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A PRELIMINARY ANALYSIS OF FAILURE DETECTION AND GUIDANCE SYSTEM MONITORING DURING THE LM POWERED DESCENT

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SUMMARY

An analysis of the PNGS and AGS guidance failures and their effect on the powered descent trajectory has been made. The critical failures have been identified and the probability of their detection by the crew determined. Failed system isolation using the MSFN and landing radar is discussed. Finally, two possible methods for monitoring the onboard guidance systems are discussed and evaluated.

INTRODUCTION

The present LM strategy requires the LM to be controlled automatically by the guidance throughout almost the entire descent to the lunar surface. While the crew does not exercise spacecraft control until late in the descent, they can monitor the trajectory through the primary (PNGS) and abort (AGS) guidance systems in conjunction with navigational data received from the manned space flight network (MSFN) data.

Among the functions the crew performs during the descent is that of detecting and isolating guidance failures before a safe abort is no longer possible. Some measure of confidence that the systems are operating satisfactorily can be achieved by comparing the trajectories computed by the PNGS and AGS. In addition, monitoring of the trajectories can provide failure detection, but identification of the failed system is not always possible. The studies of references 1 and 2 indicate that differencing of parameters of the two systems also provides failure identification, but as before, the technique does not provide failed system isolation. In both cases, some other system must be used to positively identify the failed system.

The present study is directed toward the problems involved in the monitoring of guidance systems and the detection and isolation of failed systems during the LM descent to the lunar surface. The report examines monitoring concepts, identifies critical failures and their probability of detection by the crew. Finally, two techniques for providing the onboard monitoring requirements are evaluated.

DESCENT TRAJECTORY

The LM descent trajectory is divided into a braking phase, a final approach phase, and a landing phase. The braking phase, which starts at 50,000 feet pericynthion altitude and ends at a high gate altitude of the order of 9,000 feet, is a near fuel-optimum trajectory. The final approach starts at hi-gate and is designed to provide the crew visibility of the landing site and sufficient time to assess the landing area. The landing phase begins at a low gate altitude of the order of 500 feet approximately 1,500 feet from the landing site. The phase is designed to afford the crew time to make a close-in assessment of the landing site with a trajectory easy to control.

GUIDANCE AND NAVIGATION SYSTEMS

Control of the LM is through the primary navigation and guidance system (PNGS). Backing up the PNGS for aborts following any PNGS failure is the abort guidance system (AGS). In addition, the MSFN is able to track the LM and provide certain navigational data.

Primary Guidance System

The main elements of the PNGS are the guidance computer (LGC), the inertial measuring unit (IMU), the display and keyboard (DSKY), the landing radar (LR), and the rendezvous radar (RR) or optical tracker (LORS). The LGC, IMU, and DSKY function as a unit, but the LR and RR have the capability to operate independently of the other equipment.

The PNGS computes the descent guidance steering commands throughout the descent maneuver. The PNGS is initialized from lunar orbit navigation data processed onboard or obtained from the MSFN prior to separation and realined before the start of the descent burn. The PNGS operates as a pure inertial system to an altitude of the order of 25,000 feet. At this altitude, the landing radar measurement of altitude is combined with PNGS estimate of altitude to begin correcting the inertial drifts to wash out terrain altitude uncertainties. To prevent large transients in PNGS operation in the event of significant differences between the two altitudes, the LR altitude is weighted at a ratio of 0.1 with the PNGS altitude at the first update. The LR altitude weighting is increased from 0.1 to 0.55 at an altitude of the order of 15,000 feet where it remains for the remainder of the landing maneuver. The update of PNGS velocity by the LR begins at an altitude of about 15,000 feet with a weight of 0.1 increasing to 0.4 at an altitude of the order of 5,000 feet.

Abort Guidance System

The primary elements of the AGS are the abort sensor assembly (ASA), the abort electronics (AEA), and the DEDA. The AGS is initialized by the PNGS just prior to the descent burn and provides backup ascent guidance for abort following PNGS failure. The AGS does not provide descent guidance, but it can compute and display various trajectory parameters on the flight instruments or DEDA.

Manned Space Flight Network

The discussion of reference 3 indicates the determination of the LM state vector from MSFN observations is relatively poor. The position and velocity are determined to a 3σ accuracy of 10 n. mi. and 120 ft/sec, respectively, with the errors being very nearly equally distributed in height, downrange, and crossrange. However, the MSFN measurement of $\dot{R}$ (doppler or velocity along the station-LM line of sight) is quite good with a 3σ error of about 1.5 ft/sec. The MSFN data are not processed through a statistical filter but are simply subjected to coordinate transformation and smoothing. Thus, the information delay is essentially that of transmission, perhaps of the order of three seconds.

GUIDANCE FAILURE MODES

Both the PNGS and AGS are subject to a number of failures that either cause a complete disruption of the output or so seriously degrade their performance they are of no further use for controlling the spacecraft. Many of the failures that the onboard guidance systems are subject to are detected automatically, and the crew is informed of these failures through caution and warning devices. Among these are the two guidance computers (LGC, AEA) which contain self-checks for failure detection, electrical power supplies, engine trim gimbals, RCS jets, and others. Failures of this type will be detected by the crew without difficulty.

Failure Types

In general, the failures that are of concern in monitoring can be grouped into one of two categories: (1) "hard-over" and (2) slowly deteriorating. Hard-over failures, such as a gyro, cause fairly rapid deviations from the expected spacecraft performance. Such failures generally cannot endanger the crew because the slow attitude and translational characteristics of the LM during descent allow the crew time to assess the failure and take corrective action. The failures in the second category are more difficult for the crew to detect because they result only in relatively slow divergence from the normally expected conditions. While this type of failure creates no immediate danger to the crew, allowing it to persist will ultimately drive the spacecraft into flight conditions which reduce the possibility of safe abort. However, as these failures must also exist for extensive periods of time, the crew is afforded a reasonable amoung of time for detection of the failure.

Primary Failure Sources

From a guidance viewpoint, the primary area of failures that affect the trajectory arise from inertial measuring component characteristics. In the LM, the equipment subject to these failures are the IMU (PNGS) and the ASA (AGS). The characteristics of major importance are gyro drifts, accelerometer biases, component misalinements, and other items such as mass unbalance and nonlinearities of the gyros and accelerometers.

Effect of Failures on PNGS and AGS Trajectories

The primary sources of the failures affecting trajectory computations arise from failed or partially failed IMU or ASA accelerometers and gyros. To determine the effect of these failures on the trajectories computed by the PNGS and AGS, the presently expected 1σ accelerometer and gyro errors were inserted into the descent trajectory program. The deviations in the PNGS and AGS positions and velocities were then determined assuming a final inertial alinement five minutes prior to the start of descent burn. Error propagation was terminated at 430 seconds into the descent burn, corresponding to a hi-gate altitude of the order of 8,600 feet. From this, the magnitude of the deviations required to cause the LM to penetrate the deadman's curve were determined assuming that linear analysis techniques were valid. The actual IMU and ASA errors used in the analysis were: (1) accelerometer bias, scale factor, scale factor nonlinearity, misalinement, and cross-axis sensitivity, (2) initial platform misalinement, and (3) gyro drift, mass unbalance, and anisoelasticity. The data from the linear analysis provided the expected 1σ deviations about the descent trajectory for both the PNGS and AGS as a function of time into the descent burn (figure 1).

FAILURE DETECTION TECHNIQUES

A check on PNGS performance can be obtained by using the PNGS and AGS estimates of the trajectory in at least two ways: (1) monitoring trajectory bounds or (2) differencing selected PNGS and AGS trajectory variables.

Failure Detection by Reference Trajectory Monitoring

In reference trajectory monitoring, the expected deviations of each system from the reference trajectory are determined as a function of time from final update or time into descent burn. The two bounds of selected parameters of each system can be checked at specified intervals of time. As long as the two guidance systems agree, there is a high probability that both systems are working. (There is a certain probability that both systems have almost identical failures in which case both trajectories would be wrong but agree.) On the other hand, if the expected limits of either system are exceeded, there is a high probability that a failure of some type has occurred, and the crew must decide which system has failed. The decision as to which one has failed is not altogether easy for the trajectory bounds do not provide an immediate answer. Assume, for example, that the AGS limits have been exceeded and that the PNGS is within the expected limits. On the surface, it would appear that the AGS had failed because the PNGS is on trajectory whereas the AGS is not. Remember, though, that the PNGS is insensitive to IMU errors and that the LGC accepts the IMU outputs as being correct and computes the trajectory based on the available sensor outputs. That is, the trajectory is adjusted to meet the sensor data. Hence, even though the AGS trajectory is out of limits, it can be indicating a PNGS failure.

Failure Detection by PNGS-AGS Trajectory Differencing

Failure detection can also be performed by using the difference between the PNGS and AGS trajectory parameters (reference 1). In this technique, the expected 3σ differences (or any other desired difference level) of PNGS and AGS trajectory parameters are precalculated and plotted on charts as a function of time from the start of descent burn. For failure detection, the PNGS/AGS difference is determined at specific time intervals during the descent and compared to the expected 3σ difference for that time. If the difference is within the 3σ bounds, both systems are assumed to be functioning correctly; if the difference exceeds the 3σ difference, one of the two systems is assumed failed. For the latter event, the crew still has the problem of determining which one has failed.

GUIDANCE SYSTEM FAILURE ISOLATION

Regardless of the type of failure detection technique employed, the use of only two systems does not provide a positive identification of the failed system. In the case of the LM, three systems are available for this purpose: (1) landing radar, (2) rendezvous radar, and (3) MSFN. The landing radar should provide a useful measure of altitude after some 200 seconds into the descent burn and altitude rate data some 100 seconds later (reference 4). The rendezvous radar provides usable CSM-LM relative range and range-rate data throughout most of the braking phase. MSFN tracking data of primary significance are velocities, in particular range rate between the LM and earth-based MSFN tracking stations. Note, however, that once the PNGS has been updated by the landing radar, the two systems are no longer independent, and the radar cannot (rather should not) be used for failure isolation thereafter.

Isolation of Failed System

Once it has been established that a guidance system has failed, the procedures for isolation are straightforward. The use of the three systems is:

Landing Radar - The landing radar measurement of altitude can be compared directly with the PNGS and AGS estimates of altitude following the detection of a failure. The primary disadvantage is that the terrain altitude uncertainty may require an extremely large difference to exist between the two systems before a positive identification can be made. Later on in the descent burn, the measurement of altitude rate can be used for a direct check of PNGS and AGS altitude rate estimates which gives a fairly quick isolation check as only the radar velocity uncertainties must be considered.

Rendezvous Radar - The rendezvous radar measurement of range-rate must be compared against a chart showing expected time history of range rate as the LGC and AEA do not compute this variable. Following a failure, it must be assumed that the PNGS has failed if the RR measure of range rate does not agree with the chart and AGS failed if the measured range rate agrees with the cart. The system can be used during the entire braking phase.

MSFN - As in the case of the RR, the MSFN measurement of station-LM range rate must be compared to a chart showing the nominally expected range rate for the particular landing site as neither the LGC or AEA have programs to compute this. From the viewpoint of crew operations, it would be better to compare altitude rate directly, but the expected velocity uncertainty along the moon radius vector and LM almost precludes this. However, as the station-LM range rate is well defined for a specified pericynthion and subsequent powered descent trajectory, the use of station-LM range rate provides a sufficient means of failure isolation. Use of range rate following a failure requires that the MSFN measurement be compared to a chart showing a time history of expected range rate for the given landing site. If the MSFN measurement agrees with the expected range rate, the AGS is assumed to be failed. Should the MSFN measurement disagree with the expected range rate, the crew must assume the PNGS has failed.

RESULTS AND DISCUSSION

Providing a scheme for the detection and isolation of failures requires a knowledge of the effect of various failures on the PNGS and AGS and the magnitude of failures that can be detected. Because the guidance monitoring is based on maximizing the probability of safe abort, the failures of primary concern are those having the greatest effect on altitude and altitude rate. In particular, it is necessary to determine the magnitude of the failures that cause the LM to penetrate the deadman's curve and to establish the probability of their detection by the crew.

Trajectory Deviations Required to Penetrate Deadman's Curve

The deviations in altitude, altitude rate, and lateral velocity arise principally from X and Z-axis accelerometer bias, X-axis and XY cross-axis accelerometer misalinements, and X- and Y-axis gyro drifts. To obtain the trajectory deviations causing penetration of the deadman's curve, these errors were added linearly to the reference trajectory. Figure 2 contains the altitude-altitude rate profile of the normal descent plus deadman's curves for T/W ratios of 0.35 and 0.55 (roughly the expected ratios for the descent engine in the region of interest). The PNGS deviations from the normal trajectory were then extended at various intervals of time to hi-gate until they passed through the two deadman's curves. The multiple of 1σ deviations required to penetrate were then determined and have been plotted in figure 3. This figure indicates that the region of principle interest lies between 300 and 430 seconds into the descent burn. Figure 3 shows that penetration will occur only for 60σ trajectory deviations that have existed for some 300 seconds (600 seconds after final alinement) and that deviations of the order of 10σ must exist nearly 430 seconds to cause penetration. Also, notice the relative insensitivity of T/W to deadman's curve penetration, at least for the range of T/W considered here. For practical considerations, errors in the trajectory of the order of 60σ are not likely to exist unless something is radically wrong with either the spacecrafr or guidance system. The lower deviations causing penetration of the deadman's curve near 430 seconds are reasonable but still represent highly degraded guidance operation.

<u>Magnitude of Component Errors Required for Deadman Curve Penetration</u> - The magnitude of the individual errors required to drive the LM into the deadman's curve can be calculated using the formula:

$$K_X^2 = 1 + \frac{(K_\sigma^2 - 1)\ \sigma_M(t)^2}{\sigma_{ME}(t)^2} \qquad (1)$$

where K_X is the number of 1σ of the error source required for deadman curve penetration

K_σ is the multiple of 1σ trajectory deviations of the combined error sources causing deadman curve penetration

$\sigma_M(t)$ is the normal 1σ trajectory deviation at time t

$\sigma_{ME}(t)$ is the normal 1σ error source contribution to $\sigma_M(t)$ at time t

Using the information from figures 1, 2, and 3, the values for K_X for the PNGS were calculated and have been tabulated in table 1.

		$K_X(\sigma)$					
		Z-axis accel. bias	X-axis accel. bias	XY-cross axis misaline	Y-axis misaline	Y-axis gyro drift	
Time	K_σ					h	$\dot{h}$
300	60	202	67	403	202	703	600
350	30	101	34	200	100	333	277
420	8.5	27	9.3	54	27	87	71

Table 1 - Size of failure for deadman curve penetration

Probability of Crew Detecting Errors

After determining the magnitude of the errors that cause penetration of the deadman's boundary, it is necessary to determine the level of the component errors the crew is able to detect. While the effort in this report is basically aimed at detecting errors causing penetration of the deadman's curve, it should be apparent that the same principle can be applied to the other trajectory variables. However, a complete analysis of the entire trajectory is beyond the scope of this effort, and from the viewpoint of providing safe aborts, not expressly required.

Theory of Detection - The principle effect of errors in the IMU and ASA is to cause an off-nominal trajectory. This off-nominal trajectory will, at some time, exceed the expected normal 3σ boundary. Now, if only the error source is considered, once the mean error forces the normal 3σ boundary deviation to be exceeded, the crew would have an indication of failure by time t_1 as shown on the figure A below.

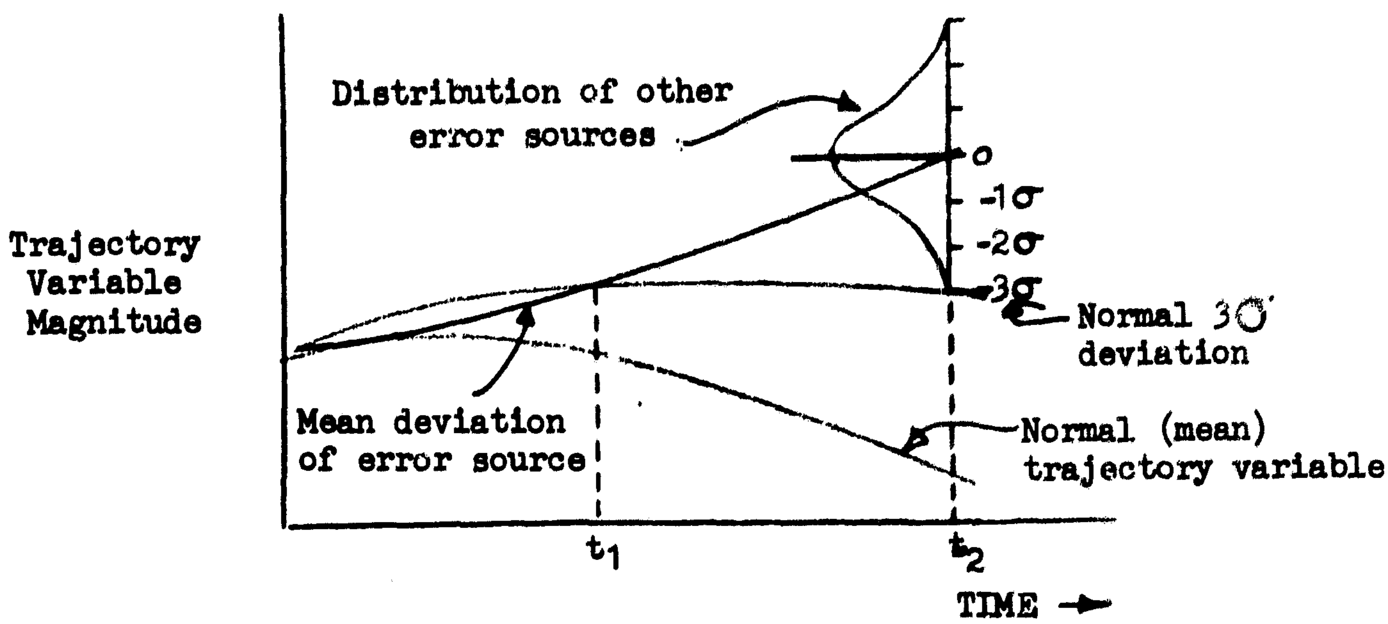

Figure A = Determination of Probability of Detecting Errors

However, the statistical properties of the remaining error sources cause a Gaussian distribution about the new trajectory. Because of this, the actual trajectory is equally likely to be above or below the mean. Hence, there is exactly a 50% chance of detecting the error at t_1. Now if the trajectory is allowed to deviate further, the chance of detection increases accordingly. At t_2 in the above figure, the trajectory has deviated to the point where the distribution about the mean lies at the -3σ point of the remaining error sources. Hence, the crew at this time has a 99.86% chance of detecting the failure. Thus, it can be seen that a correct statistical combination of all error sources allows the 99.86% (or any other level of probability of detection) detection level to be calculated. As shown in As shown in reference 1, the 99.86% level can be determined using the equation:

$$m(t) = 3\ \sigma_M(t) + 3\left[\sigma_M(t)^2 - \sigma_{ME}(t)^2\right]^{\frac{1}{2}} \qquad (2)$$

where $m(t)$ = mean value required for 99.86% detection level and $\sigma_M(t)$ and $\sigma_{ME}(t)$ are as defined previously

Because $m(t) = K\sigma_{ME}(t)$ and $\sigma_{ME}(t)$ is a function of the normal 1σ IMU or ASA errors, the magnitude of component error required for a 99.86% detection level is readily determined. This assumes, of course, that the linear analysis holds for large trajectory and component deviations.

Magnitude of Component Failures for 99.86% Detection Level - Using the numerical values associated with the individual PNGS and AGS component failures in equation (2) yields the results shown in tables 2 and 5. The table indicates the size of a failure yielding a 0.9986 probability of detection remains essentially constant throughout the descent for all failures examined (except for Y-axis gyro drift). As was the case in the size of PNGS failures causing penetration of the deadman's curve, the failures for 0.9986 probability of detection level correspond to essentially complete failures except for PNGS and AGS X-axis accelerometer bias and Y-axis accelerometer misalinement. The monitoring of $\dot{Y}$, as shown in table 3, will pick up an X-axis accelerometer bias failure in either the PNGS or AGS at approximately the same time as the monitoring of h and $\dot{h}$. Also, relatively low level failures in AGS X-axis misalinement can be detected early, but an X-axis gyro must essentially fail before there is a high probability of detection.

	m(t) (σ)											
	Z-axis Accel. Bias		X-axis Accel. Bias		XY Accel. Cross-Axis Misaline		Y-axis Accel. Misaline		Y-axis Gyro Drift			
TIME	PNGS	AGS	PNGS	AGS	PNGS	AGS	PNGS	AGS	PNGS		AGS	
									h	h	h	h
100	20.7	29.8	4.7	8.3	43	17.5	21.1	7.4	81	83	21	20
200	22.6	30	4.8	8.7	42	17	21.0	7.3	78	78	18	17
300	19.7	29.8	4.9	9.0	40	16.3	18.5	7.2	69	60	17	15
400	19.3	30	5.0	9.3	39	17	18.7	7.2	62	52	15	13

Table 2 - PNGS and AGS Failure Magnitudes for a Detection Level of 99.86% Using h and $\dot{h}$

		m(t) (σ)					
		Y-axis Accel. Bias		X-Axis Accel. Misalinement		X-axis Gyro Drift	
		PNGS	AGS	PNGS	AGS	PNGS	AGS
Time (sec)	100	4.3	8.7	19.4	7.1	78	19.1
	200	4.5	9.4	18	7.1	63	16.3
	300	4.7	10.1	16.9	7.1	52	17.3
	400	4.1	11.5	14.5	6.9	40	12.3

Table 3 - Magnitude of Failure for 99.86% Detection Probability by Monitoring $\dot{Y}$

Margin of Safety in Failure Detection - There is initially a considerable margin of safety in detecting the errors resulting in penetration of the deadman's curve. This is shown quite clearly in figure B which shows the two curves for an X-axis accelerometer bias error. At 300 seconds into the burn, it requires roughly an error 15 times greater than the error required for 99.86% level of detection to drive the LM into the deadman's curve. The ratio decreases to 3:1 at 400 seconds and to 2:1 at 430 seconds, which is still a comfortable safety margin. It must be noted that both these curves were calculated assuming no update of the PNGS with the landing radar. Hence, the curve is optimistic in that a much larger error is required for detection. However, the error required to cause penetration of the deadman's curve is also greater so that the margin should not be radically changed, but the detailed analysis necessary to determine the exact effect is beyond the scope of this present effort.

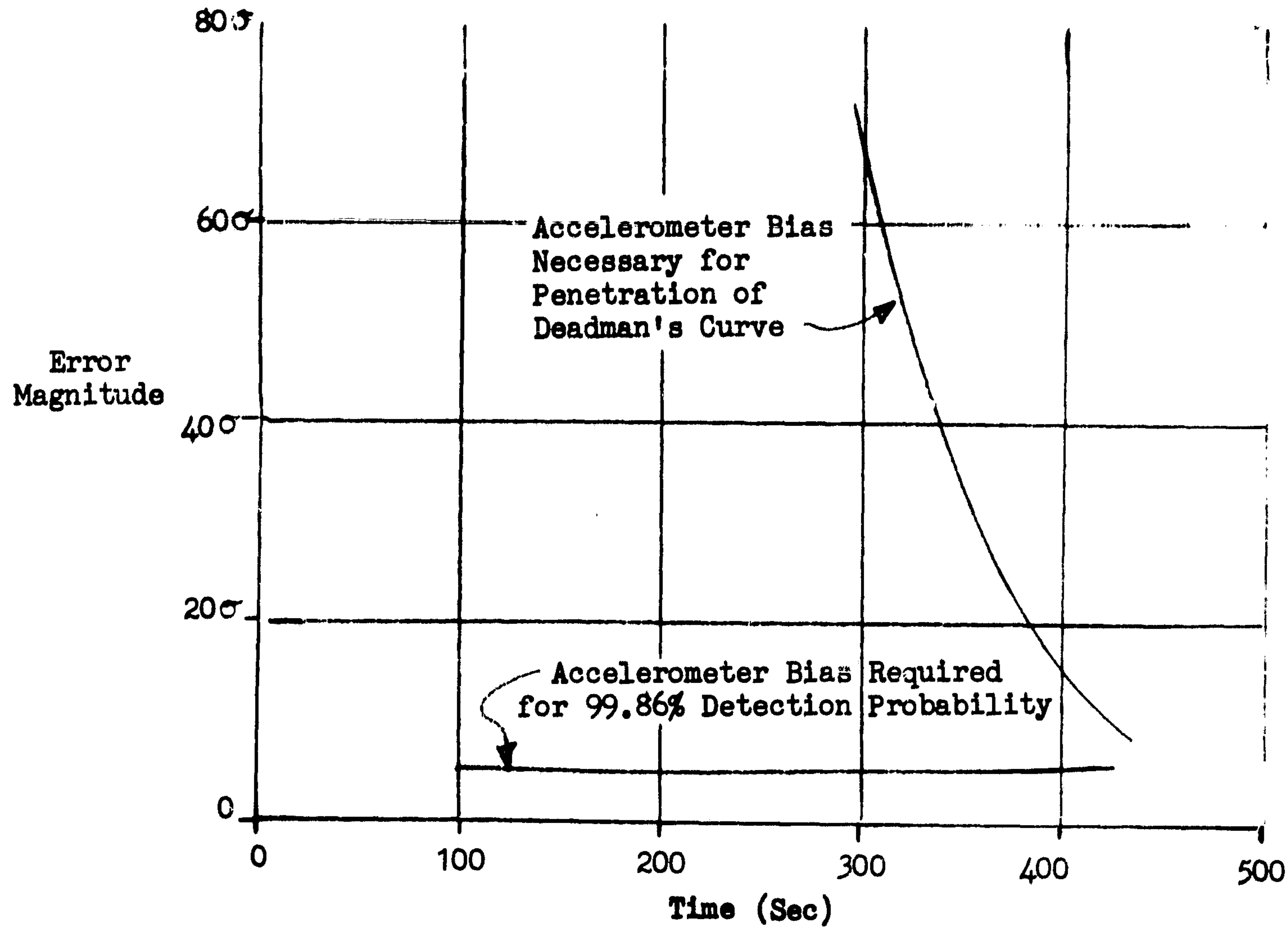

Figure B - Margin of Safety for Error Detection

Error Analysis Summary

The linear error analysis and the discussion of error detection reveal two primary items of interest to guidance monitoring--(1) the trajectory deviations required to cause penetration of the deadman's curve are large except in the region below high gate, and (2) except for isolated cases, the magnitude of a failure that will assure a high level of detection essentially constitutes complete disruption of component function. From the viewpoint of being able to effect a safe abort, it would appear that the dynamic characteristics of the LM preclude anything but a gradual penetration of the deadman's curve. Late in the descent, the picture changes slightly in that smaller deviations affect the safe abort possibilities, but the penetration is still gradual, and the detection probability high. From a monitoring viewpoint, the low dynamic characteristics of the LM coupled with the excessive trajectory deviations necessary to penetrate the deadman's curve means the crew has a relatively long time to assess the guidance operation before dangerous flight conditions are reached. The second fact indicates the crew has an extremely high probability of detecting any and all failures that lead to unsafe abort conditions. Further, while some failures may not have a high detection probability, neither do they radically affect the trajectory and consequently offer little danger to the crew. For these reasons, it makes very little difference whether the trajectory is monitored or PNGS-AGS differencing is used for failure detection. Both provide the same end results with about the same amount of effort on the part of the crew as is indicated in the following discussion.

Reference Trajectory Monitoring

In the trajectory monitoring scheme, the crew periodically examines selected trajectory variables to determine whether they be within their expected bounds. The crew does not perform failure isolation checks unless one of the variables being monitored falls outside its limit. The variables selected for monitoring are contained in table 4; a possible time line of events is given in table 5 for the entire descent.

<u>Braking Phase Monitoring</u> - During the braking phase, the crew shares the monitoring task. The commander monitors PNGS altitude and altitude rate engine T/W, and DPS fuel through the flight instruments. The systems engineer interrogates the DSKY and DEDA for more precise data and compares these to the precomputed bounds of the monitored parameters. In addition, the systems engineer also checks the engine T/W and DPS fuel and monitors delta V through the AGS. Altitude monitoring with the LR is relegated to the commander and should occur as early in the descent as possible for comparison with the PNGS estimate of altitude MSFN provides the LM with an overall velocity check and establishes earth-LM relative range-rate for use in guidance system failure isolation.

With the exception of altitude, the trajectory variables are velocities. The use of velocities as a primary check is based on the assumption, which is reasonable, that if the insertion results in a correct pericynthion state vector and the landing site altitude uncertainty is of the order predicted in reference 5 (RMS = 544 feet), then a safe trajectory is assured if the vertical and lateral velocities remain within their expected 3σ limits. In any event, the present trajectory almost precludes the use of downrange position or landmarks for monitoring because of the uncertainty in correlating range and time from a reference trajectory. The downrange uncertainty exists because of the descent engine calibration during the transfer burn which cannot be predicted beforehand. Crossrange position should not be affected by the engine uncertainties, but correct velocity limits mean the out-of-plane position error is within the expected bounds. Altitude is not required for monitoring early in the descent, but it is a required check later in the descent, and there is no reason to have a break in the monitoring procedures, and therefore, is checked to maintain continuity in the procedure. The monitoring of T/W, fuel, and ΔV provides a check on engine performance.

Table 6 shows the upper and lower bounds for some of the monitored variables during braking. The variables do not need to be more accurate than the nearest 500 feet for altitude and the nearest 5 feet/second for altitude rate because of the difficulty in reading rapidly changing numbers. The times shown for checking are arbitrary and stop at 400 seconds into the powered descent. In the actual operational case, a series of charts or nomograms based on the expected trajectory and descent engine performance will probably be necessary.

Final Approach and Landing Monitoring - Monitoring of the trajectory during the braking phase consists of integrating visual cues with the PNGS trajectory information. MSFN can still track, but, except for one AGS check near the half-way mark of the final approach, AGS monitoring is discontinued because the crew will be too busy with more important duties. In any event, the crew should be able to evaluate the trajectory performance in this phase better than MSFN or the AGS. The commander should monitor the flight instruments, including the LPD and LR. The systems engineer transfers his major effort toward evaluating the PNGS trajectory data, comparing it with the LR, and informing the commander of the DSKY/LPD readout. It is not anticipated that extensive charts containing trajectory data can be used because of the small time allowed for cross checking and because of the very likely possibility that a change of landing site will be made. As long as the final approach trajectory is designed to be within the capability of the crew to a ssess the changes as they occur, visual cues and knowledge of the LPD effects on the trajectory should be sufficient.

PNGS-AGS Trajectory Differencing

The principle operation in guidance monitoring using PNGS-AGS differencing requires the crew to periodically determine the difference in the PNGS and AGS estimates of selected variables. The difference is then compared to a chart (table 7) or graph which contains the expected PNGS-AGS difference for that time in the descent. If the difference is less than the expected difference, the guidance systems are assumed to be operating correctly. Should the difference be greater than the normally expected value, one of the two systems is considered to be failed. The failed system is isolated using one of the three independent systems available for this purpose.

<u>Braking Phase Monitoring</u> - The monitoring of the two systems in this technique follows much the same line as for trajectory monitoring. The additional requirement is that the crew must difference the PNGS and AGS estimates of altitude, altitude rate, and lateral velocity at intervals along the descent. The monitoring chart used is shown in Table 7. The division of responsibility is essentially that of Table 4 with the major change being the differencing performed by the systems engineer. The time line of events is contained in Table 5.

<u>Final Approach and Landing Monitoring</u> - Once hi-gate has been reached, the differencing is discontinued because the time is not available to perform the check. In any event, it would seem that if a guidance failure has not been detected by this time in the descent, there is very small chance of detecting it during the final approach and landing phase. Thus, the monitoring after hi-gate is the same as that for trajectory monitoring.

CONCLUSIONS

Based on the results of this study, the following conclusions are made:

1. There is a high probability of detecting guidance failures that lead to unsafe trajectories, but it is still possible to have failures that cause large guidance errors that are not readily detected by trajectory monitoring.

2. Trajectory deviations of the order of 60σ must exist for over 300 seconds into the powered descent to cause the LM to penetrate the deadman's curve and deviations of the order of 10σ must exist for over 430 seconds to cause penetration of the deadman's curve.

3. The guidance component failures causing the trajectory deviations cited in (2) above, except for accelerometer biases, constitute essentially complete failures of the components. In the case of accelerometer biases, component deviations of the order of 9σ drive the LM into the deadman's curve.

4. Even under the worst possible case, the crew has a better than 99.9% chance of detecting all failures leading to unsafe abort conditions within 100 seconds after the start of descent burn.

5. Either of the two monitoring techniques examined in this study provides the crew with a satisfactory method for detecting guidance system failures; however, the reference trajectory monitoring technique offers some time advantage over the PNGS-AGS differencing technique and is more in line with normal piloting procedures.

6. With the present system configuration, there is very little chance that the crew will be able to identify the failed component using trajectory monitoring techniques.

7. While the rendezvous radar can be used as a failure isolating device, it cannot be identified as being an absolute requirement for guidance monitoring during the LM powered descent to the lunar surface because the landing radar and MSFN are each capable of providing the additional data source required.

RECOMMENDATIONS

The following recommendations are made:

1. The effect of landing radar update on the error analysis should be made to obtain more realistic values for the magnitude of failures that can be detected by the crew.

2. An analysis should be made to determine the full capabilities of the MSFN during powered descent.

3. A piloted simulation study should be made to evaluate the guidance monitoring techniques of this report in an operational environment.

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3. "Current Estimate of Manned Space Flight Network (MSFN) Capability for the First Apollo Lunar Mission", memo to FA/Director of Flight Operations from FM/Chief, Mission Planning and Analysis Division, 66-FM41-177, June 15, 1966

4. "LM Landing Radar Performance Analysis", Memo EG26-66-77 from EG26/Guidance Development Branch to EG/Chief, Guidance and Control Division

5. "Determination of LM Landing Site Inertial Coordinates by CSM Landmark Type Sightings", MSC Internal Note No. 66-EG-18

	MONITORED BY	
VARIABLE	COMMANDER	SYSTEMS ENGINEER
ALT	PNGS-LR	AGS-PNGS
ALT RATE	PNGS-LR	AGS-PNGS
EARTH - LM RANGE-RATE	MSFN	
CROSSRANGE VELOCITY	PNGS	AGS-PNGS
T/W	Meter	Meter
ΔV		AGS
Fuel	Meter	Meter

Table 4 - Monitored Trajectory Parameters

TIME (SEC)	TRAJECTORY MONITORING: COMMANDER	TRAJECTORY MONITORING: SYSTEMS ENGINEER	PNGS-AGS DIFFERENCING: COMMANDER	PNGS-AGS DIFFERENCING: SYSTEMS ENGINEER
B-20	Checks PNGS pitch attitude = 89°, PNGS alt = 50,000 feet, PNGS alt rate = 0 ft/sec, PNGS lat vel = 0. Asks MSFN for state vector update, verifies engine status and that descent engine throttle in minimum position.	Checks AGS pitch attitude = 89°, AGS alt = 50,000 feet, AGS alt rate = 0 ft/sec, AGS lat vel = 0, AGS for vel = 5550 ft/sec. Confirms engine and fuel status.	Same	Same
B+0	Confirms engine firing. Notes thrust decrease from 30% to 10% then to maximum. Verifies T/W reading correct.	Verifies pericynthion state on PNGS and AGS.	Same	Same
B+50	Verifies PNGS pitch attitude = 87°, PNGS alt = 50,000 feet, PNGS alt rate = +15 ft/sec, PNGS lat vel = 0 ft/sec. Confirms engine and fuel status.	Verifies PNGS and AGS alt, alt rate, forward and lateral velocities are correct. Checks engine and fuel status.	Same	Same
B+100	Verifies on tape ind. PNGS alt between 49,000 and 48,500 feet, alt rate between -30 and -25 ft/sec, lat vel between ±1 ft/sec, T/W = 3.4. Requests MSFN velocity check. Confirms engine and fuel status.	Verifies on DSKY and DEDA that PNGS alt between 49,000 and 48,500 feet, alt rate between -30 and -25 ft/sec, lat vel between ±1 ft/sec, AGS alt between 49,500 and 48,500 ft, alt rate between -33 and -23 ft/sec, lat vel between ±2 ft/sec. Verifies PNGS and AGS estimates of forward velocity and informs Commander.	Same	Verifies PNGS-AGS difference alt = 300 feet, alt rate = 6 ft/sec, lat vel = 6 ft/sec. Checks PNGS and AGS trajectory parameter and informs Commander, verifies engine fuel status.
B+150	Verifies PNGS pitch attitude = 83°, T/W ≅ 3.6, alt ≅ 46,000 feet, alt rate ≅ 60 ft/sec. Checks LR for operation.	Verifies AGS estimates of trajectory and informs Commander. Checks engine and fuel status.	Same	Same
B+200	Verifies PNGS alt between 43,000 and 42,000 feet, alt rate between -140 and -125 ft/sec, lat vel between ±2 ft/sec, T/W=3.9. Requests MSFN velocity check. Checks LR for operation.	Verifies PNGS and AGS trajectory estimates of DSKY and DEDA and relays status to Commander. Verifies engine performance.	Same	Verifies PNGS-AGS difference alt = 1100 feet, alt rate = 12 ft/sec, lat vel = 13 ft/sec and informs Commander Checks PNGS and AGS trajectory estimates and informs Commander. Verifies engine status.

Table 5 - Time Line of Events

Table 5 (Continued)

TIME (SEC)	TRAJECTORY MONITORING		PNGS-AGS DIFFERENCING	
	COMMANDER	SYSTEMS ENGINEER	COMMANDER	SYSTEMS ENGINEER
B+250	Verifies PNGS pitch attitude = 78^o, T/W $\approx$ 4.2, alt $\approx$ 35,000 feet, alt rate ; 165 ft/sec. Checks LR for operation.	Verifies PNGS and AGS trajectories estimates and informs Commander. Checks engine and fuel status.	Same	Same
B+300	Verifies PNGS alt between +28,000 and 26,000 feet alt rate between -170 and -150 ft/sec, lat vel between ± 2 ft/sec. T/W $\approx$ 4.5, request MSFN vel. check. Verifies decrease of descent rate. Checks LR for operation and updates PNGS.	Verifies PNGS and AGS trajectory estimates in limits and informs Commander. Checks engine and fuel status.	Same	Verifies PNGS-AGS difference alt $\approx$ 2800 feet, alt rate $\approx$ 19 ft/sec, lat vel 17 ft/sec. Checks PNGS and AGS trajectory estimates and informs Commander. Verifies engine status.
B+370	Verifies increase of descent rate, and that it is $\approx$ -120 ft/sec. PNGS att = 70^o, alt 15,000 feet.	Verifies PNGS and AGS trajectory estimates, engine status and informs Commander.	Same	Same
B+400	Verifies PNGS alt between 14,500 and 10,500 feet, alt rate between -155 and -135 ft/sec, lat vel between ± 3 ft/sec. Requests MSFN velocity check. T/W=5.3. Notes decrease of engine thrust to $\approx$ 60% at t 410 sec.	Verifies PNGS and AGS trajectory estimates and informs Commander. Checks engine and fuel status. Verifies engine thrust change.	Same	Verifies PNGS-AGS difference alt $\approx$ 5,000 feet, alt rate $\approx$ 27 ft/sec, lat vel $\approx$ 24 ft/sec. Checks PNGS and AGS trajectory estimates and informs Commander. Verifies engine status and engine thrust change.

Table 5 (Continued)

TIME(SEC)	COMMANDER	SYSTEMS ENGINEER
FA-10	Alines eye with LPD for expected 52° reading	Verifies PNGS Alt = 7100 feet, Forward velocity = 700 ft/sec Thrust = 6000#, Pitch att = 62° Alt rate = 145 ft/sec, Lat vel = 0 Calls out trajectory data over interphone and VHF command
FA-6	Verifies pitch to final approach attitude	Verifies pitch to final approach att and 6°/sec att rate
FA-0+	Evaluates visibility problems. Performs safety of flight trajectory evaluation. Begins search for landing site and alinement with LPD = 52°. Starts assessment of trajectory and landing site by visual cues.	Verifies pitch angle = 38° LPD reading = 52°, alt = 6100 feet, alt rate = 135 ft/sec, thrust = 5300. Calls out hi-gate event and data over interphone and VHF command.
FA+10	Verifies LPD reading = 52° and LEM is on-trajectory, site good.	Calls out LPD reading, altitude, altitude rate, T/W, and fuel remaining (interphone only)
FA+20	Continues LPD monitoring and trajectory and site assessment. Visually estimates altitude and evaluates altitude rate against values called out by systems engineer.	Verifies and calls out alt = 3800 feet, alt rate = 95 ft/sec, pitch = 38° forward vel = 380 ft/sec, lat vel = 0 ft/sec, LPD = 52, thrust = 5100#, fuel =
FA+30	Verifies LPD = 52° and on-trajectory	Calls out LPD reading alt and alt rate (interphone only).
FA+40	Continues visual LPD monitoring and trajectory and site assessment checks	Verifies and calls out over interphone and VHF command alt = 2500 feet, alt rate = 65 ft/sec, pitch = 38°, forward vel = 260 ft/sec Lat vel = 0 ft/sec, LPD = 52, thrust = 4700#, Fuel =

Table 5 (Continued)

TIME (SEC)	COMMANDER	SYSTEMS ENGINEER
FA+50	Verifies visually LPD = 52° and that trajectory and site good.	Checks AGS alt = alt rate = Calls out LPD, alt, alt rate (interphone only)
FA+60	Verifies LPD = 52° and continues trajectory and site assessment.	Verifies alt = 1200 feet, alt rate = 35 ft/sec, pitch = 35° for vel = 140 ft/sec, LPD = 52° lat vel = 0 ft/sec, thrust = 4200#
F+70	Verifies LPD = 52° and trajectory and site are good.	Verifies and calls out over phone alt = 800 feet, alt rate = 25 ft/sec, pitch = 32° for vel = 90 ft/sec, thrust = 4000#
F+75	Rotates LEM from 30° to 11° pitch back attitude. Switches LR to second position.	Switches PNGS from AUTO to RCAH, manually. Verifies fuel quantity level is satisfactory and relates this to Commander.
L+0	Evaluates vehicles handling qualities. Verifies pitch att = 11°, alt = 500 feet, alt rate = 15 ft/sec, lat vel = 0, for vel = 50 ft/sec Verifies landing site satisfactory.	Calls out lo-gate over VHF command and interphone. Monitors flight instruments and begins terrain assessment from right-hand window
L+10	Verifies pitch att = 11° alt = 350 feet, alt rate = 15 ft/sec, for vel = 40 ft/sec Checks landing site. Reduces alt rate to 8 ft/sec. Evaluates performance of rate-of-descent command mode. Estimates engine gimbal angle (roll attitude for zero $\dot{Y}$)	Checks fuel status. Advises commander of landing site assessment.

Table 5 (Concluded)

TIME (SEC	COMMANDER	SYSTEMS ENGINEER
L+30	Verifies pitch att = 11° alt = 190 feet, for vel = 20 ft/sec, lat = 0. Reduces lat rate to 5 ft/sec. Checks landing site. Evaluates performance of rate-of-descent command mode. Estimates gimbal angle (roll).	Advises Commander of fuel status.
L+40	Landing site passes from view. Alt = 140 feet alt rate = 5 ft/sec forward vel = 10 ft/sec lat vel = 0 Checks for dust agitation.	Advises Commander of landing site, fuel status.
L+45	Begins rotation forward and nulls forward and lateral velocities.	Checks for final radar update of PNGCS.
L+50	Continues to null forward and lateral velocities. Notes alt = 90 feet, alt rate = 5 ft/sec. Checks fuel. Compares forward and lateral velocity null visually. Estimates pitch and roll component of engine gimbal angle.	Advises Commander of fuel
L+60	Continues velocity nulling. Notes alt = 40 feet. Checks fuel. Estimates pitch and roll component of engine gimbal angle.	Continues advisement.
L+62	Continues velocity nulling. Notes alt = 30 feet, reduces alt rate to 4 ft/sec. Arms DE shutoff switch.	Continues advisement
L+70	Shuts off DE on probe light indication	

TIME	COMMANDER										SYSTEMS ENGINEER					
	ALT		ALT RATE		CROSSRANGE VELOCITY		VELOCITY CHECK	T/W	ΔV	FUEL	ALT		ALT RATE		CROSSRANGE VELOCITY	
	PNGS	LR	PNGS	LR	PNGS	LR	MSFN	Meter	AGS	Meter	PNGS	AGS	PNGS	AGS	PNGS	AGS
0			0		0		5550	0.1					0 0	0 0	0 0	0 0
100	49,000 48,500		30 25		+1 -1		4900	3.4			49,000 48,500	49,500 48,500	30 25	35 25	+1 -1	+2 -2
200	43,000 42,000		140 125		+1.5 -1.5		3760	3.9			43,000 42,000	44,000 41,500	140 125	140 120	+1.5 -1.5	+3 -3
300*	28,000 26,000		170 150		+2 -2		2400	4.5			28,000 26,000	29,500 24,500	170 150	180 155	+2 -2	+5 -5
370**			135 115		+2 -2								135 115	150 100		
400	14,500 10,500		155 135		+2.5 -2.5		1100	5.3			14,500 10,500	17,000 8,000	155 135	175 120	+2.5 -2.5	+7 -7

Table 6 - Reference Trajectory Monitoring Charts

*Note Decrease of $\dot{h}$

**Note Negative Increase of $\dot{h}$

	COMMANDER										SYSTEMS ENGINEER					
TIME	ALT		ALT RATE		CROSSRANGE VELOCITY		VELOCITY CHECK	T/W	ΔV	FUEL	ALT		ALT RATE		CROSSRANGE VELOCITY	
	PNGS	LR	PNGS	LR	PNCS	MSFN/LR	MSFN	Meter	AGS	Meter	EXP	ACT	EXP	ACT	EXP	ACT
0	50,000		0		0		5550	0.1			0		0		0	
100	49,000		30		±1		4900	3.4			300		6		6	
200	42,500		130		±2		3659	3.9			1100		12		13	
300*	27,000		160		±2		2400	4.5			2800		19		17	
370**			125										26			
400	14,000		150		±3		1100	5.3			5000		27		24	

Table 7 - Monitoring Chart for PNGS-AGS Trajectory Differencing

*Note Decrease of $\dot{h}$

**Note Negative Increase of $\mathring{h}$

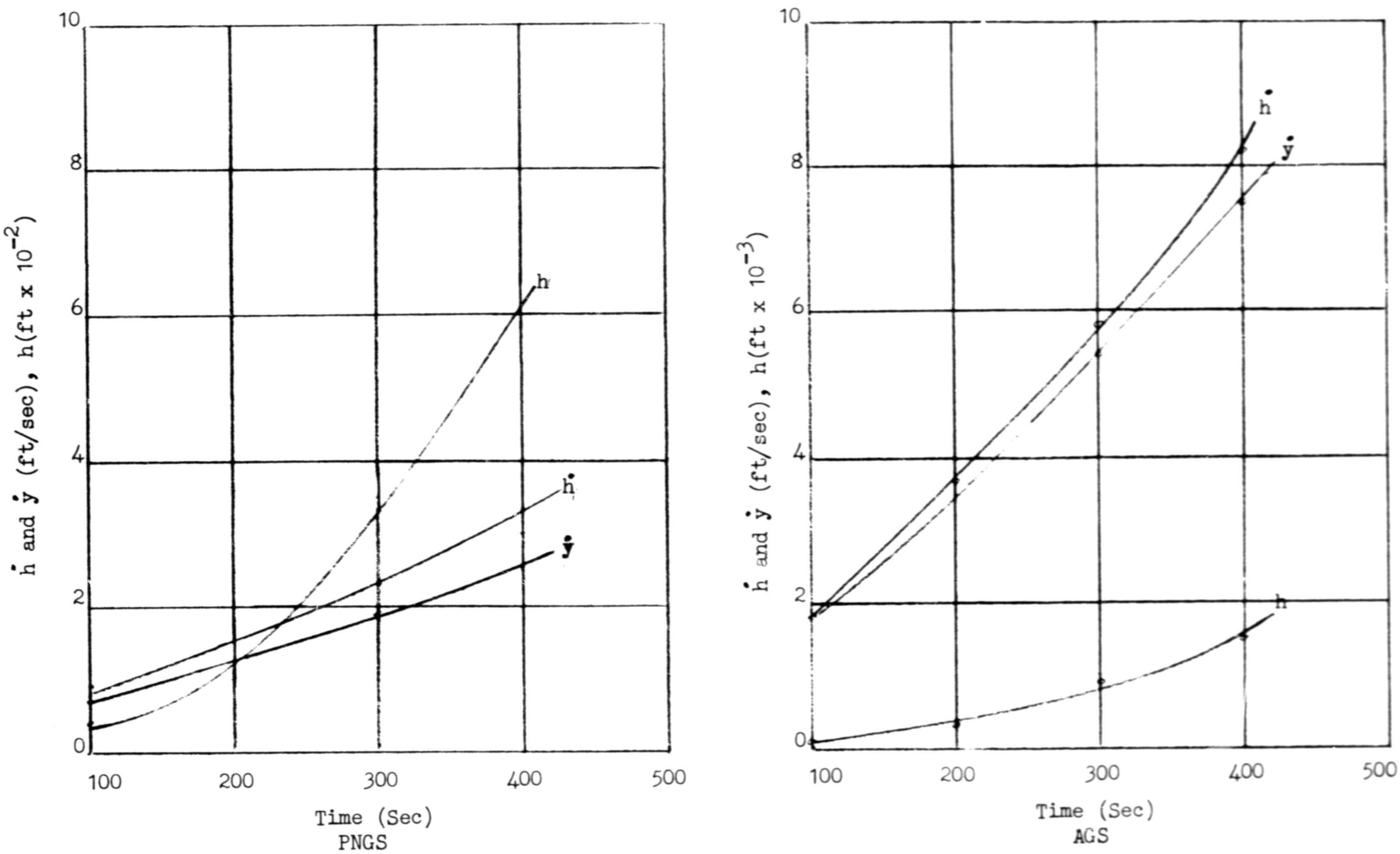

Figure 1 - PNGS and AGS 1σ Trajectory Deviations

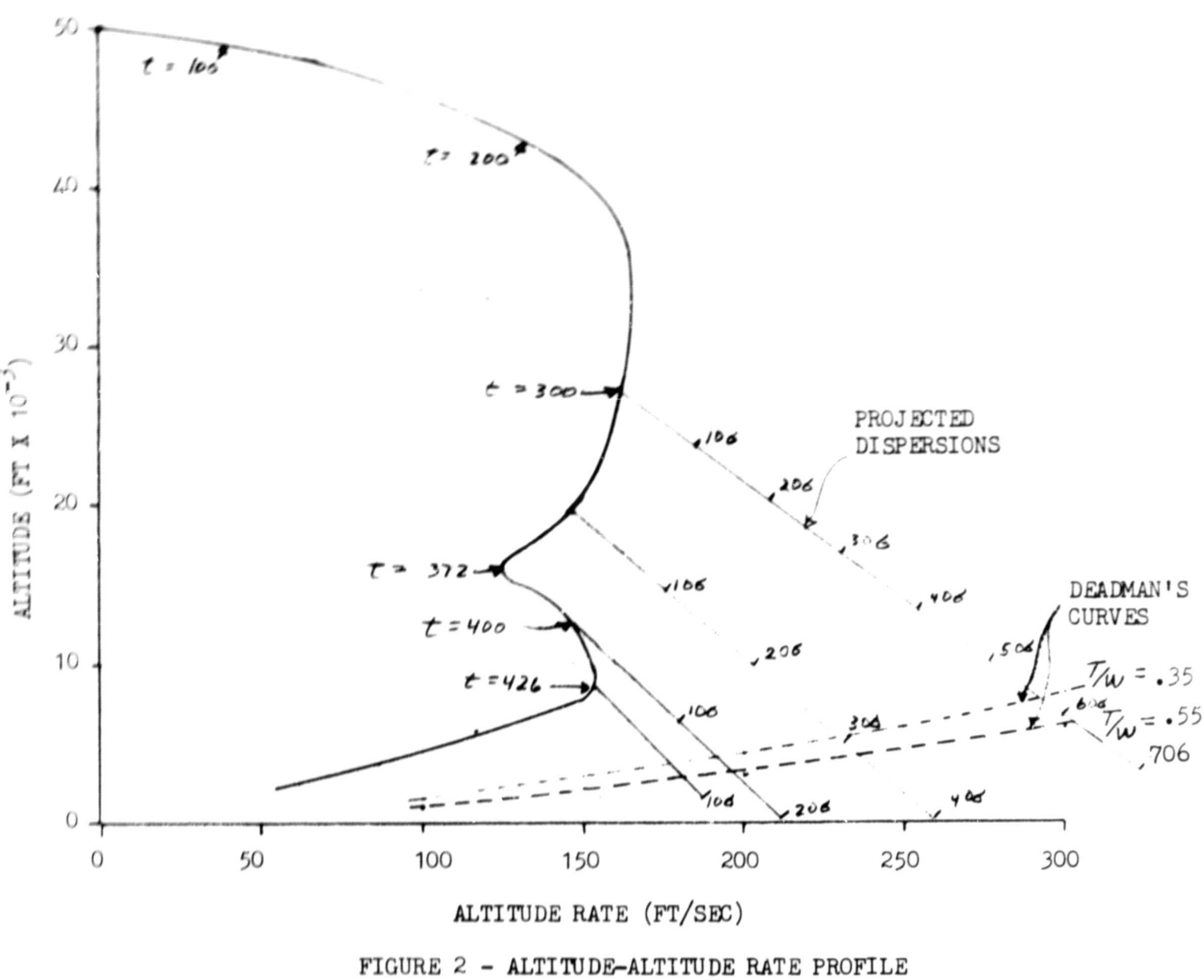

FIGURE 2 - ALTITUDE-ALTITUDE RATE PROFILE

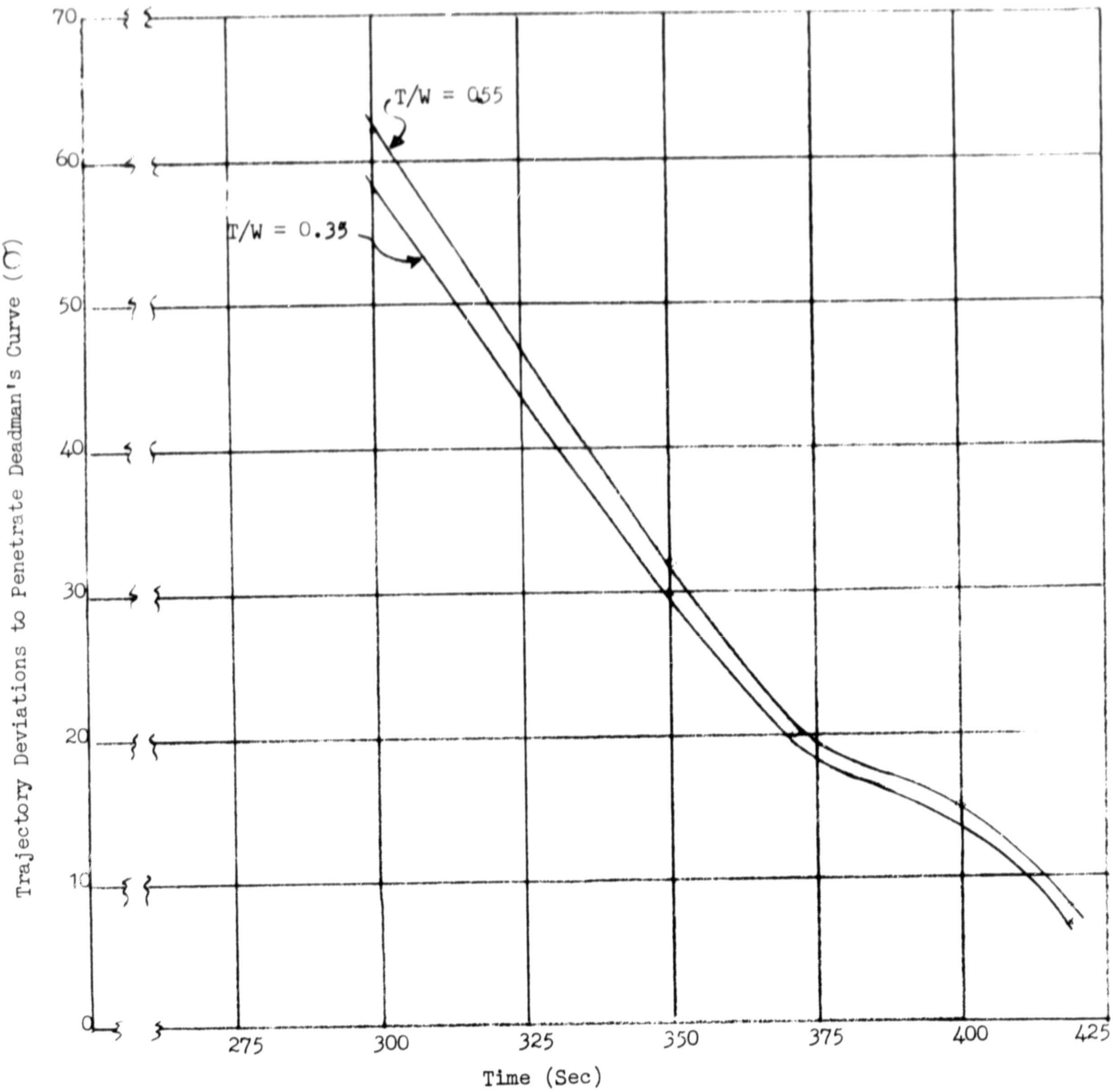

Figure 3 - Effect of T/W Ratio on Deadman's Curve Penetration