A System for In-Flight Determination of Lunar Orbit Injection Conditions

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SUMMARY

This paper presents a description of the Douglas AIMP-E Real Time (DART) System and its operation during the translunar flight of Anchored Interplanetary Monitoring Platform-E, which is now in lunar orbit and designated Explorer XXXV. The DART system is basically an on-line computer program for determination of the fourth stage (retro-rocket) ignition time and orientation which yield lunar orbits that best satisfy mission objectives. This paper describes the history of the AIMP-D and E missions, the development of the computer software used for the missile preflight analysis, and integration of that software into a real time computer program which ultimately became DART. The program gained a significant speed advantage over standard methods of orbit prediction by making use of a method of dynamic lifetime prediction developed by one of the authors. This speed advantage enabled probability analyses to be performed in real time. Results obtained from these analyses are shown to provide a far better basis for decision-making than do solely deterministic data. Included in the paper are samples of the program output and graphs which show how the information was presented to the mission director during an actual mission. The authors feel that one of the primary reasons for the success of the DART system was the very close man-computer relationship inherent in the system's design. In this paper emphasis is placed on that relationship.

1. INTRODUCTION

The Anchored Interplanetary Monitoring Platform (AIMP) Program was established for the purpose of placing two scientific information-gathering spacecraft (AIMP-D and AIMP-E) into lunar orbits by means of the Improved Delta Launch Vehicle. The nature of the program was such that in-flight decisions regarding the use of control parameters had to be made in a short period of time. A computational system was required to provide the mission director with sufficient information to make these decisions. The necessary characteristics of such a system are the following:

a. rapid response time
b. flexible operation
c. clear, concise output

Efficient response to a variety of situations, both foreseeable and unforeseeable, was essential. The Douglas AIMP Real Time (DART) system was designed to meet these needs. For the AIMP mission, a flexible procedure of successive refinements in the generation of data was required. Rapid simulation techniques were needed to establish approximate firing time and attitude selections for a retro-rocket in the vicinity of the moon. Probability analysis capability was also needed since uncertainties would exist in the real time tracking data, retro-rocket attitude, and in the performance of the retro-rocket. As will be shown in the paper, probability analyses and the determination of the attitude orientation of the retro-rocket each involve the simulation and examination of literally thousands of lunar orbits. Without the availability of an approximate method for rapidly determining the dynamic lifetime of lunar orbits, performance of these analyses in the limited time available for real time operation would not have been possible. As regions of interest were established using approximate techniques, integration of selected lunar orbits was desired to confirm results obtained from approximate solutions and to aid in establishing final values for time of retro-fire and orientation of the retro-rocket.

All information generated by the DART program was presented in an output format which is easily identifiable, concise, and familiar to the mission director. An on-line output was incorporated into the program to enable the operations engineers to monitor data as they were generated.

This paper describes the DART system and its operation at the Goddard Space Flight Center during the launch of the AIMP-E spacecraft (now designated Explorer XXXV) in July of 1967. Emphasis is placed on the need for probability information in support of real time decisions. It is shown that incorrrect decisions can result when only deterministic information is considered in the decision process. The paper stresses the need for careful design of the man-machine interface in the computation system. An effective system must utilize the flexibility and judgement of the operations engineer while making fully advantageous use of the high-speed digital computer.

2. MISSION DESCRIPTION

In 1964 the National Aeronautics and Space Administration awarded the Douglas Aircraft Company a study contract to analyze the task of placing a spacecraft in orbit about the moon by use of the Improved Delta vehicle. Primary emphasis for such a mission was to be the attainment of a lunar orbit with a dynamic lifetime in excess of six months to permit prolonged gathering of scientific data in the vicinity of the moon. Early feasibility analyses revealed that a high probability of achieving an initial lunar orbit with the Delta vehicle could be expected without the use of a midcourse correction system. This fact, coupled with the limited spacecraft weight capability of the Delta vehicle, led to the decision to proceed with mission analysis assuming the absence of any midcourse correction system.

With no midcourse correction system, trajectory errors introduced by the launch vehicle, particularly those caused by the spin-stabilized solid propellant third stage, propagate through the transfer trajectory and thereby cause large, nonlinear dispersions in the vicinity of the moon. These potential large dispersions and the availability of only two control variables, viz., fourth-stage ignition time and attitude of the fourth-stage impulse, led to analysis methods and results which are unique to this lunar mission. The mission analysis objective, basically, was to design a launch trajectory which would maximize the probability of achieving a lunar orbit with greater than six months dynamic lifetime and to select suitable launch opportunities for that trajectory. This task implied the need for probability analyses which required, in order that they be performed at reasonable cost, a rapid means for approximating lunar orbit lifetimes given initial lunar orbit conditions. It was necessary to develop a means of describing the dynamic behavior of lunar orbits for a complete spectrum of initial lunar orbit conditions. The highly nonlinear effects resulting from launch vehicle dispersions dictated the need for Monte Carlo analyses used in conjunction with a rapid lifetime determination method to obtain meaningful probability data. Numerical integration of each sample orbit in a Monte Carlo analysis would require exorbitant amounts of computer time. Such analyses would undoubtedly be required for numerous trial design trajectories, and they would create a prohibitive analysis cost if no approximate lunar orbit lifetime program were available.

The results obtained from the study effort were most encouraging. A detailed description of that effort and its findings is available for the interested reader.[1] Those tools developed under the analysis effort which became integral parts of the DART system are listed below:

a. Satellite Lifetime Analysis Program (SLAP): SLAP is an approximate method for determining lunar orbit dynamic behavior given initial orbit conditions.

b. GAUSS: GAUSS is a four-body numerical integration program used for obtaining a detailed description of lunar orbit behavior.

c. Monte Carlo Statistical Analysis Program: This program combined Monte Carlo capability with an interplanetary trajectory simulation program to enable statistical analyses of pertinent mission parameters.

A detailed description of these tools will follow in section 2.1 of this paper.

Based on the results of the mission analysis effort, plans were made to launch two spacecraft (AIMP-D and AIMP-E) in the summers of 1966 and 1967, respectively. In addition to the orbit lifetime requirement, several other mission requirements were established. Also established was the need for presentation of several trajectory parameters in real time. It is not the intent of this paper to present the reasons for the various mission and output data requirements. It suffices to state that they arose either from basic hardware limitations, or from their significance to the scientific experiments associated with the mission. It is, however, necessary to state the various mission constraints and required output parameters since the DART system was designed to present data pertinent to those needs. Before specific mission requirements are stated, the reader should be acquainted with the following fact. Because a finite probability exists that primary mission objectives could not be met given certain combinations of boost vehicle performance anomalies, provisions were made to select an alternate mission which would yield a large percentage of the desired scientific data in the event the actual launch resulted in a condition indicating a low probability of achieving a satisfactory lunar orbit. The alternate mission would be a highly elliptic earth orbit achieved by an early firing of the spacecraft retro-rocket. Figure 1 provides a pictorial view of the primary and alternate missions. Preliminary analysis results indicated that a retro-rocket firing as early as three hours after launch could be required to achieve the desired alternate mission orbit.

A major objective of the DART system is to determine rapidly whether to proceed with the primary mission or to elect the alternate mission. This decision will henceforth be referred to as "GO/NOGO" with regard to primary mission requirements. The providing of sufficient information to make this decision adequately would require knowledge of the probability of satisfying primary mission requirements.

The mission requirements for the AIMP missions were as follows:

Primary Mission

a. orbit lifetime $\geq$ 6 months (desired)

b. 2250 km (radius) $\leq$ pericynthion $\leq$ 3500 km (radius)

c. largest shadow duration < 2.5 hours

d. shadow to sunlight ratio < 15% average during lifetime

Alternate Mission

In the event the mission requirements for the primary mission could not be satisfied, the following criteria would be used to select the retro-rocket firing time for an alternate (earth orbit) mission:

a. apogee altitude = 450,000 km

b. orbit lifetime $\geq$ 6 months with minimum shadow conditions

c. perigee $\geq$ 30,000 km

The DART system was designed to provide rapidly all information necessary to make the decisions pertinent to meeting the above requirements. Section 2.1 describes in detail the structure of the DART program.

2.1 Program Description

The DART system was designed to provide information sufficiently complete to allow a tentative GO/NOGO decision to be made for the AIMP lunar orbit mission within 10 minutes after receipt of tracking data describing the translunar trajectory. These data along with a numerical confirmation of the GO/NOGO decision were to be available to the mission director approximately 15 minutes after receipt of the tracking data. The output of the DART system is designed to provide maximum ease in identifying the firing time and attitude which will satisfy the various mission criteria, while providing complete data necessary to study the resulting orbits in detail. Data provided directly in terms of the specific mission requirements, including

a. orbit lifetime
b. pericynthion radius
c. shadow exposure,

are calculated and output by DART as functions of retro-rocket firing time and attitude. In addition, the following data are provided:

a. alternate mission data
b. initial spin axis-sun angle
c. occultation and umbral passage
d. maximum apocynthion radius during satellite life
e. minimum pericynthion radius during satellite life
f. apocynthion-moon-sun angle
g. miscellaneous orbit data

In the event the initial evaluation produces a NOGO decision, or if it is desirable to alter the lunar orbit to improve orbit stability or the scientific value of the orbit, an attitude reorientation of the spacecraft may be desired. DART is equipped to perform a rapid attitude scanning analysis to determine what attitude orientation (if one exists) would be required to produce a satisfactory lunar orbit. This analysis requires about 5 minutes.

To ascertain the probability of satisfying mission criteria associated with a decision (firing time and attitude), DART is equipped to perform a Monte Carlo analysis in real time. This analysis will permit an evaluation of the effects of uncertainties in the tracking data, spacecraft attitude angle, and retro-rocket performance on the GO/NOGO decision. The probability of satisfying the lifetime, pericynthion, and shadow requirements as a function of retro-rocket firing time is provided by this analysis. Shadow computations can be included in the Monte Carlo analysis, although the computation time will thereby be increased by a factor of about 10.

The DART system is coded in Fortran IV and has been implemented for use on the IBM 7094 computer. The computing times quoted in this paper are based on using a 7094 Model II computer.

2.2 Major Program Elements

Because of the nature of a real time computing system, it is desirable to be able to perform initial first order calculations as quickly as possible. These first order calculations can be refined later. To accomplish this, the DART system uses the following computational modes:

a. Alternate mission calculation mode: trial retro-rocket firings are simulated along the early portion of the lunar transfer trajectory, and resulting orbit parameters with respect to the earth are displayed.

b. Analytic GO/NOGO analysis mode: trial retro-rocket firings are made in the vicinity of the moon. The resulting lunar orbits are classified GO if they satisfy the decision criteria (i.e., mission requirements) or NOGO if they do not. Lunar orbit lifetime is calculated by the SLAP program (to be described later). A sample printout of this mode is shown in table 1.

c. Two numerical confirmation modes: since the SLAP method of determinining orbit lifetime is an approximate method, numerical confirmations are included to provide the DART system with an accurate method of determining lunar orbit lifetime.

d. Monte Carlo statistical analysis mode: errors associated with the tracking data, the spacecraft attitude, and the retro-rocket thrust are randomly varied. By performing an analytic GO/NOGO analysis on each random sample, the probability of satisfying the mission requirements can be determined for a range of firing times. An example of a printout of this phase is shown in table 2.

e. Attitude reorientation mode: information for determining the best spacecraft attitude based on mission requirements is generated. A sample printout of this phase is shown in tables 3 and 4.

In the analytic GO/NOGO analysis phase, the Monte Carlo phase, and the attitude reorientation phase, the quick first order computational methods are used to describe the evolution of the lunar orbit. The numerical confirmation phases use precise numerical integration of the lunar orbit.

The DART system is coded in a manner to allow a flexible control of the system at all times. Flow can be transferred from one computational phase to another by simply issuing a single command at the 7094 control panel, thereby allowing a rapid evaluation of the firing time and attitude for all possible situations. A considerable amount of effort was expended to make the program as easy to operate as possible. Since the pressures of an actual launch are very great, the possibility of an error is reduced if the complexity of the operation is kept at a minimum. This also allows more time for the operations engineers to analyze the data.

The system was coded using a block programming technique. Although this method required more programming effort initially, the effort was more than compensated for later when alterations and additions to the system were necessary. For example, the AIMP-E spacecraft had an attitude control system, whereas the AIMP-D spacecraft had no such system. Thus, after the AIMP-D launch, some type of attitude determination scheme had to be incorporated into the DART system. By using block programming, this scheme could be assembled independently and incorporated by adding only a few statements to the basic DART coding. The five basic blocks of the DART system are described below.

FLIP (Fortran Lunar Interplanetary Program):

FLIP is an N-body trajectory program based on Cowell's method. The program includes the sun, the earth, the moon, the effects of the earth's oblateness, and the triaxiality of the moon. The capability to include the other planets of the solar system exists within FLIP, but they are omitted for this application. The equations are integrated using a nine-cycle Runge-Kutta integration scheme.

FLIP is used to integrate the motion of the spacecraft from anchor vector epoch to insertion into the lunar orbit. The anchor vector is the instantaneous position and velocity at some epoch as determined by tracking. As the integration proceeds, the alternate mission calculations are first performed by simulated trial firings of the retro-rocket at specified times during the early portion of the trans-lunar trajectory. When the spacecraft approaches the sphere of influence of the moon, simulated retro-rocket firings are again made at specified times. These firings continue until the spacecraft passes the moon. Initial lunar orbit elements as a function of retro-rocket firing time result from these calculations. At each point the computations for the orbit lifetime and shadow life are performed using SLAP (discussed below).

FLIP can be used to integrate numerically an alternate mission trajectory. It can also be used for numerical confirmation of the approximate lunar orbit calculations; this confirmation, however, is relatively slow (requiring about 13 minutes for an orbit with a lifetime of 180 days). Another subprogram, GAUSS, was included in the DART system to perform a more rapid numerical confirmation of the orbit. GAUSS is discussed below.

SLAP (Satellite Lifetime Analysis Program):

SLAP is used to evaluate the evolution of the lunar orbital elements and thus the lunar orbit lifetime.[2] Also, the time to a shadow of a duration of specified length is determined. The inputs to SLAP are the initial orbit elements which result after each retro-rocket trial firing. A single lifetime prediction for a 180-day duration including shadow calculations and printing can be accomplished in approximately 15 seconds (average) using SLAP. If the shadow calculations are omitted, a single lifetime calculation requires only about 1 to 2 seconds. For a typical transfer trajectory with 100 trial firing times (e.g., trial firings at 65 to 90 hours after launch at 15 minute intervals), approximately 5 minutes will be required to complete the analysis including shadow calculations. Only 5 minutes is required because many of the 100 trial firings will result in very short lifetime.

When doing the probability analysis, SLAP can be operated in the fast (limit value) mode, provided shadow-life probabilities are not required. In this mode the actual lifetime is not computed; instead, the program simply determines whether or not the lifetime is greater than 180 days by calculation of a lifetime index. This index is defined by

$$LI = e_{crit} - e_{max}$$

where

LI is the lifetime index

e_{max} is the maximum eccentricity calculated by SLAP

and

e_{crit} is the eccentricity at which the spacecraft will impact the moon assuming that the orbit's semi-major axis is constant. This is a valid assumption for semi-major axes below 30,000 km.

Thus

$$e_{crit} = 1 - \frac{\text{radius of moon}}{\text{orbit's semi-major axis}}$$

The above parameters are illustrated in figure 2, which is a typical eccentricity history. It is evident that a negative lifetime index corresponds to an orbit that results in an impact. The lifetime index is a measure of the stability of the lunar orbit. The calculation of the lifetime index (LI) requires less than one-tenth of a second.

Monte Carlo Statistical Analysis Program:

During the Monte Carlo statistical analysis phase, FLIP is used to integrate the trans-lunar trajectory for the N samples of the randomly selected anchor vector, the spacecraft's attitude, and retro motor thrust. The random samples are based on the best estimate of the anchor vector covariance matrix, attitude errors, and retro-rocket impulse variations. In the vicinity of the moon, retro-rocket trial firings are simulated over a preselected range of times and the probability of satisfying mission criteria are determined at each firing time. The probability analysis mode was incorporated into DART because it was discovered, in the early mission planning stages, that even though the deterministic phases (analytic GO/NOGO and numerical confirmations) might indicate a GO condition at a specific firing time, the probability of achieving a satisfactory orbit could be very low.

A 50 sample size Monte Carlo run with 12 trial firing times requires only 8 minutes of computing time when using SLAP in the fast mode. If an estimate of the lifetime is required for those samples whose lifetimes are less than 180 days, SLAP can be run in the normal mode. In this case the same probability analysis will require about 30 minutes of computing time. If it is further desired to have shadow-life probabilities computed (the time before occurrence of impact/escape or shadow time exceeds a specified value), the shadow calculations must be added to SLAP. One to three hours of computing time are required to complete the probability analysis in this case.

GAUSS

GAUSS is a four-body numerical integration program based on Gauss's variation or parameters formulation. The four bodies are the earth, moon, spacecraft, and sun; the earth's oblateness and lunar triaxiality are neglected in this program. The position of the earth relative to the moon is defined by its mean elements. Because of the approximations inherent in GAUSS, it is not so precise as FLIP. However, for orbits in the region of interest, GAUSS has been verified to be sufficiently accurate to provide satisfactory results. Numerical confirmation runs made using the GAUSS program require approximately 2 minutes (average) to describe the dynamic behavior of the lunar orbit for 180 days.

Attitude Analysis Program:

The attitude analysis program is used to determine the best attitude of the spacecraft's spin axis in order to achieve the mission criteria. The analysis is performed by first integrating to the moon using FLIP. The hyperbolic orbital elements with respect to the moon at closest approach are used for the remainder of the analysis. The spacecraft's attitude is systematically varied in two orthogonal directions, with an approximate analytic GO/NOGO analysis performed for each attitude. One direction is in the plane formed by the spacecraft spin axis and a vector pointing from the spacecraft to the sun (in-plane); the other is perpendicular to that plane (out-of-plane). By making use of Keplerian two-body theory and SLAP in its fast mode for lifetime calculations, an approximate analytic GO/NOGO analysis can be performed in a very short time. For an average attitude analysis, a 15 by 15 array of attitude angles is analyzed; thus 225 approximate analytic GO/NOGO analyses are performed with 60 simulated trial retro-rocket firings made in each GO/NOGO analysis. In all, about 13,500 lifetime calculations are made for each attitude analysis.

A page of output is generated which contains the maximum lifetime index attainable for each combination of attitudes, thereby providing a rapid means of determining the region of attitudes which produce satisfactory lifetimes (table 3). Also provided are output pages presenting the approximate GO/NOGO analysis results for each spacecraft attitude (table 4). The computer time required is only 3 minutes.

2.3 Printout

The DART system is capable of printing in both on-line and off-line modes. The on-line printing is primarily used by the personnel operating the system. The immediate availability of information allows the operations engineers to prepare in advance for subsequent analyses and to participate actively in the decision process.

The off-line outputs are used by the mission director. These pages were designed in a compact, easy-to-read format for this purpose. Only those items of data which the mission director personally requested are included in this printout. Examples of selected off-line printout are presented in tables 1 through 4.

2.4 Computational Flow

The GO/NOGO evaluation performed by the DART system begins with receipt of anchor vector and spacecraft attitude data from the tracking and orbit determination phases of the Goddard AIMP Real Time Program. Figure 3 is an illustration of the data flow associated with DART. The solid lines indicate the primary or initial steps to be followed; the dashed lines indicate optional capabilities. The anchor vector, anchor vector epoch, and covariance matrix of the anchor vector errors are received on magnetic tape. The spacecraft attitude must be keypunched on an IBM data card. All data input to the system via cards are punched in a standardized format. Because of the nature of the inputs many of the input cards can be keypunched in advance. Inputs determining the control of the program are input via keys on the 7094 control panel. The first phase of computation is an output of all the important inputs and control flags on-line so that they can be checked for accuracy. If an error is detected, flow can be easily recycled back through the input routine. The program also has the capability of returning to the data input point from any portion of the program if it is necessary to terminate a run for any reason.

The alternate mission calculations are performed first. Then, the program begins the GO/NOGO computations using the approximate method (SLAP); these calculations will span the full range of firing times during which a lunar orbit can possibly be achieved, at 15 minute intervals. Complete orbit data including lifetime and shadow-life calculations are provided for each simulated trial firing. Thus, complete visibility of the possible firing time spectrum is provided. The data are displayed on-line and are printed off-line. Once the analytic GO/NOGO phase is completed, several options exist for further analyses (indicated by the dashed arrows in figure 3). These options and their associated computation times are the following:

a. The off-line output can be printed and sent to the mission director.

b. A new anchor vector can be input and the approximate analysis repeated (requires about 5 minutes).

c. A more detailed analysis, at shorter intervals, can be accomplished using the approximate analysis (requires less than 5 minutes).

d. Numerical confirmation runs can be made (requires about 3 minutes per case).

e. A Monte Carlo probability analysis can be performed (requires about 8 minutes per case for a 50 sample size with 12 trial firing times each,or one to three hours when shadow calculations are included).

f. An attitude reorientation analysis can be performed (requires about 3 minutes).

g. A numerical confirmation of an alternate mission can be made using FLIP (requires about 7 minutes).

h. A numerical confirmation run of a lunar orbit can be made using FLIP. However, this mode is necessary only for the final evaluation of lunar orbits when very precise simulations are desired (requires about 13 minutes per case).

3. LAUNCH SUPPORT OPERATIONS

The Delta vehicle, carrying the AIMP-E spacecraft, lifted from its launch pad at 10:19:02 EDT on Wednesday, July 19, 1967. Sixty-seven hours later, at 05:19 EDT on Saturday, July 22, the retro-rocket was fired and the spacecraft was successfully placed into orbit about the moon. In this section, the operation of the DART system at the NASA Goddard Space Flight Center during the translunar flight is described.

The likelihood that the primary mission was GO was established with the processing of the first anchor vector at approximately two and one-half hours after launch. An hour later data were available which could have been used to make a final decision, if required, as to what time to fire the retro-rocket on the basis of the measured attitude of the spacecraft. The resultant orbit would have satisfied the mission requirements as originally specified. Such a decision was not required at that time and thus was deferred pending further analysis. The analysis which followed showed that a class of orbits existed which would significantly enhance the scientific value of the mission. The various modes of the DART system were utilized to great advantage during the next two and one-half days to select, with a high degree of confidence, the attitude orientation and the retro-rocket firing time which would result in a better orbit. The probability analysis and attitude reorientation analysis modes were especially useful in making the selection.

A total of ?? runs were processed with the DART system during the period from one hour after launch to nine hours prior to lunar orbit insertion. The analysis was conducted essentially in three phases on July 19, 20, and 21, respectively. The three phases are indicated below:

1. Wednesday, July 19:

   a. primary mission GO/NOGO evaluation

   b. retro-rocket time-to-fire evaluation for measured attitude

   c. attitude reorientation analysis to explore possibility of improving the orbit

2. Thursday, July 20:

   selection of attitude orientation

3. Friday, July 21:

   selection of retro-rocket firing time

The operation of the DART system and the major decisions that were made are highlighted in the discussion which follows.

Wednesday, 19 July 1967:

Liftoff occurred at 10:19:02. The first anchor vector was received at 12:15 and within five minutes the approximate analysis, using the analytic GO/NOGO mode, was completed. The analysis indicated that a region of retro-rocket firing times existed between 65.0 and 70.5 hours after liftoff where orbits with greater than 180 day lifetimes could be achieved. The run indicated that all mission constraints could be met by firing at between 67 and 67.5 hours. The output from this run is presented in table 1. A GO decision for the primary mission appeared to be a certainty. Numerical runs made using the GAUSS mode at selected time points verified the results given by the analytic GO/NOGO mode; these runs were made assuming the spacecraft attitude to be nominal since telemetry data giving the measured attitude of the spacecraft would not be available for another hour.

Measured spacecraft attitude data were received at approximately 13:40 EDT. The data showed the spacecraft attitude to be very nearly nominal. Both the right ascension and the declination were within 0.6 degrees of nominal, and the standard deviation measurement errors were less than 0.1 degrees. A second, updated anchor vector (AV #2) was provided. Examination of the covariance matrix for AV #2 indicated that the uncertainties in the tracking data were quite small, which meant that the anchor vector had very nearly achieved stability. The standard deviation position and velocity uncertainties were 0.86 n.mi. and 0.46 ft/sec, respectively. A run was made using the analytic GO/NOGO mode; it showed little change from the first run. At this point, it appeared that the choice of retro-rocket firing time should be in the vicinity of 67.25 hours, yielding an initial orbit having nominally the following parameters:

$r_a = 7795$ km

$r_p = 2962$ km

$i = 169$ deg. (w.r.t. ecliptic)

$\tau = 9.83$ hours

These orbit parameters satisfied all of the originally specified mission constraints (refer to section 2 of this paper for the mission constraints). Thus, a highly successful lunar orbit mission was in the making which did not use a mid-course correction and which did not require orientation of the spacecraft. The only control required to achieve a lunar orbit which would satisfy the mission constraints was the retro-rocket firing time.

To provide the reader with an understanding of the events which took place during the next two and one-half days, three important considerations are now discussed. These considerations and the analyses and decisions which followed from them clearly demonstrate the important role that the DART system, with its flexibility and speed of operation, played in selecting the final orbit. First, an orbit with a period greater than 10 hours was desired because of possible degradation in the power supply from the solar cells as a result of temperature cycling fatigue. The low inclination of the achievable orbits meant that the satellite would pass through the moon's shadow once every orbit. In the event of such a problem, the number of thermal cycles and, hence, the useful life of the spacecraft would be directly proportional to the period of the orbit selected. Thus, a minimum orbit period of 10 hours was desired to guarantee a minimum useful life of the spacecraft.* Second, although original requirements called for the pericenter radius to be between 2250 and 3500 km, the AIMP-E project scientist indicated that the scientific value of the mission would be significantly enhanced if the pericenter radius were less than 2500 km. The lower pericenter was desired so that the nature of the bow of the solar plasma wave near the moon could be investigated. The data in table 1 show that the pericenter radius is about 3000 km and increasing for a period of 10 hours or greater. The third consideration concerns the reliability of the spacecraft's attitude control system. As will be shown, the desired conditions could be achieved only by reorienting the spacecraft prior to firing the retro-rocket. While the reliability of the attitude reorientation system was considered to be high and was available for use if required, some doubt existed as to the advisability of exercising the system since all of the required mission constraints could be satisfied without reorienting the spacecraft. To make this decision with confidence in real time, the mission director had to be provided with information that would adequately assess the risks involved. He could then weigh this information against the scientific advantages to be gained.

As a first step in eliminating or at least minimizing the possible thermal cycling fatigue problem, a decision was made to investigate the possibility of orienting the spacecraft prior to firing the retro-rocket such that the resultant orbit would have a high inclination. Increasing the orbit inclination would reduce the number of shadow passages. The attitude reorientation analysis mode of DART was used to obtain the required data. The simulated spacecraft attitude was displaced plus and minus 70 degrees, in 10 degree increments, both in the in-plane and in the out-of-plane directions. The array of lifetime indices given in table 3 defined the bounds of attitude orientation for which, deterministically, stable lunar orbits could be achieved. Examination of the trial firing data for each of the acceptable attitude orientations revealed that the orbit's inclination could not be increased sufficiently to eliminate shadows. Further examination of the data from the ±70 degree attitude reorientation run revealed that a small in-plane change of the spacecraft attitude would yield an orbit with a pericenter radius less than 2500 km and a period greater than 10 hours (e.g., see table 4). Table 4 presents the resulting orbits as a function of the retro-rocket firing time for the -10.0 point (circled) in the attitude matrix of table 3. A page similar to that shown in table 4 was provided for each attitude for which the lifetime index was positive. The firing times that produce stable lunar orbits are denoted whereever the lifetime index is positive. To define the relationship between the pericenter radius and the orbit period in the region of interest, a second attitude reorientation analysis run was made using a finer grid. The simulated spacecraft attitude was displaced plus and minus 10 degrees in 2 degree increments. The -6.0 point from that run is given in table 5. The relationship between the pericenter radius and the orbit period as a function of the change in spacecraft attitude (in-plane) and the retro-rocket firing time is presented in figure 4.

The data in figure 4 show that the desired conditions could be satisfied by appropriately selecting the spacecraft attitude and the firing time. Examination of these data suggests that an orbit with a period well in excess of 10 hours and a pericenter radius below 2500 km could have been achieved by reorienting the spacecraft about 10 degrees. Having established the possibility of obtaining an orbit that would meet the desired conditions, the tasks of choosing the exact spacecraft orientation and retro-rocket firing time and evaluating the risks associated with these choices remained. Several factors had to be taken into account in making the final decision. For example:

a. How important was it that the orbit period be above 10 hours since the desired pericenter radius could be achieved without reorienting the spacecraft if an orbit period of about 6.5 hours could be tolerated?

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*As of the writing of this paper, the Explorer XXXV satellite has been orbiting the moon for about five months, and no appreciable degradation of the power supply has occurred.

b. How might the attitude reorientation system fail, and what would the effect be if it did fail?

c. To what extent should the system error souces be increased when performing the probability analysis? Since a highly successful mission was in the making without reorientation, it was felt that the analyses supporting a decision to reorient should be conservative so as to maintain high confidence that the mission would not be jeopardized.

A discussion of these and other factors is given, where appropriate, in the material which follows.

The details of the possible solar power degradation are not pertinent to the intent of this paper. Let it suffice to say that it was deemed highly desirable to have an orbit period of greater than 10 hours.

Close examination of the attitude reorientation system revealed that the risk involved in exercising the system was very small. Only a random ordered failure of the attitude reorientation system could have resulted in a fatal attitude error. However, such a circumstantial failure was considered to be extremely remote.

It has been stated previously that an attitude change on the order of 10 degrees appeared to be a good choice. The discussion that follows shows that this would have been a bad choice. Deterministically, this choice would have been a good one. However, when consideration was given to the fact that the state conditions (e.g., anchor vector, spacecraft attitude, and retro-rocket impulse) were not known precisely, it became evident that the probability of success fell below an acceptable value. Results from practice simulations showed that a strong correlation existed between the probability of achieving a stable lunar orbit and the width of the acceptable retro-rocket firing time window, as evaluated using deterministic data. To understand this correlation, assume for the moment that conditions are such that a small window exists. Runs made during the practice simulations revealed that the beginning and ending of the window shifted in time as the simulated attitude of the spacecraft was changed. For a small window, the possibility existed that the beginning and ending of the window for the perturbed attitude would not overlap that determined for the measured attitude, as illustrated in figure 5. Thus, a retro-rocket firing time selected on the basis of the measured attitude might not fall within the acceptable window for the perturbed, unknown attitude. The probability of this situation occurring decreases as the width of the window increases, provided, of course, that the firing time is not chosen too close to the window's edge.

Examination of the lifetime indices (tables 1, 4 and 5) revealed that the width of the window of acceptable retro-rocket firing times was beginning to decrease rapidly as the attitude reorientation angle approached -10 degrees. Thus, it was concluded that the angle change should be less than 10 degrees in order to maintain a high probability of achieving a stable lunar orbit. The data in figure 4 and tables 4 and 5 show that an attitude change of from 4 to 8 degrees would yield the desired orbit.

Based on these findings, it was concluded that the risk associated with reorienting the spacecraft was quite low. Since the scientific gains made possible through this maneuver were significant, the decision was made to reorient.

Thursday, 20 July 1967:

Computations with the DART system got underway on Thursday at about 11:15 EDT. In order to achieve the lowest possible pericenter radius consistent with an orbit period greater than 10 hours and a high probability of mission success, the Probability Analysis mode of the DART system was run at several attitude orientations in the region of interest. To insure that the mission would not be jeopardized, the one-sigma value of the dominant error source (spacecraft attitude) was varied. This provided a measure of the sensitivity of mission success to this error source and permitted the best attitude orientation to be chosen with a high degree of confidence. An updated anchor vector (AV #3) was used for this phase of the analysis even though changes from the previous vector were minor. A decision was made not to update the anchor vector further in this phase of the analysis so that the data to be compared would be consistent. The description of the runs made during this phase is not presented in the real time sequence; rather, it is presented in a manner which allows the results to be summarized most conveniently.

As a first step, the analytic GO/NOGO mode was run for IN PLANE attitude reorientation angle changes of 0, -4, -6, and -8 degrees to update and more accurately define the orbit data. Recall (cf. section 2.2) that the orbit calculations performed by the attitude reorientation mode make use of approximate analytical expressions for the approach trajectory and the addition of the retro-rocket impulse. Numerical confirmation runs were made using the Gauss mode at selected firing times to confirm the validity of the approximate analysis in the region of interest. A series of runs made using the probability analysis mode resulted in the conclusion that the amount by which the spacecraft should be reoriented was between 5 and 7 degrees. Fifty samples were used for most of the runs, requiring about 8 minutes per case. Table 6 presents the probability of having an orbit with a lifetime of greater than 180 days, as extracted from the probability runs, for a -6 degree attitude reorientation. Standard deviation errors of 0.5, 1.0, and 2.0 degrees were assumed in making these runs. The accepted value was 0.5 degrees. However, as stated previously, larger values were also run to introduce conservatism. Similar probability data were obtained for the 0, -4, and -8 degree reorientation angles. The data showed that the probability of achieving a stable lunar orbit was beginning to fall off rapidly as the reorientation angle approached -8 degrees. Following the examination of the above data, a reorientation angle of -6.7 degrees was selected. An angle change of 6.7 degrees was chosen since the attitude of the spacecraft could be reoriented in steps of no larger than 1.675 degrees per command. To simplify the reorientation maneuver, the use of fractional commands was avoided. Four commands yielded a total angle change of 6.7 degrees. Two runs were made using the probability analysis mode for this orientation: one had a standard deviation attitude error of 0.5 degrees, and the other an error of 1.0 degrees. The results of the probability analysis verified this angle to be a good selection. A reorientation angle of 6.7 degrees, as shown in figure 4, would result in an orbit with a pericenter radius below 2500 kilometers and a period as high as approximately 15 hours. Thus, the analysis for choosing the angle by which the spacecraft was to be reoriented was completed.

Friday, 21 July 1967:

Reorientation of the spacecraft attitude was successfully accomplished during the night. The actual IN PLANE change of the spacecraft attitude was determined to be 5.6 degrees. During the process of reorientating the spacecraft, each command was assessed to change the angle by about 1.86 degrees. This amount was greater than that previously expected. Thus, a decision was made to use only three commands, resulting in a total angle change of 5.6 degrees. The use of a fourth command would have caused the angle change to be in excess of 7 degrees.

The task which remained was to establish the exact time for firing the retro-rocket based on the reoriented attitude of the spacecraft. Again, it was desired to select an orbit with the lowest possible pericenter radius, consistent with an orbit period in excess of 10 hours and a high probability of mission success. An updated anchor vector (AV #4) was used for the analysis on the final day. The first run of the day was made using the analytic GO/NOGO mode for firing times between 66 and 68 hours in one-minute steps to define in detail the type of orbit which could be achieved as a function of retro-rocket firing time. This run was followed by a series of numerical confirmation runs to verify the adequacy of the analytic GO/NOGO mode in the region of interest.

The desired retro-rocket firing time was assessed to be about 67 hours after liftoff. The next run processed was a 200 sample size Monte Carlo probability analysis for trial firing times between 66.4 and 67.3 hours in steps of 0.05 hours. A standard deviation attitude error of 1.5 degrees was assumed. An attitude error of this magnitude was believed to account conservatively for measurement uncertainties and errors due to thrust misalignment or mass unbalance. Before the run was made, it was decided to change the upper and lower limits on the pericenter constraint to 1838 kilometers and 2400 kilometers, respectively. The change in pericenter radius limits was made so that the probability of satisfying the desired condition could be assessed; little value would have been derived from using the original constraints (2250 kilometers to 3500 kilometers) since a decision had already been made to emphasize the region below 2500 kilometers. The ability to change these constraints was an online capability of the DART system. The output from this run is given in table 2. The probability of achieving a stable lunar orbit is seen to be 100 percent, based on a sample size of 200, for trial firing times through 67.05 hours, with only a slight drop in probability through 67.30 hours. Similar probability runs were made for standard deviation attitude errors of 0.5 and 1.0 degrees; a sample size of 100 was used in making these runs. These runs show, as expected, that the probability of achieving a stable lunar orbit was 100 percent throughout the range of firing times investigated. Data from the first probability analysis run were presented as a scatter diagram on a plot of eccentricity versus semimajor axis (or period), as shown in figure 6. Lines of constant pericenter radius beginning with the lunar surface are shown in the figure. This presentation permitted a clear understanding of the possible behavior of the resultant orbits; the density distribution and the range of the parameters of prime interest (viz., pericenter radius and orbit period) are illustrated.

The effect of a larger-than expected impulse error of the retro-rocket was investigated. To accomplish this investigation, a run was made in which the retro-rocket impulse standard deviation was increased 50 percent. The standard deviation attitude error used in this run was 1.0 degree. The effect of increasing the impulse error of the retro-rocket was shown to have little effect on the probability of success.

To further evaluate the possible effects of errors on the resultant orbit, a run was made using the attitude reorientation analysis mode of the DART system. The attitude was displaced plus and minus 7 degrees in one degree steps from its current orientation. The purpose of this run was to define deterministically the amount by which the attitude could be in error and still achieve stable lunar orbit. The allowable IN PLANE attitude error as defined by the data from this run was seen to be about 3.5 degrees for a retro-rocket time of 67 hours, as shown in figure 7.

Based on the results of the attitude reorientation and probability analysis modes, a retro-rocket firing time of 67 hours was selected. The resultant orbit for this firing time had a pericenter radius of 2353 kilometers and an orbit period of 11.09. A final run was made using the FLIP numerical integration mode to obtain a detailed history of the orbit which results from a retro-rocket firing time of 67 hours.

The above discussion has shown how the probability analysis and the attitude reorientation modes of the DART system were used to select the attitude orientation and the retro-rocket firing time that produced an orbit which significantly enhanced the scientific objectives of the Explorer XXXV mission. The use of the DART system, with its flexibility and speed of operation, permitted a detailed analysis to be completed in a relatively short period of time, and the analysis provided a high degree of confidence that the proper decision had been made.

The retro-rocket was fired at 67 hours after liftoff, and the orbit that resulted was assessed to have the following parameters:

a. Orbit period = 11.4 hours

b. Pericenter radius = 2531 kilometers

c. Apocenter radius = 9362 kilometers

d. Orbit inclination = 169.47 degrees (w.r.t. ecliptic)

4. CONCLUSIONS

This paper has presented a description of a real time system for the determination of lunar orbit injection conditions. The development of the DART system has been traced from its inception as several separate mission analysis techniques to its culmination as a highly flexible on-line computer program capable of performing many analyses quickly and in rapid succession. The Monte Carlo error analysis and the orbit lifetime determination techniques used for the AIMP-D and E mission analyses became subroutines in the DART program, addressable in many cases by a single FORTRAN statement. This block-type design of the basic program elements made it possible to combine those elements in many different ways with a minimum of coding difficulties. A description of these elements has been presented along with an outline of their basic roles in the real time selection of the firing time and orientation necessary to inject a spacecraft into a suitable lunar orbit.

An account of the operation of the DART system during the launch and translunar flight of the AIMP-E spacecraft has been included in the paper. Emphasis has been placed on the kinds of decisions which were necessary and how these decisions were made in real time. A description of the analyses which were required has been included as well as an account of how the results of the analyses were presented to the mission director. The need for a high degree of flexibility in the DART system and how that flexibility was provided through a very close man-machine relationship has been stressed.

The development of an accurate approximate lifetime prediction method made it possible to perform Monte Carlo analyses in real time so as to determine the probability of mission success versus retro-rocket ignition time and orientation. The availability of this real time Monte Carlo analysis made it possible for the mission director to make a decision, with confidence, which resulted in a more scientifically valuable mission. The benefit to the AIMP-E mission of making the decisions on a probabilistic rather than a deterministic basis has been emphasized throughout the paper.

REFERENCES

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2. Uphoff, C. W. "An Approximate Method for Predicting Lunar Satellite Lifetimes and its Application to a Lunar Orbit Mission Analysis," presented to AAS Space Flight Mechanics Conference, Preprint 66-130, July 1966.

FIRINGTIMES MEASURED IN HRS PAST LIFT-OFF

ANCHOR VECTOR AT 15. 30. 33.
RT.ACC.= 235.430 DEC.= -24.000

RETRO ROCKET FIRING TIME DECISION CRITERIA GO = + NOGO = -
(IN ORDER OF IMPORTANCE)
1. DYNAMIC LIFETIME ** GO ONLY IF (LIFETIME IS .GE. 180. DAYS)
2. PERICENTER RADIUS ** GO ONLY IF (RP .GT. 2250. KM. AND RP .LT. 3500. KM.)
3. USEFUL LIFETIME ** GO ONLY IF (THE LIFETIME UNTIL 2.5HR. SHADOW TIME/ORBIT IS .GT. 180. DAYS)
4. AVERAGE SHADOW TIME/ORBIT PERIOD ** GO ONLY IF(TOTAL SHADOW TIME .LT. 0.15*LIFETIME)
NOTE-- AVERAGE ST/OP EXPRESSED AS PER CENT OF ORBIT PERIOD

FIRING TIME	RA (KM)	RP (KM)	AMS ANGLE	INC (DEG)	PERIOD (HR)	ECC	AVE ST/OP	MIN RP	MAX SHAD	USEFUL LIFETIME 2HR	2.5HR	3HR	LIFE TIME	L.T. INDEX	DECISION 1 2 3 4
65.000	29062.	9492.	345.	172.3	66.71	0.508	2.6	5899.	3.95	96.	102.	108.	180.	0.22	+ - - +
65.250	25535.	8581.	351.	172.1	55.53	0.497	2.6	5962.	3.23	78.	90.	99.	180.	0.25	+ - - +
65.500	22414.	7699.	357.	171.9	46.05	0.489	1.9	5820.	2.21	87.	180.	180.	180.	0.27	+ - + +
65.750	19642.	6853.	4.	171.6	38.01	0.483	1.8	5531.	1.64	180.	180.	180.	180.	0.29	+ - + +
66.000	17146.	6051.	11.	171.4	31.14	0.478	2.0	5146.	1.41	180.	180.	180.	180.	0.29	+ - + +
66.250	14890.	5299.	18.	171.0	25.28	0.475	2.3	4698.	1.30	180.	180.	180.	180.	0.29	+ - + +
66.500	12850.	4605.	26.	170.7	20.32	0.472	2.9	4219.	1.25	180.	180.	180.	180.	0.28	+ - + +
66.750	10994.	3977.	35.	170.2	16.14	0.469	4.0	3739.	1.22	180.	180.	180.	180.	0.27	+ - + +
67.000	9307.	3425.	45.	169.7	12.66	0.462	6.2	3284.	1.19	180.	180.	180.	180.	0.24	+ + + +
67.250	7795.	2962.	57.	169.1	9.83	0.449	9.7	2884.	1.16	180.	180.	180.	180.	0.21	+ + + +
67.500	6479.	2610.	72.	168.4	7.64	0.426	12.8	2571.	1.14	180.	180.	180.	180.	0.18	+ + + +
67.750	5417.	2398.	92.	167.6	6.09	0.386	15.8	2382.	1.10	180.	180.	180.	180.	0.16	+ + + -
68.000	4709.	2373.	119.	166.8	5.25	0.330	17.5	2362.	1.08	180.	180.	180.	180.	0.18	+ + + -
68.137	START OF OCCULTATION BY MOON														
68.225	VEHICLE ENTERS UMBRA OF MOON														
68.250	4481.	2582.	154.	166.1	5.23	0.269	16.3	2531.	1.09	180.	180.	180.	180.	0.22	+ + + -
68.500	4820.	3017.	195.	165.6	6.11	0.230	13.1	2906.	1.10	180.	180.	180.	180.	0.30	+ + + +
68.618	END OF OCCULTATION BY MOON														
68.750	5744.	3623.	234.	165.4	7.99	0.226	9.8	3507.	1.06	180.	180.	180.	180.	0.38	+ - + +
68.773	VEHICLE LEAVES UMBRA OF MOON														
69.000	7225.	4349.	267.	165.3	10.97	0.248	7.1	4275.	1.06	180.	180.	180.	180.	0.44	+ - + +

Table 1. Douglas AIMP-E Real Time Analysis: Quick-Look Firing Time Analysis

COVARIANCE MATRIX

```
0.17125933E-01-0.22256366E-02 0.29877344E-01 0.56256603E-07-0.37314937E-07 0.19508465E-06
-0.22256366E-02 0.24058929E-01-0.47020456E-01-0.51115026E-08 0.15288173E-06-0.29467093E-06
0.29877344E-01-0.47020456E-01 0.13056542E 00 0.93771906E-07-0.33259362E-06 0.82822786E-06
0.56256603E-07-0.51115026E-08 0.93771906E-07 0.18831943E-12-0.10937842E-12 0.61531038E-12
-0.37314937E-07 0.15288173E-06-0.33259362E-06-0.10937842E-12 0.11939969E-11-0.24259087E-11
0.19508465E-06-0.29467093E-06 0.82822786E-06 0.61531038E-12-0.24259087E-11 0.58313946E-11
```

SIGMA THETA= 0.15000000E 00 SIGMA PSI= 0.70000000E-01

PARTIAL RT. ACC. W.R.T. THETA= 0.120000E 00
PARTIAL DEC. W.R.T. THETA= 0.780000E 00

PARTIAL RT. ACC. W.R.T. PSI= 0.109000E 01
PARTIAL DEC. W.R.T. PSI= -0.130000E 00

REORIENTATION ERROR MODEL
0.00000000E-38 0.00000000E-38 0.00000000E-38 0.00000000E-38

FOURTH STAGE ERRORS

SIGMA RT. ACC.	SIGMA DEC.	SIGMA THRUST
1.50000000	1.50000000	5.13999999

PROBABILITY OF SATISFYING CRITERIA
VS.
FOURTH STAGE FIRING TIME

TIME	LIFE TIME	PERICENTER	SHADOW LIFE TIME	AVERAGE ST/OP
66.400	100.00	2.00	********	********
66.450	100.00	3.00	********	********
66.500	100.00	4.00	********	********
66.550	100.00	5.50	********	********
66.600	100.00	5.50	********	********
66.650	100.00	11.50	********	********
66.700	100.00	13.50	********	********
66.750	100.00	17.00	********	********
66.800	100.00	20.50	********	********
66.850	100.00	29.50	********	********
66.900	100.00	39.50	********	********
66.950	100.00	45.50	********	********
67.000	100.00	60.00	********	********
67.050	100.00	70.00	********	********
67.100	99.50	76.00	********	********
67.150	99.00	79.50	********	********
67.200	99.00	83.00	********	********
67.250	99.00	87.00	********	********
67.300	98.50	88.50	********	********

Table 2. Douglas AIMP-E Real Time Analysis: Probability Analysis

CURRENT ATTITUDE

R.A. = 236.00 DEC. = -23.40
ANCHOR VECTOR AT 16. 0. 55.
DELTA V = -1866.000 N.M./HR.

OUT OF PLANE ANGLE	IN PLANE ANGLE														
	-70.0	-60.0	-50.0	-40.0	-30.0	-20.0	-10.0	0.0	10.0	20.0	30.0	40.0	50.0	60.0	70.0
70.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00
60.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00
50.00	-9.00	-9.00	-9.00	-9.00	-9.00	-0.14	-0.02	-0.04	0.05	0.04	-0.02	-1.03	-9.00	-9.00	-9.00
40.00	-9.00	-9.00	-9.00	-0.18	0.08	0.21	0.30	0.33	0.31	0.25	0.17	0.09	0.00	-9.00	-9.00
30.00	-9.00	-9.00	-0.38	0.04	0.19	0.35	0.49	0.58	0.54	0.39	0.26	0.15	0.05	-0.06	-9.00
20.00	-9.00	-9.00	-0.05	0.08	0.24	0.41	0.60	0.45	0.53	0.52	0.34	0.19	0.08	-0.02	-9.00
10.00	-9.00	-9.00	-0.03	0.08	0.24	0.43	0.63	0.46	0.47	0.60	0.39	0.22	0.09	-0.02	-9.00
-0.00	-9.00	-9.00	-0.03	0.07	0.22	0.40	0.62	0.48	0.46	0.62	0.41	0.23	0.09	-0.02	-9.00
-10.00	-9.00	-9.00	-0.04	0.05	0.17	0.33	0.53	0.51	0.46	0.58	0.40	0.22	0.09	-0.04	-9.00
-20.00	-9.00	-9.00	-0.08	0.01	0.11	0.23	0.37	0.54	0.59	0.47	0.33	0.19	0.06	-1.92	-9.00
-30.00	-9.00	-9.00	-9.00	-0.05	0.04	0.12	0.19	0.25	0.27	0.26	0.20	0.11	-0.32	-9.00	-9.00
-40.00	-9.00	-9.00	-9.00	-9.00	-0.02	-0.04	0.01	0.03	0.03	0.01	-0.09	-9.00	-9.00	-9.00	-9.00
-50.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00
-60.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00
-70.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00	-9.00

Table 3. Douglas AIMP-E Real Time Analysis: Attitude Reorientation Analysis - Lifetime Index Grid - Current Attitude

IN PLANE ANGLE	OUT OF PLANE ANGLE	FIRING TIME	RA (KM)	RP (KM)	AMS ANGLE	INC (DEG)	PERIOD (HR)	ECC	LIFE INDEX
-10.00	-0.00								
		60.963	52610.	10610.	4.	174.8	140.09	0.664	-0.167
		63.010	35354.	6479.	17.	174.1	75.43	0.690	0.046
		64.210	26741.	4550.	25.	173.5	48.78	0.709	0.082
		64.995	21527.	3499.	32.	173.0	34.89	0.720	0.081
		65.546	18026.	2967.	37.	172.5	26.62	0.726	0.068
		65.954	15516.	2461.	41.	172.0	21.24	0.726	0.052
		66.269	13630.	2187.	46.	171.5	17.53	0.723	0.036
		66.519	12155.	1997.	50.	171.1	14.85	0.718	0.020
		66.723	10997.	1864.	54.	170.8	12.85	0.710	0.007
		66.893	10046.	1771.	57.	170.4	11.32	0.700	-0.005
		67.038	9261.	1707.	61.	170.1	10.12	0.689	-0.015
		67.162	8603.	1665.	64.	169.8	9.17	0.676	-0.022
		67.272	8045.	1642.	68.	169.6	8.40	0.661	-0.027
		67.368	7569.	1632.	71.	169.4	7.78	0.645	-0.030
		67.455	7160.	1635.	74.	169.1	7.27	0.628	-0.030
		67.534	6806.	1649.	77.	169.0	6.85	0.610	-0.027
		67.606	6499.	1673.	81.	168.8	6.51	0.590	-0.021
		67.673	6231.	1707.	84.	168.6	6.23	0.570	-0.013
		67.735	5998.	1749.	87.	168.5	6.01	0.548	-0.002
		67.793	5794.	1801.	90.	168.3	5.83	0.526	0.012
		67.848	5617.	1861.	93.	168.2	5.70	0.502	0.029
		67.901	5464.	1932.	96.	168.1	5.60	0.478	0.048
		67.952	5331.	2012.	99.	168.0	5.55	0.452	0.071
		68.001	5219.	2104.	102.	167.9	5.52	0.425	0.096
		68.049	5123.	2207.	105.	167.8	5.53	0.398	0.124
		68.096	5045.	2324.	108.	167.7	5.58	0.369	0.156
		68.142	4983.	2456.	110.	167.6	5.65	0.340	0.190
		68.188	4936.	2605.	113.	167.6	5.77	0.309	0.227
		68.234	4904.	2773.	115.	167.5	5.93	0.277	0.267
		68.280	4887.	2964.	117.	167.5	6.13	0.245	0.309
		68.327	4885.	3181.	119.	167.4	6.38	0.211	0.355
		68.375	4899.	3429.	120.	167.4	6.70	0.177	0.403
		68.424	4930.	3712.	120.	167.3	7.08	0.141	0.455
		68.474	4983.	4036.	117.	167.3	7.55	0.105	0.508
		68.527	5066.	4402.	108.	167.3	8.12	0.070	0.561
		68.582	5223.	4783.	77.	167.2	8.82	0.044	0.606
		68.641	5573.	5059.	26.	167.2	9.69	0.050	0.617
		68.703	6202.	5231.	4.	167.2	10.77	0.085	0.600
		68.769	6994.	5393.	357.	167.2	12.15	0.129	0.574
		68.841	7996.	5573.	355.	167.2	13.93	0.179	0.542
		68.920	9276.	5781.	356.	167.2	16.28	0.232	0.504
		69.007	10952.	6025.	358.	167.2	19.49	0.290	0.460
		69.104	13216.	6314.	360.	167.2	24.05	0.353	0.407
		69.213	16409.	6656.	3.	167.2	30.87	0.423	0.342
		69.338	21190.	7064.	6.	167.2	41.85	0.500	0.258
		69.482	29001.	7556.	9.	167.3	61.60	0.587	0.144
		69.653	43719.	8153.	13.	167.3	104.12	0.686	-0.040
		69.857	80379.	8888.	16.	167.3	245.04	0.801	-0.797
-10.00	-0.00								

Table 4. Douglas AIMP-E Real Time Analysis: Attitude Reorientation Analysis - Approximate Quick Look Firing Time Analysis

IN PLANE ANGLE	OUT OF PLANE ANGLE	FIRING TIME	RA (KM)	RP (KM)	AMS ANGLE	INC (DEG)	PERIOD (HR)	ECC	LIFE INDEX
-6.00	-0.00								
		60.963	70375.	17044.	337.	174.7	297.78	0.610	-0.771
		63.010	40835.	10448.	357.	174.1	102.35	0.593	0.022
		64.210	29064.	7197.	10.	173.5	60.85	0.603	0.142
		64.995	22666.	5396.	20.	173.0	41.43	0.615	0.167
		65.546	18616.	4303.	27.	172.5	30.58	0.625	0.164
		65.954	15811.	3594.	33.	172.0	23.82	0.630	0.150
		66.269	13753.	3111.	39.	171.6	19.30	0.631	0.133
		66.519	12180.	2769.	44.	171.2	16.11	0.630	0.115
		66.723	10941.	2522.	48.	170.8	13.77	0.625	0.099
		66.893	9942.	2340.	53.	170.4	12.00	0.619	0.084
		67.038	9122.	2205.	57.	170.1	10.62	0.611	0.071
		67.162	8439.	2105.	61.	169.8	9.54	0.601	0.060
		67.272	7853.	2032.	65.	169.5	8.67	0.589	0.051
		67.368	7373.	1981.	69.	169.3	7.97	0.576	0.044
		67.455	6953.	1948.	73.	169.0	7.40	0.562	0.040
		67.534	6591.	1929.	77.	168.8	6.93	0.547	0.039
		67.606	6277.	1924.	80.	168.6	6.54	0.531	0.040
		67.673	6004.	1929.	84.	168.4	6.23	0.514	0.044
		67.735	5767.	1946.	88.	168.3	5.97	0.495	0.050
		67.793	5560.	1972.	92.	168.1	5.76	0.476	0.058
		67.848	5380.	2009.	96.	167.9	5.60	0.456	0.070
		67.901	5224.	2054.	99.	167.8	5.47	0.436	0.084
		67.952	5089.	2110.	103.	167.7	5.38	0.414	0.100
		68.001	4975.	2175.	107.	167.6	5.33	0.392	0.119
		68.049	4878.	2251.	111.	167.4	5.31	0.368	0.141
		68.096	4798.	2339.	116.	167.3	5.31	0.345	0.165
		68.142	4734.	2439.	120.	167.2	5.35	0.320	0.192
		68.188	4686.	2553.	124.	167.2	5.43	0.295	0.221
		68.234	4653.	2683.	129.	167.1	5.54	0.269	0.253
		68.280	4635.	2829.	134.	167.0	5.68	0.242	0.287
		68.327	4633.	2996.	140.	166.9	5.87	0.215	0.323
		68.375	4646.	3155.	146.	166.9	6.11	0.187	0.362
		68.424	4678.	3399.	153.	166.8	6.40	0.158	0.403
		68.474	4729.	3643.	161.	166.8	6.75	0.130	0.445
		68.527	4807.	3918.	173.	166.7	7.18	0.102	0.489
		68.582	4924.	4221.	191.	166.7	7.71	0.077	0.533
		68.641	5114.	4531.	220.	166.7	8.35	0.060	0.571
		68.703	5443.	4800.	259.	166.6	9.14	0.063	0.594
		68.769	5945.	5018.	289.	166.6	10.12	0.085	0.596
		68.841	6613.	5223.	307.	166.6	11.35	0.117	0.585
		68.920	7466.	5443.	319.	166.6	12.93	0.157	0.565
		69.007	8556.	5694.	329.	166.6	14.99	0.201	0.539
		69.104	9971.	5986.	336.	166.6	17.76	0.250	0.504
		69.213	11840.	6330.	343.	166.6	21.62	0.304	0.461
		69.338	14478.	6739.	349.	166.6	27.24	0.365	0.406
		69.482	18297.	7232.	355.	166.6	35.95	0.433	0.333
		69.653	24297.	7831.	1.	166.7	50.79	0.512	0.233
		69.857	34793.	8569.	6.	166.7	79.37	0.605	0.082
		70.107	57223.	9492.	12.	166.7	151.87	0.715	-0.232
-6.00	-0.00								

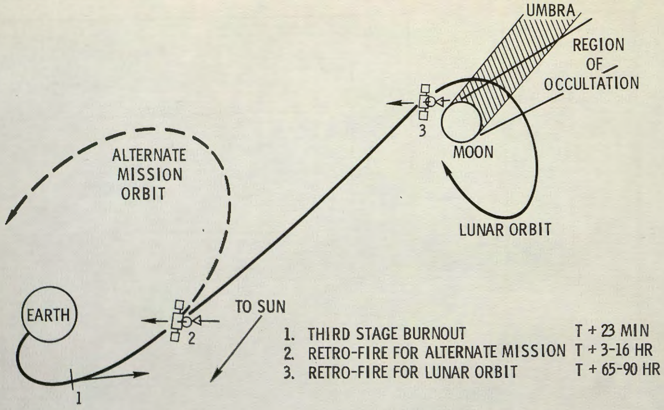
Table 5. Douglas AIMP-E Real Time Analysis: Attitude Reorientation Analysis - Approximate Quick Look Firing Time Analysis

SAMPLE SIZE = 50

Retro-Fire Time (Hrs)	Probability (Lifetime 180 Days)		
	$\sigma\theta = 0.5^{o}$	$\sigma\theta = 1.0^{o}$	$\sigma\theta = 2.0^{o}$
66.0	100	100	100
66.2	100	100	100
66.4	100	100	100
66.6	100	100	98
66.8	100	100	98
67.0	100	100	94
67.2	100	100	90
67.4	100	98	90
67.6	100	100	90
67.8	100	100	100
68.0	100	100	100

Table 6. Probability Analysis: Variation With Standard Deviation Attitude Error

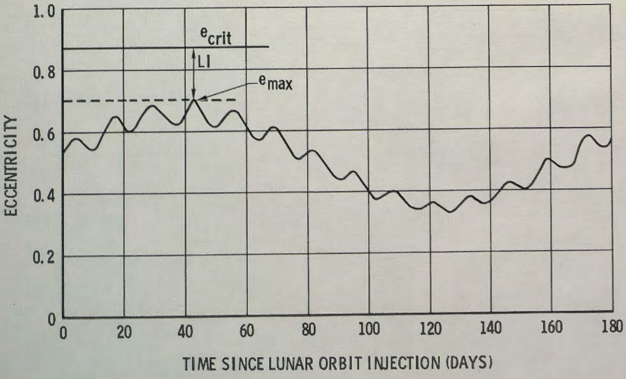
MISSION PROFILE



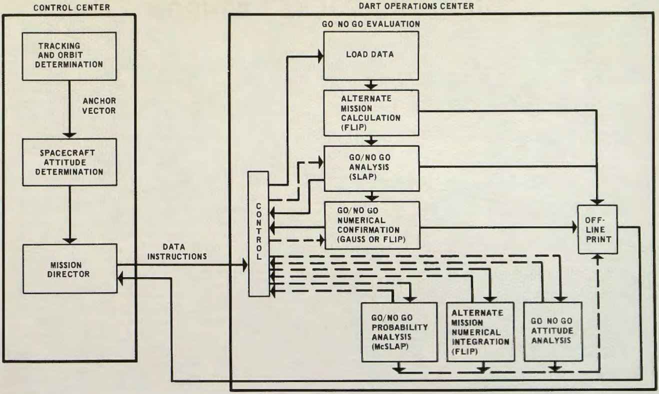
ECCENTRICITY HISTORY

SEMI-MAJOR AXIS = 13,620 KM

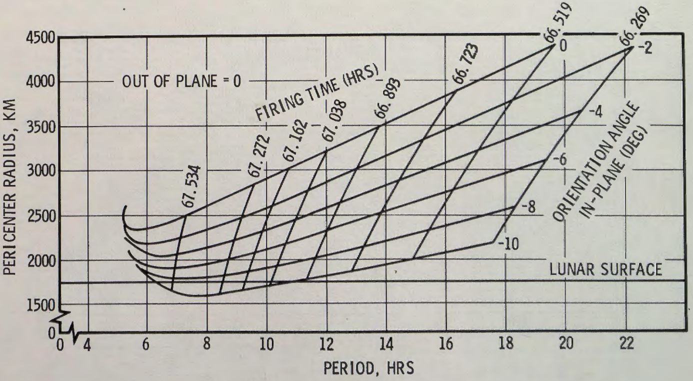
e_{crit} = 0.872



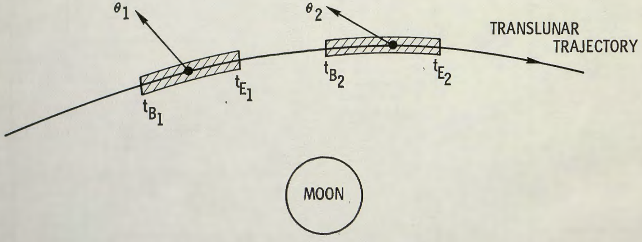
COMPUTATIONAL FLOW OF DART SYSTEM



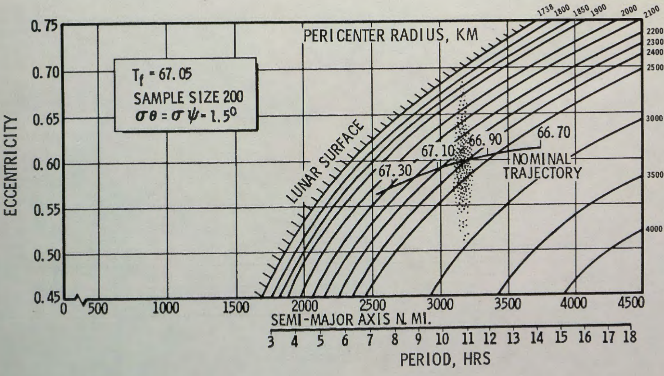
EFFECT OF ORIENTATION ON ORBIT PARAMETERS



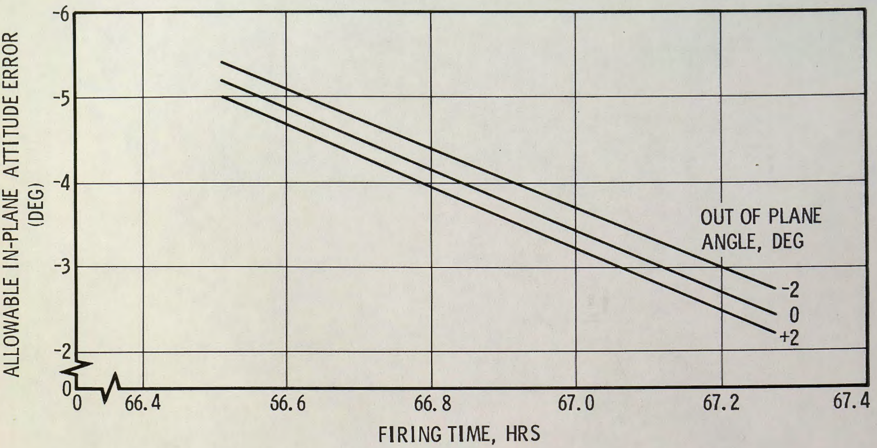
EFFECT OF ATTITUDE ON RETRO-ROCKET WINDOW



PERICENTER DISTRIBUTION DENSITY



ALLOWABLE IN-PLANE ATTITUDE ERROR BASED ON LIFETIME INDEX



11.3-17