Felleman (Larson)

MISSION TECHNIQUES MEMO #29A

TO: Distribution
FROM: Malcolm W. Johnston
DATE: May 21, 1969
SUBJECT: "G" Descent Aborts

1. MIT feels that pipa bias can be corrected (by ground monitoring and subsequent compensation adjustment) to an accuracy of about 0.003 ft/sec^2. This number, rather than a 3σ value, should be used in error analysis if MSC agrees to this compensation update philosophy (i.e., always update). See enclosed STG memo #1347 by G. Edmonds.

2. Gyro drift, particularly y axis, is a predominant descent/ascent error source (along with the pipa bias mentioned above). A proposal for reducing the uncertainty in our knowledge of this drift rate prior to PDI is outlined in the enclosed STG memo #1349, by G. Edmonds, dated May 22, 1969. Basically, it involves AOT sightings on the sun via P52 and subsequent comparisons of actual and desired attitudes.

3. An alternate to the present docked PGNCS alignment procedure is described in the enclosed APM #49-69 by R. White, dated May 16, 1969. The method is independent of knowledge of the relative orientation of the two navigation bases, and is possibly more accurate than an AOT alignment.

4. The enclosed Luminary Memo #81, by D. Eyles, reviews the landing guidance dependence on a nominal landing site platform alignment. Deviations from this nominal of up to 5^o are seen to pose no problems.

5. Can control be switched from AGS to PGNCS during ascent?

Ans. Yes! P12, P70, and P71 must be properly initiated at the outset of the ascent. Also, if the AGS targetting differs from the PGNCS the associated transient must be tolerable.

6. PDI and post PDI (no ignition obtained) aborts will utilize a P30 / P40 burn sequence. Can the P30 abort ΔV's and TIG be loaded prior to P63 execution without subsequent overwriting?

Ans. The ΔV's (N81) can be, but the TIG (N33) will have to be re-loaded.

7. The post insertion (P12, P70, or P71) N85 residual velocity display will continue to be updated after shutdown. (i.e., guidance is continually trying to satisfy the cut-off conditions even though the S/C is moving towards apolune). Therefore, residual trimming should be done immediately or this display will become misleading.

8. In P12, P70, and P71 the RR self track enable bit is cleared when the program (average G) is terminated. Therefore, unless another system is controlling the radar, the no track light should come on.

9. An on-going error analysis effort is presently being conducted by B. Kriegsman (Descent) and R. White (Descent aborts and Ascent). Included in the analysis are the effects of individual error sources such as initial platform misalignment, pipa bias etc., initial state vector uncertainties, LR uncertainties, terrain variations, etc. Of particular interest is the determination of a velocity difference (vs MSFN) criterion that represents a failed PGNCS. Mission Simulation Memo #11-69, by B. Kriegsman, dated May 12, 1969, summarizes the initial descent work. Due to its length, it has not been addended, (though it has been distributed under separate cover).

10. A premature DPS failure after an abort (a double failure), followed by execution of an abort stage will result in an APS burn which utilizes a cut-off velocity calculated via the DPS rather than the APS "VHF" polynomial. What errors result at orbit insertion?

Ans. This has an insignificant effect if aborting from hover. At intermediate points in the descent, where the effect is greatest, a horizontal cut-off velocity error of approximately 10 ft/sec results.

11. The ascent programs (P12, 70, and 71) have two thrust level thresholds: a very low level which, if not exceeded, will trigger the thrust fail routine after 10-12 secs, and a higher level (corresponding to about 60% DPS thrust) which must be exceeded before ascent steering replaces attitude hold. Actually, full thrust is requested twice, both before and after auto throttle is requested by the LGC, then steering will commence if the 60% DPS acceleration threshold is exceeded.

12. Pre-PDI aborts for the "F" mission were designed to utilize P40 and the AGS. An alternate technique, utilizing P70 and/or P71, is outlined in the enclosed AG#114-69 memo by G. Cherry, dated March 3, 1969. (Currently, this alternate has been abondoned. If resurrected, MIT has some new reservations concerning its use).

Malcolm W. Johnston

Malcolm W. Johnston

M. Johnston
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MIT/IL
Apollo Guidance and Navigation
System Test Group Memo No. 1347

To: A. Laats

From: G. P. Edmonds, Jr.

Date: 15 May 1969

Subject: LM PIPA Bias Measurement Before Lunar Landing

Reference: 1. Mission Simulation Memo No. 11-69, by B. Kriegsman, "Effect of IMU errors During DOI and PD on Powered Landing PGNCS Performance"

Introduction

Reference 1 shows that the X and Z accelerometer bias are critical parameters in the lunar landing performance, and so an inflight measurement is desired. This memo will estimate how accurate a measurement can be achieved (based on data from Apollo 9) and make recommendations for Lunar Landing flights.

Analysis of Data

Table I is the accelerometer data taken from the LM3 during Apollo 9 and gives calculated biases. The data was obtained on the ground from the downlink. The standard deviation of the measured X accelerometer biases is 0.03 cm/sec^2, and the Z accelerometer is .012 cm/sec^2.[(1)] A variety of measurement times from 180 seconds to 2890 seconds were used and there is no indication of degraded accuracy at the shorter times. It is assumed that this data was taken during times of no RCS thrust firings and low(near orbital) spacecraft rotational rates.

It is apparent that the Apollo 9 accelerometer bias was accurately measured, and so the compensation could have been updated with an accuracy of 0.03 cm/sec^2 on a 1σ basis. Similar results should be expected on lunar landing flights, if the uncompensated biases measured are about the same as those existing in Apollo 9.[(2)] This results should be checked again on the Apollo 10 LM.

---

(1) The Y accelerometer bias standard deviation was also calculated to be 0.028 cm/sec^2.

(2) If large biases exist, somewhat larger measurement errors for short measurement times can be expected. The 2 May 1969 LM5 values are: x=.08, y=+.17, z=.06 cm/sec^2, and the x average uncompensated bias in Apollo 9 was

Recommendation [3]

The following ground procedure is recommended for lunar landing flights[4].

1. Allow 30 minutes for temperature stabilization after operate power is applied..
2. Make three measurements of X and Z accelerometer bias during times when no RCS jet firings are taking place and when S/C rates are low (about orbital rates). The measurements are made by reading the PIPA counters and LGC time on the downlink at two times separated by 180 seconds or more. The bias is then calculated by the formula:

$$\text{Bias}(\text{cm/sec}^2) = \frac{\text{change in contents of PIPA counter (pulses)}^{(5)}}{\text{time difference (secs)}^{(6)}}$$

3. Each measurement should differ by less than 3σ or 0.09 cm/sec^2 from the average. If a greater difference is noted additional measurements should be made until confidence exists that a good average has been obtained.
4. Update X and Z accelerometer compensation to the average value if it differs from the prelaunch load by more than 0.03 cm/sec^2. This update will probably be required. (Always updating at this point is acceptable to simplify procedure
5. Continue to monitor the bias when practical and update again if three measurements indicate a change in bias of 0.09 cm/sec^2 or more. These updates probably will not be required. (0.09 cm/sec^2 is the existing update criterion)

---

(3) These recommendations apply to the X and Z accelerometers. The Y bias should also be measured and updated, and it would be reasonable to update with the same criterion although updating only if the bias exceeeds the existing criterion of 0.09 cm/sec^2 will meet the requirements.

(4) The inflight astronaut procedure could be used but will give reduced accuracy.

(5) Coast and align downlist words 77b and 78b changes

(6) Coast and align downlist word 51 change

George P. Edmonds Jr.
G.P. Edmonds, Jr.
System Test Group

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From GET	To GET	Uncomp. Calculated Bias X cm/sec^2	Uncomp. Calculated Bias Y cm/sec^2	Uncomp. Calculated Bias Z cm/sec^2	Sample Time sec	PIPA Counter Contents (No compensation running) X Counts	Y Counts	Z Counts	Remarks
47:44:00	47:47:00	+0.36	-0.045	+0.015	180	452/518	-60/-68	34/37	
47:44:00	48:01:12	+0.33	-0.037	+0.016	1032	452/835	-60/-97	34/+51	
48:14:00	48:24:58	+.36	-.016	0	658	1120/1360	-112/-125	64/65	
91:07:44	91:22:03	+.33	+.05	+.015	859	436/684	66/107	26/39	
91:07:44	91:55:59	+.32	+.056	+.01	2895	436/1291	66/215	26/56	
94:46:03	95:01:15	+.28	+.075	-.003	912	101/375	24/92	-3/-6	
97:09:43	97:15:49	+.31	+.054	-.016	366	295/182	66/46	-18/-12	
100:21:03		+.297	+.082	-.017	1436	1897/2324	509/627	-64/-88	

TABLE I

MIT/IL
Apollo Guidance and Navigation
System Test Group Memo No. 1349

To: Distribution

From: G. P. Edmonds, Jr.

Date: 22 May 1969

Subject: LM Gyro Drift Check Before PDI Using a Sun Sighting

Reference:
1. Mission Simulation Memo No. 11-69, Bernard A. Kriegsman Effects of IMU Errors During DOI and PD on Powered Landings PGNCS Performance.
2. Preliminary Apollo 11 Flight Plan 15 April 1969
3. APM 49-69, R. White, A Proposed Method of Fine Alignment When the LM is Docked to the CSM
4. MSC Memo No. 69-PA-T-78A

Introduction

Reference 1 shows that uncompensated Y gyro drifts of 0.5^{o}/hr (33 meru) will cause an unsatisfactory lunar landing. Therefore, it is desired to make a measurement of this parameter before PDI even though gyro reliability is good and MIT would not consider a drift measurement essential.

This memo will discuss using a partial alignment check before PDI based on a sun sighting following the pre-DOI AOT P52 to obtain a drift measurement.

Proposed Method

The proposed method is to compare the LGC computed IMU gimbal angles to point the AOT at the sun to those actually required. The difference is a measurement of misalignment error and is transformed to gyro axis and divided by time since the pre-DOI alignment (about 1 hour) to obtain gyro drift.

The detailed procedure by R. Larson is as follows:

1. Call P52
2. V04 N06

   R2 = 3 (REFSMMAT)
3. V50 N25

   00015

   ENTR (STAR ID = 0)

4. V01 N70
   XX546; place AOT in rear detent
5. V62E
6. V50 N18 (R, P, Y)
7. V06 N18 (Auto Maneuver)
8. V50 N18 (R, P, Y)
9. Key V06 N20
10. Go to PGNCS pulse
11. Center target vertically
12. Key ENTR as target crosses horizontal reference line in AOT
13. Record R1, R2, R3
14. Repeat steps 12 and 13 two times
15. V37E00E
16. Ground transform gimbal angles from step 13 (average values) and step 7 to angles about gyro axis and compute gyro drifts.

Error Analysis

No information on misalignment can be obtained about the LOS to the sun (near Z_{sm}) however the following table shows expected errors on any axis perpendicular to the sun LOS (i.e., X_{sm} or Y_{sm} approximately)

Error Source	1σ Error per Axis	Data Source
Basic AOT inflight overall accuracy	0.025°	AG 472-67
Additional sighting error due to sun diameter of 1/2 deg.	0.05°	G. Karthas estimate
Error from previous P52 which will effect this drift measurement.	0.025	STG 1327 Rev. 1
Gimbal angle bit size. 40 sec for each time read	$.011^{\circ}$	
	RSS= 0.064°	

The rss total alignment error is 0.064° 1σ. This gives a drift uncertainty, assuming 1 hour between the sun measurement and the pre DOI alignment, of 0.064°/hr 1σ or 0.19°/hr (12 meru) 3σ.

Recommendation

Work should be continued on other methods such as that in Reference 3* or a complete P52 before PDI if stars are found to be available. Limits on the results of a drift measurement should be set as a compromise between the following objectives:

1. Low probability that a gyro is "broken".
2. Low probability of not landing when system is actually good.
3. Low probability that uncompensated drift is near or above 33 meru.

Reference 4 suggests the mission rule that if misalignment exceeds 0.25° (17 meru drift since pre DOI alignment) then PDI is delayed 1 rev. and another P52 is performed in the dark. This criterion is about the optimum compromise to meet the above objectives and should be used. If this second P52 is used and the indications are that there is a stable drift the compensation should be updated and the landing made.

*The ideal situation would be to:

(1) use the method of reference 3 at the docked alignment

(2) measure drift and perhaps update compensation after the pre-DOI P52

(3) verify the results of (2) before PDI with the sun sighting method or a complete P52 if possible.

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Massachusetts Institute of Technology
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Cambridge, Massachusetts 02142

APOLLO PROJECT MEMORANDUM #49-69

TO: Distribution

FROM: Robert White

DATE: 16 May 1969

SUBJECT: A proposed method of LM IMU fine alignment when the LM is docked to the CSM.

I. Introduction

Recently I proposed to Bill Tindall a method of LM IMU fine alignment during the LM-CSM docked state which is totally independent of the knowledge of relative orientation of the LM with respect to the CSM as long as the relative orientation remains fixed during attitude changes of the docked configuration. In other words, this method, unlike the present one being used, is independent of the structural alignment errors associated with the CM and LM docking rings, navigation base mounting surfaces, etc. The IMU alignment accuracy obtainable with the proposed method seems to be better than that obtained by sighting on stars with the AOT. The method consists of reading the CSM and LM IMU CDU angles at three different but somewhat arbitrary attitudes of the docked configuration and using this data to determine the gyro torquing angles needed to perform the fine alignment of the LM IMU. It is assumed that the CSM IMU orientation with respect to the Basic Reference Coordinate System is known and, as a matter of convenience, represents the desired orientation of the LM IMU. The attitude changes required in order to obtain the three different attitudes of the docked configuration may be executed by either vehicle (preferably the CSM) and could conceivably be those already being performed for some other purpose such as landmark tracking. Probably the best way of implementing this method at the present time is to place the desired CDU data on downlink and have the earth compute and send up the required gyro torquing angles.

II. General Description of Proposed Method

The basic approach used in the proposed method is somewhat analogous to that used in the IMU Realignment Program (P-52) in that two known directions in inertial space are obtained both with respect to the present and desired IMU Stable Member Coordinate Systems. The two directions arbitrarily denoted as A and B can be represented in the two coordinate systems as shown in Figure 1 where $\underline{s}_A$ and $\underline{s}_B$ denote the unit vectors for the directions A and B in present stable member coordinates, and $\underline{s}'_A$ and $\underline{s}'_B$ are the unit vectors for the corresponding directions in desired stable member coordinates. With these four vectors it is possible to compute the orientation of the desired IMU Stable Member Coordinate System with respect to the present one and thereby determine the gyro torquing angles required to drive the IMU to the desired orientation. In Program P-52, the vectors $\underline{s}_A$ and $\underline{s}_B$ are obtained from optical sightings on two celestial bodies A and B (usually stars), and $\underline{s}'_A$ and $\underline{s}'_B$ are obtained from data stored in the computer or loaded by the astronaut. There is, however, nothing magical about using only celestial bodies for this purpose as clearly shown in the Lunar Surface Alignment Program (P-57) where the direction of lunar gravity is substituted for a celestial body. The main goal is to simply obtain two directions in inertial space with respect to the present and desired IMU Stable Member Coordinate Systems.

In the proposed method the two directions (A and B) are the directions of rotation of the docked configuration with respect to the CSM and LM IMU's when going from the first to the second and from the second to the third attitudes. It is assumed that the CSM IMU has already been aligned and represents the desired orientation of the LM IMU.

What is meant by a direction of rotation is the following: When a vehicle changes attitude the orientation of its navigation base with respect to its inertially fixed IMU stable member also changes. If the orientation of the navigation base axes with respect to the IMU Stable Member Coordinate System is determined for two different attitudes of the vehicle, a direction of rotation can be computed in IMU Stable Member Coordinates about which the navigation base axes can be rotated from the first to the second orientation. This direction of rotation is a function only of the two orientations of the navigation base and does not depend on the manner in which the orientation is changed when going from first to the second orientation. Likewise, if two docked vehicles containing IMU's are placed at two different attitudes, a direction of rotation can be established in each vehicle for its navigation base with respect to its IMU. In this latter case it is apparent that the direction of rotation in each vehicle will have a common direction in space. This method of determining a common direction in space with respect to two IMU's is therefore analogous to determining the direction to a star in both the present and desired IMU Stable Member Coordinate Systems.

To obtain two directions of rotation in space with respect to the CSM and LM IMU Stable Member Coordinate Systems, use is made of the IMU CDU or gimbal angles recorded in each vehicle for each of three attitudes of the docked configuration. For each set of IMU gimbal angles, recorded in a given vehicle for a given attitude, a matrix [SMNB] can be computed as follows: where IGA, MGA and OGA denote the inner, middle and outer IMU gimbal angles respectively:

$$Q_1 = \begin{bmatrix} \cos IGA & 0 & -\sin IGA \\ 0 & 1 & 0 \\ \sin IGA & 0 & \cos IGA \end{bmatrix} \quad (2)$$

$$Q_2 = \begin{bmatrix} \cos MGA & \sin MGA & 0 \\ -\sin MGA & \cos MGA & 0 \\ 0 & 0 & 1 \end{bmatrix} \quad (3)$$

$$Q_3 = \begin{bmatrix} 1 & 0 & 0 \\ 0 & \cos OGA & \sin OGA \\ 0 & -\sin OGA & \cos OGA \end{bmatrix} \quad (4)$$

$$[SMNB] = Q_3 \; Q_2 \; Q_1 \quad (5)$$

The matrix $[SMNB]$ is the standard matrix used in the Apollo Guidance Computer (AGC) to transform vectors from stable member to navigation base coordinates. It should be noted that $[SMNB]$ completely defines the orientation of the navigation base with respect to the IMU Stable Member Coordinate System since it may be expressed as follows:

$$[SMNB] = \begin{bmatrix} \underline{x}_{NB}^T \\ \underline{y}_{NB}^T \\ \underline{z}_{NB}^T \end{bmatrix} \quad (6)$$

where $\underline{x}_{NB}$, $\underline{y}_{NB}$ and $\underline{z}_{NB}$ are the unit vectors defining the directions of the x, y and z axes of the navigation base with respect to the vehicle's IMU Stable Member Coordinate System.

For three different attitudes of the docked configuration there will be three different values of $[SMNB]$ for each vehicle. Let $[SMNB_1]_{LM}$, $[SMNB_2]_{LM}$ and $[SMNB_3]_{LM}$ be the LM values of $[SMNB]$ for the three different attitudes, denoted as 1, 2 and 3, and let $[SMNB_1]_{CM}$, $[SMNB_2]_{CM}$

and $\left[SMNB_3\right]_{CM}$ be the corresponding CSM values. With the three values of $\left[SMNB\right]$ for the LM, two directions of rotation ($\underline{s}_A$ and $\underline{s}_B$) of the LM navigation base with respect to the LM IMU can be deteremined as follows:

$$\left[L\right]_{12} = \left[SMNB_1\right]^T_{LM} \left[SMNB_2\right]_{LM} \tag{6}$$

$$\left[R\right] = \left[L\right]^T_{12} - \left[L\right]_{12} = \begin{bmatrix} r_{11} & r_{12} & r_{13} \\ r_{21} & r_{22} & r_{23} \\ r_{31} & r_{32} & r_{33} \end{bmatrix} \tag{7}$$

$$\underline{s}_A = \text{UNIT}\ (r_{32},\ r_{13},\ r_{21}) \tag{8}$$

$$\left[L\right]_{23} = \left[SMNB_2\right]^T_{LM} \left[SMNB_3\right]_{LM} \tag{9}$$

$$\left[T\right] = \left[L\right]^T_{23} - \left[L\right]_{23} = \begin{bmatrix} t_{11} & t_{12} & t_{13} \\ t_{21} & t_{22} & t_{23} \\ t_{31} & t_{32} & t_{33} \end{bmatrix} \tag{10}$$

$$\underline{s}_B = \text{UNIT}\ (t_{32},\ t_{13},\ t_{21}) \tag{11}$$

It should be noted that the above procedure will not yield the correct values for $\underline{s}_A$ and $\underline{s}_B$ if either of the two rotations is equal in magnitude to 0^o, 180^o, 360^o, etc.

Using the above procedure with the three values of $\left[SMNB\right]$ for the CSM, two directions of rotation ($\underline{s}'_A$ and $\underline{s}'_B$) of the CSM navigation base with respect to the CSM IMU can be determined:

$$\left[C\right]_{12} = \left[SMNB_1\right]^T_{CM} \left[SMNB_2\right]_{CM} \tag{12}$$

$$\left[P\right] = \left[C\right]^T_{12} - \left[C\right]_{12} = \begin{bmatrix} p_{11} & p_{12} & p_{13} \\ p_{21} & p_{22} & p_{23} \\ p_{31} & p_{32} & p_{33} \end{bmatrix} \tag{13}$$

$$\underline{s}_A' = \text{UNIT } (p_{32}, p_{13}, p_{21}) \tag{14}$$

$$\left[C\right]_{23} = \left[SMNB_2\right]^T_{CM} \left[SMNB_3\right]_{CM} \tag{15}$$

$$\left[S\right] = \left[C\right]^T_{23} - \left[C\right]_{23} = \begin{bmatrix} s_{11} & s_{12} & s_{13} \\ s_{21} & s_{22} & s_{23} \\ s_{31} & s_{32} & s_{33} \end{bmatrix} \tag{16}$$

$$\underline{s}_B' = \text{UNIT } (s_{32}, s_{13}, s_{21}) \tag{17}$$

Since the vectors $\underline{s}_A'$ and $\underline{s}_B'$ will have the same directions in space as the corresponding LM vectors $\underline{s}_A$ and $\underline{s}_B$, the four vectors can be used to determine the orientation of the desired (CSM) IMU Stable Member Coordinate System with respect to the present (LM) IMU Stable Member Coordinate System in the manner described for the routine AXISGEN in Section 5.6.3.2 of the Colossus or Luminary GSOP. Afterwards, the required gyro torquing angles may be computed as shown for the routine CALCGTA in the same section of the above GSOP's.

III. Additional Details on Implementation of Proposed Method

At present the most convenient way of implementing this method for a mission such as G is to place the desired CDU data on downlink and have the earth compute and send up the required gyro torquing angles.

As indicated earlier, IMU CDU angles will be required from each vehicle for three different attitudes of the docked configuration. Since the attitude rates are seldom zero, even in "attitude hold", it is important that the CDU angles obtained from each vehicle for a given attitude be as near to each other in time as possible in order to minimize the error in the alignment process. Probably the best way of obtaining simultaneous CDU angles from each vehicle and having this data present on the downlink is to use V06N20 simultaneously in each vehicle for each of the three attitudes. V06N20 causes a DSKY display of the present IMU CDU angles and places this data in a special location on the downlist. It is understood that this method of obtaining a simultaneous set of CDU angles is used at least once in the present method of fine alignment.

Another way of obtaining the desired CDU angles at the earth is to use the CDU data which is always present on every downlist from each vehicle. This approach is attractive in that it does not require the astronauts in each vehicle to repetitively use V06N20 in order to get the data. During the transmission of a downlist, which usually occurs every two seconds, the actual CDU angles are read twice (separated by one second) and placed on the downlist. A timing diagram illustrating the approximate times when the CDU angles are read and placed in specific word locations on the downlists is shown in Figure 2. These angles are present in word locations 9/10 and 59/60 for the CSM, and in 37/38 and 87/88 for the LM. The ground elapsed time is given in word location 51 for all downlists. Since the downlink transmissions of the CSM and LM are not synchronized with respect to each other, the CDU angles obtained from each vehicle will also not be synchronized. However, note that a set of CDU angles obtained from one vehicle will always be within 0.5 seconds of a set obtained from the other vehicle. When the CSM is docked to the LM it is understood that the attitude rate is less than 0.01 degrees per second during attitude hold. If the attitude rate is 0.01 degrees per second and the difference in CDU read times is 0.5 seconds, an error of 0.005 degrees will occur, which is well within the desired accuracy. In fact, an error a few times larger than this would seem acceptable. However, there is apparently some problem on the earth with respect to quickly identifying and collecting the best CDU values from the above word locations. Since I don't know the nature of this problem, I cannot make a proper evaluation of this approach.

The attitude changes required in this method are somewhat arbitrary but should be at least 30 degrees (90^{o} is optimum) between attitudes 1 and 2 and between attitudes 2 and 3. In addition, the angle between the two directions of rotation should be at least 30 degrees (90^{o} is optimum). The attitude maneuvers may be executed by either vehicle and could conceivably be those already being performed for some other purpose. Actually, a simple roll maneuver followed by a pitch or yaw maneuver would be suitable for this purpose.

Robert L. White
Robert White
Assistant Director

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C. F. Wasson, EG2
MIT at GAEC
MIT at KSC
MIT at NAA
J. Lawrence
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MIT at MSC

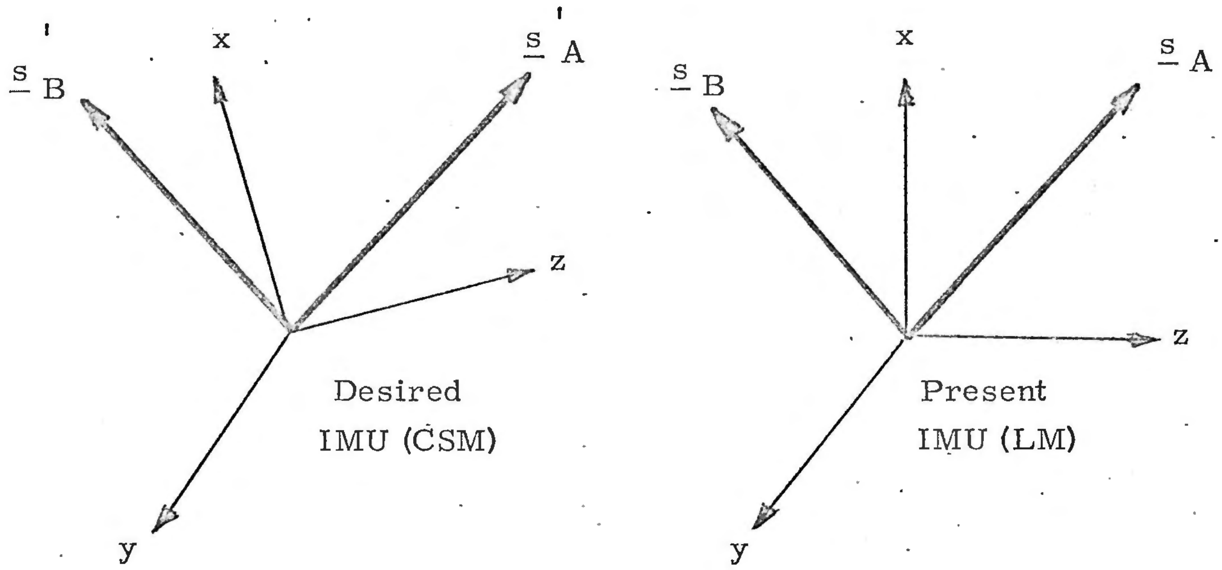

Figure 1

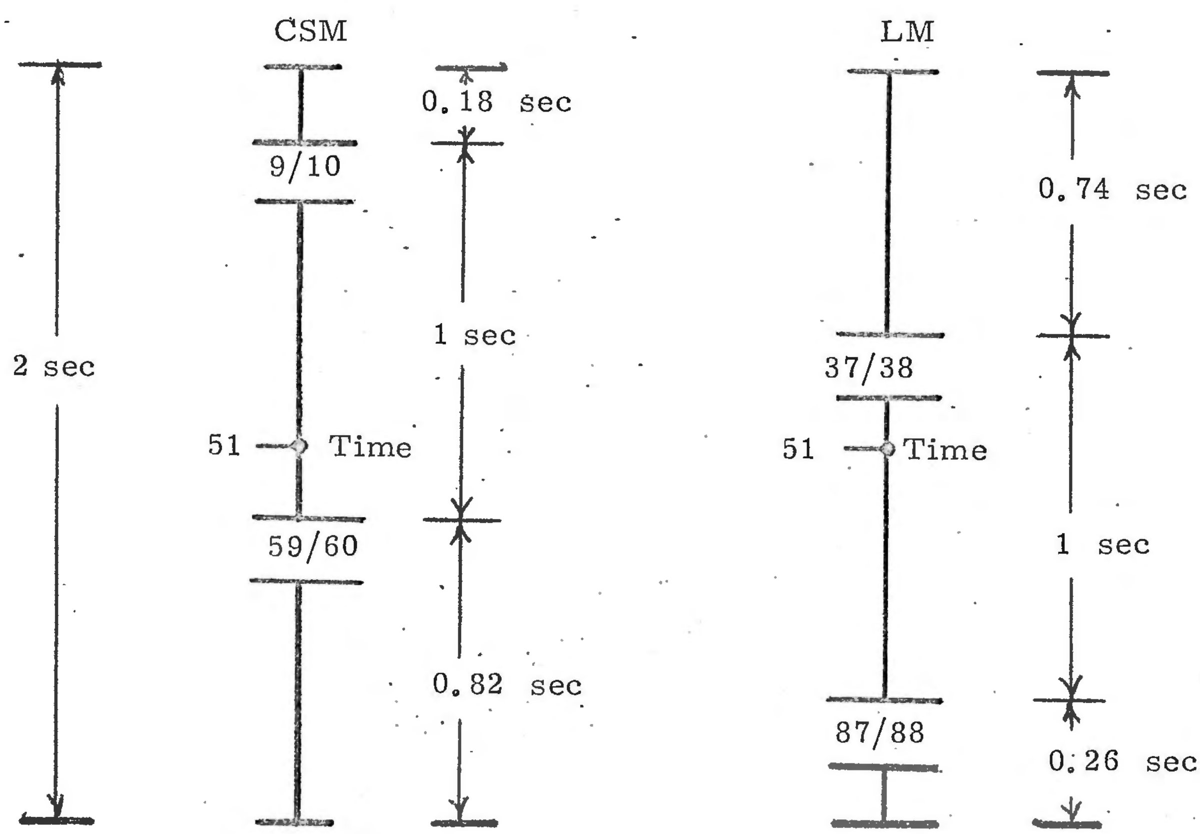

Figure 2

Massachusetts Institute of Technology
Instrumentation Laboratory
Cambridge, Massachusetts

LUMINARY Memo #81

To: Distribution
From: D. Eyles
Date: 28 April 1969
Subject: Landing Dependence on Platform Alignment

The landing, unlike other powered flight programs, depends on a particular platform alignment, the "lunar landing alignment" described in the GSOP section 5.1.4.2, also section 5.3.4.2.3 and in chapter 4 assumption 3 of programs 63 - 67. The landing site alignment is the local vertical at the site at TLAND with the z-axis in the CSM orbital plane.

Questions have arisen of how sensitive the landing is to wrong alignments - that is alignments different from the lunar landing alignment, but accurately reflected in REFSMMAT. Slipping the landing one orbit without realigning the platform is one case in which wrongness develops - about one degree. Redesignations by shifting the site introduce wrongness, but of negligible magnitude.

The permissable wrongness was first cautiously estimated as one degree. It was hypothesized that five degrees in any direction is okay, and runs were made which confirm this. Five degrees can be noticed, but can easily be tolerated.

There follows some background and a description of the runs.

The lunar landing alignment is assumed in three places in the landing programs:

(1) in the radial control logic where the x and y axes of the platform define the directions, by assumption radial and out-of-plane, which are favored in the allotment of available thrust.

(2) in the redesignation logic where certain very wrong alignments could cause the site to be involuntarily redesignated, or cause the new site in the event of a voluntary redesignation to be wrong.

(3) in the ignition algorithm where the guidance-to-platform transformation matrix is initialized as the identity matrix: in this case the matrix would soon be corrected, but meanwhile the time-to-go computation may have blown up.

The runs were four: two in which the platform is wrong by a rotation of 5 degrees each way about its y-axis, arbitrarily called Y+5 for the case where the x-axis pierces the moon to the east of the site, and Y-5 for the occidental case; one run each with 5 degree x-axis and z-axis rotations, on the assumption that these cases are symmetrical.

The y runs were perceptibly different from the nominal in ignition time, throttle-down time, landing time, and achievement of high gate.

	ignition time (TIG-0)	throttle-down time	touch-down time	altitude and altitude rate from first P64 display (nominal: 7783 feet, -143.4 f/s)
Y-5	early .01 second	late about 7 seconds	early about 4 seconds	7594 feet -130.0 f/s
Y+5	late .01 second	early about 5 seconds	late about 2 seconds	7730 feet -154.3 f/s

There is nothing to worry one here.

The x run was very close to the nominal, differing slightly in throttle-down time.

The z run was indistinguishable from the nominal, except for noise. This is as expected since only a z-axis rotation does not affect radial control (because <u>both</u> the x and y axes are given thrust priority). The closeness of this run indicates that to radial control alone should be attributed the differences seen in the x and y cases.

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AG# 114-69
March 3, 1969

National Aeronautics and Space Administration
Manned Spacecraft Center
Houston, Texas 77058

Attention: Mr. W. Kelly
Project Officer
Guidance and Navigation

Thomas Gibson FS5

Through: NASA/RASPO at MIT

Subject: Aborting from the Hohlmann Descent in Mission G

Gentlemen:

Tom Gibson asked me the following question. My answers follow.

Tom: George, if the astronaut decides during the Hohlmann orbit descent prior to powered descent ignition that he wants to abort (with DPS or APS) what procedure should he use? How can he avoid ignition in a retrograde attitude in P63 since he knows he wants a posigrade burn?

George: Tom, he has two ways of aborting in this situation. The first way is useful if he knows which engine he will start and complete the abort with.

<u>1st Way: Target P30 with the PDI Abort Velocity</u>

1. Load N33 in P30 with TIG (PDI).

2. Load N81 with

   R1 +5644.2 for DPS (5646.6 for APS)

   R2 +0.0

   R3 -19.5

3. Select P40 for DPS (P42 for APS)

The problem with this procedure is that if the abort is started with the DPS (P40) and cannot be concluded with the DPS, bringing the APS mass, APS DAP, APS tailoff, et cetera on line is awkward. R03 would have to be used. The APS tail-cff, ATDECAY, would have to be loaded or the error in tail-off would have to be accepted, The same sort of problem exists in DPS P40 burns in LUMINARY 1. We are investigating now how a burn started with the DPS could be concluded with the APS.

(What one would like to see, of course, is a an abort stage capability in P40. If there were an abort stage monitor in P40, the astronaut could cause the LGC to set the correct APS mass, select the APS DAP, the APS engine tail-off, et cetera simply by pushing the abort stage button. The monitor could also switch the program number from 40 to 42 just to show it recognizes his input, are you interested?)

The advantage of this first way is that the maneuver to the abort burn direction is done automatically by P40.

2nd Way: Use P70 and P71

The astronaut can abort with P70 or P71 if he is willing to start the engine in P63 as though he were going to land. The abort discretes monitor and the abort programs are not enabled until TIG in P63. The maneuver to the retrograde orientation in P63 can be avoided by keying in an ENTER response to the FL V50 N18 display in R60. The astronaut can steer the S/C to the approximate posigrade abort burn attitude by putting the mode control switch into ATT HOLD, steering to the approximate burn attitude, and then returning the mode control switch to AUTO. The P63 landing equations are not used until throttle-up time in P63. Thus, by avoiding the KALCMANU (R60) maneuver in P63, the astronaut has complete control of the spacecraft attitude between P63 entry time and throttle-up time. If he presses the abort button between TIG and

TIG + ZOOMTIME, P70 will be entered before the landing equations are activated. Since P70 selection immediately throttles up the DPS, he should delay pressing abort until thrust and attitude control have stabilized at 10%.

The advantage of this technique is that P30 does not have to be targetted. The disadvantage is that the crew must perform a manual maneuver to the approximate burn attitude.

George W. Cherry
George W. Cherry
LUMINARY Project Manager

GWC/db

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