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Application of Titan III Guidance and Navigation System to Lunar Missions

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Introduction

The current series of Titan III R and D flights includes missions of considerable complexity. The mission profiles are designed to take full advantage of the combination of the Titan IIIC vehicle configuration and of the accuracy and flexibility provided by the AC Electronics Guidance and Navigation (G & N) System.

The fact that the Titan IIIC is capable of completing a synchronous equatorial mission from an ETR launch implies that a lunar impact mission could be flown with a payload having two to three times the synchronous payload weight. This paper discusses the changes which would be required in the present Titan III guidance hardware and software to perform a certain class of lunar missions. The changes considered are those that require minimum modification of the existing Operating Ground Equipment (OGE), Missile Guidance Computer (MGC) ground programs, and MGC flight equations.

The accuracy of the G & N System is of interest to the payload user; since any decrease in the midcourse correction requirements can immediately be translated into an increase in useful payload weight. One of the primary considerations in the discussion of system modifications is that of minimizing system errors. The propagation of system errors and the lunar impact sensitivity coefficients are treated in some detail.

Mission Definition and Requirements

Payloads

Most of the mission constraints and requirements are ultimately determined by the payload. The types of payloads considered here are those designed to transmit information about the lunar surface. Typical payload requirements include viewing and lighting constraints before and perhaps after lunar impact, as well as constraints on the velocity of the payload either at the moon's sphere of influence or at lunar impact.

Because of the mission limitations, a suitable launch window exists only on certain days during the lunar month. Several factors are responsible for these limitations: (1) range safety considerations at ETR exclude all but a small range sector of launch azimuths, (2) accuracy requirements limit the length of the parking orbit coast time, and (3) requirement for visibility of lunar impact from earth tracking stations restricts the launch opportunity, both with respect to the time of the month and the time of day.

Payloads which soft-land on the moon have rather stringent lighting and viewing constraints before and after landing. The number of launch opportunities for such payloads each month may be limited to about seven. On the other hand, payloads requiring only that lunar impact be visible from one or more tracking stations can be launched from ETR on twenty or more days each month. The launch schedule for the latter payloads is considerably more flexible, but also requires more targeting to provide coverage for late schedule changes.

Reference Trajectory

Given only the requirement of lunar impact, launch could occur at any time and lunar impact could be attained along any one of an infinite number of trajectories. Practical limitations, however, are such that only flight times of about two or three days can seriously be considered for Titan III application. With this constraint, the launch opportunities for which a direct inject trajectory may be used are very limited. The logical choice is a trajectory that includes a period of coast in a parking orbit.

As will be seen, the adaptation of the Titan III guidance philosophy to a lunar mission requires that the trajectory be unique for a given time of launch. The guidance is adaptive only to the extent that compensation for non-nominal vehicle performance may be included. The form of the nominal trajectory is, therefore, the same throughout a launch window; that is, the sequence of guidance events remains unchanged.
The basic trajectory used for lunar guidance studies is composed of segments similar to those in existing Titan III missions. The first segment is a boost phase into a circular parking orbit of about 100 nautical miles. This is followed by a coast segment during which the vehicle is maintained in a "belly-down" orientation by an attitude control system. The third segment is a second powered flight phase which terminates in the injection of the payload into an orbit that intersects the moon.

In order to make the trajectory unique for a given time of launch, it is necessary to specify the azimuth of the orbit plane and the length of the coast. In general, the G & N System errors will increase with increasing total flight time from liftoff to payload release. Since the assumption of minimum coast time (hence, minimum total flight time) for a given azimuth is a convenient targeting criterion, most of the studies have been carried out with this consideration.

**Launch-on-Time**

The most significant difference between a lunar mission and an earth-orbital mission is the change from a two-body problem to a three-body problem. The moon's ephemeris must be included in the guidance equations in some form. The Titan III guidance philosophy in its present form is not compatible with the explicit use of the moon's motion. However, the launch azimuth and range angle do appear in the Titan III guidance equations. The range angle is essentially equivalent to the time of coast. From the results of targeting, the time dependence of the azimuth and range angle can be determined and these quantities initialized in the flight program at liftoff.

Figure 1 shows the time dependence of the launch azimuth and total range angle to translunar injection for the 26 May 1966 launch window. The details of the trajectory will be discussed below in connection with Table 1 and Figure 4. The shape of the curves in Figure 1 is typical of those for any of the missions investigated. The most notable feature is the nearly linear character of the curves, suggesting that a simple, yet accurate, functional representation should be easy to generate.

**Guidance and Navigation System Modifications**

**Missile Guidance Computer**

The form of the reference trajectory was chosen specifically to make use of parts of current Titan III MGC flight programs. This decision was strongly influenced by the ease of implementation in the flight computer. Since nearly all of the necessary equation blocks are immediately available, the computer capacity requirements for a lunar mission can be estimated with very little uncertainty.

The problem of computer capacity is a very real one for any system which uses an airborne computer. The Titan III MGC is similar to the Titan II computer, except that the drum has been lengthened to increase the total memory by 35 percent. The Titan III drum contains 78 tracks, with locations for 9,792 instructions and 1,152 constants. The incorporation of a "wired-in" divide has also increased the programming efficiency by allowing parallel computations.

The most complex program yet written for the Titan III computer is for the synchronous satellite mission. This mission includes three Stage III (restartable final stage) burns, two plane changes, eight payload ejections (each at a different velocity), and several other maneuvers during the coasting phases. The MGC capacity requirements of this program far exceed those of the lunar missions discussed here. The conclusion is that the present Titan III computer is completely adequate.

**Flight Programs**

**Flight Initialization.** During the final phase of the Titan III countdown, a discrete signal, FLY, is sent to the flight computer through the ground equipment. Three seconds after the return signal, FSEQ (Flight Sequence) has been received from the MGC, the engines are fired. Forty-five milliseconds after the MGC sends FSEQ the computer program enters the first major cycle of flight computations. The inertial platform, which had been slaved to earth-fixed coordinates, is set free in inertial space at the same time.

The flight equations are initialized in the first major cycle. For a lunar mission two of the initialization equations are for the azimuth and range angle. Two polynomials are constructed by fitting the curves in Figure 1. The value of time is set in from an accumulator which was initialized at some earlier point in the countdown. The polynomials used might apply to the total launch window or to only part of the launch window. The higher the order of the polynomial used, the greater the range of times over which one would expect it to apply. The linearity of the curves in Figure 1 is such that there would be no need for more than one pair of polynomials for the entire launch window.

**Flight Equations.** The Titan III guidance equations, which can be directly applied to generate the trajectory described above, employ aim point values of radial distance (R), radial rate (R), and speed (V). Guidance shutdown is accomplished by use of time-to-go, Tg, which is a function of the measured thrust acceleration and the difference between the aim point speed, Vf, and the measured speed, V. Since Tg approaches zero as (Vf - V) approaches zero, the final speed is controlled directly.
If the mission requirements can be fulfilled by injecting into the translunar orbit with a constant energy throughout the launch window, the aim points are independent of the launch time. The variation in the total flight time will be almost entirely determined by the variation in the parking orbit coast time.

If constant time of arrival at the lunar surface is a mission requirement, the aim point values can no longer be held constant during the launch window. These values can be treated in the same manner as the azimuth and range angle. They can be expressed as polynomials in time and computed in the MGC initialization block at the start of flight computations. The coefficients of these polynomials would also have to be changed from day to day in the event of a launch delay.

The targeting for a mission is done under the assumption of nominal missile and environmental characteristics. Non-nominal conditions will introduce dispersions in the injection position and velocity. Since the engines cannot be throttled, it is impossible, even in theory, to maintain the same state vector for both nominal and non-nominal cases.

The guidance equations control $R$, $\dot{R}$, and $V$, but do not control the range angle. This does not present any problems in an earth-orbital mission, but it can lead to large errors in lunar impact missions. The start of the second Stage III burn is controlled by the nominal range angle. The time might be slightly different for non-nominal boost conditions, but this introduces very little error at lunar impact. However, non-nominal thrust during the second burn results in range angle dispersions that require compensation.

Fortunately, the method of in-flight compensation is quite simple. An error in range angle can be almost totally compensated for by adjusting the flight path angle. The relationship is nearly linear and is such that a one-degree change in range angle requires about 1/2-degree change in flight path angle. The final range angle can be predicted during the burn and the final flight path angle computed. The aim point value of $\dot{R}$ is then changed to yield the proper end conditions.

If a vernier trim is required after the final Stage III shutdown, the propellant settling mode of the attitude control system (ACS) can be used to adjust the total energy. The thrust during the propellant settling mode is 180 pounds versus 16,000 pounds for the main Stage III engines. Because of the much smaller thrust level, the uncertainty in ACS shutdown is correspondingly smaller so that the final orbital energy can be closely controlled even for light payloads.

The axis of spin stabilized satellites can be oriented in any direction prior to release. Since the G & N System errors are very small, the accuracy of the orientation is determined by the limit cycle of the ACS. For the configuration of the ACS currently in use, the accuracy is ±0.5 deg. Since the quantization of the gimbal angles is 0.17 deg, little would be gained by reducing the amplitude of the limit cycle very far below its present value.

Ground Programs

The establishment of an accurate time reference is a special problem and requires minor changes to the existing MGC ground programs and countdown procedures. The new ground programs include a mode in which the MGC searches for a discrete signal from the OGE. This discrete signal is sent at a predetermined time of day. The computer then accumulates time subsequent to receipt of this signal. The overall error introduced by this scheme can be reduced to about 1 ms. This magnitude of error has been found to give insignificant azimuth and range angle errors.

For launch delays of from one to several days, the coefficients in the polynomials for azimuth and range angle must be changed on a daily basis. On Titan III, three of the constants tracks on the MGC drum can be addressed directly from the OGE van. Up to $64 \times 3 = 192$ constants can be changed in this manner without removing the computer from the missile. This is certainly a sufficient number to provide for changing all of the launch-date-dependent constants.

Guidance and Navigation System Accuracy

Guidance Equation Mechanization Errors

The significant new error sources that appear in the lunar mission equations are the launch-time-dependent initializations and the range angle compensation. All other guidance equation error sources are present in the current Titan III programs. The contribution of the current error sources in the guidance equations is only a small fraction of the total G & N System error, and this contribution would not be expected to change significantly for lunar missions.

The polynomials representing the azimuth and range angle must be very accurate in order to avoid large position errors at payload release. An error of 1 arc-min in either of these angles can lead to an error of 6,000 feet in range or out-of-plane position at injection.

The sources of errors in these polynomials are the targeting routine, the polynomial curve fitting routine, and the value of time. The accuracy of the data determined from the targeting routine is limited only by the validity of the simulation as a true representation of the real world. The polynomial curve fit errors can be minimized by the proper choice of the degree of the polynomial, coupled with optimum scaling of the MGC equations. However, the latter is the largest of the three error sources.
As mentioned above, the error introduced by the uncertainty in the value of time can be ignored.

Calibration and Compensation

Ground Programs. To realize the inherent accuracy of the G & N System, the MGC ground and flight programs include the pre-flight calibration of, and the in-flight compensation for: (1) accelerometer bias, scale factor, and nonlinearity, and (2) gyro drift, spin axis unbalance, input axis unbalance, and compliance. The accelerometer and gyro coefficients are first measured on the tilt-table in the AC Electronics laboratory at ETR. The accelerometer bias and scale factor are recalibrated by a special ground program after the Inertial Measurement Unit (IMU) has been installed in the missile. The final calibration enables updating of the accelerometer bias and gyro spin axis unbalance coefficients during the final countdown.

Flight Programs. The navigation equations in the flight programs process the raw accelerometer counts in each channel by applying the final accelerometer coefficients. The gyro coefficients are used to calculate the drift rate of the platform to find the matrix relating the orientation of the drifted platform to the platform orientation at launch. The final transformation to the earth-centered inertial computational frame includes the measured angular misalignments of the accelerometers on the platform.

The gimbal angle commands computed in the guidance equations may or may not include compensation for the known platform drift. If the guidance equations contain some form of compensation which applies to thrust misalignment from any source during a powered flight phase, compensation for platform drift does not have to be explicitly inserted. However, if the attitude is specified by gimbal angle commands which are input constants (for example, at payload release), direct compensation for the known drift is required for maximum accuracy.

Error Propagation

The time histories of the G & N System errors during a typical flight are shown in Figures 2 and 3. Numerical values are not shown because of the security classification of the information. The $3\sigma$ vector displacement and velocity errors are resolved into radial, normal, and tangential components. These directions are determined by $R$, $R \times V$, and $(R \times V) \times R$, respectively, where $R$ and $V$ are the nominal radius vector and velocity. The errors are found by differentiating the non-nominal $R$ and $V$ and the nominal $R$ and $V$ at the same value of time.

The case illustrated is for a mission with more than 1/2 orbit of coast. This is somewhat longer than the longest mission, which is consistent with the assumption of minimum coast time for a given azimuth.

However, it is used to show the behavior of the errors during coast. In general, the errors increase during the first burn and then oscillate during the coast. The second burn occurs at 4,000 seconds and yields a net increase in the magnitudes of the position and velocity errors. Because of the correlation assumed between the errors in the first burn and those in the second burn, some of the component errors actually decrease during the burn.

If the second burn had been initiated at an earlier point in the coast, the errors would have been the same up to that point. Assuming that the launch azimuth varies from 90 deg to 114 deg (Titan III ETR launch), the time histories for any available azimuth will look qualitatively the same as those shown, except for the time of the second burn. Quantitatively, the principal difference will be the errors added during the second burn. These errors will decrease as the coast time is made shorter.

Because of the periodic behavior of the errors during coast, the error in a given component might either decrease or increase as the coast time increases. Although one expects intuitively that the magnitudes of the vector displacement and velocity errors increase with time, the distribution of the errors among the components changes such that some decrease while others increase. Since the effect of, say, a radial error at injection on the error in lunar impact position is not the same as that of an equal normal or tangential error, not only the magnitude but also the direction of the vector displacement error determines the size of the CEP on the moon. By including the error analysis results in the targeting routine, it would theoretically be possible to replace the criterion of minimum coast time for a given azimuth by the criterion of minimum CEP for a given azimuth. Implementation of the scheme would greatly complicate the targeting, although the principle is not unlike that of Kalman filtering techniques for aided-inertial systems.

Lunar Impact Errors

The data presented in Table 1 and Figures 1 and 4 are all derived from the same set of trajectories. Launch is from ETR with a launch azimuth that varies from 90 deg to 114 deg. Lunar impact occurs within sight of both the Goldstone and Woomera tracking stations with a minimum elevation angle of 5 deg. The optimum launch azimuth for viewing is 102 deg. The inertial velocity vector at lunar impact is parallel to the radius vector from the moon's center. The selenographic latitude and longitude of impact are variable throughout each launch window as well as from day to day.

The values in Table 1 cover almost the entire range of May 1966 launch opportunities for a three-day mission. For comparison, the same entries are also shown for a two-day mission that impacts
the moon on the same date as would the 26 May launch. The angular error error on the moon (in degrees) is given for the indicated injection errors in the parameters listed. For example, if the injection radius is changed by 1,000 feet, the square root of the sum of the squares of the resulting changes in selenographic latitude and longitude is 2.60 deg on 9 May.

A 1 arc-min error in declination or right ascension represents a position error of about 6,000 feet as compared with the 1,000-foot reference error for the radius in Table 1. Comparison of the corresponding lunar impact errors shows that the 1,000-foot radial error yields larger impact errors than the 6,000-foot error in either declination or right ascension.

In the same way, an error of 0.01 deg in flight path angle or azimuth represents a velocity component error of nearly 6 ft/s. The sensitivity to errors in speed is seen to be at least six times the sensitivity to errors in the other velocity components. The variation in the magnitudes of the impact errors for errors in radius and speed is also the greatest, increasing by a factor of about three from 9 May to 26 May.

It is not surprising that the errors in radius and speed are the most significant since these two quantities essentially determine the energy, which in turn determines the time of flight. Errors in the tangential position and radial velocity have the effect of rotating the orbit about an axis perpendicular to the orbit plane. Errors in the out-of-plane position and velocity components have the effect of rotating the orbit plane. Neither of these two rotations produces errors comparable with those introduced by changing the shape of the trajectory and the time of flight as a result of changing the energy.

A further insight into the reasons for the variation in sensitivity to orbital energy changes can be gained from Figure 4. The figure contains two curves. The first is a plot of the radial distance of the moon from the earth's center during May 1966 as a function of the right ascension of the moon. The other curve is a plot of declination versus right ascension.

Each of the circles along the latter curve represents the time of impact for a reference trajectory having an azimuth of 102 deg. The dates of impact are 12 May 1966 through 1 June 1966. The two arrows associated with each of these points show the directions of the moon's velocity and the velocity of the satellite at lunar impact. This angle remains less than 10 deg from 12 May to 20 May and then increases rapidly to 74.25 deg on 26 May.

If a satellite approaches the moon in a direction nearly parallel to the moon's velocity, the effect of a difference in time of arrival at the moon's sphere of influence on the impact position would be expected to be less than if the angle of approach were large.

The focusing by the moon's gravitational field is more effective if the descent to the lunar surface is more nearly along the radius vector. The errors in radius and speed produce the largest variation in the time of flight, and a comparison of the values in Table 1 with the data in Figure 4 shows that these errors are the largest when the angle between the velocities is the greatest. The most meaningful comparison is between the 13 May launch and the 26 May launch, since the flight times are nearly the same.

The correlation of the Table 1 errors with the radial distance from the earth to the moon is much less direct. Intuitively, one would expect that the errors might decrease as the flight time increases. This effect is masked in Table 1 by the difference in flight times. The dispersions increase as the flight time increases from 67 hours for a 1 May launch to 74 hours for an 18 May launch. This increase in flight time more than compensates for the decrease in distance travelled.

The Table 1 entries for the 46-hour mission show that the decrease in transit time does not make any significant difference in the errors. Since the trajectory is nearly a straight line along the line of sight at injection, the errors in flight path angle and azimuth become more significant and the errors in speed and radius less significant. Also, the velocity at lunar impact is greater so that the effect of the relative velocity at impact is less. For most payloads of interest, the larger velocity at impact is a disadvantage. The three-day mission time appears to be the most satisfactory for the cases studied.

Conclusion

The modifications to the current Titan III G & N System hardware and software to accommodate a lunar mission of the class described above are minor. The only hardware changes involve the ground equipment. One new MGC ground program is required to maintain an accurate time reference in the computer. Most of the necessary flight equations already appear in existing Titan III programs.

In general, it can be stated that the accuracy of the Titan III G & N System is such that the CEP on the moon has a radius much less than the moon's radius. By observing the way in which the system errors propagate and the variation of the lunar impact errors with the launch date, careful scheduling and targeting can result in a minimum CEP. It is, therefore, possible to seriously consider the Titan III G as a launch vehicle for lunar satellites that have no mid-course correction capability. For payloads requiring pinpoint accuracy, the satellite mid-course correction requirements would be minimal.
<table>
<thead>
<tr>
<th>PARAMETER (Change in Value)</th>
<th>9 May 1966 67 Hours</th>
<th>13 May 1966 70 Hours</th>
<th>18 May 1966 74 Hours</th>
<th>26 May 1966 70 Hours</th>
<th>27 May 1966 46 Hours</th>
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<tr>
<td>Radius (1,000 ft)</td>
<td>2.60*</td>
<td>3.30</td>
<td>5.42</td>
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<td>2.76</td>
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<td>Declination (1 arc-min)</td>
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<td>2.09</td>
<td>1.95</td>
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<td>1.05</td>
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<td>Speed (1 ft/s)</td>
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<td>4.04</td>
<td>6.60</td>
<td>9.03</td>
<td>3.35</td>
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<tr>
<td>Flight Path (0.01 deg)</td>
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<td>1.48</td>
<td>2.46</td>
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<td>0.52</td>
<td>0.42</td>
<td>0.26</td>
<td>0.70</td>
</tr>
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*Errors given in degrees of selenographic latitude and longitude dispersions

Table 1. Lunar Impact Errors
Figure 1. Launch Azimuth and Range Angle Versus Launch Time for 26 May 1966 Launch Window
Figure 2. Time History of Position Errors
Figure 3. Time History of Velocity Errors
Moon and probe position projected upon the Earth at the time the probe reaches the Lunar sphere of influence, for each 102° reference trajectory of May, 1966. Vectors show relative direction of motion of probe and Moon. $\Delta AZ$ is the difference in azimuths of these vectors. Also shown is the Radial Distance of the Moon for May, 1966, versus Right Ascension.

- $\Delta AZ = 5.74°$ (May 18)
- $\Delta AZ = 8.55°$
- $\Delta AZ = 5.65°$
- $\Delta AZ = 6.53°$
- $\Delta AZ = 34.44°$
- $\Delta AZ = 3.47°$
- $\Delta AZ = 5.65°$
- $\Delta AZ = 72.18°$ (May 26)
- $\Delta AZ = 74.25°$ (May 22)
- $\Delta AZ = 67.25°$
- $\Delta AZ = 5.53°$
- $\Delta AZ = 73.63°$
- $\Delta AZ = 6.31°$
- $\Delta AZ = 5.37°$
- $\Delta AZ = 5.42°$ (May 13)
- $\Delta AZ = 7.18°$
- $\Delta AZ = 6.31°$
- $\Delta AZ = 5.74°$
- $\Delta AZ = 9.41°$ (May 9)

Figure 4. Lunar Impact Conditions for May 1966