axis to the $\overline{a}_3$ axis. It is seen then that the direction of the control moment on the vehicle is dependent upon the precession angle. This continual rotation of the moment vector requires that some form of compensation be incorporated to allow the resultant moment vector to be fixed in the vehicle. This compensation can be in the form of two gyros back-to-back which are precessed in opposite directions such that the cross-coupling moments cancel. Compensation can also be achieved by appropriately commanding two or more gyros not located on the same vehicle axis.

The precession of a gyro can be approximated by two methods which represent the minimum and maximum values of torquer size required for a given response time. In the first method the torquer both accelerates and decelerates the gyro to a new position by applying a constant torque for one half of the precession angle and then a reversed torque to stop the gyro during the last half of the precession angle. In the second method, the torquer accelerates the gyro to the desired final position and then some form of an external braking mechanism is used to stop the gyro at this position. Equations and appropriate curves were generated which relate the torque and weight requirements for both torqueing methods with single gyros per vehicle axis and back-to-back gyros per vehicle axis.

The curves for the second torqueing method are shown in Figure 6. Six curves are presented, the upper three curves pertaining to a single gyro along each axis and the lower grouping of curves pertaining to one gyro of a gyro pair along each axis. The following characteristics and parameters were used in computing the curves:

Vehicle inertia $J_2 = 4.36 \times 10^6$ ft-lb-sec^2

Change in vehicle angular rate $\Delta\psi_2 = 1 \times 10^{-3}$ rad/sec

Response time $t_o = 0.4$ sec

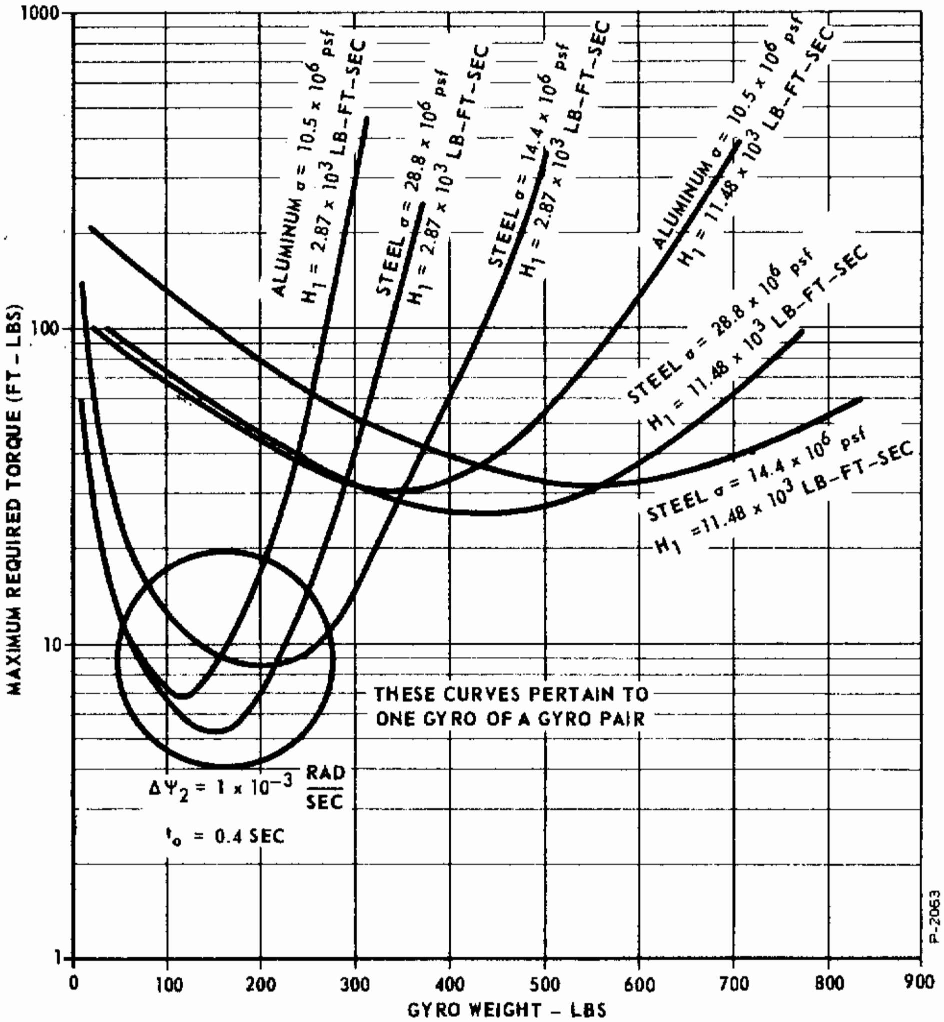


Figure 6 - Torque vs. Weight for Gyro Precession (Method 2)

Maximum precession angle $\theta_1 = \pi/2$ rad for gyro pairs

Maximum precession angle $\theta_1 = \pi/6$ rad for single gyro*

*This value of θ_1 was selected to minimize the cross-coupling effect of the gyro precession. This angle will produce 50 percent of the saturation impulse around the primary axis and only 14 percent of the saturation impulse on the orthoganal axis; a figure which lends readily to compensation by alternate gyros. For gyro pairs, however, a full 90 degree precession is possible.

The following materials were used for the gyro rotors:

(1) Aluminum with a maximum allowable stress of 10.5×10^6 psf (73,000 psi)

(2) Steel with a maximum allowable stress of 14.4×10^6 psf (100,000 psi)

(3) Steel with a maximum allowable stress of 28.8×10^6 psf (200,000 psi)

These curves are useful guides for selecting gyro size and estimating torque requirements.

3.1.3 Comparison of Control-Moment Gyro and Reaction Wheel

It is interesting and significant to compare the power requirements of gyros and reaction wheels of the same weight and inertia. Using, for example, the minimum torque point on the curves for the gyro pair as given in Figure 6, two gyros weighing a total of 320 pounds and two 5 ft-lb torquers are required per axis. The requirements upon which the gyro curves were derived are such that the vehicle must have an average acceleration of 2.5×10^{-3} rad/sec^2 during an attitude correction. If this acceleration is to be produced by a reaction wheel, the reaction torque driving the wheel must be 11,900 ft-lbs. Obviously, the reaction wheel driving torque requirement is excessive. In terms of power consumption, the comparison is similar, with the gyros using a total of 10 ft-lb of energy and the 320 pound reaction-wheel requiring an energy input of 1.143×10^6 ft-lb.

3.2 SECONDARY INJECTION THRUST VECTOR CONTROL

3.2.1 Description

Thrust vector control by means of secondary injection is a basically simple technique. The injectant fluid is fed into the diverging section of the rocket nozzle where it creates a bow shock. The resulting asymmetrical pressure distribution within the nozzle plus the jet reaction force of the injectant creates a side thrust. In effect, the rocket thrust vector is increased in magnitude and canted in the plane of the injector nozzle through an angle equal to the arctangent of the ratio of the side thrust to the axial thrust.

The secondary injection of inert liquids has received the most attention thus far because a liquid system is simple to operate and uses essentially state-of-the-art components. Hot gas, however, is the most effective injectant, but it is the most difficult to handle and control. The assumption made in this study is that hot gas bled from the rocket combustion chamber is used as the injectant.

It has been found that the following relationship describes the performance of the secondary injection phenomenon over most of the range of injectant flow rate:

$$K_a = \frac{F_s \dot{W}_a}{F_a \dot{W}_s}$$

where

K_a = amplification factor

F_s = side thrust, lb

F_t = total axial thrust, lb

$\dot{W}_s$ = injectant weight flow rate, lb/sec

$\dot{W}_a$ = total weight flow rate out of rocket nozzle, lb/sec

Indications are that amplification factors of 2 should be possible for vector angles up to 5 degrees when using hot gas injection.

3.2.2 System Requirements

The basic function of the secondary injection thrust vector control system is to correct for thrust vector misalignments, the maximum imposed moment being 400,000 ft-lb. Since the engine is located approximately 40 feet from the vehicle center of gravity, this represents a side thrust of 10,000 pounds and a vector angle of 1.4 degrees. Other important parameters are:

(a) Specific impulse 400 sec

(b) Amplification factor 2

(c) Main engine thrust 350,000 lb

The injectant flow rate required to obtain a 10,000 pound side force is 12.8 lb/sec.

3.2.3 Control Valves

The most critical item involved in implementing thrust vector control is the injectant flow control valve, because the performance of the control loop is directly dependent upon the accuracy, response, and reliability of this device. Two types of valves for this application were investigated, a simple poppet valve and a multi-stage vortex valve.

The poppet valve is shown schematically in Figure 7. It is a two-stage device which uses control gas from a source other than the main engine combustion chamber. A simple valve such as this is desirable from the standpoint of reliability, so long as stability and adequate gain can be achieved. A practical problem associated with this valve is the compatability of materials with the hot gases. The use of a relatively cool control gas will promote reliability, but the extreme corrosive and erosive characteristics of the combustion chamber gas strain the capabilities of presently available materials. The assumption

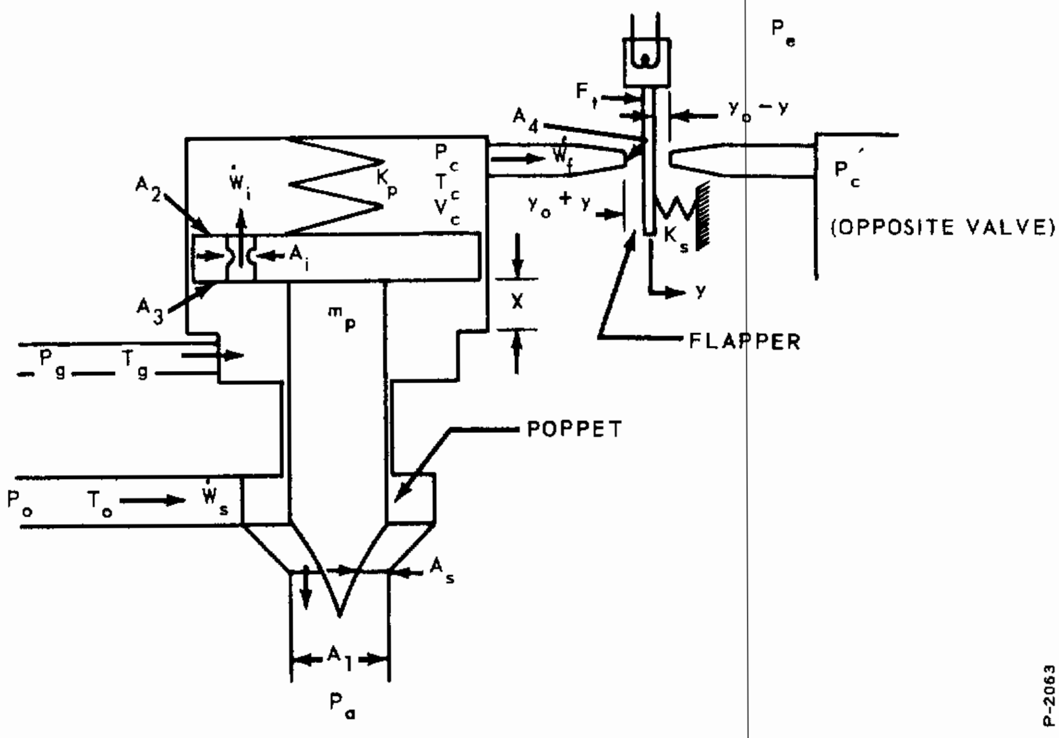


Figure 7 - Schematic Diagram of Secondary Injection Poppet Valve

is therefore made that materials technology will progress sufficiently far in the next few years to make this valve practical.

A linear analysis and subsequent design of the poppet valve resulted in the following two transfer functions which represent the extreme cases of zero and maximum flows:

$$\left(\frac{\Delta \dot{W}_s}{\Delta F_t}\right)_{\text{zero flow}} = \frac{1.23\,(3.82 \times 10^{-4}\, S + 1)}{\left[0.636\times10^{-12}\right] S^4 + \left[5.67\times10^{-8}\right] S^3 + \left[1.395\times10^{-4}\right] S^2 + \ldots \left[8.70 \times 10^{-4}\right] S + 1,}$$

$$\left(\frac{\Delta \dot{W}_s}{\Delta F_t}\right)_{\text{max. flow}} = \frac{0.69\,(2.32 \times 10^{-4}\, S + 1)}{\left[0.547\times10^{-12}\right] S^4 + \left[1.42\times10^{-8}\right] S^3 + \left[1.27\times10^{-4}\right] S^2 + \ldots \left[3.34 \times 10^{-4}\right] S + 1,}$$

where $\Delta \dot{W}_s$ is the incremental change in valve weight flow and ΔF_t is the incremental change in the force exerted on the flapper.

A vortex valve is an "all-fluid" control element having no moving mechanical parts. Valving action is produced by interaction between a control flow and the supply flow. A schematic diagram of the valve is shown in Figure 8. The main supply flow $\dot{W}_o$ of gas enters the valve downstream of the control inlet orifice and then passes into the main vortex chamber. In the absence of control flow, the main flow proceeds radially toward the central outlet orifice of the vortex chamber, while the addition of control flow imparts a rotational flow component to the supply flow as it passes the control inlet orifice. The function of the control flow is to attenuate the outlet flow, which it accomplishes by varying the "strength" of the fluid vortex. Over the range of operation, an increase in control flow produces a decrease in outlet flow.

The complete vortex valve system can consist of one or more vortex stages, depending upon the gain required. For a valve with three vortex stages, for example, the final stage is sized to handle the range of flow required. The control flow port is connected to the inlet of a smaller vortex second stage, and the control port of the second stage is in turn connected to the inlet of a still smaller, but

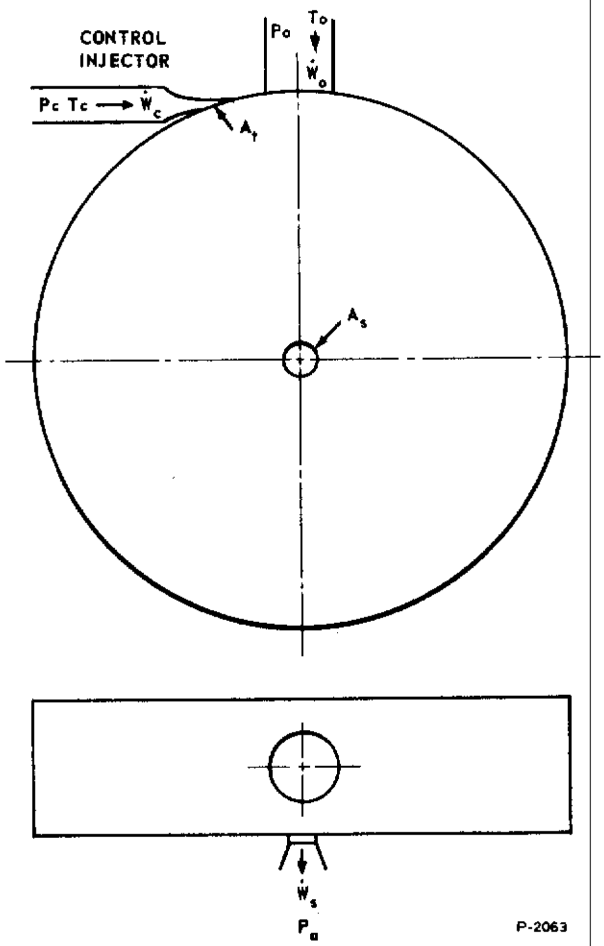


Figure 8 - Schematic Diagram of Vortex Valve

similar, vortex first stage. The control flow from the first stage can be controlled by a flapper nozzle valve in accordance with an electrical signal. Power gain in each vortex stage is achieved through the ability of a small change in angular momentum of the fluid in the chamber to greatly affect the flow through the chamber. The angular momentum is induced by tangential jets near the outer wall of the chamber which bias the valve at the high gain portion of its operating range. Changes in angular momentum are created by bleeding off a small flow to reduce the effect of the bias flow. When vortex stages are coupled, they must be properly matched so that the flow range of each driving stage is compatible with the control flow requirements of its following stage at the existing operating pressure level.

A linear analysis of the vortex valve was performed, and a preliminary design was made. The following are transfer functions for the extreme cases of minimum and maximum flows:

$$\left(\frac{\Delta \dot{W}_s}{\Delta F_t}\right)_{\text{min. flow}} = \frac{1.556\left[(8.22 \times 10^{-6})\, S + 1\right]}{\left[(2.98\text{x}10^{-6})\, S + 1\right]\left[(22.8\text{x}10^{-18})\, S^3 + (37.0\text{x}10^{-12})\, S^2 + \ldots\ldots\ldots\ldots (12.65\text{x}10^{-6})\, S + 1\right]}$$

$$\left(\frac{\Delta \dot{W}_s}{\Delta F_t}\right)_{\text{max. flow}} = \frac{1.376\left[(6.03\text{x}10^{-6})\, S + 1\right]}{\left[(2.98\text{x}10^{-6})\, S + 1\right]\left[(18.75\text{x}10^{-18})\, S^3 + (31.5\text{x}10^{-12})\, S^2 + \ldots\ldots\ldots\ldots (11.74\text{x}10^{-6})\, S + 1\right]}$$

The final selection of the injectant metering valve depends upon considerations of weight, reliability, and response. Without going into a detailed design, only a qualitative comparison can be made. As stated earlier, the reliability of the poppet valve would be poor, based upon the present state-of-the-art. Conversely, the reliability of the vortex valve can be expected to be excellent because of the absence of moving parts in contact with the main gas flow. When considering weight, the vortex valve will undoubtedly be heavier, but this disadvantage may be of secondary consideration. Insofar as response is concerned, the vortex valve is significantly faster than the poppet valve as seen from the respective transfer functions. From this study, there is no compelling reason for selecting one valve in preference to the other. In terms of potential, however, the vortex valve has a decided edge.

3.3 ATTITUDE CONTROL REACTION ENGINES

3.3.1 Introduction

The mass expulsion attitude control system, while being one of the more conventional means of developing space vehicle control torques, is nonetheless an ultimate choice in meeting the requirements for response, efficiency, weight, reliability, and versatility. The system basically consists of propellant tankage, manifolding, a positive shut-off control valve, a combustor and a converging-diverging nozzle. This section is concerned with propellant selection, tankage evaluation, system design and definition, and dynamic analysis including analogue computer simulation. Weight estimates have been made of the propellant and tankage of the controller. The valve design is discussed with regard to optimizing certain parameters to achieve optimum performance: Two possible modes of control, on-off and proportional are discussed and possible system design concepts presented.

3.3.2 Propellant Selection

The selection of a propellant combination for the reaction attitude control system involves considerable tradeoff with regard to the other vehicle systems. It is especially influenced by the availability of the main propulsion system propellants. Liquid hydrogen and liquid oxygen have been chosen for the propulsion system because of high specific impulse. They do have an inherent problem however in not being hypergolic and thus requiring an ignition system. The control system analytical studies show that the on-off or pulsing mode of control operation is the most desirable. This conclusion is then complicated by the requirement for the fairly sophisticated ignition system when capitalizing on the already available non-hypergolic propellants.

It was decided to make a tradeoff involving both a storable hypergolic bipropellant and the cryogenic liquid hydrogen - liguid oxygen. This comparison is made on the basis of performance, storage requirements, handling and contribution to system weight. Considerable analysis and application data is available for the variety of propellant combinations available.

Several convenient parameters are available for comparing the various promising propellant combinations. Those parameters chosen for this study were specific impulse, characteristic velocity, density impulse, and total system weight. An accurate system weight comparison must include propellant, tankage, and insulation weights. The rest of the control system would be basically the same for all propellant combinations.

The values of specific impulse for the two propellant combinations selected for comparison assuming an expansion ratio of 150 to 1 are:

$(I_{SP})_{HBP}$ = 290 sec (nitrogen tetroxide plus 50/50 UDMH and hydrazine)

$$(I_{SP})_{LOH} = 410 \text{ sec (liquid hydrogen-oxygen)}$$

For an estimated total impulse of 1,500,000 lb-sec, the propellant weights are:

$$(W_P)_{HBP} = 5180 \text{ lbs}$$

$$(W_P)_{LOH} = 3660 \text{ lbs}$$

Adding 15 percent for tankage, the comparative weight of the hypergolic bipropellant system is 5950 pounds. Adding 80 percent for tankage and 440 pounds for insulation, the comparative weight of the hydrogen-oxygen system is 7028 pounds. The comparative weight of the hydrogen-oxygen system is seen to be 18 percent greater than that of the bipropellant system.

This comparison was made assuming the hydrogen and oxygen for the attitude control system would be stored at supercritical pressures in separate tanks. However, if they were stored in the main engine tanks and pumped to the A.C.S., the weight necessary for comparison would essentially consist only of the propellant and insulation. When this is done the LOH system provides a 30 percent weight saving. It is therefore desirable to use liquid hydrogen and oxygen from the main engine tanks as the propellant for the mass expulsion attitude control system.

3.3.3 System Description

A schematic diagram of the reaction control system selected for this application is shown in Figure 9. This system is unique in that it can be operated in on-off or proportional modes or a combination of both. This provides unlimited flexibility in the design of the over-all attitude control loop.

The control valve without the proportional control feature is shown in Figure 10. The pilot-operated, two-stage configuration was chosen to meet response requirements. A single-stage valve capable of handling the large flows dictates excessively large electrical actuation devices, which should be avoided due to their limited response capabilities. The proposed valve utilizes a small solenoid capable of working from conventional power supplies. The valve operation is as

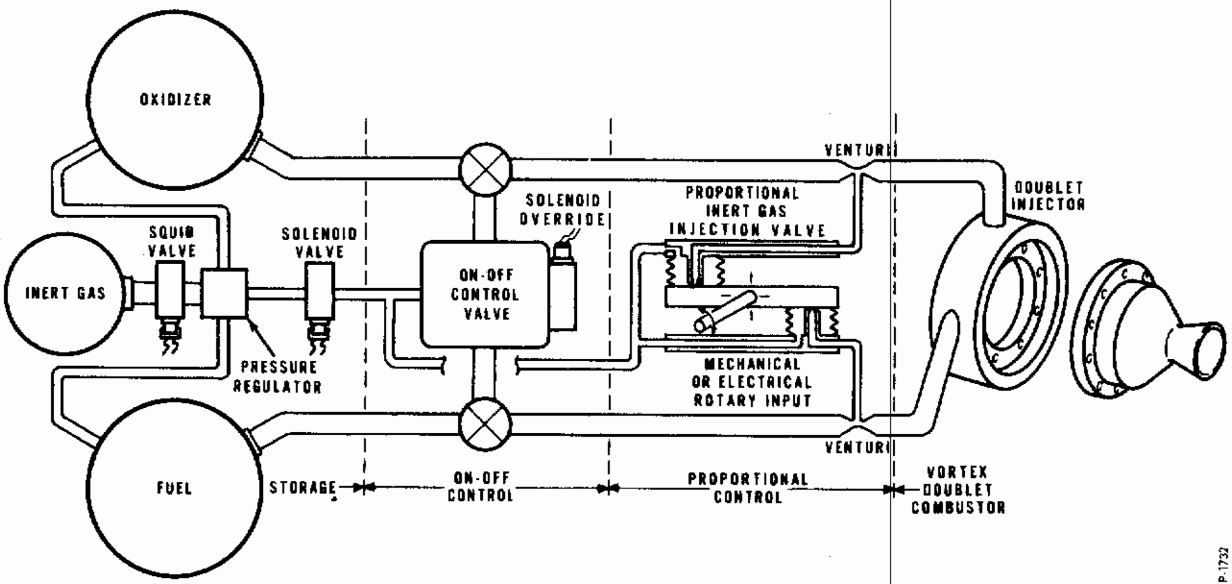


Figure 9 - Reaction Control System

follows: The pilot stage consists of a simple unbalanced area poppet actuated by a solenoid. The control signal actuates this solenoid allowing the control gas to flow into the main stage. When this pressure force builds up to a point sufficient to overcome the restraining forces the main stage piston moves. Attached to this piston by a tie bar are the main poppets which then open allowing fuel and oxidizer to flow into the combustor. When the signal is removed, the pilot stage poppet is closed by the gas acting on the unbalanced area, the gas in the main stage exhausts, and the spring preload closes the main poppets.

The method used in this system to achieve proportional control of thrust is an all-fluid technique which promotes good mixing for stable combustion at all chamber pressures. The fuel and oxidizer flow through cavitating venturis just prior to entering the injectors. At the throat of these venturis the propellants are at their vapor pressures and it is relatively simple to inject a control gas at this point. The injected gas displaces some of the propellant and flows into the combustor. The amount of gas displacement depends on its supply pressure. The system is designed so that the supply pressure is the same on each side and the oxygen-to-fuel ratio is maintained constant throughout the throttling range. The proportional valve, shown schematically in Figure 9, controls the gas flow to both sides. This valve

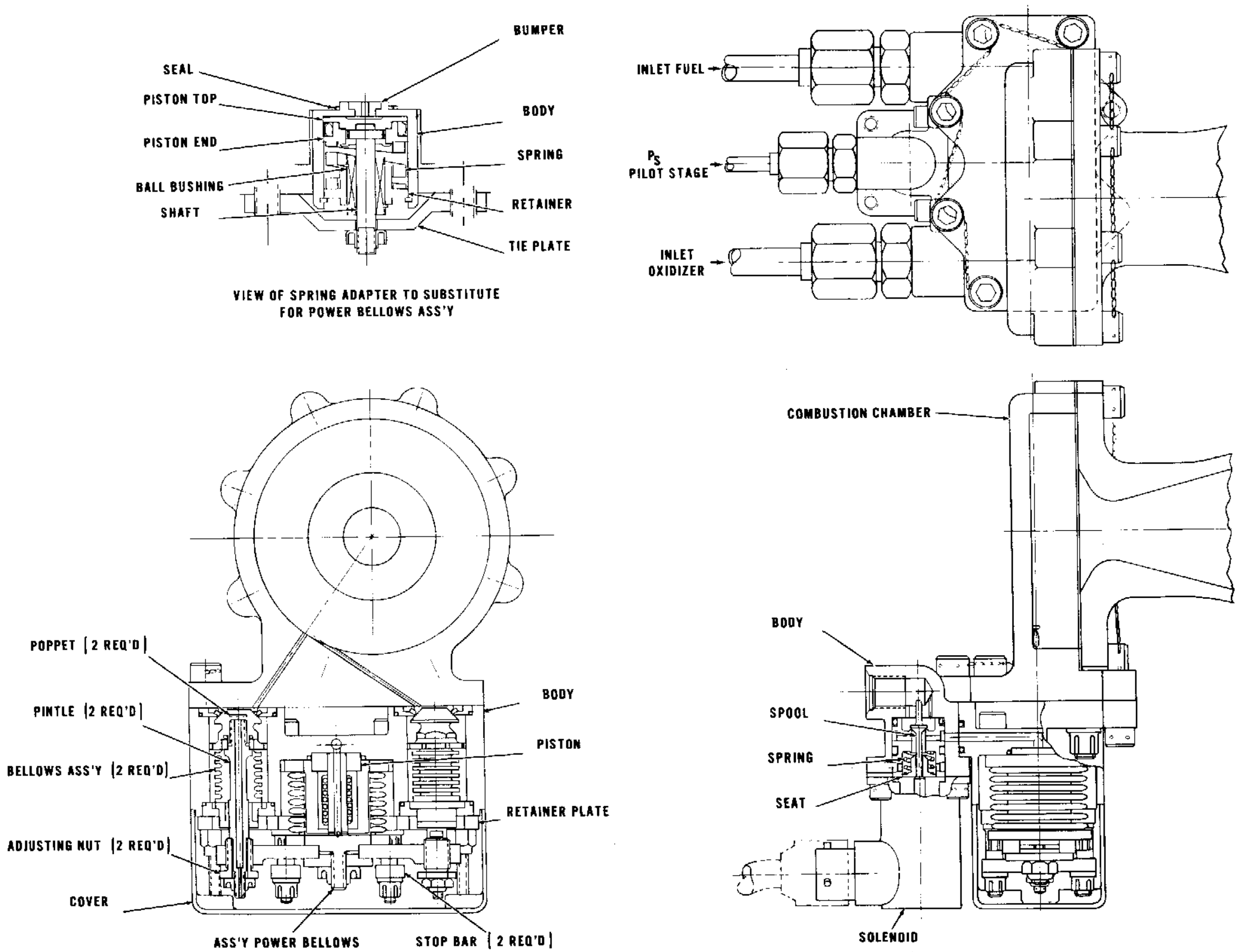


Figure 10 - Reaction Control Valve

is a very simple device and will not add much complexity to the system. During proportional control, the on-off valve is held open since the propellants flow through this valve at all times. This also allows the system to be operated in the on-off mode at different thrust levels by varying the control gas pressure.

This system will use a vortex doublet combustion chamber. In this device, the oxidizer is admitted radially and the fuel tangentially so that a vortex is induced with combustion occurring as the liquids spiral toward the center. Excellent mixing is accomplished and a long average burning path is inherent. The mixing is done as the liquids swirl away from the injector holes which allows the igniter to be positioned such that combustion does not affect the propellant flow into the chamber. Another advantage of this design is that the walls of the chamber are relatively cool because the higher temperature gases are confined to the center of the combustor.

3.3.4 Dynamic Analysis

A dynamic analysis of the valve-combustor system was performed, and the system performance was simulated on an analogue computer. The factors having the greatest effect on response time and stability were found to be the ignition time delay τ_i, combustion time constant τ_b, combustion chamber volume V_b, main stage control volume V_c, and pilot stage preload F_{sl}. We do not have much control over τ_i and τ_b, and the volumes V_b and V_c are determined by other considerations. Therefore the remaining effective parameter, F_{sl}, determined the optimum system and was increased to give approximately the same response for charging and discharging the main stage. The resulting optimum response is shown in Figure 11.

The valve-combustor system is mathematically complex and would make the computer simulation of the complete attitude control system a difficult task. Its response, however, indicates that it can be represented by a simpler block diagram, so after a series of computer manipulations it was found that the solenoid can be represented by a linear first order transfer function, and the valve dynamics by a single integration, saturation, and positive feedback. The combustion process was simplified by leaving out the ignition time delay. The parameters were adjusted in the computer circuit for the simplified dynamics to get the same response time, overshoot, and same number of oscillations, as for the original complete simulation.

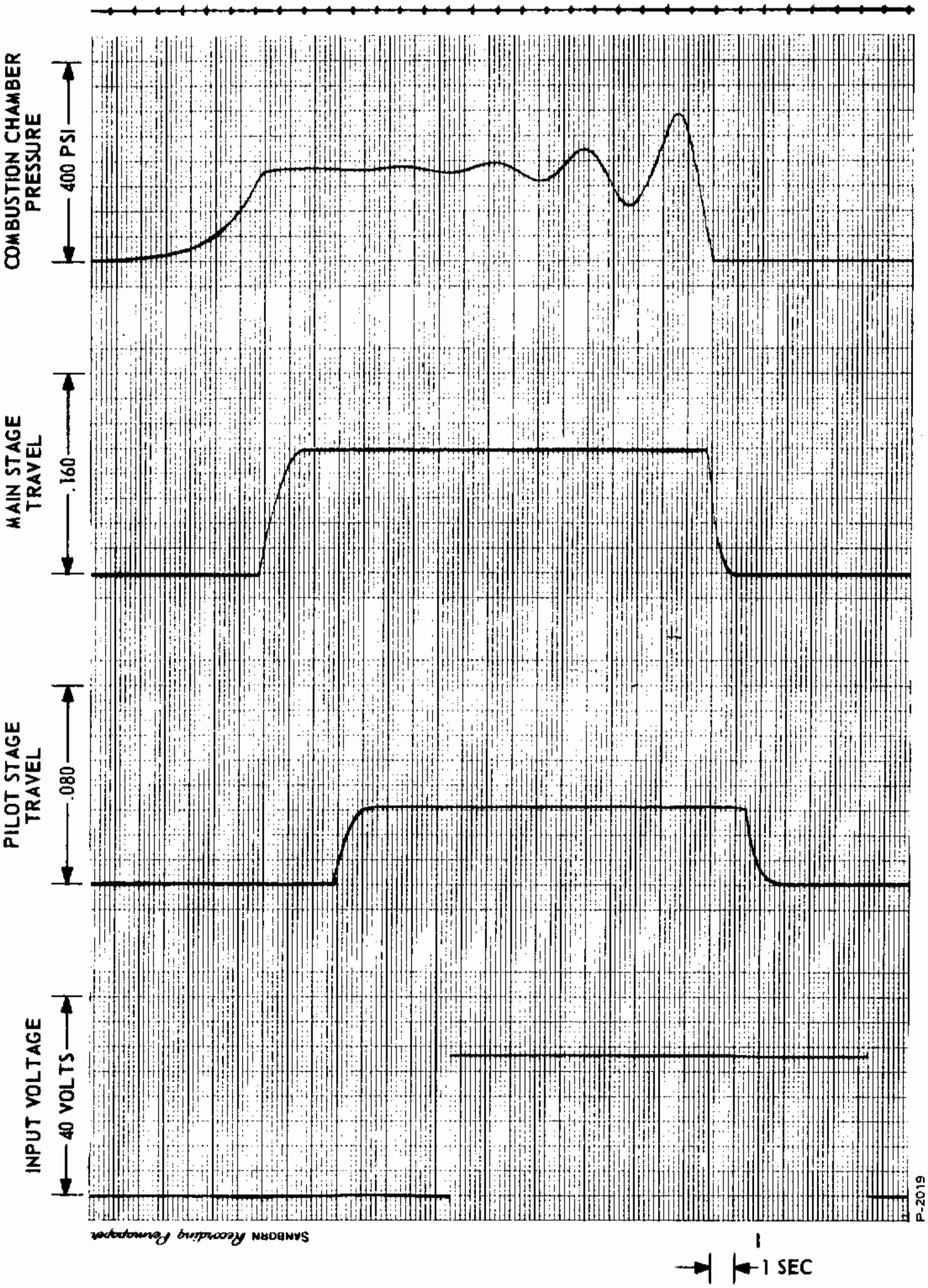


Figure 11 - Step Response of Optimum Valve - Combustor System

SECTION 4

ATTITUDE CONTROL DURING RENDEZVOUS AND ORBITAL TRANSFER

4.1 INTRODUCTION

4.1.1 Objective

The objective of this study is to select and devise an attitude control system which will meet all of the requirements dictated by orbital transfer and rendezvous maneuvers. In meeting this objective, a simulation of the control loop and vehicle dynamics must be accomplished.

4.1.2 Requirements

During rendezvous, range rate and angular rate corrections are made at frequent intervals. The guidance logic computes the required corrections from measured values and accordingly commands thrust pulses from the guidance engines. The attitude control system properly aligns the guidance engines' thrust vectors with respect to the line-of-sight between the vehicle and target. If sizable attitude errors occur, large errors are introduced into the guidance system, resulting in an unsuccessful rendezvous.

The average accuracies required by the attitude control system during a rendezvous guidance correction are specified as follows:

Axis	Maximum Average Error
Pitch	± 0.010 radian
Yaw	± 0.010 radian
Roll	± 0.020 radian

During orbital transfer, attitude errors of ± 0.020 radian about all axes can be tolerated. It is seen that the rendezvous accuracy requirements are the most stringent. On this basis, it will be assumed that a system which provides successful rendezvous performance will also be adequate for orbital transfer.

The desired control system must satisfy the functional requirement stated above, and in addition:

(a) Have a low response time

(b) Use a minimum amount of propellant in making attitude changes

(c) Minimize the number of switchings of the attitude jets

(d) Minimize the propellant consumption in the attitude limit cycle

(e) Minimize the complexity of the system.

4.2 ANALYSIS OF ATTITUDE CONTROL LOOP

The following four control loops were analyzed and optimized for use in the reaction engine control system:

(a) On-off control with lead-lag compensation

(b) On-off control with rate feedback

(c) Proportional control with lead-lag compensation

(d) Proportional control with rate feedback.

The choice of control loop is based on three criteria: average error, total propellant consumption, and reliability. The maximum average errors given in Subsection 4.1.2 must not be exceeded. While this criterion is necessary to accomplish rendezvous, the probability of rendezvous is increased by decreasing the average error during corrective action by the guidance engines. The propellant consumption is a critical factor since it determines the payload available for the vehicle. Sufficient fuel and oxidizer must be carried to account for the maximum or worst conditions that could be encountered. In this case, the propellant requirements are not severe because the attitude control system uses the same propellant as the propulsion system. The propellant requirements for attitude control for a short mission are much less than the propulsion engine requirements. System reliability may be evaluated on the basis of equipment complexity and duty cycle. The additional valve and electronic equipment required to operate a proportional reaction engine certainly increases the chance of failure even if these components are well developed and proven components. On the other hand, if a reaction engine operates many times due to a high frequency limit cycle, the probability of failure increases.

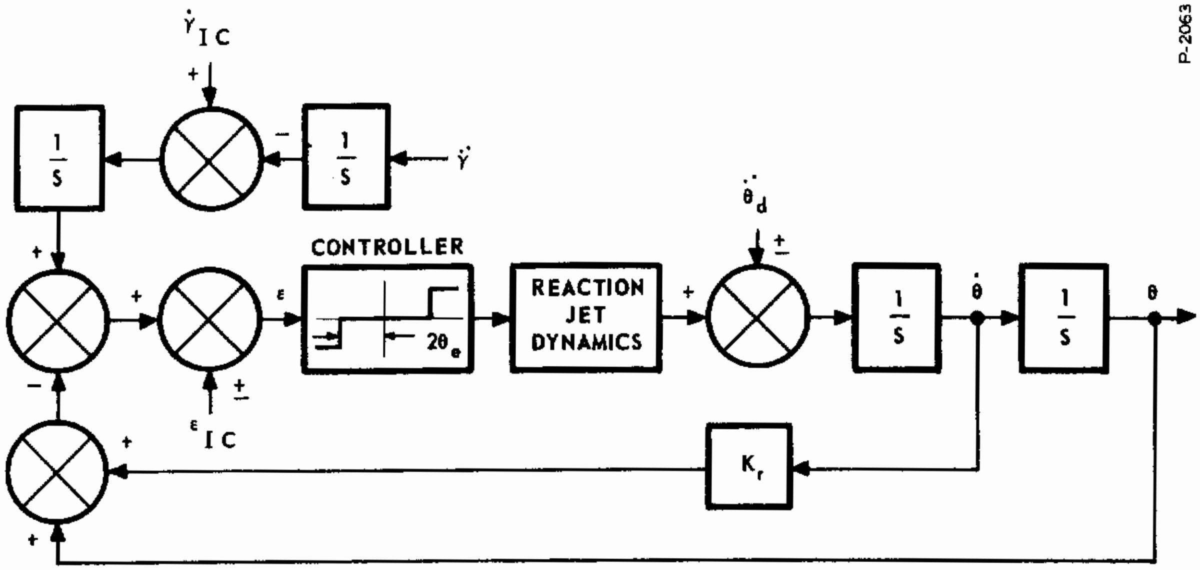


Figure 12 - Block Diagram of On-Off System with Rate Feedback Compensation

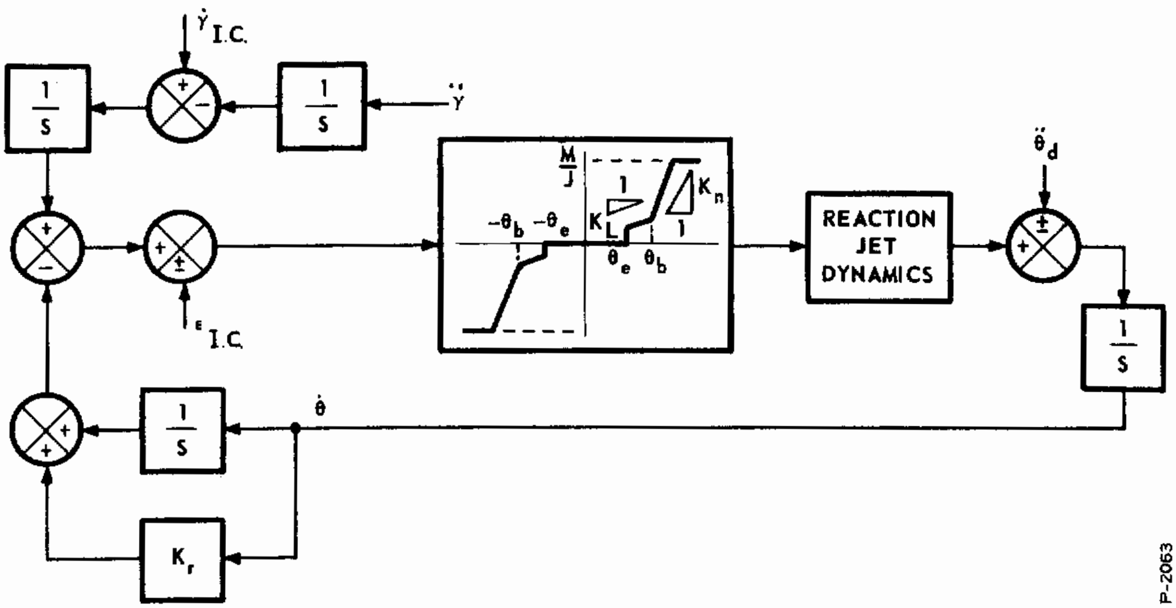


Figure 13 - Block Diagram of Proportional System with Rate Feedback

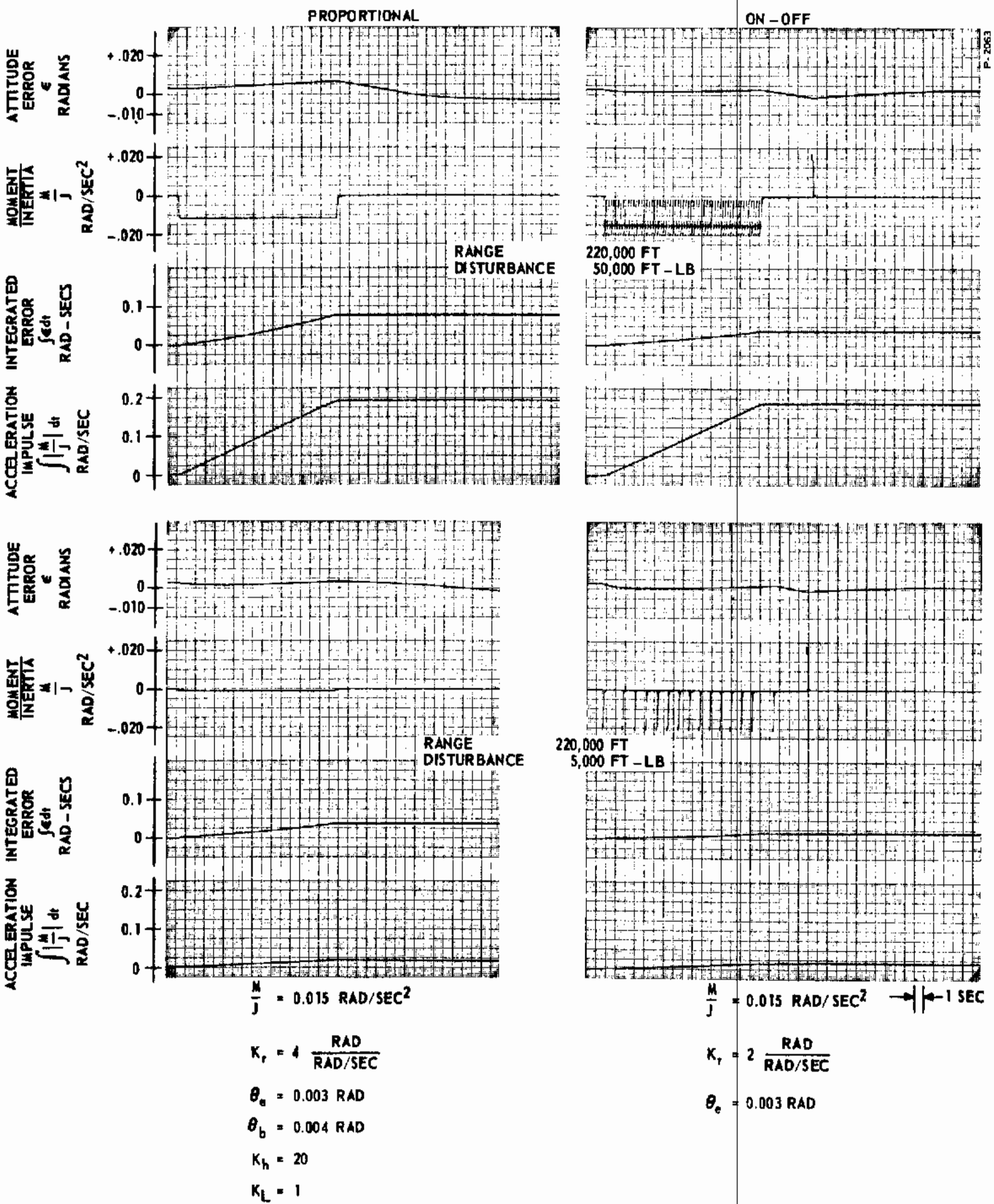
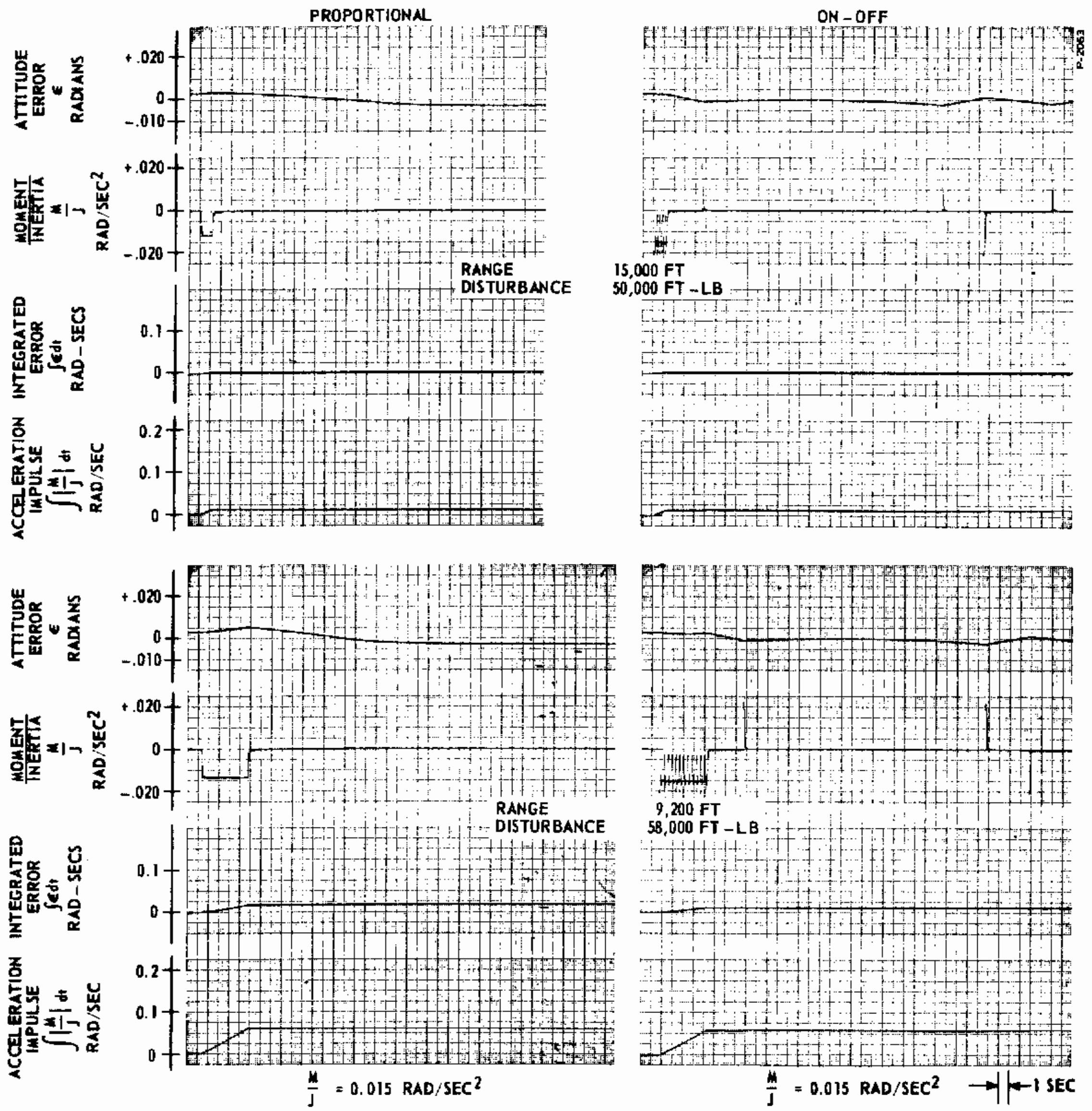


Figure 14 - Comparison of On-Off and Proportional Attitude Control Systems with Rate Feedback



$\frac{M}{J}$ = 0.015 RAD/SEC^2

$K_r = 4 \frac{RAD}{RAD/SEC}$

θ_e = 0.003 RAD

θ_b = 0.004 RAD

K_h = 20

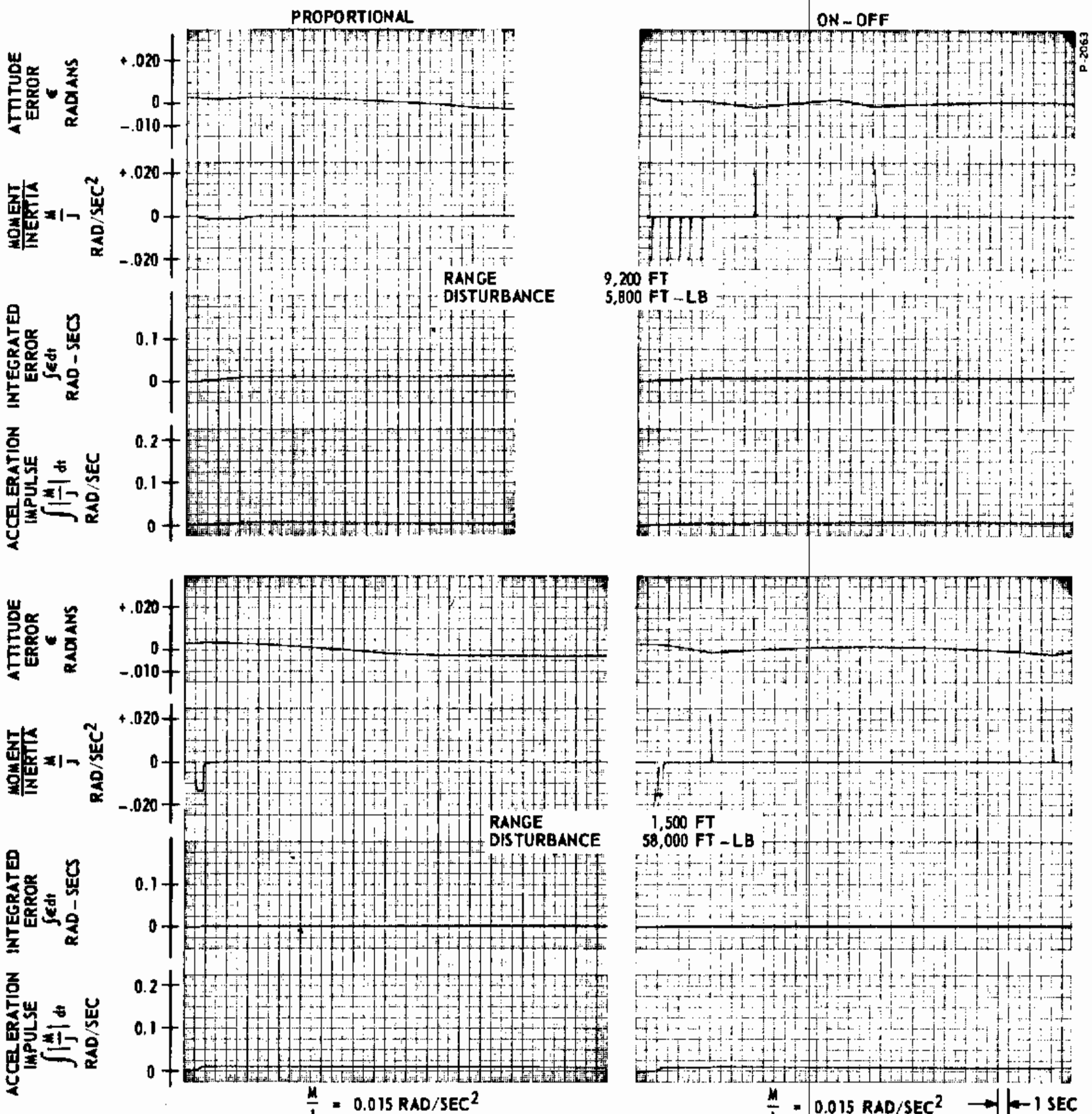
K_L = 1

$\frac{M}{J}$ = 0.015 RAD/SEC^2

$K_r = 2 \frac{RAD}{RAD/SEC}$

θ_e = 0.003 RAD

Figure 14 - Comparison of On-Off and Proportional Attitude Control Systems with Rate Feedback



$\frac{M}{J} = 0.015 \text{ RAD/SEC}^2$

$K_r = 4 \frac{\text{RAD}}{\text{RAD/SEC}}$

$\theta_e = 0.003$ RAD

$\theta_b = 0.004$ RAD

$K_h = 20$

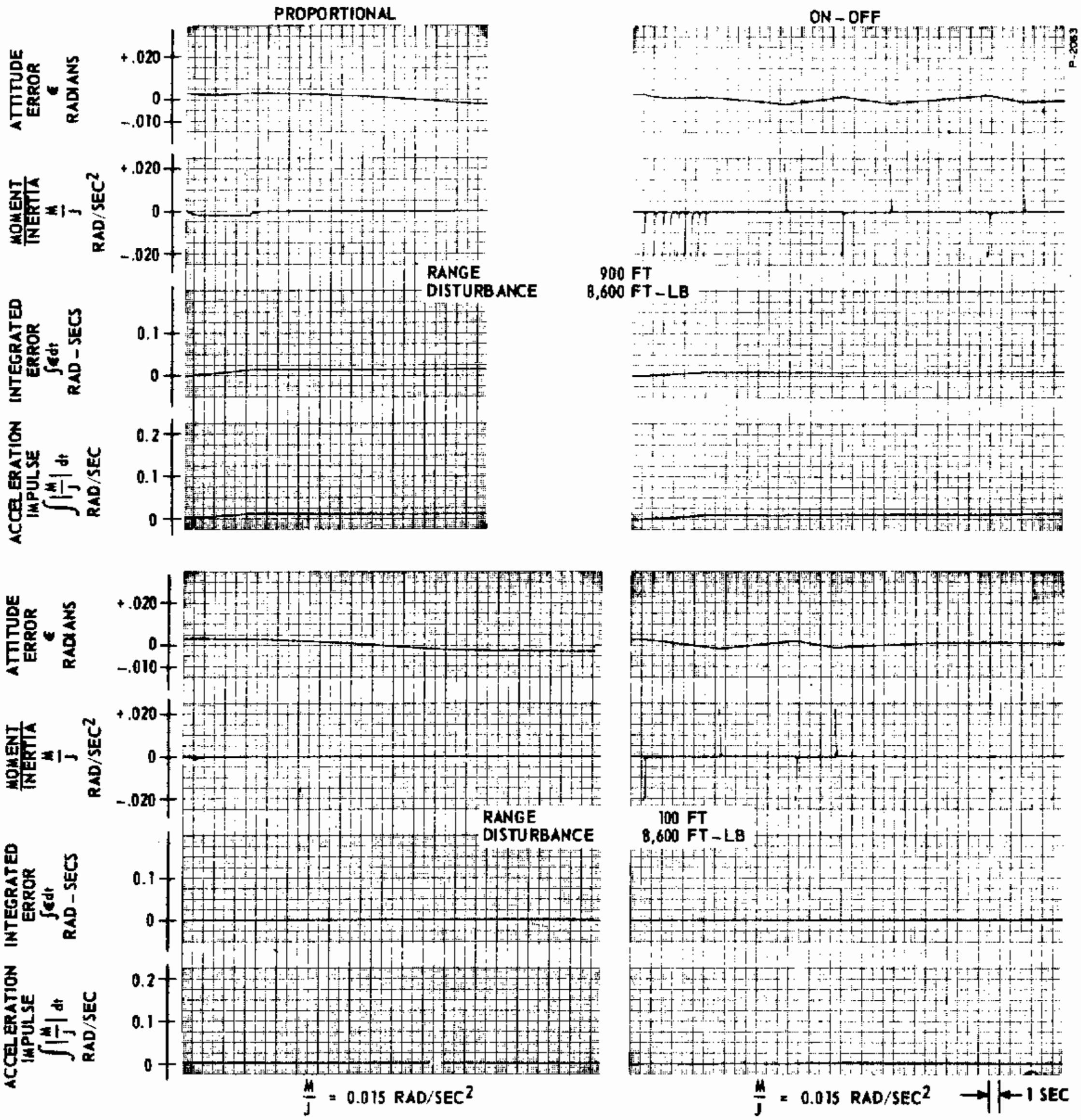
$K_L = 1$

$\frac{M}{J} = 0.015 \text{ RAD/SEC}^2$

$K_r = 2 \frac{\text{RAD}}{\text{RAD/SEC}}$

$\theta_e = 0.003$ RAD

Figure 14 - Comparison of On-Off and Proportional Attitude Control Systems with Rate Feedback



$\frac{M}{J}$ = 0.015 RAD/SEC^2

$K_r = 4 \frac{\text{RAD}}{\text{RAD/SEC}}$

θ_e = 0.003 RAD

θ_b = 0.004 RAD

K_h = 20

K_L = 1

$\frac{M}{J}$ = 0.015 RAD/SEC^2

$K_r = 2 \frac{\text{RAD}}{\text{RAD/SEC}}$

θ_e = 0.003 RAD

Figure 14 - Comparison of On-Off and Proportional Attitude Control Systems with Rate Feedback

It was found that rate feedback compensation is better for both the on-off and the proportional systems. (Refer to the block diagrams of these systems in Figures 12 and 13.) There remains the selection of one of the two systems as optimum based on the previously mentioned criteria. The response of the two systems to the conditions of rendezvous are shown in Figure 14. In each and every case the average error for the on-off system is equal to or less than the average error for the proportional system. The two average errors are about the same at a range of 15,000 feet, and at 100 feet the average errors are both negligible. On a propellant consumption basis, the proportional system generally uses less. This difference is due to the small amount of propellant consumed during the coasting limit cycle. It can be noted that the on-off system did not always go into a limit cycle immediately after removal of the disturbance. This certainly can occur given the proper minimum impulse obtainable and an appropriate set of initial conditions. For all comparisons of propellant consumption, it was assumed that the on-off system goes into a limit cycle immediately upon the removal of the disturbance and the accompanying data has been adjusted accordingly. If a rendezvous requires that, in each range, four guidance corrections are required, the total impulse or propellant consumption can be calculated based on maximum disturbances. For the proportional system, a total impulse of 1.15 rad/sec is required compared to 1.26 rad/sec for the on-off system. The 9.5 percent advantage amounts to about 11 pounds of propellant. This certainly is a small amount compared with the total amount of propellant that must be carried for guidance and propulsion.

From a reliability point of view, the added complexity of the proportional system has already been mentioned. An interesting phenomenon can be seen during coasting with the proportional system. In all cases the controller chatters many more times than the equivalent on-off system. For each condition except one, there are more firings of the proportional engine than the on-off engine. The one exception is for the 220,000 foot range with maximum disturbance where there are probably more oscillations during the disturbance for the on-off system than during coast for the proportional system. The total number of operations or firings whether maximum or minimum thrust is a reliability criterion because in each case the pilot stage and main stage valves must operate full stroke.

On the basis of the above discussion, the on-off system with rate feedback compensation was selected as the optimum system for the reaction engines. Further analysis showed this system to be adequate for the thrust vector control loop as well.

SECTION 5

ATTITUDE CONTROL DURING RE-ENTRY

5.1 INTRODUCTION

5.1.1 Objective

The attitude control system comprises an important loop within the re-entry guidance system. In fact, its performance determines the degree of success of the entire re-entry maneuver, since the vehicle angle-of-attack (pitch attitude) controls the lift and drag acting on the vehicle. Errors in angle-of-attack can cause the vehicle to enter the atmosphere too rapidly thereby burning up, or they can force the vehicle to skip out of the atmosphere into an undesirable orbit or trajectory. The performance of the attitude control system then is most properly defined in terms of deviations of the trajectory characteristics from optimum or commanded values. Subsection 2.3.2 defines what is considered to be an optimum trajectory. Accordingly, the objective of this study is to analyze and define an attitude control system for re-entry which will allow the vehicle to closely follow this hypothetical trajectory.

5.1.2 Problem Definition

The aerodynamic forces and moments acting on the vehicle during re-entry are correlated by expressions which contain the dynamic pressure q as a factor. Thus, before any numerical calculations of moments acting on the vehicle can be performed, the dynamic pressures must be known. The dynamic pressure is given by:

$$q = \frac{\rho V^2}{2}$$

where ρ is the atmospheric density and V is the vehicle velocity. During the early stages of re-entry, the dynamic pressure is small. Specifically, q is less than 2 psi for the first 96 seconds of re-entry and is never less than this value thereafter. This value of dynamic pressure is considered adequate for complete aerodynamic control of attitude. It is therefore concluded that during the first 96 seconds of re-entry, an additional source of moments for attitude control is required. The re-entry maneuver is considered to be made up of two parts, the first part involving both aerodynamic and mass expulsion control and the second part using only aerodynamic control.

The vehicle attitude during the first part of re-entry is controlled primarily by moments created by small rocket motors. The initial altitude is 400,000 feet, the initial speed is 26,000 ft/sec, and the initial path angle is 2.5 degrees. This portion of the flight lasts approximately 96 seconds and finally achieves an altitude of approximately 294,000 feet and a speed of approximately 25,200 ft/sec. A limit of 2 g's path deceleration is set for the entire re-entry but is not approached during the initial 96 seconds, therefore, no angle-of-attack α corrections are required. The flap angle δ_f is maintained at -30 degrees (flap up) which causes the vehicle to naturally trim at an angle of attack of 77 degrees. The aerodynamic attitude control moments are too small to provide adequate response during this portion of the re-entry, so augmentation by mass expulsion is necessary.

The second part of re-entry begins after approximately 90 to 100 seconds when the dynamic pressure reaches a sufficient level to allow complete aerodynamic control of the vehicle attitude. During this portion, the path deceleration reaches 2 g's, thus causing the guidance system to move the aerodynamic flaps in commanding a new angle-of-attack. From this point on, the angle-of-attack is corrected at somewhat regular intervals until pilot control is initiated.

5.2 FIRST PHASE OF RE-ENTRY

In the first phase of re-entry, the dynamic pressure q builds up from essentially zero to a level which is on the threshold of the range of values which it will assume for the remainder of the re-entry period. In the early stages of this initial period of re-entry, the level of dynamic pressure is so low that very low natural frequencies result if pitch control by purely aerodynamic means is attempted. On the other hand, if a zero flap deflection is assumed and an attempt to control pitch by using only mass expulsion is attempted, the required range of thrusts or impulse bits is excessively large. Thus one reaches the conclusion that a combination of the two modes of pitch control is required.

The vehicle dynamics are essentially provided by the mass expulsion system, since the aerodynamic flap is held fixed at an angle such that the natural aerodynamic forces will attempt to trim the angle-of-attack to its desired value. As soon as the aerodynamic moments reach a sufficient level, the mass expulsion system is turned off.

5.3 SECOND PHASE OF RE-ENTRY

The second phase of re-entry is defined as the trajectory where the vehicle is controlled solely by aerodynamic surfaces. Detailed interpretation of the vehicle aerodynamic characteristics together with the imposed dynamic pressures reveals that the control moments obtainable are more than adequate for attitude control. The aerodynamic natural frequency of the vehicle was found to lie in the range of 0.5 to 2 rad/sec.

The re-entry maneuver is described by an extremely complex block diagram. The complexity of this maneuver requires an extensive computer simulation program to determine if the vehicle can follow the proposed trajectory. In view of the fact that this study is concerned more with attitude control than guidance, only the flap servo loop was studied in detail.

5.4 FLAP ACTUATOR SELECTION

Three schemes for flap actuation were considered: reaction-jets on the flap, a piston-cylinder driving a linkage, and an expansion vane motor driving through a high-ratio transmission. It was immediately determined that the thrust and fuel requirements of the reaction jets are enormous in comparison with the other schemes under consideration. With the reaction-jets rejected, a more detailed examination of the piston-cylinder and vane motor characteristics was made.

An expansion-type vane motor controlled by a spool-type servo valve was selected for actuating the flaps. This device was shown to have superior response, stiffness, and fuel consumption characteristics. A sketch of the valve-motor combination is shown in Figure 15.

5.5 AERODYNAMIC CONTROL LOOP

The flap control loop was analyzed and designed using appropriate linearizations and simplifications to put it into a reasonably simple but representative form. In so doing, the loop shown in Figure 16 was obtained. The closed loop transfer function of the simplified system is of the form:

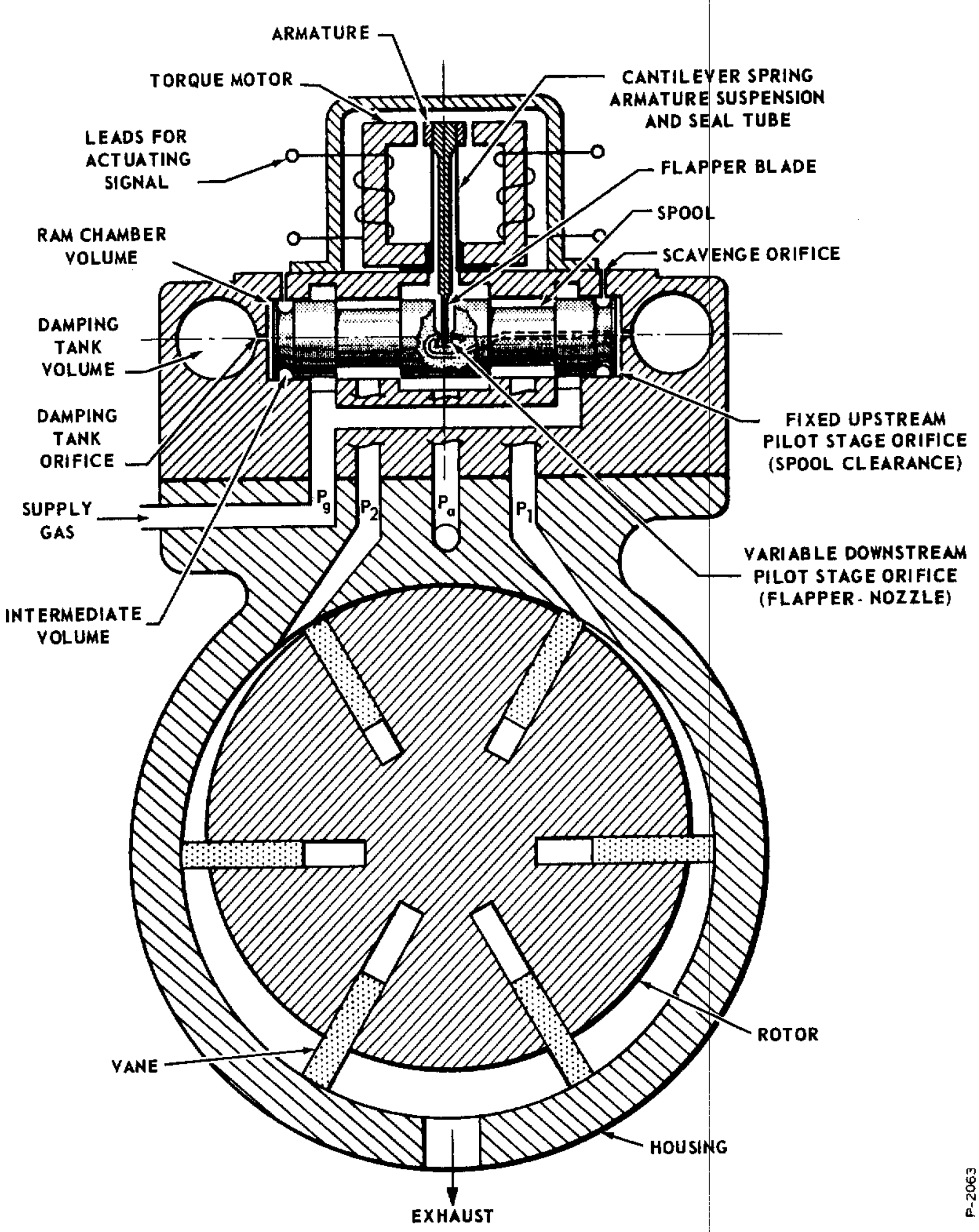


Figure 15 - Pneumatic Servo-Valve and Motor

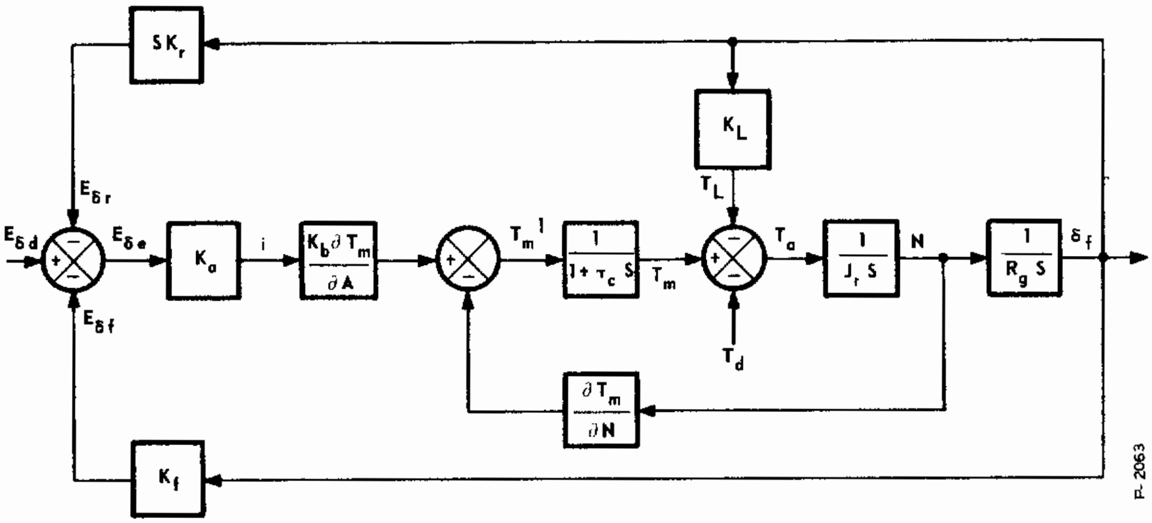


Figure 16 - Simplified Flap Servo System Block Diagram

$$\frac{\delta_f}{E_{\delta d}} = \frac{b_1}{a_3 S^3 + a_2 S^2 + a_1 S + a_o}$$

where:

$$b_1 = \frac{K_a K_b \frac{\partial T_m}{\partial A}}{K_L + K_a K_b K_f \frac{\partial T_m}{\partial A}}$$

$$a_3 = \frac{R_g J_r \tau_c}{K_L + K_a K_b K_f \frac{\partial T_m}{\partial A}}$$

$$a_2 = \frac{R_g J_r}{K_L + K_a K_b K_f \frac{\partial T_m}{\partial A}}$$

$$a_1 = \frac{K_L \tau_c + R_g \frac{\partial T_m}{\partial A} + K_a K_b K_r \frac{\partial T_m}{\partial A}}{K_L + K_a K_b K_f \frac{\partial T_m}{\partial A}}$$

$$a_o = 1$$

The parameters and constants selected and computed for the system given by Figure 16 are:

$$K_a = 8780 \text{ amp/volt}$$

$$K_b = 0.0232 \text{ in.}^2\text{/amp}$$

$$\frac{\partial T_m}{\partial A} = 2.66 \times 10^4 \text{ in.-lb/in.}^2$$

$$\frac{\partial T_m}{\partial N} = -1.576 \text{ in.-lb-sec/rad}$$

$$\tau_c = 0.00123 \text{ sec}$$

$$J_r = 0.0482 \text{ in.-lb-sec}^2$$

$$R_g = 9580$$

$$K_L = 519 \text{ in.-lb/rad}$$

$$K_f = 57.3 \text{ volt/rad}$$

$$K_r = 0.139 \text{ volt-sec}$$

The frequency response of this system has a resonance peak of 2 db occurring at a frequency of 4.35 cps. The phase shift at this point is 135 degrees. The -3 db point occurs at 5.32 cps with a phase shift of 205 degrees. The transient response has a 10 percent overshoot and four oscillations.

5.6 SUMMARY AND CONCLUSIONS

There is no problem in maintaining attitude control during the initial part of re-entry where both mass expulsion and aerodynamic control are used. With the flap set to trim the vehicle at the required angle-of-attack, the mass expulsion system need only maintain stability and overcome extraneous disturbance torques exerted on the vehicle. The mass expulsion attitude control system which is used for the rendezvous maneuver will be more than adequate for re-entry.

Attitude control during the latter part of re-entry is dependent upon the servo system which actuates the flaps. Essentially, the frequency response of the servo system must be substantially greater than the 0.5 to 2 rad/sec aerodynamic natural frequency of the vehicle.

The actuator selected for driving the flaps is an expansion vane motor controlled by a spool-type servo valve. Reaction jets are completely out of the question for flap control because of the enormous quantities of fuel required. A piston-cylinder lacks the performance and fuel economy of an expansion vane motor.

The flap control system was designed which has adequate performance for controlling vehicle angle-of-attack during re-entry. Based upon a simplified mathematical model, the servo system has a frequency response at least an order of magnitude greater than the maximum aerodynamic natural frequency of the vehicle.

The analytical approach involved a great deal of linearization and simplification. This was done because it allowed a preliminary system design to be readily accomplished without the expense of a complex computer simulation of re-entry. The accomplished work is considered to be adequate to the extent of defining and sizing the high moment-producing method and system required.

SECTION 6

CONCLUSIONS AND RECOMMENDATIONS

6.1 CONCLUSIONS

A manned space vehicle which executes orbital transfer, rendezvous, and re-entry maneuvers has been studied with regard to its attitude control requirements and the means for meeting these requirements. For each maneuver, a particular trajectory or guidance logic was selected, the attitude control requirements were determined, and an attitude control system was selected and designed. For the relatively short mission duration of the hypothetical vehicle considered, the optimum attitude control system for space maneuvers employs mass expulsion reaction engines and uses secondary injection thrust vector control when necessary. The re-entry maneuver requires a combination of mass expulsion and aerodynamic control during the initial stages and pure aerodynamic control during latter stages of the maneuver.

Momentum exchange systems provide greater attitude accuracy than do mass expulsion systems, but they become large and complex when high control moments are required. Also, a momentum exchange system requires the presence of a properly integrated mass expulsion system to provide de-saturization. For the application studied in this report, a mass expulsion system can provide the necessary attitude control accuracy, thus eliminating the need for an auxiliary system.

A large space vehicle with a long mission duration might use a momentum exchange system to advantage, particularly for the lower moment control functions such as stabilization. For this case, control moment gyros are preferred to reaction wheels because of their significantly lower power requirements.

The guidance engines for the hypothetical vehicle were assumed to use liquid hydrogen and liquid oxygen as the propellant combination. A comparison of this fuel and oxidizer with other bipropellants reveals that liquid hydrogen and liquid oxygen are preferred for use with the attitude control reaction engines. Strong points in this argument are the advantages inherent in having a common propellant supply for all engines on board.

Secondary injection thrust vector control is a convenient and efficient means for compensating for engine misalignment and maintaining attitude control during main engine firing. Fast response of the secondary injectant flow is obtained by using an all-fluid valving system; however, sufficient information concerning the dynamic response of the thrust vector deflection phenomenon is not available to allow the complete loop characteristic to be accurately defined.

The selection of a control system for the rendezvous phase of a manned space vehicle with a space station has been studied and a conclusion formed on what constitutes a good control system. The final configuration selected consists of a secondary injection thrust vector control system for major corrections when the main engine is firing and a reaction control system for lower magnitude attitude corrections and orientation. The secondary injection system is a simple control loop which uses hot gases from the main engine combustion chamber. An optimum control loop for both the reaction engine system and the secondary injection thrust vector control system uses an on-off controller with deadband and rate feedback. The two systems cross-couple through the vehicle dynamics but function in complete harmony.

Attitude control during the latter part of re-entry is dependent upon the servo system which actuates the aerodynamic flaps. A linearized analysis indicates that a proportionally controlled pneumatic system using a closed center valve and an expansion vane motor will provide adequate response with low gas consumption.

6.2 RECOMMENDATIONS

Further study is desirable in some of the many areas covered by this report. It was not possible in all of the analyses and investigations to go into as much detail as possible, and in some cases, the selection of a particular component, system, or technique had to be made on the basis of intuitive judgment due to lack of information available. Specific recommendations for further research and study are contained in the following paragraphs.

It is altogether possible that certain guidance techniques have less stringent attitude control requirements than others. In particular, the rendezvous maneuver, including the docking phase, can be executed with a number of different logic schemes. Rendezvous guidance should be studied from the standpoint of performing a successful maneuver with the least demands upon the attitude control system, the goal being to increase reliability and reduce propellant consumption. The scope of this study should include the analogue simulation of the complete dynamics of the integrated attitude control and guidance systems.

The use of liquid hydrogen and liquid oxygen as the propellant for attitude control engines should be studied further and in greater detail to more firmly establish feasibility and practicability. There are a number of problems involved in providing the propellant in a liquid state at the reaction engines. The application of efficient thermal insulation must be considered as well as the problems associated with the boil-off which will result due to heat transfer and heat generation. The effect of the cryogenic liquids upon the propellant supply system design and operation should be determined, particularly with regard to the dynamic elements.

Additional analysis of the reaction engine attitude control system is required, now that the system has been defined, to establish complete system design criteria with emphasis on the following considerations:

(1) Establish more realistic analytical expressions for the combustion dynamics.

(2) An often neglected part of this system is the tankage and manifolding. The valves and nozzles are remotely located from the propellant storage tanks. The line dynamics become complex and are significant in optimizing performance. A general design analysis anticipating the variety of operating conditions and configurations is required. A distributed parameter analysis and analog simulation would be invaluable in this area.

(3) Further analysis and analog computer simulations is required with regard to choice of the proportional or on-off control mode, since a more accurate description of system non-linearities is being achieved as the technology is advanced. The relative effects of significant parameters on performance must be defined to assure the best design based on tradeoff parameters such as accuracy, simplicity, weight and reliability. The basic control system models have been established, and further analysis would improve their validity and worth to the vehicle designer.

A computer simulation of the complete re-entry maneuver should be performed wherein all of the nonlinearities of the dynamics are included. This is a sizeable effort, but it is the logical next step beyond the work reported here. Such a simulation will allow the attitude control system to be optimized for all of the dynamic pressure conditions of the complete re-entry trajectory. Also, it will positively verify that the attitude control system will allow the vehicle to follow the desired trajectory.