#### SECTION 3

## BLOCK II GNC SYSTEM DESCRIPTION

## 3.1 GNC Hardware Configuration

System 204 will be GNC system for Mission 278. It is a normal Block II system except for the Signal Conditioning Assembly which processes FLIGHT QUALIFICATION measurements as well as OPERATIONAL measurements. The system is comprised of the following major assemblies:

Inertial Measurement Unit (IMU)
PIPA Electronics Assembly (PEA)
Coupling Data Unit (CDU)
Navigation Base (NB)
Optical Unit (including Space Sextant-SXT and Scanning Telescope - SCT)
Command Module Computer (CMC)
Two (2) Display Keyboard Assemblies (DSKY)
Display and Control Group
Power and Servo Assembly (PSA)
Signal Conditioning Assembly (SCA-FQ)
G&N Interconnect Harness Group
Eyepiece Storage Unit (ESU)

## 3.2 GNC/Spacecraft Signal Interfaces

# 3.2.1 Interface Control Documents (ICD's)

Below are presented the Interface Control Documents (ICD's) which are pertinent to definition and understanding of the electrical/functional interfaces between the CG&N system and the spacecraft, the launch vehicle and the astronauts. Additional existing ICD's pertaining to mechanical interfaces, thermal interfaces, etc. are not listed because they are not considered to be relevant to this document.

| General Inter-   |   |   |   |
|------------------|---|---|---|
| facing Area      | ICD Title   | ICD No.                                     | <u>Description</u>  |
| Electrical Power | G&N Electrical Input Power  | MH-01-01327-216<br>(IRN 0055)               | Total a-c and d-c Power specification from CSM to GNC.  |
| Displays         | Altitude Error  | MH 01-01324-216 (IRN's 1499, 3605)          | Signal interface for following:  a) Pitch Error  b) Roll Error  c) Yaw Error  d) 800-cps reference  |
|                  | Total Attitude Signals  | MH 01-01325-216<br>(IRN's 0823, 3614, 3616) | Signal interface for following 800 cps modulated signals to FDA1 and altitude set resolver; a) Sine AIG b) Cosine AIG c) Sine AMG d) Cosine AMG e) Sine AOG f) Cosine AOG |
|                  | G&N Master<br>Warning and<br>Caution Lamps                              | MH-01-01342-216                             | Electrical interface<br>from S/C Alarm<br>system to GNC dis-<br>plays   |
| Telemetry        | G&N Data Transmission to Operational and Flight Qualification Telemetry |   | Electrical interface<br>for GNC PCM and<br>Flight Recorder<br>measurements.   |
| Steering         | CDU to TVC Servo<br>Amplifiers  | MH-01-01307-216<br>(IRN's 0826, 3604)       | Electric (d-c) interface to SPS for: a) Pitch Command   |

Yaw Command

| General Inter-<br>facing Area | ICD Title                                       | ICD No.                         | Description   |
|-------------------------------|---|---------------------------------|---|
|                               | G&N Attitude<br>Signals to Saturn<br>Guidance   | MH-01-01386-216                 | Electrical (d-c) interface to Saturn Guidance for:  |
|                               | φ····································           |                                 | <ul> <li>a) Roll Attitude Error</li> <li>b) Pitch Attitude</li> <li>Error</li> <li>c) Yaw Attitude Error</li> </ul> |
| Moding                        | Mode Control<br>Signals S/C to<br>ISS Discretes | MH-01-01344-216<br>(IRN's 3601) | Switch closure discrete to ISS: a) IMU Cage b) GNC A/P Control c) S/C Control of Saturn                             |
| CMC                           | CMC Electrical<br>Interfaces-Block II           | MH-01-01380-216<br>(Rev. A)     | Defines the electrical interfaces between 'CMC and CSM. For details see sec. 3.2.2                                  |

# 3.2.2 CMC/CSM Signal Interface

The following section lists the complete interface between the CMC and CSM as described in detail by ICD MH-01-01380-216 Rev. A. Figure 3-1 presents a schematic representation of the interface.

# 3.

| . 2. 2. 1 | CI                      | MC/CSM Interface - Inp  | uts to CMC              |
|-----------|-------------------------|---|-------------------------|
| . 2, 2, 1 | 1) 2) 3) 4) 5) 6)       | LEM Attached SPS Ready Attitude Hold Free Drift Accept Uplink CM/SM Separate S IV B Separate Lift Off | Control Mode and Status |
|           | 9)<br>10)<br>11)<br>12) | Guidance Reference Re Ullage Thrust Present Plus Roll Minus Roll Plus Pitch Minus Pitch               |                         |

Figure 3-1

(To be supplied)

Similar to Figure 3-1

Vol.II

Pulsed altitude encrements 15) Plus Yaw from Gyro Display Coupler 16) Minus Yaw 17) Plus Roll Command Discretes from 18) Minus Roll 19) Plus Pitch Rotational Hand Controller 20) Minus Pitch 21) Plus Yaw 22) Minus Yaw 23) Plus X Command Discretes from 24) Minux X 25) Plus Y 26) Minus Y Translational Hand Controller 27) Plus Z 28) Minus Z 29) Start PCM Downlink 30) Stop 31) Bit Sync 32) Uplink "One" Uplink 33) Uplink "Zero" CMC/SCM Interface - Outputs from CMC 3, 2, 2, 2 1) + Pitch/+X2) - Pitch/+X 3) + Pitch/-X4) - Pitch/-X 5) + Yaw/+X 6) - Yaw/+X7) + Yaw/-X8) - Yaw/-X SCS Reaction Control 9) + Ro 11/+ZSystem Jet-Firing 10) -Ro11/+ZCommands 11) + Roll/-Z12) - Roll/-Z 13) + Roll/+Y14) - Roll/+Y 15) + Roll/-Y16) - Roll/-Y 17) SPS Firing Command To SPS

18) S IV B Fire Cycle

19) S IV B Engine Cutoff

20) Master Clock (1024K)

21) Digital Downlink Data

22) CMC Warning

23) ISS Warning

24) PGNC Caution

To Saturn

Engine

Central Timing Equipment

PCM

Caution & Warning

Subsystem

## 3. 2. 3 GNCS Failure Monitor

The GNCS performance and operational readiness are self-monitored and caution and warning information are displayed to the crew. Two warning (red) lamps are actuated by the GNCS on the Caution/Warning Panel: CMC Warning indicates computer failure; ISS warning indicates failure of the inertial subsystem. Also a GNCS Caution (amber) lamp is actuated to indicate non-critical problems in the system. Further detail regarding the caution items is displayed by means of the DSKY event lamps and the DSKY data registers (in the event of a program alarm). The failure monitor mechanization is shown in Figure 3-2.

## 3.2.3.1 CMC Warning

A CMC warning alarm is generated in the event of CMC power failure, scaler failure of either of two types, restart or counter fail during CMC operate or in response to an alarm test program. As is shown in Figure 3-2 a scaler fail or prime power fail results in immediate alarm indication whereas the other inputs are buffered by a filter so as to prevent momentary transient disturbances which recover from causing a warning alarm. In this subsection the various inputs and conditions associated with CMC warning are defined.

- (a) SCAFAL Occurs if scaler stage 17 (1.28 sec period) fails to produce pulses. This provides a check on the timing for all logic alarms.
- (b) COUNTER FAIL Occurs if counter increments happen too frequently or else fail to happen following an increment request.

  "Too frequently" means continuous counter requests and/or incrementing for from .625 ms to 1.875 ms.
- (c) SCADBL Occurs if the 100 pps scaler stage operates at a pulse rate of 200 pps or more.

Figure 3-2 (To be supplied) Identical to Figure 2 Vol. II except that all Tracker inputs and outputs are deleted. 3-7

- PARITY FAIL Occurs if any accessed word in fixed or erasable memory whose address is octal 10 or greater contains an even number of "ones".
- RUPT LOCK Occurs if interrupt is either too long or too in-(e) frequent. The criterion for "too long" is phase dependent varying from 140 ms to 300 ms. Likewise the criterion for "too infrequent" varies from 140 ms to 300 ms.
- TC TRAP Occurs if too many consecutive TC or TCF instruc-(f) tions are run or TCF instructions are too infrequent. The criterion for "too many" varies from 5 ms to 15 ms duration. The criterion for "too infrequent" varies from 5 ms to 15 ms absence.
- NIGHT WATCHMAN Occurs if the computer should fail to (g) access address 67 within a period whose duration varies from .64 sec to 1.92 sec.
- V FAIL Occurs if the CMC voltages (28, 14, 4) are out of (h) limits. This signal produces STRT1 if it stays on for a period of between 157 and 470 µsec. If the computer is in the STAND-BY mode, an input to the CMC WARNING FILTER is generated simultaneous with STRT1. The following criteria apply for V FAIL:

16V

14 V Supply > 12,5V 4 V Supply > 4.4 V14 V Supply < 28 V Supply < ~22.6V 4 V Supply < 3.65V

STANDBY - This is a signal which turns on RESTART and (i) turns off the switchable +4 and +14 voltage, thus putting the CMC into a low power mode where only the scaler, timing signal, and a few auxiliary signals are operative. STANDBY is initiated by first setting the ENABLE STANDBY outbit (CH13 B11), and then pressing the STANDBY button on the DSKY for a time which varies from .64 sec to 1.92 sec, at the end of which time the STANDBY LIGHT is turned on. (All CMC alarms are inhibited during the standby mode with the exception of CMC WARNING, which can be caused by VOLTAGE FAIL or SCALER FAIL; and TEMPERATURE CAUTION, which can be caused by TEM ALARM.) Normal operation is resumed by pressing the STANDBY button on the DSKY again, time of depression same as above.

(j) RESTART - RESTART occurs at next time 12 following occurrence of any one or more of the parameters shown in Figure 3-2 except OSCILLATOR FAIL.

RESTART occurs immediately and forces time counter to 12 upon occurrence of OSCILLATOR FAIL. (See paragraph "1" below.)

RESTART causes the computer to transfer control to address 4000 as soon as it disappears. It sets a flip-flop which lights the RESTART CAUTION lamp in the DSKY. The Flip-flop is reset either by the ALARM RESET hard wired signal or by the CAUTION RESET outbit CH 11 B 10. ALARM TEST operates the lamp but not the flip-flop.

- WARNING FILTER This circuit is used to operate the CMC (k) WARNING output following repeated or prolonged occurrence of parameters shown in Figure 3-2. All occurrences of these signals are stretched so that no more than one input to the filter is generated in each 160 millisecond period. Approximately six consecutive stretched pulses cause CMC WARNING to turn on for about 5 seconds. Non-consecutive stretched pulses may also cause CMC WARNING after an interval dependent on the frequency of the pulses. The output will not occur if input pulses occur at a frequency of less than 0.9 pps; and the output will remain on if The threshold pulses occur at a frequency of 0.7 pps or more. of the filter resumes its normal level with a time constant of many seconds after the filter has received inputs. An immediate reset of the CMC FAIL due to a WARNING FILTER output is therefore not possible.
- (1) OSCILLATOR FAIL Occurs if the oscillator stops. Has nominal 250-millisecond delay to keep signal present after the oscillator starts. Also occurs when CMC is in STANDBY because of loss of power to front end of circuit. This gives 250-millisecond delay in starting when CMC comes out of STANDBY into OPERATE. Causes immediate restart without waiting for time pulse 12.

#### 3.2.3.2 ISS WARNING

The ISS WARNING signal is the logical "OR" of the following parameters, any one of which will cause an ISS WARNING under the following conditions:

## (a) IMU Fail

- (1) IG Servo Error greater than 2.9 mr for 2 sec
- (2) MG Servo Error greater than 2.9 mr for 2 sec
- (3) OG Servo Error greater than 2.9 mr for 2 sec
- (4) 3200 cps decrease to 50% of normal level
- (5) 800 wheel supply decrease to 50% of normal level
  These parameters are generated in the Inertial Subsystem.
  However, the "FAIL" signal itself is under CMC program control.
  It is ignored by the CMC program when the G&N system is in the
  Coarse Align Mode and during the 5 second interval following
  Coarse Align. During this mode the servo errors normally exceed
  the above criteria.

#### (b) PIPA FAIL

Occurs if no pulses arrive from a PIPA during a 312.5 $\mu$ sec period, or else if both plus and minus pulses occur, or if a "long time" elapses without at least one plus pulse and at least one minus pulse arriving. By "long time" is meant a period of between 1.28 sec and 3.84 sec.

This FAIL signal is generated totally within the CMC and thus is completely under CMC program control. Its generation is enabled by the CMC only during CMC controlled translation or thrusting maneuvers.

#### (c) ISS CDU FAIL (Monitored for each of 3 CDU's)

- (1) CDU fine error in excess of 1.0V rms
- (2) CDU coarse error in excess of 2.5V rms
- (3) READ COUNTER limit cycle in excess of 160 cps
- (4) Cos ( $\theta \phi$ ) below 2.0V
- (5) +14 DC Supply decrease to 50% of normal level

These parameters are generated in the Inertial Subsystem. However, the "FAIL" signal itself is under CMC program control. It is ignored by the CMC program when the G&N System is in the CDU Zero Mode. During this Mode the CDU errors normally exceed the above criteria.

#### 3.2.3.3 GNCS Caution

The GNCS Caution lamp is actuated by the following undesirable but non-critical events:

- (a) CMC Restart during Operation. In the event of Restart during operate a latch is set in the CMC which maintains the GNCS Caution alarm and the RESTART lamp on the DSKY until the latch is reset by program or until the latch is manually reset by ALARM RESET. For further detail see section 3.2.3.4.
- (b) Temperature out of Limits. The CMC receives a signal from the IMU when the stable member temperature is in the range 126.3°F to 134.3°F. In the absence of this signal, the Caution alarm and the TEMP lamp on the DSKY are actuated.
- (c) Gimbal Lock. When the CMC determines that the middle gimbal angle (MGA) of the IMU is greater than 70°, the Caution alarm and the Gimbal Lock lamp on the DSKY are actuated. When MGA exceeds 85° the ISS is downmoded to Coarse Align and the No Attitude lamp on the DSKY is actuated.
- (d) Program Alarm. Under a variety of situations a program alarm is generated. Figure 3-2 illustrates one example, that of PIPA fail when the vehicle is not in a thrusting mode. Under program control the CMC inhibits this alarm for 10 sec after system turn-on. The program alarm actuates the Caution alarm and the Program lamp on the DSKY. For further information see section 3.2.3.4.

#### 3.2.3.4 Restart and Program Alarms

When the Restart or Program Alarm lamps are illuminated on the DSKY, either V 05 N 31 will automatically appear on the DSKY with a key to the source of the alarm displayed in R1 or by astronaut call up from the DSKY this information can be made available. This allows the astronaut to identify and normally correct the alarm condition. The listing of program alarms is included in Section 4.

## 3.3 Digital Autopilot for the CSM

#### 3.3.1 Introduction

There are four distinct digital autopilots employed in the GNC system to control the Block II spacecraft. Two autopilots relate to RCS operation: one during space flight and one during entry. Two autopilots are associated with thrust-vector control (TVC): one with LEM docked and one without LEM docked. (Although attitude command of the Saturn vehicle cannot be strictly classed a GNC autopilot operation, this command utilizes several DAP service programs.) The use of the various autopilots is essentially mutually exclusive i.e., except for brief transitional periods at the start of thrusting, no more than one autopilot is active at any given time. The activation of any A/P and its interface with other programs is governed by the major mission program in process.

## 3.3.2 Spacecraft Characteristics

The various DAP's described in this section were originally designed predicated upon spacecraft and engine characteristics officially transmitted before 1 January 1966. Unofficial information during the Spring of 1966 indicated that the bending mode frequencies and the slosh damping coefficients were lower than the official values; furthermore, there is considerable uncertainty as to the exact values of these parameters, which significantly influence the compensation design.

of the DAP's to accommodate this new information is underway, but the new design will not be completed in time to be serviceable in Mission 278. Instead, the DAP's designed as optimal on the basis of the previously official information have been detuned for use on Mission 278.

All-digital and hybrid simulations have demonstrated satisfactory performance for the optimal DAP's during all phases of the mission (all representative mass, inertia, fuel-configuration situations tested). Less complete simulations indicate that the detuned DAP's provide stable, if degraded, performance which will be satisfactory for the 278 earth-orbital mission.

The spacecraft and engine data utilized for this detuned design are included in Section 6.

## 3.3.3 General Moding Information

The various DAP's described in this section control spacecraft operations after S IV B separation. DAP operations are sensitive to the following modes, status and control inputs:

GNC A/P Control
Attitude Hold
Free Drift
S IV B Separate
CM/SM Separate
LEM Attached
Translational Hand Controller Inputs
Rotational Hand Controller Inputs
Minimum Impulse Inputs (integral to GNC)
Fine Align (NO ATTITUDE), (integral to GNC)

Two additional discretes, SPS Ready and Ullage Thrust Present, also affect GNC operational during DAP modes, but do not directly interface the DAP.

The GNC does not initiate any DAP operations unless it receives the GNC A/P Control discrete.

The significance of the hand controllers in the various modes is outlined in Table 3.1.

The values of rate limit during AUTO or HOLD for various phases of the mission are presented below:

CSM Abort 5°/sec.

CSM Lunar Orbit 0.5°/sec.

CSM Transearth

CSM-LEM Translunar

Economy Mode 0.05°/sec.

CM Pitch/Yaw 5°/sec.

CM Roll 5°/sec.

None of these CSM or CSM-LEM phases are pertinent to GNC performance on Mission 278. For this earth orbited mission maneuvers will be limited to 0.5 deg/sec or 4 deg/sec.

\* When the GNC takes over Saturn control (only attitude control for Mission 278), the CDU's provide rate commands to the Saturn Guidance System.

Talle I CSM Dry La Author at Moder

| CAP                                   |               |  |                        | m A M  | 130  |  |
|---------------------------------------|---------------|--|------------------------|--|--|--|
| Mode                                  | DISCRETES     | シント  | TVCM RC Sa 40.05 70.05 | 40.05  | 70.055   | Saturu   |
| AUTO                                  | GAUC A/PCOUL. | (3) (3)  | (S)                    | (5)  | <u>(M)</u> 3   |  |
|                                       | Free          | Man to the same  |                        | 2  |  |  |
|                                       | GUC A/P Cont. | (3)  | 50                     | <u> </u>   | Page and   | es es la constant de l'arres   |
| HOLD                                  | 313           | notice and a state of the state | <b>6</b>               | 5  | estimo   | mpt.   |
|                                       | GUCAPPORT     | E  | 5                      | (4)  | 3  |  |
| 1                                     | Te Car        |  | 6                      |  | 6-Acres and 60   | 1  |
|                                       | Frae          |  | Fellowalle             | A CONTRACTOR OF THE PROPERTY O |  | STATE OF THE PARTY |
|                                       | SICCUS Sast.  |  |                        | 1  |  | 3  |
| S S S S S S S S S S S S S S S S S S S |               |  |                        |  | June manufacture and the second secon |  |

| 8) Wells attended of the of access command    | 10) fellows 124C in rall only - note. | 11) fellens RAC as in hold mole; | referent to AUTO upon relience |                               |                                    |
|---|---------------------------------------|----------------------------------|--------------------------------|-------------------------------|------------------------------------|
| 1) wanny SPS<br>2) molubing TVC with RCS with | 3) automatic atomic                   | 5) fellewer RHC and THE ex       | attled hall often release      | 6) belowe RHC as note command | 1) fellews THC as accel. commissed |

The S IV B separate discrete is required to initiate CSM DAP operations; the CM/SM separate discrete is required to initiate CM Entry DAP operations; the LEM attached discrete is used by the RCS and TVC DAP's to select the proper DAP compensation sections. During thrusting operations, no DAP will recognize a change in status of these discretes.

The Fine Align ISS mode is required for all DAP operations. Loss of Fine Align will rapidly result in an ISS Warning and a Program Caution and thrusting termination. The Astronaut must manually switch to a backup mode.

## 3.3.4 TVC Digital Autopilots

## 3.3.4.1 Operational Requirements

#### 3.3.4.1.1 General

During the thrusting period of the CSM, the Apollo Guidance Computer (AGC) generates vehicle attitude commands and compares these commands with the IMU-measured vehicle attitude to generate a set of attitude errors expressed as rotations about the pitch, yaw, and roll axes of vehicle. These attitude errors are generated in three steps:

- 1. Tranform the CMC-generated increments in attitude command into equivalent angular increments of the IMU gimbals.
- 2. Accumulate the IMU angle command increments and compare with the measured IMU angles.
- 3. Transform the angular errors of (2) into pitch, yaw and roll errors about the body axes.

CMC computer treats the control of each of the three attitude errors as a separate problem. Three independent control signals are generated by the computer to control the vehicle about the pitch, yaw and roll axes in accordance with the attitude error about these axes.

Implementation of roll control is achieved by reaction jets whose firing is controlled by a separate roll autopilot program in the CMC computer.

Control of the pitch and yaw attitude of the vehicle is achieved by pitch and yaw deflections of the gimballed rocket engine of the CSM. These engine deflections are produced by pitch and yaw gimbal servos whose command signals are generated by independent pitch and yaw autopilot programs in the CMC computer.

#### 3. 3. 4. 1. 2 Roll Autopilot

The control of the roll angle of the vehicle is not critical in achieving overall guidance objectives. A simple, on-off autopilot program to operate

the vehicle roll jets has been developed to control the roll angle to within  $\pm$  5° of a constant, preset value.

## 3. 3. 4. 1. 3 Pitch and Yaw Autopilot

The overriding consideration in the design of the pitch and yaw autopilot programs of the CSM and CSM-LEM vehicles is the ability of these autopilots to perform in conjunction with the external guidance loop. The primary factor in evaluating this performance is the value of the velocity pointing error achieved at thrust-cutoff. This pointing error must be confined within limits determined by orbital requirements for the various thrusting periods of the Apollo flights. The combined guidance-autopilot system must also limit transient excursions of the vehicle attitude and the thrust vector as required to minimize propellent usage and to provide a proper dynamical environment for the Astronaut. In addition, the autopilot must be designed with a view to minimizing the activity of the gimbal servos so as to maintain the somewhat critical servo clutch wear at a low level.

The evaluation of the performance of the combined pitch-yaw autopilot and guidance system must consider the transient disturbances to this system caused by the following conditions existing at the instant of thrust initiation:

- 1. A mis-alignment of the thrust vector to the vehicle center of gravity at 1 degree.\*
- 2. Initial turning rates of the vehicle about its pitch, yaw and roll axes of 0.1 degrees per second.
- 3. Initial displacements of the fuel and oxidizer slosh masses in the CSM tanks of 0.6 times the tank radius.

The autopilot-guidance system evaluation must also include that particular combination of vehicle and engine gimbal servo constants, within the specified tolerances, which gives the worst case performance characteristics of digital autopilot.

Initial evaluation and parametric studies of the pitch-yaw autopilot programs have been carried out independently of the guidance system interaction with major considerations being given to achieving a stable autopilot with satisfactory insensitivity to specified tolerances in vehicle and engine servo characteristics. During this separate evaluation the requirements of combined guidance-autopilot system have been taken into account by

\* Prior to thrust initiation the AGC computer supplies bias signals to the engine gimbal servos which orient the thrust vector through the estimated position of the vehicle c.g. Errors in the c.g. estimation and in the alignment of the engine thrust vector relative to its mechanical null position can result in as much as a one degree misalignment of the thrust vector.

ICE POL Communication placing lower limits on the speed of response and the attitude-error excursions of the autopilot system.

The dynamical characteristics of the CSM-LEM docked vehicle are sufficiently different from those of the CSM alone as to require the development of separate autopilot programs tailored to the characteristics of each vehicle. The three principle differences in the two vehicles:

- 1. The bending mode frequencies of CSM-LEM vehicle are as low as approximately 1 cps, while the bending mode frequencies of the CSM alone are more than a decade higher.
- 2. There are substantial differences in the moment arm of the gimballed engine to the center of gravity and in the vehicle moment of inertial between the CSM and CSM-LEM vehicles. As a result of these differences a given deflection of the gimballed engine of the CSM alone produces roughly 4 times the angular acceleration which would be produced by the same deflection in the CSM -LEM vehicle.
- 3. The fuel and oxidizer slosh picture in the CSM-LEM vehicle is different from the CSM alone, because of the additional slosh masses in the LEM tanks and the effects of the increase mass of the over-all vehicle.

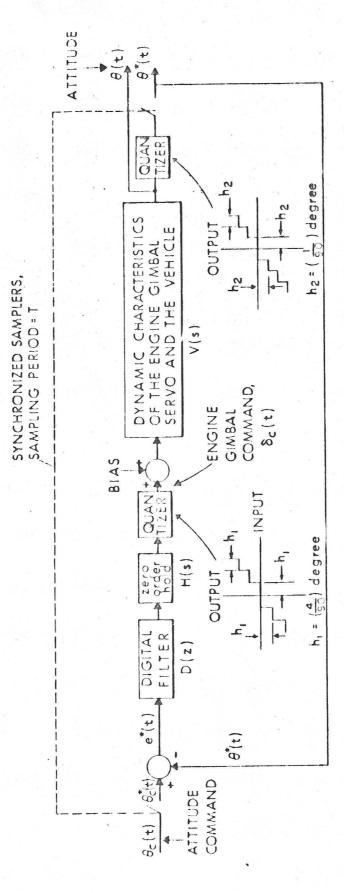
The problem of minimizing the excitation of slosh modes and preventing any de-stabilization of these modes is of comparable importance in both CSM and CSM-LEM vehicles. The problem of avoiding the excitation of the bending modes and preventing their potentially de-stabilizating influence is primarily a problem in CSM-LEM autopilot design.

## 3. 3. 4. 2 CSM Pitch-Yaw Autopilot

The design of the pitch-yaw autopilot for the CSM vehicle was carried out initially without the complication of the interaction of the guidance loop. The only restriction imposed on this design by consideration of guidance loop interaction was the specification that the main cross-over frequency of the autopilot should be higher than a guidance loop cross-over frequency of 0.1 radians per second.

Figure 3-3 shows a schematic block diagram of one channel (pitch or yaw) of a pitch-yaw autopilot which can serve as a basis for describing the autopilots of both the CSM and CSM-LEM vehicles. For the sake of simplification this diagram neglects the various transformations involved

(Chin File) (a) mask Alle (Alle (Chin) (a) mask (Alle (Chin) (a) mask (Alle (Chin))



A

Simplified Schematic Block Diagram of One Channel of the CSM Pitch-Yaw Autopilot 

Figure 3-3

endikielikananalahikaner

in generating the pitch and yaw attitude errors, and treats these errors as if they were generated by the comparison of command and feedback signals expressed in body coordinates. The role of the CMC computer in this autopilot is to perform a dynamic filtering operation on the sampled attitude error,  $e^*(t)$ , as required to generate a suitable sampled command gimbal servo,  $\int_{c}^{A}(t)$ . A digital-to-analog converter changes the gimbal servo command signal to an analog voltage which it holds for each sample period. The output of this digital-to-analog converter is quantized with increments of .0237 degrees. Both the command signal and the feedback signal are also quantized with increments of .0111 degrees.

The operation of the CSM autopilot is best explained by first going through a description of the method by which the autopilot was designed.

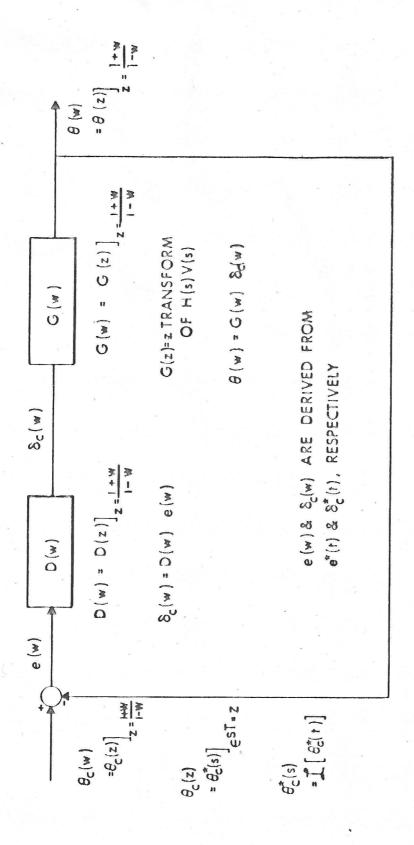
The design of the autopilot was carried out in terms of a linearized model of the vehicle, engine gimbal servo, and autopilot filter which neglected the signal quantization and other non-linearities in the engine-vehicle system. Using this linearized model, the characteristics of the digital filter and the engine-vehicle system could be described in terms of pulse transfer functions, which could be conveniently combined to describe the overall system. The use of these pulse transfer functions to describe the system is illustrated in Figure 3-4. Here, the transfer function of the vehicle plus engine gimbal servo plus digital-to-analog converter is represented by the pulse transform G\*(s). This transform may be alternatively expressed as the z transform,

or in terms of the w transform,

Similarly the digital filter transfer function is represented by D\*(s), D(z), and D(w). The open-loop transfer function of the autopilot system is then given by the product of the two component transfer functions, e.g., D(w)G(w). The overall transfer function of the system is given by

$$\frac{\Theta(\omega)}{\Theta_{c}(\omega)} = \frac{D(\omega) G(\omega)}{1 + D(\omega)G(\omega)}$$

The autopilot design can be described and analyzed conveniently either in terms of the z transforms or in terms of the w transforms. The z transforms offer the advantage that the D(z) can be translated directly into computer program equations. The w transforms on the other hand offer the advantage



0

5 3 20

that the characteristics of the engine-vehicle system and any variations in these characteristics can be readily expressed in terms of the poles and zeros of G(w).

It can be shown that  $G(w)|_{w=ju} = G(ju)$  when plotted versus u for compared represents the real frequency response of the sampled engine-vehicle system over the real frequency range  $0 < \omega < \frac{\omega_s}{2}$  That is

Real-frequency response function

$$G^*(J\omega) = G(J\omega)|_{\omega=\tan\frac{\omega\tau}{2}}$$

for

where

T = sampling period, seconds

$$W_S = \frac{2 \pi r}{T} = \text{sampling frequency, radians/second}$$

Similarly the substitution of u = tan - transforms

D(ju) G(ju) and 
$$\frac{D(ju) G(ju)}{1 + D(ju) G(ju)}$$
 into

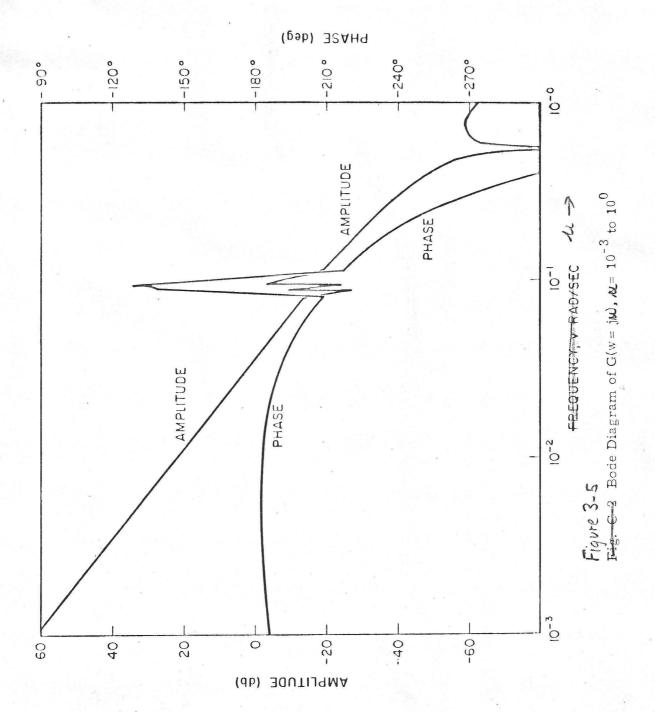
open loop and closed loop real-frequency response functions. It should be noted that for most of the frequencies of interest in the autopilot design the relationship between u and w may be approximated by

$$u \approx {wT \over 2}$$
 [where  $u < 2$ ]

The conventional, open-loop frequency-response methods of design which have been developed for continuous-data systems may be applied also to the sampled-data autopilot, using D(ju)G(ju).

Both z-transform and w-transform methods have been used to develop the autopilot design for the CSM. The w-transform method will be described here.

Figure 3.5 shows the magnitude and phase characteristics of the uncompensated CSM vehicle with a 26% propellent loading. The plots of Figure 35 shows the system to be unstable, since the phase shift is more than 180 degrees at the cross-over frequency, u=0.04. This cross-over frequency in u corresponds to a real cross-over frequency of  $\omega=2$ . Even if



the phase shift at this main cross-over frequency were less than 180 degrees, the system would still be unstable because of the magnitude and phase contours at the slosh frequency, u=.06, which corresponds to a real frequency of  $\omega=3$ .

The open loop magnitude and phase plots in Figure 3-6 show the effect of adding compensation D(w) to the system of Figure 3-5. The system is seen to be stabilized with the phase shift less than 180 degrees at both the main cross-over frequency and the slosh frequency. A good measure of the stability is the difference between the phase shift where a 0-db cross-over occurs and the stability-limit value of 180 degrees. This difference is called the "phase margin." An adequate phase margin of roughly 40 degrees occurs at the main cross-over frequency in Figure 3-6 and a phase margin in excess of 30 degrees is seen to occur at the slosh frequency. The main cross-over frequency in Figure 3-6 is u = .03 which corresponds to a real frequency of  $\omega = 1.5$ . Figure 3-7 shows the close-loop magnitude and phase characteristics of the compensated system. The compensation function D(w) is given by:

$$D(w) = 0.288 \left[ \frac{w + 0.01}{w + 0.2} \right]$$

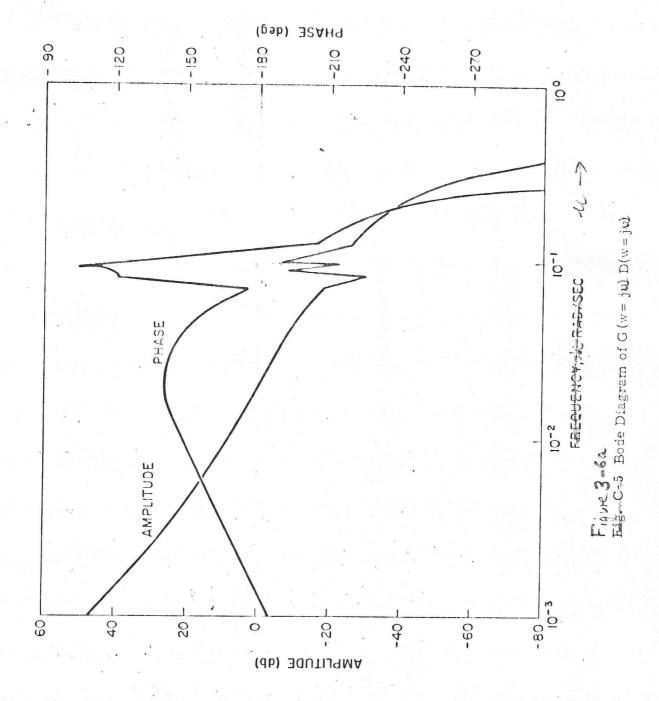
which has the corresponding z transform,

$$D(z) = 0.243 \left[ \frac{2 - 0.98}{2 - 0.67} \right]$$

This filter function appears to give satisfactory response characteristics both with and without the guidance loop for the conditions of Mission 278. The step response of 26% full CSM employing this autopilot function is shown in Figure 3-7.

As mentioned previously, one of the major obstacles to be overcome by the digital autopilot is the effect of an initial error in the alignment of the thrust vector through the vehicle center of gravity. The response of the CSM system to a 1 degree initial offset of the thrust vector is shown in Figure 3-8. The attitude of the vehicle is observed to rise to a constant level which is approximately given by  $1/K_{SS}$  where  $K_{SS}$  is the steady-state gain of the digital filter, as given by:

$$K_{ss} = D(w)_{w=0} = D(z)_{z=1}$$

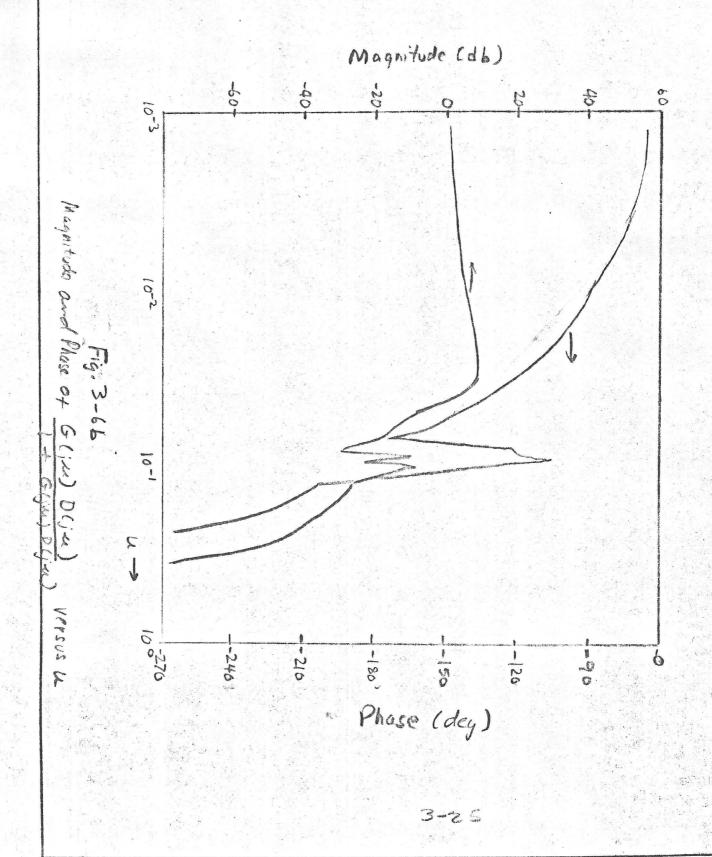


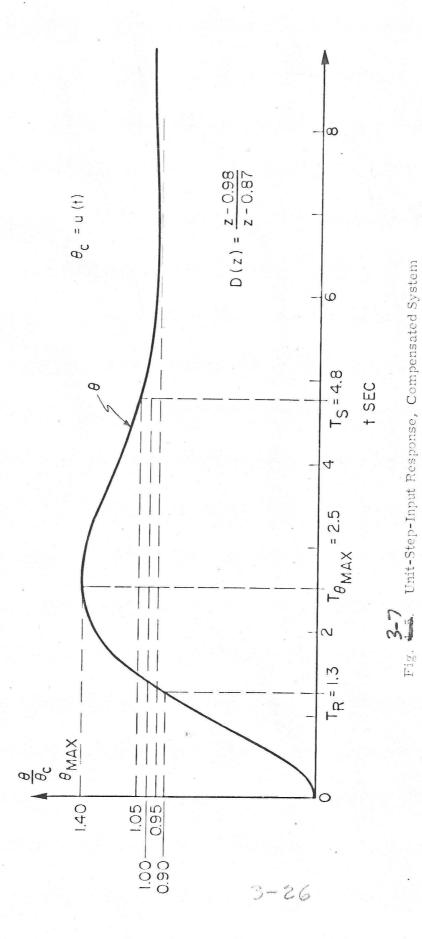
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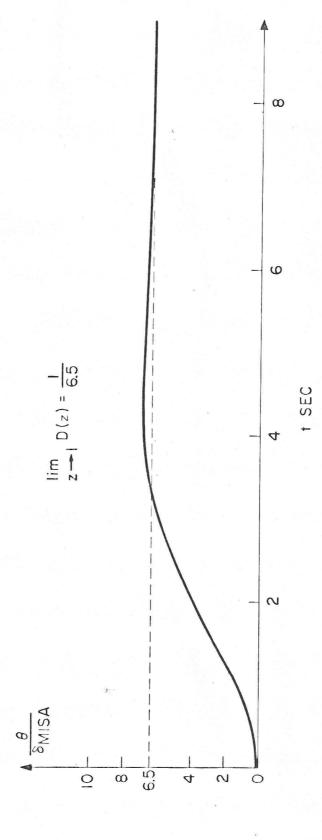
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3-8 Fig. 4-6. Unit Thrust-Vector Misalignment Response

Thus, in the steady state, the value of the attitude error must be such that the resulting output of the digital filer produces an engine deflection which is equal and opposite to the initial thrust misalignment. It is important to note that this steady state attitude error produced by thrust misalignment does not in general mean a steady state error in the velocity pointing of the guided vehicle, because the external guidance loop will act to readjust the attitude command signal to the autopilot to compensate for this attitude error. However, the pointing error can be appreciable during the short thrusting periods when there is insufficient time for the external guidance loop to react to the thrust vector misalignment before thrust cutoff. Investigations of the CSM autopilot with a closed guidance loop for short thrusting periods indicate that the resulting pointing errors are acceptable for Apollo missions. A discussion of the performance of the combined autopilot-guidance-system is presented in a later section of this report.

#### 3.3.4.3 CSM-LEM Pitch-Yaw Autopilot

One of the problems of generating a final design for the pitch-yaw autopilot of the CSM and CSM-LEM vehicles has been the uncertain nature of the data on vehicle characteristics and engine gimbal servo characteristics. This uncertainty in data has had a more severe effect on the design of the CSM-LEM autopilot than it has had on the design of the autopilot of the CSM alone. Of very great concern is the uncertainty in the bending mode characteristics of the CSM-LEM vehicle. Since no experimental programs to measure these characteristics are planned, it will be necessary to rely on theoretical estimates for Mission 278. A final set of theoretical estimates of the bending mode characteristics is not yet available as of the writing of this report. Another data problem which has been partially alleviated recently is that of the sloshing of fuel and oxidizer in the CSM tanks. Recent scaledmodel tests by NAA of low-amplitude sloshing have yielded drastic reductions in the slosh damping ratios below the original theoretical estimates. Thus the damping ratio originally estimated at 0.5% for all tanks has now been reduced to as low as 0.07% under some conditions.

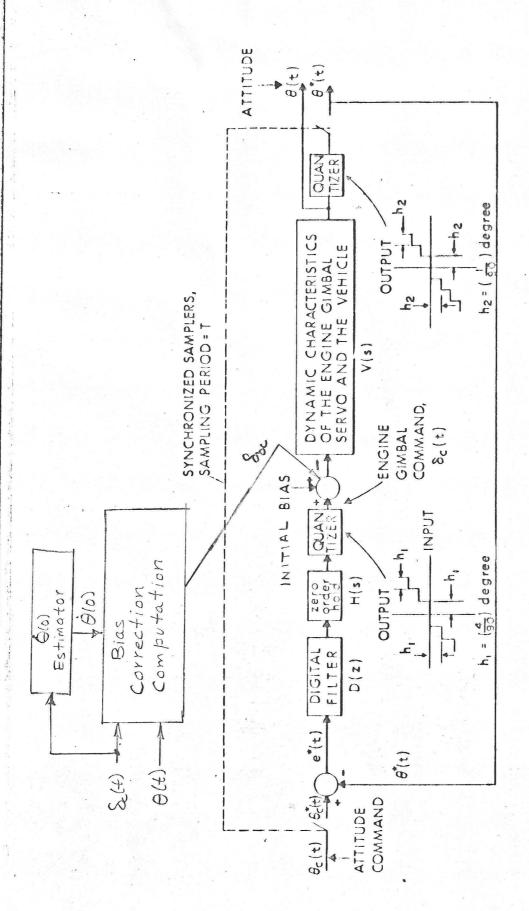
The original CSM-LEM autopilot filter described in R-503 was designed with a view to allowing a large uncertainty in the first bending mode frequency from 1 to 4 cps and to making the system insensitive to the characteristics of higher order bending modes. This design policy was implemented by the use of a fourth order attenuation factor in D(w) that produced a rapid dropoff in the open loop gain above the slosh frequency. This gain drop-off was

3-20

sufficient to attenuate to a negligible level any resonance peaks occurring above the first bending mode. The attenuation factor was selected to produce a rapidly increasing phase lag, which rotated the gain vector approximately -260 degrees from its -135 degree angle ( 45degree lead) at the main crossover frequency of 0.1 cps to a phase angle of roughly -395 degrees or -35 degrees at 1 cps. This phase angle at 1 cps was such as to permit the +180 degree phase excursion associated with the first bending mode's zero-pole combination without coming closer than 35 degrees to the  $\pm 180$  degree stability limit. Beyond the 1 cps frequency the open-loop phase angle was shaped to provide a phase stabilization of a first bending mode resonance out to 4 cps. Now, this rapid phase change was achieved at the expense of not being able to phase compensate the slosh modes at whose frequencies the negative phase shift was greater than 180 degrees. However, since the open loop gain of the compensated system at the slosh frequencies was about -15 db, as compared to slosh peaks less than +4 db, the slosh modes were adequately gain stabilized. The N + 30db peaks of the new, experimentally measured slosh modes arenot gain stabilized by this compensation filter design.

Subsequent studies of the redesign of the autopilot filter to phase-lead stabilize the slosh mode have indicated that this design policy greatly increases the sensitivity of the autopilot to bending mode characteristics. Unfortunately, the original bandwidth of the autopilot can be maintained only through the phase lead stabilization of the slosh modes.

Faced with this problem of designing an autopilot filter that would be sensitive to uncertain bending mode data, it was decided to look into the possibility of reducing the autopilot bandwidth and speed of response requirements so as to allow an alternative design approach that would be less sensitive to the bending mode characteristics. It was found that the major problem affecting bandwidth was that of compensating for an initial misalignment of the engine thrust vector. The filter gain and system bandwidth of the R-503 autopilot were considered minimum with respect to thrust misalignment correction. Fortunately, an additional autopilot feature has been found that allows these filter requirements to be relaxed. This innovation involves the addition of a computer program to generate an estimate correction to the initial gimbal bias signal, as shown in Figure 3-8. This correction is computed and inserted approximately 4 seconds after thrust initiation.



Simplified Schematic Block Diagram of One Channel of the CSM-LEM Pitch-Yaw Autopilot Figure 3-9

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The gimbal bias correction computation is based on the double integration of a simplified, rigid-body equation of motion,

where

I = moment of inertia of the vehicle about the pitch (or yaw) axis.

= angular acceleration of the vehicle about the pitch (or yaw) axis.

T = engine thrust.

 $l_{\rm v}$  = moment arm from the engine hinge point to the vehicle c.g.

 $\int_{c}$  = engine gimbal command supplied by the autopilot filter.

 $\mathcal{S}_{bc}$  initial offset of the thrust vector (to be computed as a bias correction).

Integration of this equation gives the expression for computing the bias correction:

$$S_{bc} = \frac{z}{t_o^2} \left( \frac{T}{t \ell_x} \right) \left[ \Theta(t_o) - \Theta(o) - \Theta(o) + o \right] - \int_0^t \left[ \int_0^t S_c(t) dt \right] dt$$

where  $t_0$  = time at which the bias correction is inserted, measured from the instant of thrust initiation (with proper allowance for the thrust buildup time). This relationship requires as inputs the initial turning rate of the vehicle,  $\Theta(0)$  and the initial and final attitude angles,  $\Theta(0)$ ,  $\Theta(t)$ . In order to minimize the effects of slosh induced oscillations of the vehicle on the bias correction estimate, the computation period,  $t_0$ , is made equal to the estimated period of the predominent slosh mode.

Using the bias correction computation scheme the original autopilot filter of R-503 can be easily modified to make the autopilot insensitive to the characteristics of any bending modes or other resonances occurring above 1 cps. The required modification is simply to reduce the filter gain by a factor of 4 and to increase the sampling time from 40 milliseconds to 80 milliseconds. This modification has the result of shifting the open-loop gain and phase versus real frequency,  $\omega$ , downwards in frequency by a factor of two. Thus, the bandwidth of the new autopilot system is one half that of the original system, but the effects of its gain and bandwidth reductions are compensated for by the added bias correction compilation. The effects of this frequency

plot shifting are (1) to phase stabilize the slosh mode at 0.5 cps in the same way as the bending mode at 1 cps was compensated originally, (2) to phase stabilize the first bending mode also, out to 2 cps, and (3) to gain stabilize any bending modes occurring above 2 cps.

The filter function, D(z), of the Mission 278 autopilot for the CSM-LEM vehicle is as follows:

$$D(z) = Rc$$
  $\frac{i=7}{11}(3-pi)$   $\frac{i=7}{11}(3-gi)$ 

where

$$K_c = 5.747 \times 10^{-3}$$

and where the zeros and poles of D(z), and their corresponding factors, are as follows:

Zeros:

$$z = 0.61088$$
  
 $z = -0.9920$   
 $z = +0.9920$   
 $z = +0.4279 + j 0.6821$   
 $z = +0.7521 + j 0.4858$ 

Corresponding Factors:

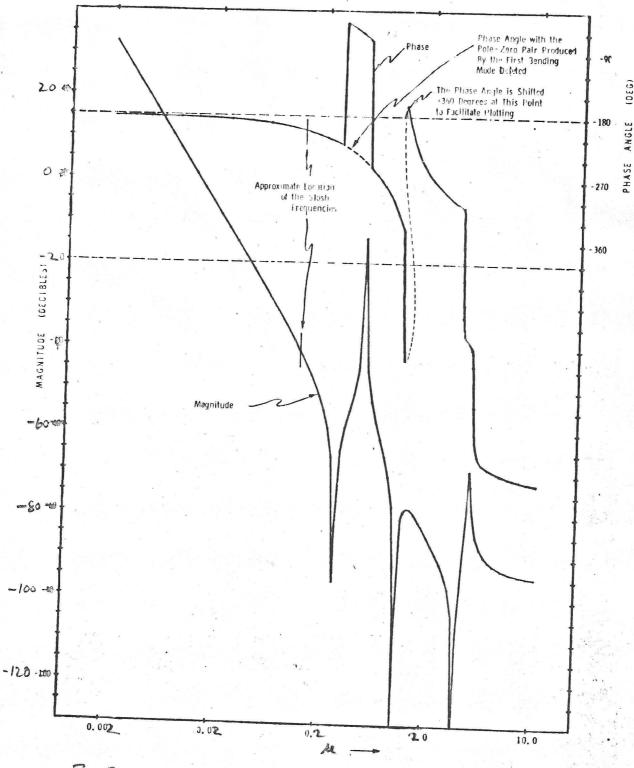
$$(z - 0.61088)$$
  
 $(z + 0.992)$   $(z^2 - 0.9841)$   
 $(z - 0.992)$   
 $(z^2 - 0.8558z + 0.6483)$   
 $(z^2 - 1.5042z + 0.8016)$ 

Poles:

$$z = 0.000$$
  
 $z = 0.5385$   
 $z = 0.5385$   
 $z = 0.9257 + j 0.1129$   
 $z = 0.9257 + j 0.1129$ 

Frequency response and transient response characteristics of the CSM-LEM autopilot without the external guidance loop and the thrust vector bias correction are presented in Figures (0 to 13.

Response characteristics of the CSM-LEM autopilot with the guidance loop closed and the bias correction added are discussed in a later section of this report.

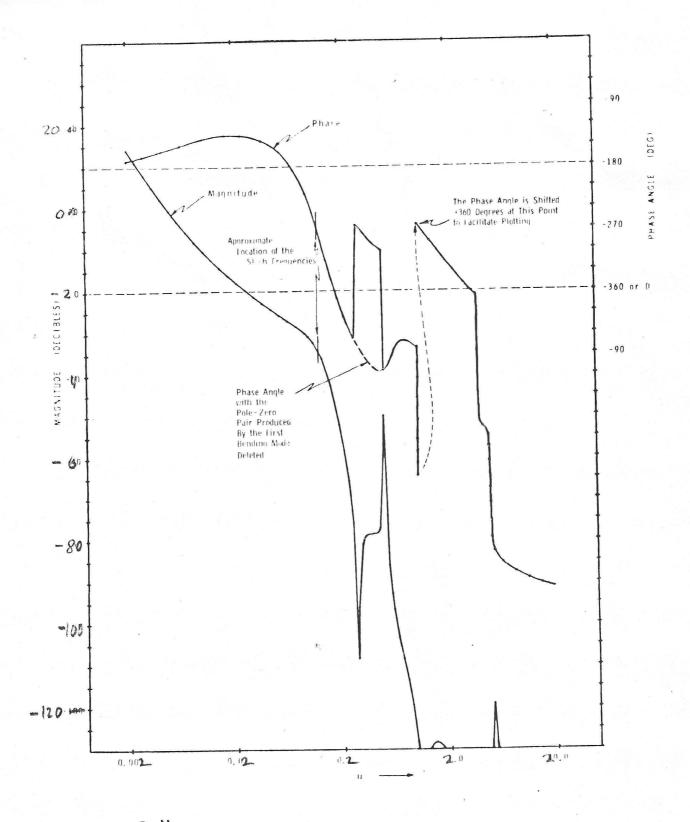


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\* "自己的证明是提供的。"

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Fig. 3-10
Fig. 3 Magnitude and phase of G(ju) versus u [slosh effects deleted].



3-11
Fig. \* Magnitude and phase of D(ju) G(ju) versus u [slosh effects deleted].

3-34

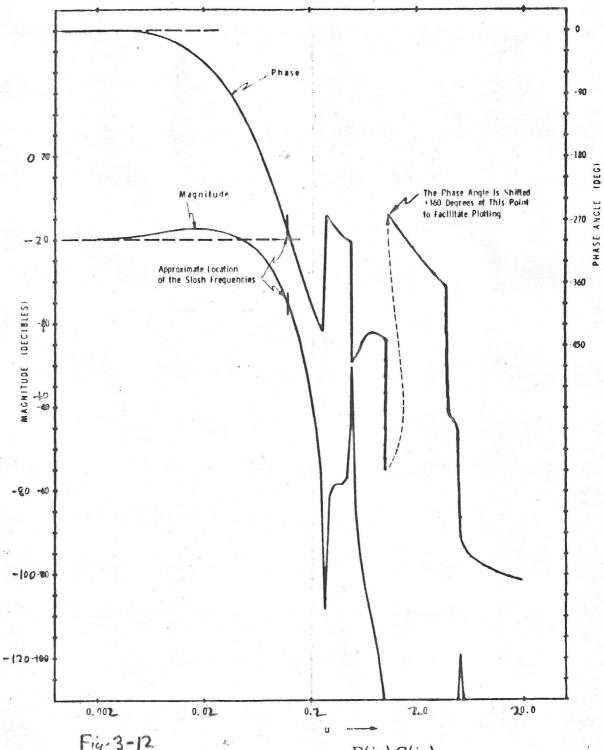
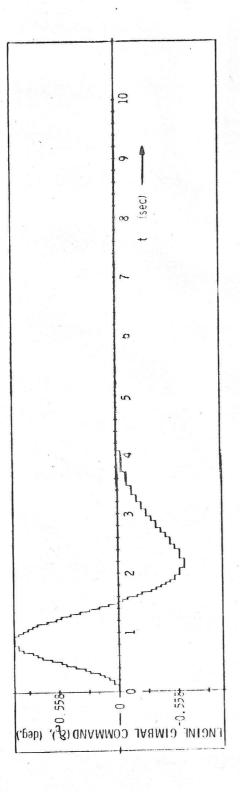


Fig. 3-12

Fig. 6 Magnitude and Phase of  $\frac{D(ju) G(ju)}{1 + D(ju) G(ju)}$  versus u [slosh effects deleted].



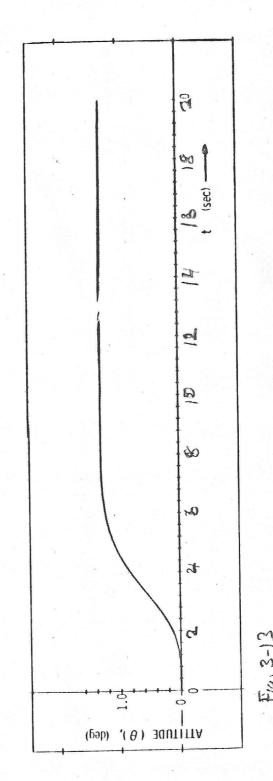


Fig. 3-13

Fig. 9. Response of the autopilot to a one degree step in attitude command; compensation gain set at the design value.

[ Bias Correction Composation Omitted]

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#### 3.3.4.4 Roll Autopilot

During thrust vector control using the SPS engine, "roll control" is effected by maintaining the outer gimbal angle (OGA) close to a preselected value. Prior to ignition the astronaut may roll the vehicle to a preferred orientation, if desired. The OGA corresponding to the orientation at ignition will be maintained throughout the powered maneuver. It is expected that under nominal conditions the inner and middle gimbal angles will be near zero at the start of a major powered maneuver. As a result OGA control is equivalent to roll control. With a sizeable initial MGA (e.g., 45°) some roll motion will occur but this is not considered significant.

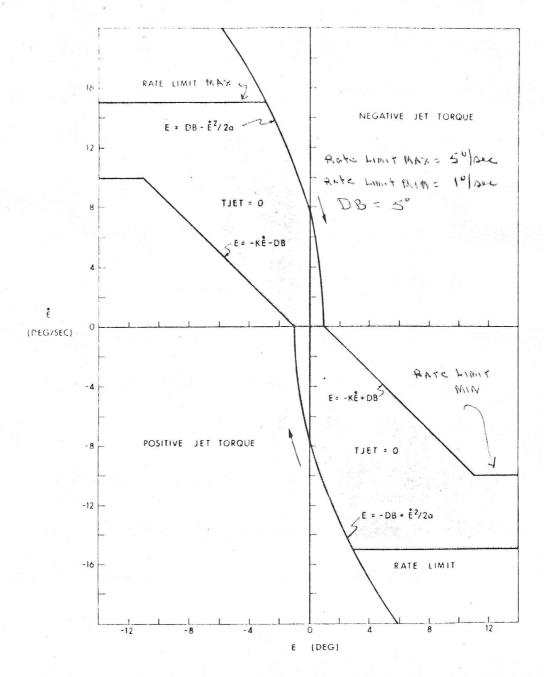
The roll autopilot is an adaptation of the LEM p-axis control as documented in Chapter III of R-499, 7/65. The phase plane switching curve is shown in Figure 3-14 for convenience. The slope of the straight line is chosen at  $K = -2^{\circ}/\sec/\deg$  which is a good compromise between fuel usage and speed to response.

Since the autopilot operates in a thrusting environment, subject to disturbing torques generated by actuator limits-cycling and cross-coupled inertias, minimum impulse performance is unlikely. In Figure 3-15 the tables indicate the lb/sec consumption of roll fuel for lunar orbit insertion and transearth injection. Two-jet firing is assumed and the mid-burn roll inertia is selected as an average value. It is observed that a limit cycle rate of 1°/sec yields satisfactory fuel economy for an undisturbed flight. Since disturbances will be significant any attempt to constain the rate to a smaller value would be futile. By sampling every 1/2 sec and simply deriving the OGA rate by differencing, the theoretical accuracy due to quantization is .0222°/sec. Thus at this sampling rate .1°/sec can be detected quite easily.

The dead band has been set at  $\pm$  5° which is adequate for this mode. Figure 3-16 is an illustrative run with initial conditions set at  $(OGA_o)^{error}$ ,  $(OGA_o)^{error} = 1^{\circ}/sec$ . In addition the slosh masses have been pulled at the start to simulate the results of an ullage maneuver. (Note: actual conditions at the start of a powered maneuver should be close to  $(OGA_o)^{error} = \pm 1/2^{\circ}$ ,  $(OGA_o)^{error} = .05^{\circ}/sec.*$ ) The OGA settles quickly into its  $\pm$  5° dead band with jet firings occurring about every 100 sec.

Since roll control driving TVC is not a precise function the design philosophy, correspondingly, has been kept simple. However in order to increase reliability during this automatic sequence; i.e., the astronaut

\* Maintained by the RCS Autopilot.



3-14A
3-14A
Fig. 14b X-axis RCS laws in the phase plane.

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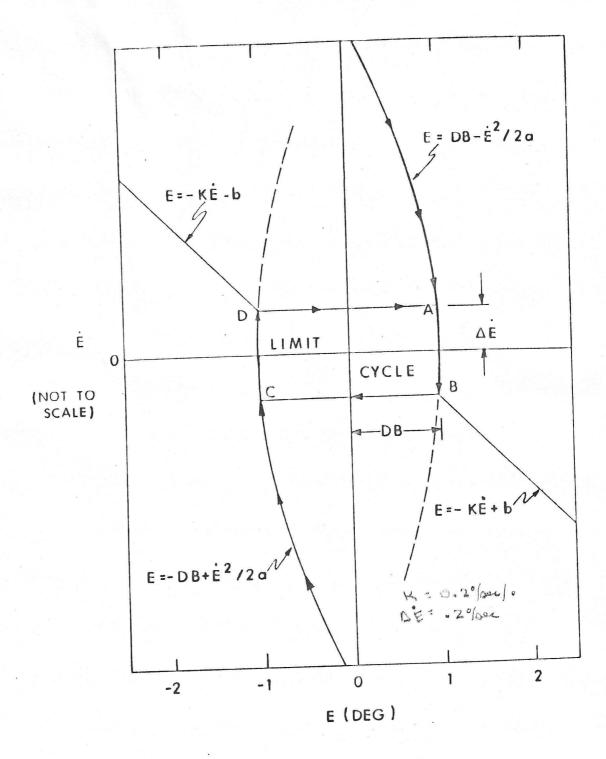


Fig. 14a X-axis-limit cycle and switching parabolas.

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| A         | · Juna 0, 2, 2        | was string              |                           |  |
|-----------|-----------------------|-------------------------|---------------------------|--|
|           | Ixx = 43,500 plus-1/2 | Mintemp = 1400 1/2 2/2. |                           |  |
|           | Lund Cycle            | Ils/sec                 | 10823. Vonge 9,7 3,20 see |  |
|           | 1.0                   | ,6791                   | 35.4 lbs.                 |  |
|           | 0.3                   | 0196                    | 6.35                      |  |
|           | 0.7                   | .00316                  | \. O \                    |  |
| Solected  | , 0.1                 | 19 7000.                | 0.25                      |  |
|           | 0.05                  | 891000.                 | 0.06                      |  |
| NW.       | 0.01                  | .0000079                | 0.0025                    |  |
| Subject > | 6.0093                | 8300006                 | 0.0022                    |  |

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also Fig \$ 3-15

B. Then Eath. Subjection

Ixx = 16,600 plug &"

TSP - 275 Dac

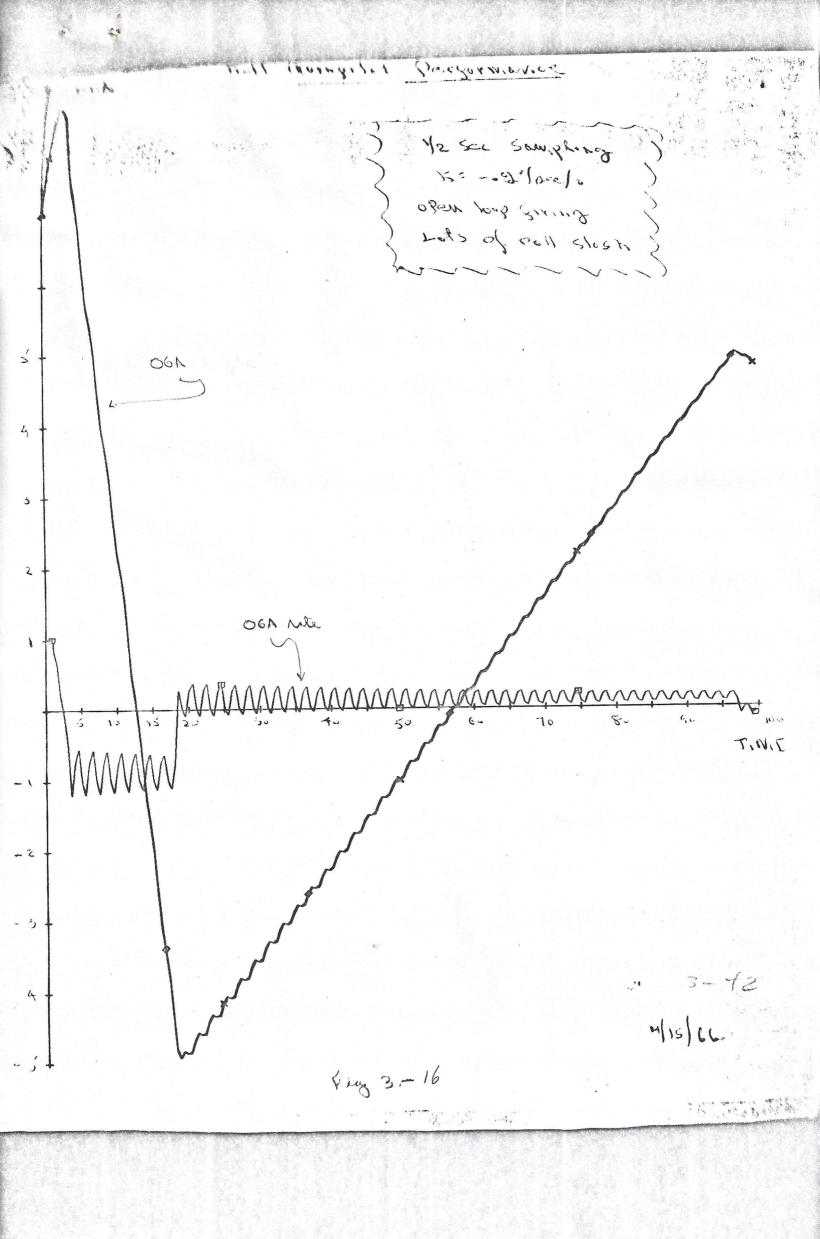
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| Total Jeans Sc 150 Dar |
|------------------------|
| 3.62 910               |
| 0.905                  |
| 0.144                  |
| 0.0362                 |
| 0.00965                |
| 0.00246                |
| 0.00036                |
|                        |

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cannot exercise manual control in the primary mode, the following features have been included in the roll autopilot:

- (1) the maximum firing time for any one firing is set at 2.5 sec.
- (2) following any firing, there is a delay of at least 1/2 sec before the next firing. This delay enhances the rate estimation and breaks up s long maneuver into 2.5 sec firings and at least 1/2 sec measurements.
- (3) each jet pair fires alternately. That is, if two consecutive positive roll firings are required, these will consist of separate fet pairs. Roll control can remain operative under a combination of jet failures.

## 3.3.4.5 The TVC Digital Autopilot Program

The digital autopilot compensation filters for Flight 278 are based upon the MOD 0 design documented in R-503 and SGA Memo #9-66. The data assumed for design purposes are also included in the same Memo. The slosh and bending characteristics for the 278 LEM-docked vehicle are sufficiently different to warrant a modification in the MOD 0 design. Essentially, the dynamic compensation afforded by the filters has been shifted to provide increased stability margins at the slosh frequencies. This shift is accomplished by increasing the sampling interval from 40 ms. to 80 ms. In addition, the design gain has been reduced by a factor of 4. In all, the autopilot for the docket configuration for Flight 278 will provide satisfactory performance for the 10 sec. normal burn and the 35 sec. stroking test. The larger pointing errors attending lower d.c. filter gains will be reduced by adjusting the actuator trim angles one or two slosh periods after engine ignition.

From a programming point of view a digital filter is a difference equation (or set of equations) in which old outputs and inputs are processed through appropriate gain coefficients. The Fourier transform, of a delayed unit impulse is simply  $e^{-sT}$  or 1/z. Regarding the sampled data as mathematical impulses, the resulting polynomials in z represent a difference equation involving present and past, inputs and outputs.

Thus

$$D(z) = K_{z} \frac{z^{r} + N_{1} z^{r-1} + \dots + N_{i} z^{r-1} + N_{r-1} z + N_{r}}{z^{s} + D_{1} z^{s-1} + \dots + D_{j} z^{s-j} + D_{s-1} z + D_{s}}$$

or

$$D(z) = K_z \frac{\pi_i (z - z_i)}{\pi_j (z - p_j)}$$

For Flight 278, the coefficients  $N_i$ ,  $D_j$  are

$$N_1 = -2.971$$
  $D_1 = -4.7798972$   $N_2 = -3.195$   $D_2 = 9.4452763$   $N_3 = -0.4096$   $D_3 = -9.8593475$   $N_4 = -2.518$   $D_4 = 5.7231811$   $N_5 = 2.963$   $D_5 = -1.7484750$   $D_6 = 0.21933335$   $N_7 = 0.3124$ 

and the difference equation for the filter may be written

$$\delta_{k} = -[D_{1} \delta_{k+1} + \dots + D_{6} \delta_{k+6}] + K_{z} [E_{k} + N_{1} E_{k+1} + \dots + N_{7} E_{k+7}]$$

where

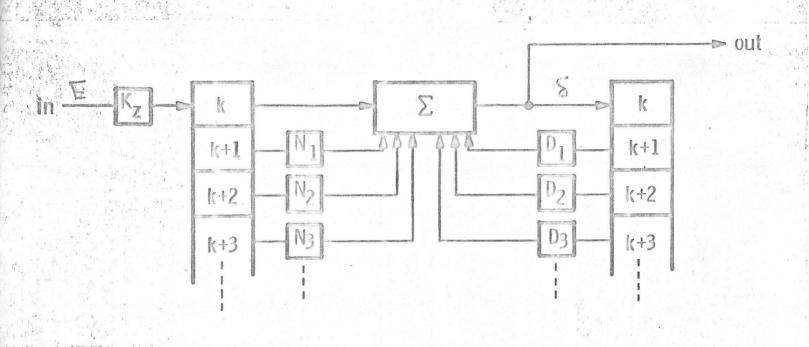
$$E_k = filter input$$

$$\delta_{\mathbf{k}} \equiv \text{filter output}$$

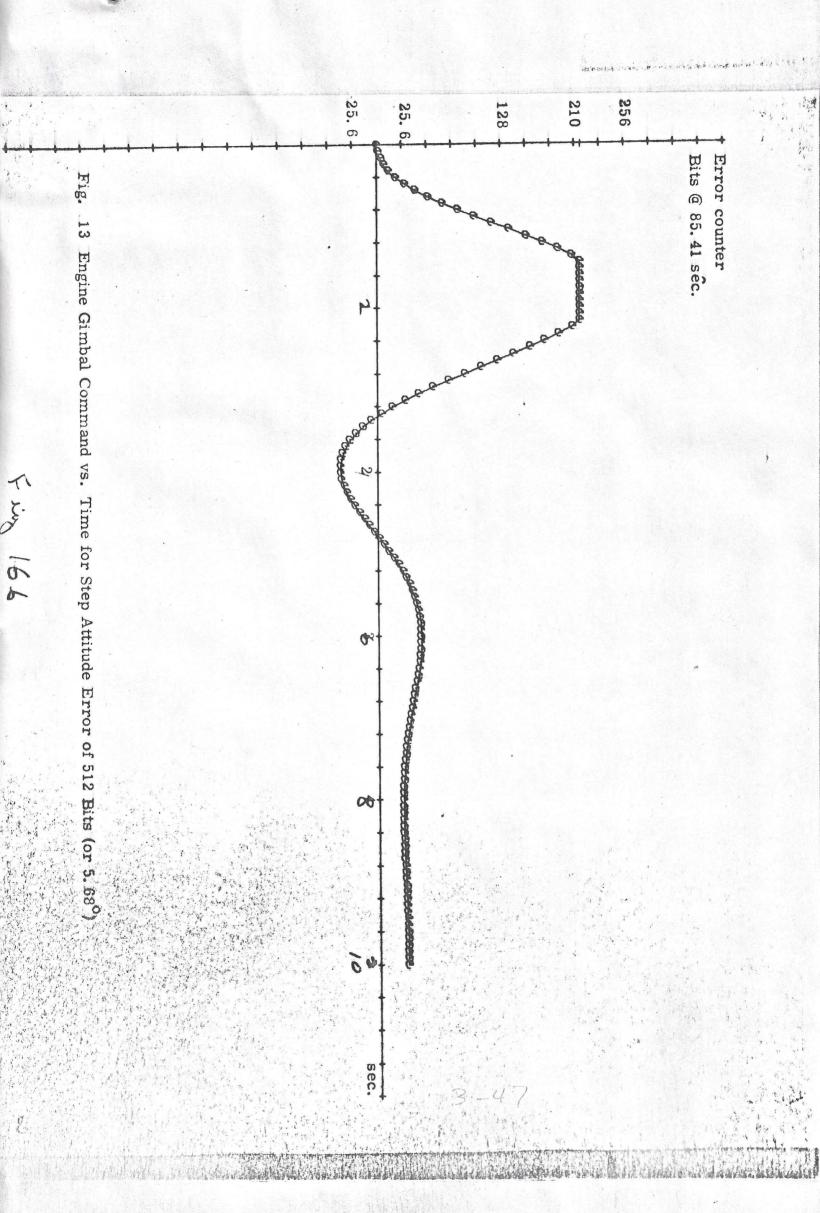
A schematic of this implementation is shown in Fig. 1. Note that the coefficients  $N_i$  are held in single precision while the  $D_i$ 's are double precision constants. The multiplications  $E_k N_k$  are  $SP \times SP$  but the  $D_k \delta_k$ 's must be performed as  $DP \times DP$  (see SGA Memo # 26-65.). In Fig. 2 and 3 the step responses of the filter as programmed in the AGC, for a 40 ms. sampling interval, (i.e. MOD 0) are shown for large\* (5.8°) and small (.17°) input steps. In each case, the solid underline represents the ideal response neglecting the finite word length of the AGC. The filter evidence proper transient response over a wide range of input amplitudes. The modifications in design for Flight 278 reduce the ordinate by 4 and doubles the time scale.

The LEM-off MOD 0 digital filters have been documented in R-533, Guidance Programming Memo 1-66 and SGA Memo #9-66. For Flight 278 this design is adequate with a slight modification of the lag break frequency (see above autopilot description). The Z-transform of the original filter for which guidance-autopilot performance data is available (in Guidance Programming Memo # 2-66) is

<sup>\*</sup> The actuator signal has been limited to a command of  $+5^{\circ}$ .



Feigure 16a



$$D(1) = K_c \frac{a_0 + a_1 Z^{-1}}{b_0 + a_1 Z^{-1}}$$

where

$$a_0 = 1$$
  $b_0 = 1$   $a_1 = -0.98$  ,  $b_1 = -.87$ 

Figure 4 shows the analytical results for a 1° step input. This was obtained by using a MAC (FORTRAN-like) program solving the basic difference equation, Fig. 5 shows the excellent agreement between the analytical and AGC results; their difference is less than 0.1%. Figure 6 shows the normalized output (with respect to a 100-bit input) for step-input to 5, 10, 100, 1000 + 2000 bits (at 39.55 sec./bit). The program shows a linearity of 0.1% for input steps between .05° and 20°.

The filter gains for both LEM on and off autopilots are time varying quantities reflecting the torque to inertial ratio of the vehicle during SPS burn. The selection of the gains and their variations are commented upon in the next section. 3.3.4.5.1

#### Time Variable Parameters

The control loop gains are made time-variable to offset the variation of control effectiveness ( $T_x^l/I_{yy}^l$ ). In particular, the desired gain:

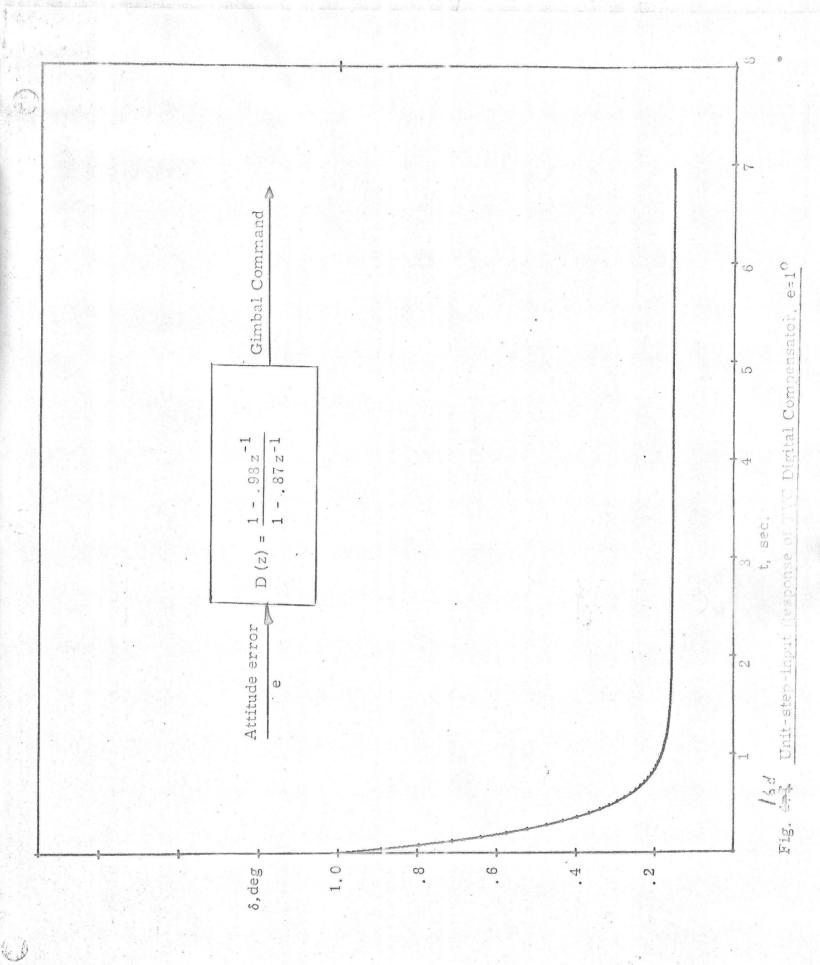
$$K_{p}(t) = \frac{\left[K_{p}(Tl_{x}/Iyy)\right] \text{ DESIGN}}{\left[Tl_{x}/Iyy\right] \text{ CURRENT}}$$
(1)

Analysis shows that  $K_p(t)$  can be approximated by a linear function of elapsed time for a particular burn:

$$K_{p}(t) = K_{p}(0) - K_{p}(t - t_{0})$$
 (2)

The start burn value,  $K_p$  (0), in turn is also approximated by a linear function of the "expected acceleration", EXACC: thus

$$K_{p}(0) = K_{p_{1}} + K_{p_{2}} / EXACC$$
 (3)



-7- 3-50

# Table 4.1 Unit-Step Response-Comparison of MAC and AGC Programs

| Time  | Unit-Step I | Response (deg) |
|-------|-------------|----------------|
| (Sec) | MAC         | AGC            |
| .000  | 1.0000      | . 9995         |
| . 040 | . 8900      | . 8896         |
| .080  | . 7943      | . 7940         |
| . 120 | . 7110      | . 7107         |
| . 160 | . 6386      | . 6384         |
| . 200 | . 5755      | . 5754         |
| . 0   | .1798       | .1799          |
| 2.0   | .1546       | . 1547         |
| 3.0   | .1538       | . 1539         |
| 4. 0  | . 1538      | . 1539         |
| 5. 0  | .1538       | .1539          |

Fing 16e

Table 4.2: Normalized Step-Input Response-Linearity Test

|       |   |         |         | Marie Control of the |         |  |
|-------|---|---------|---------|---|---------|--|
| Time  | Normalized output = output $\times \frac{100}{C}$ (deg) |         |         |   |         |  |
| (Sec) |   | ê (bi   | ts)     | ě.  |         |  |
|       | . 5   | 10      | 100     | 1,000   | 2,000   |  |
| 0     | 1.09846   | 1.09846 | 1.09846 | 1.09846   | 1.09846 |  |
| 1     | . 20207   | . 20202 | . 20200 | . 20200   | . 20200 |  |
| 2     | .17024  | .17018  | .10716  | . 17016   | .17016  |  |
| 3     | .16931  | .16923  | .16918  | .16918  | .16918  |  |
| 4     | .16931  | .16923  | .16915  | .16915  | .16915  |  |
| 5     | .16931  | .16923  | .16915  | .16915  | .16915  |  |

Fing 164

"EXACC" is entered by the Astronaut before each burn.

Mechanization of variable K<sub>p</sub> within the AGC requires a discrete form of equation (2):

$$K_{p}(nT) = K_{p}((n-1)T) - \Delta K_{p}$$
 (4)

where  $\Delta K_p = T(K_p)$ , and the initial  $K_p(0)$  is form equation (3).

The constants  $K_{p_1}$ ,  $K_{p_2}$ , and  $\Delta K_p$  differ significantly for LEM on and LEM-off burns. Two sets of these parameters are required to closely match the desired  $K_p(t)$ .

Empirical data is similarly employed to estimate the roll acceleration during a burn and the slosh period at engine ignition.

## 3.3.4.5.2 Computer Timing and Storage Estimates

For Flight 278 the LEM-docked digital autopilot will operate at a basic sampling interval of 80 ms. That is, the pitch and yaw filters are called, alternately, every 40 ms. Each filter requires an interrupt time of about 5 ms., thus the filters are expected to occupy .125 sec. every 1 sec. or 12.5 % of main frame time. For LEM-off, the interval is 40 ms. with an interrupt time of 1.25 ms. or only 3%. The roll autopilot (described elsewhere) operates every 1/2 sec. and occupies an additional 1% of computer time. Thus for Flight 278, the autopilot functions are expected to take less than 15% of main frame time during a powered maneuver. Based on Block I experience for steering and matrix updating and taking into account the increased speed of the Block II interpreter, the steering equations for Flight 278 will be solved every 2 sec., occupying .5 sec. of computer time or 25%. Thus for guidance, steering and autopilot a powered maneuver can be expected to take 40% of main frame time.

The filters (LEM on and off) and roll control require approximately 1500 fixed locations. The filter gains, coefficients and roll control acceleration estimate are carried in erasable. In addition, provision is made for the astronaut to key in the pitch and yaw actuator trim angles during pre-thrust. The total erasable load will be close to 110 registers.

# 3.3.4.6 Guidance - Autopilot Performance

As a result of simulations for the LEM on and off cases using the AGC programs and cross-product control the achieved pointing errors appear to be within acceptable limits. These runs incorporate an initial uncertainty of 1° in the knowledge of c.g. location, and mass loadings corresponding to Flight 278.

| Burn Time | Δ٧        | Pointing Error |
|-----------|-----------|----------------|
| 10 sec    | 29 m/sec  | 5.40           |
| 35 sec    | 110 m/sec | 1.0°           |

For LEM-off the pointing error has been plotted in Fig. 7 as a function of desired  $\Delta V$ , The pointing error peaks at 2.3° for  $\Delta V = 20$  m/sec.

3-54

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#### 3.3.5 RCS DAP for the CSM

#### 3. 3. 5. 1 Introduction

The Apollo Spacecraft (Fig. 3-17) consists of a reentry capsule - the Command Module - attached to a propulsion and support unit - the Service Module. During translunar flight the spacecraft is docked with the Lunar Excursion 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.

The primary requirement for a DAP for this task is to conserve RCS fuel. The policy for jet firings is to achieve prerequisite attitude and/or maneuvers in a reasonable time with a minimum expenditure of fuel. Only in the FREE mode does the manual control permit acceleration control without "fuel-conscious" rate limiting.

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

- (a) The CMC is to have reaction control capability at all times when the GN&C system is on.
- (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 CMC and are to have priority at all times with rotations taking precedence.
- (d) A three-position mode control switch interfaces with the CMC. 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,

In the "Auto" mode, the mission program will determine the control mode; however, in both Hold and Auto modes manual rotation

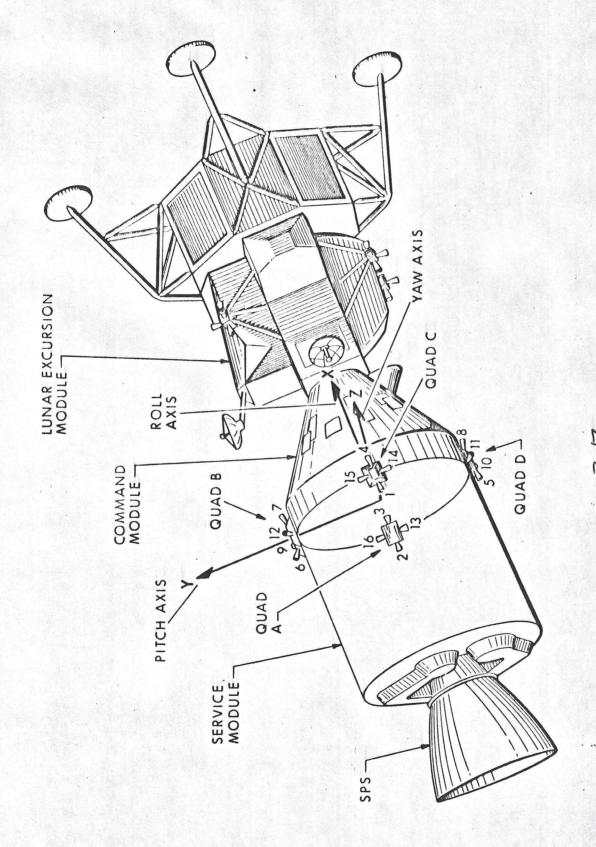


Fig. My Apollo Spacecraft

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 CMC 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.

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 singlejet 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.

### 3.3.5.2 General Description

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 (autopilot) 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. Because 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 Figure 3-18.

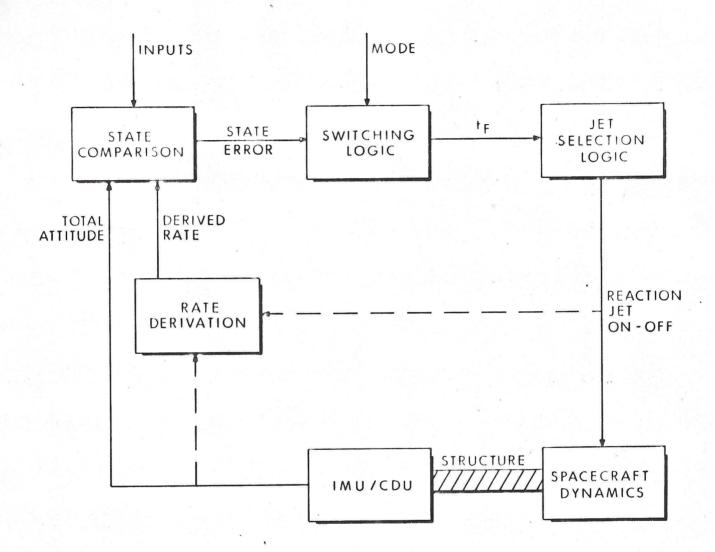
In designed the switching logic for this system, the performance of the reaction jets was studied to achieve efficient operation. Figure 3-19 shows the total impulse of a jet plotted against the length of time the electrical "on" signal is applied. Also plotted is the specific impulse, ie. the ratio 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.

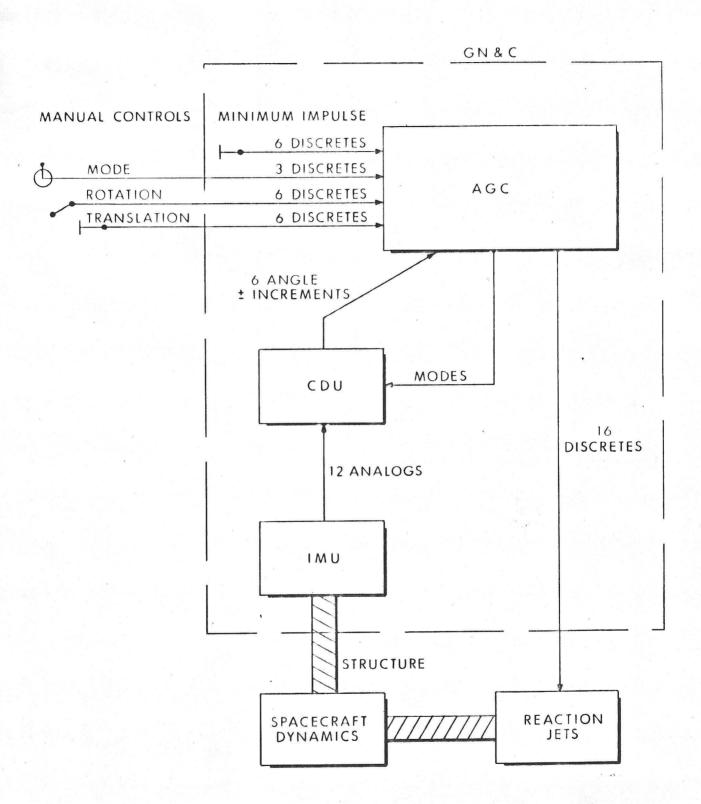
Since the vehicle dynamics can be modeled as a secondorder system, phase plane methods can be conveniently used to describe the vehicle state. Thus, it is convenient to look on the control problem as one of estimation of state, and to use the phase plane together with the associated switching logic to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

#### 3.3.5.2.1 Spacecraft 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 Figure 3.20. 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.



3-18
Fig. 24A Block Diagram of Autopilot



3-20
Fig. App Model used in the Design

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 CMC. 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 they Digital Analog Converters to drive displays (also used in the GNC - S IV B 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 sumed 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.

3.3.5.2.2. Computer Design

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

3.3.5.2.2.1. Channels

Inputs channels are provided for "SM separate", "S IV B separate"; rotation control\*, translation controls\*, "Hold", "Free", "G&C control"; minimum impulse\*, "LEM attached".

The asterisks above indicate those manual controls which cause an interrupt in the CMC. Once made, this interrupt can no longer be obtained until a trap is reset (see below).

Output channels are provided for pitch and yaw Reaction Control jets; roll control jets;

reset manual control interrupt trap, enable T6 interrupt, (see below); CDU modes; CDU-DAC activity bits.

3.3.5.2.2.2 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 counter 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.

#### 3.3.5.2.2.3 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 digitial autopilot is wanted. The following description is for the case where Reaction Control is required (see Figure 3.21).

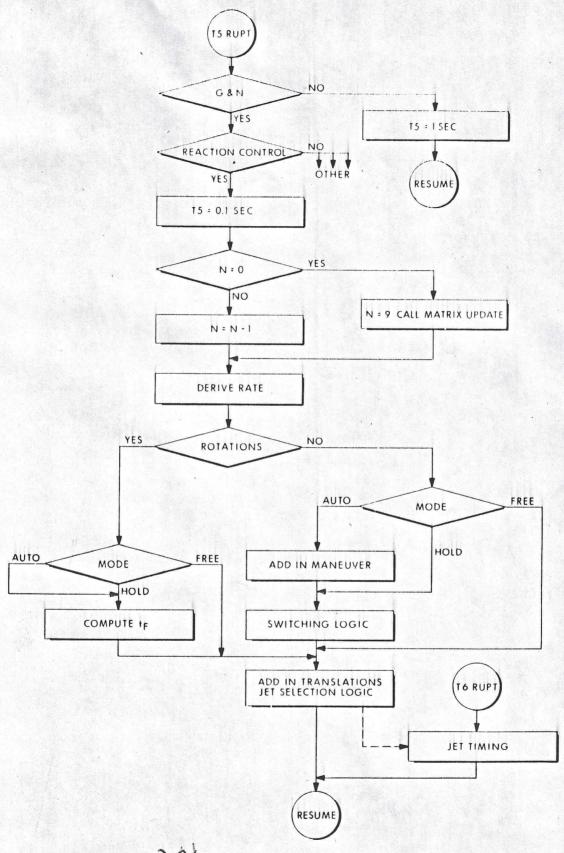


Fig. 4/4 Flow Chart of Program

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.

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.

#### 3.3.5.3 Detail Design

3.3.5.3.1. Rate Derivation

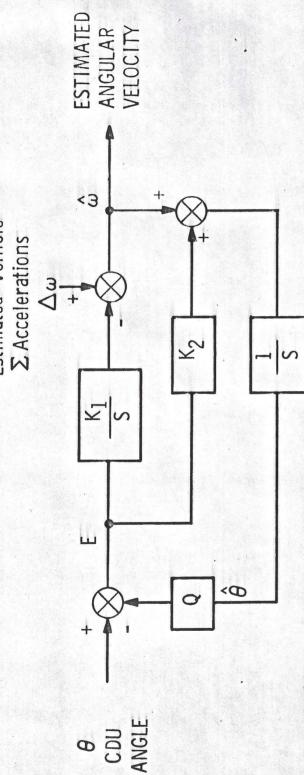
Derivation of the body rates of the spacecraft from the gimbal angles of the Inertial Measurement Unit is complicated by two factors.

- 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 deadband. 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 thereis no following error, and thus no disturbance of the rate estimate (see Figure 3.29).





25 wn

DIFFERENCE EQUATIONS: <3 **Φ**> Fig. ## Filter Block Diagram

EQUIVALENT TRANSFER FUNCTION:

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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, the times of firing of the jets, and a nominal Moment of Inertial for each phase of the mission.

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.

Tentatively, the characteristics of the estimating filter are  $\omega_n$  = 0.4 rad/sec. 5 = 0.8

3.3.5.3.2 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 3.3.5.2.3). 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.

3.3.5.3.2.1. 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 3.3.5.3.3, 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 3.3.5.3.3.3.

## 3.3.5.3.3.2 Auto and Hold Modes

In these modes the rotation hand controller is mechanized to give a fixed 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.

3.3.5.3.3 Jet Selection Logic

For the purpose of jet selection we can divide the jets into three groups as shown in Tables 3-2, 3-3 and 3-4. 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.

3.3.5.3.3.1 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 3-2 for example, we will see that the result will be zero net translation. Thus the first step of the logic (see Fig. 3.3) is to ignore translations in the presence of reported fails. Wherethere 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.

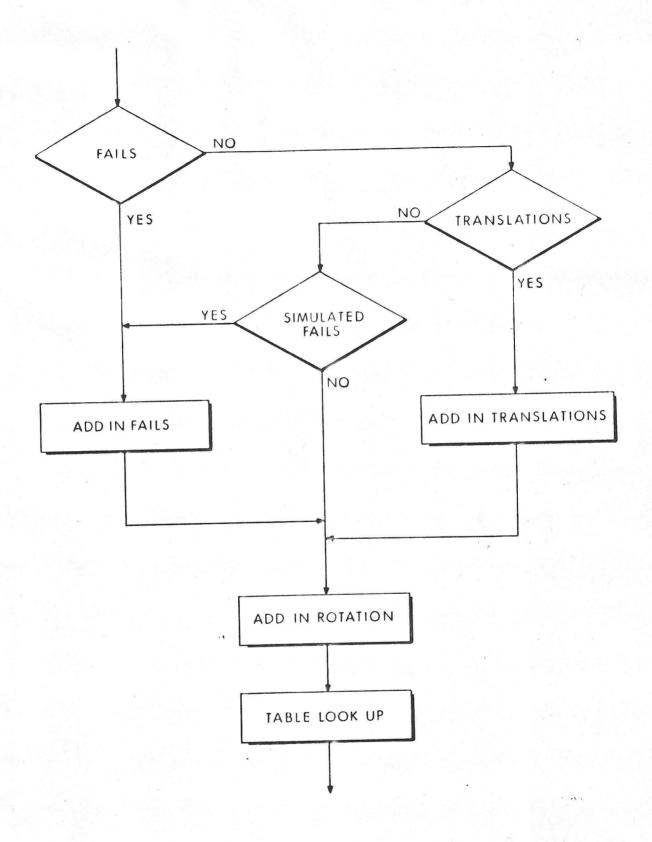
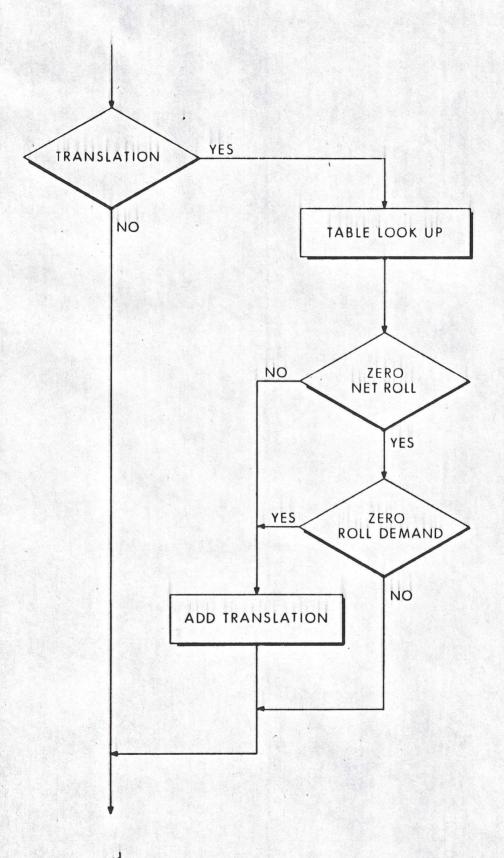


Fig. 3.23 Logic for Rotation Quads



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Fig. 3. Logic for "Roll" Translation Quads

Table 30

# Pitch Jets also Control ± X Translation

| Quad | Jet | Pitch            | Translation |
|------|-----|------------------|-------------|
| Α    | 3   | +                | -X          |
| C    | 1   | +                | , +X        |
| Α    | 2   | -41 <b>-</b> 017 | +X          |
| C    | 4   | _                | -X          |
|      |     |                  |             |

7 Table 3.4

# Yaw Jets also Control ± X Translation

| Quad | Jet | Yaw                | Translation |
|------|-----|--------------------|-------------|
| В    | 7   | +                  | -X          |
| D    | 5   | +                  | +X          |
| В    | 6   | 10 s. <del>.</del> | +X          |
| D    | 8   |                    | -X          |

Table 3. \$2

# A-C Jets Controlling ± Y Translation

| Quad | Jet | Roll | Translation |
|------|-----|------|-------------|
| Λ    | 13  | +    | +Y          |
| С    | 15  | +    | -Y          |
| A    | 16  |      | <b>-</b> Y  |
| C    | 14  |      | +Y          |

4b Table 3. M

# B-D Jets Controlling ± Z Translation

| Quad | Jet | Roll | Translation |
|------|-----|------|-------------|
| В    | 9   | +    | +Z          |
| D    | 11  | +    | -Z          |
| В    | 12  |      | -Z          |
| D    | 10  | - 1  | +Z          |

The pitch rotation selections are illustrated in Table 3-5

Table 3-5

#### Pitch Rotation Select

| Item | Pitch    | X-XLN | Quad Fail  | Jets | Torque |
|------|----------|-------|--|------|--------|
| i    | 0        | 0     | The state of the s |      | 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    | + [      |       | The state of the s | 3    | i      |
| 9    | <u>-</u> | 4     |  | 4    | -1     |
| 10   | 0        |       | A  |      | 0      |
| 11   | +        |       | Α  | i    | +1     |
| 12   | •        |       | A  | 4    | -1     |
| 13   | 0        |       | Ċ  |      | 0      |
| 14   | 1        |       | 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.

## 3.3.5.3.3.2 'Roll' Translation Quad Logic

If the unused roll quad pair has a translation demand  $(\pm Y)$  on A-C or  $\pm Z$  on B-D) then we look up in a table (such as Table 3-6) the appropriate jets to fire allowing for failures.

Table 3-6

#### A-C Translation Only

| Item | Translation | Quad Fail | Jets   | Torque |
|------|-------------|-----------|--------|--------|
| 1    | 0           |           |        | 0      |
| 2    | +Y          |           | 13, 14 | 0      |
| 3    | <b>-</b> Y  |           | 15,16  | 0      |
| 4    | 0           | Α         |        | 0      |
| 5    | <b>+Y</b>   | Α         | 14     | -1     |

Table 3-6 (con't)
A-C Translation Only

| Item | $T_1$ | ranslation    | Qu  | ad Fəil | Jets | Torque |
|------|-------|---------------|-----|---------|------|--------|
| 6    |       | $-\mathbf{Y}$ |     | A       | 15   | +1     |
| 7    |       | 0             | - 1 | 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 Figure

# 3.3.5.3.4 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 Figure 3-25) All jet burnings are limited to a minimum 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 in this case is simply the number of jets scaled by the sampling interval of 0.1 sec.

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.

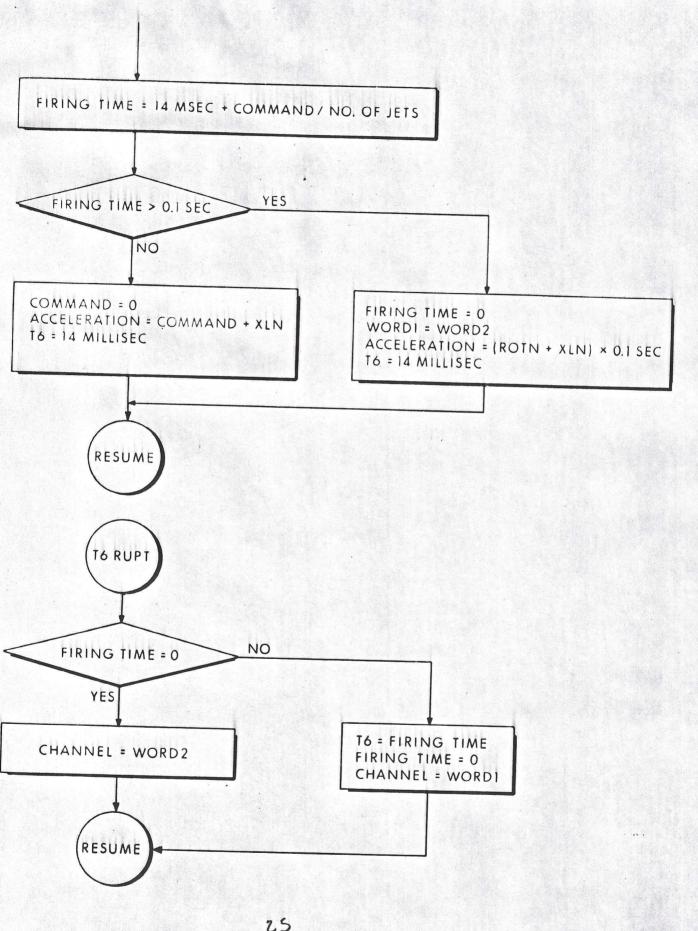


Fig. 3.4 Jet Timing Logic
27.

# 3.3.5.3.5 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 Figure 3-26. 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 inertial, 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 the 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.

The value of rate limit is  $0.2.0^{\circ}/\text{sec}$ . The Astronaut may select his deadband (between  $0.5^{\circ}$  and  $5.0^{\circ}$ ) by Keyboard command. The program for this control has not yet been written.

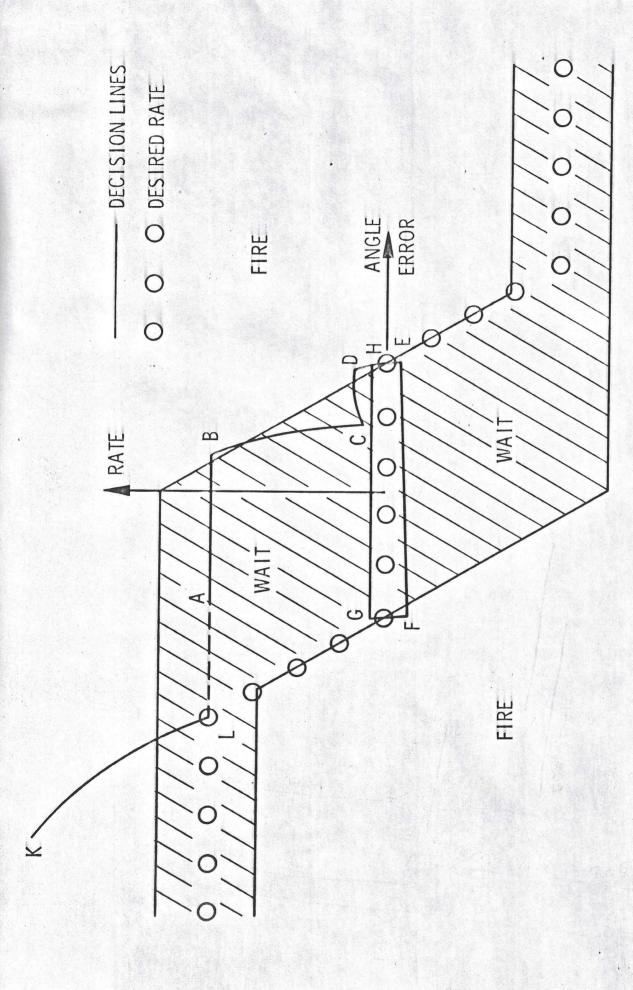


Fig. 3.4 Phase Plane Showing Switching Logic

Because of uncertainty as to the values of certain inertias, the characteristics of the attitude hold logic have not been finalized as yet. The tentative value of the rate/angle slope is 3/4 deg./sec./deg. The current width of the rate deadband is 0.05°/sec.

3.3.5.3.6 Automatic Maneuver Computation (Kalemanu)

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 maneuver rates are normally specified by the major program, but in certain instances may be specified by the crew as a Keyboard command.

The inputs from the maneuver computation program to the RCS DAP being described are those three spacecraft rates, and desired spacecraft attitude. Where gimbal lock is to be avoided, the two component maneuvers are sent to the control system separately.

During a mission phase such as Pre Thrust which calls for automatic maneuver, the DSKY will DISPLAY the desired final CDU angles and will flash V N . If the Astronaut wishes the maneuver to be performed automatically he will press Enter. If at anytime during the maneuver he takes the RHC out of detent, the DAP will follow in a rate control mode and will leave the S/C in attitude hold after the stick is returned to detent. He must reinitiate the maneuver if he wishes the automatic maneuver to continue. Final roll is normally left to the discretion of the Astronaut.

During thrusting maneuvers the KALCMANU program will position the thrust vector, not the X-axis of the spacecraft.

## 3.3.5.3.7. 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 toward the final position.

Due to the requirement for high vehicle maneuver rates which have 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.

#### 3.3.5.3.8 Prethrust Operations

The final task of a pre-SPS thrusting maneuvers is to perform ullage. The 2-jet X-axis translation is initiated X sec before scheduled SPS firing and persists until 2 sec after SPS firing. During this period the RCS DAP holds the vehicle in minimum deadband attitude hold. The RCS DAP is removed 2 sec after SPS firing.

#### 3.3.5.3.9 Post-thrust Maneuvers

At SPS cutoff the RCS DAP is restarted in the maximum deadband attitude hold.

#### 3, 3, 5, 3, 10 Saturn Control

During AUTO and HOLD modes the CMC responds to the RHC to provide limited rate commands to the CSM. The same programs for interpreting the RHC will be used during Saturn control to provide rate error signals via the DACS.

# 3.3.5.4. Performance

The system described in this section has been thoroughly exercised in digital simulation. Fuel economy proved to be close to the roughly calculated optimum. The system did not appear to be significantly affected by slosh induced disturbances (which were not included in the design model); however the slosh mechanism were not simulated in sufficient detail to render conclusive findings.

# 3.3.6 Entry Autopilots

The terminal phase of extra-atmospheric return to the earth at a specified landing point is achieved by Entry Guidance, utilizing atmospheric aerodynamic lift. The guidance equations determine proper orientation of the lift vector relative to the trajectory plane as the means of control in achieving the desired landing point. The component of lift in the plane of the relative velocity and the position vector controls the down range flight. Cross range is controlled by the out-of-plane component of lift. By rolling the CM about the velocity vector, orientation of the lift vector is achieved such that the desired in-plane component obtains. The out-of plane component of lift causes lateral drift of the vehicle. This sidewise motion is used for plane changing, if any is needed. Otherwise laterial drift is an unwanted byproduct of the steering, and is constrained by causing the CM to roll so as to reverse the sign of the out-of-plane component of lift. Since the inplane component is the fundamental controlled quantity, it is important that its sign be unchanged during lateral reversals or switchings. In effect this demands that the CM roll through the smaller of the two possible angles at lateral switching time. Thus Entry Guidance provides a single output quantity, commanded roll angle ROLL C, to be achieved by the CM.

Within the atmosphere the CM is aerodynamically stable, a property that simplifies the autopilot design. In pitch and yaw, since stability is no problem, simple rate dampers are indicated. Furthermore, aerodynamic forces can be utilized to do most of the work during a coordinated turn. The roll axis torque of the CM is mostly inertia, having negligible damping and cross-coupling terms. A simple predictive autopilot utilizing the phase plane is used.

3.3.6.1 Coordinate System

3.3.6.1.1 Trajectory Triad

The Inertial Measurement Unit is the source of input data for Entry Guidance and the autopilots. Accelerations in the form of velocity increments are provided by the PIPAs, and the gimbal angle values are communicated by the CDU's. The coordinate system used during Entry is defined in Figure 3-37. During entry the axis of symmetry of the CM is pointed backward along the trajectory and the triad used by the autopilots is -UVA, UYA, UNA. In the figure, the body axis set UBX, UBY, UBZ is shown for ROLL C = 0 (corresponding to lift up) and for  $\alpha = \alpha_{\rm t}$  (corresponding to trimmed flight).

LOCAL
HORIZONTAL

UIIA

VREL

UBY

UBY

BODY AXES DRAWN

FOR LIFT UP

ROLLC=0

UR = UNIT (R)

VREL = V - WE x R

UVA = UNIT (VREL)

UVA = UNIT (UVA x UR)

UNA = UVA x UVA

3-27
FIG AF ENTRY AUTOPILOT COORDINATE TRIADS

3. 3. 6. 1. 2 Body Triad

VV

The body axis triad of the CM is defined relative to the platform axis set in Figure 3-28 in terms of the three gimbal angles. When all three gimbal angles are zero, the body set UBX, UBY, UBZ and the platform set UPX, UPY, UPZ coincide. At other times the body set is obtained from the platform set by using the Euiler gimbal sequence: inner, middle, outer, abbreviated I, M, O. These rotations are pictured in Figure 3-28 along with the transformation equation, (I). With these, the autopilot determines the CM unit body triad UBX, UBY, UBZ expressed in platform coordinates, (II).

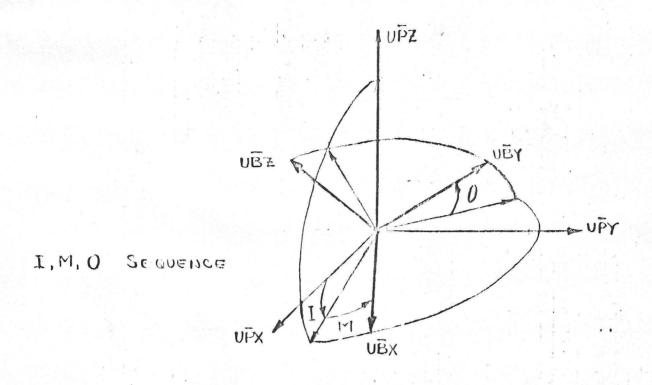
3. 3. 6. 1. 3 Angular Velocity Measurement in CM Body Axes

The body angular velocities, roll p, pitch q, and yaw r are measured in the positive sense about the respective axes UBX, UBY, UBZ and are obtained from the gimbal angle rates using the sequence I, M, O of Figure 3-28. When the gimbal angles are zero, body axes coincide with platform axes and the rates p, q, r are equal to the respective gimbal rates O, I, M. When the gimbal angles are not zero, the body rates are obtained from equations (III) of Figure 3-28.

Measurement of the gimbal rates, and hence of the body rates is not continuous, but occurs at regular time intervals as indeed do all calculations done by the CMC. The basic computation cycle for Navigation and Guidance is 2 seconds. Body rates must be determined more frequently, and a computation cycle of 0.1 second is used. Thus each interval  $\Upsilon = 0.1$  sec, the gimbal angular rates required by (III) are given by normalized CDU angle differences as 0 /180, 1 /180, M /180 and yield p /180, q /180, r /180. These body rates are exact only if the angular accelerations are zero. A better estimate is obtained from the measurements by the simple algorithm, given here and in Figure 3-36.

$$\dot{\Theta}_{\gamma} = \Delta\Theta_1 + \left(\frac{\Delta\Theta_1 - \Delta\Theta_0}{2}\right)$$

where the subscript <u>0</u> refers to the previous value of a difference and <u>1</u> refers to the present value. Since constant acceleration jet thrusters are located so as to generate body rates, the averaging is done in terms of the body rates using the values given by the equations



$$\begin{pmatrix} UBX \\ UBY \\ UBZ \end{pmatrix} = \begin{pmatrix} 1 & 0 & 0 \\ 0 & C0 & S0 \\ 0 & -S0 & C0 \end{pmatrix} \begin{pmatrix} CM & SH & O \\ -SM & CH & O \\ 0 & 0 & 1 \end{pmatrix} \begin{pmatrix} CI & O & -SI \\ O & I & O \\ SI & O & CI \end{pmatrix} \begin{pmatrix} UPX \\ UPZ \\ UPZ \end{pmatrix}$$

BODY TRIAD EXPRESSED IN PLATFORM COORDINATES

BODY ANGULAR VELOCITIES.

WHERE SME SIN M BTC.

(III)

(I)

(11

FIG 112 PLATFORM TO BODY TRANSFORMATION

(III). The algorithm is exact when the angular acceleration is constant.

# 3.3.6.1.4 CM Body Attitude

The CM attitude angles are defined relative to the trajectory triad -UVA, UYA, UNA and the body triad UBX, UBY, UBZ. Both are expressed in platform coordinates. The defining Euiler sequence R,  $\alpha$ ,  $\beta$  is pictured in Figure 3-29 along with the transformation equation (I). When all angles are zero, UBX points along the negative relative velocity vector away from the heat shield, and UBY lies along the normal to the trajectory plane. UBZ lies in the trajectory plane and points along the lift vector. The condition R = 0 corresponds to positive lift or lift up. Rotation of the body triad by rolling about UBX (-UVA) through the angle R, causes the lift to have a component out of the trajectory plane. The direction of UBZ can be called UL since it points in the direction of the lift vector. The direction of UBY can be called YBN since for any R it points in the nominal direction for Y body. That is, it is normal to the relative velocity, lying in the UNA, UYA plane. The second rotation is pitch about YBN (UBY) through the angle  $\alpha$ , shown negative in the picture. The new direction of UBX is called XBN and represents the nominal X body. The last rotation is yaw about UBZ through the angle  $\beta$ . The result of the final rotation is to locate the actual CM unit body triad UBX, UBY, UBZ relative to the trajectory triad - UVA, UYA, UNA in terms of the Euiler attitude angles R,  $\alpha$ ,  $\beta$ . The equations for calculating the angles R,  $\alpha$ ,  $\beta$ and also their derivatives are summarized in Figure 3-30.

# 3. 3. 6. 2 Functional Behavior of Autopilots

In Entry Guidance, control is based on the components of lift in and normal to the trajectory plane, components that are specified in terms of an angle defined the same as R but having value ROLL C. If the CM is to achieve the desired lift direction, then the X and Y body axes must lie along XBN, YBN. This condition is met when  $\beta = 0$ . Thus one property of Entry autopilot is to keep  $\beta = 0$  and the other is to have the actual vehicle roll angle R agree with the desired value ROLL C.

Within the atomosphere, aerodynamic forces tend to keep  $\beta$  essentially zero, and  $\alpha$  assumes a trim value  $\alpha_{\rm t}^{\prime}=-22^{\rm O}$  so that attitude control is required only in roll.

es i somen a con la cadratos

UER VBN

Rαβ SEQJENCE

XBH

UEX

-α

LOCAL

HOROZONIAL

PLANE

FIG. 相外 CM BODY ATTITUDE ANGLES - DEFINITION

38.3

COMBINE MATRICES IN FIG. 1.32 INTO SINGLE MATRIX, REARRANGE TO BE ANALOGOUS TO CALCGA MATRIX, THEN BODY ATTITUDE ANGLES ARE OBTAINED FROM

YBN = UNIT ( UVA x UBZ)

CR = YBN · UVA , SR = YBN · UNA

Sd = - UBZ · UVA

SP = UBX · YBN , CP = UBY · YBN

FOR UPDATING ANGLES ROOB BETWEEN MATRIX CALCULATIONS, USE THE RATE EQUATIONS

$$cd \dot{R} = p c\beta - q s\beta$$

$$\dot{d} = p s\beta + q c\beta$$

$$\dot{\beta} = r - \dot{R} sd$$

FIG. COMPUTATION OF BODY ATTITUDE ANGLES

DIAKSFAMET 294

As the CM rolls, UBY should remain normal to UVA. When  $\beta$  =0 and  $\alpha$  is constant, such a maneuver causes UBX to describe a cone of angle  $2\alpha$  about -UVA while UBZ similarly describes a cone about +UVA. This condition of rolling about the velocity vector, or making a coordinated roll, is equivalent to requiring that the CM angular velocity vector: lie along -UVA. In the atmosphere then, coordinated roll maneuvers are enforced by the angular rate autopilots already required to damp pitch and yaw body rates caused by dynamic behavior of the CM.

Figure 3-29 shows that for the nominal case,  $\beta = 0$ , only roll and yaw rate are present if the body angular velocity is to lie along -UVA. Further, this angular velocity is equal to the rate of change of roll angle R. A third property of the atmospheric autopilot is that it enforce the proper proportionality between yaw and roll rate necessary to produce coordinated roll.

Outside the atmosphere the three angles R,  $\alpha$ ,  $\beta$  could be used to arbitrarily orient the CM. However, the policy is to hold R to the last specified value ROLL C, and the maintain trim attitude of the vehicle relative to VREL by holding  $\alpha = \alpha_t = -22^{\circ}$  and  $\beta = 0$ .

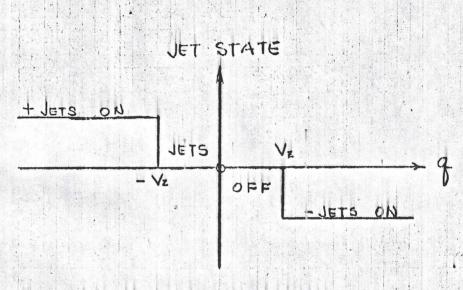
3.3.6.3 Pitch and Yaw Digital Autopilot

3.3.6.3.1 In Atmosphere (> 0.05g)

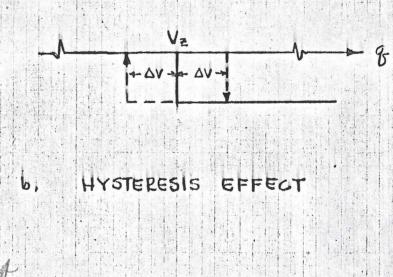
In the atmosphere mode, rate damping is used to main-yain pitch and yaw angular velocities within prescribed limits. The digital autopilots perform this function in a discontinuous manner by producing a control action each time new input data are available. Thus, the pitch and yaw autopilots are exercised each .1 second immediately following the measurement of angular velocity.

3. 3. 6. 3. 1. 1 Pitch Damper DAP

The pitch axis autopilot for the CM performs rate damping action using the simple logic of Figure 3-3/c. As long as sampled pitch rate is less than  $2^{\rm O}/{\rm sec.}$ , no action is taken and the jet thrusters are activated to produce opposing torque. The jets remain firing until the pitch rate is observed to be less than  $2^{\rm O}/{\rm sec.}$  Because of the digital nature of rate measurement, jet switching is based on the value of the rate at the time of measurement, which in general will not be exactly  $2^{\rm O}/{\rm sec.}$  Hysteresis, a result of the 0.1 sec. interval between control actions, is present at the nominal  $2^{\rm O}/{\rm sec.}$  switching discontinuity and prevents the information of high



a. PITCH RATE DAMPER



The state of the s

FIG. 24 PITCH RATE AUTOPILOT

a newsparagement as har statuent

frequency limit cycles in presence of pitch acceleration. The hysteresis zone is of variable width and can be as large as twice the total  $\Delta V$  achievable by the jets in one sample time (Figure 3-3/b.).

# 3.3.6.3.1.2 Yaw Damper DAP

The yaw axis requirements are somewhat different than pitch. The CM must roll about the velocity vector in order to maintain nominal trim attitude and to keep the stagnation point in a fixed region on the heat shield. Figure 3-32 illustrates the situation in the UBX, UBZ plane. The UBY component of VREL is kept essentially zero by the aerodynamic force so that the velocity shown can be considered to be the total velocity VREL. In the coordinated turn illustrated, the angular velocity vector lies along -VREL if the angular velocities in yaw and roll are related by

$$r = -p \tan |\alpha_t|$$

if pitch angular velocity  $\ q=0$  . Even when the latter condition is not exactly fulfilled, the equation is still useful.

During periods of angular acceleration in roll, a corresponding acceleration in yaw is quickly established and supported by aerodynamic forces. We utilize these forces to encourage the CM to make a coordinated turn. The bulk of the work is done by the aerodynamics with the yaw jet thrusters operating only as needed according to Figure 3-33. The jets fire when necessary to keep the vehicle within the central band. This allows r to be proportional to p and at the same time keeps yaw oscillations damped to within  $+2^{\circ}/\text{sec}$ .

The yaw damper operates at the .1 sec. sample interval established by the angular velocity measurements and exhibits the same switching boundary hysteresis as the pitch damper.

# 3.3.6.3.2 Outside Atmosphere (< 0.05 g)

Out of the atmosphere all three autopilots are attitude control systems. The desired values for R,  $\alpha$ ,  $\beta$  are ROLL C,  $\alpha_{\rm t}$ , 0 where ROLL C is the last value provided by Entry Guidance and  $\alpha_{\rm t}$  is nominal trim angle -22°.

The roll system will continue to operate in its usual way to hold the specified attitude. The pitch and yaw channels now have the added task of maintaining attitude but unlike the predictive roll system, will operate as sampled systems.

YAW

RATE

3636

AUTOPILOT

BAIRUSFATHER 394

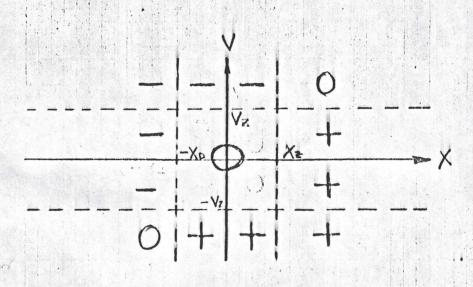
Both pitch and yaw use the same phase plane logic.

Consider the pitch damper. In Figure 3-34 $\alpha$ , X =  $\alpha$  com - $\alpha$ represents the error and V represents the velocity  $\alpha$ and V are to be kept within specified limits: the range (-Vz, Vz) for V and the range (-Xz, Xz) for X. Within the area bounded by these limits, jet thrusters are turned off. If |X| < Xz, but V > z+Vz, negative jets are desired and are indicated by the minus sign. Similarly, if V exceeds the lower bound, positive jets are required. If |V| < Vx, and X > +Xz, positive jets are required to establish a positive velocity to reduce the error X. As indicated, the positive jets remain on until at a sample interval V has exceeded Vz. The zero means turn all jets off. The positive velocity causes the point (X, V) to drift leftward. At a subsequent sample instant, X is observed to be less than Xz but V>+Vz and negative jets are actuated. In this manner arbitrary initial conditions (X, V) of reasonable value are caused to enter the central box. As the vehicle encounters air, there is coupled motion in pitch and yaw and the phase trajectory under acceleration becomes more sinusoidal then parabolic. However the jet select logic of Figure 3-34a will damp' oscillations.

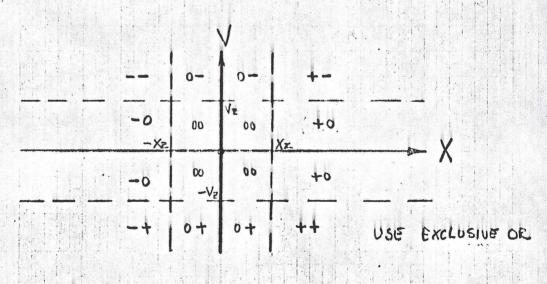
In actual operation both X and V are separately tested against limits. A tag is set to zero if the limit is not exceeded, and to + 1 bit depending on the sign of the limit exceeded. In Figure 3-34b, the pairs of signs refer to the result of bothtests, where the two tags are shown in the order (X, V). The tags are combined in an 'exclusive or' using ones complement arithmetic to generate the single tags shown in Figure 3-34b. The resultting tag is used to select the proper jet code from a list for the axis being controlled. When a jet code is executed, only the jets designated by the code are fired or are left on if already firing. All other jets are quenched. A tag of value zero quenches any jets that might be firing.

For the yaw system, the error is  $X = \beta_{com} - \beta$  and  $V = \beta$ .  $\beta_{com} = 0$  unless in 'Hold' mode. The block diagram gives the calculations in detail starting with Figure 3-35 and through Figure. 3-39.

15, 5



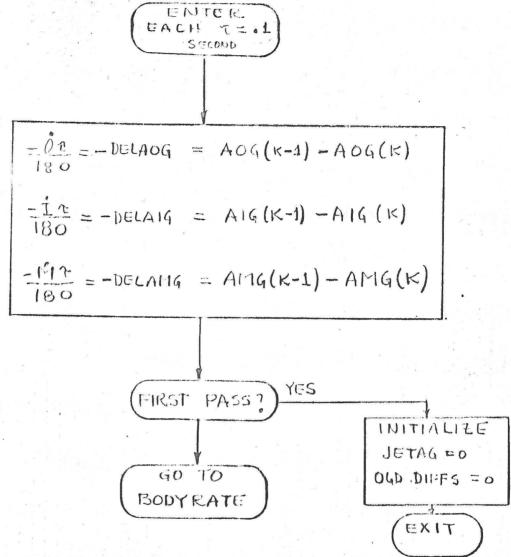
a. JET SELECT LOGIC



hi Micro Logic

FIG PITCH YAW ATTITUDE PHASE PLANE LOGIC

READGYMB

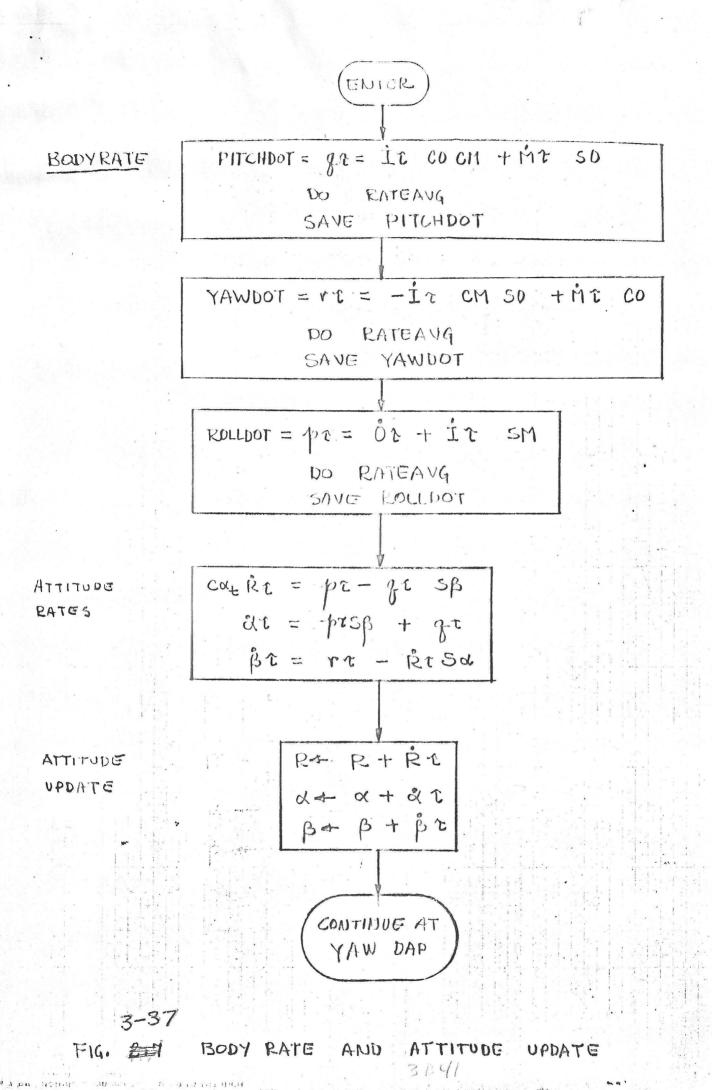


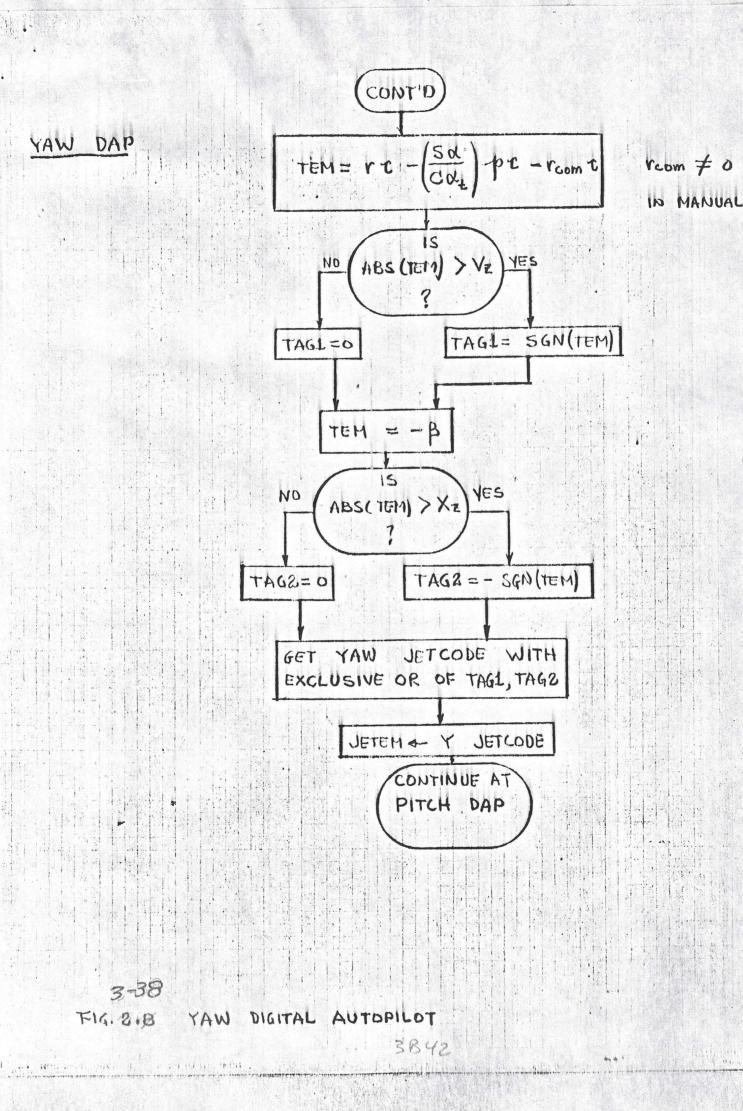
3-35 FIG. 2.5 READ GIMBAL ANGLES

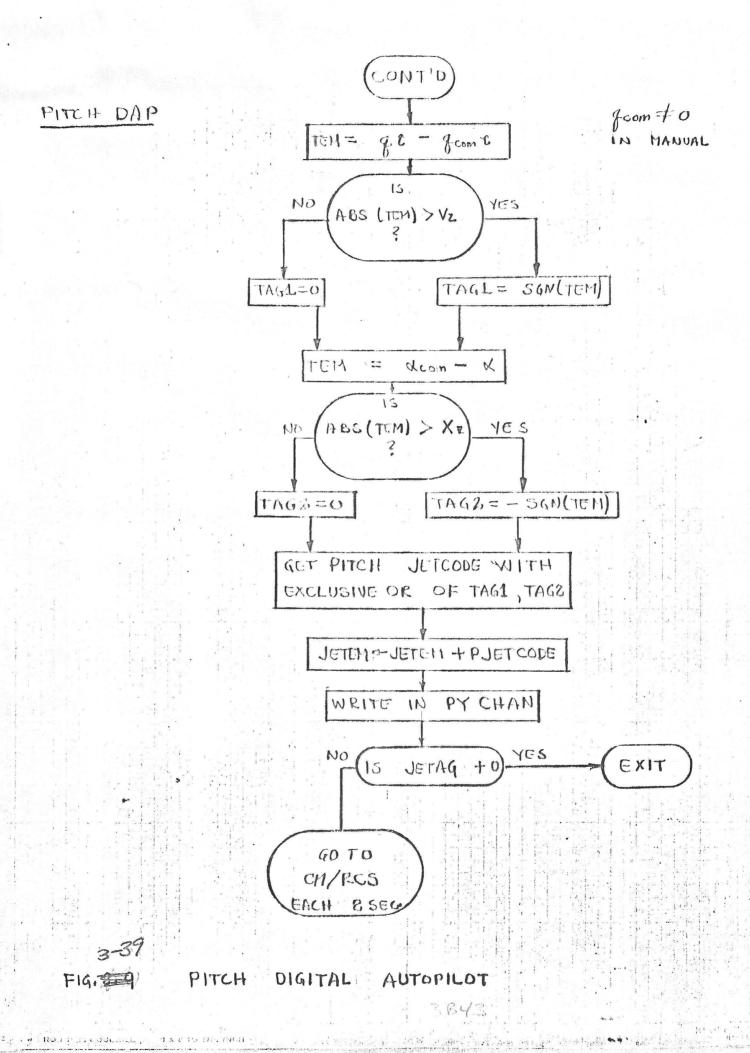
DELV ANG = DELV + (DELV - OLD DELV)/2

(RETURN)

3-36 FIG. 届 BODY RATE AVERAGING ALGORITHM 3840 16 FA TAUS! - 134







In the atmosphere the  $\alpha$   $\beta$  attitude calculations are allowed to continue and provide continuous attitude information. The attitude control loops are disabled since the aerodynamic forces keep both  $\alpha$  and  $\beta$  errors well below the threshold. This is insured in pitch even in the transonic region by setting  $\alpha$  com to the value of  $\alpha$  at update time, instead of to  $\alpha$ . The vector calculations of Figure 3-30 provide a fresh evaluation of attitude each 2 sec. Attitude is integrated in between times. The block diagram CM/POSE, of Figure 3-40 and Figure 3-44, apply.

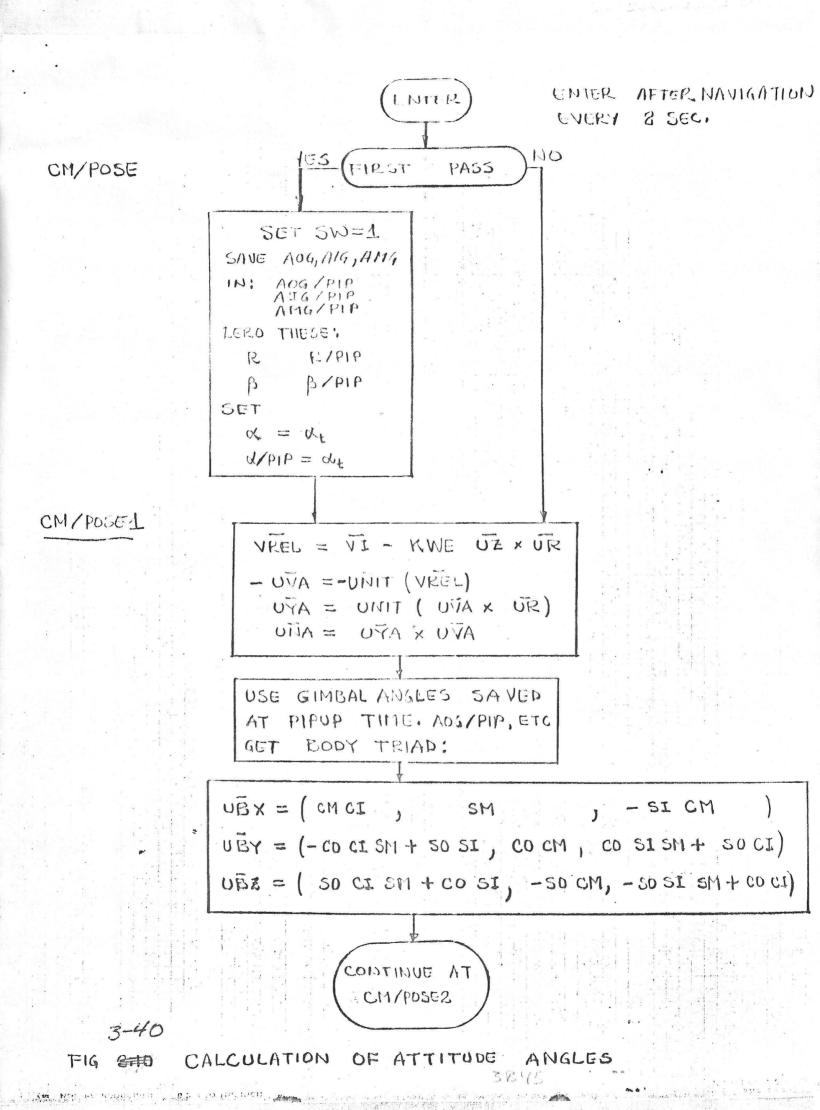
# 3. 3. 6. 4 Roll Attitude Control

Every 2 seconds during the entry phase, Entry Guidance provides a roll command ROLL C. It examines vehicle current position and velocity and also landing point position, decides on the proper orientation of the lift vector, and generates the commanded roll attitude of the CM necessary to achieve that orientation. The roll autopilot uses the command ROLL C, the present roll attitude R and roll rate p to generate firing times for the jet thrusters. In general, there are three time intervals generated each 2 seconds. Two are thrust durations, and one is quiescent duration. Having obtained the time intervals, there remains the bookkeeping task of implementing the result of the calculations.

Bookkeeping consists of maintaining a short list containing pairs of entries: a jet thruster selection code and a time duration. The jet code specifies which jets are to be turned on or off and the time duration specifies the interval until the next change in jet activity occurs. Bookkeeping is carried out with the aid of the TIME 3 preset timer (WAITLIST) in the CMC. The list contains only entries which can be executed within a single two second interval.

# 3.3.6.4.1 Logic in Phase Plane with Control Line

The roll model for the CM is a simple inertial. This makes the phase plane an attractive tool because motion is described by straight lines and parabolas. The (X, V) phase plane of Figure 3-42 is used where X = ROLL C -R and V = R. When the CM has the commanded attitude, both X and V are zero and the point (X, V) representing the state of CM motion is at the origin. While not necessary to the understanding of the roll DAP operation, bear in mind the fact that roll rate p is related to R by cos a in a coordinated roll. The DAP assumes coordinated roll obtains and the trim value



(cont,t)

CM/POSE2

OBTAIN ATTITUDE ANGLES, VALID AT

YEN = UNIT ( UVA x UBZ)

CR = YEN. UTA, SR = YEN. UNA

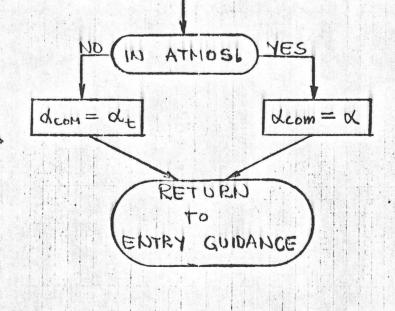
SW = - UEZ. UVA

SB = UEX. YEN, CB = UEY. YEN

OBTAIN NEWR, NEW &, NEW B

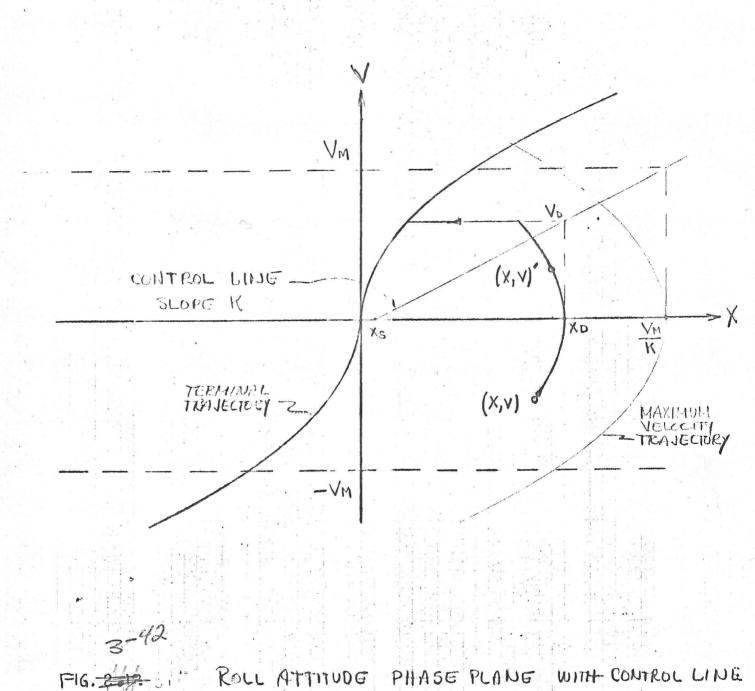
CORRECT FOR INCREMENT SINCE PIPUP TIME.

 $R \rightarrow NEWR + R - R/PIP$   $\alpha \rightarrow NEW\alpha + \alpha - \alpha/PIP$  $\beta \rightarrow NEW\beta + \beta - \beta/PIP$ 



3-41

FIG. EM ATTITUDE CALCULATION, CONT'D.



Duce Spacen

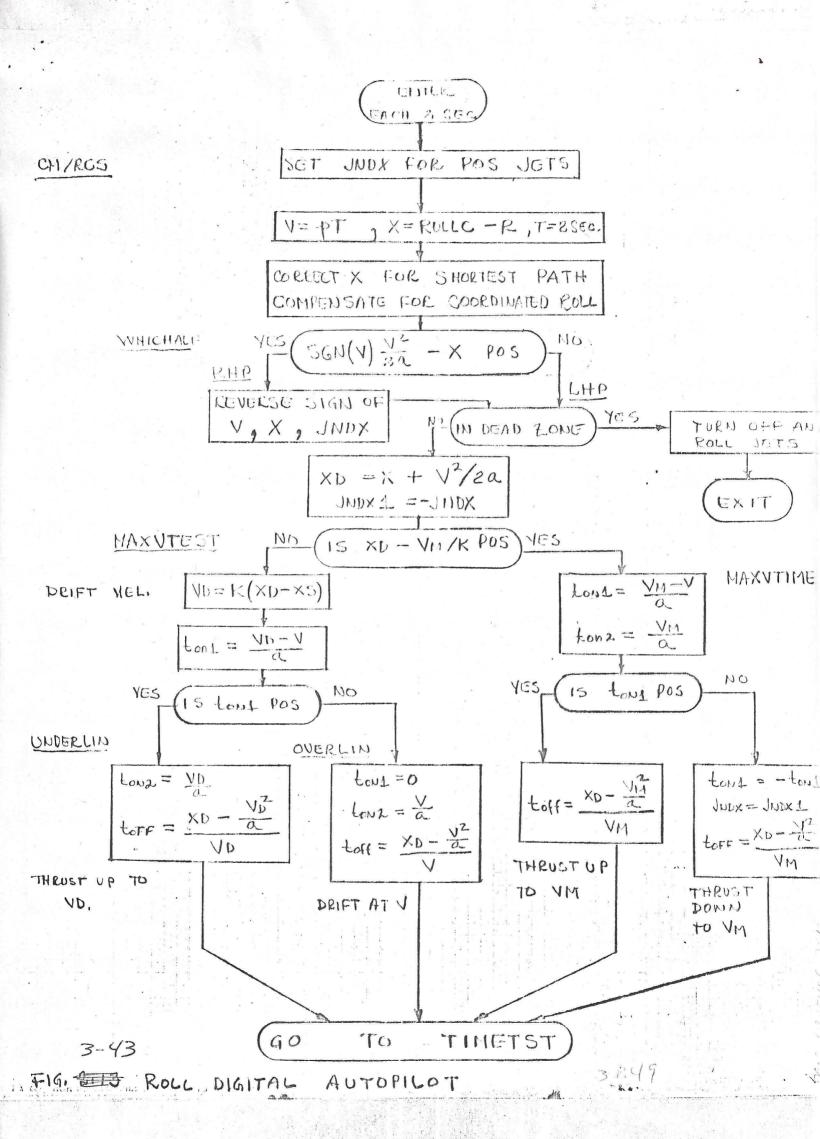
T. Burne 11 in

 $\cos \alpha_{\rm t}$  is used to scale one axis of the phase plane so that velocity V and the change in error  $\Delta {\rm X}$  are compatable. Incompatability results in overshooting the origin and costs fuel.

The salient feature of the control technique is the use of a line of slope K to determine the rate with which an error is reduced. This has several advantages. The fundamental and motivating one is that fuel consumption ( $\Delta V$ ) becomes somewhat proportional to error. The CM responds rapidly to large error and proportionately slowly for small error. The slope K is, chosen as small as is consistent with the rate of response needed during Entry. A by-product is that the equations involved are simplified. Furthermore, this system will efficiently follow ramp inputs by establishing the necessary rate.

In operation, the roll system has initial values (X, V), marked in Figure 3-42. The system will calculate the trajectory indicated. Positive jet acceleration causes the parabolic path shown. Jets remain on until the specified drift velocity Vd is reached, at which time the jets are turned off. The trajectory drifts at constant velocity Vd until it intercepts the terminal trajectory. Negative jets are turned on and remain on until the origin is reached. The drift velocity is defined geometrically by the control line as shown in the picutre. Were the initial point (X, V), the parabola would be extended back to the X axis to define XD, and the same geometry applied. If the trajectory from (X, V) to the origin is unable to be completed within the interval T, it does not matter. The roll system computes a completely new trajectory each T seconds from whatever (X, V) it finds. In this way, changes in command, noise in measurement, and other disturbances are always taken into account.

Because of symmetry, the control equations are instrumented only for the rhp, where the plane is bisected by the terminal trajectory. To determine which half plane contains (X, V), evaluate the equation for the terminal trajectory at the initial point (X, V). If the point is in the lhp, reverse the sign of variables to reflect about the origin into rhp. If the point (X, V) is to the right of the maximum velocity trajectory through X = VM/K, then thrust up (or down) to V = VM. If the point is to the left, and below the control line, thrust up to Vd. But if the point is to the left and over the control line, keep the present velocity as the drift velocity if Vd <V. In all cases where the point (X, V) lies to the left of the maximum velocity trajectory,



calculate the complete trajectory to the origin. The equations are summarized in the flow chart.

The result of the trajectory calculations are three time intervals:

ton1 the time to thrust from V to Vd (or VM)
toff the time required to drift at Vd (or VM)
ton2 the time to origin or terminal trajectory from Vd
(or VM)

Intervals ton1 and ton2 are firing intervals and toff is quiescent interval. The attitude system can generate three, two, one or none of the time intervals. None occurs if the point (X, V) lies within the deadzone at the origin. Jets are quenched, if on, and the program exits.

# 3.3.6.4.2 Timetest and Jetcall 1

The program portion named TIMETEST, Figure 3-44 examines the three possible time intervals and prepares to execute them in sequence if

- 1. no interval is less than two centiseconds
- 2. the sum of the intervals plus time used in computation is less than two seconds.

Time intervals which fail to meet the above requirements are ignored, and the jet codes corresponding to such intervals are appropriately executed.

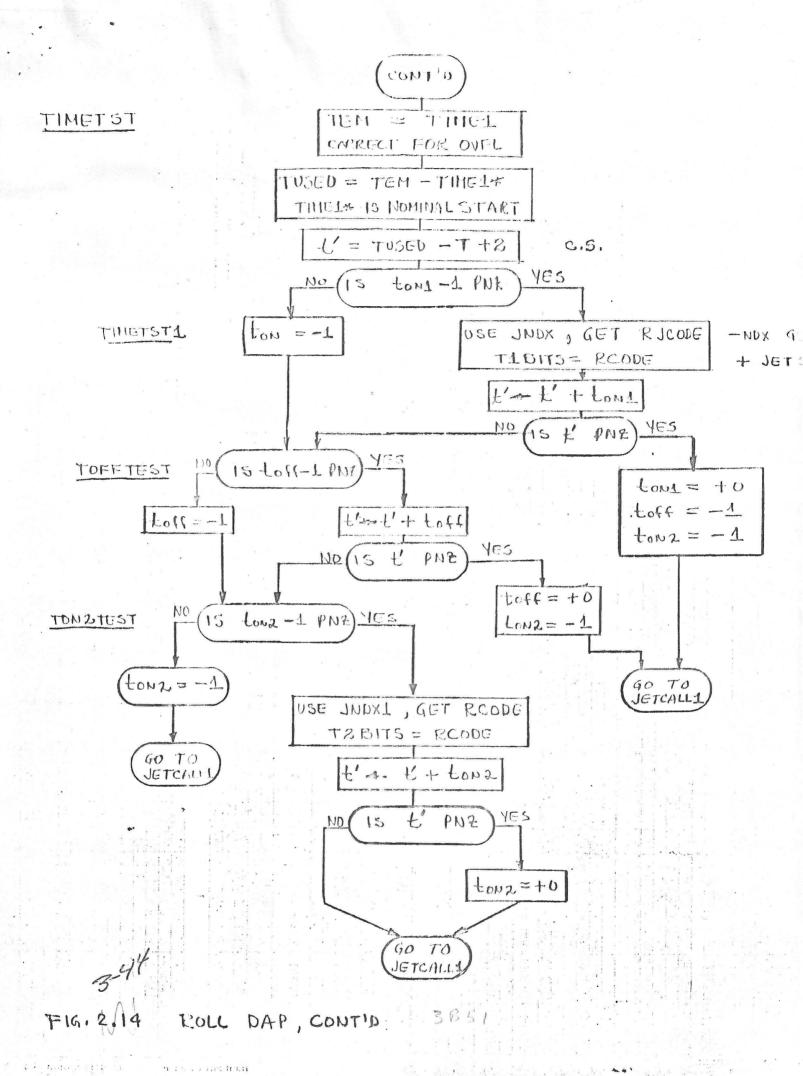
The program portion named JETCALL1, Figure 3-45 performs the actual bookkeeping with the jet list, executing the jet codes and establishing WAITLIST calls as required.

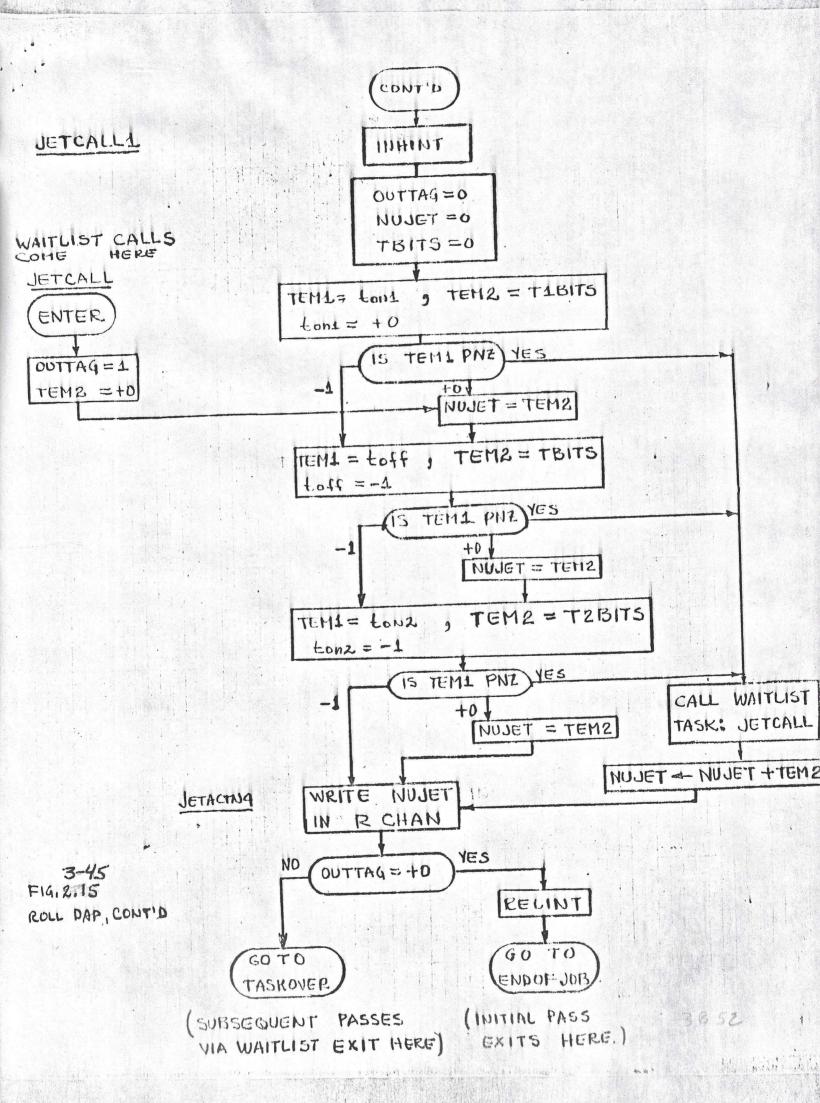
# 3.3.6.5 Entry DAP Implementation

In the body of this section, the various parameters of the system have been designated only by letters. At this time simulations are not sufficiently definative to establish firm numbers for the parameters; however, the nominal values in Table 3-7, which are being used in the simulations, are probably quite good.

3.3.7 Interface between DAP and other Computer programs such as Steering and Constant Heat Load rotation programs

(To be provided)





# Table 3-7

# CONSTANTS - MOMINAL VALUES

| $\circ$ | 200  |
|---------|------|
| ROLL    | DAP  |
| VC Con  | 0171 |
| -       |      |

|                | ERROR PEADBAND RATE DEADBAND RATE LIMIT                                     |                | ± 4°<br>± 2°/500.<br>± 20°/sec.                 |
|----------------|---|----------------|---|
|                | ACCELERATION (DUAL)   | 154576m) a     | 9.1°/sec <sup>2</sup><br>4.55°/sec <sup>2</sup> |
|                | CONTROL LINE SLOP<br>UPDATE TIME<br>CONTROL LINE INTERCEP<br>(DEPENDS ON K) | T              | 0.22<br>2 sec.<br>-1.9                          |
| PITCH, YAW D.  | MP  |                |   |
|                | ERROR DEADBAND RATE DEADBAND  |                | ± 4°<br>± 2°/sec.                               |
|                | COORDINATED YAW MAX R   | ATE = ±VM tand | ~ ±8°/sec.                                      |
| NOT GXPLICITLY | PITCH ACCELERATION  | (DUAL)         | +11.3 /sec2 -7.7 %sec<br>5.6 -3.9               |
| used.          | YAW ACCELERATION  | (DUAL)         | + 12 %sec2 -12 %sec2 + 6 + 6                    |
|                | UPPATE TIME   | r              | 0.1 Sec.  |
|                | TRIM ANGLE OF ATTA  | ck X           | -22°  |

# 3.4 Digital UPLINK to CMC

By means of the CMC UPLINK, the ground can insert data or instruct the CMC in the same manner normally performed by the crew using the DSKY keyboard. The CMC will be programmed to accept the following UPLINK inputs:

- 1) LIFTOFF (backup to discrete input)
- 2) CMC CLOCK ALIGNMENT
- 3) CSM POSITION and VELOCITY data (provides ground capability to update CSM STATE VECTOR data in the CMC.)
- 4) LEM POSITION and VELOCITY data (provides ground capability to update LEM STATE VECTOR data in the CMC.)
- 5) EXTERNAL DELTA-V
- 6) TIME UPDATE (provides ground capability to update CMC clock and DTEPOCH)

Operational procedures governing the use of these Uplink inputs must be developed to ensure proper operations within program constraints.

All information received by the CMC from the Uplink is in the form of keyboard characters. Each character transmitted to the CMC is triply redundant. Thus, if C is the 5-bit character code, then the 16-bit message has the form:

#### 1 CCC

when  $\overline{C}$  denotes the bit-by-bit complement of C. To these 16 bits of information the ground adds a 3-bit code specifying which system aboard the spacecraft is to be the final recipient of the data and a 3-bit code indicating which spacecraft should receive the information. The 22 total bits are sub-bit encoded (replacing each bit with a 5-bit code for transmission.) If the message is received and successfully decoded, the receiver on-board will send back an 8-bit "message accepted pulse" to the ground and shift the original 16 bits to the CMC (1  $\overline{CCC}$ ).

Any ground command sequence via the uplink may be duplicated by the astronaut via the keyboard. All uplink words given in this section are in the form transmitted from the uplink receiver to the CMC. Therefore they do not contain the vehicle or sub-system addresses added on by the ground facilities. For the purpose of this section, the following definitions hold:

- 1) 1 uplink word = 1 character
- 2) 5 characters or uplink words = contents of one CMC register
- 3.4.1 LIFTOFF to send the backup liftoff discrete to the CMC the following words should be sent via the uplink:

| Binary Uplink Word  | Equivalent Keyboard Character |  |  |
|---------------------|-------------------------------|--|--|
| (1 CCC)             | (C)                           |  |  |
| 1 10001 01110 10001 | . VERB                        |  |  |
| 1 00111 11000 00111 | 7                             |  |  |
| 1 00101 11010 00101 | 5                             |  |  |
| 1 11100 00011 11100 | ENTER                         |  |  |

3.4.2 CMC CLOCK ALIGN - to align the CMC clock two procedures are required. To set the CMC clock to a specific value, the following uplink words must be sent.

| Binary Uplink Word  | Equivalent Keyboard Character |
|---------------------|-------------------------------|
| (1 CCC)             | (C)                           |
| 1 10001 01110 10001 | VERB                          |
| 1 00010 11101 00010 | 2                             |
| 1 00101 11010 00101 | 5                             |
| 1 11111 00000 11111 | NOUN                          |
| 1 00001 11110 00001 | i                             |
| 1 00110 11001 00110 | 6                             |
| 1 11100 00011 11100 | ENTER                         |

This must be followed by +00XXX ENTER +000XX ENTER +0XXXX ENTER. Where the first three X's are decimal digits representing the hour, the next two X's represent the minutes, and the last four X's represent the centiseconds (.01 sec) to be loaded into the CMC clock.

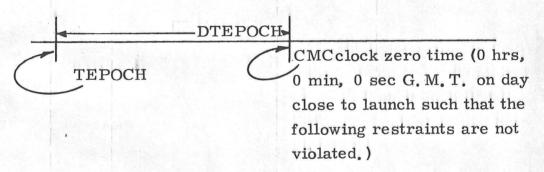
Since there are uncertainties in time of transmission, et certera, it is anticipated that a time increment may be needed. To increment the CMC clock, the following uplink words must be sent.

| Binary Uplink Word  | Equivalent Keyboard Character |
|---------------------|-------------------------------|
| (1 CCC)             | (C)                           |
| 1 10001 01110 10001 | VERB                          |
| 1 00101 11010 00101 | 5                             |
| 1 00101 11010 00101 | 5                             |
| 1 11100 00011 11100 | ENTER                         |

This must be followed by ±00XXX ENTER ±000XX ENTER ±0XXXX ENTER where the X's represent the hours, minutes, and centiseconds to be added to or subtracted from the CMC clock. If the increment is positive (i.e. to increase the clock), all the signs must be positive; if the increment is negative, all the signs must be negative.

The CMC must have had the latitude, azimuth and time DTEPOCH described below, loaded as three double-precision quantities during erasable memory initialization. The CMC uses these quantities to generate the matrix which relates the reference inertial co-ordinate system to the stable member co-ordinate system.

The CMC will be zeroed at 00 00 00 G. M. T. and DTEPOCH will reference the CMC zero time to TEPOCH. When the CMC detects lift-off, it will add the contents of the CMC clock to DTEPOCH and zero the clock. In this way the L/O time will be given G. M. T. (the reading of the CMC clock at L/O) and all other times will be in elasped time from lift-off since the CMC clock was re-zeroed at L/O.



The following restraints must be observed on the magnitudes of the times shown above.

- 1) DTEPOCH + CMC clock at guidance reference release must be less than 2<sup>28</sup>C.S. since the CMC must use this time to determine the inertial platform co-ordinates at guidance reference release.
- 2) CMC Clock during the flight must be less than 2<sup>28</sup>C.S. to prevent overflow.
- 3.4.3 CSM STATE VECTOR to begin a CSM STATE VECTOR update on flight 278 the following 4 words should be sent via uplink.

| 1 10001 01110 10001 |     | VERB  |
|---------------------|-----|-------|
| 1 00111 11000 00111 |     | 7     |
| 1 00110 11001 00110 | i i | 6     |
| 1 11100 00011 11100 | )   | ENTER |

The ground station should then await confirmation via Downlink that the CMC is in Major Mode 07.

In Major Mode 07 the CMC will accept a complete ground STATE VECTOR update in the format to be described below.

The data itself will take the form of three (3) double precison components of position, three (3) double precision components of velocity, and double precision time. The position and velocity components should be given in navigation co-ordinates (see Sec. 6) and the time should be in the time of the "fix" referenced to CMCCLOCK ZERO. The data must be sent in the following sequence:

| octal      |
|------------|
| identifier |
|            |

| <u>identifier</u> |       |                                 |       |
|-------------------|-------|---------------------------------|-------|
| 1                 | XXXXX | (most sig. part of X position)  | ENTER |
| 2                 | XXXXX | (least sig. part of X position) | ENTER |
| 3                 | XXXXX | (most sig. part of Y position)  | ENTER |
| 4                 | XXXXX | (least sig. part of Y position) | ENTER |
| 5                 | XXXXX | (most sig. part of Z position)  | ENTER |
| 6                 | XXXXX | (least sig. part of Z position) | ENTER |
| 7                 | XXXXX | (most sig. part of X velocity)  | ENTER |
| 108               | XXXXX | (least sig. part of X velocity) | ENTER |
| 118               | XXXXX | (most sig. part of Y velocity)  | ENTER |
| 128               | XXXXX | (least sig. part of Y velocity) | ENTER |
| 138               | XXXXX | (most sig. part of Z velocity)  | ENTER |
| 148               | XXXXX | (least sig. part of Z velocity) | ENTER |
| 158               | XXXXX | (most sig. part of time from    |       |
|                   |       | CMC clock zero)                 | ENTER |
| 168               | XXXXX | (least sig. part of time from   |       |
| . 0               |       | CMC clock zero)                 | ENTER |

where each "X" and "ENTER" above represents an uplink word. If, for some reason, the ground wishes to resend any 5 uplink word group before the ENTER associated with that group has been transmitted, the following "CLEAR" word would be sent

1 11110 00001 11110

and the 5 word group retransmitted.

If the ground station wishes to terminate the load at any time, the following 4 uplink words must be sent.

| Binary Uplink Word  | Equivalent Keyboard Character |  |  |
|---------------------|-------------------------------|--|--|
| (1 CCC)             | (C)                           |  |  |
| 1 10001 01110 10001 | VERB                          |  |  |
| 1 00011 11100 00011 | 3                             |  |  |
| 1 00100 11011 00100 | 4                             |  |  |
| 1 11100 00011 11100 | ENTER                         |  |  |

which will return the CMC to the mode it was in before the update was initiated.

After the ENTER associated with the least sig. part of time, the ground station must verify via Downlink that the CMC has correctly received the navigational update before sending the following 4 uplink words to signal the CMC that it can use the data in guidance computations.

| 1 | 10001 | 01110 | 10001 |  | VERB  |
|---|-------|-------|-------|--|-------|
| 1 | 00011 | 11100 | 00011 | description of the control of the co | 3     |
| 1 | 00011 | 11100 | 00011 |  | 3     |
| 1 | 11100 | 00011 | 11100 |  | ENTER |

If the STATE VECTOR update is associated with a pre-thrusting mode, the entire load must be completed at least - - sec\* before SPS ignition.

If, during the final verification period after the ENTER associated with the least sig. part of time, it is found that the data in the CMC are not correct, the ground station may change the load in either the following ways:

1) If only a few parts must be changed the ground station should send the octal identifier of the part to be changed, (I.E., if the least sig. part of the Y velocity were to be changed)

| 1 | 00001 | 11110 | 00001 | i     |  |
|---|-------|-------|-------|-------|--|
| 1 | 00010 | 11101 | 00010 | 2     |  |
| 1 | 11100 | 00011 | 11100 | ENTER |  |

then the 5 uplink words corresponding to the part to be changed are sent followed by an ENTER. This procedure, octal identifier ENTER et cetera, must be repeated for each part to be changed. When all changes are made and verified via Downlink the VERB 33 ENTER must be sent to signal the CMC that it can use the data.

2) If many parts must be reloaded, the ground station may choose to start the load from the beginning. To do this, as stated above, the ground station must send VERB 34 ENTER which will terminate the load and allow the CMC to return to its pre-update condition.

If the CMC receives an improperly coded word from the uplink receiver during the load (not CCC) it will turn on bit 4 of OUT 1 which is transmitted via Downlink (see Sec. 3.4). When this occurs the ground station should send the following 3 uplink words:

<sup>\* &</sup>quot;Fix" time must be within - - - sec of SPS + X translation time.

| Binary Uplink Word . | Equivalent Character     |
|----------------------|--------------------------|
| (1 CCC)              | (C)                      |
| 1 00000 00000 00000  | (to clear uplink buffer) |
| 1 10010 01101 10010  | ERROR RESET              |
| 1 11110 00001 11110  | CLEAR                    |

The ground station should then begin loading with the first word of the 5 word group it was sending when the alarm condition occurred.

If insufficient time remains to SPS + X translation, the CMC will change it major mode and proceed with the internally computed data.

The scale factors for CMC navigation updating are:

position  $meter/2^{24}$ velocity  $(meter/C.S.)/2^{7}$ "fix" time  $C.S./2^{28}$ (1C.S. = .01 sec)

The CMC is a fixed pt. machine with the pt. just to the left of the most significant bit.

The scaling indicated above will be sufficient to force the 3 components of position and the 3 components of velocity and time to numbers less than one.

To form the double precision quantities ready for coding and transmission the scaled magnitudes of time and each component of position and velocity should be expressed as two binary words as follows:

Each X above represents a binary bit of the appropriate magnitude, the place value of which is indicated below the corresponding X. Once the magnitude of the component is accounted for in the above 28X's, the sign must be considered.

If the component is positive, the words remain as formed; if the component is negative, the "1's complement" of the 2 words is used (all 1's are replaced by 0's and all 0's by 1's.)

The first word is then transformed into a 5 character octal word. The first character is the octal equivalent of the first three bits, the second character is the octal equivalent of the next three bits, et cetera. This word is referred to as the "most significant part" of data in the text above.

Similarly the second word is transformed into a 5 character octal word which is the "least significant part" of data.

Each character must now be coded into a 16 bit uplink word for transmission. A table of the characters and their uplink word follows. (Table 3-1).

## 3.4.4 State Vector Update - LEM

To update the LEM state vector via the uplink the ground should send Verb 70 Enter and then enter the parameters in a manner identical to that described for CSM updates in Section 3. 4. 3.

### 3.4.5 External Delta - V

To load the CSM for an External Delta - V maneuver the ground should send Verb 72 Enter followed by the parameters listed below, in the order given below.

| identi |  |  |
|--------|--|--|
|        |  |  |

| 1   | XXXXX Enter | (most sig.   | part of Tign)  |
|-----|-------------|--------------|--|
| 2   | XXXXX Enter | (least sig.  | part of T <sub>ign</sub> ) part of X component of V <sub>g</sub> ) |
| 3   | XXXXX Enter | (most sig.   | part of X component of Vg)   |
| 4   | XXXXX Enter | `(least sig. | part of X component of V   |
| 5   | XXXXX Enter |              | part of Y component ov $\vec{V_g}$ )                               |
| 6   | XXXXX Enter | (least sig.  | part of Y component of Vg)   |
| 7   | XXXXX Enter |              | part of Z component of $V_g$ )                                     |
| 108 | XXXXX Enter | (least sig.  | part of Z component of Vg)   |
|     |             |              |  |

The ground should use the same procedure for correcting, terminating or accepting this data as outlined for Verb 76 above.

3.4.6 TIME UPDATE - to begin a time update for flight 278 the following four words should be sent via uplink.

| Binary Uplink Word  | Equivalent Keyboard Character |
|---------------------|-------------------------------|
| (1 CCC)             | (C)                           |
| 1 10001 01110 10001 | VERB                          |
| 1 00111 11000 00111 | 7                             |
| 1 00111 11000 00111 | 7                             |
| 1 11100 00011 11100 | ENTER                         |

This must be followed by XXXXX ENTER where the 5 X's represent 5 octal characters which give the time change scaled as C.S./ $2^{14}$ .

A positive value indicates that the clock reading should be decreased and DTEPOCH should be increased, while a negative value indicates that the clock should be increased and DTEPOCH decreased.

TABLE 3-1

| Uplink Word         |
|---------------------|
| 1 10000 01111 10000 |
| 1 00001 11110 00001 |
| 1 00010 11101 00010 |
| 1 00011 11100 00011 |
| 1 00100 11011 00100 |
| 1 00101 11010 00101 |
| 1 00110 11001 00110 |
| 1 00111 11000 00111 |
| 1 01000 10111 01000 |
| 1 01001 10110 01001 |
| 1 10001 01110 10001 |
| 1 11111 00000 11111 |
| 1 11100 00011 11100 |
| 1 10010 01101 10010 |
| 1 11110 00001 11110 |
| 1 11001 00110 11001 |
| 1 11010 00101 11010 |
| 1 11011 00100 11011 |
|                     |

NOTE: It is good operational procedure to end every uplink message with a KEY RELEASE.

### 3.5 CMC Digital Downlink

The CMC digital downlink is only transmitted during high bit rate for mission 278. The CMC digital downlink consists of 50 words per second. Each word consists of 33 bits; the first bit is a "word order code" bit, the next 16 bits are the contents of one 16 bit CMC register and the last 16 bits are the contents of another 16 bit CMC register. (Because the PCM telemetry format requires data in multiples of 8-bit words, the CMC downlink program repeats the last 7 bits from the first 16-bit register, after the 33-bit transmission, in order to "fill" the 40-bit downlink allocation.) In this manner the contents of 100 CMC registers are transmitted in one second.

The downlink format is controlled by an CMC program which loads the next 2 registers to be transmitted into channels 34 and 35. This program is entered on an interrupt caused by an "endpulse" from the telemetry system.

The general CMC downlink format is shown in Table 3-2. Each list contains 50 words (i.e., 100 CMC registers). The first word in any list contains the ID and sync registers and has a word order code bit of zero. The ID register marks the beginning of a list and also identifies which list is being transmitted. The sync register always contains the same 16 bits. These bits (000 0001000111111) are used to sync certain remote site downlink processing equipment. The next 12 words (24 registers) may also change for different lists. These twelve words will have word order code bits of one. Since certain data on the downlink are only meaningful when considered in multiregister arrays (i.e. state vectors) and since the programs which compute these arrays are not synchronized with the downlink program, a "snapshot" is taken of these registers so that changes in their values will not occur while these arrays are being transmitted to the ground. The next 26 words (52 registers) will have word order code bits of one and will probably change from list to list. Since both registers of a word are loaded at the same time, no synchronization problems can arise on double precision parameters. The next 6 words (12 registers) have word order code bits of one and transmit the contents of the latching relays in the DSKY. As such they indicate all displays and certain other functions (i.e. G&N mode switching) handled through the latching DSKY relays. All downlink lists will contain DSPTAB registers in this position in the list.

The next downlink word contains the double precision contents of the CMC clock (i.e. TIME1 and TIME2). This word will have a word order code bit of one and will always retain the position indicated in all lists. The last 4 downlink words (8 registers) contain the contents of channels 11, 12, 13, 14, 30, 31, 32 and 33 in that order. These channels have similar functions to the in and out registers for the Block I computer. Therefore, they will contain the status of a number of discrete interfaces of the CMC. These 4 words will have word order bits of one and

Table 3-2
GENERAL LGC DOWNLINK FORMAT

| Word #   | Word Order Code | Contents of      |                       |
|----------|-----------------|------------------|-----------------------|
|          |                 | 1st register     | 2nd register          |
| 1        | 0               | ID               | sync                  |
| 2        | 1               | ("snapshot" data | a which will probably |
| 13       |                 | change with dif  | ferent lists          |
| 14       | 1               | (data which will | probably change       |
|          | Į.              | with different 1 |                       |
| 39<br>40 | 1               | DSPTAB regis     | ters                  |
| 45       | i               | {                |                       |
| 46       | 1               | TIME 2           | TIME1 '               |
| 47       | 1               | channel 11       | channel 12            |
| 48       | 1               | channel 13       | channel 14            |
| 49       | 1               | channel 30       | channel 31            |
| 50       | 1               | channel 32       | channel 33            |

3.6.2 PCM Telemetry

| Identification | Function                         | Туре                       | Sample<br>Rate/Sec |
|----------------|----------------------------------|----------------------------|--------------------|
| GG 0001 V      | Computer Digital Data            | PCMD (OP)                  | 50 <sup>**</sup>   |
| GG 1040 V      | +120 VDC PIPA Supply             | PCM (OP)                   | 1                  |
| GG 1110 V      | 2.5 VDC Telemetry Bias No. 1     | PCM (OP)                   | 1                  |
| GG 1111 V      | 2.5 VDC Telemetry Bias No. 2     | PCM (FQ)                   | 1                  |
| GG 1201 V      | IMU 28 V 800 CPS 1PCT OD,<br>RMS | PCM (OP)                   | 1                  |
| GG 1331 V      | 3.2K 28V Supply                  | PCM + (OP)                 | 1                  |
| GG 1513 X      | +28 VDC IMU Stby/off             | PCME + (OP)                | 1                  |
| GG 1523 X      | +28 VDC LGC Operate              | PCME + (OP)                | 1                  |
| GG 2001 V      | X PIPA SG Out in Phase           | PCM (OP)                   | 50                 |
| GG 2021 V      | Y PIPA SG Out in Phase           | PCM (OP)                   | 50                 |
| GG 2041 V      | Z PIPA SG Out in Phase           | PCM (OP)                   | 50                 |
| GG 2107 V      | IG Servo Error In Phase          | PCM (FQ)                   | 200                |
| GG 2137 V      | MG Servo Error In Phase          | PCM (FQ)                   | 200                |
| GG 2167 V      | OG Servo Error In Phase          | PCM (FQ)                   | 200                |
| GG 2112 V      | IG 1X Res Out-SINE               | PCM + (OP)                 | 10**               |
| GG 2113 V      | IG 1X Res Out-COSINE             | PCM + <b>(</b> OP <b>)</b> | 10*                |
| GG 2142 V      | MG 1X Res Out-SINE               | PCM + (OP)                 | 10*                |
| GG 2143 V      | MG 1X Res Out-COSINE             | PCM + <b>(</b> OP <b>)</b> | 10*                |
| GG 2172 V      | OG 1X Res Out-SINE               | PCM + (OP)                 | 10*                |
| GG 2173 V      | OG 1X Res Out-COSINE             | PCM + (OP)                 | 10*                |
| GG 2121 V      | IG 1X Res Out-SINE 11 Deg.       | PCM (FQ)                   | 100                |
| GG 2151 V      | MG 1X Res Out-SINE 11 Deg.       | PCM (FQ)                   | 100                |
| GG 2181 V      | OG 1X Res Out-SINE 11 Deg.       | PCM (FQ)                   | 100                |
| GG 2110 C      | IG Torque Motor Current          | PCM (FQ)                   | 100                |
|                |                                  |                            |                    |

<sup>\*</sup>If low bit rate (PCM+) is selected in the LEM T/M system the sample rate on these functions is reduced to 11 sec.

<sup>\*\*(</sup>Refer to Sec. 3.5)

will be in the same position on all downlink lists.

More detailed information will be included as addenda to this document as it becomes available.

## 3.6 Analog Data Telemetry

#### 3.6.1 Types

The inflight information from the Block II CSM G&N system is of four types:

- a) PCM telemetry of the CGC Digital Downlink (described in sec. 3.5),
- b) PCM telemetry of OPERATIONAL (required for all Block II missions) analog measurements, which include the following categories:
  - 1) PCM Pulse-code-modulated analog measurements digitally coded into 8-bit words and transmitted at the normal (51, 2k-c) operational bit rate.
  - 2) PCM+ Flight-critical PCM measurements, which continue to be monitored when the PCM is operated in the "slow-format" mode.
  - 3) PCME PCM monitor of discrete events (1 bit per word).
  - 4) PCME+ Flight critical PCME measurements monitored in "slow-format." mode;
- c) PCM telemetry of FLIGHT QUALIFICATION measurements which is discontinued after the flight qualification requirements are fulfilled.
- d) On board recordings of FLIGHT QUALIFICATION measurements for retrieval by subsequent transmission or by examination after S/C recovery.

#### 3.6.2 Authority

The PCM telemetry and on-board recorder allocations have been defined by NASA in SID 65-1642 (2 March 1966), CSM Measurement Requirements Specification for Block II Spacecraft Apollo CSM System. The G&N measurements are listed below.

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3.6.3 OPERATIONAL PCM Measurements

| Measurement # | Description                   | T/M  | Response S/S |
|---------------|-------------------------------|------|--------------|
| CG 0001V      | Computer Digital Data 40 Bit  | PCMD | 50           |
| CG 1040V      | +120 VDC PIPA Supply DC Level | PCM  | 1            |
| CG 1110V      | 2.5 VDC T/M Bias              | PCM  | 1            |
| CG 1201V      | IMU28V .8KC 1 PCT 0 Deg. RMS  | PCM  | 1            |
| CG 1331V      | 3. 2KC 28V Supply             | PCM  | 1            |
| CG 1513X      | +28V IMU Standby              | PCME | 10           |
| CG 1523X      | +28V CMC Operate              | PCME | 10           |
| CG 1533X      | +28V OPTX Operate             | PCME | 10           |
| CG 2001V      | X PIPA SG Output In Phase     | PCM  | 50           |
| CG 2021V      | Y PIPA SG Output In Phase     | PCM  | 50           |
| CG 2041V      | Z PIPA SG Output In Phase     | PCM  | 50           |
| CG 2112V      | IG 1X Resolver Output SIN     | PCM  | 10           |
| CG 2113V      | IG 1X Resolver Output COS     | PCM  | 10           |
| CG 2117V      | IG Servo Error In Phase       | PCM  | 100          |
| CG 2142V      | MG 1X Resolver Output SIN     | PCM  | 10           |
| CG 2143V      | MG 1X Resolver Output COS     | PCM  | 10           |
| CG 2147V      | MG Servo Error In Phase       | PCM  | 100          |
| CG 2172V      | OG 1X Resolver Output SIN     | PCM  | 10           |
| CG 2173V      | OG 1X Resolver Output COS     | PCM  | 10           |
| CG 2177V      | OG Servo Error In Phase       | PCM  | 100          |
| CG 2300T      | PIPA Temperature              | PCM+ | 1 -          |
| CG 3721V      | Shaft CDU DAC Output          | PCM  | 10           |
| CG 3722V      | Trunnion CDU DAC Output       | PCM  | 10           |
| CG 5040X      | CMC Warning                   | PCME | 10           |

CAL WARNING

# 3.6.4 FLIGHT QUALIFICATION PCM Measurement for CSM 101

| Measurement # | Description                    | T/M  | Response S/S |
|---------------|--------------------------------|------|--------------|
| CG 1211V      | OPTX 28V .8KC 1 PCT 0 Deg. Rms | PCM  | i            |
| CG 2219V      | Pitch Att. Error-CDU DAC Out   | PCM  | 10           |
| CG 2249V      | Yaw Att. Error-CDU DAC Out     | PCM  | 10           |
| CG 2279V      | Roll Att. Error-CDU DAC Out    | PCM  | 10           |
| CG 2301T      | IRIG Temperature               | PCM  | 1            |
| CG 2302X      | IMU Heater Current             | PCME | 1            |
| CG 2303X      | IMU Blower Current             | PCME | ı i          |
| CG 3011V      | Trunnion CDU Fine Error        | PCM  | 10           |
| CG 3021V      | Shaft CDU Fine Error           | PCM  | 10           |
| CG 3140V      | SXT Shaft Tachometer Output    | PCM  | 10           |
| CG 3150V      | SXT Trun Tach Output           | PCM  | 10           |
| CG 3160V      | SCT Shaft Tach Output          | PCM  | 1            |
| CG 3170V      | SCT Trun Tach Output In Phase  | PCM  | 1 1          |
|               |                                |      |              |

# 3.6.5 FLIGHT QUALIFICATION Flight Recorder Measurement for CSM 101

| Measurement # | Description                       | Recoverable B/W<br>CPS |
|---------------|-----------------------------------|------------------------|
| CG 2120C      | IG Torque Motor Current           | 35                     |
| CG 2122V      | IG 1X Resolver Output SIN 11 Deg. | 35                     |
| CG 2150C      | MG Torque Motor Current           | 25                     |
| CG 2152V      | MG 1X Resolver Output SIN 11 Deg. | 25                     |
| CG 2180C      | OG Torque Motor Current           | 20                     |
| CG 2182V      | OG 1X Resolver Output SIN 11 Deg. | 20                     |
| CG 6001D      | Vibration Navigation Base Roll    | 2K                     |
| CG 6002D      | Vibration Navigation Base Pitch   | 2K                     |
| CG 6003D      | Vibration Navigation Base Yaw     | 2K                     |

2 A 15