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APOLLO

GUIDANCE, NAVIGATION AND CONTROL

E-1964

APOLLO COMMAND AND SERVICE MODULE REACTION CONTROL BY THE DIGITAL AUTOPILOT

by
R. Crisp & D. Keene

May 1966

MIT INSTRUMENTATION LABORATORY
CAMBRIDGE 39, MASSACHUSETTS

INSTITUTE OF TECHNOLOGY

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GUIDANCE AND NAVIGATION

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APOLLO COMMAND AND SERVICE MODULE
REACTION CONTROL, BY THE DIGITAL AUTOPILOT

ABSTRACT

An Apollo Guidance Computer Program developed to control the reaction jets of the Service Module of the Apollo spacecraft is described. Design philosophy is discussed, although the main design restraints are the existing hardware design, and maneuver requirements evolved at the implementation meetings for Apollo Block II CSM G&C systems. In general, the translation and rotation manual controls are implemented in the same way as the Block I SCS except simultaneous translation and rotation accelerations are possible. Automatic maneuver and attitude hold are instrumented in such a way as to conserve reaction control propellants.

The maneuver instrumentation was designed and evaluated using a flexible vehicle model with no propellant motion. Current slosh models look on propellant motion as a source of disturbing torques. However, analog simulations have been made with new slosh models where the propellant motion is coupled with vehicle response.

A theoretical study has been made of propellant utilization in generalized automatic maneuvers, and is compared with figures from three-degree-of-freedom digital simulations. The theoretical figures give a good estimate of the simulated propellant utilization.

Conclusions are made regarding the current design, future work, and simulation plans.

by Robert Crisp
Donald Keene

April 1966

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SECTION 1

INTRODUCTION

Early in 1964 the concept of incorporating a digital autopilot into the Apollo Guidance and Navigation System was discussed at a series of implementation meetings leading to the Block II CSM Guidance, Navigation and Control. system. Basically, by increasing the logical powers of the Apollo Guidance Computer (AGC) and the size of the system interface, an increase in the system redundancy and reliability was obtained for little additional equipment size or weight.

The design of the digital autopilot fell into three parts; thrust vector control, reaction control, and reentry control (aside from those in the LEM). The part discussed in this report is the reaction control of the Command and Service Module (with and without LEM docked). The design of the computer program has been through many alterations due to hardware and mission constraints and changes to these, and it would be impossible to chronicle all the phases that the design has been through. The presentation here is only of the current design.

1.1 Apollo Reaction Control Requirements

The Apollo Spacecraft (Fig. 1. 1) consists of a reentry capsule - the Command Module - attached to a propulsion and support unit - the Service Module. During trans-lunar flight the spacecraft is docked with the Lunar Excusion Module (LEM, shown on right side). In order to maintain attitude control, to make rotational maneuvers, and to perform small translations of the spacecraft, the Service Module is equipped with sixteen reaction jets mounted in four Quads as shown.

Major propulsive maneuvers are performed by the main SPS engine. In the Block II system the Apollo Guidance Computer has direct interface with the actuators of the Command and Service Module reaction jets and the SPS engine. We are concerned here with the AGC program that actuates the Service Module reaction jets when not performing an SPS engine firing.

Briefly the requirements of the free-fall reaction control program are as follows:

a. The AGC is to have reaction control capability at all times when the GN&C system is on.

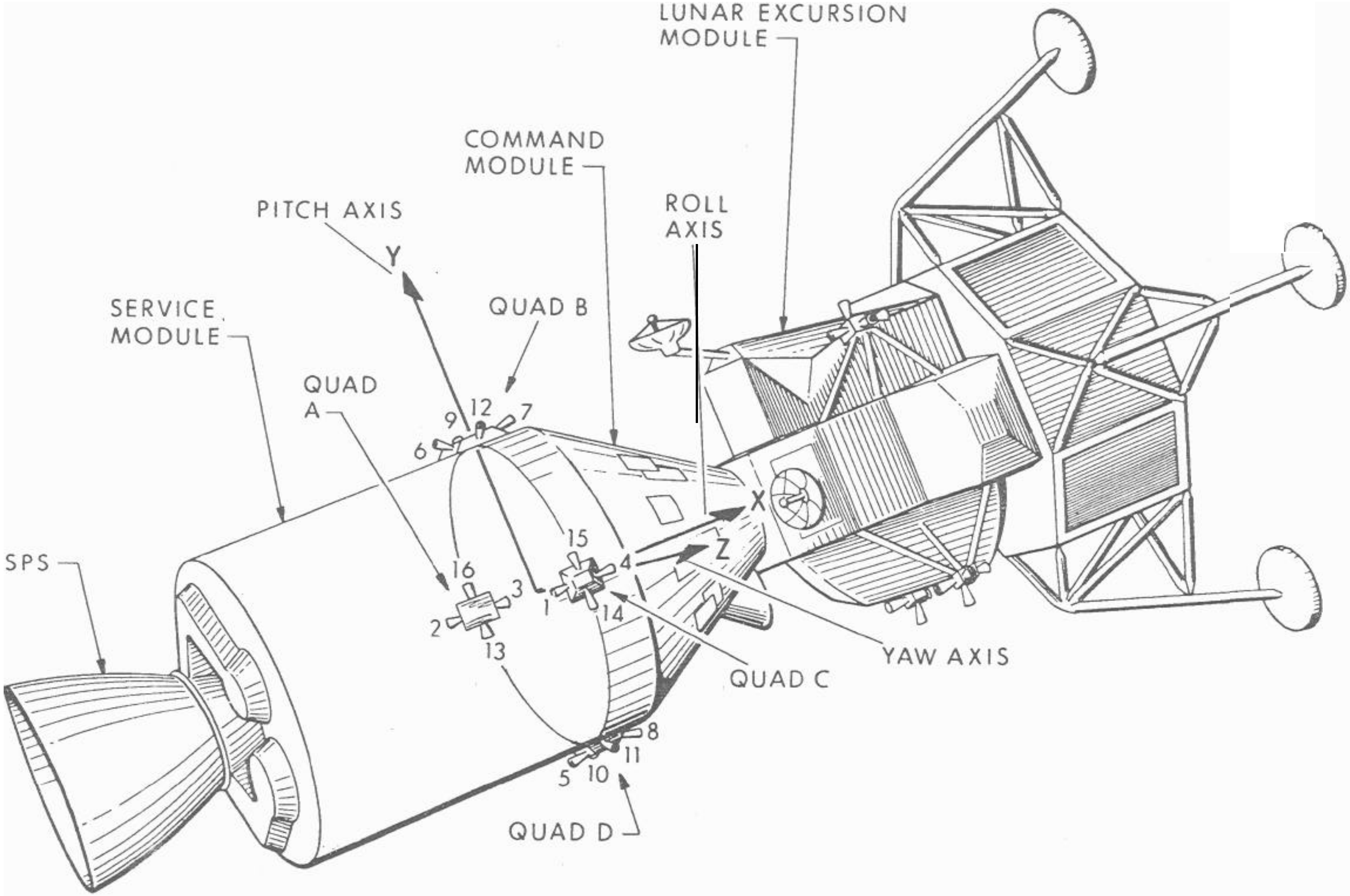


Fig. 1 1 Apollo Spacecraft

b. Attitude control feedback is to be obtained via the IMU and CDU's ie., the gimbal angles relative to the stable member.

c. Manual translation and rotation controls interface with the AGC and are to have priority at all times with rotations taking precedence.

d. A three-position mode control switch interfaces with the AGC. In the "Free" mode, no jet firings will be made except for rotational and translational accelerations in response to the manual controls, including the minimum impulse control. In the "Hold" mode, attitude is held close to that reached on switching to hold, or on terminating a manual rotation, except that, if the vehicle is tumbling, the rate will be limited first (see g. below). In the "Auto" mode, the mission program will determine the control mode; however, in both Hold and Auto modes manual rotation commands will cause the vehicle to move at a predetermined rate about the appropriate axes, and translation commands will cause all the appropriate jets to fire whenever possible.

e. The AGC is to have an automatic maneuver capability to perform simultaneous 3-axis maneuvers with a limited angular rate. As far as possible the jets are to be fired so as to achieve the highest specific impulse.

f. Rate damping will be provided to limit the maximum vehicle angular rate to a level somewhat above maneuver rates.

g. In attitude hold the system must settle to a minimum impulse limit cycle except in the presence of disturbing torques where the largest possible limit cycle amplitude should be maintained within the deadband. Both a wide and a narrow deadband are required.

h. The system must be operable should any single quad, or any two adjacent thruster quads become inoperative.

i. The system must be capable of operating in a single-jet mode in attitude hold.

j. Where possible simultaneous translation and rotation should be made.

k. The system operates throughout in control axes aligned with the SM thrusters, ie., displaced -7.25 deg. in roll from structure axes.

SECTION 2

DESIGN PHILOSOPHY

In essence, the reaction control system is required to make commands to the reaction jets on the Service Module in order to control vehicle attitude and vehicle rates. Modern control theory shows that an optimal or quasi-optimal controller (auto-pilot) requires measurement or estimation of the complete state of the plant (vehicle) dynamics. Since the spacecraft can be modeled as a double-integral plant, measurement of both attitude and the derivative of attitude (attitude rate) will be needed. Be cause the inertial measuring unit acting through the Coupling Data Unit provides attitude information only, attitude rates must be computed from the CDU information. For an optimal estimation of the vehicle rates, the commanded angular accelerations must also be included in this computation.

Once the measurements of the state, or in this case the errors in the state, have been obtained, they may be processed by nonlinear switching functions in the computer to generate ON-OFF commands to the RCS jets. A jet selection logic can then be used to select the individual jets to be fired. This design philosophy leads to a control loop as shown in Fig. 2. 1.

In designing the switching logic for this system, the performance of the reaction jets was studied to achieve efficient operation. Figure 2. 2 shows the total impulse of a jet plotted against the length of time the electrical “on” signal is applied. Also plotted is the specific impulse, ie. the ratio of the time integral of thrust to the weight of propellants burned.

Whereas the total impulse varies linearly with “on” time, the specific impulse drops seriously for short firings. Thus, in order to obtain a high specific impulse and to use the least propellant possible, changes in vehicle rate should ideally be made by one long firing rather than several shorter firings. To achieve these long firings we require a good estimate of vehicle rate so that the firing time to obtain a demanded rate can be accurately computed.

Since the vehicle dynamics can be modeled as a second-order system, phase plane methods can be conveniently used to describe the vehicle state. Thus, it is convenient to look on the control problem as one of estimation of state, and to use the phase plane together with the associated switching logic to describe the changes in state of the vehicle.

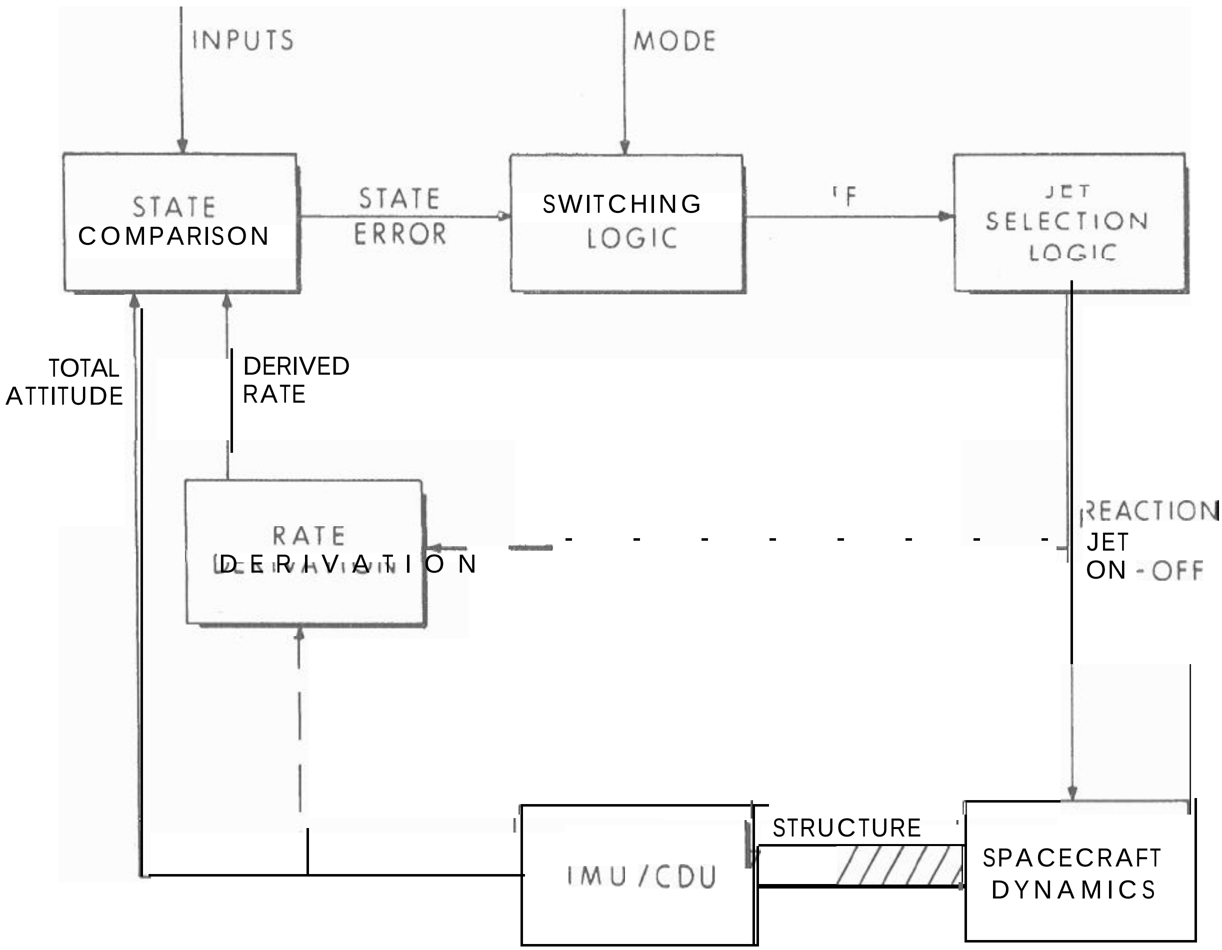


Fig. 2. 1 Block Diagram of Autopilot

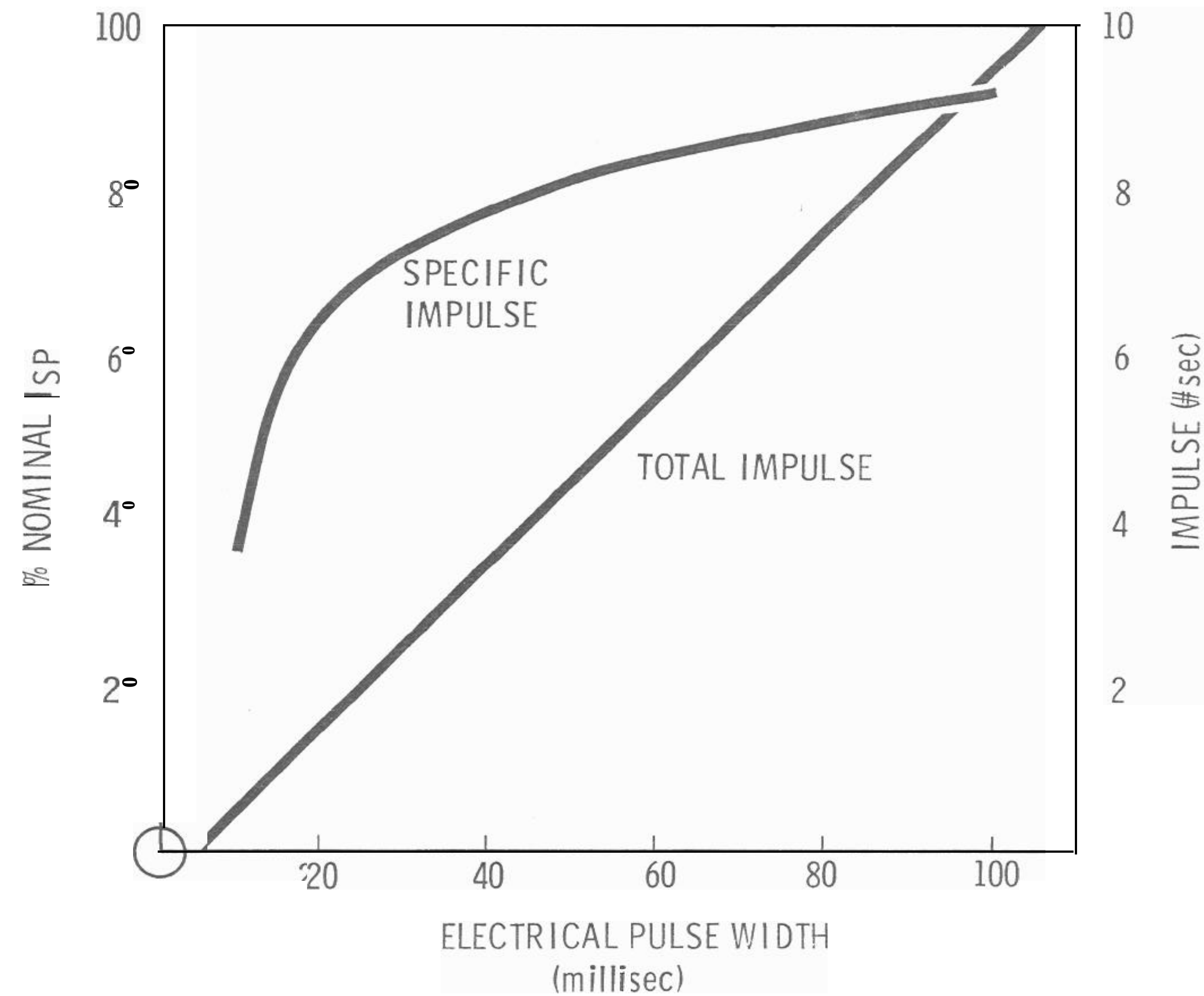


Fig. 2.2 Typical Reaction Jet Performance

2.1 Space craft Model

The parts of the GN&C equipment and the spacecraft important in the design and analysis of the free-fall reaction control system are shown in Fig. 2. 3. The manual controls for mode selection, vehicle rotations and translations, and minimum impulse firings interface with the computer by means of 21 input discretes as shown. Spacecraft attitude information is derived from the Inertial Measuring Unit as coarse and fine resolver analog signals dependent on the three gimbal (Euler) angles.

These signals are converted into total angles by the CDU's and fed in incremental form to the computer. CDU moding discretes are used to synchronize the angle counters in the CDU and AGC. The outputs from the Autopilot program are the 16 discretes controlling the reaction jets which close the loop through the spacecraft dynamics and the IMU-CDU's. Analog attitude error signals are generated by the Digital Analog Converters to drive displays (also used in the GN&C - SIVB interface).

In designing the system a model for the reaction jets has been obtained from figures given in North American Aviation G&C data book entry NAA-S-34. It has been assumed that total impulse is a linear function of thruster electrical “on” time, and that specific impulse is the sum of two exponential terms with electrical “on” time as the independent variable. For the spacecraft dynamics, inertia data were taken from NAA data book entry NAA-S-5 Addendum I (8.12.65). For the S/C dynamic model the complete Euler equations of motion are used in simulations. Data from vehicle bending modes were taken from NAA-S-37. The LEM-docked first bending mode was used in the design model. The attenuation of other modes by the rate filter is very high. The model used for zero-g propellant slosh was taken from NAA-S-22, supplemented by NAA-S-73. Since no model is given for rotational excitation of the slosh, this has been treated as a disturbing torque.

The IMU has been treated as a perfect attitude transducer. Since the bandwidth of the stabilization loops is so high compared with the frequencies that can be sampled by the AGC, this is a very reasonable approximation. The CDU's have been treated as a perfect quantizer, although in practice there is also hysteresis present.

2.2 Computer Design

Certain special features were included in the design of the Block II AGC to enable the digital autopilot functions to be carried out. Those additions which are important to Free-Fall Reaction Control are considered below.

2. 2. 1 Channels

Input channels are provided for "SM separate", "SIVB separate"; rotation controls:*, translation controls*, “Hold”, “Free”, "G&C control"; minimum impulse*, "LEM attached". The asterisks* above indicate those manual controls which cause an interrupt in the AGC. Once made, this interrupt can no longer be obtained until a trap is reset (see below).

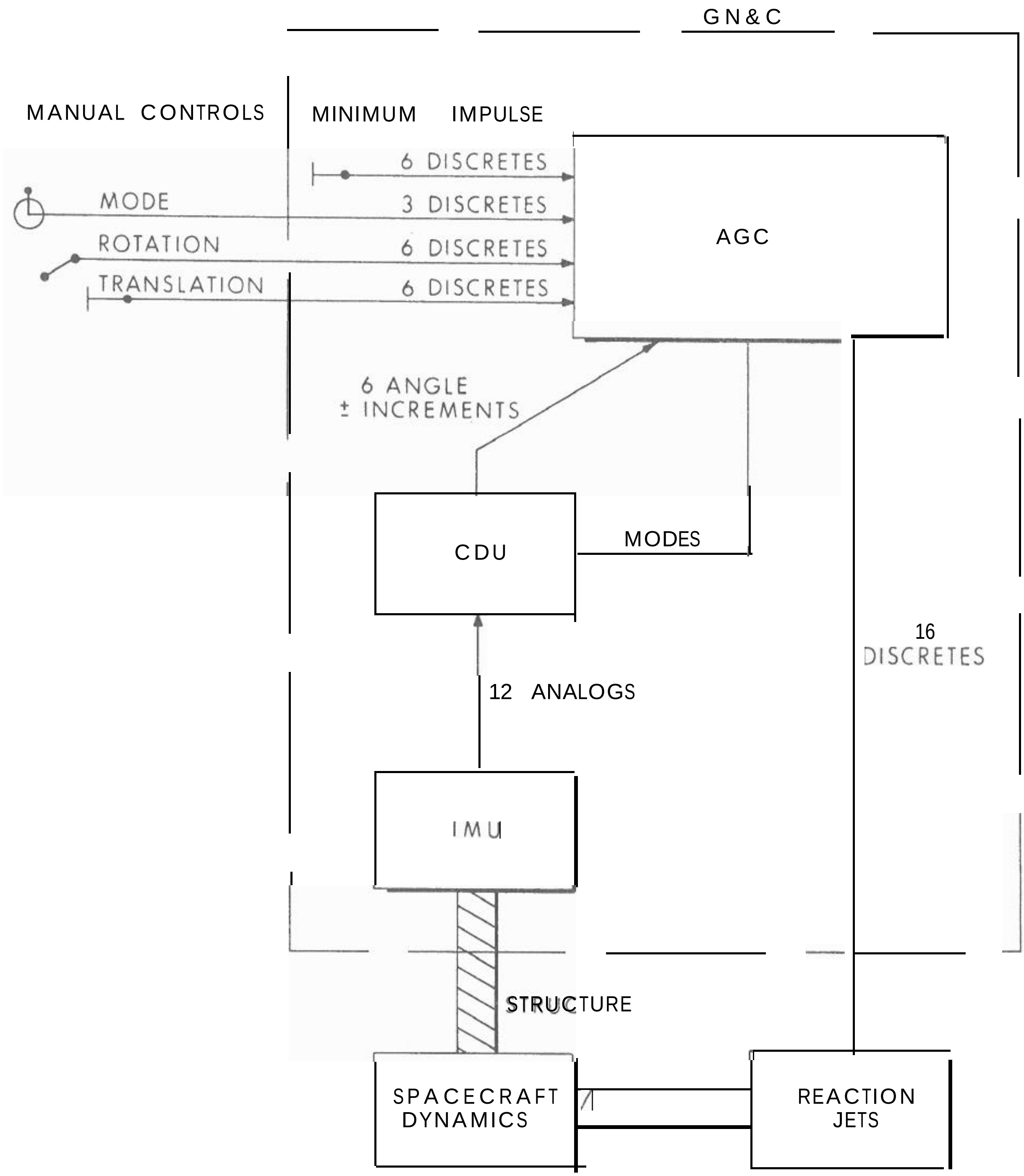


Fig. 2. 3 Model used in the Design

Output channels are provided for pitch and yaw Reaction Control jets; roll control jets; relay codes* for "Auto", Hold", Free" displays; reset manual control interrupt trap, enable T6 interrupt, (see below); CDU modes; CDU-DAC activity bits.

2.2. 2 Counters

Counter T5 is provided for timing the iteration rate of the free-fall Reaction Control and other digital autopilot programs. The counter may be set to any number, and the counter is incremented at 100 pps and causes an interrupt (T5 RUPT) on counter overflow.

Counter T6 is provided for timing the length of firings of the Reaction Control jets. The counter may be set to any number, and the count is diminished at 1600 pps when the T6 activity bit is set. When the counter has reached state zero, an interrupt occurs on the next pulse (T6 RUPT) and the activity bit is knocked down.

In addition to the above modifications, the erasable and fixed memories were expanded, the operation code set augmented, and some codes revised to decrease operation time. In general, these changes were made to speed operation of the real time programs, such as the digital autopilot.

2.3 Program Layout

In order to give the broad picture of the operation of the reaction control program we will look at the major building blocks of the program which will be discussed individually later. The program is initiated by the occurrence of a T5 interrupt (T5 will be initiated by a fresh start, and in any case would occur within 164 seconds). The first action is to look at the GN&C spacecraft control input bit. If no spacecraft control is required, T5 is reset to sample again in one second and no further action is taken. If spacecraft control is required, then the mission phase is examined to see which variety of digital autopilot is wanted. The following description is for the case where Reaction Control is required (see Fig, 2.4).

Free-fall Reaction Control programs are initiated by the T5 interrupt as described above. The first thing that is done is to reset the T5 counter to cause another interrupt in 100 msec, thus establishing the iteration cycle of the main program. Every tenth iteration a program to compute the gimbal-to-body transformation matrix is called under executive. The IMU gimbal rates are derived and transformed to body axes. Then, if rotation commands are present, they are processed in a manner appropriate to the mode, and a flag is set indicating to the attitude hold logic that a new attitude is being set, to be called for on resumption of attitude hold.

*Now deleted.

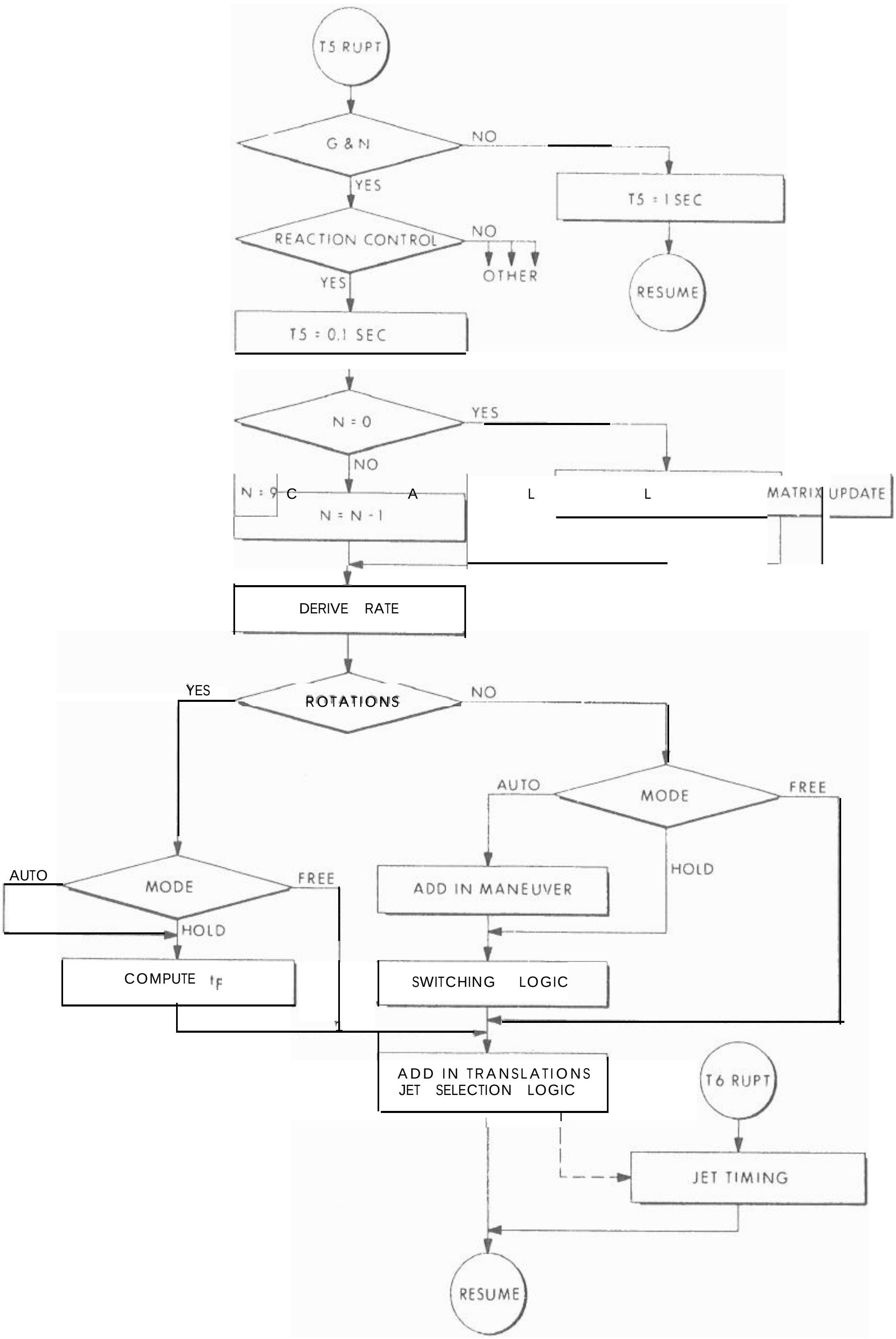


Fig. 2.4 Flow Chart of Program

Translations are then examined and the Jet Selection Logic done. The times of firing are sorted for use in the T6 counter. Finally, the AGC is allowed to resume its previous activity.

Alternatively, if there were no rotation hand controller demands, the modes are examined, and if in “Auto”, automatic maneuvers are processed; if in “Hold”, a reference attitude is held; if in “Free”, the minimum impulse controller inputs are examined and the jet selection logic made as before, firing times computed, and other AGC activity resumed.

SECTION 3

DETAIL DESIGN

3.1 Rate Derivation

Derivation of the body rates of the spacecraft from the gimbal angles of the Inertial Measurement Unit is complicated by two factors.

a) Quantization of the angles, 40 arc sec.
b) Body bending modes of the vehicle.

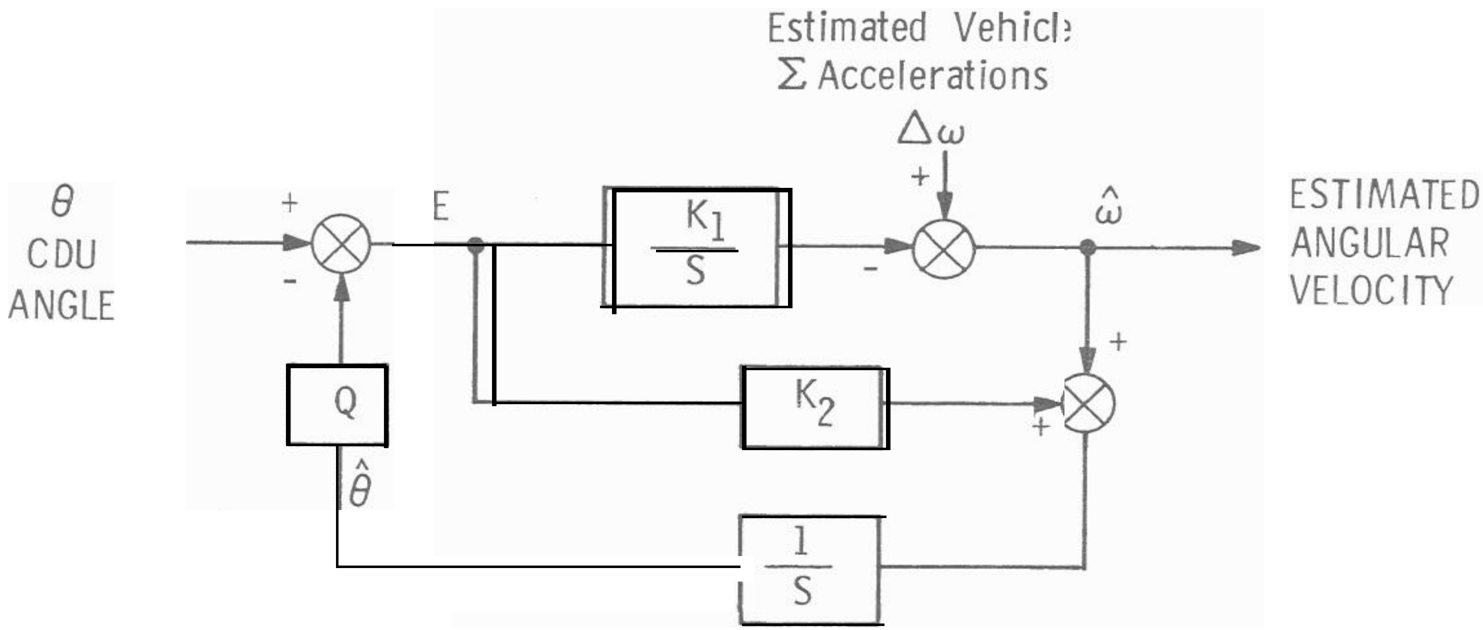
The requirement for rate measurement accuracy is such that the reaction jets can be fired to reduce the body rate to a minimum impulse limit cycle within the dead-band. In the case of the LEM-docked configuration, this limit cycle rate can be as low as 10 microradians/sec, ie., much less than the earth's rotational rate (73 microrad/sec), in the presence of much higher body bending rates.

In order to overcome the quantization problem, an estimating filter has been used to measure rates about the' gimbal axes. The filter is second-order, with a quantizer in the feedback, so that in the steady state there is no following error, and thus no disturbance of the rate estimate (see Fig. 3. 1).

In order to overcome the body bending problem, the characteristics of the filter are shaped so as to attenuate the maximum expected bending to less than 10 microradians per second. The worst case is assumed to be that resulting from a prolonged firing of a pair of reaction jets.

The resulting filter has a serious time delay in responding to change of input rate. To improve the response, estimates of vehicle angular accelerations are used to update the velocity estimates. The estimated accelerations are derived from the number of jets firing, and the times of firing of the jets.

It will be noted that there is some formal similarity between the filter used here and the recursive estimating filter used in the LEM digital autopilot, although the objective there is to measure disturbing torque accurately, rather than to estimate rate accurately.



EQUIVALENT TRANSFER FUNCTION:

$$\frac{\hat{\omega}(S)}{\hat{\theta}(S)} = \frac{S}{S^2/K_1 + K_2/K_1\, S + 1}$$

$$K_1 = \omega_n^2$$
$$K_2 = 2\zeta\,\omega_n$$

DIFFERENCE EQUATIONS:

$$\underline{\hat{\omega}} = \hat{\omega}_{-1} \pm K_1 T E_{-1} \pm \Delta\omega$$

$$\underline{\hat{\theta}} = \hat{\theta}_{-1} \pm K_2 T E_{-1} \pm T\hat{\omega}_{-1}$$

Fig. 3.1 Filter Block Diagram

3.2 Manual Controls

The manual control input bits are sampled every iteration of the T5 interrupt, ie., every 0.1 sec, except for the G&N control bit, which is only sampled every second when it has been turned off (as explained in subsection 2. 3). The minimum impulse controller is only sampled when in the "Free" mode. As a result of the sampling, manual controls will be quantized in time, but so fast it will not be apparent to the operator.

3. 2. 1 Free Mode

In the "Free" mode both the rotation and translation controllers will result in vehicle acceleration. The combination of rotations and translations is discussed in subsection 3. 3, Jet Selection Logic. The minimum impulse controller fires a single pulse (14 msec) of rotation for each occurrence of the discrete, that is, the controller must be returned to the central position giving a zero discrete before another pulse can be fired. Single-jet operation is obtained by the use of simulated fails as described in subsection 3. 3.

3. 2. 2 Auto and Hold Modes

In these modes the rotation hand controller is mechanized to give a fixed vehicle rate on each axis. This rate may be chosen from a discrete set, The desired rate is compared with the actual rate as derived by the filter, and, if the difference exceeds some threshold, the reaction jets are fired for a length of time to reduce the rate error. Due to uncertainties in the thrusters and vehicle inertias, the times are computed for the minimum possible inertias in that mission phase. As explained before, a flag is set by a manual rotation to indicate a new desired attitude on return to the attitude hold mode.

3. 3 Logic

3. 3. 1 Jet Selection Logic

For the purpose of jet selection we can divide the jets into three groups as shown in Tables 3. 1, 3. 2, and 3. 3. Each group's rotational effect on the vehicle is essentially independent (ignoring c. g. displacement effects in the presence of quad fails or single-jet operation).

In the case of the ROLL jets, only one pair of quads is used to perform rotations, in order to save propellants. These roll rotation quads use the same logic as pitch and yaw rotation quads. The remaining pair, the "roll" translation quads, are used only for the appropriate translation when there is no prejudice to rotations being performed by the other pair.

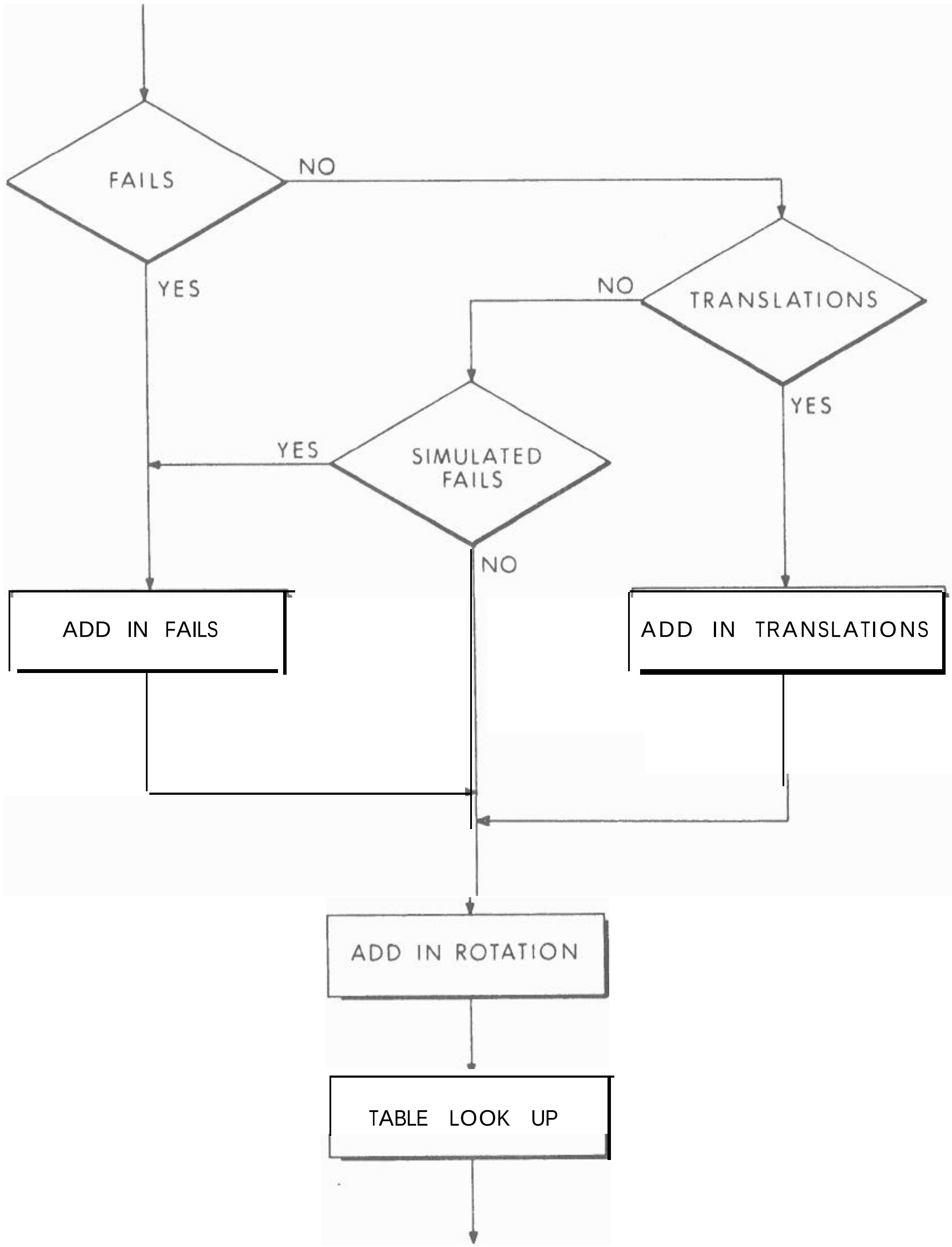


Fig. 3.2 Logic for Rotation Quads

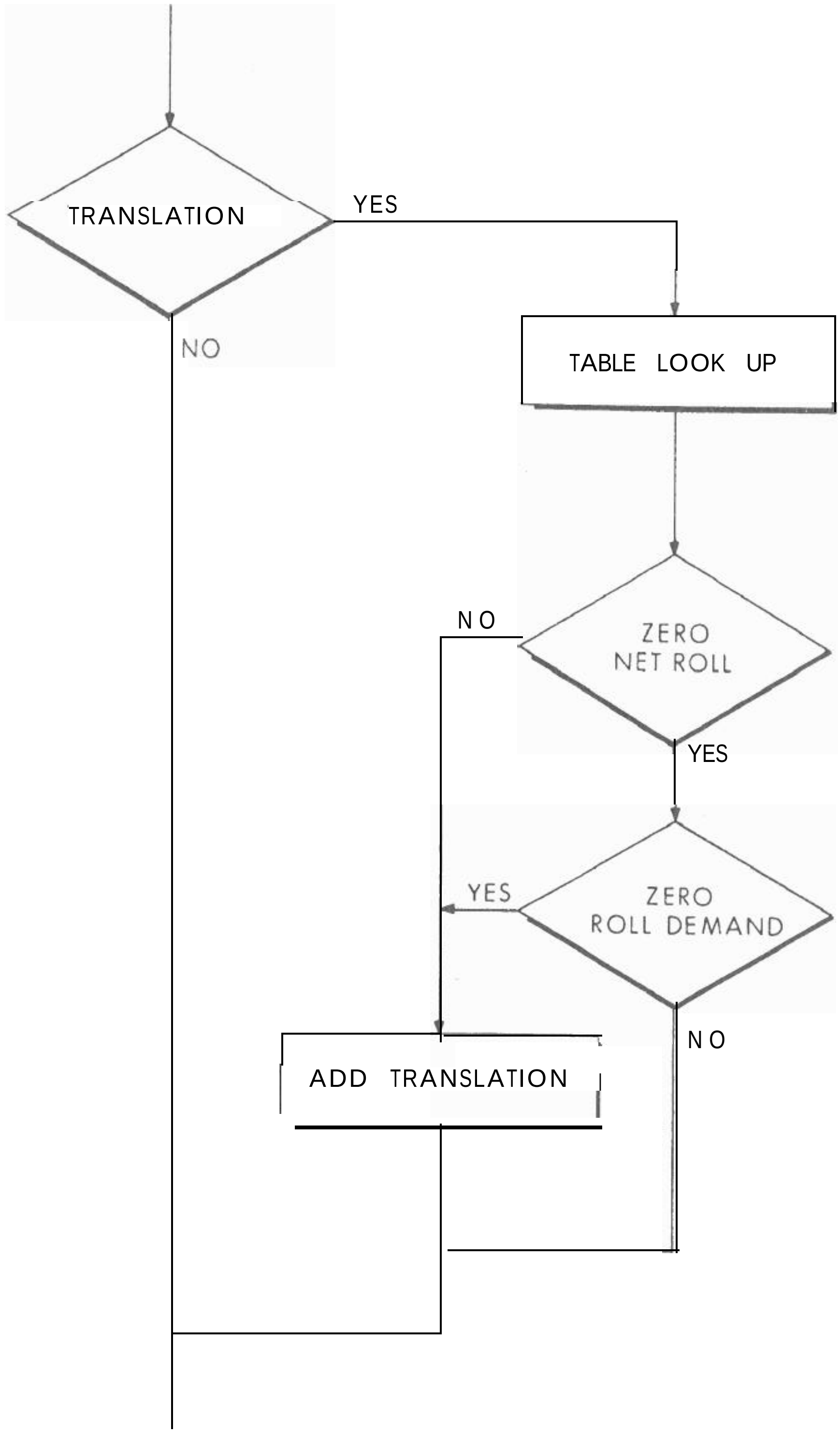


Fig. 3. 3 Logic for “Roll” Translation Quads

Table 3.1

Pitch Jets also Control ± X Translation

Quad	Jet	Pitch	Translation
A	3	+	-x
C	1	+	+x
A	2		+x
c	4		-x

Table 3.2

Yaw Jets also Control ± X Translation

Quad	Jet	Yaw	Translation
B	7	+	-X
D	5	+	+X
B	6		+x
D	8		-X

Table 3.3a

A-C Jets Controlling ± Y Translation

Quad	Jet	Roll	Translation
A	13	+	+Y
C	15	+	-Y
A	16		-Y
C	14		+Y

Table 3.3b

B-D Jets Controlling ± Z Translation

Quad	Jet	Roll	Translation
B	9	+	+z
D	11	+	-Z
B	12		-Z
D	10		+z

3. 3. 1 Pitch, Yaw, Roll Rotation Logic

If any quad of a rotation pair fails, it is impractical to attempt translations, since the resulting rotation torque will immediately cause the control system to ask for an opposite torque. Looking in Table 3. 1, for example, we will see that the result will be zero net translation. Thus the first step of the logic (see Fig. 3. 2) is to ignore translations in the presence of reported fails. Where there is a fail, there is no attempt to fire jets on the failed quad, but an increase is made in the firing time of the jet in the opposite quad to allow for the reduced torque.

In order to allow single-jet operation in attitude hold, the program also allows for simulated quad fails; however, for the reasons stated above, these must be discounted in the presence of translations. The final part of the logic is done by table look-up. The table is in two sections; rotations with translations, and rotations with failures. The logic generates the appropriate pointer for the table look-up. The pitch rotation selections are illustrated in Table 3. 4.

Table 3. 4
Pitch Rotation Select

Item	Pitch	X-XLN	Quad Fail	Jets	Torque
1	0	0			0
2	+	0		1, 3	2
3		0		2, 4	-2
4	0	+		1, 2	0
5	+	+		1	1
6	-	+		2	-1
7	0	-		3, 4	0
8	+	-		3	1
9	-	-		4	-1
10	0		A		0
11	+		A	1	+1
12	-		A	4	-1
13	0		C		0
14	+		C	3	+1
15			C	2	-1

The torque part of the table is used to correct firing times and to provide correct acceleration estimates to the rate filter which was described previously. The tables for Yaw, A-C quad Roll and B-D quad Roll are essentially the same as the pitch table, 3.3.2 "Roll" Translation Quad Logic

If the unused roll quad pair has a translation demand (± Y on A-C or ± Z on B-D) then we look up in a table such as Table 3.5 the appropriate jets to fire to allowing for failures.

Table 3.5

A-C Translation Only

Item	Translation	Quad Fail	Jets	Torque
1	0			0
2	+Y		13, 14	0
3	-Y		15, 16	0
4	0	A		0
5	+Y	A	14	-1
6	-Y	A	15	+1
7	0	C		0
8	+Y	C	13	+1
9	-Y	C	16	-1

As can be seen, in cases of failure, we introduce a roll torque and, thus, we must not make the translation if a previous roll demand is nullified. The logic for this is shown in Fig. 3.3.

3.4 Jet Timing

The jet selection logic generates two words for each axis of control. "Word one" contains the rotation jets with translation, and "word two" contains the translation only for use when the rotation has finished. The commanded rotation firing time was computed based on a single-jet firing. Now that we know how many jets are to be fired, we can compute the firing time (see Fig. 3.4). An extra 14 millisecond is added to the firing time to ensure the firing of at least a minimum impulse and the reversal of rate in the limit cycle.

If the firing time is less than 0.1 sec, the feedback acceleration time for the Rate Derivation is set equal to the command, plus any effect due to unbalanced translation, and the rotation command is zeroed,

Otherwise, the firing time is zeroed, and the "word one" jets used throughout this interval by setting "word two" equal to "word one". The acceleration feedback time is in this case is simply the number of jets scaled by the sampling interval of 0.1 sec.

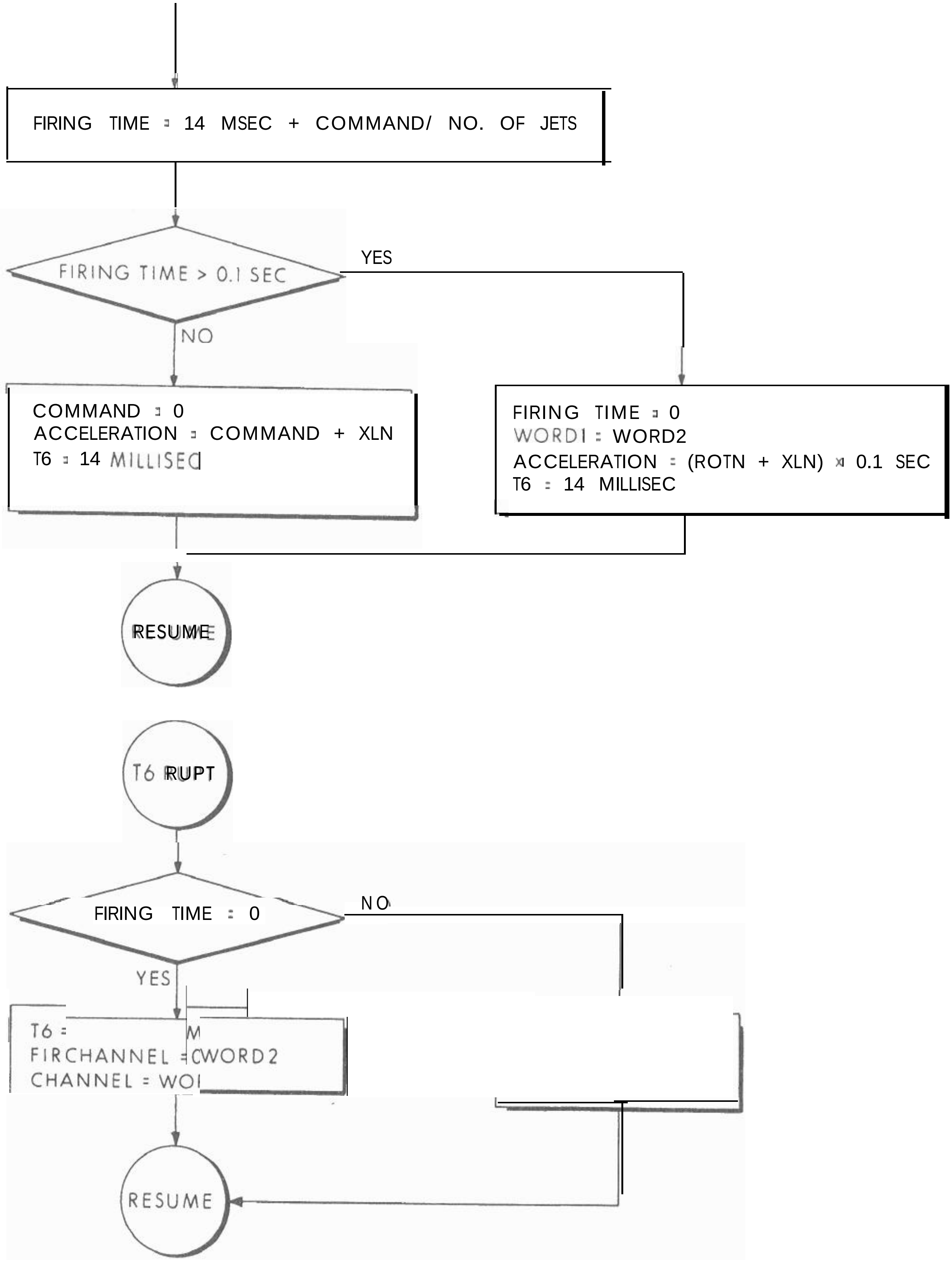


Fig. 3.4 Jet Timing Logic

If the firing time is not zero, the T6 counter is set up to interrupt after that time, the jet control channel is loaded with "word one", and the firing time zeroed. On the occurence of the T6 interrupt "word two" replaces "word one" in the channel.

In practice the T6 counter is shared between three channels. When there is more than one firing to be timed, the shortest is loaded into T6 first. Subsequently, the counter is loaded with the differences between times. However, the words are changed at the appropriate time as before.

3.5 Attitude Hold and Rate Limiting

The attitude hold and rate limiting functions of the autopilot are performed in the same part of the computer logic. The inputs to the logic are the vehicle attitude error and the body rates. The attitude errors are assumed to be small and are equal to the difference between vehicle attitude and the desired vehicle attitude resolved into the three body axes, Rate has been derived in vehicle axes. We will consider the logic used in only one channel since the others are identical.

The dynamics of the switching logic can be best understood by interpretation in the phase plane as shown in Fig. 3. 5. The abscissa is the vehicle angular error; the ordinate is the vehicle body rate. Paths of zero acceleration are horizontal lines and paths of constant acceleration are parabolas.

In the attitude hold mode we can consider what happens to the vehicle when its starts in the state A. The velocity drives the state along the error axis until it reaches the decision line at B. The logic then fires an impulse on the reaction control jets to attain the desired rate corresponding to that attitude error (zero in this case). Due to tolerances in the torque produced by the jets and to uncertainties in the vehicle moment of inertia, the impulse fired is actually short (20%) of the desired (plus a minimum impulse). Thus the state drops to C, and now moves more slowly towards the decision line. During this time the Rate Derivation filter drives the rate estimate from the predicted value towards the actual value. Now the state hits the decision line again at D and fires the jets again to bring it to E. The effect of the extra minimum impulse is to reverse the velocity and the vehicle now goes into a minimum impulse limit cycle EFGHE, etc.

In the rate limiting mode the vehicle is at some high rate condition such as at point K. Then the jets fire to bring the operating state within the rate limiting deadband at L. Then the state moves until it reaches the attitude deadband, say at A for example, and the vehicle moves into an attitude hold mode,

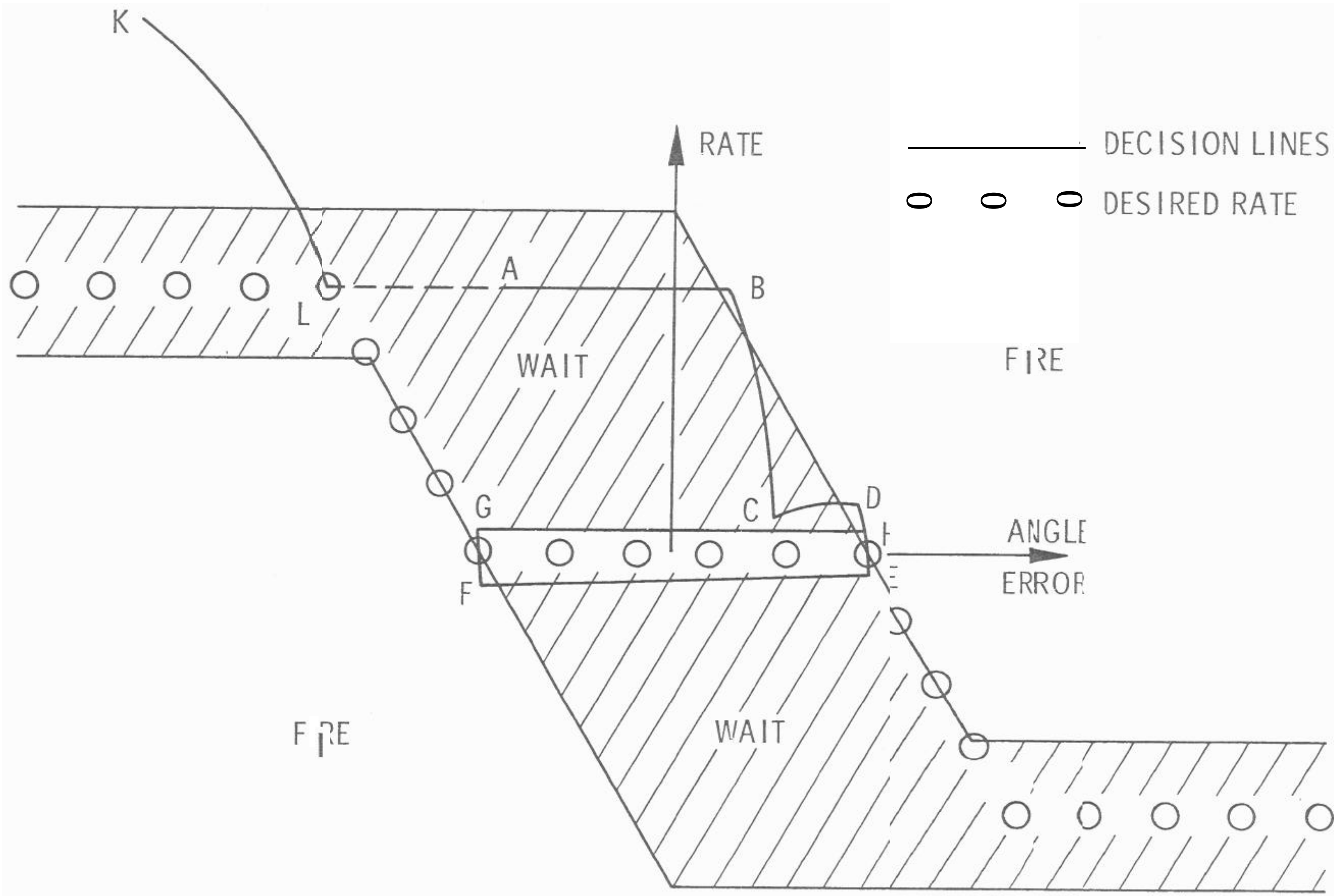


Fig. 3.5 Phase Switching Logic

3.6 Automatic Maneuver Computation

It has been shown in MIT/IL Report E-1832 that it is convenient to perform attitude maneuvers by simultaneous maneuvers in three axes. However, under certain circumstances this leads to maneuvers through the area of gimbal lock warning on the Inertial Measurement Unit. In this event the maneuver is split into two parts such that the gimbal lock area is avoided.

The inputs to the attitude maneuver computation are the three gimbal angles desired as the final orientation of the spacecraft with respect to the IMU stable member. In order to convert from spacecraft axes to control axes, all outer gimbal angles are modified by the 7.25-degree reaction jet offset. The rotational rate to be used by the maneuver is also required.

From the present gimbal angles and the required final gimbal angles a rotation matrix is computed which describes the transformation from the initial to the final attitude. From this matrix the eigenvector giving the direction of required rotation is derived by partitioning the matrix into its symmetric and antisymmetric components. The equivalent angle of maneuver is obtained, and using the magnitude of maneuver rate, the time of maneuver is computed, In addition, the rotation vector of the maneuver rate is resolved into the three control axes of the spacecraft,

The inputs from the maneuver computation program to the Reaction Control System being described are these three spacecraft rates, and the time of maneuver. Where gimbal lock is to be avoided, the two component maneuvers are sent to the control system separately.

3.7 Attitude Maneuvers Execution

Attitude maneuvers are implemented by exactly the same logic as that used in attitude hold, except that the inputs to the logic are error rate and attitude error. The difference is that the rate ordinate is the difference between actual rate and desired rate. Thus, throughout a maneuver, the vehicle attitude is being held about a desired attitude moving from the initial vehicle position towards the final position.

Due to the requirement for high vehicle maneuver rates which has become apparent, it has been found that the above scheme has one disadvantage; the initial error rate in the phase plane can be great enough to prevent the initial firing of the jets from driving the operating point into the attitude deadzone. As a result, the vehicle rate limits on the opposite side before returning to the deadzone. This results in opposite firing of the jets, which is wasteful of propellants. In order to overcome this problem, a bias has been introduced into the attitude error, The bias is the ratio of the square of the maneuver rate to twice the torque-to-inertia ratio. With this bias present the vehicle state always moves into the correct rate limiting deadband before moving into the attitude hold deadzone.

SECTION 4

DESIGN EVALUATION

The evaluation of the design of the attitude hold logic, and the automatic maneuver features of the digital autopilot, was performed by simulations on the M-H 1800 computer. A complete three-dimensional model of the autopilot and spacecraft was set up using the MAC compiler language. The vehicle simulation included a model for propellant utilization, complete Euler equations of motion, and dynamics of the principal bending mode (LEM attached). Subsequently, the AGC Program Logic itself has been checked out on the Apollo Digital Simulation (the mandatory software verification simulation). Typical phase plane plots from the digital simulation are shown in Fig. 4. 1. The plots show the execution of a small maneuver, and the attitude hold at the completion of the maneuver. The effect of cross coupling in the vehicle is to cause the spacecraft to limit cycle (bounce) on one end of the attitude deadzone.

Analog computer simulations have been made using a single-axis model in order to study the effects of parameter variations and of fuel slosh. No destabilizing effects have been found in these studies, which are reported below.

4.1 Propellant Utilization

A theoretical study of propellant utilization was made in MIT/IL Report E-1832. In order to get a complete paper model to compare with simulations, we must add in an extra term to allow for the limit cycle existing in the deadband due to the control action. Thus, the expected propellant utilization is due to three terms. The first of these is the propellant to start and stop the maneuver, P_1, given by

$$P_1 = \sum_i \frac{2J_i \omega_i}{I_{SP} L}$$

where J_i = inertia of the vehicle about the i axis

ω_i = vehicle maneuver rate about the i axis (absolute value)

I_{SP} = specific impulse

L = lever arm

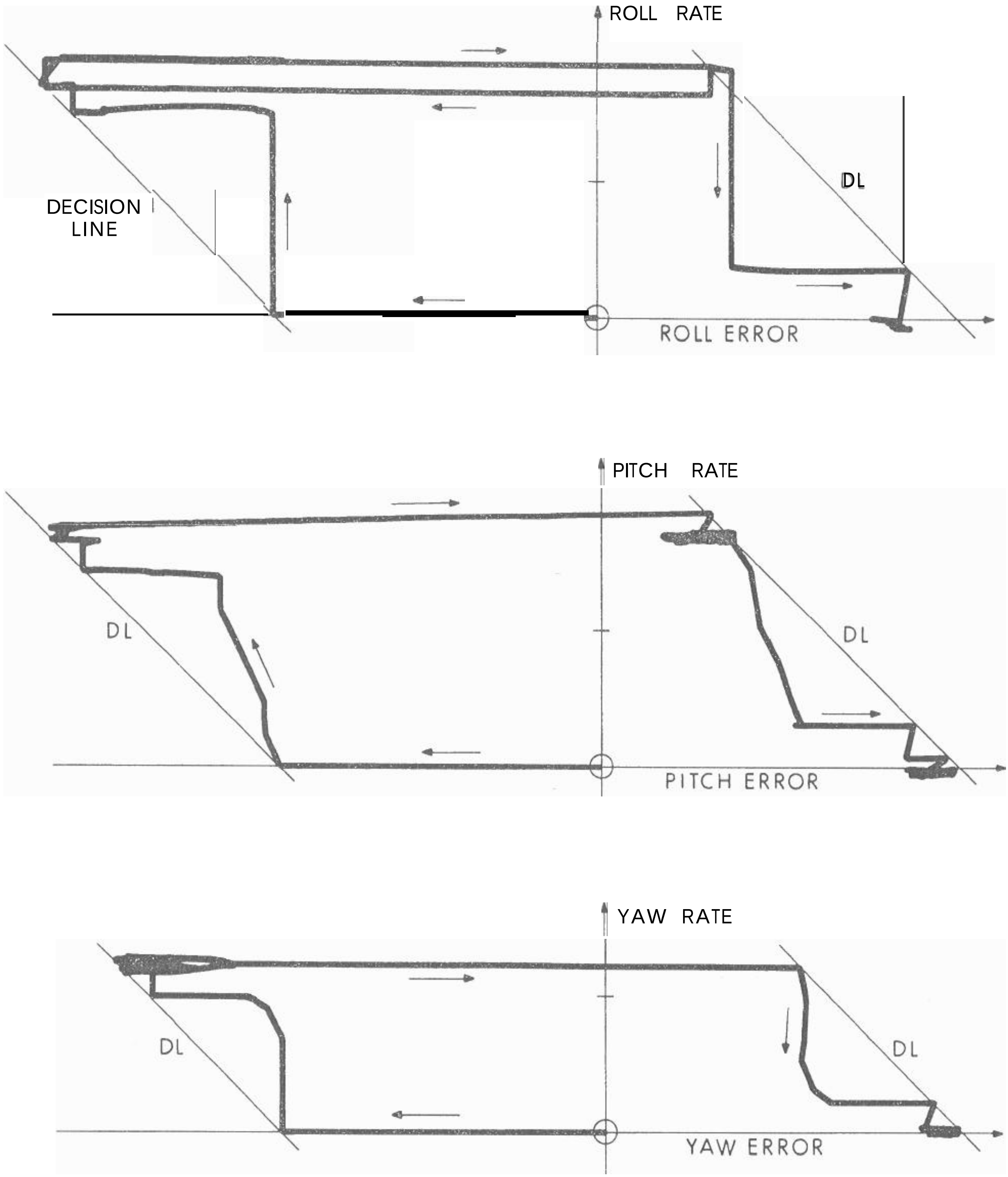


Fig. 4.1 Phase Planes for 3-Axis Maneuver

The second term is the propellant used to cancel cross coupling, P_2, given by

$$P_2 = \frac{(J_Y - J_X)}{I_{SP} \quad L} \quad \omega_X \; (\omega_Y + \omega_Z) \; t_M$$

where t_M = time of maneuver, and

$J_Y = J_Z$.

The third term is the propellant used to maintain the limit cycle, P_3, given by

$$P_3 = \frac{I_M^2 \; L \; t_M}{2 I_{SP} \; AD} \sum_i \frac{1}{J_i}$$

where I_M = minimum impulse of jet pair

AD= angular deadband

The total propellant used in then given by

$$P = P_1 + P_2 + P_3$$

In making the comparison between this figure and simulations, we have assumed that, for computing P_1, I_{SP} = 275 because the firings are quite long, and for computing P_2 and P_3 that I_{SP} = 155 since the firings are mostly minimum impulses. Table 4. 1 gives the comparison between the above theortical model figures for propellant utilization, and the figures from simulations. The maneuvers are given as commanded gimbal angles corresponding to pitch, yaw and roll which were initially zeroed. Typical inertias have been chosen for translunar, lunar orbit, and transearth phases of the mission. The figures agree well despite the crude theoretical model used; the discrepancies are due to the real assymetric vehicle inertias and to the true variation of specific impulse.

4.2 Disturbing Torques

The "Zero G" propellant slosh data in NAA-S-73 give the disturbing torques expected from residual slosh as a result of main SPS engine firings, and also as a result of an attitude maneuver. Because no relationship is given for the phase of the oscillations with respect to vehicle motion, the only analysis that has been done is to estimate propellants used in cancelling these torques. As there are a wide variety of possible SPS firings and a wide range of figures is given for the slosh from attitude maneuvers, only order-of-magnitude estimates have been made by integrating the absolute value of the torque with respect to time.

Table 4.1

Simulated and Theoretical Propellant Consumption

Maneuver	Units	A	B	C	D	E
Pitch, IGA	deg.	20	0	20	0	180
Yaw, MGA	deg.	20	0	0	20	0
Roll, OGA	deg.	20	20	0	0	20
Translunar 0.2 deg/sec						
Simulated	lbs.	2.87	0.24	2.00	2.02	2.59
Theoretical	lbs.	2.81	0.22	1.18	1.78	2.29
Lunar Orbit with LEM 0.5 deg/sec						
Simulated	lbs.	5.25	0.51	3.70	3.46	4.77
Theoretical	lbs.	4.93	0.36	3.22	3.22	3.85
Lunar Orbit CSM only 0.5 deg/sec						
Simulated	lbs.	1.09	0.40	0.80	0.68	
Theoretical	lbs.	0.88	0.23	0.57	0.57	1.07
Transearth 0.2 deg/sec						
Simulated	lbs.	0.57	0.23	0.33	0.30	
Theoretical	lbs.	0.56	0.23	0.34	0.34	1.56

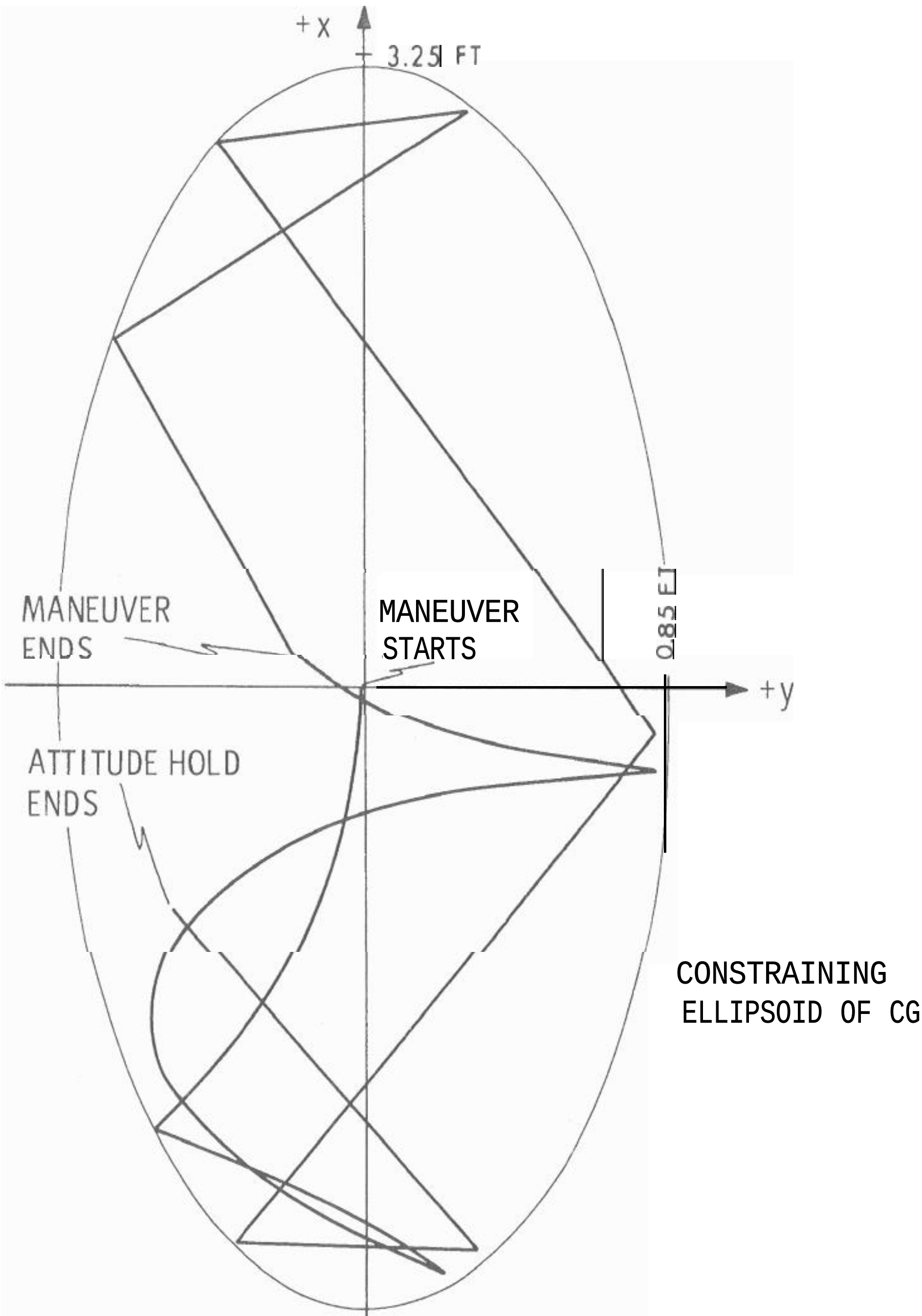
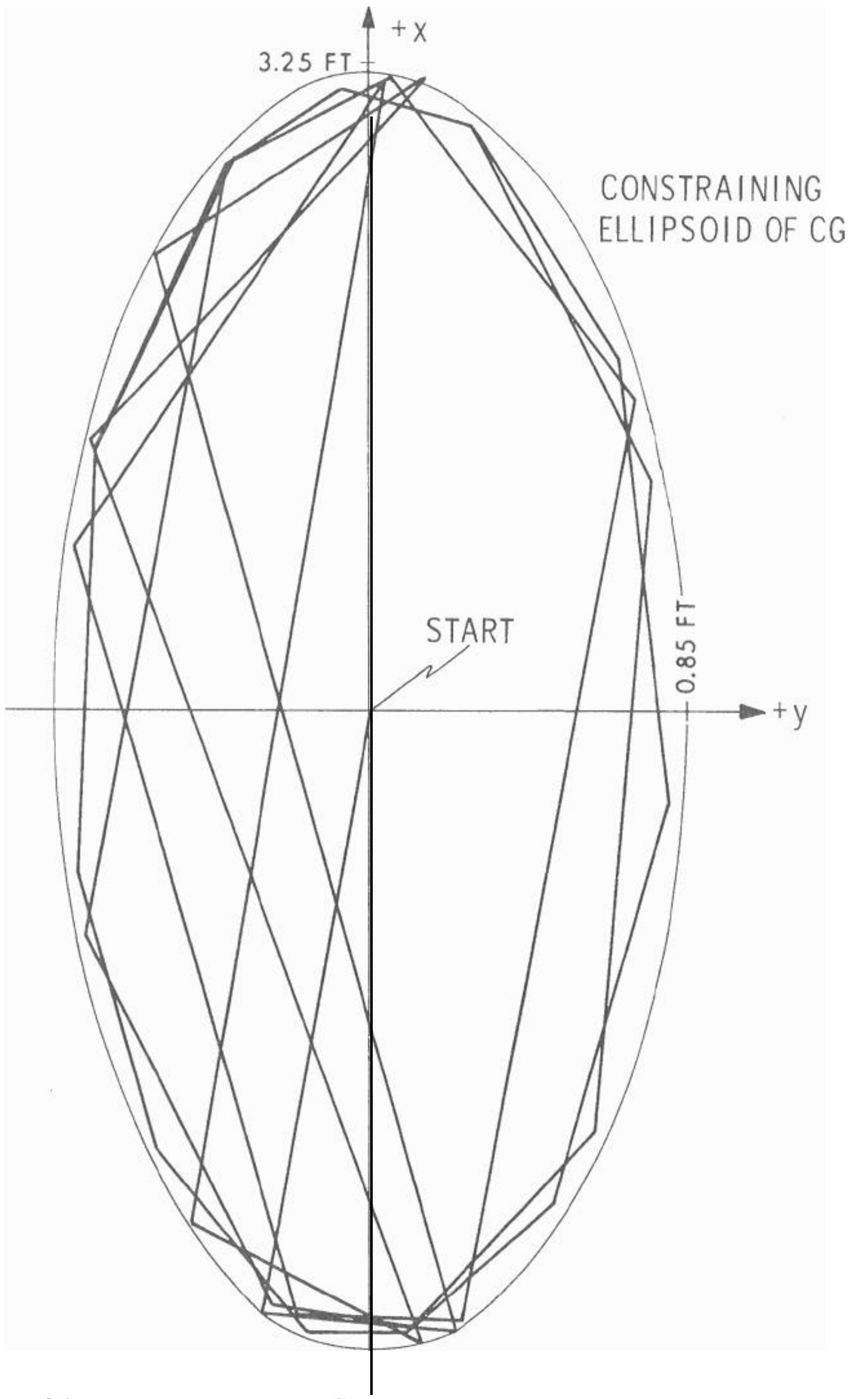


Fig. 4. 2 Propellant Motion Due to Attitude Maneuver



$\dot{x}_d = -0.015$ FT/SEC
$\dot{y}_d = -0.15$ FT/SEC
IC ON FUEL

Fig. 4. 3 Propellant Motion in Attitude Hold

SECTION 5

CONCLUSIONS

The digital autopilot for the reaction control system has been designed for, and verified with, the current spacecraft model contained in the Block II G&N Data Book. While the design is adaptable to a wide range of spacecraft and RCS jet parameters, any significant changes to the model of the spacecraft may require changes to the program, and extensive re-verification.

Requirements for simultaneous translation and rotation accelerations, and single-jet operation in attitude hold, have considerably increased the complexity of the program. However, simultaneous translation and rotation use less propellants than where rotations interrupt the translation. Similarly, the single-jet operation uses less propellants then two-jet operation in attitude hold.

Propellant motion in free-fall requires further study. It is recommended that the “elliptical” model be further developed, and entered into the Data Book. In any case, this model should be incorporated in the Apollo Digital Simulator. Then the reaction control can be verified using vehicle dynamics coupled to propellant motion. If such a model for dynamics were incorporated into the Data Book, the Apollo Digital Simulator could be easily used in verification runs for mission programs.

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MIT/IL Report E-1832 "Attitude Maneuver Optimization to Conserve Reaction Control Propellants" R. Crisp & D. Keene, August 1965.

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