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505 KING AVENUE COLUMBUS, OHIO 43201

## THIRD INTERIM SCIENTIFIC REPORT

on

DEVELOPMENT OF AN EVALUATION TECHNIQUE FOR INTERPLANETARY MISSION ASTRIONICS

Submitted to

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION Electronics Research Center Cambridge, Massachusetts

February 28, 1970

Contract No. NAS 12-550

BATTELLE MEMORIAL INSTITUTE Columbus Laboratories 505 King Avenue Columbus, Ohio 43201

## DEVELOPMENT OF AN EVALUATION TECHNIQUE FOR INTERPLANETARY MISSION ASTRIONICS

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#### February, 1970

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## Prepared under Contract No. NAS 12-550 by BATTELLE MEMORIAL INSTITUTE Columbus Laboratories 505 King Avenue Columbus, Ohio 43201

#### Electronics Research Center

## NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

#### FOREWORD

This interim scientific report presents the results of a twelve-month study conducted by Battelle Memorial Institute, Columbus Laboratories, for the NASA Electronics Research Center in partial fulfillment of the work requirements of Contract NAS 12-550.

The objective of this study was to extend the evaluation techniques developed for astrionics systems which employ aided inertial guidance systems operating on interplanetary flyby missions to include the astrionics required for orbiter, lander, and multiple planet swingby missions.

This volume presents a summary of the study results, detailed technical discussion, recommendations, and conclusions.

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# LIST OF MAJOR SYMBOLS AND DEFINITIONS

Exponent used in Weibull distribution

Control Moment Gyroscope

α

CMG

$^{\mathrm{D}}\mathrm{T}$	Degrees-of-freedom of the target miss covariance
D <sub>V</sub>	Degrees-of-freedom of the midcourse $\Delta V$ covariance
DSIF	Deep Space Instrumentation Facility
ERP	Effective Radiated Power
Ig	Product of specific impulse of the propulsion system and gravity
ISU	Inertial Sensing Unit
IWAC	Inertia Wheel Attitude Control
K <sub>DC</sub>	Propulsion system constant weight
к <sub>DV</sub>	Propulsion system tankage factor (system weight/fuel weight)
MCTF	Mean cycles to failure
MTTF	Mean time to failure
м <sub>v</sub>	Mean of the $\Delta V$
P <sub>FA</sub>	Probability of mission failure attributable to the astrionics system
P <sub>FR</sub>	Probability of mission failure due to inadequate hardware reliability
P <sub>FT</sub>	Probability of exceeding target miss criteria
P FV	Probability of having insufficient $\Delta V$ fuel
P <sub>FTR</sub>	Probability of failure due to inadequate reliability or target miss
R <sub>T</sub>	Square root of the trace of the target miss covariance matrix, or the standard deviation of any single miss parameter of interest
Rv	Square root of the trace of the $\Delta V$ covariance matrix

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# LIST OF MAJOR SYMBOLS AND DEFINITIONS (Continued)

RTG	Radioisotope thermoelectric generator
WAC	Weight of attitude control unit
WAS	Weight of the entire astrionics system
W <sub>DV</sub>	Weight of the entire propulsion system
W <sub>F</sub>	Weight of $\Delta V$ fuel
WICP	Weight of onboard inertial sensing unit, computer, electrical energy source, electro-optical sensors, communications subsystem, radars, attitude control, and wiring
W <sub>NA</sub>	Weight of spacecraft less astrionics
$w_{_{\mathrm{T}}}$	Total spacecraft weight
X <sub>MISS</sub>	Allowed magnitude of any miss parameter, or vector of interest, at the target
∆V	Velocity change capability
μ	Spacecraft mass ratio
${}^{\psi}\mathbf{T}$	X <sub>MISS</sub> /R <sub>T</sub>

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## DEVELOPMENT OF AN EVALUATION TECHNIQUE FOR INTERPLANETARY MISSION ASTRIONICS

Interim Report for the Period of

February 1, 1969, to February 1, 1970

BATTELLE MEMORIAL INSTITUTE Columbus Laboratories

#### INTRODUCTION

This report presents the results of the work on "Development of an Evaluation Technique for Strapdown Guidance Systems", performed in accordance with modification No. 2 dated February 1, 1969, to the statement of work of Contract No. NAS 12-550. The purpose of this modification was to extend the evaluation techniques developed for astrionics systems which employ aided inertial guidance systems operating on interplanetary flyby missions to include the astrionics required for orbiter, lander, and multiple planet swingby missions. This volume presents a summary of the study results, detailed technical discussion, recommendations, and conclusions. To further the reader's understanding of the organization of this report, the principal items of work for this reporting period are listed below.

#### Added Study Elements

The various tasks performed to extend the evaluation techniques developed for interplanetary flyby missions astrionics systems, to orbiter, lander, and multiple planet swingby mission astrionics were:

- The additional astrionics required for the approach phase of orbiter, lander, and multiple planet swingby missions were incorporated into the effectiveness evaluation by inclusion of their weight, electrical power, reliability, and accuracy.
- (2) The capability to investigate the tradeoffs between attitude control mechanizations employing gas reaction jets, control moment gyroscopes (CMG's), and reaction wheels was provided. The impact of flight control requirements on astrionics effectiveness was included.

- (3) Two specific multiple midcourse correction strategies approved by the NASA/ERC Technical Monitor were investigated to determine their influence on the penalty functions.
- (4) The impact of the requirements for communications between the spacecraft and the Earth were investigated to determine the requirements placed on the other astrionics and the resultant influence on the penalty functions. Pointing accuracy, spacecraft stabilization techniques, information rates, and power requirements were considered in determination of the communication subsystem parameters. The case of the spacecraft being eclipsed by any celestial body was examined to determine the impact on the onboard astrionics.
- (5) The effect of switching on and off the various astrionics subsystems during the mission was modeled to include the reliability degradation associated with the switching.
- (6) The computer programs developed under this contract were modified and exercised at the written technical direction of the NASA/ERC Technical Monitor.

#### Guidelines

The additional astrionics required for the approach phase of orbiter, lander, and multiple planet swingby missions, in addition to the propulsion system requirements for these missions, required making modifications to the previously reported work (Reference 1). The factors which have to be considered in applying the evaluation method are primarily those associated with (1) the class of missions for which the evaluation technique has been specifically developed and (2) the astrionics design philosophy.

#### Interplanetary Missions

The evaluation technique, as presently structured, provides a measure of index of astrionics system performance for the class of interplanetary missions. These missions include flyby, orbiter, lander, and others such as multiple planet swingby.

<u>Flyby</u>. A single planet flyby mission requires that the spacecraft pass close to the target planet at some nominal periapse. No propulsion system is carried for orbit insertion about the target planet. The evaluation of astrionics for this mission requires specifying the acceptable periapse uncertainty and probability of mission failure which can be attributed to astrionics system failure.

Orbiter. An orbiter mission requires that the spacecraft carry a propulsion system capable of inserting the spacecraft into a nominal orbit about the target body. The insertion burn usually occurs at nominal periapse. The evaluation of astrionics for this mission requires specifying acceptable uncertainties of the target body orbital elements and probability of mission failure which can be attributed to astrionics system failure.

Lander. A lander mission frequently has all the requirements of an orbiter mission as well as the required propulsion system and astrionics needed to accomplish a "soft" landing. The evaluation of astrionics for this mission requires specifying the nominal descent burn time, landing area, and acceptable probability of mission failure which can be attributed to astrionics systems.

<u>Multiple Planet Swingby Missions</u>. A multiple planet swingby mission requires specifying the velocity and position vectors at each planetary encounter. Deviations from these nominal conditions will, unless adequately corrected, result in excessive miss parameters at subsequent encounters. The astrionics evaluation for such missions requires specifying the acceptable position and velocity uncertainties at each encounter and the probability of mission failure which can be attributed to the astrionics systems.

#### Mission Selected for Analysis

In choosing the missions to be used for the present work, the Jupiter flyby mission described in Reference 1 was retained and a second mission was sought for orbiter-lander illustration. A Jupiter orbiter and/or lander was initially considered. Investigation of the feasibility of such a mission revealed that knowledge of the planet is limited. The facts that the planet is large and cold are undisputed. From this point, however, fact and assumption become increasingly inseparable.

It is generally assumed that the planet is composed of a solid core, permanently covered with a thick layer of ice, and surrounded by an atmosphere of hydrogen and ammonia. This atmosphere exhibits a density of about 0.3

 $\frac{gm}{cm}$  near the surface. Therefore, it has been said that there exists no definite  $cm^3$ 

dividing line between surface and air, but that the surface changes gradually from solid to slush to gas. It is therefore not entirely clear what "lander" should mean, but it is defined here as signifying a device that will come to rest on surface material of sufficient density to support the spacecraft's weight.

In examining the requirements for landers, there arise two distinct cases: direct landers and indirect landers. Direct landers are launched from Earth, and land on the planet without first going into orbit around it. It is immediately apparent that in the case of Jupiter it is virtually impossible to achieve a soft landing by the direct method. The  $\Delta V$  to be overcome is the

 $Vv'_{escape} + v'_{escape}$  approach. This is roughly equal to the escape velocity at the surface of the planet for reasonable approach trajectories, or about 197,300 feet/second (Reference 2). There are, at this time, no rockets that can provide such a  $\Delta V$ .

For an indirect lander, the spacecraft is first put into an orbit around the planet, and then a  $\Delta V$  is applied to effect the landing. Taking as a typical case an orbiter at 2 planet radii (note that 1 radius is the nominal surface), the  $\Delta V$  required to land depends on whether this is an all-propulsion system or an aerodynamic system.

At 2 radii, the escape velocity for Jupiter is about 139,700 feet/second, and the circular orbital velocity is  $139,700/\sqrt{2} \approx 98,700$  feet/second.

The only method that may conceivably work is to employ purely aerodynamic braking in the early phase. At the velocity that the spacecraft will arrive, direct descent is out of the question. Rather, the spacecraft should be allowed to go into orbit, and then as it repeatedly passes through the outer atmosphere it will slow down enough to fire a retromotor for the landing. The spacecraft cannot be allowed to employ a purely gravitational descent because the high density of the lower atmosphere would cause its complete destruction. Consider a spacecraft which is to be initially placed in an orbit of 2 radii perijove and 20 radii apojove. For the present reference trajectory approach velocity of 58,000 feet/second, the AV required for insertion into this eccentric orbit is 18,150 feet/second (see Reference 3 for curves used in calculation), as compared to 52,400 feet/second necessary for insertion into a circular 2-radii orbit. (Note that the present reference trajectory has a much higher approach velocity than desirable for orbiters or landers). Obviously, it is more economical from fuel and scientific payload considerations to have a highly eccentric orbiter rather than a circular one. The retro required for landing from the eccentric orbit would, even with aerodynamic braking, exceed technological capabilities for the immediate and foreseeable future. It was therefore decided that alternate orbiter-lander missions be examined.

Approved NASA missions were examined and a Mars orbiter-lander was selected due to its high probability of being realized. For this reference mission, it was assumed that the lander was carried as part of the orbiter's payload. The reason for this approach is that this corresponds with the previously planned Voyager and presently planned Viking missions.

For the Voyager and Viking missions, the requirements was established that anything reaching the Mars surface should be sterile. In addition, the density of the planet's atmosphere is very low which limits the available aerodynamic braking. This necessitates the lander mass be kept as low as possible.

Therefore, the astrionics subsystems used during the interplanetary and insertion phases of the mission should not be included in the lander's mass. Only those subsystems necessary for a successful mission completion should be utilized. The lander astrionics are independent of the ones analyzed during the earlier phases of the mission. The results presented in this report consider only the astrionics used from Earth launch to Mars orbit insertion.

#### Astrionics Design Philosophy

Navigation and guidance of the launch vehicle which would be used for launching of interplanetary probe spacecraft were assumed to be under control of the astrionics subsystems contained above the final launch vehicle stage. Ιt was assumed that the astrionics considered in the present work are an integral part of the spacecraft as shown in Figure 1. It would not be necessary for the astrionics to be an integral part of the spacecraft if only launch vehicle navigation and guidance were considered. In this case, certain subsystems such as the spacecraft attitude control would not be considered in the analyses. In addition, alternate electrical energy sources such as batteries would be used in the launch vehicle astrionics. Since the results presented in this report consider the astrionics to be part of the spacecraft, the spacecraft electrical energy source, propulsion system, and attitude control system are considered in the evaluation. In the case of flight control systems, it is quite possible that certain components of the system will be located on lower stages of the launch vehicle. It is also possible that power may be supplied to these components from a source other than the spacecraft's. In such case, only those components inside the spacecraft are considered for weight and power calculations. All components are considered in reliability calculations, taking into account their effective periods of operation.

For a flyby mission, such as the Jupiter flyby examined in this report, the onboard approach radars shown in Figure 1 may not be required. For an orbiter-lander mission, such as the Mars mission examined in this report, Figure 1 represents the astrionics of the orbiter spacecraft from launch through orbit insertion. The lander is assummed to carry its own astrionics subsystems that become active prior to separation from the orbiter. In the present evaluation scheme, the entire lander is considered part of the payload.

The attitude control unit provides the required torques for stabilizing and maneuvering the spacecraft. Various mechanizations are possible. The mechanizations that led to the results reported herein are:

- A set of six pairs of thruster nozzles driven with cold gas from a single tank;
- (2) A set of four control-moment gyros; and
- (3) A set of three orthogonal reaction wheels.

Although any of the three mechanizations may be employed, combinations of the three are not permissible under the present structure of the program.



\* Deep Space Instrumentation Facility

FIGURE 1. BLOCK DIAGRAM OF INTEGRATED ASTRIONICS

The inertial sensing units can be either strapdown or gimballed. The results presented in this report consider only strapdown inertial sensing units.

A centralized general purpose digital computer is assumed to provide all data management. For example, this includes:

- (1) Navigation, guidance, and control computations;
- Processing of data input and output to the communicaations subsystem;
- (3) Control of all subsystem functions such as sequencing; and
- (4) Data storage and processing.

Components of the communications subsystem include both onboard equipment and Earth-based tracking radars. The onboard equipment is assumed to consist of the necessary antennas, transmitter, command decoder, and multiplexer.

Approach radars include both range and range-rate units. They are used in the orbiter-lander mission and are omitted in the flyby mission.

Electro-optical sensors include horizon sensors, sun sensors, planet sensors, and gimballed or strapdown star trackers. Excluding planet sensors, the remaining sensors have been evaluated for the Jupiter flyby mission (Reference 1). Sun sensors and strapdown star trackers were used in the determination of the results presented in a later section of this report.

The flight-control system includes, in addition to the computer, one or more sets of rate gyros, one or more rate-integrating gyros, a lateral accelerometer, an angle of attack sensor, passive filters, wiring, and electromechanical actuators. These sensors may be located on the launch vehicle as well as the spacecraft. Their location is determined by control system stability analysis. One or more of these components may be missing for a particular mission.

The electrical power source and distribution network include the source of the electrical energy such as a radioisotope thermoelectric generator (RTG) or batteries, as well as power supplies and wiring.

#### SUMMARY

The evaluation techniques developed for astrionics systems have been extended to include alternate attitude control configurations, flight control subsystems, communications requirements, astrionics switching considerations, and optimum midcourse correction strategies.

Mission requirements, mission event schedules, and spacecraft design characteristics are considered in the evaluation of the effectiveness of candidate astrionics systems, and in the determination of the effectivenesss of specific navigation updating and midcourse correction schedules.

Effectiveness evaluation is based on a cost effectiveness approach with cost defined to be the total astrionics system weight and effectiveness defined to be the probability that the astrionic system operates correctly. Using this cost or weight effectiveness model, several performance indices have been developed. These may be broken into two categories. The first category requires a specified effectiveness or probability of success and uses weight as the performance index, while the second category has a specified weight allowance for the astrionics and uses the ineffectiveness or probability of failure as the performance index.

#### Penalty Functions

Three different penalty functions were developed during the first phase of this contract and are discussed in detail in the Technical Discussion section of this report. The three penalty functions (modes) are defined as follows:

Mode 1. The probability of mission failure due to lack of astrionics reliability and accuracy,  $P_{FA}$ , is

a specified constant. Another specified constant is all nonastrionics weight,  $W_{NA}$ . The penalty function is the astrionics system weight,  $W_{AS}$ , and is obtained by complete analysis of the astrionics, mission schedule, and spacecraft data. The total astrionics weight is defined to be the sum of the weights of: (1) the astrionics hardware including the inertial sensing unit; (2) the electrical energy source and distribution network; (3) the attitude control unit,  $W_{AC}$ ; and (4) the propulsion system,  $W_{DV}$ . An increase in the combined astrionics system weight necessary to assure a given influence, by the astrionics system, on probability of mission success is reflected in an increased launch weight,  $W_{T}$ .

- Mode 2. The total launch weight, equal to the sum of the nonastrionics weight plus the combined astrionics system weight, is a specified constant. In addition, the nonastrionics weight is specified as is the combined astrionics system weight. Any decrease in astrionics system hardware or power source weight is offset with an increase in propulsion system weight or vice versa. The probability of mission failure due to lack of reliability or accuracy is the penalty function.
- Mode 3. The third mode involves specified total launch weight and probability of mission failure due to lack of astrionics reliability and accuracy. The combined astrionics system weight is the penalty function. In this mode, the nonastrionics weight (useful payload) is the difference between the launch weight and combined astrionics system weight. Thus, for increasing W<sub>AS</sub>, W<sub>NA</sub> is reduced.

The three penalty functions are shown in Table I for comparison.

Mode	P <sub>FA</sub>	W <sub>NA</sub>	WAS	W <sub>T</sub>	Remarks
1	F	F	Р	V	Fixed Nonastrionics Weight and Probability of Astrionics Failure
2	Р	F	F	F	Fixed Total Weight and Astrionics Weight
3	F	v	Р	F	Fixed Total Weight and Probability of Astrionics Failure

TABLE I. THREE PENALTY FUNCTIONS FOR EVALUATION OF ASTRIONICS SYSTEMS\*

\* V  $\Delta$  Variable with System, F  $\Delta$  Constant, P  $\Delta$  Penalty function.

For each of the modes, the minimum value of the penalty function defines the best system.

Evaluations discussed in this report were made using Mode 3. The probability that the astrionics system operates correctly,  $1 - P_{FA}$ , was specified as a mission constraint and the combined astrionics system weight,  $W_{AS}$ , is the penalty and is obtained by complete analysis of the astrionics, mission schedule, and spacecraft data.

Sensitivity of each penalty function with respect to specific system hardware parameters is expressed as the percent change in penalty per percent change in data. These sensitivities allow easy determination of the system parameters and components which affect the penalty function most directly (large sensitivity magnitude). The algebraic sign indicates which direction the penalty changes for an increase in the system parameter. Further explanation of the penalty functions is contained in the Technical Discussion section of this report.

#### System Parameters

The parameters used in the evaluation techniques are, in general: (1) weight; (2) power; (3) mean-time-to-failure (MTTF); (4) mean cycles to failure (MCTF); and (5) performance which depends upon the functions of the particular subsystems. Of these parameters, the estimation of performance (accuracy) of aided inertial guidance systems which utilize aid measurements and Kalman filtering in the updating of system errors is the most difficult to achieve.

Techniques to calculate the weight of the inertial sensing unit (ISU), propulsion subsystem, computer subsystem, and power subsystem were developed under Item 2 of the contract and are discussed in Reference 4. The total system weight is the summation of the weights of each of the subsystems. The weight of the attitude control system is estimated by the methods described in Reference 1 and in the Technical Discussion section of this report. The weight of the flight control system is estimated by summing the weights of the various components. The communication subsystem weight is estimated for the onboard transmitter and antenna. The weight of the electro-optical subsystem is the total weight of all electro-optical sensors used during the mission.

The power required by the astrionics system is estimated by summing the power required by each of the subsystems as a function of the system operating schedule for the mission of interest. The weight of the power sources is estimated from the resulting mission power load profile. The peak load determines the capacity of the RTG. The total weight of the power subsystem is the summation of the weights of the RTG, power conditioning and distribution equipment, and the wiring between subsystems.

The reliabilities of the ISU, flight control system, propulsion subsystem, computer subsystem, and power subsystem are estimated as discussed in Reference 4. The reliability of the attitude control subsystem is estimated by the methods described in the Technical Discussion section of this report and Reference 1. The Weibull distribution (Reference 4) with  $\alpha = 1$  is used for the communications subsystem as well as the electro-optical sensors. The operating time for the various candidate aids depends upon the mission schedule. In addition, the effects of subsystem switching have also been included in the reliability calculation.

#### Computer Programs

The calculation of the three penalty functions and the necessary estimation of the system parameters have been coded into a deck of FORTRAN subroutines. The subroutines, with a short, simple, main program calculate the necessary system parameters and evaluate them according to the specified penalty function.

#### Data Requirements

Data needed to run the program are divided into four categories. The first three involve data describing the mission and spacecraft and include: (1) injection error sensitivities as computed by the Strapdown Error Analysis Program (SEAP) or Platform Error Analysis Program (PEAP); (2) state transition matrices generated by the n-body program; and (3) data describing mission values, ISU design values, and spacecraft subsystems. The fourth category is data describing candidate components (accelerometers, gyroscopes, electrooptical sensors, communication subsystem, and computers) and includes: component (1) weight, (2) dimensions, (3) excitation power, (4) reliability, and (5) error coefficients. Computer data required are similar to that for gyros and accelerometers except that the navigation errors are estimated based upon the specified number of bits used to store each element of the attitude matrix, attitude update integration frequency, integration scheme (rectangular, Runge-Kutta second order, or Runge-Kutta fourth order).

### Output Options

The following types of output are available from the program:

- Level 1 Evaluation. A level 1 evaluation produces

   a 1-page report summarizing the astrionics subsystem
   parameters and the effectiveness evaluation
   calculations.
- (2) Level 2 Evaluation. Level 2 evaluation includes a detailed printing of all mission operations and error analysis quantities as a function of time from the beginning of the mission to arrival at the target point.
- (3) Sensitivity Analysis. Sensitivity is defined to be the percent change in effectiveness per percent change in any data value. Sensitivity reports may be generated for all mission, spacecraft, and astrionics data or selected subsets of data. These reports aid in identifying the subsystem parameters and mission values with the greatest impact on astrionics effectiveness.

- (4) Optimum System Selection. The optimum suite of astrionics is found by successive substitution of candidate subsystems for evaluation. The substitution algorithm is similar to a steepest descent technique with the possibility of finding only local minima. Multiple starting points are used to minimize the probability that the system found is a local rather than a global optimum.
- (5) Optimum Multiple Midcourse Corrections. Selection of an optimum sequence of updates and midcourse corrections is possible, as an option. A table of all sequences tried by the algorithm is printed.

The first four types of output were programmed during the first two years of this study and extensive examples of these are shown in References 1 and 4. The optimum midcourse correction algorithm is discussed at length in this report and a sample output is shown in Appendix A.

#### Mission Characteristics

Two mission are considered in this report. The first, a Jupiter flyby has been discussed in detail in Reference 1 and 4. The second mission is a Mars orbiter/lander mission which is discussed in Appendix A. This mission is based on Viking data when possible but is not a conclusive study of astrionics for the Viking mission. (Reference 5).

#### Mission Schedule

The computer program accepts a flexible mission schedule defining astrionics operations. To avoid lengthy repetition of similar sequences, subschedules have been introduced. A set of scheduled astrionics operations such as a midcourse correction sequence can be defined as a subschedule. The mission schedule then states the times at which the subschedule is to be executed.

#### Summary of Effort on Added Study Elements

#### Planetary Approach Sensors

Approach sensors carried onboard the spacecraft are used: (1) to provide information for state updating at various times as specified in the schedule for any particular mission; and (2) to aid in correct attitude orientation of the spacecraft prior to midcourse corrections.

In determining whether any particular sensor should be employed, the assumption was made that, except in cases when the spacecraft is eclipsed by another body, Earth-based updating information is available from the DSIF. With this assumption, only sensors capable of improving on the accuracy of the DSIF-updated navigation system need be considered. Onboard range and rangerate radar and electro-optical sensors were examined. Range and rangerate radars were modeled and incorporated in the computer program. Planet angle sensors were analyzed but not modeled. If the assumption is made that the DSIF provides range as well as range-rate data with the accuracies stated in the available literature (Reference 6), the planet angle sensors do not offer a significant improvement in updating information. Sun sensors and star trackers are carried onboard the spacecraft and measurements with these are possible during the approach phase. These measurements are used for updates as specified in the mission schedule.

A Mars orbiter mission was analyzed, as an example, from launch to retro-firing for orbit injection. Although sensors necessary for landers were considered, no modeling of landers was done.

#### Alternate Attitude Control Schemes

In addition to cold gas reaction jets, discussed in Reference 1, two alternate attitude control mechanizations have been analyzed and modeled. These are control moment gyros (CMG) and inertia wheel attitude control (IWAC). Unlike gas reaction jets, both these schemes are mass-conservative mechanizations, operating by shifting the orientation of the spacecraft's momentum vector.

The CMG system is not only mass conservative but, in most cases, momentum conservative. The gyros are kept rotating at constant speed and momentum is transferred between spacecraft axes by torquing the gyros. IWAC systems are not momentum conservative. They employ wheels with fixed orientations with respect to the vehicle axes, and the spacecraft total momentum vector is changed by slowing down or speeding up the wheels.

Sizing of the system is accomplished by calculating the torque necessary to perform the specified maneuvers and overcome external disturbances. Once the minimum adequate torque is established and the maximum precession velocity is postulated for the system, then the necessary momentum is known. The postulated precession velocity is not permitted to exceed the maximum angular velocity for which successful acquisition of the Sun and star by the onboard sensors is possible.

The various possible attitude control mechanizations are compared in Table II. Only one set of inertia wheels is used in the IWAC system. If separate fine and coarse control wheels were employed, it is reasonable to expect a small increase in weight and some decrease in the required power.

Both CMG and IWAC mechanizations result in a higher penalty than the gas-reaction jets, mainly due to the subsystem weight difference. In multiplemidcourse missions, the penalty may also be affected by degraded reliability.

Attitude Control Mechanization	Attitude Weight (lbs)	Control Power (watts)	Penalty <sup>*</sup> (1bs)
Gas Reaction Jets (Equal Thrusts)	21.218	10.00	384.047
Gas Reaction Jets (Unequal Thrusts)	21.215	10.00	384.326
Control Moment Gyros	29.019	9.407	391.567
Inertia Wheels	30.422	14.978	394.965

# TABLE II.COMPARISON OF ATTITUDE CONTROL SYSTEMS<br/>ON A JUPITER SWINGBY MISSION

\* All other subsystems contributing to the penalty are identical for each mechanization.

This effect did not become apparent in the Jupiter flyby mission on which the systems were evaluated. Detailed analysis and results appear in the Technical Discussion part of this report.

#### Flight Control Requirements

An investigation of the impact of flight control requirements on astrionics effectiveness was carried out. The analysis was hampered by the fact that the flight control components differ according to the launch vehicle as well as the mission characteristics. No model was therefore possible for a generalized approach towards synthesis of such systems.

If the flight control components and their locations are known, it is possible to calculate the effect that their power requirements, reliability, and in some cases, weight will have on the overall penalty.

In addition, evaluation of the requirements flight control imposes on the onboard computer is possible on the basis of computational speed and memory capacity. For a particular mission, these requirements should be input as data. The data bank is searched for a computer meeting the given requirements. More detailed evaluations were not implemented in this study, due to their dependency on individual mission and vehicle parameters.

#### Effects of Subsystem Switching

The effect of switching subsystems on and off on the astrionics reliability is considered. An exponential model is assumed for computing the probability of failure due to on and off switching with the mean number of cycles to failure (MCTF) of each subsystem specified as input data.

## Communications Requirements

The impact of the spacecraft communications subsystem upon the penalty was considered by optimizing the communications subsystem weight. In this application, the transmitter must deliver a sufficient amount of power to the spacecraft antenna in order to satisfy a given requirement on the effective radiated power (ERP). The required ERP is calculated from trajectory and Earth based station data. A comparison of an optimized transmitter versus a specified 20 watt transmitter weighing 10 lbs. is shown in Table III.

	Transmitter	
	Optimized	Specified
ERP (KW)	3.36	3.36
Antenna Gain (DB)	23.6	26.2
Antenna Weight (1b)	14.615	20.98
Antenna Pointing Tolerance (deg)	1.084	.802
Iransmitter Output Power (watts)	36.536	20.0
Transmitter Weight (lbs)	3.727	10.0
Energy Source Weight (1bs)	114,772	109.067
Penalty <sup>**</sup> (lbs)	377.114	384.047

# TABLE III.COMPARISON OF OPTIMIZED VERSUS SPECIFIEDTRANSMITTER ON A JUPITER SWINGBY MISSION

\* Includes electrical power requirements of other astrionics subsystems which are identical for both cases.

\*\* Attitude control using equal thrust gas jets.

## Multiple Midcourse Correction Strategies

A critical review of seven selected papers relating to the development of an optimum midcourse correction policy was made and one of these, a paper by C. G. Pfeiffer, was selected for implementation. This optimization procedure, an adaptation of Bellman's dynamic programming algorithm, was applied to the astrionics penalty function to determine an optimum midcourse correction strategy. The results of this application are discussed in Appendix A. A second midcourse correction policy was implemented. This policy was developed by assuming that velocity corrections would be made at given times to minimize the uncertainty in the resulting target miss. A matrix which relates the desired velocity correction vector to the vector of instantaneous computed deviations from the nominal trajectory is determined for two cases. In the first case, execution errors are neglected while, in the second case, an approximation of the errors generated by the midcourse correction is considered. Improvement using this technique was slight and it was removed from the program.

#### TECHNICAL DISCUSSION

## Evaluation Criteria for Interplanetary Multiple Planet Swingby. Orbiter, and Lander Mission Astrionics

Evaluation criteria for interplanetary flyby mission astrionics were formulated and implemented through computer programs in previous phases of work on this contract. This Interim Scientific Report encompasses the third year effort and resultant modifications to the computer programs required to accomplish the objective of this phase. The extension of the effort to include evaluation of astrionics for multiple planet swingby, orbiter, and lander missions necessitated making modifications to the previous astrionics evaluation criteria and system concept.

#### Astrionics System Concept

It was assumed that the modular astrionics system design philosophy (References 1 and 4) is applicable to this study. This philosophy permitted addition of another module to the integrated astrionics configuration used during development of the evaluation techniques for interplanetary flyby mission astrionics (Reference 1). The module added is approach guidance radar. Note that the electro-optical aid subsystem now includes planetary approach sensors. In addition, as depicted in Figure 2, the propulsion subsystem was changed from a midcourse correction propulsion subsystem to a  $\Delta V$  propulsion subsystem. This change was necessary since the same propulsion subsystem is used for midcourse correction  $\Delta V$ , orbit insertion  $\Delta V$ , and orbit trim  $\Delta V$  in many spacecraft designs (Reference 5 and 7).

To evaluate the modified astrionics system, some modifications were made to the penalty function as originally presented in References 1 and 4. The modified penalty function is discussed in the following section.

### Penalty Function Analysis

The effectiveness of an astrionics system on a specific spacecraft and mission is evaluated by one of three penalty functions. The astrionics system is described by the seven system parameters shown in Table IV.  $M_V$ , the mean  $\Delta V$ , is the nominal retro burn for orbit insertion if accomplished with the same engine used for the midcourse corrections. Since the orbit insertion burn may include guidance to correct velocity deviations, it is necessary to include the propulsion subsystem in the astrionics analysis. If a separate engine is used for orbit insertion and corrective guidance does not occur,  $M_V$ is set to zero and the retro subsystem is not included in the astrionics analysis. The penalty functions are calculated from the above system parameters and the mission and spacecraft parameters shown in Table V.



Power Subsystem - - Supplies all required electrical power.

\* Horizon Sensor --- Parking Orbit
 Star Tracker/Sun Sensor --- Injection to Midcourse
 Planetary Approach Sensor --- Encounter

FIGURE 2. BLOCK DIAGRAM OF ONBOARD INTEGRATED ASTRIONICS SUBSYSTEMS

# TABLE IV. ASTRIONICS SYSTEM PARAMETERS

Symbol	Definition
P <sub>FR</sub>	Probability of mission failure due to inadequate hardware reliability.
<sup>R</sup> V	Square root of the trace of the $\Delta V$ covariance matrix.
DV	Degrees-of-freedom of the midcourse $\Delta V$ covariance.
MV	Mean of the $\Delta V$ .
R <sub>T</sub>	Square root of the trace of the target miss covariance matrix, or the standard deviation of any single miss parameter of interest.
D <sub>T</sub>	Degrees-of-freedom of the target miss covariance.
WICP	Weight of onboard inertial sensing unit, computer, electrical energy source, electro- optical sensors, communications subsystem, radars, attitude control, and wiring.

# TABLE V. MISSION AND SPACECRAFT PARAMETERS

Symbol	Definition
P <sub>FA</sub>	Probability of mission failure attributable to the astrionics system.
W <sub>NA</sub>	Nonastrionics spacecraft weight.
W <sub>T</sub>	Total spacecraft weight.
X <sub>MISS</sub>	Allowed magnitude of any miss parameter, or vector of interest, at the target.
K DV	Propulsion system tankage factor (system weight/ fuel weight).
K <sub>DC</sub>	Propulsion system constant weight.
Ig	Specific impulse of the propulsion system times gravity.

The three penalty functions are defined in Table VI.

			T.7	
Mode	<sup>P</sup> FA	WNA	WAS	<sup>w</sup> Т
1	F	F	Р	V
2	Р	F	F	F
3	F	V	Р	F

TABLE VI. PENALTY FUNCTION DEFINITION\*

\* F  $\Delta$  Fixed, P  $\Delta$  Penalty, V  $\Delta$  Variable with System

Penalty function, Mode 1, assumes that a certain probability of mission failure attributable to astrionics ( $P_{FA}$ ) is reasonable and that non-astrionics spacecraft weight ( $W_{NA}$ ) is fixed. The astrionics system weight is calculated and used as the penalty function.

Penalty function, Mode 2, assumes nonastrionics, astrionics system, and total spacecraft weights are constants with the probability of mission failure attributable to astrionics ( $P_{FA}$ ) variable and used as the penalty function.

Penalty function, Mode 3, assumes that a certain probability of mission failure attributable to astrionics (P<sub>FA</sub>) is reasonable and that total spacecraft weight (W<sub>T</sub>) is fixed. The astrionics system weight (W<sub>AS</sub>) is variable and used as the penalty function.

Calculation of the penalty function under any of the three modes will involve calculating the intermediate quantities defined in Table VII.

Symbol	Definition		
P <sub>FV</sub>	Probability of having insufficient $\Delta V$ fuel.		
P <sub>FT</sub>	Probability of exceeding target miss criteria.		
P <sub>FTR</sub>	Probability of failure due to inadequate reliability or target miss.		
W <sub>F</sub>	Weight of $\Delta V$ fuel.		
W <sub>DV</sub>	Total weight of propulsion system.		
$\Delta V$	Velocity change capability.		
$\psi_{\mathrm{T}}$	X <sub>MISS</sub> /R <sub>T</sub>		

## TABLE VII. INTERMEDIATE QUANTITIES USED IN CALCULATING THE PENALTY FUNCTIONS

A detailed discussion of the steps used to calculate each penalty mode is given below.

<u>Penalty Mode 1</u>. Probability of missing the target  $(P_{FT})$  is calculated from the system parameters describing accuracy at the target as follows:

$$P_{FT} = Prob \Psi_{T}, D_{T}$$

where

$$\psi_{\rm T} = X_{\rm MISS}/R_{\rm T}$$

Note that the definition of  $X_{\rm MISS}$  has been expanded to be completely general.  $X_{\rm MISS}$  for an orbiter would be the acceptable deviation in one or more of the orbital elements. For a lander,  $X_{\rm MISS}$  could include acceptable position or velocity vector deviations at touchdown.  $X_{\rm MISS}$  on multiple planet swingby missions in the acceptable periapsis deviation at the final planet. Deviations from the nominal periapsis at intermediate planets will be accounted for by subsequent midcourse corrections.

The function  $Prob(\psi, D)$  is the probability distribution of the magnitude of a vector with normal, zero-mean components as discussed in Reference 8. A table of this distribution is shown in Figure 3.

The combined probability of missing the target or failing due to inadequate reliability is obtained from

$$P_{FV} = \frac{P_{FA} - P_{FTR}}{1 - P_{FTR}} .$$
 (1)

Note that if  $P_{\rm FTR}$  exceeds  $P_{\rm FA}$ ,  $P_{\rm FV}$  does not exist. In other words, if the probability of failure due to inadequate reliability or target miss is greater than  $P_{\rm FA}$ , even a perfect system (zero probability of insufficient fuel) will exceed  $P_{\rm FA}$ .

The  $\Delta V$  capability needed to achieve the required P  $_{\rm FA}$  is calculated from

$$\Delta V = REQ(P_{FV}, M_V, R_V, D_V)$$

where REQ(P,M,R,D) is a modified version of  $Prob(\psi,D)$  as discussed in Appendix B.

X0000 6816 6989 1.1053 1.2436 1.4433 1.4709 1.5009 1547 3284 3443 3620 3819 4629 5035 5597 . 6662 .7892 .8880 .9910 1.5340 1.5712 1.6138 1.66643 1.7268 .6108 .9446 ,9640 5600 0685 4045 8058 9376 .9501 9638 9969 3.0779 0368 4259. 8556 6847 H6E\*\* **.5**786 .9854 3.0 กักกักกักกับ NNNNNNN 0000X0 NNNNN .1052 1949 2635 .8715 .9231 .0091 .0220 .0930 3.2002 E 0723 ა ç Ö 3.0884 3.1018 3.1165 .1116 алуууу алууу алуууу алууу алуууу алууу алууууу алуууу алуууу алуууу алуууу алуууу алуууу алуууу алуууу •1460 •1874 •2395 •3110 .7936 .8102 3,2359 2.6 1.6995 1. 2.0511 2. 2.0731 2. 2.0975 2. 2.1547 2. 2.1557 2. 2.1916 2. 2.2222 2.2249 2.2249 2.2249 2.5249 2.5249 2.5249 2.5249 2.54935 2.54935 2.54935 2.54935 2.54935 2.549555 2.54955555555555 3.1906 3.2062 3.2237 3.2439 0 X 0 0 0 X 0 1.1004 1.25324 1.55324 1.55476 1.55476 1.55476 1.6573 1.46513 1.47520 .7343 .7343 .8476 2.0955 -3522 .9669 
 Model
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X indicates extrapolation from Gamma Function Table

CUMULATIVE PROBABILITY DISTRIBUTION FUNCTION [Pr( $|x| > \psi \sqrt{\text{TRACE}}$ )] OF

ZERO MEAN, COMPONENTS

VECTOR WITH NORMAL,

ന്

FIGURE

Magnítude/√Trace = ψ

4

The familiar rocket equation,

$$\Delta V = Ig \log_{2}(\mu)$$

,

is used to obtain the spacecraft mass ratio,

$$u = e^{\Delta V/Ig}$$

The spacecraft mass ratio is the initial spacecraft weight divided by the final spacecraft weight and is calculated as follows:

$$\mu = \frac{W_{\rm T}}{W_{\rm T} - W_{\rm F}}$$

The weight of the required fuel is

$$W_{\rm F} = W_{\rm T} \frac{(\mu - 1)}{\mu}$$

The propulsion system weight is estimated using the equation

$$W_{\rm DV} = W_{\rm F} K_{\rm DV} + K_{\rm DC}$$

Defining

$$W_{T} = W_{ICP} + W_{NA} + W_{DV}$$

substitution into the equation for  $W^{}_{\rm F}$  in terms of  $W^{}_{\rm T}$  and  $\mu$  yields

$$W_{\rm F} = \frac{(\mu - 1)(W_{\rm ICP} + W_{\rm NA} + K_{\rm DC})}{\mu - (\mu - 1) K_{\rm DV}}$$

The effective weight of the complete astrionics system is then

$$W_{AS} = W_{ICP} + W_{DV} \qquad (2)$$

This is the desired penalty function. The above equations are shown in flow chart form in Figure 4.

<u>Penalty Mode 2</u>. The probability of failing due to inadequate reliability of miss at the target ( $P_{\rm FTR}$ ) is calculated as in Penalty Mode 1. The combined astrionics system probability of failure ( $P_{\rm FA}$ ), the penalty of Mode 2, is calculated by including the probability of insufficient midcourse correction fuel ( $P_{\rm FV}$ ).

The total spacecraft weight (WT) and nonastrionics spacecraft weight (WNA) define the total astrionics system weight to be

$$W_{AS} = W_{T} - W_{NA}$$

The propulsion system weight is assumed to be

$$W_{\rm DV} = W_{\rm AS} - W_{\rm ICP}$$

Thus, the fuel weight is determined from the equation,

$$W_{\rm F} = \frac{W_{\rm DV} - K_{\rm DC}}{K_{\rm DV}}$$

and the spacecraft mass ratio is

$$\mu = \frac{W_{\rm T}}{W_{\rm T} - W_{\rm F}}$$

The  $\Delta V$  capability is found from the rocket equation,

$$\Delta V = Ig \log_{e}(\mu)$$

and the probability of insufficient fuel is

$$P_{FV} = PRM(\Delta V, M_V, R_V, D_V)$$
System Parameters



FIGURE 4. CALCULATION OF PENALTY, MODE 1

where PRM(V,M,R,D) is the inverse function of REQ(P,M,R,D)

The combined probability of mission failure attributable to the astrionics is

$$P_{FA} = P_{FTR} + P_{FV} - P_{FTR} P_{FV}$$
(3)

and is the desired penalty, Mode 2. The above equations are shown in flow chart form in Figure 5.

Penalty Mode 3. Penalty Mode 3 is similar to Penalty Mode 1 in that the penalty is the effective weight of the astrionics system (WAS). However, the total spacecraft weight (WT) is held constant under Mode 3 unlike Mode 1 where the nonastrionics weight was held constant.

The probability of failing due to reliability or miss at the target (PFTR) is calculated as under Mode 1. The probability of having insufficient fuel is obtained from the equation

$$P_{FV} = \frac{P_{FA} - P_{FTR}}{1 - P_{FTR}}$$

The result is used to compute the required  $\Delta V$  capability from the equation

$$\Delta V = REQ(P_{FV}, M_{V}, R_{V}, D_{V})$$

With the  $\Delta V$  requirement known, the mass ratio is

$$u = e^{\Delta V / Ig}$$

Since the total spacecraft weight is known, the required fuel weight may be obtained directly from

$$W_{\rm F} = W_{\rm T} \frac{(\mu - 1)}{\mu}$$

The total effective astrionics system weight is obtained by adding the propulsion system weight to the weight of the other astrionics subsystems as shown below,

$$V = REQ(P_{FV}, M_{V})$$





FIGURE 5. CALCULATION OF PENALTY, MODE 2

$$W_{AS} = W_{ICP} + W_{DV}$$

The equations for Penalty Mode 3 are shown in Figure 6.

## Astrionics Subsystem Analysis

The following sections of this report describe the analyses of the added astrionics subsystems and the modifications to the error analysis and penalty evaluation techniques.

### Deep Space Instrumentation Facility.

The Deep Space Instrumentation Facility (DSIF) provides communications with and tracking of the spacecraft through most of the mission. Brief periods may exist when the DSIF can not view the spacecraft due to occultation by celestial bodies such as the target planet or the moon.

The communications function of the DSIF is considered under the communications requirements discussion later in this report. The tracking function is included in the error analysis as updating of the onboard navigation estimates of the states. Prior to the updating of the onboard system, extensive ground based calculations are made using the DSIF doppler radar measurements. The accuracy of the onboard update is a function of the trajectory, the tracking time, and the frequency and accuracy of the doppler measurements. The ground based calculations and basic doppler errors are not modeled in this study. It is assumed that the results of these measurements and calculations, when transmitted to the onboard navigation system constitute an update of the onboard system range and range rate errors. The accuracy of the range and range rate information, after tracking and statistical smoothing by the ground based calculations are referred to in this report as the DSIF accuracies. As mentioned above, these accuracies are a function of the trajectory and sampling frequencies, but are assumed to be constants, specified as data in this study. The one sigma values used, 50 ft and .004 ft/sec., are achievable with the DSIF under certain trajectory and tracking conditions [one doppler sample per minute for several days (Reference 6)]. Further study is required to ascertain the exact figures for any mission.

### Planetary Approach Sensors

Capability for evaluating planetary approach sensors has been developed and added to the existing computer program.

The approach sensors are treated like any other astrionics subsystem for penalty evaluation purposes. Weight, power, MTBF, MCTF, and accuracies are

# (4)

System Parameters



FIGURE 6. CALCULATION OF PENALTY, MODE 3

required as input data, and the effect of the system on the penalty is then calculated.

Measurements discussed include those made by electro-optical sensors and range and range-rate radars. Sensors actually modeled in the computer program include star and sun sensors and range and range-rate radars. Sensors for planet angle measurements were not modeled. The accuracy of DSIF earthbased tracking minimizes the effect of updates using these less accurate measurements.

A Mars orbiter mission was used to exercise the computer program. Results for this mission appear in Appendix A.

### Radar Measurements

During the approach phase, it is assumed that onboard radars capable of measuring range and range rate are available for updating the navigation system errors. These updates are in addition to the DSIF and star-sun electrooptical sensors used through most of the mission. To include the effects of updates using these radars in the navigation error analysis, it is necessary to know the covariance or standard deviations of the errors in the measurements and the measurement matrix relating the navigation states to the measured quantities.

<u>Radar Accuracy</u>. The fundamental accuracy of range, and range rate (doppler) radars is discussed in many texts (e.g., References 9 and 10). Additional errors associated with the design and installation of radar subsystems contribute significantly to the total error in approach radar measurements. These are discussed in detail in Reference 11. In summary, the errors discussed in Reference 11 are broken into the following three categories:

- (1) Bias errors, which are constant offsets and vary from point to point in the trajectory and thus are functions of the nominal state vector. These include:
  - (a) Terrain bias errors,
  - (b) Uncompensated dynamic-lag errors,
  - (c) Doppler compensation errors, and
  - (d) Pre-amp slope errors.
- (2) Installation and environmental errors associated with the misalignments of the sensors on the spacecraft. These include:
  - (a) Initial mounting errors, and
  - (b) Vehicle distortion due to vibration and temperature changes.

- (3) Fluctuating errors which vary randomly with statistics that are a function of the nominal trajectory state vector. These may be broken down to include:
  - (a) Spread spectrum error,
  - (b) Oscillator drift,
  - (c) Range beam modulation error, and
  - (d) Quantitization errors due to digital processing.

All the errors are treated as Gaussian uncorrelated sources and the total measurement errors may be found by taking the RSS of the 10 values of each source. The computer program does not model the approach radars in detail. A single number, the total standard deviation of each measurement, is loaded as data. If additional data is available the program could easily be modified to make the standard deviation of range and range rate.

<u>The Measurement Matrix</u>. The measurement matrix necessary for error analysis is calculated from the partial derivatives of range and range rate with respect to the state vector of position, velocity, and attitude. The range may be expressed as:

$$R = \sqrt{\underline{r} \cdot \underline{r}}$$

and the partial derivatives are:

$$\frac{\partial R}{\partial r_i} = \frac{r_i}{R}$$

Range rate is given by

$$S = \frac{dR}{dt} = \frac{\underline{r} \cdot \underline{v}}{R}$$

with the partial derivatives

$$\frac{\partial S}{\partial r_{i}} = -\frac{\underline{r} \cdot \underline{v}}{R^{2}} \frac{\partial R}{\partial r_{i}} + \frac{v_{i}}{R} = -\frac{(\underline{r} \cdot \underline{v})r_{i}}{R^{3}} + \frac{v_{i}}{R}$$

and

$$\frac{\partial S}{\partial v_i} = \frac{r_i}{R}$$

The complete 2 by 9 measurement matrix is then

$$H = \begin{bmatrix} \frac{r_{i}}{R} (i = 1, 2, 3) & 0 & 0 \\ \frac{R}{R} & 1 & \frac{r_{i}}{R} (i = 1, 2, 3) & \frac{r_{i}}{R} (i = 1, 2, 3) \\ \frac{r_{i}}{R} & \frac{r_{i}}{R} & 0 \end{bmatrix}$$

### Electro-Optical Measurement

Electro-optical sensors are assumed to be carried onboard the spacecraft. These sensors are used for updating the state of the spacecraft and also for attitude orientation. A sun sensor and a Canopus sensor are both employed and their function and operation have already been discussed in Reference 1.

On-board electro-optical sensors may be used for the performance of a variety of measurements. Most of these involve measuring the angle between two bodies such as star-star, star-sun, planet-star, planet-sun, etc. A discussion of these possibilities and their associated measurement matrices can be found in Reference 12.

Choosing between the various possibilities depends on the ultimate purpose of the measurements. For the planetary approach phase, the function desired of the electro-optical sensors is to supplement the information derived from the ISU for state estimation. It has been pointed out (Reference 7) that planet angular diameter measurements and planet-star angle measurements using a star near the normal to the ecliptic are the best candidates for this purpose.

The planet angle measurement technique described below has not been implemented in the present computer program because the accuracies used for the DSIF are such that the effect of the planet angle measurement would not significantly improve the results. Should the need arise, however, it can be readily included in the program.

<u>Planet Angle Measurement</u>. Stadiometry is a well-known method of determining the distance from spacecraft to the sun or planets in our solar system, given their diameters. In this method the angle subtended is inversely proportional to the range of the measuring device. One convenient method of estimating this angle utilizes the measurement of the illuminated portion of the planet. Since the sun will not always be behind the spacecraft, the illuminated portion of the planet as observed from the spacecraft will vary from the full circle to the eclipse with the various crescent shapes in between. An angular sensor would, by necessity, have to be able to distinguish between the possibilities.

One measuring technique is provided by sectioning the illuminated image into an array of cells as illustrated in Figure 7. This could be accomplished by vidicon scanning or by a mosaic sensor. The mosaic sensor,

33

-7



FIGURE 7. CELLED ARRAY OF ILLUMINATED PLANET

although not as fully developed as the vidicon, offers the possibilities of greater reliability, lower power requirements, less computation, and less memory storage to determine the subtended angles. Advances in mosaic sensor state-of-the-art offer considerable competition to the vidicon [i.e., photoconductor-diode arrays with as many as 360 x 360 elements have been produced by thin-film techniques (Reference 13)].

The celled array approach provides two significant measurements as illustrated in Figure 7. The distance between the cells which are the farthest apart, AB, can be used to determine the stadiametric angle. CD, the distance between cells farthest apart but perpendicular to AB, can be used to determine the crescent angle. From these measurements, the distance to the planet and the planet-sun angle can be determined as shown below.

Consider the two diagrams in Figure 8. In the top diagram the Sun is assumed perpendicular to the page. If the Sun's reflection from points A and B can be sensed, then

$$\sin \frac{\alpha_1}{2} = \frac{D_p}{2R_{sc}}$$

where  $D_p$  is the planet diameter and R is the distance from the spacecraft to the planet center. From this

The angle,  $\beta$ , is defined to be the angle measured from the spacecraftplanet vector to the spacecraft-Sun vector.  $\beta$  can be related to  $\alpha_2$  as follows:

$$\theta = \frac{\alpha_1}{2} - \alpha_2 \tag{5}$$

$$\frac{D}{2}\sin(\beta - 90^{\circ}) = -\frac{D}{2}\cos(\beta) = d$$
 (6)

Therefore,

$$\frac{d}{R_{p}} = \tan \theta \tag{7}$$

$$R_{p} = R_{sc} - \frac{D_{p}}{2} \cos (\beta - 90^{\circ})$$
 (8)

$$\alpha_1 = 2 \sin^{-1} \frac{D_p}{2R_{sc}}$$

$$= 2 \sin^{-1} \frac{D_p}{2R_{sc}}$$



۰.

FIGURE 8. DIAGRAMS OF PLANET ANGULAR MEASUREMENT

$$R_{p} = R_{sc} - \frac{D_{p}}{2} \sin(\beta)$$
 (9)

Using Equations (6) and (9) in Equation (7),

$$\theta = \tan^{-1} \left[ \frac{\frac{-\cos(\beta)}{2R}}{\frac{2R}{p} - \sin(\beta)} \right] .$$
(10)

In terms of  $\alpha_2$ ,

$$\alpha_2 = \tan^{-1} \left[ \frac{\frac{-\cos(\beta)}{2R_{sc}}}{\frac{2R_{sc}}{p} - \sin(\beta)} \right] - \sin^{-1} \frac{\frac{D_{p}}{2R_{sc}}}{\frac{2R_{sc}}{p} - \sin(\beta)}$$

The measurement matrix can now be obtained by taking the partial derivatives of the measured quantity  $\alpha_2$  with respect to the elements of the spacecraft-to-planet state vector R, i.e.,

$$\frac{\partial \alpha_2}{\partial \beta} \frac{\partial \beta}{\partial r_i} + \frac{\partial \alpha_2}{\partial R_{sc}} \frac{\partial R_{sc}}{\partial r_i}$$

where  $r_i$ , i = x, y, z, are components of the state vector R.

If the angular field of view (FOV) of the mosaic sensor is  $\psi$  degrees and there are N cells covering this field of view, then the angular resolution

is  $K = \frac{\Psi}{N}$ . K or  $\psi$  can be varied by changing the lens focal length. Hence, a system might be self-adjusting as the spacecraft approaches the planet.

Each cell of the mosaic is either on or off for any one frame as a function of the threshold and illumination. It is assumed that the threshold will be set so that the on state will occur when a section of image having a defined intensity is exposed to 1/2 the active cell area. This would provide an angular error of at least 1/2 K. If two cells are used in determining the subtended angles, the minimum error in measurement for a single frame would be <u>K</u>

 $\frac{K}{\sqrt{2}}$ . Repeated computation after the image is shifted slightly to provide uncorrelated frames would make it possible to average the cell resolution error. This would depend on the correlation between frames and the spacecraft travel during the averaging time.

Other errors in angle measurement attributed to the mosaic are random noise, crosstalk, and nonuniformity of cell sensitivity. Prolonged observation

helps considerably in reducing thermal noise and other random excitations. Errors caused by crosstalk and nonuniformity are strictly a function of the mosaic fabrication and structure. These errors are considered as bias errors and are not reduced by time or frame averaging.

Errors in angle measurement not related to the instrument are those caused by so called background clutter (i.e., stars, meteroids, cosmic dust, planet dark spots, etc.). It is believed that image enhancement techniques will help considerably in reducing these errors. The errors attributed to this factor may be lumped into one variance figure.

The total error in angular measurement can be determined by summing the squares of the contributing variances as follows:

 $\sigma_{\text{TOT}}^2 = (\sigma_{\text{CEL}}^2 + \sigma_{\text{INST}}^2 + \sigma_{\text{CLUT}}^2)$ 

where  $\sigma_{CEL} = \frac{K}{N\sqrt{2}}$  variance attributed to resolution

where N is the factor provided by frame averaging  $\sigma_{\rm INST}$  = variance attributed to crosstalk and nonuniformity  $\sigma_{\text{CLUT}}$  = variance attributed to clutter.

The measurement matrix previously derived and the measurement errors defined above are all that is necessary for the implementation of the angular sensors in the Kalman-filtered update model.

Planetary Approach Phase Navigation. Analysis of orbiter-lander approach guidance systems is divided into two parts: the first part includes guidance from the point of entry into the planet's sphere of influence until orbit; the second part includes guidance from orbit to landing. Since a sizable package is left in orbit after the lander separates, navigation aids may be quite different in the two cases.

For approach, optical aids will usually be employed. It is assumed that the spacecraft employs a sun sensor and a star sensor, and, if desired, a sensor which can lock on the illuminated portion of the target planet. DSIF information is assumed available during this phase.

For the terminal phase, optical aids may or may not be employed. The lander may include a TV transmitter, with resultant monitoring from Earth. At planetary distances, however, the time lag excludes earth-based guidance. DSIF information may not be available due to the spacecraft's being shielded by the target planet, and due to the difficulty of locking onto the spacecraft when in close proximity to the surface.

The possible errors sources for each part and their impact on the mission success probability must now be identified.

For the approach phase, the end objective is to place a spacecraft into an orbit around the target planet. The desired periapsis and apoapsis for this orbit are fixed as  $r_p$  and  $r_a$  respectively. Figure 9 shows the approach path to achieve such an orbit.

Orbit insertion consists of altering the path at point S, so that the spacecraft enters on the path S-S' with firing of the retro engine occurring at point P for orbit injection. The line S-A, tangent to S-S' at S, is called the approach asymptote, and the line segment B, from the planet's center perpendicular to SA, is called the impact parameter.

Onboard measurements normal to the approach trajectory involve determining the angle between the vectors locating the planet center and the reference star with respect to the spacecraft. This is shown in Figure 9 as the angle  $\theta + \emptyset$ . The angle  $\theta$  is known from trajectory calculations, so the impact parameter angle  $\emptyset$  is readily calculated from this measurement. If a second star in another axis is viewed, then the rotation of the impact parameter about the planet center can also be determined.

Use of DSIF updating provides range and range-rate accuracies of sufficient magnitude (see Appendix A) that angular measurements of the planet with state-of-the-art electro-optical sensors does not improve the state estimate for the Mars mission on the trajectory used. Should significantly more accurate electro-optical sensors for planet approach measurements be available, they may be incorporated into the computer program without undue difficulty. In such a case, the improvement in accuracy should be weighed against the degradation in the penalty that would result from the weight, power, and reliability considerations of these sensors.

Additional errors will arise during retro-firing due to errors in the spacecraft's orientation when firing. These attitude errors will generate velocity and position errors. For a Martian orbiter, the retro  $\Delta V$  is about 3000 to 4500 ft/sec depending on the trajectory selected. The velocity errors generated during approach are much smaller in magnitude and correction requires very little additional fuel.

are:

The factors which may affect the penalty during approach navigation

- (1) Electro-optical sensor and radar MTBF, weight, and switching reliability degradation.
- (2) Electro-optical sensor and radar measurement uncertainties.
- (3) Position and attitude errors at the craft's entry into the planet's sphere of influence. (For the approach phase, these are essentially biases).
- (4) Retro-thrust magnitude and duration uncertainties.



FIGURE 9. ORBIT INSERTION (FROM REFERENCE 14)

(5) Velocity and position errors generated during thrusting by improper initial alignment of spacecraft. Although these are essentially second-order effects, the error magnitudes may be the largest contributors to the penalty.

For the terminal phase, it is assumed that the spacecraft is already in orbit around the planet. Errors that may exist in the orbit can be propagated via the state-transition matrix. Altitude errors may be assumed to have been previously corrected by a Hohmann transfer, or they can be propagated.

From this parking orbit, the lander is launched and descends to the surface of the planet. The moment of release of the lander may be computed on-board, or may be commanded from Earth. Obviously, the landing site will be directly dependent on the time of release. The DSIF can measure certain orbital parameters very accurately, so the time of release can be calculated equally accurately.

The lander enters the planet's atmosphere at an entry angle  $\beta$  as shown in Figure 10. For a direct lander, this path entry angle is very critical. Too steep an angle will cause the lander to burn up and too shallow an angle may cause it to skip back out. For entry from orbit, however, much shallower angles are permissible without danger of skip-out. Following atmospheric entry, the lander is slowed by aerodynamic drag and possible parachutes. The retro engines are then fired to effect a soft landing. To avoid surface contamination by the exhaust, the engines are cut at some height over the surface. If the horizontal velocity is not greater than some critical value, the lander will then drop to the surface intact without toppling over. In practice, horizontal velocities of less than 2 ft/sec are achievable with state-of-the-art guidance systems. In addition, attitude and attitude rate must also be controlled as excessive tilt of the craft is unacceptable. Inclination of the landing surface should also be taken into account.

The spacecraft is assumed to contain a pulsed radar (doppler) altimeter, and a CW radar. Other configurations are possible, but will not be discussed herein. The radar altimeter provides information for release of the parachute prior to final descent, and acts as a backup to the range beam down to some low altitude limited by pulse width constraints (Reference 15). The CW radar posesses four CW velocity beams and one FM/CW range beam, separated from each velocity beam by a "squint" angle  $\psi$ .

The ISU output can be used for derivation of steering commands.

Factors to be considered in the terminal guidance portion of the mission must include:

- (a) Inertial sensing unit errors
- (b) Retro engine thrust and firing time uncertainties
- (c) Altimeter (doppler shift) errors
- (d) CW Radar errors (Frequency drift, noise, measurement uncertainties)



FIGURE 10. LANDING GEOMETRY (FROM REFERENCE 14)

- (e) Optical aid errors
- (f) Flight path entry angle error (This is not an independent error but will be a function of initial condition errors and retro engine errors)
- (g) Failure probability due to various MTBF's.

Given more information on the terrain of the planet, the probability that the lander lands upright can also be calculated. This, for the present, is secondary.

The terminal phase is not included in the present form of the computer program due to lack of sufficient information on the lander vehicle.

# Alternate Attitude Control Mechanizations

Cold gas reaction jet attitude control was studied in detail in Reference 1. In this section control moment gyros and inertia wheels are considered.

<u>Control Moment Gyro Attitude Control</u>. An investigation has been conducted on the use of control moment gyros in the attitude control system in place of the previously examined gas-reaction jets (Reference 1). Although it is entirely possible to use both control moment gyros (CMG) and gas-reaction jets in the same attitude control system, the following analysis concentrates on the use of CMG alone.

Unlike the gas-reaction jet, which is a mass-expulsion device, the CMG is a mass-conservative system, working on the momentum-exchange principle. While the total momentum of the system is constant, quantities of it can be absorbed by the CMG by changing the orientation of the spin vectors of the gyros. In this way, the total momentum vector of the vehicle can change direction, resulting in a change of attitude for the spacecraft.

In addition, external torques acting on the spacecraft can be compensated by an appropriate change in the gyro orientation. In this case, the total momentum vector may or may not remain constant in magnitude, depending on the nature of the external disturbance. For impulsive disturbances (e.g., meteorite impact) the total momentum will, except in the most severe cases, remain constant while for continuous disturbances (e.g., solar pressure), the total momentum will eventually change.

Design Considerations. The CMG is basically a power gyro, and its main output is torque. Its basic design goals should be symmetry, stability, avoidance of severe temperature gradients, and design at stress levels well below the elastic limit. This last requirement is particulary important, since the control torques can be applied to the spacecraft only through the gimbals and bearings, and yet the gimbal inertias should be low (Reference 16).

Since the prime output of a power gyro is torque, the extreme position accuracy which is obtainable in instrument gyros cannot be achieved. Electromagnetic and electrostatic suspensions which are utilized in instrument gyros cannot be utilized here since they preclude the transmission of sizable torques between the gyro and the vehicle. Thus, the CMG uses conventional gimbals and bearings, with all the losses and inaccuracies associated with such a setup. The spin bearings are the prime factor in power consumption and life of the CMG, while the gimbal bearings determine the angular rate limits and sensitivity thresholds.

The rate threshold of a CMG limits the accuracy of control to about ten seconds of arc (Reference 16). If greater accuracies are desired, floated gyros must be employed, with their associated disadvantages of lower torque gain, more required power, and smaller momentum to weight ratios. In this analysis, conventional bearings will be assumed.

Equipment Associated with CMG. For the CMG to operate effectively in an attitude control system, it is necessary to have, in addition to the gyro itself, a sensor, or pickoff, to provide a signal proportional to the gimbal angle. In usual CMG applications, this signal is fed to a torque computer and a reset computer. In addition, there exists a need for a set of torquers, which will provide a given torque to the CMG given some command signal. Accurate servo motors are usually utilized as torquers.

The torque computer mentioned above assumes the existence of a multiple gyro system. It is common practice to utilize at least three single degree-of-freedom gyros, with their output axes aligned as close as possible to the body axes of the vehicle. In control operations, each gyro will exert a portion of control effort in any given direction. The function of the torque computer is the solution of the geometric problem of how to move the momentum vectors in space to insure that the torque exerted on the vehicle is about the correct axis and of the correct direction and magnitude.

The reset computer, also mentioned above, is basically a threshold measuring device. The computer determines the amount of momentum being provided along the three axes, and when it reaches some pre-established value, the computer calls for a measured reset pulse. Current practice is to combine the torque and reset computers into a single package.

For reliability analysis of a CMG system, reliability values for the computer package may be as significant as those for the gyros themselves.

Analysis of a CMG System. A complete CMG attitude-control system may have several different configurations. The simplest would comprise one singledegree-of-freedom gyro per principal vehicle axis. Moving up in complexity, one finds twin single-degree-of-freedom gyros per axis, single two-degree-of-freedom, twin two-degree-of-freedom, and, finally, configurations involving four gyros, single or twin, of one or two degrees of freedom. Analyses for several of these cases exist in the literature (References 16, 17, and 18). In this section, two cases will be examined; one single-degree-offreedom (SDF) gyro will be analyzed as an illustration of the type of procedures necessary, and then the four SDF gyro configuration will be analyzed. The reasons for these choices will be presented with the analysis.

One Single Degree-of-Freedom Gyro. The configuration of interest is shown in Figure 11. Note that the final results will depend on the choice of coordinate systems. Assume both coordinate systems possess a common origin at the center of mass of the spacecraft. The body axes are  $x_b$ ,  $y_b$ ,  $z_b$  and, the gimbal-centered coordinate system  $x_g$ ,  $y_g$ ,  $z_g$ , has  $z_g$  always coincident with the spin axes.  $\theta$  is the angle describing the gimbal rotation and is positive when rotated about  $x_g$  according to the right hand rule.



FIGURE 11. SINGLE-DEGREE-OF-FREEDOM CONTROL MOMENT GYRO

The components of the vectors are related by:

$$\begin{bmatrix} x_{g} \\ y_{g} \\ z_{g} \end{bmatrix} = C \begin{bmatrix} x_{b} \\ y_{b} \\ z_{b} \end{bmatrix}$$

where the transformation matrix C is

$$C = \begin{bmatrix} 1 & 0 & 0 \\ 0 & \sin \theta & -\cos \theta \\ 0 & \cos \theta & \sin \theta \end{bmatrix}$$

The total angular momentum about the origin is

$$\underline{\mathbf{H}}_{\mathrm{T}} = \underline{\mathbf{H}}_{\mathrm{b}} + \underline{\mathbf{H}}_{\mathrm{g}} \tag{11}$$

and

$$H_{b} = I(\underline{\omega}_{b} + \underline{\omega}_{r})$$
$$H_{g} = J(\underline{\omega}_{G} + \underline{\omega}_{g} + \underline{\omega}_{b} + \underline{\omega}_{r})$$

where

 $\underline{w}_{b}$  = angular velocity of body frame with respect to (w.r.t.) reference frame

$$\begin{split} & \underline{w}_{r} = \text{angular velocity of reference frame w.r.t inertial space} \\ & \underline{w}_{G} = \text{angular velocity of gyro frame w.r.t gimbal frame} \\ & \underline{w}_{g} = \text{angular velocity of gimbal frame w.r.t body frame} \\ & I = \text{inertia dyadic of vehicle} = |I_{x}| + |I_{y}| + |I_{z}| \\ & J = \text{inertia dyadic of rotor} = |A_{x}| + |A_{y}| + |G_{z}| \end{split}$$

where

$$A = A_{x} = A_{y} = \int_{0}^{m_{0}} m \left[ y^{2}(m) + z^{2}(m) \right] dm = \int_{0}^{m_{0}} m \left[ x^{2}(m) + z^{2}(m) \right] dm$$

$$G = G_{z} = \int_{0}^{m_{0}} m \left[ x^{2}(m) + y^{2}(m) \right] dm$$

= generic particle of mass

and

= mass of rotor

x(m), y(m), z(m) = coordinates of generic particle.

Equation (11) can be rewritten as

mo

m

$$\mathbf{H}_{\mathbf{T}} = (\mathbf{I} + \mathbf{J})(\underline{\omega}_{\mathbf{b}} + \underline{\omega}_{\mathbf{r}}) + \mathbf{J}(\underline{\omega}_{\mathbf{G}} + \underline{\omega}_{\mathbf{g}})$$

and since, in practice, I >>> J,

$$\underline{H}_{T} = I(\underline{\omega}_{b} + \underline{\omega}_{r}) + J(\underline{\omega}_{G} + \underline{\omega}_{g}) \qquad (12)$$

The equation of motion is derived by setting the torque acting on the vehicle equal to the derivative of Equation (12):

$$\frac{d}{dt} \left[ I(\underline{w}_{b} + \underline{w}_{r}) + J(\underline{w}_{G} + \underline{w}_{g}) \right] = \underline{T}$$
(13)

Equation (13) can be rewritten, after considerable manipulation, as

$$\begin{bmatrix} I_{x} \dot{p} + (q - w_{o}) r(I_{z} - I_{y}) \end{bmatrix} \underline{x}_{b} + \begin{bmatrix} I_{y} \dot{q} + pr(I_{x} - I_{z}) \end{bmatrix} \underline{y}_{b} + \\ \begin{bmatrix} I_{z} \dot{r} + p(q - w_{o}) (I_{y} - I_{x}) \end{bmatrix} \underline{z}_{b} + \begin{bmatrix} A\ddot{\theta} + G\Omega(q \sin \theta - r \cos \theta - w_{o} \sin \theta) \end{bmatrix} \underline{x}_{b} + \\ \begin{bmatrix} G\dot{\Omega} \cos \theta - G\Omega(\dot{\theta} + p) \sin \theta + A\dot{\theta} r \end{bmatrix} \underline{y}_{b} + \\ \begin{bmatrix} G\dot{\Omega} \sin \theta + G\Omega(\dot{\theta} + p) \cos \theta - A\dot{\theta}(q - w_{o}) \end{bmatrix} \underline{z}_{b} = \underline{T}$$
(14)

where  $x_b$ ,  $y_b$ , and  $z_b$  are unit vectors and p, q, r are the components of the angular rate  $\underline{\omega}_b$  and

 $\omega_{o}$  = magnitude of the angular rate with respect to inertial space

 $(\underline{\omega}_r = -\omega_o \underline{y}_b)$  .

Also

 $\Omega$  is defined by  $\underline{w}_{G} = \Omega \underline{z}_{g}$  and

 $\theta$  is defined by  $\underline{w}_{g} = \theta \underline{x}_{g}$ , for the coordinate system as shown in Figure 11.

A detailed derivation of Equation (14) is given in Reference 17.

Four Single-Degree-of-Freedom Gyros. A power gyro can exert torque only about its sensitive axis. If, during the course of a maneuver, a situation arises where all the gyros of a multiple-gyro system precess sufficiently so that all their output axes end up in the same plane, the system can no longer provide any control torque about axes parallel to this plane. This situation is known as "bindup", and for a control system employing one single-degree-of-freedom gyro per principal axis it has been shown (Reference 16) that bindup will occur when the momentum level reaches 0.4 H along any control axis (where all gyros are assumed identical, each with a momentum H). For a three-gyro system, the capacity of the system is  $\sqrt{6}$ H (Reference 16), but the bindup considerations dictate that the actual limiting momentum is 0.4 H. This waste of momentum becomes necessary in order to maintain control about all axes.

By employing a control system with four gyros, bindup conditions at momentum levels below the system's capability are avoided. In addition, a higher momentum capability is achieved, as well as added reliability due to redundancy.

The additional weight and power that the fourth gyro demands is included in the evaluation.

The equations of motion for the four-gyro system are complex, and the complete derivation will not be presented here. An excellent derivation can be found in Appendix 1 of Reference 16. Only the pertinent assumptions and results are discussed.

Consider a four-CMG configuration as shown in Figure 12, with the following description:

- (1) Gimbal axis "a" lies in the first quadrant of the y,z plane and is  $\alpha^{O}$  from the y axis
- (2) Gimbal axis "b" lies in the second quadrant of the y,z plane and is  $\alpha$  from the -y axis
- (3) Gimbal axis "c" lies in the fourth quadrant of the x,z plane and is  $\alpha^{\circ}$  from the x axis.
- (4) Gimbal axis "d" lies in the third quadrant of the x,z plane and is  $\alpha$  from the -x axis.

For this configuration, equal gains about the cardinal occur when  $\alpha$  = 54.7°.



FIGURE 12. FOUR-GYRO SYSTEM ORIENTATION

Beginning with the gimbal torque equations with respect to the a, b, c, and d axes, and making the following simplifying assumptions

- Reaction torques are small compared to the control torques and may be neglected
- (2) Gimbal inertia terms are considered as a part of the vehicle inertia
- (3) Gimbal acceleration terms are neglected
- (4) Vehicle accelerations are small relative to the gimbal motions,

the system equations can be written, after considerable manipulation, as

$$\begin{bmatrix} \mathbf{I}_{\mathbf{x}} \ \mathbf{S}^{2} \ \theta_{\mathbf{x}} \\ \mathbf{I}_{\mathbf{y}} \ \mathbf{S}^{2} \ \theta_{\mathbf{y}} \\ \mathbf{I}_{\mathbf{z}} \ \mathbf{S}^{2} \ \theta_{\mathbf{z}} \end{bmatrix} + \frac{\mathrm{H}}{\mathbf{I}_{\mathbf{G}}^{\mathbf{S} + \mathbf{D}}} \begin{bmatrix} \mathrm{B}_{11} & \mathrm{B}_{12} & \mathrm{B}_{13} \\ \mathrm{B}_{21} & \mathrm{B}_{22} & \mathrm{B}_{23} \\ \mathrm{B}_{31} & \mathrm{B}_{32} & \mathrm{B}_{33} \end{bmatrix} \begin{bmatrix} \mathrm{HS} \theta_{\mathbf{x}} + \mathrm{K} \theta_{\mathbf{x}} \\ \mathrm{HS} \theta_{\mathbf{y}} + \mathrm{K} \theta_{\mathbf{y}} \\ \mathrm{HS} \theta_{\mathbf{z}} + \mathrm{K} \theta_{\mathbf{z}} \end{bmatrix} = \begin{bmatrix} \mathrm{T}_{\mathbf{x}} \\ \mathrm{T}_{\mathbf{y}} \\ \mathrm{T}_{\mathbf{z}} \end{bmatrix} .$$
(15)

In the equation above,

 $θ_i = vehicle angle with respect to inertial space$ S = Laplace operator
H = gyro momentum
I<sub>G</sub> = gyro inertia about the gimbal axis (gimbal inertia)
D = gimbal damping
K = combined gain of sensor and torque-motor (lb-ft/rad)
T<sub>i</sub> = torques on vehicle axes.
B<sub>11</sub> = sin<sup>2</sup>a + sin<sup>2</sup>b + (sin α cos c)<sup>2</sup> + (sin α cos d)<sup>2</sup>
B<sub>12</sub> = -sin α (sin a cos a + sin b cos b + sin c cos c + sin d cos d)

$$B_{13} = -\cos \alpha (-\sin a \cos a + \sin b \cos b - \sin \alpha (\cos^{2}c - \cos^{2}d))$$

$$B_{21} = B_{12}$$

$$B_{22} = (\sin \alpha \cos a)^{2} + (\sin \alpha \cos b)^{2} + \sin^{2}c + \sin^{2}d$$

$$B_{23} = -\cos \alpha [\sin \alpha (\cos^{2}a - \cos^{2}b) + \sin c \cos c - \sin d \cos d]$$

$$B_{31} = B_{13}$$

$$B_{32} = B_{23}$$

$$B_{33} = (\cos \alpha)^{2}(\cos^{2}a + \cos^{2}b + \cos^{2}c + \cos^{2}d).$$

n

n

where a, b, c, and d are the gimbal displacement angles from the zero momentum configuration. The above terms depend entirely on the gyro angles, and these angles determine the transfer function between input and output. The off-diagonal terms represent the cross-coupling characteristics of the system. Note that when gimbal angles all equal zero the system is decoupled, and all off-diagonal terms vanish.

Weight and Power. For the CMG, power requirements are two-fold: power is needed at the drive motor to overcome bearing and windage losses, and power is needed at the torquers to move the gimbals during maneuvers.

The power to overcome losses must be supplied by the drive motor. The same motor is used to spin the gyro up to its operational speed. To overcome the need for large drive motors, long spin-up time is necessary.

The power required per torquer is

where

 $T_{M} = motor torque$ 

 $\dot{\theta}$  = angular rate desired.

This can be written:

$$P = T_{M} \theta = (I\theta + \omega H \cos \theta) \theta$$
(16)

where H = momentum about the spin axis

 $\omega = \text{spin velocity.}$ 

Equation (16) is derived by writing the equation of motion for a gimbal, assuming the output axis parallel to the vehicle x-axis:

$$T_{M} - D\theta - w_{X} H \cos \theta - w_{Y} H \sin \theta \cong I\theta$$
 (17)

Assuming small damping and no angular velocity normal to the output axis, (17) reduces to

$$T_{M} = I\theta + \omega H \cos \theta$$

which appears in Equation (16) (Reference 18).

For the four-gyro configuration, in the uncoupled states and assuming identical gyros, the power limit is four times that of Equation (16). In actual practice, required power is less, since one gyro acts on more than one axis at a time.

Figure 13 shows a power versus momentum curve for a four-gyro controller. Only spin power is shown, so the plot should be interpreted as continuous power required.

Figure 14 shows a weight versus momentum curve for the four-gyro controller. To this should be added the weight of the computers and electronics associated with the system. Of course, graphs such as these are constructed on the basis of common-practice materials, bearings, etc. Specialized applications may require individual calculation.

Momentum Requirements. To determine the momentum requirements, consider  $H_1$  to be the minimum angular momentum necessary for performing the prescribed maneuvers, and  $H_2$  to be the minimum angular momentum necessary to overcome the worst case of meteorite impact, solar pressure, or midcourse engine misalignment.

Given a maneuver which requires traversing a given angle  $\theta,$  in a specified time  $\Delta t,$  the torque necessary is

$$f = \frac{4\theta I}{\Delta t^2}$$





Continuous Power Required (Watts)





# System Momentum Capability (ft-lb-sec)

where I = moment of inertia about the axis of rotation.

Now, consider that this torque will be supplied by a gyroscope, capable of a velocity of precession  $\omega$ . For a successful operation

$$\omega_{\rm p} \ge \frac{\Theta}{\Delta t} \tag{19}$$

otherwise the maneuver cannot be performed. Assuming that relation (19) holds true, from simple gyro theory

 $T = H_1 \omega_p$ 

 $H_1 = \frac{4\theta I}{\Delta t^2 \omega}$ 

or

Now, given the  $\omega$  mentioned above, the other torquing requirements are examined.

From meteorite impact, solar pressure, and midcourse engine misalignment, the worst case is picked, and this torque is called T  $_{\rm max}$  . From this

 $T_{max} = H_2 \omega_p$ 

The momentum required, H3, for the particular mission, is then

 $H_3 = Max (H_1, H_2)$ 

H<sub>gyro</sub> 
$$\geq \frac{1}{3}$$
 H<sub>1</sub>

$$H_2 = \frac{T_{max}}{\omega_p} .$$
 (21)

(20)

or

This is the maximum momentum <u>per axis</u>. Since, for a four-gyro system, the total momentum available per axis is three times the momentum of an individual gyro, the final sizing can now take place.

$$H_{final} = \frac{1}{3} H_3$$
 per gyro

Reliability Considerations. The electrical parts reliability, minus the computer, for a four-gyro attitude control system, was calculated in 1965 (Reference 16) as  $9 \ge 10^{-6}$  failures/hour, per gyro. The governing figure in reliability analysis in a gyro package is the gyro hardware. Table VIII gives a detailed breakdown, circa 1965, and a figure of 59.26  $\ge 10^{-6}$  failure/hour per gyro is arrived at.

Today, advances in electronic components and packaging may be assumed to cut the failure rate of the electronics to, roughly,  $3 \times 10^{-6}$  failures/hour. Likewise, gyro hardware failure rates can be assumed (References 16) to have decreased to about 40  $\times 10^{-6}$  failures/hour. This yields a reliability figure of 43  $\times 10^{-6}$  failures/hour per gyro channel, and this figure will be used in the calculations.

Since a four-gyro system is employed, it is also assumed that if three of the four gyros survive the mission, the system performance, though degraded, will still be considered satisfactory. Under this assumption, the probability that at least three of four channels will survive is given by:

$$PS = P(s,n) = \sum_{x=s}^{x=n} \frac{n!}{x! (n-x)!} p^{x} (1-p)^{n-x}$$
(22)

where s = number of survisors (=3)

n = total number of elements (=4)

p = probability of survival of a single element.

The probability of failure for this case is

$$PF = 1 - PS$$
.

If an actual numerical calculation is attempted, it immediately becomes apparent that the attitude control system cannot be left on continuously for a long mission. In the case of a Jupiter flyby, leaving the system on for 400 days will result in a reliability of about 30%, clearly an unacceptable figure.

# TABLE VIII.FAILURE RATES FOR GYRO PARTS<br/>(From Reference 16)

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	Part	Failure/Hour
1	Potor	$5.00 \times 10^{-6}$
2	End hall bearings (2)	7 00 10
2. 2	Stator (including windings)	5.00
4.	1 piece (spin) shaft	
5.	Gimbal	
6.	Balance weights (adjustable)	
7.	Gimbal bearings (2) at 0.875	1.75
8.	Torquer coil assembly	0.30
9.	Torquer coil assembly frame	
10.	Flex lead cap assembly	
11.	Flex leads	4.00
12.	Limit switch assembly (mag. read)	2.00
13.	Magnet return path	
14.	Outer casing	
15.	Gimbal bearing support - sensor end	
16.	Gimbal bearing support - torquer end	
17.	Permanent magnet	
18.	Reduction gear	2.0
19.	Slip ring bearing support	
20.	Slip ring bearings (2)	3.0
21.	Gimbal lock solenoid	2.0
22.	Servo motor	5.20
23.	Electrical header	2.0
24.	End caps (2)	
25.	Pickoff - sensor stator	1.0
26.	Pickoff - sensor rotor	1.0
27.	Assorted nuts	2.0
28.	Slip rings	10.0
29.	Slip ring brushes	6.0
	TOTAL	$59.25 \times 10^{-6}$

Note: From G.E. Memo 62-45-501

The system must, therefore, be turned off when it is not needed. This will create tradeoff conditions due to reliability degradation by switching. This tradeoff is examined elsewhere in this report.

It must also be kept in mind that at least two hours of warmup time will be necessary after the system is turned on and before it is operated in order to bring the previously stationary gyros up to operating speed (in the case examined here, 24,000 RPM). This will have an effect on the total reliability figure, as well as the total energy required. Assuming that no more power is required to spin the gyro up than to keep it spinning, no power increase will be necessary.

Implementation. In implementing the above in the computer program, the system sizing is done in much the same way as for the gas reaction jets (Reference 1). There are two main torque requirements: first, torque required to perform the maneuvers called for in the schedule; and second, torque required to overcome disturbances due to solar pressure, meteorite impact, and midcourse engine misalignment. Each of the two requirements is sized independently and a value of angular momentum required is obtained for each. The greater value is then retained as the minimum momentum required for successful completion of the schedule.

Since the torque generated by the CMG system is directly proportional to the angular velocity of precession of the gyro, care must be exercised so that the angular velocity picked is not too great. If the spacecraft is allowed to rotate exceedingly fast, there exists the danger of the sensors being unable to acquire a star when such an acquisition is called for.

To combat the arbitrary selection of overly large values for the angular velocity of precession, a new subroutine has been written and has been incorporated into the system. This new subroutine, STDET, checks whether, for a particular angular velocity of the spacecraft, the probability of detecting a star is sufficiently high as to warrant the use of that particular value of angular velocity. The subroutine allows the use of either rectangular line-scanned detectors, or circular, circularly scanned detectors. The equations for the probability of detection while employing either of the two configurations have already been presented in Reference 1. The routine also requires as input a minimum probability of detection which would be acceptable to the user. In the examples cited in this report, this probability is set at 0.99. With the use of STDET, the attitude control subroutine is allowed to employ reasonably high velocities of precession for the CMG gyros without jeopardizing the process of star detection and acquisition.

Ten exercise runs were made using the new CMG attitude control subroutine on a Jupiter flyby mission and the results are summarized in Table IX. Detailed program output is shown in a later section of this report. It is evident that the dimensions and scanning frequency of the star tracker are of great importance in sizing the system. The values shown are arbitrary and there exists

Remarks	est.		Star Detection Probability Constraint			Best	·	Star Detection Probability Constraint		
Penalty (Mode 3) (1bs)	347.844	350.978	372.604	359.479	368.426	347.835	350.970	372.603	359.470	368. 424
Actual Angular Velocity (rad/sec)	0.8	0.6	0.6	0.35	6.0	0.8	0.6	0.6	0.35	6.0
Maximum Angular Velocity Allowed (rad/sec)	0.8	0.6	0.6	0.9	6.0	0.8	0.6	0.6	6.0	6.0
Scan Frequency ( cps)	104	104	$0.5 \times 10^{4}$	10 <sup>3</sup>	$0.5 \times 10^4$	104	10 <sup>4</sup>	$0.5 \times 10^{4}$	10 <sup>3</sup>	0.5 x 10 <sup>4</sup>
Dimensions	3" x 4"	3" x 4"	2" x 5"	2" x 4"	2" x 5"	3" x 4"	3" x 4"	2" x 5"	2" x 4"	2" x 5"
Star Tracker	ITT-LUN. OB.	ITT-LUN. OB.	GIMB. ST.	ITT-LUN. OB.	GIMB. ST.	IIT-LUN. OB.	ITT-LUN. OB.	GIMB. ST.	ITT-LUN. OB.	GIMB. ST.
Schedule	1	1	Ч	M	Ę	5	7	7	2	5
Figure No.	19	20	21	22	23	24	25	26	27	28

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TABLE IX. SUPMARY OF CMG-ATTITUDE-CONTROL SIZING RESULTS ON JUPITER FLYBY MISSION

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a very definite need for accurate data. In the cases shown, it is evident that the CMG attitude control system will weigh slightly more than the gas reaction jet system (Reference 1); however, the difference is not so great as to be prohibitive. In the cases marked as "best" in Table IX, the total penalty is only about 4 pounds more than in the gas reaction jet case. When the dimensions and scanning frequency of the star tracker are such that the star detection probability constraint comes into play, the total penalty becomes approximately 16 pounds more than the previous value. Although sizable, such a difference still cannot be considered prohibitive.

<u>Reaction Wheel Attitude Control</u>. The reaction wheel system is a mass conservative system exerting control on the spacecraft by changing the spacecraft's momentum vector. Whereas a control moment gyro system changes the momentum vectors by tilting the spin axes of the gyros, a reaction wheel system has its rotating masses permanently aligned with respect to the spacecraft. Since the momentum vector cannot, in this way, be changed by moving the individual gyro momentum vectors, the change is effected by altering the magnitudes.

Thus, the control moment gyro system is basically momentum conservative, whereas the reaction wheel system is not. There are both advantages and disadvantages to this latter type of system when it is compared to the control moment gyro. Perhaps the main advantage is a lack of drift, an important source of error in a CMG system. In addition, there are fewer bearings, since there are no gimbals, which increases the reliability.

On the other hand, the reaction wheel requires a large motor in order for commanded momentum changes to occur within reasonable times. This places a more severe requirement on power and energy, even considering the possibility of employing regenerative braking which is a common practice in such a system.

Some reaction wheel attitude control systems employ two sets of reaction wheels, one for coarse and one for fine control. This may improve the power requirements, but it adversely affects the reliability. Only a single set of reaction wheels is considered in this analysis.

Single Reaction Wheel Analysis. The motor in a reaction wheel attitude control system exerts a control torque on the vehicle and an equal and opposite torque on the wheel. The torque axis is fixed with respect to the vehicle. Therefore, for complete attitude control, three such wheels with mutually perpendicular spin axes are needed. Normally, the spin axis of each wheel would be parallel to a principal axis of the vehicle to reduce the coupling, evident in Equation (23) (Euler's equation for the motion of a rigid body).

$$\underline{\mathbf{T}} = \begin{bmatrix} \mathbf{I}_{\mathbf{X}\mathbf{X}} & \hat{\mathbf{w}}_{\mathbf{X}} + (\mathbf{I}_{\mathbf{Z}\mathbf{Z}} - \mathbf{I}_{\mathbf{y}\mathbf{y}}) & \mathbf{w}_{\mathbf{y}} & \mathbf{w}_{\mathbf{z}} \end{bmatrix} \underline{\mathbf{x}} \\ + \begin{bmatrix} \mathbf{I}_{\mathbf{y}\mathbf{y}} & \hat{\mathbf{w}}_{\mathbf{y}} + (\mathbf{I}_{\mathbf{x}\mathbf{X}} - \mathbf{I}_{\mathbf{z}\mathbf{z}}) & \mathbf{w}_{\mathbf{x}} & \mathbf{w}_{\mathbf{z}} \end{bmatrix} \underline{\mathbf{y}} \\ + \begin{bmatrix} \mathbf{I}_{\mathbf{z}\mathbf{z}} & \hat{\mathbf{w}}_{\mathbf{z}} + (\mathbf{I}_{\mathbf{y}\mathbf{y}} - \mathbf{I}_{\mathbf{x}\mathbf{x}}) & \mathbf{w}_{\mathbf{x}} & \mathbf{w}_{\mathbf{y}} \end{bmatrix} \underline{\mathbf{z}} \end{bmatrix}$$

where  $\underline{T}$  = sum of control and disturbance torques  $I_{ii}$  = mass moment of inertia of vehicle along principal axes  $\omega_i$  = vehicle angular velocity  $\underline{x}, \underline{y}, \underline{z}$  = unit vectors.

For a single reaction wheel, with spin axis parallel to the vehicle X axis, the equation becomes-

$$T_{\text{control}} = T_{D} + I_{xx} \overset{\circ}{w}_{x} + (I_{zz} - I_{yy}) \overset{\omega}{v}_{y} \overset{\omega}{z}$$
(24)

where  $T_D$  is disturbance torque and the assumption is made that the moments of inertia of the vehicle approximate those of the system (Reference 18).

For attitude changes in interplanetary space, the control torque required for a reasonable response time is normally much greater than the disturbance torque, a possible exception being the disturbance torque resulting from misalignment of the main engine thrust vector. Under these conditions, Equation (24) becomes, for the case of commanded reorientation

$$T_{\text{control}} = I_{\text{xx}} \overset{o}{\overset{}}_{x} + \left(I_{zz} - I_{yy}\right) \overset{o}{\overset{}}_{y} \overset{\omega}{\overset{}}_{z} \qquad (25)$$

Power required (less motor, bearing, and windage losses) is (Reference 18)

 $P = T_{control} \Omega = I_{R} \left( \begin{array}{c} 0 & - \begin{array}{c} 0 \\ \Omega & - \end{array} \right) \Omega$ (26)

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(23)
where  $\Omega$  = angular velocity of the wheel relative to the vehicle

 $I_p$  = mass moment of inertia of wheel about its spin axis.

Optimization of any mass conservative attitude control system consists of arriving at the proper blend of the following objectives while meeting cost and reliability specifications:

- (1) Maximum control torque
- (2) Maximum angular impulse capacity
- (3) Minimum power and energy
- (4) Minimum weight
- (5) Minimum space requirements.

With respect to a reaction wheel, the following points apply: (a) objective (4) and the need for large wheel moment of inertia dictate a wheel with its mass concentrated in the rim; and (b) objectives (3) and (4) are in direct conflict. For a given control torque decreasing  $I_R$  necessitates a higher wheel angular acceleration with resulting increase in power and energy (Reference 18).

Three Axis Reaction Wheel Control. In order to determine the effect on inertia wheel control of such phenomena as gyroscopic cross coupling and vehicle control to a rotating reference, the three axis equations of motion were derived. The exact derivation is given in Reference 16. The equations presented below are subject to the following constraints: first, the inertia wheel spin axes lie along the principal axes of the vehicle; second, the reference coordinates are either inertially fixed or represent vehicle orientation to the local vertical; third, vehicle attitude deviation from the reference axis is small; fourth, wheel inertia is much less than the principal inertias of the vehicle; fifth, the motion of the vehicle about its center of mass has negligible effect on the motion of the center of mass; and sixth, the products of the Euler angles and their rates are negligible.

Under these assumptions, consider the vehicle and reference coordinates to be those shown in Figure 15

where X, Y, Z = principal axes of vehicle (spin axes of wheels are along these axes)

U, V, W = reference axes

 $\theta_x$ ,  $\theta_y$ ,  $\theta_z$  = angles defining the orientation of the vehicle with respect to reference axes.



FIGURE 15. REACTION WHEEL COORDINATE SYSTEMS

The generalized equation of motion (rigid body) is

$$\underline{\mathbf{T}} = \underline{\mathbf{\dot{H}}}_{\mathrm{S}} + \underline{\boldsymbol{\omega}}_{\mathrm{V}} \times \underline{\mathbf{H}}_{\mathrm{S}}$$
(27)

where  $\underline{T}$  = external torque vector (vehicle coordinates)

 $\underline{H}_{c}$  = system momentum vector (vehicle coordinates)

 $\underline{w}_V$  = angular velocity of vehicle coordinate system with respect to inertial space.

This equation may be written in matrix form, under the additional assumption that the vehicle is being controlled to an inertial reference (i.e., the angular velocity of the reference frame with respect to inertial space is zero) as

$$\begin{bmatrix} I_{xx} S^{2} & I_{w} \omega S & -I_{w} \omega S \\ -I_{w} \omega S & I_{yy} S^{2} & I_{w} \omega S \\ I_{w} \omega S & -I_{w} \omega S & I_{zz} S^{2} \end{bmatrix} \begin{bmatrix} \theta_{x} \\ \theta_{y} \\ \theta_{z} \end{bmatrix} + I_{w} S \begin{bmatrix} \omega_{x} \\ \omega_{y} \\ \omega_{z} \end{bmatrix} = \begin{bmatrix} T_{Dx} \\ T_{Dy} \\ T_{Dz} \end{bmatrix}$$
(28)

where I = Reaction wheel moment of inertia
I = Principal moments of inertia of the vehicle, i = x,y,z

w = Angular velocities of inertia wheels with respect to
 inertial plane
T = Components of the external disturbance torque vector
 S = d/dt

Assuming that the initial wheel angular velocity is low and does not change appreciably during vehicle motions, then the cross product terms in the position coefficient matrix are near zero, and vehicle motion as described in the single axis case is valid for three axis control. If the above assumption does not hold, which is usually the case, wheel speed build up due to disturbing torques will create a cross coupling torque if there is a vehicle angular rate. The effect of cross coupling is that the vehicle rate in one axis will introduce torques about the other two axes, the result being that inertia wheels in all three axes will be more active than in the single axis case and will respond to inputs about any axis. Such cross coupling conditions are highly deterimental to the efficiency of the total system. Unlike the case of the control moment gyro attitude control system however, cross coupling effects in a reaction wheel system do not seem to affect the stability of that system (Reference 17). Implementation of the above theory into the existing computation scheme involved the following assumptions:

- The inertia wheels are located on the principal axes of the vehicle.
- (2) Only one set of wheels is used (there is no separate fine and coarse control).
- (3) The drive motors are A-C servomotors. These motors are much superior in their speed-torque characteristics to the D-C shunt motors at the momentum levels used in the Jupiter flyby mission (H ≤ 10 lb-ft-sec).

Since momentum changes are effected by speeding up or slowing down the motors, rather than by reorienting the momentum vectors, it is often necessary to allow longer maneuver times, so that the size of the motors does not become excessive. Although it was not necessary to alter the schedule in the Jupiter flyby mission, it may be necessary to do so in other cases.

The weight of the IWAC was computed from the curves given in Reference 16. It is assumed that the nominal spin rate of the wheels is 6000 RPM, and that maximum torque is not required from the motors more than 5 percent of the time.

In computing the power required, an averaging technique was employed, where peak power was utilized 5 percent of the time, and, during the remaining

95 percent of the time, power to overcome friction losses was used. Under the additional stipulation that the motor stall torque required for maneuvering was 40 percent of the rated torque, and the stall torque required to overcome disturbances was 20 percent of the rated torque (since longer times are then available), the power requirements are (Reference 16):

P<sub>1 (watts/axis)</sub> = (0.0118) x (Stall Torque) x (Rated RPM)

By employing regenerative braking, the power requirement for all three axes is reduced to

$$P_{TOT} = 1.18 \cdot P_{1}$$

Finally, reliability must be examined. The number of failures per hour for a single-axis IWAC is 19.6 x  $10^{-6}$  (MTBF = 51,000 hours), circa 1965 (Reference 16). Since there is no redundancy employed in the system, the probability of failure is simply

$$PF = 1.0 - \left(1.0 - \frac{\text{Total Hours}}{\text{MTBF}}\right)^3$$

Implementation of the above calculations in the computer program has been accomplished. Several exercise runs have been made on a Jupiter flyby mission, and the results are presented following the Communications Requirements section later in this report.

## Flight Control Requirements

In the discussion in this section, the flight control system is assumed to include a number of state sensors, a central computer, and the actuators necessary to carry out the computer's commands.

In investigating the impact of flight control requirements on astrionics effectiveness, it was found that a detailed analysis in general terms is almost impossible. There are simply too many components directly dependent on the specific vehicle's structural and dynamic characteristics.

In examining the impact of flight control requirements on astrionics effectiveness two approaches are available:

- (a) The flight control components, their functions, and their location on the launch vehicle are assumed known and a method is sought to calculate their impact on mission effectiveness.
- (b) The mission launch information and launch vehicle are assumed known and a method is sought to determine the

components necessary for flight control, their function, location on the launch vehicle, and impact on the mission effectiveness.

The first method must include calculation of the additional memory and speed requirements imposed on the on-board computer in order to monitor and direct the functions of the flight control components, and calculation of the effect that the power, weight, and reliability of the flight control system will have on the penalty.

The second method must include all calculations of the first method plus design calculations considering vehicle bending, variable mass and moments of inertia of the launch vehicle, propellant sloshing on vehicles using liquid propulsion, aerodynamic instability, and influence of wind. Variable vehicle loading due to varying payload shapes and weights must also be considered.

An analysis incorporating the second method would require an effort in excess of one man-year and as such was considered outside the scope of this contract. The analytical task is extremely complex and, in practice, is usually attempted only for a particular mission or launch vehicle with no general solutions being developed. The first method, although more limited in terms of scope of application can still be quite useful and is adaptable to computer solution. Implementation of the first method would be applicable principally in the examination of tradeoffs between several candidate flight control systems. Since this agrees with the purpose of the general effort under this contract, it was decided to implement this method and include it in the computer program.

The flight control system contains the following components:

- (1) Computer usually shared with a guidance system.
- (2) Gyros rate and position.
- (3) Accelerometers lateral.
- (4) Angle of attack sensor.
- (5) Passive filters
- (6) Electromechanical or hydraulic actuators
- (7) Wiring

One or more of these components may be missing or replaced by multi-function components for any given system.

For penalty function considerations, power will be required by the computer, gyros, accelerometer, angle of attack sensor, and actuators. Weight will be contributed by all except the actuators which are considered to be included in the total weight of the vehicle. If, however, any of the components are located outside the spacecraft, their weight should not enter into the penalty calculation. Reliability figures must be supplied for all except the passive

filters and wiring. These are considered redundant and have a probability of failure so much smaller than the remainder of the components that they can be ignored. Although, in practice, it may not be generally true, failure of any of the above components is assumed to render the entire system useless, so each component carries the same weight in reliability calculations.

In addition to the effects of weight, power, and reliability data necessary for each of the components, the penalty function may also be affected by utilizing a computer of inadequate size or speed. The necessary size (i.e., memory word capacity) and speed are functions of the trajectory flown, the launch vehicle characteristics, the mechanization of the equations to be solved by the computer, and the number of flight control components utilized. Since the same computer can be utilized for navigation and guidance as well as flight control, the requirements for both must be considered. In general, the operations to be performed by the on-board computer will include: (Reference 20)

- (a) Prelaunch checkout and initialization
- (b) Computation of direction cosines
- (c) Coordinate transformation and navigation computations
- (d) Euler angle and rate computations
- (e) Processing of discretes from the ground tracking network
- (f) Output telemetry discretes
- (g) Solution of guidance steering laws
- (h) Solution of thrust vector and reaction jet control laws.

Additional requirements, such as on-board experiment monitoring may be required for a particular mission. Each of the operations above would affect computer memory and timing requirements. These requirements will determine the memory capacity and operations-per-second capability that the on-board computer must possess.

<u>Computer Sizing Considerations</u>. For the purpose of analysis of the computer sizing implications, it is assumed that the principal guidance function computations which must be performed by the computer include:

> (1) Navigation - defined as determination of the vehicle state in an appropriate coordinate system at any moment of time based upon sensor inputs and the appropriate navigation equations.

- (2) Guidance defined as the solution of a selected set of equations expressing the relation between the present state and the desired state to derive the error signal required for control (steering law).
- (3) Control (steering) computation of the steering signals which are used in control of the direction of the vehicle flight path through commanded changes in the vehicle's attitude so that the guidance function is satisfied. Control also involves stabilization and usually requires filtering and compensation of sensor inputs.

Additional computations, which are related to the guidance function but are not directly implied if only flight control is considered, were listed in the preceeding section. These requirements are not directly related to the flight control aspect of the problem but do affect the selection of the onboard computer.

The equations which the onboard computer must solve should be specified, or else the number of operations which the computer must perform each computational cycle should be specified, along with the number of words of memory required. The number of operations the computer must perform each computation cycle, as well as the number of words of memory required, is a function of the number of propulsion stages.

For example, if a specific set of guidance equations such as the explicit linear tangent guidance equations (Reference 21) were assumed, an estimate of the total number of operations which the computer must perform each computational cycle could be made. Table X summarizes the estimate for the explicit linear tangent guidance equations. Note that the number of operations is dependent upon the number of propulsion stages.

Operation	Symbol	Word Storage	Maximum Use <sup>*</sup>	Minimum Use
Sine, Cosine	Sin	5	4	3
Square root	$\sqrt{-}$	16	39	11
Dot Product	D.P.	21	20	18
Cross Product	С.Р.	8	8	7
Natural Logarithm	LOG .	2	18	6
Arc Sine or Cosine	Sin <sup>-1</sup>	1	1	0
Multiply or Divide	Х	245	444	265
Plus or Minus	+	174	365	195
Branches	an an an Anna a	33	69	35

TABLE X. TOTAL OPERATION COUNT FOR EXPLICIT LINEAR TANGENT GUIDANCE EQUATIONS (Reference 21)

\* Assuming three guided stages remaining. For more or less than three, add or substract the following for each stage: 90+, 9 and 6 Log. For each candidate computer, it will be necessary to specify the computational frequency, the time associated with each of the arithmetical operations listed in Table X, and the number of words in memory.

It must be emphasized that the adequacy of the candidate computers will be dependent upon the validity of the information describing the equations and their solution requirements. For example, alternate sets of boost guidance equations such as those discussed in Reference 22 could be considered. Mechanization of each set of equations places different requirements on the onboard computer. Since the guidance equations are only part of the guidance loop, consideration should be given to the other computational requirements and their compatibility with the candidate guidance equations before selecting a specific set and a computer which can satisfy all computational requirements.

The present work assumes the Kalman filter used in the state estimation is implemented but does not consider where this implementation is performed, i.e., in a ground-based or onboard computer. A complete study of the implications of the navigation equations including a Kalman filter approximation is beyond the scope of the present contract.

<u>Implementation</u>. In implementing the model for flight control appearing earlier in this report a new subroutine has been generated.

Subroutine FLCODE determines the contribution of the flight control system to the penalty. A system vector of seven components must be provided. The components are the computer, three rate gyros, one position gyro, one lateral accelerometer, and one angle of attack sensor. One or more of the above components may be missing for any particular case.

In the case of extremely flexible launch vehicles, such as the Atlas/ Centaur where more than one set of rate gyros is carried, the subroutine may be called repeatedly, after separation of each stage. It is thus possible to analyze a changing flight control system, as would be the case of the Atlas/Centaur.

The computer, specified as the first component of the flight control system vector, is checked against the prespecified core and speed requirements. Depending on the result a check-constant is set to zero, if everything is acceptable, to one if insufficient core is available, and to two if the speed is not adequate. The system optimization routine (SYSOPT) is utilized, if the checkconstant is not zero, to select an acceptable computer.

Components 1 through 6 of the system vector are integers referring to the number in the appropriate data bank, which is already built into the existing program. Component 7 is an integer referring to the component number in a data bank built into subroutine FLCODE. Also built in as data are such constants as wiring and filter weight, and actuator power requirements and reliability.

Subroutine FLCODE calculation are not included in the results shown in this report. Inclusion of FLCODE can be easily accomplished when sufficient data is available.

## Effects of Subsystem Switching

The switching on and off of astrionics subsystems can increase the number of failures over those expected from the total operating time. This effect might be represented by estimating the probability of successful operation after time t and n cycles for a single subsystem as

$$P_{\rm S} = e^{-t/T} e^{-n/N} \tag{29}$$

where T is the mean time between failures and N is the mean number of cycles between failures. The probability of failure due to lack of reliability and switching is given by

$$P_{\rm F} = 1 - P_{\rm S} = 1 - e^{-(t/T + n/N)}$$
 (30)

For multiple subsystems

$$P_{F} = 1 - \frac{\pi}{\pi} e^{-(t_{i}/T_{i} + n_{i}/N_{i})}$$
(31)

where the index i indicates each subsystem.

In addition to the reliability effects, operational problems must be considered. For example, restarting the ISU necessitates restarting the gyroscopes, and realigning the gimbals in gimballed systems or resetting the direction cosines in strapdown systems. Restarting the computer requires reinitializing the memory or assuring that the memory is not destroyed when power is turned off and on.

As cited in Reference 23, a correlation is thought to exist between the number of times that systems are cycled on and off during individual test periods and the number of failures observed. The problem is that the data acquired are often not originated with this sort of experiment in mind, and the lack of any concrete result does not necessarily prove that the turn-on stress is negligible. The majority of tests are characterized by rather frequent on-off cycles and fairly short continuous periods of operation. The estimates of the operating failure rates provided by manufacturers reflects to some extent any on-off cycle stress factor which might exist (Reference 23).

The capability has been provided in the computer programs to evaluate the effects of switching subsystems on and off. This capability can be used to evaluate feasible switching mechanizations, but experimental verification should follow the evaluation with the objectives of: (1) validating the data used and (2) possible development of analytical techniques for estimating system or component failure as a function of the number of on-off switching cycles.

A new counter has been provided to keep track of the number of switchings for each subystem as the schedule is executed. The only other necessary information is the mean number of cycles to failure for each subsystem.

Very little information is available as to the mean number of cylces to failure. In exercising the program, the mean number of cycles to failure has been assumed to be 500 for all components. Better data is needed for this item.

The probability of failure due to switching has been programmed as a simple exponential. A Weibull distribution could be used to include the effect of increased probability of failure per cycle as the system accumulates more cycling history. Such a model would require an additional parameter, the Weibull constant for switching, as was discussed for time dependent failures in Reference 4. If a Weibull distribution is adopted, it must be kept in mind that the data needed refer to the status of the components at the start of the mission. That is, the subsystem have already been degraded by switching during testing, and, therefore, the accumulated number of previous on-off cycles should be specified as well as the mean cycles to failure and Weibull constant.

Example runs employing the subsystem switching effects on the system reliability are shown following the Communications Requirements section of this report.

## Communications Requirements

According to Reference 24, the effective radiated power (ERP) required to maintain a given information rate H in bits/sec is given by

$$ERP = \left[\frac{4\pi R^2}{A}\right] [KT] \left[\frac{E}{N_0}\right] H$$

where R = the range

A = the effective area of the Earth based antenna

K = Boltzmann's constant

T = the system noise temperature

 $\frac{E}{N}$  = the ratio of energy-per-bit to noise spectral density o required to meet an acceptable error probability.

The ERP is the product of the power output of the transmitter P and the gain of the antenna G, i.e.,

ERP = PG

Thus, given either P or G, the unknown parameter is uniquely determined by the ERP required to transmit information at a given rate.

The communications subsystem weight is then estimated from P and G. The transmitter weight,  $W_{\rm P}$ , required to achieve a given radiated power P is given, as shown in Reference 24, by

$$W_{p} = W_{1} + \omega_{p}P$$

where  $W_1$  is a fixed weight, and  $w_p$  (pounds per watt) is the incremental weight of the transmitter associated with an increase in power output.

To reflect completely the increase in system weight due to transmitter output power, energy source weight change must be included. Thus an overall weight coefficient,  $\omega'_p$ , can be found from

$$w_{p}^{\prime} = w_{p} + (K_{p} + K_{e}T_{o})/\eta$$

where  $K_p$  = the energy source weight per unit power

K = the energy source weight per unit energy

 $\eta$  = the transmitter efficiency, and

 $T_{op}$  = the transmitter operating time.

The weight of the onboard antenna is assumed (Reference 24) to be given by

$$W_{a} = W_{2}G^{0.6}$$

where  $W_2$  is a fixed constant. The total weight of the onboard communications subsystem is

$$W = W_{p} + W_{a} = W_{1} + \omega'_{p} P + W_{2} G^{0.6}$$

If neither P or G is fixed, then both the onboard antenna and the transmitter can be designed to satisfy a given constraint on the effective radiated power and yield a minimum total subsystem weight. In this case, for fixed ERP = PG, the optimum choice of P and G to minimize W is



The total weight of this optimum communications subsystem is given by

$$W_{\min} = W_1 + 1.9 W_2^{5/8} \omega_p^{\prime} {}^{3/8} (ERP)^{3/8}$$

The accuracy required in the pointing of the spacecraft antenna is related to the antenna beamwidth. For a high-gain paraboloidal antenna, the beamwidth between half-power points is approximated by

$$\theta^2 = \approx 2.7 \times 10^4 / \text{G}$$

where  $\theta$  is the beamwidth in degrees and G is the antenna gain in absolute units. In most cases it is reasonable to require the pointing accuracy to be 1/10 of the onboard antenna beamwidth.

To establish the onboard antenna pointing direction, knowledge of the angle subtended by vectors from the spacecraft to the Sun and Earth is necessary (see Figure 16).

This information is easily obtained from a time history of the spacecraft and sun positions with respect to the Earth. Table XI lists this angle (EARTH-SUN ANG) at discrete points in the Jupiter flyby mission. Also included in Table XI is the range of the spacecraft as observed from the Earth and the angle subtended by vectors from the Earth to the spacecraft and Sun (SC-SUN ANG). The maximum range encountered during the mission is recorded and used to determine the antenna gain and transmitter power required to transmit information at a given rate.

Communications problems can be anticipated when the SC-SUN ANG passes through zero. This occurs approximately 315 days into the Jupiter flyby mission and does not interfer with critical operations in the schedules used. When crossing the orbit of Mars on the Jupiter mission, calculations show the spacecraft is about 2,000 Mars diameters out of the Mars orbital plane. Thus, the specific location of Mars in its orbit is of no concern, for this particular mission. It is quite possible, however, that different trajectories may result in other planets blocking communications between Earth and the spacecraft. Should this occur at critical times, such as during an Earth-based update, it may become necessary to choose a different trajectory to overcome this difficulty.



Nomenclature: SC  $\underline{\mathbb{A}}$  Spacecraft

ANG 🛕 Angle

FIGURE 16. COMMUNICATIONS GEOMETRY

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TABLE XI. COMMUNICATIONS GEOMETRY SUBROUTINE OUTPUT FOR THE JUPITER FLYBY TRAJECTORY

## Computer Program Results

The computer runs presented in this section are for a Jupiter flyby mission with the data as shown in Figure 17. These runs demonstrate the effects of (1) control moment gyro (CMG) attitude control, (2) inertia wheel attitude control (IWAC), (3) reliability effects of switching subsystems on and off, and (4) communications transmitter and antenna parameter estimation. The results of including these features are summarized in Tables IX (shown in the section on Alternate Attitude Control Schemes) and XII. The runs were made with the data shown in Figure 17 on the mission schedules shown in Figure 18. Penalty analysis reports for the runs used in compiling Tables IX and XII are shown in Figures 19 through 36. Figures 19 through 28 do not include the effects of subsystem switching and communications requirements.

It is apparent that the main drawback of the reaction-wheel control is not so much in the additional weight, which is only about 1.4 pounds more than the control moment gyro system, but in the additional power required, which exceeds that of the CMG by a factor of 1.6.

Whether this is a prohibitive drawback or not will depend on the rest of the mission requirements. This total power could probably be reduced employing separate coarse and fine IWAC systems. It is not apparent how this would affect the reliability of the system, and it has not been modeled.

The onboard transmitter and antenna weight and power estimation discussed in the previous section have been included in the program. The user may specify a transmitter by giving its weight, power input, MTTF, MCTF, and power output. An antenna is then designed to provide the necessary gain. If no transmitter is specified the transmitter and antenna are designed to minimize the net weight (physical weight and energy source weight) while meeting the required effective radiated power (ERP). With the gain of the antenna known, the pointing tolerance is calculated. This tolerance is used as a constraint on attitude control deadband when the transmitter is operating.

The results shown in this report are for a designed transmitter and a hypothetical transmitter used to demonstrate the capability of specifying or designing the onboard transmitter.

		) tts) b)	Penalty (1bs)	384.326	384.047	391.567	394.965
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5 PARAMETER 1	Specified T	smitter Weigh smitter Powen na Weight	Attitude ( Weight (1bs)	21.215	21.218	29.019	30.422
JNICATION		Trans Trans Anter	Fig. No.	33	34	35	36
NBOARD COMM	tarta tarta	(1b) (watts) (1b)	Penalty (1bs)	377.392	377.114	384.633	388.032
TIONS, AND C	Transmitter	ht = 3.727 r = 36.536 = 14.615	Control Power (watts)	10.000	10.000	9.407	14.978
MECHANIZA	Designed '	imitter Weig imitter Powe ina Weight	Attitude Weight (1bs)	21.215	21.218	29.019	30.422
		Trans Trans Anter	Fig. No.	29	30	31	32
1			Attitude Control Scheme	Gas Jets (Unequal Thrusts)	Gas Jets (Equal Thrusts)	Control Moment Gyros	Inertia Wheels

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SUMMARY OF RESULTS FOR JUPITER SWINGBY COMPARING ATTITUDE CONTROL TABLE XII.

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FIGURE 17a. JUPITER FLYBY COMPUTER PROGRAM DATA

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FIGURE 18b. MISSION SCHEDULES EOR JUPITER FLYBY (Continued)

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FIGURE 18c. MISSION	N SCHEDUI	ES FOR JUPITER FLYBY (Continued)		
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JOULES WATTS 347.84395 •00063 33•463 176°711 141°375 43.500 09.511 33971 °231 106.500 .09213 48.100 101.582 110.000 28.434 LBS 8.306 90.109 26.264 .00000 EXCIT.POWER = TOTAL P.FAIL= TOTAL WEIGHT= PONER = P.FAIL= WEIGHT= ENERGY= POWER = ENERGY= 0.8 RAD/SEC EXCIT.ENERGY= WEIGHT= n TOTAL WEIGHT= TH0I3W 334-A REQUIREMENTS 11 11 PROB. FAIL.= WEIGHT= TOTAL TOTAL TOTAL TOTAL TOTAL OTAL TOTAL ဗ္ဗ n ê ENERGY POWER sys. STRAPDOWN STAR TRACKER, PENALTY (MODE 334-A 0000 000000 0.00000 2.517 10.849 2.1750 1.0875 ຍິຍ i Ħ n (HR) 334-A TIME (HR) 347.844 1652-156, 2000-000 HORIZ.SEN. NONE 000000 0.00.0 0.000 0000000 00000 COND. WEIGHT 1.345 10.000 6.000 TIME 99 TOTAL ENERGY= 34238.051 zo zo GYRO PRECESSION VELOCITY, SCHEDULE 1 MAX.THERMAL MIN.THERMAL GYROSCOPES= ASTRIONICS= SPACECRAFT= TOTAL= MCR-503 9601.033 33603.617 3.500 .09155 3.100 JUPITER FLYBY PENALTY EVALUATION, COM. SYST. INSULATION= ELECTRONICS= COMPONENTS= 27.870 WEIGHT ELAPSED=13.292 (SEC.) CAPABILITY= 0000 1SU/C.P.S. 0.00000 0000-0 D-4E NONE 97.875 ARMA 8.703 4.549 2.867 .06315 0.00000 .09271 .15000 14.985 DOF=1.000 ŝ 256.181 SENSOR --1402 10.849 2.000 54.246 5.000 (MUMIT40 00011 PENALTY (MODE 0-4E PROHABILITIES 53.04. POWER= WEIGHT BASE= COVER= BLOCK= INSUF.MIDCOURSE FUEL= п n н SUN SUN EXCESSIVE TGT. MISS ARMA •000122 • 046 1.102 .007 •881 . U0001 4 TOTAL POWER= ASTRIONICS TOTAL START= 39.74, END= MAX.HEATER MIN.HEATER •00012 ITT-LUN.03 10.849 86.793 STAR TRCKR 8.000 UNPELIABILITY DESIGN NUMBER 0-4E 19 (SCHEDULE NO. 1) CONTROL ANALYSIS MAN. TORQ.= FINAL H = TORQUE == FIGURE ŧi 81 n h CAPABILITY= DIMENSIONS 10.450 5.450 9.350 UELTA-V= ARMA FINAL H MAX. VELOCITY TORQUE TORQUE SHT RUK-2 2.517 226.575 90.000 .00042 36.000 ACCELERUM.= CUMPUTERS SUBSYSTEM PAHAMETERS ANALYSIS UATA (HORIZONTAL EXPECTED DATA =151]= HEIGHI= OUTSIDE LENGIH= SUMMATION MAX. TOTAL MUMENIUM EXECUTION TIMES. SOLAR PRESSURE WINCOURSE MIS. ENGINE ERROR ANALYSIS ISU COMPONENTS SYSTEM SIZING SOURCE ATTITUDE THERMAL METEORITE P.FAIL= WEIGHT= MANEUVER ENERGY= PONEK= TIME= MIDCOURSE PENALTY ENERGY WIRING 1SU ISU 0WO 88

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		66 334-A		EXCIT.ENERGY= EXCIT.POWER = TOFAL P.FAIL= TOFAL WEIGHT=	TOTAL ENERGY= Total Power =	TOTAL ENERGY= 101AL POWER = 101AL P.FAIL= 101AL W.FGHL=		EQUIREMENTS	ER = 10 36Y = 10 3. FAIL.= 0	. WEIGHT= 2	TOTAL WEIGHT= TOTAL WEIGHT=	TOTAL WEIGHT=	= (E
•		66 334-A	2.517		2+1750 1-0875		10,849	R	POW	SYS			PENALTY (MODI
	a de la companya de la composición de l	i= 66 334−A	N TIME (HR)=	1.345 10.000 6.000	L COND.=	HORIZ.SEN. NONE 0.000 0.000 0.000 0.000 0.0000	N TIME (HR)=				101		WEIGHT 372.604 1627.396 2000.000
- <b>Company of the Annual States of States</b>	novem and a second s	GYRUSCOPES	0	WEIGHT AULATION= CTRONICS= PONENIS=	HAX.THERMA NIN.THERMA	COM. SYST. MCH-503 4601.033 3503.617 3503.617 3.500 3.105 3.105	0	ran, beine instal bereinigen is werden beine der Antonio eine stere			IERGY= 34315.	28.061	ASTRIONICS= SPACECGAFT= TOTAL= (.)
	and the state of managements and the state of the state o	RMA 0-4E		703 INS 549 ELEC 867 COM	. 875 . 000	15J/C+P+S+ NONE 0.000 0.000 0.000 0.000 0.0000		n mag gran na bi u ring. gran ta an gana da gana ang ang ang ang ang ang ang ang a			TOTAL EN	CAPAHILITY=	303 000 242 000 €0=13•300(SE
TY (MODE 3)		RMA 1)-4E A	OPTIMUM)	WETGHT LOCK= 8. dase= 4.		SUN SENSOR ADCL-1402 10.944 54.54 54.64 54.600 -0001 2.000		nga da nasar ang nga nga nga nga nga nga nga nga nga	07 43 72 22 22	н] 69 69	=	1 DOF=1.000	1[[TTTES 066 EUFL= 0600 TSS = 0.000 = 099 = 15 62.49. FLAPS
NO. 1)	4. The second se Second second sec	RMA 1)-4E A	GN NUMBER 4	1045 350 HI 450 C	MAX.HEATER	51AR TRCKR 51AR TRCKR 51AH.51 10.449 10.449 14.000 26.500 26.500			ни и и и и и и и и и и и и и и и и и и	$H = 1.40$ $H = 1.40$ $I \uparrow Y = 1.40$	TOTAL POWER:	-V= 15.04	
S (SCHENDLE	Ş	ELEROM .= AI	ZONTAL DEST	STDE DTAEUS (61H= 9+ 01H= 10+	MALYSTS	20-19 LTERS 20-19 LTERS 20-19 LTERS 20-10 LTERS 20-000 20-000 36-000	CONTROL ANAL	()	AK. VELACIT AK. MAN. TUR FINAL TODDE	IIS. FOODE FINAL		INE FCIED DFLTA-	LTON TNSUI EXCE ASTR ASTR
HOR ANALYST	1 COMPONENT	אככ	SU DATA (HUH)	100 130	SU THERMAL N	0.58851EM PAR 1186 - 0 1186 - 0 2006 - 10 2006 - 1	40 ATTITOF	MALEN SIXIN	AANEUVER A Second Frank Second Praces	TOTAL MUNEU	VERGY SUUPCE	TOCHJASE Eve	сайцтү Зајача колттор креднттор

11.792 WATTS 127.935 JOULES 176.711 141.375 359.47951 •00063 33•463 106.500 43.500 109.511 102.785 48.100 110.000 33971 °231 28.434 LBS 36.697 • 000000 EXCIT.ENERGY= EXCIT.POWER = Tofal P.FAIL= ENERGY= POWER = P.FAIL= WEIGHT= H 11 wEIGHT= ENERGY= WEIGHT= TOTAL WEIGHT= WEIGHT 334-A POWER REQUIREMENTS 0.35 RAD/SEC 0 0 FAIL.= WEIGHT= 99 TOTAL TOTAL TOTAL TOTAL TOTAL TOTAL TOTAL TOTAL н Э POWER Energy PROB. sys. PENALTY (MODE 334-A 2,517 0.000 000000 0.0000.0 000.0 10.849 JUPITER FLYBY PENALTY EVALUATION, STRAPDOWN STAR TRACKER, 2.1750 99 11 = ( HH ) = 334-A (HR) 0.000.0 00000 0.00.0 359.480 1640.520 00000000 0.000 HOHIZ.SEN. 2000.0005 COND.= COND.= **MACN** 1.345 WEIGHT ON TIME TIME 6.000 99 34275.877 zo MIN. THERMAL 67HOSCOPES= MAX. THERMAL ASTHIONICS= SPACECRAFT= TOTAL= 3.100 9601.033 33603.617 3.500 CUM. SYST. 27.870 GYRO PRECESSION VELOCITY, SCHEDULE 1 ELECTRONICS= WE IGH INSULATION= COMPONENTS= MCH-503 ENERGY= 30.44. ELAPSED=13.292 (SEC.) ]4.9H5 DOF=].000 CAPABILITY= 0.000.0 0.000 0.000 000000000 TSU/C.P.S. TOTAL **ARMA D-4E** NONE 97.875 4.549 .06315 0.00000 R.703 .15000 17990. m 259.667 5.000 -00011 2-000 10.449 SUN SENSOR 945.96 (MIMI I do AUCL-1402 (MODF ARMA U-45 PROHABILITES POWER= WEIGHT FUFL= BLOCK= 11 12 HASE= COVFR= EXCESSIVE TGT. MISS .000122 PENALTY .007 •046 2.525 2.528 .000011 . 441 TOTAL POWER= \$ MIN.HEATER ASTHTOMICS TOTAL SIARI= 1/.20. ENDE MAX.HEATER TNSUE \* MTUCUURSE 7.000 STAR TRCKR 10.849 66.793 ITT-LUN.OR 8.000 .00012 ISU DATA(HOMIZONTAL DESIGN NUMBER 1200FLTARTLTTY ARMA D-4E 6 AMALYSIS • GN CAPANILITY= **UIWENSIONS** # 11 11 н U. 11 22. 11 ]0.450 0.350 DELTA-V= 5.450 MAN. 1080. I τ VELACITY FINAL (SCHEDINE Topont FTWAL Toport THOODE FIGURE 90.000 712.9 226.415 5 400 U.V. 36.000 :: CO IPUTERS SHI RUK-2 CONTROL ANALYSTS PARAMETERS ACCELERON EXPECTED NATA 001510F LENGTH= \* 1014= HE 16HT= NULLYMENS PiaX. ₩Aχ. **PÚRENTUR** SARCHTTON TINES SOLAP PRESSIVE 115. F.N.G.I.M.F AWALYSIS 3 2 2 SYSTEM SIZING SOURCE CMG ATTITUDE SECTANCO THERMAL all40413m EnF 26 Y= PONEX= P.FAJUE =149136 IANFINEN TINEE SURSYSTEM RJUCOURSE INTAL PFUALTY FNERGY SULTIA 40883 : SC 151 91

JOULES WATTS •00063 33•463 112•500 •09224 67•600 176•711 141•375 368.42466 43.500 34036.326 09.511 103.526 110.000 28.490 LBS 86.124 25.346 7.938 .00000. EACII.POWER = TOTAL P.FAIL= TOTAL WEIGHT= POWER = P.FAIL= EXCIT.ENERGY= 0 0 ENERGY= WEIGHT= WEIGHT= WEIGHT= WEIGHT= ENERGY: POWER = EXCIT.POWER 334-A REQUIREMENTS PROB. FAIL.= 11 - 11 WEIGHT= T01AL T01AL T01AL 99 TOTAL TOTAL TOTAL TOTAL TOTAL n ຄ ENERGY POWER SYS. PENALTY (MODE 334-A TRACKER 0000000 2,517 0000.0 00000.0 0.00.0 10.849 2.1750 99 JUPITER FLYEY PENALTY EVALUATION, GIMBALLED STAR 0.9 RAD/SEC GYRO PRECESSION VELOCITY, SCHEDULE 1 8 11 334-A (HR) (HR) 0000.0 368•425 1631•575 2000•000 HORIZ SEN 0.000 00000000 0.000 COND.= NONE 1.345 10.000 6.000 ON TIME WEIGHT TIME 00 TUTAL ENERGY= 34299.161 GYROSCOPES= MAX.THERMAL MIN.THERMAL zo ASTRIONICS= SPACECHAFT= TOTAL= 9601.033 33603.617 3.500 3.100 COM. SYST. INSULATION= ELECTRONICS= COMPONENTS= WEIGHT 28.06] MCR-503 ELAPSED=13.292 (SEC.) 15.081 DOF=1.000 CAPABILITY= ISU/C.P.S. NONE 0000-0 0.00.00 0.00000 0.000 0-4E 97.875 8.703 4.549 2.867 ARMA .09282 • 06303 0.0000.0 .15000 ŝ 261.813 10.849 SUN SENSOR 54-245 5.000 (MUMI 1 40 .00011 2.000 ADCL-1402 PENALTY (MUDE ARMA D-4E **PRUBABILITIES** MAX.HEATER POWER= MIN.HEATER POWER= WEIGHT BLOCK= BASE= 62.42. CUVER= INSUF.MIDCOURSE FUEL= EXCESSIVE TGT. MISS = H 11 •000122 •000122 .043 +007 612. 67.6. .881 4 TOTAL POWER= ASTRIUNICS TUTAL EXECUTION IIMES. START= 49.13. END= 151.849 151.888 14.000 STAH TRCKH 26.500 •00024 DESIGN NUMBER UNRELIABILITY 0-4E 23. GIMB.ST NU. 1) LUNTROL ANALYSIS UUTELUE DIMENSIONS ŧ MAN. TORU.= 11 11 11 п 11 FIGURE ARMA 9.350 10.450 CAPAULL [Y= 5.450 UELIA-V= FINAL H r VELOCITY ERROR ANALYSIS (SCHEDULE รีบชุญรี TURNUE F INAL TURUUE 5H1 HUK-2 2.517 226.575 90.000 .00042 36.000 ACCFLEROM.= CUMPUTERS ISU THERMAL ANALYSIS SUHSYSTEM PARAMETERS SU DATA (HURIZONTAL EXPECTED ENERGY SUUNCE UATA LENGIME wluin= nelonl= • XTN PENALIY SUMMAIION NAX . TUTAL HUMENIUM MIUCUURSE MIS. SULAR PRESSURE LNULNE ISU COMPONENTS SYSTEM SIZING CMG ATTITUDE METEORITE TIME= ENERGY= PO#EK= MANEUVER WEIGHT= MIDCOURSE WIHING 92

C ERROR ANALYSIS (SCHEDULE NO, 2)	
ISU COMPONENTS	
ACCELEROM.= ARMA D-4E ARMA D-4E ARMA D-4E GYROSCOPES= 6G 334-A 6G 334-A	GG 334=A
ISU DATA(HORIZONTAL DESIGN NUMBER 4 OPTIMUM)	
OUTSIDE DIMENSIONSWEIGHTWEIGHTEXCLENGTH=9.350BLOCK=8.703INSULATION=1.345EXCMEIDHT=10.450BASE=4.549ELECTRONICS=10.000TOTHEIDHT=5.450COVER=2.867COMPONENTS=6.000TOT	EXCIT.ENERGY= 154.461 EXCIT.POWER = 43.500 TOTAL P.FAIL= 000089 TOTAL WEIGHT= 33.463
ISU THERMAL ANALYSIS Max.HEATER POWER= 97.875 Min.HEATER POWER=000 MIN.THERMAL COND.= 2.1750 TO Min.HEATER POWER=000 MIN.THERMAL COND.= 1.0875 TO	TOTAL ENERGY= 249.244 TOTAL POWER = 141.375
SUBSYSTEM PAMAMETERS       STAP TRCKR       SUN SENSOR       ISU/C.P.S.       COM. SYST.       HORIZ.SEN.         CUMPUTERS       STAP TRCKR       SUN SENSOR       ISU/C.P.S.       COM. SYST.       HORIZ.SEN.         SHT PUK-2       ITT-LUN.03       ADCL-1402       NONE       MCR-503       NONE         SHT PUK-2       11.982       11.882       0.000       9601.033       0.000       0.000         ENERGY=       319.575       95.060       59.412       0.000       33603.617       0.000       0.000         POWER=       90.000       8.000       59.412       0.000       33603.617       0.000       0.000       0.000       000       000       000       000       0000       0000       0000       0000       0000       0000       0000       0000       0000       0000       0000       0000       0000       00000       0000       000000       00000       00000<	TOTAL ENERGY= 34077.664 TOTAL ENERGY= 34077.664 TOTAL POWER = 106.500 TOTAL P.FAIL= 009231 TOTAL WEIGHT= 48.100
CMG ATTITUDE CONTROL AMALYSIS	
SYSTEM SIZING	QUIREMENTS
MANEUVER MAX. VELOCITY = .007 MAX. MAN. TORG.= .046 Einan H	R = 8.306 WATTS GY = 98.692 JOULES
METEORITE TORAUE = .000122 SOLAR PRESSURE TORAUE = .000122 SOLAR PRESSURE TORAUE = .000011	• FAIL.= •000004
MIDCOURSE MIS. TORRUE = .681 FINAL H = 1.102 TOTAL MUMENIUM CAPABILITY= 1.102 IOTAL MUMENIUM CAPABILITY= 1.102	WEIGHT= 26.264 LBS
ENERGY SOURCE DATA TOTAL POWER= 256.181 TOTAL ENERGY= 34425.600	TOTAL WEIGHT= 101.582
TOT	TOTAL WEIGHT= 110.000
MIDCOURSE EMGINE EXPECTED DELTA-V= 14.945 DOF=1.000 CAPABILITY= 27.841 TOT	TOTAL WEIGHT= 28.426
PEWALTY SUMMATION PROBABILITIES WEIGHT INSUF.MIDCOURSE FUEL= .066272 ASTRIONICS= 347.835 EXCESSIVE TGT. MISS = 0.00000 'SPACECRAFT= 1652.165 UNMELLARILITY = .09312 TOTAL= 2000.000 ASTRIONICS TOTAL = .16000	•
EXECUTION TIMES, START= 21.13, END= 39.74, ELAPSED=18.612(SEC.)	3)= 347.83533
FIGURE 24. JUPITER FLYBY PENALTY EVALUATION, STRAPDOWN STAR TRACKER, 0.8 RAD/SEC GYRO PRECESSION VELOCITY, SCHEDULE 2	

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PENALTY (MODE 3)

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e

MALITY (MONE 3)       ARMA D-4E       ARMA D-4E	PENALTY (MODE 3)         PLAN D-4E         ARMA D-4E         FISIONS         WETGHT         9.450         D.450         HONCE         ASS         HLOCK         ARAS	NUMPLYSIS     (5C-F-NULE NO. 2)       NUMPLYSIS     (5C-F-NULE NO. 2)       NUMPLYSIS     (5C-F-NULE NO. 2)       NUMPLYSIS     (5C-F-NULE NO. 2)       NUMPLYSIS     (1004120NTAL INFSIGN NUMMER 4 0PT1411M)       NUTSTUF NUMBLYSIS     NAX-HFATER POWER       NUMLYSIS     NAX-HFATER POWER       NUMLYSIS     NAX-HFATER POWER       NUML INS.     NAX-HFATER POWER       NUML INS.     NAX-HFATER POWER       NUML ANALYSIS     NAX-HFATER POWER       NAX-HFATER POWER     0.000       NAX-HFATER POWER <t< th=""><th></th><th></th><th>GYHUSCOPES= 66 334-A 66 334-A 66 334-A</th><th>0N TIME (HR)= 3.551</th><th>WEIGHT EXCIT.ENERGY= 154.461</th><th>ATION= 1.345 EXCIT-POWER = 43.500 ANICS- 10.000 TOTAL P.FATE - 00080</th><th>NENTS= 6.000 TOTAL WEIGHT= 33.463</th><th>MAX.THERMAL COND.= 2.1750 TOTAL ENERGY= 249.244</th><th>M. SYST. HORIZ.SEN.</th><th>14-503 NONE 0601.033 U.000 0.000</th><th>13603-617 0.000 0.000 TOFAL ENERGY= 34077.664</th><th>3.500 0.000 0.000 0.0000 101AL POWER = 106.500 0.00155 0.0000 0.0000 101AL P.FATL= 00231</th><th>3.100 0.000 0.000 TOTAL WEIGHT= 48.100</th><th>ON TIME (HR) = 11.882</th><th>REQUIREMENTS</th><th>POWER = 9.407 WATT ENERGY = 111.785 JOUL</th><th>PROB. FATL.=000002</th><th></th><th>SYS. WEIGHT= 29.019 LAS</th><th></th><th>16Y= 34438.693 TOTAL WEIGHT= 101.962</th><th>TOTAL WEIGHT= 110.000</th><th>27.840 TOTAL WEIGHT= 28.425</th><th>WEIGHT WEINICS= 350.970</th><th>iPACECRAFT 1649.030 TOTAL= 2000.000</th><th></th></t<>			GYHUSCOPES= 66 334-A 66 334-A 66 334-A	0N TIME (HR)= 3.551	WEIGHT EXCIT.ENERGY= 154.461	ATION= 1.345 EXCIT-POWER = 43.500 ANICS- 10.000 TOTAL P.FATE - 00080	NENTS= 6.000 TOTAL WEIGHT= 33.463	MAX.THERMAL COND.= 2.1750 TOTAL ENERGY= 249.244	M. SYST. HORIZ.SEN.	14-503 NONE 0601.033 U.000 0.000	13603-617 0.000 0.000 TOFAL ENERGY= 34077.664	3.500 0.000 0.000 0.0000 101AL POWER = 106.500 0.00155 0.0000 0.0000 101AL P.FATL= 00231	3.100 0.000 0.000 TOTAL WEIGHT= 48.100	ON TIME (HR) = 11.882	REQUIREMENTS	POWER = 9.407 WATT ENERGY = 111.785 JOUL	PROB. FATL.=000002		SYS. WEIGHT= 29.019 LAS		16Y= 34438.693 TOTAL WEIGHT= 101.962	TOTAL WEIGHT= 110.000	27.840 TOTAL WEIGHT= 28.425	WEIGHT WEINICS= 350.970	iPACECRAFT 1649.030 TOTAL= 2000.000	
NALTY (MODE       ARMA U-4E       4     OPTIMUM       WETGHT       RLOCK=       BASE=       COVER=       BASE=       COVER=       BASE=       COVER=       BASE=       COVER=       BASE=       COVER=       BACL-1402       BACL-1402       0       SON       0       SON       0       SON       AUCL-1402       BAL       0       SON       SON       AUCL-1402       BAL       0       SA       0       SE       MISS       MISS	PENALTY (MODE       PLAE     ARMA D-4E     ARMA D-4E       FSIGN NUMMER     4     PFIMUM       FSIGN NUMMER     4     PFIMUM       FNSIONS     WETSHT     9.450       FNSIONS     WETSHT     9.450       FNSIONS     WETSHT       PASS     RLOCK=       JAAX-HEATER     POWER=       JAAX-HEATER     POWER=       ANALYSIS     RLOCK=       ANALYSIS     00013       ANALYSIS     000122       ANALYSIS     00011       PASE     000122       PASE     000122       PASE     00011       PASE     000122       PASE     00011       PASE     000122       PASE     00012       PASE     00012       PASE     00012       PASE     00012       PASE     000011	ML_YSIS (SCHERULE NO. 2)     PENALITY (MODE       ML_YSIS (SCHERULE NO. 2)     MML_YSIS (SCHERULE NO. 2)       OUSTUE NTWENSIONS     METGHT       ALGELEHUM.= ARMA D-4E     AMMA U-4E       ALOTIDE NTWENSIONS     METGHT       MINUMERR SUN SENSIONS     MARCLIADS       MTNUHEATER POWERE     NAX-MEATER POWERE       MTNUHEATER POWERE     NAX-MEATER POWERE       MINUMERR SUN SENSIONS     NAX-MEATER POWERE       MALANDONO     NAX-MEATER POWERE       MALANDONO     NAX-MEATER POWERE       MAX-WEATER POWER     NAX-MEATER POWERE       MALANDONO     NAX-MEATER POWERE       MALANDONO     NAX-MEATER POWERE       MAX-MEATER POWERE     NAX-MEATER POWERE       MALANDONO     NAX-MEATER POWERE       MAX-MALYSIS     NAX-MEATER POWERE       MAX-MALYSIS     NAX-MEATER POWERE       MAX-MALANDA     NAX-MAL	(٤		ARMA U-4E	···· ··· ··· ··· ··· ··· ··· ··· · ··· ·	-	R.703 INSUL	P.867 COMPO	97.875 	R	NUNE MC	E 0000 0	0.000	0.000					ang dhangan shirk ƙwarni Mas ya si si ya ma wa hisso a mananan ƙwarnin ƙwardo ƙasa		ar 11111 ar 1 1199 ar 1	R2 TUTAL ENER		00 CAPAHILI FY=	. 16272 A	• 00000 S	.15000
	PE     PE       PILE NO.     2)       FSIGN NUMMER       FSIGN NUMMER       FNSIONS       9.350       9.350       10.450       10.450       10.450       10.450       10.450       10.450       10.450       10.450       10.450       10.450       11.450       11.450       11.4450       11.4450       11.4450       11.4450       11.4450       11.4450       11.4450       11.4450       11.4450       11.4450       11.4450       11.4450       11.4450       11.4450       11.4450       11.4450       11.4450       11.4450       11.4450       11.1450       11.141       11.141       11.141       11.141       11.141       11.141       11.141       11.141       11.141       11.141       11.141       11.141	ML_YSTS     (SCHEMULE NO. 2)       UNFLITS     ACCFLEMULE ARMA 10-4E       ACCFLEMULE ARMA 10-4E       (HONLZONTAL DESTON NUMMER       OUTSTUE DTME DESTON NUMMER       OUTSTUE DTME DESTON NUMMER       OUTSTUE DTME DAMA 450       MIDTHE DTME STAR       MIDTHE DTME STAR       MIDTHE DAMA 4515       MAX.HEAT       MIDTHE DTME STAR       MIDTHE DAMA 4515       MAX.HEAT       MAX.HEAT       MIDTHE DAMA 4515       MIDTHE DAMA 4515       MAX.HEAT       MIDTHE DAMA 4515       MAX.HEAT       MAX.HEAT       MIN.HEAT       MIN.HEAT       MIN.HEAT       MIN.HEAT       MIN.HEAT       MIN.HEAT       MAX.HEAT       MIN.HEAT       MIN.HEAT </td <td>NALTY (MODE</td> <td>a a se anna a se anna de se a se a se a se a se a se a se a</td> <td>ARMA U-4E</td> <td>4 OPTIMIM</td> <td>WETGHT</td> <td>BLOCK= HACF=</td> <td>COVERS</td> <td>FR POWER=</td> <td>R SUN SENSO</td> <td>H AUCL-1402</td> <td>0 59.41</td> <td>00•5 0001</td> <td>0</td> <td></td> <td></td> <td>• 007 • 046</td> <td>110.</td> <td>00011</td> <td>- X4) 1 - 469 1 - 460</td> <td></td> <td>WEH= 257.2</td> <td></td> <td>•945 DOF=1.0</td> <td>RAHICITIES</td> <td>• #ISS = 0</td> <td>TAL =</td>	NALTY (MODE	a a se anna a se anna de se a	ARMA U-4E	4 OPTIMIM	WETGHT	BLOCK= HACF=	COVERS	FR POWER=	R SUN SENSO	H AUCL-1402	0 59.41	00•5 0001	0			• 007 • 046	110.	00011	- X4) 1 - 469 1 - 460		WEH= 257.2		•945 DOF=1.0	RAHICITIES	• #ISS = 0	TAL =

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SU COMPANY SUCCE	NN.E NO. 2)	4				a a se e genero, managemente en a		and a second
×CCELEROM≂	: ARMA ()-4E A	18MA 0-4E	AI3MA ()-4E	64H0SCOPE:	S= 66 334-A	66 33 <b>4</b> -A	GG 334-A	er forske en
O DATA (HORIZONTAL)	FESTGN NUMBER 4	(MUMITO	A	)	ON TIME (HR)=	3,551	a and and any start of the second sec	ana a ann aite i stàite anns an san anns an stàite anns an stàite
0015105 0174 Leutotte w101te W10341	4ENS 10MS 4.350 4.350 1.4.850 5.450 0.0	WE1GHT 4LOCK= 845E= 50VER=	8-703 IN 4-549 ELEC	WEIGHT SULATION= CTRONICS= KPONENIS=	1.345 10.000 6.000		EXCIT.ENERGY= EXCIT.POWER = TOTAL P.FAIL= TOTAL WEIGHT=	154.461 43.500 00089 33.463
APPPAR ANALYSTS	MAX+HEATER MIN+HEATER	P0%E8= 4	•7•875 -•000	MAX . THEKM	AL COND.= AL COND.=	2.1750 1.0875	TOTAL ENERGY= TOTAL POWER =	249°244 141°375
JASYSTEIN PARANETERS COMPUTERS	STAR THCKR	SUN SENSOR	TSU/C.P.S.	СUM. SYST. МС0-603	HORIZ•SEN• NOME			
TTPE TTPE Faltabyth 319,55	11 01 04 04 04 04 04 04 04 04 04 04 04 04 04	AUCL-1.07 11.882 59.412	0000-0	9601.033 33603.617	0000-0	00000	TOTAL FNFRGY=	34148,050
D0.00 = HH-00.00	14.000	000.5	0.000	3.500	0000	0000	TOTAL POWER =	112.500
P.Fn11.= 00000 WF10H1= 36.00	10 • 00026	• 00012 2• 000	00000°0	。09155 3。100	0.00000	0.0000000000000000000000000000000000000	TOTAL P.FAIL= TOTAL WEIGHT=	•09243 67•600
HATTIFUNE CONTROL	AMALYSTS	and the second se			ON TIME (HR)=	11.882		
5YS1EP 51/146		rannan	ramman yr i o'n cifa, do o'r o'r o'r o'r argenrae maerer fys Anne maerer ar a			æ	EQUIREMENTS	n baga nga na nano na Ariminin (na nano na nanona alaga nanona a
HANENVER HAX. VELO HAX. MAN.	1080.=	107 143			star Allandina and Santa and Allandina and Allandina and Allandina and Allandina and Allandina and Allandina an	POW	ER = 11	9.407 WATTS 1.785 JOULES
аңтеритте Аңтеритте Son да Риезуние торо	00015 11 00001	122 122 011	and a semicount frame of the data is the family of the set of		no de antes - como e antes e como de de antes e como	ояч	B. FAIL.= .0	00005
ALDCOURSE HIS. TOP FT TOTAL MUNENTIAM CADA	00HE =	481 469 469				SYS	• WEIGHT= 2	9.019 LRS
HERRY SUUNCE DATA	TOTAL POWFF	t= 263+2R5	P IUTAL EN	NEHGY= 34509,	•988		TOTAL WEIGHT=	104.032
1-100 F						an t	TOTAL WEIGHT=	110.000
LOCOURSE FUGLOE	LTA-V= 15.05	56 DOF=1.000	D CAPAHILITY=	28.060			TOTAL WEIGHT=	28.489
EINLTY SJAMATION	NSUF . WTUCOURSE	FUFL= .C	06260 	ASTRIONICS:	- WEIGHT = 372.604 		· · ·	
	TACESSIVE TGL. T RAVELIAGILITY A COTOMICS TOTAL		00000 09324 15200	SPACECHAR I TOTAL	= 2000.000	an sa an		
		1			an and an and a second se	PENALTY (MOD	E 3)=	372.60345

JOULES WATTS 43.500 •00089 33.463 359.470AB 249.244 141.375 106.500 34077.664 .09231 48.100 102.785 110.000 28.425 154.461 36.697 LBS 11.792 140.121 • 000005 ENERGY= POWER = POWER = EXCIT.ENERGY= H TOTAL P.FAIL= TOTAL WEIGHT= ENERGY= wE16H1= WEIGHT= wElGHT= WEIGHT= EXCIT.POWER 334-A REQUIREMENTS 0 0 FAIL.= WEIGHT= TOTAL TOTAL 99 TOTAL TOTAL TOTAL TOTAL TOTAL TOTAL H  $\widehat{\mathbb{C}}$ ENERGY PROB. POWER sys. PENALTY (MODE 334-A TRACKER 0.00.0 11.882 3.551 2.1750 99 2 JUPITER FLYBY PENALTY EVALUATION, STRAPDOWN STAR 0.35 RAD/SEC GYRO PRECESSION VELOCITY. SCHEDULE (HR) TIME (HR)= 334-V 359.471 1640.529 0.000 000000 0.00.0 00000000 0.00.0 2000-0005 HURIZ.SEN. COND.= NONE TIME 3.345 10.000 VEIGHT **6.000** 00 34467.029 MAX.THERMAL MIN.THERMAL N N GYROSCOPES= ASTHIONICS= SPACECPAFT= TOTAL= 3.500 CUM. SYST. 9601.033 33603.617 3.100 66190. 27.840 WEIGHT ELECTHONICS= COMPONENTS= INSULATION= MC4-503 ENENGY= FI\_ADSCIN=18.608 (SEC.) 14.945 DOF=1.000 CAPARILITY= 0.000 0000.0 TSU/C.P.S. 000000000 0.000 TOTAL **APMA D-4E** NONE .000.-97.875 4.549 .15312 15000 0.00000 R.703 259.667 e, AUCL-1402 11.442 59.412 5.000 SENSOR 2.010 51000 (MUNIT40 RODE ARMA 11-4E PROMARILITIES SIADT= 30.50. EMP= 49.10. POWER= WEIGHT MAX.HEATER POWER= HLOCK= н 11 HASE= COVER= TNSUE.MIDCOURSE FUEL= EXCESSIVE TGT. MISS = SUN .000122 PENAL TY .046 2.528 .007 164. 2.528 .000011 =njmOd 4 ASTRIDUICS TOTAL WIN.HEATEN 95.060 8.000 ITT-LUN.OH 11.842 7.000 STAR TRCKR .00013 27. DESIGN NUMHER HNPEL LANILITY ARMA U-45  $\widehat{\sim}$ TOTAL ANAL YSIS FIGURE • 22 UTHENS LONS 11 MAN. TURQ.= 11 11 11 11 11 CAPANTLITY= 9.350 10.450 5.450 DFLTA-V= FINAL H I VELOCITY ( כרובהוונב FINAL. ThumanE 300001 Theode 000.00 3.551 319.475 .00059 36.000 ACCFLERON.= SRT PUK-2 CUMPUTFOS ANALYSTS PARAMETERS CONTROL ISU DATA (HOHIZONTAL EXPECTED DATA 101 S J NO LE-07H= a [1) [ H= HE LGHT= · > V --ыдх. SURWATTON FOTAL HOWENTHM HIS. KECHTTON TIMES. SULAR PRESSURE FNGTRIE SISYJAUA ISU COMPONENTS SYSTEM SIVING SUURCE CMG ATIITUNË THERMAL atocoustse H1140913,0 TTNFI Elett P.F∧JL= # ] H51 34 MANUTRE #21/1001 SURSYSTEM Thenuse PENALTY ENERGY 80443 **Owlaim** 131 2 96

JOULES 368.42417 WATTS 43•500 •00089 249°244 141°375 34148°959 112.500 .09243 103.526 28.489 33.463 67.600 54.461 110.000 25.346 LHS 7.938 94.327 • 000005 EXCIT.POWER = TOTAL P.FAIL= TOTAL WEIGHT= PUWER = P.FAIL= ENERGY= POWER = EXCIT.ENERGY= ENERGY= WEIGHT= WEIGHT= WEIGHT= WEIGHT= 334=A POWER REQUIREMENTS SYS. WEIGHT= 11 - 11 PRUB. FAIL.= TOTAL 99 TOTAL **LOTAL** TOTAL TOTAL TOTAL TOTAL 9 10 10 ENERGY POWER PENALTY (MODE 334-A TRACKER. 000000 000000 0.00.0 0.00.0 11.882 3,551 2.1750 ю Ю JUPITER FLYBY PENALTY EVALUATION, STRAPDOWN STAR 0.8 RAD/SEC GYRO PRECESSION VELOCITY, SCHEDULE 2 U (HR) = (HR) 334-A 368-424 1631-576 2000-000 0000.000 000.0 HORIZ . SEN. 00000000 0 • 0 0 0 COND == NONE 1.345 TIME TIME WEIGHT 6+000 99 34492.531 z S GYROSCOPES= MAX.THERMAL MIN.THERMAL ASTRIONICS= SPACECHAFT= TUTAL= 9601.033 33603.617 3.500 SYST. 3.100 INSULATION= ELECTRONICS= COMPONENTS= 28.060 WEIGHT MCR-503 ENERGY= COM. ELAPSED=18.516(SEC.) 15.056 DOF=1.000 CAPABILITY= 0.000 ISU/C.P.S. TUTAL 0-4E 97.875 8.703 4.549 2.867 ARMA 00000000 \*09324 .15000 06260 261.813 ŝ 11.882 59.412 5.000 .00012 SUN SENSOR 2.000 (MUMIJ 40 AUCL-1402 C MODE ARMA D-4E PROBABILITIES POWER. WEIGHT ENU= 80.94, BLOCK= BASE= COVER= INSUF MIDCOURSE FUEL= 11 11 11 EXCESSIVE TGT. MISS •000122 •000122 PENALTY ۠0• .979 .979 • 007 .881 PUWER= 4 ASTRIUNICS TUTAL MAX.HEATER MIN.HEATER 11.882 266.352 14.000 STAR TRCKR GIMB.ST 26.500 DAFA (HURIZUNTAL DESIGN NUMBER UNHELIABILITY 0-4E 28 ŝ TUTAL START= 62.42. LUNTRUL ANALYSIS KHOR ANALYSIS (SCHEDULE NO. OUTSIVE DIMENSIONS 11 Ħ MAN. TORU.= 11 1I H CAPABILITY= ARMA 9.350 10.450 FIGURE 5.450 UELTA-V= FINAL H I VELUCITY TURGHE 102005 TURWUE F INAL 541 RUK=2 3.551 3.575 90°00°9 36.000 ACCELEHOM.= CUMPUTERS ISU THERMAL ANALYSIS SUNSYSTEN PARAMETERS EXPECTED SOUNCE UNTA LENGTH= WILTH= HELGH]= SUMAAL UN EXECUTION TIMES. WHX-TUTAL MUMENIUM 14X • SOLAH PHESSURE MINCUURSE MIS. ENGLOF ISU COMPONENTS SYSTEM SIZING CMG ATTITUUE METEURITE TIME= ENERGY≡ P.FAIL= WE LOH I = MANEUVER MIUCOURSE PENALTY ENERGY WIHING SUS 97
SU COMPONENTS						a server a gran and a server a server a server a		
ACCELERUM.= ARMA D-4E AF	3MA U-4E	ARMA D-4E	GYROSCOPES=	66 334-A	66 334 <b>-</b> A	GG 334-A		
U DATA(HURIZUNIAL DESIGN NUMBER4	(MUMI140	CYCLE	S= 3 0N	TIME (HK)=	2.517	And and an and a second s	an in a subscription of the subscription of the subscription of the	<b></b>
UUTSIDE DIMENSIONS     V       LEMGTH=     9.350     BL       JIDTH=     16.450     F       HELGHT=     5.450     CO	AFIGHT -OCK= R 3ASE= 4 VER= 2	.703 INS .549 ELEC .867 COM	WEIGHT SULATION= STPONICS= 1 PONENTS=	1.345 0.000 6.000		EXCIT.ENERGY= EXCIT.POWER = TOTAL P.FAIL= TOTAL WEIGHT=	109.511 43.500 03597 33.463	
U THERMAL ANALYSIS MAX.HEATER F MIN.HEATER F	00WFR= 9	7.875 000	MAX.THERMAL MIN.THERMAL	COND . II COND . II	2.1750 1.0875	TOTAL ENERGY= TOTAL POWER =	176°711 141°375	-
TENNA DESIGN GAIN (DB) = 23.6	ERP (Kw) =	3•36 P(	DINTING TOL. (D	EG.)= 1.08		TOTAL WEIGHT=	14.615	
35YSTEM PAKAMETERS         35YSTEM PAKAMETERS         SRT RUK-2         STAR         STAR	SUN SENSOR ADCL-1402 10.849 54.246 5.000 0.01410 2.000	1 SU/C.P.S. NONE 0.600 0.000 0.000 0.000 0.000	COM. RCVR. H MCR-503 9601.033 33603.617 33603.617 3.500 .09336 3.100	0.812.55.N. NONE 0.000 0.000 0.000 0.000 0.0000	XMITTER DESIGNED 0.000 0.000 36.536 3.727	TOTAL ENERGY= 3 TOTAL POWER = TOTAL POWER = TOTAL P.FAIL= TOTAL WEIGHT=	3971.231 143.036 .10654 51.827	
TITUDE CONTROL SYSTEM ANALYSIS		CYCLES= 2	NO	TIME (HK)=	10.849			1
THRUST SIZING (LB) RALL YAW RALL YAW	eltCH	-	FUEL CONSU	MPTION (LB- YAW	SEC) PITCH			
-1.1.1MPACT= 0000 0000 ATEUVERS = 0.0000 0000 TOCOURSE = 0.0000 3778 AX.THRUST= 00099 3778	.0001 .0001 .3778 .3778	SEARCHT DEAD BAI MANEUVEI TOFAL II	NG= 4.0585 NU= 0001 NS= 2793 NP= 4.3379	7.3060 4445 1547 2.9052	•003) •2560 •1917 •4508	TOTAL ENERGY= TOTAL POWER =	108.492 10.000	
). OF FIRINGS= 22440 TOTAL	INPULSE=	7.6940	FUEL WEI	GHT= .137	393	TOTAL VETCHTE	21•212 21•212	
ERGY SOURCE DATA TOTAL POWER	= 294.411	TOTAL E	VERGY= 34256.4	3 <b>4</b>		TOTAL WEIGHT=	.114°772	
21NG	:	•		•	and the second	TOTAL WEIGHT=	110-000	
OCOURSE ENGINE Ex⊁ÉCIED DELTA-V= 14.98	5 DOF=1.000	CAPARJLITY=	3R.404	· · · · · · · · · · · · · · · · · · ·		TOTAL WEIGHT=	31.501	
NALTY SUMMATION INSUF MIDCOURSE I EXCESSIVE TGT. M	TLITIES FUEL= .0.0 TSS = 0.0	1039	ASTRIONICS= SPACFCRAFT=	WEIGHT 377-392 1622-608				
UNREN.IABILITY ASIRIONICS TOTAL		4108 5000	T0TAL=	2000-0002	PENALTY (MOD	E 3)=	377.39202	

OMPONENTS				a and an analysis is the set of the	a service family and		o tan angkang a taka taka a tang atan tan ta	and the second of the second	And a second a second and and and a second as
ACCELEROM.= A	RMA D-4E	RMA D-4E	ARMA D-4	E GYPOSCI	0PES= 66	334-A	GG 334-A	GG 334-A	e gellenter denne af endere a bed abb
U DATA (HORIZONTAL DESI	GN NUMBER	(MUMITO		CYCLES= 3	ON TIME	(HR) =	2+517		a A an
0UTSIJE DIMENS LENGTH= 0. #10TH= 10. HELGMT= 5.	51 ONS 350 450 450	WE1GHT 3LOCK= 8ASE= COVER=	8.703 4.549 2.867	WEIGH INSULATION= ELECTRONICS= COMPONENTS=	1 10-00 6-000	000		EXCIT.ENERGY= EXCIT.POWER = TOTAL P.FAIL= TOTAL WEIGHT=	109.511 43.500 03597 33.463
IU THERMAL AMALYSIS	MAX.HEATER MIN.HEATER	POWER= POWER=	97.875 000	MAX.TH MIN.TH	ERMAL CONT	B B • 0	:.1750 .0875	TOTAL ENERGY= TOTAL POWER =	176•711 141•375
ITENNA DESIGN GAIN	(0B)= 23.6	ERT (XX) II	3•36	POINTING T	יר• (הבפי) =	= 1.084	· ····································	TOTAL WEIGHT=	14.615
HSYSTEM PAHAMETERS COMPUTERS SKT RUK-2 TIME= 2.517 CYCLES= 3	STAR TRCKR ITT-LUN.UB 10.849	SUN SENSUR ADCL-1402 10_449	I SU/C.F		R. HORIZ	• SEN• ИЕ 0•000	XMITTER Designed 0.000		
ENERGY= 226.575 POWER= 90.000 P.FAll= 000640 WEIGHT= 36.000	86.793 8.009 8.00411 7.000	54-246 5.000 5.000 2.000		000 33603.6 000 3.5 000 003 000 3.1	000 000 000	0.000 0.000 0.000 0.000	0.0000 36.536 0.00000 3.727	TOTAL ENERGY= TOTAL POWER = TOTAL P.FAIL= TOTAL WEIGHT=	33971.231 143.036 .10654 51.827
TITUDE CONTROL SYSTEM	ANALYSIS		CYCI.ES=	~	ON TIME	r (HR) =	10.849		
THRUST SIZING (LH) NULL NUAR PRES= .0000	YAW - DOOD	PITCH		FUEL	CONSUMPTI( _L	DN (L8-5 YAW	SEC) PITCH		
FT.IMPACT=         .0000           ANFUVERS         .0009           IUCOURSE         0.0000	.0000 .0099 .3778	•0001 •0099 •3778	SEA OEA MAR	(RCHING= 4. () BAND= .	0585 2 1310 2614	3060 4445 1547	•0031 •2560 •1917	TOTAL ENERGY=	108.492
AX.THRUST=3778 0. OF FIRINGS= 10	.3779 .33 TOTA	.3778 . IMPULSE=	7.8069	AL IMP= 4.	4509 2. _ WEIGHT=	.905ć .1394	.4508 +10	TOTAL POWER = TOTAL P.FAIL= TOTAL WEIGHT=	10.000 .00065 21.218
ERGY SOURCE DATA	TOTAL POWE	3= 294.41	1 101	AL ENERGY= 34	256.434			TOTAL WEIGHT=	114.772
RIN6.								TOTAL WEIGHT=	110.000
DCOURSE ENGINE DELTA	14.9	35 DOF=1.00	DO CAPARIL	.ITY= 37.43	с. С			TOTAL WEIGHT=	31+219
NALIY SUMMATION INSU Exce UMM2 ASTH	PROGA JF.MIDCOURSE SSIVE TGT. I ELIAHILITY	3ILITIES FUEL= MISS = 0.	01251 00000 13924 12000	ASTRION SPACECR TO	WE161 ME161 AFT=, 162 TAL= 200	HT 7.114 2.886 0.000			
						L.	ENALTY (MODE	3)=	377.11365

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			~ 01	109.511	43.500 •03597 33.463	176°711 141°375	14°615			971•231 143•036	•10654 51•827	альна У Мина — намири - т		9.407 102.064	• 000001	29•019	114.568	110-000	31.143	AACEA A	
			GG 334-A	FXCTT FNFDGY=	EXCLIPTOWER = TOTAL P.FAIL= TOTAL WEIGHT=	TOTAL ENERGY= TOTAL POWER =	TOTAL WEIGHT=			TOTAL ENERGY= 33 TOTAL POWER =	TOTAL P.FAIL= Total Weight=		EQUIREMENTS	TOTAL POWER = Total Energy=	TOTAL P.FAIL=	TOTAL WEIGHT=	TOTAL WEIGHT=	TOTAL WEIGHT=	TOTAL WEIGHT=	33	
•			66 334-A = 2.517			2.1750 1.0875	84	XMITTER Designed	000 • 0	0.000 36.536	0.00000	= 10.849	R		1		A CONTRACT OF A CO			DENNITY WOD	
			S= 66 334-A		1.345 10.000 6.000	AL COND.= AL COND.=	.(DEG.)= 1.0	HORIZ • SEN• NONE	000.0	0.000	0.00000	ON TIME (HR)					0.005			WEIGHT WEIGHT S= 1615.367 L= 2000.000	
			GYR0SCOPE		NSULATION= ECTPONICS= OMPONENTS=	MAX.THERN MIN.THERN	POINTING TOL.	COM . RCVR. MCP-503	9601.033	33603.617 3.500	.09336	•					ENER6Y= 3425		= 37.173	ASTRIONIC Spacecraf Total	SEC.)
	· · · · · · · · · · · · · · · · · · ·		ARMA D-4E CYCI		8.703 II 4.549 ELI 2.867 C	97.875 000	: 3.36	R ISUZC.P.S. NONE	000.0	0.000	0.0000	SYCLES= 2					19 TOTAL		10 CAPABILITY	01314 00000 13858 15000	APSED= .160(
	SCHEDUR F		ARMA D-4E		BLOCK= BASE= COVER=	ER POWER= Er Power=	с Евн (км) =	R SUN SENSUE R ADCI-1402	0 10 B44	3 54.246	0001	0		• 007			WER= 293.81		•985 DOF=]•00	RABILITIES SE FUFL= • MISS = 0 TAL =	n= 30.48+ EL
- - - - -	31		- ARMA D-4E		10.450 5.450 5.450	MAX.HEAT	11N(DB)= 23*1	STAR TRCK	17 10.84	75 86.79	00+1 7-00	ANALYSIS		)CITY = 		NULLIY=	TOTAL PO		:LTA-V= 14	PRO EXCESSIVE -MIDCOUR EXCESSIVE TGI JNRELLABILLITY ASTRIONICS TO	RT= 30.32+ EN
	DEMALTY IN	NENTS	ACCELERDH.=		0013105 018 LENGTH= wIDTH= HE40H]=	IAL ANALYSIS	IÉSIGN GA	I PARAMETERS COMPUTERS SHT RUK-2		1= 226.57 1= 226.57 1= 90.00		NDE CONTROL	911711	R MAX. VELO MAX. MAN.	ri TE TOR MESSIUNT TOR	TORENTON CAPA	JUPCE UATA		ENGINE Expected df	SUMMATION 5	N TIMESE STA
		15U COMPO	TAU UST			ISU ТНЕНМ	ANTENNA D	SCHSYSTEM	TIME	CTCLFS FDERGY POWER	P.FAIL WEIGHT	CHG ATTIT	SYSTEM S	MANEUVE	METEORI Solabo	TUTAL I	FNERGY SC	MIRING	MIDCOURSE	PENALTY	EXECUTION

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	4 GG 334-A	amana ama a sina ana ang ang ang ang ang ang ang ang a	EXCIT.ENERGY= 109.511 EXCIT.POWER = 43.500 TOTAL P.FAIL= 03597 TOTAL WEIGHT= 33.463	TOTAL ENERGY= 176.711 TOTAL POWER = 141.375	T01AL WEIGHT= 14.615	TOTAL ENERGY= 33971.231 TOTAL ENERGY= 33971.231 TOTAL POWER = 143.036 TOTAL P.FAIL= .10654 TOTAL WEIGHT= 51.827	DEDUTAEMENTS	TOTAL POWER = 14.978 TOTAL ENERGY= 162.504	TOTAL P.FAIL= .000618	TOTAL WEIGHT= 30.422	TOTAL WEIGHT= 116.489	TOTAL WEIGHT= 110.000	TOTAL WEIGHT= 31.216		UE 3)= 388.03198
, an ∕ sa t, s <mark>inna</mark> at, an	66 334-1	= 2.517		<b>?.1750</b> 1.0875	84	XMITTER DESIGNED 0.000 0.000 36.536 3.727	= 10.849								PENALTY (MOI
na sabah at sama si a si si si si sa da ba da si san i san i san i san si	s= 66 334-A	N TIME (HK):	1.345 10.000 6.000	VL COND.=	(DEG.)= 1.0	HOR1Z.SEN. NONE 0.000 0.000 0.000 0.0000	N TIME (HR):			f	.446			WEIGHT = 388.032 = 1611.968 = 2000.000	
ցուցյունը նունեցնությունը գործը ենքը, ու ու ու ուներիները է հետումությունը։	GYROSCOPES	S= 3	WEIGHT SULATION= CTRONICS= APONENTS=	MAX.THERM	INTING TOL.	COM. RCVR. MCR-503 9001.033 33603.617 33500 09336 3.100					VEPGY= 34310		37.425	ASTRIONICS: SPACECRAFT: TOTAL:	
	ARMA D-4E	CYCL	703 INS 549 ELFC	7-875 000	3•36 P(	ISU/C+P-S- NONE 0.000 0.000 0.000 0.000	CYCLES= 2			-	TOTAL E		CAPABILITY=	1253 0000 1221	SED= .160 (SI
SCHEDULE	18MA 0-4E	(MUMITOO	WEIGHT BLOCK= A BASE= 4.	POWER= 9. POWER=	ERP (KW) =	SUN SENSOR ADCL-1402 10,849 54,246 54,246 54,000 .00410 2,000	(SIS	)07 346	177 122 (10	38] 169 169	1 299.390	s	35 DOF=1.000	SILI IES FUEL= •0 MISS = 0•0	•1 30-65• ЕLAP
3)	RMA 0-4E A	GN NUMBER	10NS 350 H 450 A 450 C	MAX.HEATER MIN.HEATER	()B)= 23.6	STAR TRCKR ITT-LUN.0A 10.849 86.793 8.000 .00411 7.000	ONTROL ANALY	.≺ 	10000 11111 1111		TOTAL POWEN		14°36	PROBAF DF.MIDCOURSE SSIVE TGT. N LIABILITY	
ENALIY (MODE	TS CELEROM.= A	IZUNTAL DEST	TSIDE DIMENS WGTH= 9. L'DTH= 10. LGH1= 5.	ANALYSIS	9NI 6AIN(	HAMETERS CUMPUTERS SHE HUK-2 2.517 2.517 2.515 99,000 .00049 36.000	L ATTITUDE C	MAX. VELOCIT MAX. MAN. TO	FINAL TORODE SURETORQUE	MIS. TORQUE FINAL NTUM CAPARIL	E ÜATA		GINE PECTED DELTA	ATION INSU EXCE UMPE	MELS E START
Ч	COMPUNEN	UATA (HUK)	HE LOU	THERMAL /	ENNA DEST	SYSTEN PAU TIME CYCLES= CYCLES= FUEKGY= POWER= WEIGHT=	RTIA WHEEL	ANEUVER 1	ETEORLTE OLAR PRESS	UCOURSE ( UTAL MURE	ΡάΥ SUUKCt	I NG	COURSE EN	ALTY SUMA	CUTION IN

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	66 334-A	annan an an Arthur ann an Arthur an Arthu	IT.ENER6Y= 109.511 IT.POWER = 43.500 AL P.FAIL= 03597 AL WEIGHT= 33.463	AL ENERGY= 176.711 AL POWER = 141.375	AL WEIGHT= 20.980	AL ENERGY= 33971.231 AL POWER = 126.500 AL P.FAIL= 126.500 AL WEIGHT= 58.100		AL ENERGY= 108.492 AL POWER = 10.000 AL P.FAIL= 00278 AI WFIGHT= 21.215	AL WEIGHT= 109.067	AL WEIGHT= 110.000	AL WEIGHT= 31.501		384.32558
	-A GG 334-A	8)= 2.517	EXC EXC 101 101	2.1750 TOT. 1.0875 TOT	101 101	<pre>XMITTER TEST XMITER TEST XMITR TEST XMITP 0.000 0 0 0 0 0 0 0 0 0 0 0 0 0 0 0 0</pre>	-B-SEC) PITCH		101	101	T01	040	PENALTY (MOUE 3)
	SCOPES= 66 334-	ON TIME (HE	IGHT 1.345 Nu= 1.345 SS= 10.000 SS= 6.000	THERMAL COND.= THERMAL COND.=	3 TOL. (DEG.) =	3CVR. HORIZ.SEN. 3.3 1.033 1.033 1.033 1.033 1.0000 1.00000 1.0000 1.0000 1.00000 1.00000 1.0000 1.00	EL CONSUMPTION (1 ROLL YAW	4.0585 2.3060 0.001 .4449 .2793 .1544 4.3379 2.9052	34256.434		• 404	WEIGHT IONICS= 384.32 ECPAFT= 1615.67 TOTAL= 2000.00	
	A D-4E GYRC	CYCLES= 3	MEI MEINSULATIC ELECTRONIC	75 MAX. 00 MIN.	3.36 POINTING	ULC.P.S. COM. NUNE MCH-50 0.0000 33603 0.0000 33603 0.0000 33603	FUB	SEARCHING= UEAU BAND= MANEUVFRS= TOTAL IMP=	TOTAL ENERGY=	÷	PARILITY= 38.	9 ASTR 0 8 SPACI 0	=10.740(SEC.)
	ARMA U-4E ARM	(MUMITHO 4	WEJGHT R.70 BLOCK= R.70 BASE= 4.54 COVER= 2.86	POWER= 97.8	ERP (KW) =	SUN SENSOR IS ADCL-1402 10.849 24.245 54.245 5000 2,000 2,000 2,000 2,000	PITCH	• 0000 • 0001 • 0099 • 3778 • 3778	c = 277.875	- 	985 DOF=1.000 CA	APILITIES FUEL= 0.000 MISS = 0.0000 AL = 1500 AL = 1500	= 41.39, ELAPSED
(MODE _ 0)	ARMA D-4E	DESTGN NUMBER	IMENSTONS 9.350 10.450 5.450	MIN.HEATER MIN.HEATER	341N (08) = 26+2	STAR TRCKR 25 STAR TRCKR 217 10.949 33 86.793 575 86.793 200 0.00411 200 7.000 575 ANALYSTS	(LH) үдм	000 0000 000 0000 000 0000 000 0000 8775 000 8775 000	TOTAL POWE		DELTA-V= 14.5	PROB/ PROB/ EXCESSIVE TGI. UNRELIABILITY ASTRIONICS TOTA	ART= 39.65. END:
COMPONENTS	ACCELÉRON.	υάτα (θυθ1 ΖΟΝΤΑΓ	OUTSIDE D LENGTH= MIDTH= hELGMT=	THERMAL ANALYSI	NNA DESTGN	YSTEM PANAMETER COMPUTEI SRT RUM SRT RUM SRT RUM SRT RUM SRT 2.4 YCLES= POWER= POWER= POWER= POWER= POWER= FALL= S6.1	THRUST SIZING MULL	AR PRES= 0 • TMPACT= 0 • UVECS = 0 • UVES = 0 • THRUST= 0 • OF	GY SUUHCE DATA	۲۹ (۲۰۰۰ - ۲۰۰۰) د. ا	OURSE ENGINE EXPECTED I	ALTY SUNMATION	SUTION TIPES. ST

ISU COMPONENTS				• •					
ACCELEROM.=	ARMA D-4F	ARMA U-4E	AHMA D-4E	GYROSCOPES:	= 66 334=A	66 334-A	GG 334-A		
ESU DATA (HURIZUNTAL DES	IGN NUMMER	(MUM [ 140 4	כ אכו	LES= 3 Of	V TIME (HR) =	2.517	a na sa	ren Markan - Alexandro - A. and provide and definitions	
001510E DIMEN LENGTH= 7 4101H= 16 HE1GH1= 9	JS 1 ONS • 350 • 450	WEIGHT BLOCK= BLOCK= ASSE= COVER= 2	3.703 II 1.549 ELI	WEIGHT NSULATION= ECTPONICS= DMPONENTS=	1.345 10.000 6.000		EXCIT.ENERGY= EXCIT.POWER = TOTAL P.FAIL= TOTAL WEIGHT=	109.511 43.500 03597 33.463	- <b>-</b>
ISU THERPAL ANALYSIS	MAX.HEATE! MIN.HEATE!	R POWER= C	97.675 000	MAX.THERMAI MIN.THERMAI	L COND.= 2 L COND.= 1	.1750 .0875	TOTAL ENERGY= Total Power =	176.711 141.375	
ANTENNA DESIGN GAIN	1(08)= 26.2	EPP (KW) =	3•36	POINTING TOL. (1	DEG.)= .802		TOTAL WEIGHT=	20.980	
SURSYSTEM PARAMETERS COMPUTERS SKI HUK-2 SKI HUK-2 SKI HUK-2 3 CYCLES= 2.517 3 ENFRGY= 2.517 90,000 P.FAIL= .00640 WLIGHI= 30,000	STAR TRCKR 1TT-LUN.09 10.849 85.793 85.793 8.000 7.00411	SUN SENSOR AUCL-1402 10.849 54.745 54.745 54.745 2.000 2.000	ISU/C.P.S. NONE 0.000 0.000 0.000 0.000 0.000	COM+ RCVR+ MCR+503 9601.033 33603.617 3.500 .09336 3.100	HORIZ.SEN. HONE 0.000 0.000 0.0000 0.0000	EST XMITTER EST XMITR 0.000 0.000 20.000 20.000 10.000	TOTAL ENERGY= 3. Total Power = 3. Total Pomer = 1 Total Weight=	3971.231 126.500 .10654 58.100	
INERTIA WHEEL ATTITUDE	CONTROL ANAL	LYSIS	CYCLES=	5 S	N TIME (HR)=	10.849	1 - -	f	*
SYSTEM SIZING			-			RE	OUIREMENTS	1 40 <b>40</b> 1 40 <b>40</b> 1 40 1 40 1 40 1 40 1 40 1 40 1 40 1	
MANEUVER MAX. VFLOCT MAX. MAN. 7 FILO	0R0.= H -	• 0 0 7 • 0 4 6 • 0 4 5					TOTAL POWER = Total Energy=	14.978 162.504	i.
METEOPITE TURGE SOLAR PRESSURE TURGE	所に 11111 11111 11111 11111 111111	• • • • • • • • • • • • • • • • • • •					TOTAL P.FAIL=	•000618	
MIUCOURSE NIS. TOROU FIN TOTAL MONEWTUM CAPABI		. 881 • 469 • 469					TOTAL WEIGHT=	30.422	
TRERGY SOUNCE DATA	TOTAL POW	ER= 282.853	3 TOTAL	ENERGY= 34310.4	446	<pre>int int interpretation.com or interpretation into</pre>	TOTAL WEIGHT=	110.784	1
, SMING						-	TOTAL WEIGHT=	110.000	•
MIUCUUPSE ENGINE Expected DEL1	.v−V= 14.	985 DOF=1.000	0 CAPABILITY	= 37.425			TOTAL WEIGHT=	31.216	
PEMALTY SUMMATION IN Exc UNI ASY	PROR SUF.MIDCOURS CESSIVE TGT. SEL LABILITY FUIONICS TOT	ABILITIÉS E FUEL= •( MISS = ••	01253 00000 13921 15000	ASTRIONICS= SPACECRAFT= TOTAL=	WEIGHT 394.966 1605.034 2000.000				
EXECUTION TIMESF STARF	= 41.75, END	= 41.92. ELA	PSED= .162(	5EC.)	a.	ENALTY (MOI)E	3) =	394 <b>.</b> 96554	
FIGURE	36. JUPIT	ER FLYBY PE	NALTY EVALU.	ATION, INERTI	A WHEELS, SP	ECIFIED TR	ANSMI TTER		

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## <u>Multiple Midcourse Strategies</u>

## Review of Candidate Schemes

In accordance with the statement of work, six reports, References 25 through 30 were reviewed as sources of possible midcourse correction policies. An additional paper by C. G. Pfeiffer (Reference 31) was also considered. A description of each of these papers follows:

<u>Breakwell, Rauch, and Tung</u>. Breakwell, Rauch, and Tung (Reference 27) derive a correction scheme that minimizes expected fuel subject to a fixed meansquare value of the target miss. In the absence of engine mechanization errors, the optimal control policy is shown to be continuous. The times at which the midcourse propulsion system is turned on or off are determined by solving the continuous control problem using the maximum principle. If engine mechanization errors are not neglected, the optimal control policy is shown to consist of four or five impulsive corrections. The optimal spacing and magnitudes of these corrections are determined by a dynamic programming analysis.

In both of the above cases, the magnitude of the corrective thrust is proportional to the predicted deviation from the desired state at the final time. Periodic measurements of the spacecraft's state are made throughout the mission, and an update of the estimated miss is made at each observation time using the best predicted estimate and the new set of measurements just received. The observations are assumed to be made at regular intervals of time  $\Delta t$ , where  $\Delta t$  is a relatively short interval of time in comparison to the total mission time, T. Typically, for a 402-day mission to Mars, measurements are made as frequently as 1 per minute. The assumption enables the authors to utilize the continuous form of the Kalman estimation procedure.

This method of determining the optimal correction strategy is compatible with the first and third penalty modes in which a certain probability of mission failure attributable to astrionics is acceptable and the astrionics system weight is used as the penalty function. This procedure, however, is not completely adaptable, because a detailed computational procedure is established only for the special case where the rms value of one component of the state vector at the terminal time is specified.

Stern and Potter. Stern and Potter (Reference 28) present a procedure for determining a midcourse correction schedule which is optimal in the sense that total velocity correction is minimized. This is accomplished by selecting the time(s) of correction so as to maximize the miss correctible per unit of velocity correction. The principal objection to this method is that no consideration is given to the uncertainties of the navigational measurements. It is assumed that a sufficient number of measurements has been made prior to the correction so that the uncertainty in the predicted miss distance at the target is negligible.

<u>Battin</u>. Battin (Reference 29) suggests a procedure for determining appropriate times for making observations and/or velocity corrections. The procedure assumes that a set of points in time have been selected prior to the analysis. At each of these points, one of three alternative cources of action is followed:

- (1) A single observation is made
- (2) A velocity correction is implemented
- (3) No action is taken.

A measurement is made if a significant reduction in the potential miss distance would result from the measurement and updating of information. A velocity correction is made whenever the ratio of the uncertainty in the estimate to the standard deviation of the required correction is less than a fixed constant. Then an engine restart and propellant expenditure is warranted.

In the final analysis, however, the criteria for selecting times of midcourse corrections are disregarded, and a correction schedule is specified. Hence, this procedure is essentially a policy for determining an appropriate measurement schedule consisting of the times at which observations are made and the choice of the best celestial measurement to be made at this time.

Denham and Speyer. In the paper by Denham and Speyer (Reference 30), the authors present a method of varying the times of midcourse velocity corrections to minimize a function involving the terminal dispersion and a statistical measure of the total velocity change used for control. The procedure assumes a nominal measurement and correction program has been specified and through successive trials and adjustments the correction program improves the given schedule. For example, the program suggested by Battin (Reference 29) is taken as the nominal, and a 10 percent improvement in the rms uncertainty of the terminal position is realized.

Although an analytical technique for optimizing both the continuous measurement and feedback gain programs is developed, only the means of determining the optimal measurement sequence is demonstrated by an example.

<u>Pfeiffer</u>. An additional paper by C. G. Pfeiffer (Reference 31), not included among the proposed references, was reviewed as a means of determining a midcourse correction schedule. The objective discussed in this paper was to develop a guidance policy that minimizes the expected value of the target error squared, subject to the constraint that the total propellant expended in performing the corrections is less than some prespecified amount. The analysis is based on the assumption that a measurement policy, independent of the guidance policy, has been prespecified. Also, a set of points along the trajectory at which the possibility of performing a correction is to be examined must be selected prior to the analysis. The index of performance to be minimized at any decision time involves the sum of two terms. The first term is the expected value of the target uncertainty immediately after the final correction which occurs

at some prespecified time  $t_f$ . The second term is the square of the uncorrectible error due to the depletion of the correction capability prior to the final correction time  $t_f$ .

The determination of an optimal guidance policy based on this performance index falls within the category of problems subject to analysis by Dynamic Programming. As in all applications of the Dynamic Programming algorithm, one is always mindful of the "Curse of Dimensionality". It is shown in this paper that a considerable simplification is realized if the following restrictions are imposed upon the guidance policy:

- At each decision time, t<sub>i</sub>, either no correction or total correction is to be accomplished.
- At each decision time, t<sub>i</sub>, at most two corrections will be accomplished: one at the final decision time, t<sub>f</sub>, and another at some time t<sub>i</sub> < t<sub>f</sub>.

The solution obtained by application of this method will not necessarily yield an extreme value (max or mim) for the penalty considered. As indicated by Pfeiffer, the effect of imposing the two constraints is not entirely known, however, these assumptions result in a greatly simplified computational algorithm.

The validity of the constrained Dynamic Programming Algorithm outlined above is independent of the particular performance index chosen. Given an arbitrary performance measure, the same procedure can be used to obtain a correction schedule if the "two-correction" and "total-correction" constraints are imposed upon the guidance policy. Therefore, the penalty function used to evaluate the guidance system, i.e., astrionics system weight or probability of failure can be considered as a measure of performance for the midcourse correction policy. The algorithm can then be used to obtain the correction policy based on the revised penalty.

### Minimization of Expected Target Miss

Midcourse correction  $\Delta V$  may be determined from the computed deviations of position and velocity prior to the midcourse and the state transition matrix mapping errors at midcourse to errors at the target (Reference 4). The vectors and matrices of interest are defined in Table XIII.

Two types of target miss requirements have been studied. They are:

- (1) Zero all position deviations at the target.
- (2) Zero one position or velocity component deviation at the target.

When making a midcourse correction, three variables may be specified (three components of  $\Delta V$ ). Thus, up to three conditions may be satisfied at the target. Zeroing all position deviations specifies three conditions.

Symbol	Definition
[Φ]	The 6 by 6 state transition matrix from midcourse to target
<u>e, d</u> , <u>d</u> c	The 6 element error, deviation, and computed deviation vectors
<u>×</u> m	The state of the vector $\underline{\mathbf{x}}$ prior to midcourse correction
$\underline{x}_{m}^{+}$	The state of the vector $\underline{x}$ after midcourse correction
<u>×</u> t	The state of the vector $\underline{x}$ at the target
<u>×</u> ap	The 3 element position error vector subset of any 6 element vector $\underline{x}_a$
<u>×</u> av	The 3 element velocity subset of $\frac{x}{a}$
$\Delta \mathbf{V}$	The midcourse correction vector
[P <sub>d</sub> ]	The deviation covariance matrix $E[\underline{d}_{m} \ \underline{d}_{m}^{T}]^{**}$
	The error covariance matrix $E\left[\underline{e}_{m} \underline{e}_{m}^{T}\right]$
[P <sub>dc</sub> ]	The computed deviation covariance matrix $E[\underline{d}_{cm} \ \underline{d}_{cm}^T]$
[P <sub>ed</sub> ]	The error, deviation covariance matrix $E[\underline{e}_m \ \underline{d}_m^T]$

# TABLE XIII. MIDCOURSE CORRECTION DEFINITIONS

See Reference 4 for definition of these vectors.

\*\* E denotes expected value.

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In another case, however, zeroing any one component at the target specifies only one condition. In other words, there are an infinite number of possible corrections which will zero one component of the deviation at the target. In this case, it is desirable to make the correction that uses the least fuel (minimum magnitude of  $\Delta V$ ).

In either case, midcourse correction is computed by

$$\underline{\Delta}V = [D] \underline{d}_{cm} = [D] [\underline{e}_{m} + \underline{d}_{m}]$$

where D is a  $3 \times 6$  correction matrix obtained from the state transition matrix from the point of the correction to the target. The subscript m denotes vectors prior to the midcourse. If all three computed position deviations are to be zeroed at the target, then

$$[D] = - \left[ \Phi_{pv} \right]^{-1} \left[ \Phi_{p} \right]$$

where the 3 by 6 matrix  $[\Phi_p]$  and the 3 by 3 matrix  $[\Phi_{pv}]$  are partitions of the state transition matrix  $[\Phi]$  defined as follows

$\begin{bmatrix} \bar{\Phi} \end{bmatrix} = \begin{bmatrix} \Phi \\ -P \\ \Phi \\ \Psi \end{bmatrix} = \begin{bmatrix} \Phi \\ -P \\ \Phi \\ \Psi \\ \Psi \end{bmatrix}$	
--	--

If only one component of the computed position or velocity deviation is to be nulled at the target, for example,  $(\underline{d}_{ct})_i$ , then the minimum magnitude  $\underline{\Delta V}$  is determined by the matrix

[D]	=	$\begin{bmatrix} \underline{\Psi_{i}} \\ \underline{\Psi_{i}} \\ \underline{\Psi_{i}} \end{bmatrix}$
		$(\Psi_{iv}^{T}\Psi_{iv})$

where  $\underline{\varphi}_i$  is the 6 element vector whose transpose is equivalent to the i<sup>th</sup> row of the state transition matrix  $\Phi$  and  $\underline{\varphi}_i$  is the 3 element velocity subset of  $\underline{\varphi}_i$ .

In the preceding argument, the matrix [D] is chosen so as to null either one or all computed deviations in position at the target. Since the deviation is actually a random vector, it would be meaningful to base the selection of [D] on some statistical property of the target deviation.

Consider first the case where the trace of the covariance matrix of deviations in target position is to be minimized. The 3 element vector of deviations in position at the target can be written as

$$\underline{\mathbf{d}}_{tp} = [\Phi_{p}]\underline{\mathbf{d}}_{m} + [B][\underline{\mathbf{e}}_{m} + \underline{\mathbf{d}}_{m}]$$

where [B] is the 3 by 6 matrix defined by

$$\begin{bmatrix} B \end{bmatrix} = \begin{bmatrix} \Phi \\ pv \end{bmatrix} \begin{bmatrix} D' \end{bmatrix}$$

[D'] is a statistically determined correction matrix, such that

$$\underline{\Delta V} = [D']_{d_{cm}}$$

Let  $\left[ \Phi_{p} \right]$  and  $\left[ B \right]$  be partitioned into sets of column vectors as follows:

$$\begin{bmatrix} \Phi_{\mathbf{p}} \end{bmatrix} = \begin{bmatrix} \Psi_{\mathbf{1}}^{\mathrm{T}} \\ \Psi_{\mathbf{2}}^{\mathrm{T}} \\ \Psi_{\mathbf{3}}^{\mathrm{T}} \end{bmatrix} , \quad \begin{bmatrix} B \end{bmatrix} = \begin{bmatrix} \overline{b}_{\mathbf{1}}^{\mathrm{T}} \\ \underline{b}_{\mathbf{1}}^{\mathrm{T}} \\ \underline{b}_{\mathbf{2}}^{\mathrm{T}} \\ \underline{b}_{\mathbf{3}}^{\mathrm{T}} \end{bmatrix}$$

The 3 by 3 covariance matrix of position deviations at the target, i.e.,

$$[C] = E[\underline{d}_{tp} \, \underline{d}_{tp}^{T}]$$

may be expressed by

$$[\mathbf{c}]_{ij} = \underline{\varphi}_{i}^{T}[\mathbf{P}_{d}]\underline{\varphi}_{j} + \underline{b}_{i}^{T}[\mathbf{P}_{dc}]\underline{b}_{j} + \underline{b}_{i}^{T}[\mathbf{P}_{d} + \mathbf{P}_{ed}]\underline{\varphi}_{j}$$
$$+ \underline{b}_{j}^{T}[\mathbf{P}_{d} + \mathbf{P}_{ed}]\underline{\varphi}_{i} \qquad (32)$$

The trace of [C], tr[C], is

$$\operatorname{tr}[\mathbf{C}] = \sum_{i=1}^{3} \varphi_{i}^{T}[\mathbf{P}_{d}] \varphi_{i} + \underline{\mathbf{b}}_{i}^{T}[\mathbf{P}_{dc}] \underline{\mathbf{b}}_{i} + 2\underline{\mathbf{b}}_{i}^{T}[\mathbf{P}_{d} + \mathbf{P}_{ed}] \varphi_{i}$$

If tr [C] is minimized with respect to  $\underline{b}_i$  (i = 1, 2, 3), then the solution for the  $\underline{b}_i$ 's is

$$\underline{b}_{i} = - [P_{dc}]^{-1} [P_{d} + P_{ed}] \underline{\phi}_{i} \qquad i = 1, 2, 3$$

or

$$[B] = [\Phi_p] [P_d + P_{ed}^T] [P_{dc}]^{-1}$$

and

$$\begin{bmatrix} D' \end{bmatrix} = \begin{bmatrix} \Phi_{pv} \end{bmatrix}^{-1} \begin{bmatrix} \Phi_{p} \end{bmatrix} \begin{bmatrix} P_{d} + P_{ed} \end{bmatrix}^{T} \begin{bmatrix} P_{dc} \end{bmatrix}^{-1} ,$$

or

$$\begin{bmatrix} D' \end{bmatrix} = \begin{bmatrix} D \end{bmatrix} \begin{bmatrix} P_d + P_{ed}^T \end{bmatrix} \begin{bmatrix} P_{dc} \end{bmatrix}^{-1}$$

In the case where the variance of any one component of the target deviation is to be minimized, the desired midcourse correction matrix [D'] is partially defined by the 6-element vector:

$$\underline{\mathbf{b}}_{\mathbf{i}} = [\mathbf{D}']^{\mathrm{T}}_{\underline{\boldsymbol{\phi}}_{\mathbf{i}\mathbf{v}}} = - [\mathbf{P}_{\mathrm{dc}}]^{-1} [\mathbf{P}_{\mathrm{d}} + \mathbf{P}_{\mathrm{ed}}]_{\underline{\boldsymbol{\phi}}_{\mathbf{i}}}$$

Thus,

$$\underline{\varphi}_{iv}^{T} [D'] \underline{d}_{cm} = - [P_{dc}]^{-1} [P_{d} + P_{ed}^{T}] \underline{\varphi}_{i}^{T} \underline{d}_{cm}$$

The above equation defines a family of vectors  $\Delta V$  lying in a plane as shown in Figure 37. The  $\Delta V$  with minimum magnitude is the  $\Delta V$  normal to this plane, or

$$\underline{\Delta V} = [D']\underline{d}_{cm} = \frac{\underline{\varphi_{iv}}\underline{\varphi_{i}}^{T}[P_{d} + P_{ed}^{T}][P_{dc}]^{-1}}{(\underline{\varphi_{iv}}^{T}\underline{\varphi_{iv}})} \underline{d}_{cm} \qquad (34)$$

Thus

$$[D'] = [D][P_d + P_{ed}^T][P_{dc}]^{-1}$$

In both of the above cases, the correction matrix [D] which minimizes the uncertainty in the deviation at the target is determined by post-multiplying the deterministic correction matrix, [D], by

$$[P_{d} + P_{ed}^{T}][P_{dc}]^{-1}$$

The above equations apply only in the case where it is assumed that no errors are generated in making the midcourse. In order to include the effect of velocity errors generated in executing the midcourse maneuver, let it be assumed that the error is a linear function of the midcourse correction vector,  $\underline{\Delta} \mathbf{e}_{\mathbf{v}} = [\mathbf{A}] \underline{\Delta} \mathbf{V}$ . 112

(33)



FIGURE 37. MINIMUM FUEL  $\Delta V$  to minimize variance of target deviation

Thus

$$\underline{\mathbf{e}}_{m}^{+} = \underline{\mathbf{e}}_{m} + \begin{bmatrix} \mathbf{0} \\ \Delta \mathbf{e}_{v} \end{bmatrix} , \text{ and}$$
$$\underline{\mathbf{d}}_{m}^{+} = \underline{\mathbf{d}}_{m} + \begin{bmatrix} \mathbf{0} \\ \Delta \mathbf{v} \end{bmatrix} - \begin{bmatrix} \mathbf{0} \\ [\Delta \mathbf{e}_{v}] \end{bmatrix}$$

Also, assume that the generated error,  $\underline{\Delta}e_v$ , is uncorrelated with the deviation,  $d_m$ , or the error,  $e_m$ . Then the 3 by 3 covariance matrix of position deviations at the target can be written as:

$$\begin{bmatrix} C_{e} \end{bmatrix} = \begin{bmatrix} C \end{bmatrix} + \begin{bmatrix} \varphi_{pv} \end{bmatrix} \begin{bmatrix} A \end{bmatrix} \begin{bmatrix} D_{e} \end{bmatrix} \begin{bmatrix} P_{dc} \end{bmatrix} \begin{bmatrix} D_{e} \end{bmatrix}^{T} \begin{bmatrix} A \end{bmatrix}^{T} \begin{bmatrix} \varphi_{pv} \end{bmatrix}^{T}$$

where  $D_e$  is the unknown correction matrix to be determined when execution errors are not neglected and [C] is the 3 by 3 covariance matrix defined by Equation (32) with [B] =  $[\phi_{pv}][D_e]$ .

Define a 3 by 3 matrix, M, as follows:

$$[M] = [\phi_{pv}][A][\phi_{pv}]^{-1} .$$
(35)

Consequently,  $[C_{\rho}]$  can be written as

$$[C_{e}] = [C] + [M][B][P_{dc}][B]^{T}[M]^{T}$$

The trace of  $[C_e]$ , tr $[C_e]$ , is

$$tr \begin{bmatrix} C_e \end{bmatrix} = \sum_{i=1}^{3} \begin{bmatrix} C_i \end{bmatrix} + \sum_{i=1}^{3} \sum_{j=1}^{3} \begin{bmatrix} m_{j} \underline{b} \end{bmatrix}_{j}^{T} \begin{bmatrix} P_{dc} \end{bmatrix} \underline{b}_{k} \underline{m}_{ik}$$

where  $m_{11} = [M]_{11}$ . Interchanging the order of summation, tr  $[C_e]$  can be written as

$$\operatorname{tr} \left[ \begin{array}{c} C_{e} \end{array} \right] = \begin{array}{c} 3 \\ \Sigma \left[ \begin{array}{c} C \end{array} \right]_{i=1} \end{array} + \begin{array}{c} 3 \\ \Sigma \\ j, k=1 \end{array} \begin{array}{c} \Sigma \\ i=1 \end{array} \left[ \begin{array}{c} m_{i} \\ j \end{array} \right]_{i=1} \end{array} \left[ \begin{array}{c} \Sigma \\ m_{i} \\ j \end{array} \right]_{i=1} \left[ \begin{array}{c} m_{i} \\ m_{i} \\ j \end{array} \right]_{i=1} \left[ \begin{array}{c} T_{e} \\ T_{e} \\ j \end{array} \right]_{i=1} \left[ \begin{array}{c} T_{e} \\ T_{e} \\ j \end{array} \right]_{i=1} \left[ \begin{array}{c} T_{e} \\ T_{e} \\ j \end{array} \right]_{i=1} \left[ \begin{array}{c} T_{e} \\ T_{T$$

Let

$$\sum_{i=1}^{3} m_{ij} m_{ik} = n_{jk} = [M^{T}M]_{jk}$$

Then

$$tr [C_e] = \sum_{i=1}^{3} \underline{\varphi_i}^{T} [P_d] \underline{\varphi_i} + \underline{b_i}^{T} [P_{dc}] \underline{b_i} + 2\underline{b_i}^{T} [P_d + P_{ed}] \underline{\varphi_i}$$
$$+ \sum_{i,k=1}^{3} \underline{b_i}^{T} [P_{dc}] \underline{b_k} .$$

If tr  $[C_e]$  is to be a minimum with respect to  $\underline{b_i}$  (i=1,2,3), then each  $\underline{b_i}$  must satisfy the following equation:

$$[0] = 2[P_{dc}]\underline{b}_{i} + 2[P_{d} + P_{ed}]\underline{\phi}_{i} + 2\sum_{j=1}^{n} i_{j}[P_{dc}]\underline{b}_{j}$$

or in matrix form

$$\begin{bmatrix} 0 \end{bmatrix} = \begin{bmatrix} \varphi_1^T \\ \varphi_2^T \\ \varphi_3^T \end{bmatrix} \begin{bmatrix} P_d + P_{ed} \end{bmatrix}^T + \begin{bmatrix} I + N \end{bmatrix} \begin{bmatrix} b_1^T \\ b_2^T \\ b_3^T \end{bmatrix} \begin{bmatrix} P_{dc} \end{bmatrix}$$

Thus

$$[B] = -[I + M^{T}M]^{-1}[\phi_{p}][P_{d} + P_{ed}]^{T}[P_{dc}]^{-1}$$

After substitution of Equation (35) in the above equation, the following expression for  $[D_{\mu}]$  is realized:

$$[D_{A}] = [I + C^{-1}A^{T}CA]^{-1}[D^{\prime}]$$

where

$$[C] = [\phi_{pv}]^{T} [\phi_{pv}]$$

and [D'] is the correction matrix defined by Equation (33) assuming no correction errors are generated in making the midcourse maneuver.

In the case where the variance of any one component of the target deviation is to be minimized, the desired midcourse correction matrix,  $[D_e]$ , can be determined in a similar manner. If  $(\underline{d}_{ti})$  is to be minimized, then

$$[D_{e}] = (1 + \alpha_{i})^{-1} [D']$$

where  $\alpha_i$  is a scalar determined by

$$\alpha_{i} = \varphi_{iv}^{T} [A] \varphi_{iv}$$

and [D'] is the correction matrix defined by Equation (34) assuming no errors are generated during the midcourse correction.

## <u>Minimization of the Astrionics Penalty Function by a Constrained</u> <u>Dynamic Programming Analysis of Multiple Midcourse Corrections</u>

As previously discussed, in the approach suggested by Pfeiffer (Reference 31) was intended to minimize a performance index involving the expected value of the target error squared, subject to the constraint that the total propellant expended in performing the corrections is less than some prespecified amount. The exact form of the penalty does not affect the method of determining the optimal correction policy. Thus the astrionics penalty function can be substituted for the performance index considered by Pfeiffer.

The procedure for determining a suitable correction policy uses the approach by Pfeiffer (Reference 31) discussed in the previous section. It is initiated by the selection of a measurement policy, independent of the guidance policy, and a set of points along the trajectory at which the possibility of performing a correction is to be examined. The determination of an optimal guidance policy based on the selected penalty falls within the category of problems subject to analysis by Dynamic Programming. However, as indicated earlier, a considerable simplification is realized if the following restrictions are imposed upon the guidance policy:

- At each decision time t<sub>i</sub>, either no correction or a total correction is to be accomplished.
- (2) At each decision time,  $t_i$ , it is assumed that at most two corrections will be accomplished: one at the final decision time,  $t_f$ , and another at some time  $t_i < t_f$ .

If these constraints are observed, then the control policy is implemented at time t, in the following steps:

- (1) Calculate the performance index corresponding to a total correction only at  $t_f$ ,  $P_1(i)$ .
- (2) Calculate the performance index corresponding to total correction only at  $t_i$  and  $t_f$ ,  $P_2(i)$ .

- (3) If  $P_2(i) P_1(i) \ge 0$ , make no correction at  $t_i$ ; go on to the next decision time  $t_{i+1}$ . If the inequality does not hold, go on to step 4.
- (4) Calculate the performance index corresponding to total corrections only at  $t_{i+1}$  and  $t_f$ ,  $P_3(i)$
- (5) Form the switching functions  $S_i = P_2(i) P_3(i)$ . If  $S_i$  is positive, no action is taken. If it is negative or zero, a total correction is applied at  $t_i$ .
- (6) When the next decision time is reached, the process is reinitiated, with a new estimate of the error based upon the action taken at t<sub>1</sub> and the tracking data received during the interval.

A flow chart of this procedure is shown in Figure 38 and results of its application on a Mars mission are shown in Appendix A.



FIGURE 38. FLOW CHART FOR CONSTRAINED DYNAMIC PROGRAMMING ALGORITHM

### CONCLUSIONS

The evaluation techniques developed for interplanetary flyby mission astrionics systems were extended to orbiter, lander, and multiple planet swingby mission astrionics. These techniques are useful for selection of astrionics subsystems, for evaluating astrionics mission operation schedules, as an aid in the preliminary design of conceptual subsystems, and in determining research needed to improve system performance.

Using the computer program which implements these techniques, it is possible to analyze tradeoffs and establish requirements for functions such as navigation, guidance, and communication for specified missions. Candidate astrionics subsystems can be analyzed and the specifications determined for subsystems which provide the required functions.

Data required for evaluation of astrionics for interplanetary missions are identified. Numerical values for these data are quite difficult to obtain. In many cases prototype hardware may not have been developed and if hardware exists, the needed data may not have been acquired. The significance of assumed data for conceptual systems can be assessed by determining the sensitivity of the penalty function to the data. Data having high sensitivity indicates need for research, development, and testing programs.

### RECOMMENDATIONS

It is recommended that the computer program which implements the evaluation techniques be exercised for various interplanetary missions which appear likely to occur within the next decade. The objective would be to establish the astrionics specifications for the missions examined. By examining a large number of missions, possible limitations of the computer program should become apparent.

Early establishment of the astrionics specifications for these missions will identify critical data requirements. It is recommended that, testing programs be established to acquire and validate the data assumed in the analyses.

The assumption of Kalman filtering in the navigation error analysis for aided inertial systems did not consider the problems associated with the implementation of such a filter. It is recommended that a study of the tradeoffs involved in implementing a Kalman filter be made. This study should examine the feasibility of onboard implementation as opposed to implementing the filter at a site on Earth. Wherever the filter is implemented, the tradeoffs between performance of the filter and the required onboard computer capability must be determined.

Many future missions require the astrionics to operate over very long periods of time. Missions with long operating times include interplanetary missions, such as Grand Tour, and Earth orbital missions, such as a space station. Provision of the required astrionics functions on these long-life-time missions

necessitates having backup modes for each function. These modes are provided by astrionics subsystems operating in a specified manner. Failure of a subsystem required for a particular mode need not result in loss of the function if the astrionics system is designed with backup modes for critical functions. This concept of functions and modes was developed by Battelle for aerospace avionics systems under USAF contract (Reference 32).

An avionics function is defined as an operation or action performed during a mission in which employment of the aircraft's avionics is desirable. Two examples of avionics functions are navigation and communications. The term mode connotes a suite of avionics subsystems which allows a particular function to be performed. If all the subsystems are performing properly, then any given function will be conducted by utilizing the suite of avionics designated as the primary mode. Assume that one of the subsystems associated with a particular function begins to operate out of tolerance or fails . Quite often the function can still be conducted by a backup mode but with degraded performance. Usually the backup mode will employ the same suite of avionics as the primary mode, but with the unsatisfactory subsystem deleted or with another subsystem substituted for it. Of course, more than one backup mode may be possible, and the backup mode that is employed will depend on which is preferred as well as which subsystems are available. In a vehicle with sophisticated avionics, the selection of the backup mode may be made by an onboard computer or special-purpose logic circuitry rather than by the pilot. The computer program developed and described in Reference 32 uses Monte Carlo techniques to model the effectiveness of avionics systems with alternate modes for various functions. It is recommended that the concept of functions and modes be developed for astrionics systems and NASA missions. This will be necessary to properly analyze astrionics for long-lifetime missions and reusable vehicles.

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APPENDIX A

# COMPUTER PROGRAM RESULTS FOR MARS ORBITER MISSION

## APPENDIX A

### COMPUTER PROGRAM RESULTS FOR MARS ORBITER MISSION

To exercise the computer program on an orbiter mission, a mission similar to the planned Viking mission was selected as a basis for evaluation. It must be emphasized, however, that the results presented in this appendix do not constitute an evaluation of the Viking spacecraft and mission. These results are shown only to demonstrate the capability of the computer program. Considerable additional data must be obtained before a conclusive evaluation of the astrionics for the Viking mission can be performed.

#### Input Data

The first portion of the computer output is a listing of the data needed to run the program.

### Tracking Net Data

The tracking network data describing station locations, available radars, and the data associated with each radar are shown at the top of Figure A-la. This data is unchanged from the values used for the Jupiter flyby analysis reported in References 1 and 4 as well as elsewhere in this report, with the exception of the Deep Space Instrumentation Facility (DSIF) radar data. It should be noted that a range error of 50 feet is indicated for the DSIF. Although the DSIF does not measure range directly, range can be calculated if the DSIF tracks the spacecraft for a reasonable period of time (several days). As can be seen later in this appendix, the DSIF as well as all other radars are modeled as discrete one time updates rather than tracking over a period of time. It is assumed, however, that the values used for the discrete update are the results the DSIF would obtain after a reasonable tracking time.

### <u>Star Data</u>

Star data is shown in Figure A-la and is unchanged from the values used for the Jupiter studies.

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# Lewis N-Body Trajectory and State Transition Matrices

The Lewis n-body code and its modification to obtain state transition matrices was run to obtain a suitable trajectory for the Mars mission. A summary of this trajectory is printed at the bottom of Figure A-la and at the top of Figure A-lb. The data points printed in this summary are only those times, positions, and velocities at the spheres of influence crossings. As discussed in earlier reports (References 1 and 4) the Lewis n-body code performs trajectory integration separately from the launch vehicle simulation. It is thus necessary to check the matching of the end point of the launch vehicle simulation with the beginning of the Lewis trajectory. The results of this match are shown in Figure A-lb. The point on the Lewis trajectory with a radial distance from the center of the earth equal to the end point of the launch vehicle trajectory occurs 33 seconds into the Lewis trajectory. At this point, the velocities differ by 365 feet per second and the angle between the radius and velocity vectors differ by 1.82 degrees. This discrepancy is not considered serious for this analysis.

<u>Target Conditions</u>. The target point is specified as a time in seconds on the trajectory. The target conditions shown in Figure A-lb give the values of the trajectory at this time. The radius is  $1.57 \times 10^8$  feet and the velocity  $1.62 \times 10^5$  feet per second. At this time the angle between the radius and velocity vectors is approximately 90 degrees indicating periapsis of the planetary swingby.

<u>Near Earth Operations</u>. In order to insure the proper outgoing asymptote for the earth escape portion of the mission, a launch azimuth and the angle the vehicle traverses in the parking orbit must be computed. The results of these computations are shown at the bottom of Figure A-lb. A parking orbit coast angle of 171.6 degrees and a launch azimuth of 69.5 degrees were found to be necessary to obtain the proper outgoing asymptote. The launch is assumed to take place at the Eastern Test Range. This azimuth and parking orbit angle do not permit tracking of the vehicle while in the parking orbit. Thus, unlike the Jupiter mission, no parking orbit updates from ground based radars are possible.

### Mission and Subsystem Design Data

The mission and subsystem design data shown in Figures A-2a and A-2b have been modified from the values used for the Jupiter mission. The most significant changes are discussed in the following paragraphs.

<u>Spacecraft/Mission Data</u>. The spacecraft and mission data shown in Section I have the most significant changes from the Jupiter mission results. All the parameters describing the weight, sizes, and moments of inertia of the spacecraft were changed to values corresponding to the most current estimats for the Viking spacecraft. Note also that a nominal retro-delta velocity requirement has been added as Item 17 of Section I.

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MISSION AND SUBSYSTEM DATA, MARS MISSION

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MISSION AND SUBSYSTEM DATA, MARS MISSION

FIGURE A-2b.

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<u>Midcourse Engine/Energy Source Data</u>. The midcourse data shown in Section 2 have been changed to reflect the use of the propulsion system for the nominal retro fire as well as midcourse corrections. On the Viking spacecraft a single engine with 300 pounds thrust and 279 second specific impulse is used. These data are also shown in Section 2.

<u>Inertial Sensor Unit Design Data</u>. The inertial sensing unit design data shown in Section 3 are unchanged from that used for the Jupiter analysis.

<u>Thermal and Antenna Design Data</u>. The thermal subsystem and antenna design data shown in Section 4 are unchanged from the values used for the Jupiter analysis.

<u>Attitude Control Data</u>. The attitude control data shown in Section 5 have been changed to the values estimated for the Viking spacecraft.

### Candidate Subsystem Data

The candidate subsystem data shown in Figures A-3a thru A-3e are unchanged from the data used for the Jupiter analysis with the exception of the addition of the approach radar subsystem to the candidate subsystem data bank.

Approach Radar Subsystem Data. The approach radar subsystem data is shown at the bottom of Figure A-3d and the top of Figure A-3e. Three hypothetical approach radar subsystems were used in this study. The first approach radar, APP. RADR. measures range, and range rate with the errors indicated. The second approach radar, APP. RADR., is identical to the first approach radar with the exception that range is not measured. The third approach radar approximates a perfect measurement of range and range rate, indicated by errors of  $1.0 \times 10^{-6}$ feet and  $1.0 \times 10^{-6}$  feet per second as shown. This near perfect approach radar was used in a preliminary analysis to establish the most optimistic measurement of these parameters possible and their impact on penalty. The first approach radar, which measures range with an error of 300 feet and range rate with an error of 40 feet per second, is used in the evaluation presented in this appendix.

### The Communication Geometry

The geometry describing the visibility of the spacecraft from the earth is shown in Figure A-4. The number of days into the mission, the range in astronomical units, the angle between the spacecraft and the Sun as viewed from the earth, the angle between the Earth and the Sun as viewed from the spacecraft, and a label describing the portion of the mission for which the information is printed are in columns one through five, respectively. All rows in Figure A-4 are

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FIGURE A-3a. MARS MISSION, HARDWARE DATA

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FIGURE A-3b. MARS MISSION, HARDWARE DATA

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	BITS	COMP.FRE0.	INT_SCHEME -0.	0	•0-	•0-	• 0 -	•	
PLAT	FORMS								
p-1	2 4 7 1 1 1	4EIGHT 30.00000	POWER 100.00000	MTTF 2000.00000	MCTF 900.0000	LENGTH -0.00000	DIAMETER -0.00000	WIDTH -0.0000	. 0
	1=50,2=61M 1,000€∻00	• 0	• • •	• • •	•0 •	-0-	• 0 •	• 0	
tu)	¢ENT.ING	4EIGHT 80.00000	POWER 225.00000	MTTF 1350.00000	MCTF 500.00000	LENGTH -0.00000	DIAMETER -0.00000	WIDTH -0.000	o
	1=50,24GIM 2.000E+00	• 0	• 0 •	•	• • •	-0-	•	• 0 •	•
STAF				•					
e-1	ITT+LUN.OB Error(deg) 1.4005-02	KEIGHT /.00000 Fov (UEG) 8.0005+00	POWER 8.00000 DIRC55(1) 1.000£+00	MTTF 90000.00000 DIRCUS(2) U 0.	MCTF 500.00000 1RCOS(3) REF.STAR 2.000E+0	LENGTH -0.00000 W1DTH 0 3.000F+00	UIAMETER -0.00000 LEN/ANGV. 4.000E+00	WIDTH -0.0000 SC. FREQ. 1.000E+04	0 ScwD/RAD• 1.000E-02
. N	G [MB.ST	xEIG4T Z4.50000 F0V (DEG) 1.200E+02	POWER 14.00000 DTRCOS(1) 1.000E+00	MTTF 45000.00000 DIPCOS(2) D	MCTF SU0.00000 TRCOS(3) HEF.STAR - 2.000E+0	LENGTH -0.00000 #10th 0 2.0006+00	DIAMETER -0.00000 LEN/ANGV. 5.000E+00	WIDTH -0.0100 SC. FREQ. 5.000E+03	0 Scwd/rad. 7.000f-03
ราช	SENSOH								
e-1	1 204[-15UV	4516HT 2.00000	РОМЕR 5.0000	MTTF 100000.00000	MC1F 900.00000	LENGTH -0.00000	DIA4ETER 0.0000	WIDTH -0.0000	c

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FIGURE A-3c. MARS MISSION, HARDWARE DATA

A-10

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ů. -0.0000 • 0 WIDTH -0.00000 °0-•0 ••• ů 01 4.56300 -0.00000 -0.00000-4.54300 WIDIH WIDTH WIDTH WINTH WIDTH •0• •••• • •01 •0-•0= •••• UIAMETER -0.00000 DIAMETER 3.37500 -0.00000--0.00000 -0.00000 3.37500 UIAMETER DIAMETER DIAMETER DIAMETER • ; ; •••• •0= ••• •0-••• LENGTH -0.0000 -0,00000 5.25000 -0.00000 -0.00000-LENGTH LENGTH LENGTH LENGTH 0 . ? . . 。 • • • • MCTF 500-00000 MCTF 500-00000 MCTF 500.00000 MCTF 500.00000 MCTF 500.00000 MCTF 500-00000 • • • ••• ••• •••• ••• •••• •••• UTRCOS(3) 0. UIRCOS (3) (F) SO3HIO . • ••• • • • • 0 • • MTTF 2000-00000 100000\*000001 100000.000001 MTTF 10000.00000 100000.00000 1000000.00000 DIRCOS(2) 1,000E+00 DIRCOS(2) 1.000E+00 D14COS(2) 1.000E+00 MTTF MTTF MTTF MTTF • 0 -• • • •0• . • POWER 85.00000 POWER 3.50000 POWER 3.50000 POWER 5.00000 POMER 10.00000 00000°s DIRCOS(1) 0. DIRCOS(1) 0. DIRCOS(1) 0. POWER •0• • • •0• • () **•** 4.000E-01 FOV (DEG) 6.400E+01 FOV (DEG) FOV (DEG) 6.400E\*01 R. STE. ERR. 6.400E001 WEIGHT 2.00000 WEIGHT 3.10000 2.00000 14.25000 3.10000 15-00000 WEIGHT WEIGHT WEIGHT WEIGHT •0-°0-• 0 叭 ERROR (DEG) 4.000E+00 4.000E-02 8.6185-03 RANGE ERR. 3.0005405 ERROR (DEG) ERROR (DEG) APCL-1402X ADCL-1402Y 345 624-H APP . HAD . X ۳ () ۳ °() • °0~ XCD-HON 203-203 APP. RADAR COM. RCVR. ISU/C.P.S. (v)m N

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FIGURE A-3d. MARS MISSION, HARDWARE DATA

• Ú -• C = ပို -0.00000 -0.00000--0.0000 WIDTH WIDIH WIDTH •0 •••• ••• DIAMETER -0.00000 -0.00000 -0.0000-0-DIAMETER DIAMETER •0-• ••• LENGTH -0.00000 LENGTH =0.00000 -0.00000-0-LENGTH MARS MISSION, HARDWARE DATA ; ; ••• • MCTF 1000-00000 MCTF 500+00000 00000•00<u>é</u> •••• •••• ••• MCTF MTTF 1000000000 •••• • 0 • MTTF 2000-00000 2000-00000 FIGURE A-3e. MTTF -0-•0• •0: POWER 10.00000 РОМЕR 10.00000 POWER 20.00000 • 0 • •0• •0• 1.000E+06 4.0005+01 R.RTE.ERR. R.RTE.ERR. WEIGHT 10.00000 15.00000 15.00000 WEIGHT WEIGHT H.000E+00 -0. ٩ RANGE ERR. 1 . 000E ... no RANGE ERR. PWR.OUTPUT TEST XWITR APP. RADR. ° 0 PERFECT XMITTER റ N **,...**4

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Total         MANUE (201)         SC-900         MANUE (201)         SC-900         MANUE (201)         SC-900         MANUE (201)         MANUE (201) <th></th>	
2.2.5         3.5.5         11.1.41         11.2.1.4         11.2.1.4           1.1.2.1.1.1.1.1.1.1.1.1.1.1.1.1.1.1.1.1	
7.4.7       7.4.7       7.4.7         7.4.7       7.4.7       7.4.7	
31.15     31.15     31.15     31.15     31.15       31.15     31.15     31.15     31.15     31.15       31.15     31.15     31.15     31.15     31.15       31.15     31.15     31.15     31.15     31.15       31.15     31.15     31.15     31.15     31.15       31.15     31.15     31.15     31.15     31.15       31.15     31.15     31.15     31.15     31.15       31.15     31.15     31.15     31.15     31.15       31.15     31.15     31.15     31.15     31.15       31.15     31.15     31.15     31.15     31.15       31.15     31.15     31.15     31.15     31.15       31.15     31.15     31.15     31.15     31.15       31.15     31.15     31.15     31.15     31.15       31.15     31.15     31.15     31.15     31.15       31.15     31.15     31.15     31.15     31.15       31.15     31.15     31.15     31.15     31.15       31.15     31.15     31.15     31.15     31.15       31.15     31.15     31.15     31.15     31.15       31.15     31.15     31.15	31.1         31.1 <th< td=""></th<>
37.67     37.67     37.67     37.67     37.67       37.67     37.67     37.67     37.67     37.67       37.67     37.67     37.67     37.67     37.67       37.67     37.67     37.67     37.67     37.67       37.67     37.67     37.67     37.67     37.67       37.67     37.67     37.67     37.67     37.67       37.67     37.67     37.67     37.67     37.67       37.67     37.67     37.67     37.67     37.67       37.67     37.67     37.67     37.67     37.67       37.67     37.67     37.67     37.67     37.67       37.67     37.67     37.67     37.67     37.67       37.67     37.67     37.67     37.67     37.67       37.67     37.67     37.67     37.67     37.67       37.67     37.67     37.67     37.67     37.67       37.67     37.67     37.67     37.67     37.67       37.67     37.67     37.67     37.67     37.67       37.67     37.67     37.67     37.67     37.67       37.67     37.67     37.67     37.67     37.67       37.67     37.67     37.67	79-11         204-05         204-05         204-05         204-05           79-14         211000-5         210-057         204-05         211000-5           71-14         210-057         204-05         2110-057         214-05           71-15         210-057         210-057         210-057         210-057           71-05         210-057         210-057         210-057         210-057           71-05         210-057         210-057         210-057         210-057           71-05         210-057         210-057         210-057         210-057           71-05         210-057         210-057         210-057         210-057           71-05         210-057         210-057         210-057         210-057           71-05         210-057         210-057         210-057         210-057           71-05         210-057         210-057         210-057         210-057           71-05         210-057         210-057         210-057         210-057           71-05         210-057         210-057         210-057         210-057           71-057         210-057         210-057         210-057         210-057           71-057         210-057
73.06     110.07     10.07     10.07     10.07       51.45     110.07     110.07     110.07     110.07       51.45     111.40     111.40     111.40     111.01       51.45     111.40     111.40     111.40     111.01       51.45     111.41     110.47     111.40     111.01       51.45     111.41     111.40     111.40     111.40       51.45     111.41     111.40     111.40     111.40       51.45     111.41     111.40     111.40     111.40       51.46     111.41     111.40     111.40     111.40       51.47     111.41     111.41     111.40     111.40       51.47     111.41     111.41     111.41     111.41       51.41     111.41     111.41     111.41     111.41       51.41     111.41     111.41     111.41     111.41       51.41     111.41     111.41     111.41     111.41       51.41     111.41     111.41     111.41     111.41       51.41     111.41     111.41     111.41     111.41       51.41     111.41     111.41     111.41     111.41       51.41     111.41     111.41     111.41     111.41   <	37.61         177.64         177.64         17.64         17.064           37.64         17.064         17.064         17.064         17.064           37.64         17.064         17.064         17.064         17.064           37.64         17.064         17.064         17.064         17.064           37.64         17.064         17.064         17.064         17.064           37.64         17.064         17.064         17.064         17.064           37.64         17.064         17.064         17.064         17.064           77.64         17.064         17.064         17.064         17.066           77.64         27.063         27.063         27.064         27.064           77.64         27.063         27.064         27.064         27.064           77.64         27.064         27.064         27.064         27.064           77.64         27.064         27.064         27.064         27.064           77.64         27.064         27.064         27.064         27.064           77.64         27.064         27.064         27.064         27.064           77.64         27.064         27.064         27.064         <
67.60     210.00     10.4.5     10.4.5     11.40     HELLOCENT       55.65     17.40     10.4.5     11.40     HELLOCENT       57.65     17.60     10.4.5     11.40     HELLOCENT       57.65     17.60     10.4.5     11.40     HELLOCENT       57.65     17.60     10.4.5     10.4.5     10.4.5       57.65     17.60     10.4.5     10.4.5     10.4.5       57.65     17.60     10.4.5     10.4.5     10.4.5       57.65     10.4.5     10.4.5     10.4.5     10.4.5       57.65     10.4.5     10.4.5     10.4.5     10.4.5       57.75     11.4.5     11.4.5     11.4.6     11.4.6       57.75     11.4.5     11.4.5     11.4.6     11.4.6       57.75     11.4.5     11.4.5     11.4.6     11.4.6       57.75     11.4.5     11.4.5     11.4.6     11.4.6       57.75     11.4.5     11.4.5     11.4.6     11.4.6       57.75     11.4.5     11.4.7     11.4.6.	
10000       10000       10000       10000       10000	51         51<
11       10       11       10       11       10       11       10 <td< td=""><td>No.         No.         No.</td></td<>	No.
55-545     11014     111.03     111.03       67.7     1101.0     1101.0     1101.0 <td>59-50       +15114       -106-92       -11-30       HEL IOCENT         77-35       -27-24       -11-34       HEL IOCENT         77-35       -27-34       -11-34       -27-34         77-36       -11-34       -27-34       HEL IOCENT         77-37       -27-34       -11-34       HEL IOCENT         77-36       -11-34       -27-34       HEL IOCENT         77-34       -27-34       -27-34       HEL IOCENT         77-34       -27-34       -27-34       HEL IOCENT         77-44</td>	59-50       +15114       -106-92       -11-30       HEL IOCENT         77-35       -27-24       -11-34       HEL IOCENT         77-35       -27-34       -11-34       -27-34         77-36       -11-34       -27-34       HEL IOCENT         77-37       -27-34       -11-34       HEL IOCENT         77-36       -11-34       -27-34       HEL IOCENT         77-34       -27-34       -27-34       HEL IOCENT         77-34       -27-34       -27-34       HEL IOCENT         77-44
59-401     114/77     104-408     111-03     HL 1005H1       67-60     119-40     119-40     HL 1005H1       77-75     119-40     119-40     119-40       77-75     119-40     119-40     119-40       77-75     119-40     119-40     119-40       77-75     119-40     119-40     119-40       77-77     119-40     119-40     119-40       77-77     119-40     119-40     119-40       77-77     119-40     119-40     119-40       77-77     119-40     119-40     119-40       77-77     119-40     119-40     119-40       77-77     119-40     119-40     119-40 <td< td=""><td>59-51     1764-60     11.93     HELTOREN       77-52     11.93     HELTOREN     11.93       77-53     11.94     HELTOREN       77-53     11.94     HELTOREN       77-53     11.94     HELTOREN       77-53     11.93     HELTOREN       77-53     11.94     HELTOREN       77-54     11.94     H</td></td<>	59-51     1764-60     11.93     HELTOREN       77-52     11.93     HELTOREN     11.93       77-53     11.94     HELTOREN       77-53     11.94     HELTOREN       77-53     11.94     HELTOREN       77-53     11.93     HELTOREN       77-53     11.94     HELTOREN       77-54     11.94     H
6.8.48     717004     1044.46     113.45     HeL nockut       77.8.1     77.8.1     77.8.1     112.4.6     HeL nockut       77.8.1     77.8.1     77.8.1     112.4.6     HeL nockut       77.8.1     77.8.1     77.8.1     112.4.6     HeL nockut       77.8.1     77.8.1     77.8.1     77.8.1     112.4.6       77.8.1     77.8.1     77.8.1     112.4.7     112.4.7       77.8.2     77.8.1     77.8.1     112.4.7     112.4.7       77.8.2     77.8.1     142.7.8     22.8.2.7     112.1.05011       77.8.2     74.8.1     77.8.1     142.1.05111       90.8.2     73.8.1     142.7.8     23.7.4     142.1.05111       91.8.2     75.8.1     142.7.8     23.7.4     142.1.05111       91.8.2     75.8.1     75.8.1     75.8.1     176.1.1       91.8.2     75.8.1     75.8.1     75.8.1     176.1.1       91.8.1     75.8.1     75.8.1     75.8.1     177.8.1       91.8.1     75.8.1     75.8.1     75.8.1     177.8.1       91.8.1     75.8.1     75.8.1     75.8.1     177.8.1       91.8.1     75.8.1     75.8.1     75.8.1     177.8.1       91.8.1     75.8.1 <t< td=""><td>0.3-04         0.1000         -10-05         -10-05         HELTOREWING           71-04         -270-23         -10-05         HELTOREWING         HELTOREWING           71-04         -270-23         -10-05         HELTOREWING         HELTOREWING           71-04         -270-23         -10-05         HELTOREWING         HELTOREWING           71-05         -70-04         HELTOREWING         HELTOREWING         HELTOREWING           71-04         -70-04         -70-04         HELTOREWING         HELTOREWING           710-05         -70-04         HELTOREWING         HELTOREWING         HELTOREWING           710-05         -70-04         HELTOREWING         HELTOREWING         HELTOREWING           710-05         -70-04         HELTOREWIN</td></t<>	0.3-04         0.1000         -10-05         -10-05         HELTOREWING           71-04         -270-23         -10-05         HELTOREWING         HELTOREWING           71-04         -270-23         -10-05         HELTOREWING         HELTOREWING           71-04         -270-23         -10-05         HELTOREWING         HELTOREWING           71-05         -70-04         HELTOREWING         HELTOREWING         HELTOREWING           71-04         -70-04         -70-04         HELTOREWING         HELTOREWING           710-05         -70-04         HELTOREWING         HELTOREWING         HELTOREWING           710-05         -70-04         HELTOREWING         HELTOREWING         HELTOREWING           710-05         -70-04         HELTOREWIN
7.464     7.064     7.064     191.45     191.45       7.733     229.43     191.44     191.44       7.733     229.43     191.44       7.733     227.13     101.41       7.733     227.13     101.41       7.733     227.13     101.41       7.733     227.13     101.41       7.733     227.13     101.41       7.733     227.23     101.41       7.733     227.23     101.41       7.733     227.23     101.41       7.733     227.23     101.41       7.734     227.23     101.41       7.734     237.77     101.41       7.734     237.77     101.41       7.744     237.77     101.41       7.744     237.77     101.41       7.744     237.77     101.41       7.744     237.77     101.41       7.744     237.77     101.41       7.744     237.77     101.41       7.744     237.77     101.41       7.744     237.77     101.41       7.744     237.77     101.41       7.744     237.74     237.77       7.745     237.74     237.77       7.745     237.74     231.1050	71-30         2070241         -191-35         HELL TOCKNT           77-31         227-92         HELL TOCKNT         HELL TOCKNT           77-32         275-31         -195-15         HELL TOCKNT           77-35         257-92         HELL TOCKNT         HELL TOCKNT           77-35         257-92         HELL TOCKNT         HELL TOCKNT           77-35         257-92         HELL TOCKNT         HELL TOCKNT           87-37         194-97         222-25         HELL TOCKNT           87-37         272-26         HELL TOCKNT         222-25           87-87         273-31         27-27         HELL TOCKNT           91-27         27-27         22-26         HELL TOCKNT           91-27         27-27         442-27         22-26           91-27         27-27         442-27         442-10007           91-27         27-27         442-10007         442-10007 <t< td=""></t<>
1700       170       170         170       170	77.401     77.401     77.401     77.401     77.401       77.401     77.401     77.401     77.401     77.401       77.401     77.401     77.401     77.401     77.401       77.401     77.401     77.401     77.401     77.401       77.401     77.401     77.401     77.401     77.401       77.401     77.401     77.401     77.401     77.401       77.401     77.401     77.401     77.401     77.401       77.401     77.401     77.401     77.401     77.401       77.401     77.401     77.401     77.401     77.401       77.401     77.401     77.401     77.401     77.401       77.401     77.401     77.401     77.401     77.401       77.401     77.401     77.401     77.401     77.401       77.401     77.401     77.401     77.401     77.401       77.401     77.401     77.401     77.401     77.401       77.401     77.401     77.401     77.401     77.401       77.401     77.401     77.401     77.401     77.401       77.401     77.401     77.401     77.401     77.401       77.402     77.401     77.401     77.401
79.45       79.45       79.45       79.45       79.45         79.45       79.45       79.45       79.45       79.45         91.45       79.45       79.45       79.45       79.45         91.45       79.45       79.45       79.45       79.45         91.45       79.45       79.45       79.45       79.45         91.45       79.45       79.45       79.45       79.45         91.45       79.45       79.45       79.45       79.45         91.45       79.45       79.45       79.45       79.45         91.45       79.45       79.45       79.45       79.45         91.45       79.45       79.45       79.45       79.45         91.45       79.45       79.45       79.45       79.45         111.4       79.45       79.45       79.45       79.45         111.4       79.45       79.45       79.45       79.45         111.4       79.45       79.45       79.45       79.45         111.4       79.45       79.45       79.45       79.45         79.45       79.45       79.45       79.45       79.45         79.45       79.45	77.35         77.35 <td< td=""></td<>
77.35       =22.18       HELIOCEN.         77.35       =25007       =192.07       =22.18       HELIOCEN.         91.75       =339.77       =192.78       =25.43       =196.87       =25.43         91.75       =339.77       =192.78       =25.43       =196.87       =25.43         91.75       =339.77       =25.791       HELIOCEN.       =25.43         95.75       =334.77       =25.43       =334.77       HELIOCEN.         91.15       =57.991       =134.66       =33.477       HELIOCEN.         111.09       =57.993       =124.66       =33.477       HELIOCEN.         111.01       =57.993       =124.66       =33.491       HELIOCEN.         111.02       =57.993       =124.66       =33.491       HELIOCEN.         111.02       =57.493       =124.66       =37.48       =31.677       HELIOCEN.         112.05       =57.993       =124.66       =114.677       =31.661       HELIOCEN.         112.05       =57.993       =124.66       =114.677       =31.661       HELIOCEN.         122.695       =77.48       =114.677       =31.661       HELIOCEN.       =31.675         126.76       =57.993       =112.6	73-63       770-55       -752-07       -722-18       HELLOGENT         73-63       233-73       -148-68       -224-28       HELLOGENT         91-55       233-73       -148-68       -224-28       HELLOGENT         91-55       233-73       -149-67       -224-28       HELLOGENT         91-55       233-13       -149-67       -224-28       HELLOGENT         91-55       233-13       -139-67       -31-75       HELLOGENT         91-55       -139-67       -139-67       -31-75       HELLOGENT         91-55       -139-68       -139-68       -31-75       HELLOGENT         1111.00       -99-99       -123-47       -31-47       HELLOGENT         1111.01       -99-99       -123-48       -31-47       HELLOGENT         1111.02       -139-68       -123-48       -31-45       HELLOGENT         1111.02       -139-68       -123-47       -32-54       HELLOGENT         1111.02       -123-48       -31-55       -31-55       HELLOGENT         1112.02       -123-48       -31-55       -31-55       HELLOGENT         1112.02       -118-47       -37-55       -46-16       HELLOGENT         1126-19
N3-31     7194.47     22.424     H1106K11       91.72     33.67     142.66     22.66     11106K11       91.72     33.67     142.66     22.66     11106K11       91.72     33.67     142.66     22.66     11106K11       91.71     1111.07     23.66     112.66     124.66     123.66       91.71     1111.07     23.617     11106K11     11106K11       92.72     33.617     11106K11     11100K11       93.73     1111.07     1111.07     1111.07       93.73     1111.07     1111.07     1111.07       94.67     1111.07     1111.07     1111.07       1111.07     1111.07     1111.07     1111.07       1111.07     1111.07     1111.07     1111.07       1111.07     1111.07     1111.07     1111.07       1111.07     1111.07     1111.07     1111.07       1111.07     1111.07     1111.07     1111.07       1111.07     1111.07     1111.07     1111.07       1111.07     1111.07     1111.07     1111.07       1122.09     66.73     1111.07     1111.07       122.09     66.73     1111.07     1120.07       123.6.05     100.03     1111.07 <td< td=""><td>0.3-31       \$70-31</td></td<>	0.3-31       \$70-31
91.872       23487       195.63       27491       HELINGENT         91.872       23487       195.64       27491       HELINGENT         95.72       23487       195.64       195.64       HELINGENT         95.72       23487       195.64       HELINGENT       275.64         95.72       23487       195.64       HELINGENT         95.72       234.75       445.06       1144.64         100.12       45906       1145.64       234.75       445.106         111.00       45906       1145.64       235.64       445.106         111.00       45906       1125.64       235.64       46.106         111.00       1125.67       235.64       46.106       46.106         111.00       1125.67       235.64       46.106       46.106         125.66       135.64       1125.67       235.64       46.106       46.106         135.65       1125.67       1125.67       235.64       46.106       46.106       46.106         135.65       135.64       46.106       125.64       46.106       46.106       46.106       46.106       46.106       46.106       46.106       46.106       46.106       46.106	87.82       339.77       -26.26       FELTORENT         97.87       -26.26       FELTORENT       -27.77       FELTORENT         95.87       -27.87       -27.77       -27.77       FELTORENT         95.87       -27.87       -27.87       -27.16       FELTORENT         95.87       -27.91       FELTORENT       -27.97       FELTORENT         95.915       -134.67       -33.77       FELTORENT       -27.97         95.916       FELTORENT       -33.77       FELTORENT       -27.97         95.917       -134.67       -33.77       FELTORENT       -27.97         95.918       -128.68       -33.77       FELTORENT       -27.97         1114.07       -33.61       FELTORENT       -33.77       FELTORENT         1114.07       -33.61       FELTORENT       -33.77       FELTORENT         1114.07       -33.61       FELTORENT       -34.93       FELTORENT         1114.07       -1114.57       -38.91       FELTORENT       FELTORENT         122.95       -77.83       -1114.57       -38.91       FELTORENT         123.65       -77.83       -1114.57       -38.91       FELTORENT         125.65       -77.9<
9:25       332.63       1142.78       251.91       HELIOGENT         99:27       44.06       31.16       44.06       31.16       HELIOGENT         99:18       53412       1144.06       33.77       HELIOGENT       44.06         99:18       53401       114.05       114.06       44.06       33.77       HELIOGENT         111:05       54906       1144.06       33.477       HELIOGENT       44.06         111:05       55909       114.67       33.477       HELIOGENT         111:05       55909       114.67       34.75       HELIOGENT         111:05       55909       114.67       34.06       HELIOGENT         112:05       55909       112.56       5564       HELIOGENT         112:05       55909       112.56       34.06       HELIOGENT         12:05       55909       114.57       34.06       HELIOGENT         12:05       555       114.57       339.39       HELIOGENT         12:05       5570       114.57       339.39       HELIOGENT         12:05       5570       114.57       339.39       HELIOGENT         12:05       5570       114.57       339.39       HELIOGENT	51.25       96.273       -1134.677       -27.91       HEL IOCENT         99.18       -133.667       -33.777       HEL IOCENT       -110.671         99.18       -133.667       -33.777       HEL IOCENT       -110.671         111.107       -39.477       HEL IOCENT       -113.667       -33.777         111.107       -39.477       HEL IOCENT       -113.667       -33.777         111.107       -39.477       HEL IOCENT       -113.667       -33.777         111.107       -39.477       HEL IOCENT       -114.67       -33.777         111.107       -39.477       -112.944       -33.477       HEL IOCENT         111.107       -39.477       -31.776       HEL IOCENT       -114.67         111.107       -39.477       -31.916       -39.477       HEL IOCENT         111.107       -31.776       -31.776       HEL IOCENT       HEL IOCENT         111.107       -31.776       -31.910       -31.776       HEL IOCENT         112.07       -31.776       -31.910       HEL IOCENT       HEL IOCENT         112.07       -31.776       -31.910       HEL IOCENT       HEL IOCENT         113.677       -31.776       -31.910       HEL IOCENT
95.22       23.6872       -139.67       -29.70       HELTOGENT         99.18       -20.915       -146.68       -31.16       HELTOGENT         107.12       -29.915       -146.68       -33.77       HELTOGENT         111.09       -59.906       -124.66       -33.77       HELTOGENT         111.09       -59.906       -124.66       -33.77       HELTOGENT         115.07       -59.906       -124.64       -33.77       HELTOGENT         115.07       -59.906       -124.64       -33.77       HELTOGENT         115.07       -59.906       -124.64       -33.05       HELTOGENT         115.07       -59.907       -124.64       -33.05       HELTOGENT         115.07       -59.906       -124.64       -33.05       HELTOGENT         115.07       -59.907       -114.57       -38.95       HELTOGENT         126.07       -59.906       -114.53       -39.03       HELTOGENT         126.065       110.797       -38.95       HELTOGENT       HELTOGENT         136.05       -100.80       -100.903       HELTOGENT       HELTOGENT         136.05       100.800       -100.80       -39.93       HELTOGENT	99.472       -29.470       HELIDGENT         99.472       -39.477       HELIDGENT         99.472       -39.475       HELIDGENT         111.473       -99.475       HELIDGENT         111.473       -33.477       HELIDGENT         111.473       -33.403       HELIDGENT         111.473       -33.403       HELIDGENT         111.473       -33.403       HELIDGENT         122.493       -114.47       -33.403         122.403       -114.47       -33.403         122.404       -114.47       -33.403         122.405       -114.47       -33.403         122.406       -114.47       -33.403         122.407       -114.47       -33.403         122.408       -114.47       -33.
99-18       0.29916       -136.68       -31.16       HELIOCENT         1013-13       0.29916       -134.76       -33.47       HELIOCENT         1111.09       0.57996       -134.76       -33.47       HELIOCENT         1111.09       0.57996       -134.76       -33.47       HELIOCENT         1111.09       0.57996       -125.49       -33.47       HELIOCENT         1111.09       0.57996       -125.49       -33.47       HELIOCENT         1115.07       -559906       -125.49       -33.45       HELIOCENT         1115.07       -559906       -125.49       -33.51       HELIOCENT         1115.07       -559906       -125.49       -33.53       HELIOCENT         1122.59       -122.48       -33.51       HELIOCENT       -33.51         122.59       -112.44       -33.53       HELIOCENT       -33.53         122.59       -112.44       -33.53       HELIOCENT       -33.54         122.59       -122.44       -33.53       HELIOCENT       -33.54         122.59       -112.44       -33.53       HELIOCENT       -33.54         122.59       -112.44       -33.55       HELIOCENT       -33.56         155	99.18       9.9.18       9.9.16       136.66       -31.16       HELIOCENT         101.12       -32.475       HELIOCENT       -32.475       HELIOCENT         1115.07       -49.05       -126.66       -31.75       HELIOCENT         1115.07       -49.07       -128.67       -33.75       HELIOCENT         1115.07       -59.90       -39.75       HELIOCENT       HELIOCENT         1115.07       -59.90       -39.64       HELIOCENT       HELIOCENT         1115.07       -128.67       -37.55       HELIOCENT       HELIOCENT         115.07       -59.90       -39.56       HELIOCENT       HELIOCENT         115.07       -79.66       -114.87       -39.56       HELIOCENT         128.07       -79.92       -114.87       -39.93       HELIOCENT         128.07       -114.87       -39.93       HELIOCENT       HELIOCENT         128.06       -114.87       -39.93       HELIOCENT       HELIOCENT         128.06       -114.87       -114.87       -39.93       HELIOCENT         128.06       -114.87       -114.87       -39.93       HELIOCENT         128.06       -114.87       -114.84       -39.94       HELIOCENT
v       103.13       -0.2902       -135.05       HELIOCENT         101.07       -07       -33.77       HELIOCENT         111.07       -55099       -125.05       +51.05         111.07       -55099       -125.09       -33.77       HELIOCENT         111.07       -55099       -125.09       -35.77       HELIOCENT         111.07       -55090       -125.04       -35.475       HELIOCENT         112.07       -55090       -125.04       -35.64       HELIOCENT         125.07       -55090       -125.04       -35.64       HELIOCENT         125.07       -55090       -125.04       -35.010       HELIOCENT         125.07       -114.57       -38.00       HELIOCENT       HELIOCENT         125.07       -114.57       -38.00       HELIOCENT       HELIOCENT         136.57       -114.57       -38.00       HELIOCENT       HELIOCENT         136.57       -114.57       -38.00       HELIOCENT       HELIOCENT         136.56       -107.118       -39.30       HELIOCENT       HELIOCENT         156.56       107.20       -90.93       HELIOCENT       HELIOCENT         156.56       107.20       -90.33	\$\$ 1034.55       \$\$ 55900       \$\$ 132.60       #ELIOCENT         111.05       \$\$ 55703       \$\$ 132.64       #ELIOCENT         111.05       \$\$ 55703       \$\$ 123.64       #ELIOCENT         120.05       \$\$ 55703       \$\$ 113.67       \$\$ 339.66       #ELIOCENT         120.05       \$\$ 5703       \$\$ 510.68       \$\$ 500.68       #ELIOCENT         120.05       \$\$ 500.68       \$\$
10:00       :49:05       :33:77       HELIOCENT         110:07       :59:09       :12:48       :34:75       HELIOCENT         110:07       :59:09       :12:48       :34:75       HELIOCENT         110:07       :59:09       :12:48       :34:75       HELIOCENT         110:07       :59:06       :12:48       :36:04       HELIOCENT         110:07       :57:06       :12:48       :36:04       HELIOCENT         12:05:07       :57:06       :12:48       :36:04       HELIOCENT         12:05:07       :57:06       :12:48       :37:04       HELIOCENT         12:05:07       :75:06       :11:46:3       :38:01       HELIOCENT         12:05:07       :75:05       :11:46:3       :39:03       HELIOCENT         13:06:07       :77:0       :38:51       HELIOCENT       HELIOCENT         15:06       :10:26:07       :11:46:3       :39:09       HELIOCENT         15:06       :10:26:07       :11:46:3       :39:09       HELIOCENT         15:07:06       :10:26:07       :11:46:3       :39:09       HELIOCENT         15:06       :10:26:07       :10:26:07       :10:26:07       HELIOCENT         15:06 <t< td=""><td>101-12       \$\$4,75       =133.77       HELIDCENT         111.09       \$5906       =125.68       =33.77       HELIDCENT         111.09       \$5906       =125.96       =35.77       HELIDCENT         111.00       \$5906       =125.96       =35.77       HELIDCENT         111.00       \$5906       =125.96       =55.64       HELIDCENT         112.02       \$5906       =125.96       =35.64       HELIDCENT         \$65070       =17.21.08       =35.95       HELIDCENT       HELIDCENT         \$65070       =114.57       =30.95       HELIDCENT       HELIDCENT         \$76.65       =107.42       =30.95       HELIDCENT       HELIDCENT         \$155.76       =76.76       =107.42       =30.95       HELIDCENT         \$155.76       =17.42       =30.95       HELIDCENT       HELIDCENT         \$155.76       =107.91       =00.03       HELIDCENT       HELIDCENT         \$155.76</td></t<>	101-12       \$\$4,75       =133.77       HELIDCENT         111.09       \$5906       =125.68       =33.77       HELIDCENT         111.09       \$5906       =125.96       =35.77       HELIDCENT         111.00       \$5906       =125.96       =35.77       HELIDCENT         111.00       \$5906       =125.96       =55.64       HELIDCENT         112.02       \$5906       =125.96       =35.64       HELIDCENT         \$65070       =17.21.08       =35.95       HELIDCENT       HELIDCENT         \$65070       =114.57       =30.95       HELIDCENT       HELIDCENT         \$76.65       =107.42       =30.95       HELIDCENT       HELIDCENT         \$155.76       =76.76       =107.42       =30.95       HELIDCENT         \$155.76       =17.42       =30.95       HELIDCENT       HELIDCENT         \$155.76       =107.91       =00.03       HELIDCENT       HELIDCENT         \$155.76
111.09       =125.68       =125.68       =33.47       HELTOGENT         115.05       =55099       =1125.09       =35.81       HELTOGENT         119.05       =55099       =1125.09       =35.64       HELTOGENT         125.90       =55099       =1125.09       =35.64       HELTOGENT         125.90       =55099       =1125.09       =35.64       HELTOGENT         125.95       =123.64       =37.35       HELTOGENT       HELTOGENT         136.95       =123.95       =112.95       =31.95       HELTOGENT         136.95       =112.95       =112.95       =31.95       HELTOGENT         136.95       =1111.63       =32.95       HELTOGENT       HELTOGENT         136.95       =1111.63       =32.95       HELTOGENT       HELTOGENT         186.65       =1111.63       =32.95       HELTOGENT       HELTOGENT         186.65       =105.95       =105.90       =107.90       HELTOGENT         166.63       1.00.80       =105.90       =40.10       HELTOGENT         166.63       1.00.80       =105.90       =40.10       HELTOGENT         166.63       1.00.80       =0.03       =40.03       HELTOGENT	111.09       .49706       -125.64       -55.61       HELIOCENT         115.07       .55.99       -125.09       -35.61       HELIOCENT         119.07       .55.00       -125.09       -35.61       HELIOCENT         122.99       .55.99       -125.09       -35.61       HELIOCENT         122.99       .55.00       -55.00       -118.97       -35.61       HELIOCENT         122.99       .57.00       -118.97       -33.61       HELIOCENT       -57.00         122.99       .57.00       -118.97       -33.61       HELIOCENT       -57.00         122.99       .77.29       -118.97       -33.65       HELIOCENT       -57.00         130.50       .77.53       -1113.89       -99.03       HELIOCENT       -57.00         130.55       .77.66       .77.69       -107.13       -39.65       HELIOCENT         150.76       .77.66       .77.69       -107.13       -39.65       HELIOCENT         150.76       .77.66       .77.69       .70.03       -50.21       HELIOCENT         150.76       .77.66       .77.61       .70.29       -107.13       .29.65       HELIOCENT         150.76       .77.61       .70.29
115.05       57999       -125.09       -35.64       HELIOCFNT         122.90       55940       -121.44       -35.64       HELIOCFNT         122.90       55940       -121.44       -35.64       HELIOCFNT         122.90       55940       -114.57       -38.04       HELIOCFNT         122.90       55940       -114.57       -38.04       HELIOCFNT         122.90       57472       -114.57       -38.04       HELIOCFNT         125.76       •57472       -114.63       -39.03       HELIOCFNT         135.75       •74284       -1114.63       -39.03       HELIOCFNT         135.60       •77284       -1114.63       -39.93       HELIOCFNT         135.76       •77284       -1114.63       -39.93       HELIOCFNT         156.76       •77690       -107.18       -39.93       HELIOCENT         156.76       •96271       -102.90       -40.03       HELIOCENT         156.65       1905.01       -90.23       -90.23       HELIOCENT         156.66       1905.02       -90.23       HELIOCENT       HELIOCENT         156.76       96.713       -90.23       -90.23       HELIOCENT         156.65 <td< td=""><td>119.005       *57090       -1/25.09       -1/25.09       -1/25.09         119.005       *57090       -57040       -1/25.09       -55.64       HELIOCENT         128.05       *5700       -57040       -1/21.04       -31.35       HELIOCENT         128.05       *6757       -1/21.04       -31.35       HELIOCENT       -1/21.07         128.05       *6757       -1/21.04       -31.35       HELIOCENT         138.05       *114.05       -1/21.04       -32.05       HELIOCENT         138.05       *114.05       -1/23.04       -33.05       HELIOCENT         138.05       *114.05       -33.05       HELIOCENT       -100.07         138.05       *114.05       -33.05       HELIOCENT       -100.07         148.1000000       *107.02       -107.02       -33.05       HELIOCENT         158.05       107.01       -107.02       -99.05       -107.02       -107.02         158.05       100.02       -107.02       -99.05       -100.02       HELIOCENT         158.05       107.02       -99.05       -100.02       HELIOCENT       -99.05         158.05       107.02       -99.05       -90.03       HELIOCENT       -99.05</td></td<>	119.005       *57090       -1/25.09       -1/25.09       -1/25.09         119.005       *57090       -57040       -1/25.09       -55.64       HELIOCENT         128.05       *5700       -57040       -1/21.04       -31.35       HELIOCENT         128.05       *6757       -1/21.04       -31.35       HELIOCENT       -1/21.07         128.05       *6757       -1/21.04       -31.35       HELIOCENT         138.05       *114.05       -1/21.04       -32.05       HELIOCENT         138.05       *114.05       -1/23.04       -33.05       HELIOCENT         138.05       *114.05       -33.05       HELIOCENT       -100.07         138.05       *114.05       -33.05       HELIOCENT       -100.07         148.1000000       *107.02       -107.02       -33.05       HELIOCENT         158.05       107.01       -107.02       -99.05       -107.02       -107.02         158.05       100.02       -107.02       -99.05       -100.02       HELIOCENT         158.05       107.02       -99.05       -100.02       HELIOCENT       -99.05         158.05       107.02       -99.05       -90.03       HELIOCENT       -99.05
119:02       -56/33       -1/2.44       -36.64       HELIDGENT         122:99       -559906       -114.57       -37.35       HELIDGENT         126:96       -65569       -114.57       -37.35       HELIDGENT         126:96       -67472       -114.57       -37.35       HELIDGENT         126:96       -67472       -114.57       -37.35       HELIDGENT         136:07       -67472       -114.57       -37.35       HELIDGENT         136:07       -67472       -114.57       -39.03       HELIDGENT         136:07       -77.422       -1105.07       -39.93       HELIDGENT         186:07       -716.29       -1107.18       -39.93       HELIDGENT         186:07       -107.18       -107.18       -39.93       HELIDGENT         186:05       -107.18       -105.03       +00.03       HELIDGENT         159:06       107.18       -105.90       -40.16       HELIDGENT         158:06       107.18       -105.90       +00.03       HELIDGENT         158:06       107.29       -105.90       -40.16       HELIDGENT         174:06       107.913       -94.94.70       -40.13       HELIDGENT         174:05 <td>122.90       -56/23       -1/21.48       -37.55       HELINGENT         122.90       -557006       -11/4.57       -37.35       HELINGENT         120.92       -65772       -11/4.57       -37.35       HELINGENT         130.92       -65772       -11/4.57       -37.35       HELINGENT         130.92       -65772       -11/4.57       -37.35       HELINGENT         130.92       -109.92       -39.39       HELINGENT       HELINGENT         130.05       -11/4.63       -31.35       HELINGENT       -37.56         130.05       -11/4.63       -111.4.63       -39.39       HELINGENT         130.05       -1107.13       -39.39       HELINGENT       HELINGENT         140.03       -107.13       -39.46       HELINGENT       HELINGENT         150.76       -77.39       -91023       -107.42       -39.49       HELINGENT         150.76       -79.02       -107.13       -40.17       HELINGENT       HELINGENT         150.76       -77       -39.49       HELINGENT       HELINGENT       HELINGENT         150.76       -79.02       -107.13       -40.27       HELINGENT       HELINGENT         150.65       -107.42</td>	122.90       -56/23       -1/21.48       -37.55       HELINGENT         122.90       -557006       -11/4.57       -37.35       HELINGENT         120.92       -65772       -11/4.57       -37.35       HELINGENT         130.92       -65772       -11/4.57       -37.35       HELINGENT         130.92       -65772       -11/4.57       -37.35       HELINGENT         130.92       -109.92       -39.39       HELINGENT       HELINGENT         130.05       -11/4.63       -31.35       HELINGENT       -37.56         130.05       -11/4.63       -111.4.63       -39.39       HELINGENT         130.05       -1107.13       -39.39       HELINGENT       HELINGENT         140.03       -107.13       -39.46       HELINGENT       HELINGENT         150.76       -77.39       -91023       -107.42       -39.49       HELINGENT         150.76       -79.02       -107.13       -40.17       HELINGENT       HELINGENT         150.76       -77       -39.49       HELINGENT       HELINGENT       HELINGENT         150.76       -79.02       -107.13       -40.27       HELINGENT       HELINGENT         150.65       -107.42
122.50       553599       -121.04       -37.35       HELTOCFNT         128.96       653569       -11.48.57       -38.04       HELTOCENT         130.52       653569       -11.48.57       -339.03       HELTOCENT         130.52       67357       -11.3.89       -39.03       HELTOCENT         130.52       67572       -11.3.89       -39.03       HELTOCENT         130.52       77528       -11.1.6.3       -39.03       HELTOCENT         130.52       6757       -11.3.89       -39.03       HELTOCENT         150.67       77588       -1107.13       -39.03       HELTOCENT         150.76       -759.42       -1107.13       -39.93       HELTOCENT         150.63       107.33       -100.03       HELTOCENT       HELTOCENT         150.63       107.33       -40.15       -40.15       HELTOCENT         150.63       107.33       -91.27       -40.15       HELTOCENT         170.65       1.07.05       -91.25       -91.23       HELTOCENT         170.65       1.07.33       HELTOCENT       -107.33       HELTOCENT         170.65       1.07.33       -94.75       -90.35       HELTOCENT         172.55 </td <td>122.90       *5399       *12.4.04       -37.35       #ELIncENT         125.96       *65399       *114.57       -38.04       #ELincENT         130.72       *115.87       -38.04       #ELincENT         130.72       *115.87       -38.04       #ELincENT         130.72       *115.87       -38.04       #ELincENT         130.72       *115.87       -39.39       #ELincENT         130.72       *115.87       -39.39       #ELincENT         *76.88       -110.713       *39.39       #ELincENT         *76.79       *10.742       -10.942       -39.39       #ELincENT         130.723       *10.7413       *39.39       #ELincENT       #ELincENT         154.73       *0.123       *10.40       *0.17       #ELincENT         154.73       *0.124       *30.93       #ELincENT       #ELincENT         154.73       *0.124       *0.20       #ELincENT       #ELincENT         154.73       *0.124       *0.242       *10.246       #ELincENT         154.73       *0.124       *0.276       #ELincENT       #ELincENT         154.66       *0.13       #ELincENT       #ELincENT         156.65       10.040<!--</td--></td>	122.90       *5399       *12.4.04       -37.35       #ELIncENT         125.96       *65399       *114.57       -38.04       #ELincENT         130.72       *115.87       -38.04       #ELincENT         130.72       *115.87       -38.04       #ELincENT         130.72       *115.87       -38.04       #ELincENT         130.72       *115.87       -39.39       #ELincENT         130.72       *115.87       -39.39       #ELincENT         *76.88       -110.713       *39.39       #ELincENT         *76.79       *10.742       -10.942       -39.39       #ELincENT         130.723       *10.7413       *39.39       #ELincENT       #ELincENT         154.73       *0.123       *10.40       *0.17       #ELincENT         154.73       *0.124       *30.93       #ELincENT       #ELincENT         154.73       *0.124       *0.20       #ELincENT       #ELincENT         154.73       *0.124       *0.242       *10.246       #ELincENT         154.73       *0.124       *0.276       #ELincENT       #ELincENT         154.66       *0.13       #ELincENT       #ELincENT         156.65       10.040 </td
126.96       653599       1118.57       138.04       HELIOCENT         136.52       667472       1116.27       138.51       HELIOCENT         136.56       77.588       111.6.2       111.6.3       139.93         136.57       738.51       HELIOCENT       121.00         136.57       77588       111.6.3       129.93       HELIOCENT         136.79       77588       1107.13       39.93       HELIOCENT         166.79       87741       107.13       39.93       HELIOCENT         154.73       91923       107.13       39.93       HELIOCENT         156.76       91923       107.13       59.95       HELIOCENT         156.63       107.13       107.03       400.20       HELIOCENT         156.65       107.13       50.95       40.10       HELIOCENT         156.65       107.13       102.90       40.20       HELIOCENT         156.65       107.13       102.90       40.20       HELIOCENT         156.65       107.13       102.90       40.20       HELIOCENT         156.66       107.05       102.90       40.20       HELIOCENT         156.66       107.05       102.90       40.	286.95       +65.05       +114.87       -38.01       HELIOCENT         336.55       +67472       -114.87       -38.51       HELIOCENT         336.55       +7588       -114.83       -39.55       HELIOCENT         336.57       *7588       -114.83       -39.55       HELIOCENT         336.57       *7588       -114.83       -39.55       HELIOCENT         136.75       *7588       -114.83       -39.55       HELIOCENT         156.75       *87761       -114.63       -39.55       HELIOCENT         156.75       *87761       -109.07       +60.03       HELIOCENT         156.65       107.81       -100.40       -40.17       HELIOCENT         156.65       107.81       -100.40       -40.17       HELIOCENT         156.65       107.80       -107.97       -40.17       HELIOCENT         157.45       -99.65       -107.97       -40.17       HELIOCENT         158.45       -99.65       -107.97       -40.16       HELIOCENT         158.45       -99.45       -102.99       -40.17       HELIOCENT         158.45       -99.45       -99.45       -40.16       HELIOCENT         177.45       -95
136.52       -116.57       -33.51       HELIOCENT         136.52       -7472       -111.63       -33.03       HELIOCENT         136.56       -7422       -103.65       -111.63       -33.03       HELIOCENT         136.79       -7422       -105.07       -33.93       HELIOCENT       -33.93       HELIOCENT         166.79       -74223       -107.18       -33.93       HELIOCENT       -33.93       HELIOCENT         156.76       -796.03       -107.18       -105.07       -33.93       HELIOCENT       -33.93       HELIOCENT         156.76       -796.65       -107.18       -102.90       -40.03       HELIOCENT       -40.07         156.63       100.40       -40.03       -40.16       HELIOCENT       -40.17       HELIOCENT         156.63       100.40       -40.16       -40.16       -40.16       -40.17       -40.17         166.63       100.40       -90.82       -90.82       -90.82       -40.16       -40.17         172.65       107.30       -91.76       -90.13       HELIOCENT       -40.16       -40.15         173.56       100.42       -90.53       HELIOCENT       -40.15       -40.15       -17.65 <t< td=""><td>130-52       767472       -116.57       -38.51       HELTOCENT         130-52       77492       -111.65       -39.03       HELTOCENT         134.66       779       77584       -111.65       -39.03       HELTOCENT         134.66       779       87481       -107.13       -39.93       HELTOCENT         157.65       87461       -107.13       -39.93       HELTOCENT         160.79       87461       -107.13       -39.93       HELTOCENT         160.79       87461       -107.13       -39.93       HELTOCENT         160.79       -87461       -102.90       -40.03       HELTOCENT         150.76       -91923       -100.40       -40.03       HELTOCENT         151.76       -91923       -100.40       -40.20       HELTOCENT         152.66       100.80       -90.20       -40.16       HELTOCENT         152.66       100.80       -90.20       -40.16       HELTOCENT         170.60       10.90       -90.20       -90.40       -40.16       HELTOCENT         170.60       10.90730       -92.40       -92.40       -93.43       HELTOCENT         170.60       10.16       HELTOCENT       -94.016</td></t<>	130-52       767472       -116.57       -38.51       HELTOCENT         130-52       77492       -111.65       -39.03       HELTOCENT         134.66       779       77584       -111.65       -39.03       HELTOCENT         134.66       779       87481       -107.13       -39.93       HELTOCENT         157.65       87461       -107.13       -39.93       HELTOCENT         160.79       87461       -107.13       -39.93       HELTOCENT         160.79       87461       -107.13       -39.93       HELTOCENT         160.79       -87461       -102.90       -40.03       HELTOCENT         150.76       -91923       -100.40       -40.03       HELTOCENT         151.76       -91923       -100.40       -40.20       HELTOCENT         152.66       100.80       -90.20       -40.16       HELTOCENT         152.66       100.80       -90.20       -40.16       HELTOCENT         170.60       10.90       -90.20       -90.40       -40.16       HELTOCENT         170.60       10.90730       -92.40       -92.40       -93.43       HELTOCENT         170.60       10.16       HELTOCENT       -94.016
136.09       71492       113.89       39.03         136.05       7566       111.63       39.39         136.76       77588       111.63       39.39         156.79       87.741       105.07       105.03         156.79       87.741       105.07       105.07         156.79       87.741       105.07       105.07         156.79       87.741       105.07       105.07         156.63       105.07       -60.17       17         156.63       1.00.40       -60.17       17         156.63       1.00.40       -60.13       100.16         156.63       1.00.40       -60.13       100.40         156.63       1.00.40       -60.20       161.005         156.63       1.00.40       -60.20       40.16         170.60       100.40       -60.20       40.16         170.61       170.65       -99.26       461.005         170.65       1.17.66       -94.70       -39.93         170.65       1.17.66       -94.70       -39.93         170.65       1.17.66       -92.66       40.16         174.55       -93.64       -10.70       -39.93	336.09       77472         336.05       77402         134.05       77400         134.05       77400         154.75       77400         154.75       105741         154.73       105741         154.73       105741         154.73       105741         154.73       10723         154.73       10723         154.73       10723         154.73       100.17         154.73       100.17         154.73       100.17         154.73       100.17         154.73       100.17         154.73       100.17         154.73       100.17         154.73       100.17         154.73       100.17         154.73       100.17         154.69       100.17         160.13       100.401         160.14       100.401         170.60       100.17         170.60       100.13         172.60       100.401         172.60       100.401         172.51       100.401         172.51       100.18         172.51       100.18         17
134.56       75288       111.63       39.59         184.66       79.58       39.59       4EL106ENT         156.79       87.41       107.13       39.56         156.79       87.41       107.13       42.03         156.79       87.41       107.13       40.70         154.73       99.93       4EL106ENT         154.73       99.923       4EL106ENT         157.65       100.40       -40.20         158.69       1.00.40       -40.70         158.65       1.00.40       -40.70         158.65       1.00.40       -40.16         158.65       1.00.40       -40.16         158.65       1.00.40       -40.16         158.65       1.00.40       -40.16         158.65       1.00.40       -40.16         158.65       1.00.40       -40.16         177.45       -94.70       -39.93         177.45       -94.70       -39.93         177.45       -94.70       -39.93         127.45       -94.70       -39.93         127.45       -94.70       -39.93         127.45       -94.70       -39.93         125.65       -94.73<	134.86       97588       -1111.63       -339.39       HELIOCENT         166.75       87761       -107.13       -339.93       HELIOCENT         156.76       87761       -107.13       -339.93       HELIOCENT         156.76       87761       -105.07       -0.017       HELIOCENT         156.76       87761       -105.07       -0.03       HELIOCENT         156.76       87761       -105.07       -0.0.17       HELIOCENT         156.69       -0.027       HELIOCENT       -0.0.20       HELIOCENT         158.69       -0.023       -0.012       -0.0.20       HELIOCENT         166.63       10.0601       -0.0.13       +0.0.16       HELIOCENT         166.63       10.0601       -0.0.20       HELIOCENT       HELIOCENT         166.63       10.0040       -40.16       HELIOCENT       HELIOCENT         170.60       10.040       -30.33       HELIOCENT       HELIOCENT         170.56       120.052       -84.70       -39.93       HELIOCENT         170.56       170.56       -39.93       HELIOCENT       HELIOCENT         170.56       170.56       -39.93       HELIOCENT       HELIOCENT         195.
162.62       79405       -105.42       -39.66       HELIOGENT         166.79       87741       -107.13       -39.93       HELIOCENT         150.76       87741       -107.13       -39.93       HELIOCENT         150.76       87741       -100.03       HELIOCENT       HELIOCENT         150.76       91923       -100.40       -40.17       HELIOCENT         150.69       -9723       -100.40       -40.16       HELIOCENT         150.60       100.40       -40.13       HELIOCENT       -40.16         162.60       1.00.40       -40.13       HELIOCENT       -40.16         170.60       1.00.40       -39.63       HELIOCENT       -40.16         170.65       1.00.40       -39.63       HELIOCENT       -40.16         170.65       1.00.40       -39.63       HELIOCENT       -39.63         170.65       1.1766       -39.63       HELIOCENT       -30.65         100.67       -39.63       -34.42       -39.63       HELIOCENT         100.67       -39.63       -34.42       -39.63       HELIOCENT         100.60       1.3128       -34.63       -41.00       -39.63       HELIOCENT	162.52       79405       -107.42       -33.966       HELIOCENT         156.79       687461       -107.13       -33.93       HELIOCENT         156.76       687461       -107.13       -30.93       HELIOCENT         154.73       -91923       -102.907       -60.03       HELIOCENT         154.73       -91923       -102.907       -60.17       HELIOCENT         154.73       -91923       -102.907       -60.17       HELIOCENT         154.73       -91923       -102.907       -60.13       HELIOCENT         162.66       1.00.40       -40.15       HELIOCENT       HELIOCENT         162.66       1.00.40       -40.13       HELIOCENT       HELIOCENT         162.66       1.00.40       -39.93       HELIOCENT       HELIOCENT         160.63       1.00.401       -40.13       HELIOCENT       HELIOCENT         174.56       1.90.403       -93.93       HELIOCENT       HELIOCENT         174.56       1.174.6       -93.43       HELIOCENT       HELIOCENT         173.55       1.1774.6       -93.45       HELIOCENT       HELIOCENT         174.56       1.37.55       -84.04       -33.46       HELIOCENT
166.79       83959       -107.13       -39.93       HELIOCENT         150.76       87741       -105.07       -40.03       HELIOCENT         154.73       -91923       -1105.00       -40.03       HELIOCENT         158.69       -91923       -1100.40       -40.20       HELIOCENT         158.69       -9102.90       -40.20       -40.20       HELIOCENT         158.69       -90.221       -100.40       -40.16       HELIOCENT         158.60       100.40       -40.13       HELIOCENT       -40.13         170.60       1.00.40       -40.13       HELIOCENT       -40.13         170.65       1.00.40       -39.453       HELIOCENT       -40.13         170.65       1.00.40       -39.453       HELIOCENT       -39.453         170.65       1.1776       -39.453       HELIOCENT       -39.453         173.65       1.1776       -39.453       HELIOCENT       -39.453         100.43       1.2766       -39.453       HELIOCENT       -39.453         100.43       1.2766       -39.453       HELIOCENT       -39.453         100.44       -30.453       -34.45       -34.451       -34.451	166.79       687741       105.07       539.93       HELIOCENT         154.73       697823       105.07       539.93       HELIOCENT         154.73       697823       105.97       40.17       HELIOCENT         158.65       19823       100.40       40.03       HELIOCENT         158.65       190618       100.40       40.17       HELIOCENT         158.65       190618       100.40       40.13       HELIOCENT         158.65       190618       -98.76       440.15       HELIOCENT         170.60       190.930       -98.70       -39.93       HELIOCENT         170.60       170.55       -88.56       -39.93       HELIOCENT         170.60       1.1766       -39.93       HELIOCENT       HELIOCENT         170.60       1.1766       -39.55       HELIOCENT       HELIOCENT         170.63       1.3766       -39.36       HELIOCENT       HELIOCENT         170.63       1.3128       -85.66       -39.35       HELIOCENT         190.63       1.3128       -85.66       -39.36       HELIOCENT         190.65       1.35.65       -85.66       -39.55       HELIOCENT         190.65       1.3128<
150.46     -87(4)     -105.07     -40.03     HELIOCENT       154.73     -91923     -100.40     -40.17     HELIOCENT       158.69     -96.221     -100.40     -40.17     HELIOCENT       158.69     -96.221     -100.40     -40.13     HELIOCENT       158.69     -96.213     -98.75     -400.13     HELIOCENT       158.65     1.00.40     -40.13     HELIOCENT       170.60     1.00.40     -39.93     HELIOCENT       173.53     1.1766     -99.55     -39.93     HELIOCENT       173.55     -99.55     -39.93     HELIOCENT       173.55     -99.55     -39.93     HELIOCENT       173.55     -99.55     -39.45     HELIOCENT       100.43     1.1766     -39.45     HELIOCENT       100.43     1.20.47     -39.45     HELIOCENT	150.(6       -87(4)       -105.00       -40.03       HELIOCENT         154.73       -91923       -105.00       -40.17       HELIOCENT         158.69       -91923       -105.00       -40.20       HELIOCENT         162.66       100.40       -40.16       HELIOCENT       -40.17         162.66       100.60       -94.70       -39.93       HELIOCENT         170.60       100.40       -40.16       HELIOCENT       -40.16         170.60       100.70       -39.93       HELIOCENT       -40.17         170.60       100.70       -39.93       HELIOCENT       -40.15         170.60       1.70.50       -39.93       HELIOCENT       -40.15         170.60       1.17.65       -92.95       -39.93       HELIOCENT         170.50       1.17.65       -39.93       HELIOCENT       -40.15         174.55       -85.66       -339.93       HELIOCENT       -91.76         175.67       1.25.67       -85.65       -34.51       HELIOCENT         174.65       -91.26       -33.95       HELIOCENT       -91.66         175.67       1.35.65       -94.66       -35.65       -91.7       -34.51         190.63 </td
154.73       -91923       -102.90       -60.17       HELIOCENT         158.69       -96.221       -100.40       -40.20       HELIOCENT         158.69       -96.213       -98.75       -40.13       HELIOCENT         158.69       -96.23       1.00.40       -40.13       HELIOCENT         158.63       1.00.53       -98.75       -40.13       HELIOCENT         170.63       1.00.330       -99.63       -40.13       HELIOCENT         170.60       1.00.330       -99.64       -39.93       HELIOCENT         178.55       1.1766       -99.55       -39.63       HELIOCENT         178.55       -88.56       -39.63       HELIOCENT       -39.63         178.55       -88.56       -39.34       HELIOCENT       -39.65         100.43       1.3128       -36.46       -39.36       HELIOCENT	154.73       -91923       -102.90       -60.17       HELIOCENT         158.69       -906221       -90.20       -60.17       HELIOCENT         158.69       190618       -948.70       -40.20       HELIOCENT         158.69       1906213       -948.70       -40.20       HELIOCENT         158.69       1906213       -948.70       -40.20       HELIOCENT         170.60       1906213       -948.70       -39.93       HELIOCENT         170.60       1.00300       -948.70       -39.93       HELIOCENT         170.60       1.01746       -92.60       -39.93       HELIOCENT         174.55       1.01746       -99.53       HELIOCENT       HELIOCENT         178.53       1.01746       -39.93       HELIOCENT       HELIOCENT         178.55       1.01746       -39.93       HELIOCENT       HELIOCENT         178.55       1.01728       -93.93       HELIOCENT       HELIOCENT         178.55       1.05622       -88.56       -39.65       HELIOCENT         178.55       1.05622       -88.56       -39.56       HELIOCENT         199.63       1.0128       -88.50       -39.51       -48.50       1.00.0000000000000000000000
ISB0.69       966611       100.80       40.20       HELIDGENT         162.66       100618       -90.613       HELIDGENT         150.63       100.913       -96.69       -40.13       HELIDGENT         170.60       100.913       -96.69       -40.13       HELIDGENT         170.60       100.913       -96.69       -40.13       HELIDGENT         170.61       100.913       -94.70       -339.93       HELIDGENT         178.55       100.930       -99.65       -39.63       HELIDGENT         178.55       -99.65       -39.63       HELIDGENT       -39.63         100.63       1.20.672       -88.66       -39.34       HELIDGENT         100.63       1.3128       -86.46       -36.96       HELIDGENT	150:09       996261       100.80       4ELINCENT         162:05       1000209       400.16       4ELINCENT         160:03       1000209       400.16       4ELINCENT         160:05       1000209       400.13       4ELINCENT         170:05       1000209       490.53       400.13       4ELINCENT         170:05       1000.00       -90.53       -90.53       4ELINCENT         170:05       1000.00       -399.93       4ELINCENT       4ELINCENT         170:05       1000.00       -390.63       4ELINCENT       4ELINCENT         170:05       1000.00       -390.63       4ELINCENT       4ELINCENT         170:05       1000.00       -390.63       -390.63       4ELINCENT         170:05       1000.00       -390.63       4ELINCENT       4ELINCENT         170:05       1000.00       -390.63       4ELINCENT       4ELINCENT         1000.05       1000.05       -390.65       -390.65       -390.65       4ELINCENT         10000.05       1000.05       -390.65       -390.65       -390.65       4ELINCENT         10000.05       1000.05       -390.65       -390.65       -390.65       4ELINCENT         1000000
Inc.us     Inc.us     Inc.us     Inc.us     Inc.us     Inc.us       IS60.63     Inc.us     190.013     190.013     Inc.us     Inc.us       I70.60     Inc.us     190.53     Inc.us     Inc.us     Inc.us       I70.65     Inc.us     Inc.us     Inc.us     Inc.us       I70.65     Inc.us     Inc.us     Inc.us       I70.55     Inc.us     Inc.us     Inc.us       I20.562     Inc.us     Inc.us     Inc.us       I20.652     Inc.us     Inc.us     Inc.us       I20.652     Inc.us     Inc.us     Inc.us       I20.652     Inc.us     Inc.us     Inc.us       I20.652     Inc.us     Inc.us     Inc.us	166.63       1000018       -40016       HELDGENT         170.60       100209       -90.73       -90.73       HELDGENT         170.60       100209       -39.93       HELDGENT         170.60       1010209       -39.93       HELDGENT         170.60       1010709       -39.93       HELDGENT         170.60       1010709       -39.93       HELDGENT         170.61       1010709       -39.93       HELDGENT         178.53       1010705       -39.93       HELDGENT         178.53       1010705       -39.93       HELDGENT         178.53       128.56       -39.43       HELDGENT         178.55       128.56       -39.53       HELDGENT         189.63       128.66       -39.53       HELDGENT         199.63       -60       -38.10       TARGET CT         199.63       -60       -38.10       TARGET CT
150.603       1.008.13      96.69       -40.13       HELIOGENT         170.60       1.00300       -94.70       -39.93       HELIOGENT         174.56       1.15.74       -92.60       -39.93       HELIOGENT         178.53       1.17746       -90.53       -39.63       HELIOGENT         178.53       1.17746       -98.56       -39.63       HELIOGENT         182.50       1.25652       -88.56       -39.63       HELIOGENT         185.67       1.85652       -88.56       -39.63       HELIOGENT         190.43       1.31128       -86.44       -36.96       HELIOGENT	170.63       1.00234       -94.70       -40.13       HELIOCENT         170.60       1.00234       -94.70       -339.93       HELIOCENT         170.65       1.01766       -94.70       -339.93       HELIOCENT         178.53       1.17766       -98.56       -339.93       HELIOCENT         178.53       1.17766       -339.53       HELIOCENT         178.55       1.85.66       -339.53       HELIOCENT         189.67       1.3128       -88.66       -339.51       HELIOCENT         199.63       1.56522       -83.07       -34.51       HELIOCENT         199.63       1.66766       -339.51       HELIOCENT       -339.55         199.63       -60       1.4667       -334.51       -40.67         199.63       -60       1.4667       -334.51       -40.67         199.63       -60       1.4667       -334.51       -40.67
170.60 1.0000 1.00000 -39.99 HELIOCENT. 174.56 1.007.4 -92.60 -39.93 HELIOCENT. 178.53 1.177.66 -90.53 -39.83 HELIOCENT. 182.50 1.20.62 -39.63 HELIOCENT. 185.67 1.26522 -88.56 -39.34 HELIOCENT. 190.43 1.31128 -84.44 -30.96 HELIOCENT.	170.00       1.0000       1.0000       -39.93       HFLIOGENT         174.56       1.1766       -92.60       -39.93       HFLIOGENT         178.53       1.1766       -90.63       HFLIOGENT         178.53       1.1766       -93.93       HFLIOGENT         178.53       1.1766       -93.43       HFLIOGENT         178.53       1.28062       -39.43       HFLIOGENT         185.60       1.28052       -88.56       -39.34       HFLIOGENT         185.61       1.26552       -88.56       -39.34       HFLIOGENT         190.63       1.51128       -88.56       -39.51       HFLIOGENT         199.63       1.51128       -81.07       -38.51       HFLIOGENT         199.63       1.6563       -38.51       -34.51       HFLIOGENT         199.63       1.6563       -38.10       TARGET CT       -37.55         102.33       1.657.55       -37.55       1.4667       CT
178.53 1.1774 - 72.60 -39.93 HELIOCENT. 178.53 1.17746 - 49.53 -39.83 HELIOCENT. 182.50 1.22052 -88.56 -39.63 HELIOCENT. 185.67 1.26522 -86.66 -39.34 HELIOCENT. 190.63 1.31128 -84.84 -30.96 HELIOCENT.	178.55       1.17.66       -92.60       -39.93       HELIOCENT.         178.53       1.17.66       -90.63       HELIOCENT.         178.53       1.17.66       -90.63       HELIOCENT.         178.53       1.27.66       -39.63       HELIOCENT.         185.67       1.82.56       -39.63       HELIOCENT.         185.67       1.82.56       -39.34       HELIOCENT.         185.67       1.82.56       -39.34       HELIOCENT.         190.63       1.31128       -84.44       -39.34       HELIOCENT.         190.63       1.31128       -83.07       -39.51       HELIOCENT.         194.37       1.356       -81.07       -38.51       HELIOCENT.         194.37       1.40.634       -31.25       -34.51       HELIOCENT.         194.37       1.40.634       -31.25       -34.51       HELIOCENT.
178-53 1.17746 -90.53 -39.83 HELIOCENT. 182.50 1.22652 -88.56 -39.63 HELIOCENT. 185.67 1.26522 -85.66 -39.34 HELIOCENT. 190.63 1.31128 -84.84 -30.96 HELIOCENT.	173053       1.1766       -90.53       -50.53       HELIOCENT.         102.50       1.22062       -88.56       -39.63       HELIOCENT.         105.67       1.25052       -88.56       -39.63       HELIOCENT.         105.67       1.26552       -88.56       -39.63       HELIOCENT.         105.67       1.26552       -88.56       -39.34       HELIOCENT.         190.43       1.31128       -84.44       -38.51       HELIOCENT.         194.37       1.35451       -84.44       -38.51       HELIOCENT.         194.37       1.46651       -38.51       HELIOCENT.       -37.55         194.37       1.466535       -47.55       -38.10       74667       CT.
105.67 1.2052 -00 4.6202 -00.50 -59.53 HELINCENT. 105.67 1.26522 -85.66 -39.34 HELINCENT. 1931128 -84.84 -30.96 HELINCENT.	105.67     1.620522     -08.59     -39.36     HELINCENT       105.67     1.626522     -85.66     -39.34     HELINCENT       190.63     1.31128     -84.44     -39.34     HELINCENT       190.63     1.33128     -84.44     -38.51     HELINCENT       190.63     1.33128     -84.44     -38.51     HELINCENT       194.37     1.35451     -83.51     HELINCENT       194.37     1.40634     -38.10     TARGET CT       202.33     1.456535     -37.55     TARGET CT
100.03 1.31128 -50 -39.36 HELIOCENT.	100.43     1.200.43     1.2128     -00.43     HELIOCENT       190.43     1.31128     -04.44     -33.96     HELIOCENT       199.40     1.35081     -83.07     -38.51     HELIOCENT       194.37     1.40434     -81.26     -38.51     HELIOCENT       194.37     1.40454     -31.55     1.4667     -37.55       102.33     1.45635     -37.55     1.4667     CT.
190+03 1-31128 -56. HELIOCENT.	190-03 1-31128 -80.84 333-96 HELIOCENT. 196-60 1-35831 -83.07 -38.51 HELIOCENT. 194-37 1-40434 -81.29 -83.10 TARGET CT. 202-33 1-4555 -79.46 -37.55 TARGET CT.
	196.%0 1.35431 -83.07 -38.51 HELIOCENT. 198.37 1.640434 -81.24 -38.10 TARGET CT. 202.33 1.665455 -79.46 -37.55 TARGET CT.
19%-%0 1+35481 +834.61 +384.51 HELIOCENT.	194.37 1.40434 -41.24 -38.10 TARGET CT. 202.33 1.45435 -79.46 -37.55 TARGET CT.
194-37 1-40434 -41-24 -38-10 TARGET CT.	202+33 1+45435 +59+46 +37+55 TARGET CT.
202-33 1+45435 <del>1</del> /9+46 <del>-</del> 37-55 TARGET CT.	

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for the heliocentric portion of the mission with the exception of the last two points. These are within the sphere of influence of Mars. At the bottom of this table the maximum range needed for communication subsystem design calculations is printed. The maximum range for this mission is 7.138  $\times 10^{11}$  feet. Unlike the Jupiter mission, the maximum range occurs at planet encounter.

## Schedules and Subschedules

The schedule and subschedules used for the Mars mission analysis are discussed in the following sections.

Schedule No. 1. Schedule No. 1 is shown in Figure A-5. This is the only schedule used in studying the Mars mission. This schedule consists of: launch operations, turning on the communications receiver, turning off the inertial sensing unit and computer and raising the attitude control dead band to 20 degrees at booster cutoff. Twenty decision points are then shown in the schedule, starting with one immediately after booster cutoff. The second decision point occurs 10 days into the mission with additional decision points every 10 days through 190 days. Decisions points, indicated by the operation code 90, permit the insertion of various subschedules into the main schedule as indicated by the user or by the schedule optimization routines at program execution time. Following the variable 20 decision points, two fixed decision points are included. These indicate that Subschedule No. 4 is to be used. These two points are at 196 days and 198 days into the mission. Subschedule 4, described in a later paragraph, is an update and correction subschedule which utilizes planetary approach radar.

Subschedule\_No. 1. Subschedule No. 1 is shown in Figure A-6a. This subschedule is the basic update and correction subschedule for the heliocentric portion of the mission. The subschedule begins with the turning on of the computer, ISU, Sun sensor, star tracker, attitude control, and transmitter. Three minutes later, the spacecraft is allowed to begin maneuvering to acquire the Sun and Canopus, the specified star. One minute is allowed for maneuvering to the nominal orientation of the spacecraft which would provide the required lines of sight to these celestial bodies. One minute is then allowed for searching about the nominal orientation for final acquisition. Thirty-two minutes into the subschedule, the dead band is dropped to 0.1 degree. This narrow dead band is required to insure precise pointing of the spacecraft for the midcourse correction burn. Thirty-three minutes into the schedule, an update takes place with any DSIF radar which can view the spacecraft at that time. Thirty-four minutes into the subschedule, the midcourse correction is made. It should be noted that the midcourse correction is made to zero the computed deviations of position at the time indicated on the right-hand side of Subschedule No. 1. This time,  $1.6934 \times 10^7$ seconds into the mission, is the time at which the spacecraft will cross into the Mars sphere of influence. A later midcourse correction will take place at that time to zero the velocity deviations as the spacecraft comes back onto the nominal trajectory at the Mars sphere of influence. After the midcourse correction, the dead band is raised to 20 degrees and the subsystems required for making the correction are turned off to terminate the subschedule.

+2000000E+02 •0-•0= •0= : •01 •0-•0• . • •01 . . • c -• • • •0-•0-• C -;;; •0-•01 •0= • 0 •••• •0-• • • • • • •0--0 1 ÷: INAL UPDATE/CORRECTION APPRUACH CORRECTION ND OF THE SCHEDULE TURN OFF ISU TURN OFF COMPUTER RAISE DEAD BAND IURN ON COM.RCVR. START THE LAUNCH POINT POINT POINT TNIUT DECISION POINT DECISION POINT POINT PUINT TNIOG POINT POINT POINT POINT POINT POINT FUI04 PUINT PUINT PUINT DECISION POINT DECISION P DECISION P DECISION P DECISION DECISION DECISION DECISION DECISION DECISION DECISION DECESION 000 Ŷ Ŷ ŝ 0 1 Ŷ î 0 01 ĉ î î Ŷ Ŷ ĉ î î ĉ -Ŷ î ĉ C 1 °i ĉ î î î Ŷ ĉ °i î ŝ Ŷ î Ŷ Ŷ 01 î ŝ 00 î î 4 4 <sup>N</sup> 06 005 0000000 0000 5 9 30 0.6 66 05 5 050 2592000.00 7776000,00 8660000,00 5184000.00 6048000.00 6912000.00 11232000.00 (6%16000.00 16934400.00 17158949.00 844000.00 1728000.00 3447.00 4320000.00 9504000°00 0358000.00 29400000.00 3824000.00 4658000.00 00.0053625 0 • 0 n S 0.005 0.00S 0.00S 500°0 0.005 00 0H57M27.005 0.005 .00S 0.005 0.005 0.005 0,005 0.005 0°°° 0.005 0.003 500.0 0,005 0.00.5 9.005 0.003 0PT.8 ΣÔ МO 20 MO 20 ΣC Σ C МO 20 20 50 Σ чc ž 10 MO NC C N C 50 ۲. 0 20 このやまだ I I I I н С I I C C ЧC Т С H C HC л 0 25 х о I C 7 0 10 нс С I C нç I C Т С 1980 00% 7010 0031 1980 1980 0002 400 005 6009 002 9 0 D 006 001 0021 1300 000 009 (100) 000

 ALL SYSTEMS NON OPTICAL OPTICAL

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SCHEDULE NO.

e i FIGURE A-5. MARS MISSION, MAIN SCHEDULE

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<u>Subschedule No. 2</u>. Subschedule No. 2 is shown in Figure A-6b and is a simple update subschedule with only one instruction, update with any DSIF radar which can view the spacecraft at that time.

<u>Subschedule No. 3</u>. Subschedule No. 3 is a planetary approach update with no midcourse correction. The subschedule begins by turning on the approach radar and transmitter. Two minutes later the update is performed. The approach radar and the transmitter are subsequently turned off. The transmitter is used during this subschedule because it is assumed that the data obtained from the planetary approach radar must be telemetered to the earth for processing. This subschedule is shown in Figure A-6c.

<u>Subschedule No. 4</u>. Subschedule No. 4 is shown in Figure A-6d. This subschedule is a planetary approach update and midcourse correction schedule. The subschedule begins by turning on the computer, the ISU Sun sensor, star tracker, attitude control system, and approach radar. The star and Sun acquisition maneuvering and search are performed followed by dropping the dead band and updates, first with the DSIF, then the approach radar, prior to making the midcourse correction. The dead band is then raised back to 20 degrees and the subsystems are turned off to complete the subschedule.

<u>Subschedule No. 5</u>. Subschedule No. 5 is shown in Figure A-6e. This subschedule is identical to Subschedule No. 1 with the exception that the midcourse correction is made to zero the computed deviations of position at the target time (nominal periapsis of planetary swingby).

Subschedules No. 3 and 5 were not used in the example of the Mars astrionics analysis presented in this appendix.

## Midcourse Correction Optimization

The search for an optimum correction schedule on the Mars trajectory, is shown in Figures A-7a and A-7b. The times of the decision points as specified in Schedule 1 are shown at the top of Figure A-7a. Subschedule No. 2 is used for updates and Subschedule No. 1 for corrections. Schedule No. 1 includes two midcourse corrections after the 20 decision points. Thus, the final correction which is always required in Pfeiffer's technique does not appear in this printout since it does not occur at one of the decision points. The penalty for the optimum schedule is 3498.52 pounds with corrections at the first, seventh, and eleventh decision points. The algorithm encounters this combination of corrections on the first line of Figure A-7b. The remainder of Figure A-7b shows that additional midcourse corrections beyond the eleventh decision point do not decrease the penalty any further. A complete optimum schedule is shown in Figure A-8.

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FIGURE A-7a. MARS MISSION, SCHEDULE OPTIMIZATION

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FIGURE A-8a. MARS MISSION, DETAILED SCHEDULE OUTPUT

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FIGURE A-8b. MARS MISSION, DETAILED SCHEDULE OUTPUT

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0.0         0.0         0.005         6         1         -0.         SUBSYSTER 0W/OFF           0.0         8440000         1000 04 04 0.005         6         4         1         -0.         SUBSYSTER 0W/OFF           0.0         84400000         1000 04 04 0.005         6         4         1         -0.         SUBSYSTER 0W/OFF           0.0         84400000         1000 04 04 0.005         6         3         1         -0.         SUBSYSTER 0W/OFF           1800.0         84400000         1000 0430         0.005         6         3         1         -0.         SUBSYSTER 0W/OFF           1800.0         8441800         1000 0430         0.005         9         2         2         -0.         SUBSYSTER 0W/OFF           60.0         8441800         1000 0431         0.005         9         2         2         -0.         SUBSYSTER 0W/OFF           60.0         8441800         1000 0432         0.005         9         2         2         -0.         SUBSYSTER 0W/OFF           60.0         8441800         1000 0434         0.005         9         1         -0.         SUBSYSTER 0W/OFF           60.0         844100         1000         04354		8640000.0	1000 OH	00 0.00S	<b>9</b>		-	2.1000E+03	SUBSYSTEM ON/OFF
Right Cold		8640000.0	1000 OH	00 0 W0				• • • • • • • • • • • • • • • • • • •	SUBSYSTEM ON/OFF
0.0         564000.0         100         0.0         0.005         5         3         1         -0.         SUBSYSTEM ON/OFF           0.0         864000.0         1000         0         0.005         6         3         1         -0.         SUBSYSTEM ON/OFF           0.0         8641900.0         1000         0         0.005         6         3         1         -0.         SUBSYSTEM ON/OFF           1800.0         8641970.0         1000         0         0.005         9         2         2         -0.         SIAR MANYSEARCH           0.0         8641970.0         1000         0         0         0.05         9         2         2         -0.         SIAR MANYSEARCH           0.0         8641970.0         1000         0         0         0         2         2         -0.         SIAR MANYSEARCH           0.0         8641970.0         1000         0         0         0         2         2         -0.         SIAR MANYSEARCH           0.0         86471970.0         1000         0         0         0         2         0         2         0         2         0         2         0         2         0		8640000.0	LONU OH	0M 0.00S	• <b>D</b>	5 S	<b></b>	•0•	SUBSYSTEM ON/OFF
0.0         564000.0         100         0.0         0.005         5         3         1         -0.         SUBSYSTEM ON/OFF           1800.0         564000.0         1000         040         0.005         5         3         1         -0.         SUBSYSTEM ON/OFF           1800.0         564900.0         1000         04304         0.005         5         2         -0.         STAR MANY/SEARCH           60.0         664970.0         1000         04310         0.005         9         2         -0.         STAR MANY/SEARCH           60.0         664970.0         1000         0432M         0.005         9         2         -0.         STAR MANY/SEARCH           60.0         664970.0         1000         0432M         0.005         9         2         -0.         STAR MANY/SEARCH           60.0         664970.0         1000         0432M         0.005         3         0         2         -0.         STAR MANY/SEARCH           60.0         664970.0         1000         0432M         0.005         3         0         2         -0.         STAR MANY/SEARCH           60.0         664970.0         1000         0432M         0.005         0		8640000.0	1000 0H	0M, 0.00S	<b>.</b>	4		• 0 •	SUBSYSTEM ON/OFF
944000.0         10.0         0.005         6         9         1         -0.         SIGR MANVSEARCH           1800.0         8441960.0         1000         04300         0.005         9         2         -0.         SIAR MANVSEARCH           60.0         8441960.0         1000         04310         0.005         9         2         -0.         SIAR MANVSEARCH           0.0         8441960.0         1001         04320         0.005         9         2         3         0.         SIAR MANVSEARCH           0.0         8441970.0         1001         04320         0.005         9         2         3         0.         5         -0.         SIAR MANVSEARCH           0.0         8441970.0         1001         04320         0.005         9         2         3         0         5         -0.         SIAR MANVSEARCH           60.0         8441720.0         1001         04320         0.005         3         0         5         -0.         SIAR MANVSEARCH           60.0         8442100.0         1001         04350         0.005         5         1         -0.         SUBSYSTEM ON/OFF           0.0         8442100.0         1001         0435		8640000-0	100U 0H	00 0.00S	<b>9</b>	m	1	•0•	SUBSYSTEM ON/OFF
60.0         6441800.0         1040         0.033         9         2         2         -0.         51an many/5EaRch           60.0         644186.0         1000         04310         0.005         9         2         2         -0.         51an many/5EaRch           60.0         644195.0         1000         04328         0.005         9         2         3         -0.         51an many/5EaRch           60.0         644195.0         1000         04328         0.005         9         2         3         -0.         51an many/5EaRch           60.0         644195.0         1000         04328         0.005         3         0         5         -0.         1.0006E.01         CHANGE DEADHAND           60.0         8642100.0         1000         04334         0.005         3         0         5         -0.         2.0006E.01         CHANGE DEADHAND           60.0         8642100.0         1000         04354         0.005         5         0         -0         2.0006E.01         CHANGE DEADHAND           60.0         8642100.0         1000         04354         0.005         5         0         -0         2.0006E.01         CHANGE DEADHAND <t< td=""><td></td><td>8440000.0</td><td>100U 0H</td><td>S00.0 M0</td><td><b>•</b> •</td><td>6</td><td>-</td><td>•0-</td><td>SUBSYSTEM ON/OFF</td></t<>		8440000.0	100U 0H	S00.0 M0	<b>•</b> •	6	-	•0-	SUBSYSTEM ON/OFF
0000         044186.0         1000         0431M         0.005         9         2         -0.         51AR         MAUV/SEARCH           0.0         R641970.0         1000         0432M         0.005         9         2         3         -0.         51AR         MAUV/SEARCH           0.0         R641970.0         1000         0432M         0.005         8         -0         1.0000E-01         CHAMEE         DEADBAND           60.0         R641970.0         1000         0433M         0.005         3         0         5         -0         1.0000E-01         CHAMEE         DEADBAND           60.0         R641970.0         1000         0433M         0.005         3         0         5         -0         1.0000E-01         CHAMEE         DEADBAND           60.0         R641970.0         1000         0435M         0.005         5         0         -0         1.059344.01         CHAREE         DEADBAND           60.0         R642100.0         1000         0435M         0.005         5         0         -0         D         DESYSTEM         DN/OFF           0.0         R642100.0         1001         0435M         0.005         5         0		8641800.0	LOGU OH	304 0.00S		N	- -	• 0 •	STAR MANV/SEARCH
000         R641970.0         100/0 0H32M         0.005         9         2         3         -0.         51AR         MANV/SEAHCH           0.0         R641920.0         100/0 0H32M         0.005         B         -0         1.0000E-01         CHAMGE DEAUBAND           60.0         R641950.0         100/0 0H32M         0.005         B         -0         1.0000E-01         CHAMGE DEAUBAND           60.0         R641950.0         100/0 0H32M         0.005         B         -0         1.0000E-01         CHAMGE DEAUBAND           60.0         R641960.0         100/0 0H35M         0.005         B         -0         1.69934E.07         MIDCOUNSE CORRECTIO           60.0         R642100.0         100/0 0H35M         0.005         B         -0         -0         2.0000E-01         CHAMGE DEAUBAND           0.0         R642100.0         100/0 0H35M         0.005         B         -0         -0         2.0000E-01         CHAMGE DEAUBAND           0.0         R642100.0         100/0 0H35M         0.005         B         -0         -0         -0         SUBSYSTEM ON/OFF           0.0         R642100.0         100/0 0H35M         0.005         B         -0         -0         SUBSYSTEM ON/OFF		8641860.0	100U 0H	31M 0.005	·	N	2	• 0 •	SIAR MANV/SEARCH
60:0         64:1970.0         1000         0H33M         0.005         8         -0         1.0000E-01         CHANGE DEADBAND           60:0         66+1970.0         1000         0H33M         0.005         3         0         5         -0.         RADAR UPDATE           60:0         86+21970.0         1000         0H33M         0.005         3         0         5         -0.         RADAR UPDATE           60:0         86+2100.0         1000         0H35M         0.005         5         0         -0.         SUBSYSTEM ON/OFF           0.0         86+2100.0         1000         0H35M         0.005         6         3         -0.         SUBSYSTEM ON/OFF           0.0         86+2100.0         1000         0H35M         0.005         6         1         -0         SUBSYSTEM ON/OFF           0.0         86+2100.0         1000         0H35M         0.005         6         1         -0         SUBSYSTEM ON/OFF           0.0         86+2100.0         1000         0H35M         0.005         6         1         -0         SUBSYSTEM ON/OFF           0.0         86+2100.0         1000         0H35M         0.005         6         1         -0 <td></td> <td>8641920.0</td> <td>1000 OH</td> <td>32M 0.005</td> <td>6</td> <td>N</td> <td>m</td> <td>• 0 •</td> <td>STAR MANV/SEARCH</td>		8641920.0	1000 OH	32M 0.005	6	N	m	• 0 •	STAR MANV/SEARCH
000000         R641980.0         10.0         0.4334         0.005         3         0         5         -0.         RADAR UDUATE           60.0         864200.0         1000         04344         0.005         5         0         1.6934E+07         MIDCOURSE CORRECTIO           60.0         8642100.0         1000         04354         0.005         8         -0         2.0000E+01         CHANGE DEADHAND           0.0         8642100.0         1000         04354         0.005         6         3         -0         -0         SUBSYSTEM ON/OFF           0.0         8642100.0         1000         04354         0.005         6         3         -0         -0         SUBSYSTEM ON/OFF           0.0         8642100.0         1000         04354         0.005         6         2         -0         -0         SUBSYSTEM ON/OFF           0.0         8642100.0         1000         04354         0.005         6         2         -0         -0         SUBSYSTEM ON/OFF           0.0         8642100.0         1000         04354         0.005         6         2         -0         -0         SUBSYSTEM ON/OFF           0.0         8642100.0         1000         <		8641920.0	100U 0H	32M 0.005	- <b>10</b> 	ĩ	0	1.0000E-01	CHANGE DEADBAND
0.00         0.00½         0.134M         0.00½         0.135M         0.100½         0.135M         0.100½         0.135M         0.100%         0	0 * 0 ¥	8641980.0	1000 0H	500.0 MEE	<b>E</b>	0	S	• 0 •	RADAR UPDATE
0.0         8642100.0         10.0         0.435M         0.005         B         -0         2.0000E+01         CHANGE DEADHAND           0.0         R642100.0         1000         0435M         0.005         6         9         0         -0.         SUBSYSTEM ON/OFF           0.0         R642100.0         1000         0435M         0.005         6         3         -0         -0.         SUBSYSTEM ON/OFF           0.0         R642100.0         1000         0435M         0.005         6         3         -0         -0.         SUBSYSTEM ON/OFF           0.0         R642100.0         1000         0435M         0.005         6         2         -0         -0.         SUBSYSTEM ON/OFF           0.0         R642100.0         1000         0435M         0.005         6         2         -0         -0.         SUBSYSTEM ON/OFF           0.0         R642100.0         1000         0435M         0.005         6         2         -0         -0.         SUBSYSTEM ON/OFF           0.0         R642100.0         1000         0435M         0.005         6         2         -0         -0.         SUBSYSTEM ON/OFF           0.0         R642100.0         1100		8642040.0	100U 0H	34M 0.005	цу	0	0	1.6934E+07	MIDCOURSE CORRECTIO
0.0       R442100.0       1000       0.435M       0.005       6       3       -0       SUBSYSTEM ON/OFF         0.0       R.42100.0       1000       0435M       0.005       6       3       -0       SUBSYSTEM ON/OFF         0.0       R.42100.0       1000       0435M       0.005       6       1       -0       SUBSYSTEM ON/OFF         0.0       R.42100.0       1000       0435M       0.005       6       2       -0       -0       SUBSYSTEM ON/OFF         0.0       R.42100.0       1000       0435M       0.005       6       2       -0       -0       SUBSYSTEM ON/OFF         0.0       R.42100.0       1000       0435M       0.005       6       2       -0       -0       SUBSYSTEM ON/OFF         0.0       R.42100.0       1000       0435M       0.005       6       2       -0       -0       SUBSYSTEM ON/OFF         0.00       R.42100.0       1000       0433M       0.005       6       5       -0       -0       SUBSYSTEM ON/OFF         0.005       R.42100.0       1010       0433M       0.005       3       0       5       -0       SUBSYSTEM ON/OFF         .0000.0		8642100.0	100U 0H	35M 0•00S	30	Ĩ	0	2.0000E+01	CHANGE DEADBAND
0.0       8.42100.0       1000       0H35M       0.005       6       3       -0       -0       SUBSYSTEM       0N/OFF         0.0       8.642100.0       1010       0H35M       0.005       6       1       -0       -0       SUBSYSTEM       0N/OFF         0.0       8.642100.0       1000       0H35M       0.005       6       2       -0       -0       SUBSYSTEM       0N/OFF         0.0       8.642100.0       1000       0H35M       0.005       6       2       -0       -0       SUBSYSTEM       0N/OFF         0.0       8.642100.0       1000       0H35M       0.005       6       2       -0       -0       SUBSYSTEM       0N/OFF         0.0       9505980.0       11/0       0H35M       0.005       3       0       5       -0       SUBSYSTEM       0N/OFF         38600.0       9505980.0       11/10       0H33M       0.005       3       0       5       -0       SUBSYSTEM       0N/OFF         4000.0       1070       0H33M       0.005       3       0       5       -0       SUBSYSTEM       0N/OF         4000.0       1172/39990.0       1370       0H33M       0.005		R642100.0	1000 04	35M 0+00\$	<b>9</b>	6	0	• • •	SUBSYSTEM ON/OFF
0.0       8642100.0       10ñU       0435M       0.005       5       1       -0       -0       SUBSYSTEM       0N/OFF         0.0       8642100.0       100U       0435M       0.005       6       2       -0       -0       SUBSYSTEM       0N/OFF         0.0       8642100.0       100U       0435M       0.005       6       2       -0       -0       SUBSYSTEM       0N/OFF         0.0       8642100.0       100U       0435M       0.005       6       5       -0       -0       SUBSYSTEM       0N/OFF         3800.0       9505980.0       11/0U       0433M       0.005       3       0       5       -0       -0       SUBSYSTEM       0N/OFF         4000.0       1070       120U       0433M       0.005       3       0       5       -0       RADAR       UPDATE         4000.0       11233980.0       120U       0433M       0.005       3       0       5       -0       RADAR       UPDATE         4000.0       11233980.0       137U       0433M       0.005       3       0       5       -0       RADAR       UPDATE         4000.0       117233980.0       147U <t< td=""><td></td><td>8642100.0</td><td>1000 0H</td><td>35M 0.00S</td><td><b>.</b></td><td>м</td><td>0</td><td>•0 •</td><td>SUBSYSTEM ON/OFF</td></t<>		8642100.0	1000 0H	35M 0.00S	<b>.</b>	м	0	•0 •	SUBSYSTEM ON/OFF
0.0       8642100.0       100U       0435M       0.005       6       2       -0       -0.       SUBSYSTEM       0N/OFF         0.0       8662100.0       100U       0435M       0.005       6       4       -0       -0.       SUBSYSTEM       0N/OFF         3800.0       8662100.0       100U       0435M       0.005       6       5       -0       -0.       SUBSYSTEM       0N/OFF         3800.0       9505980.0       110U       0433M       0.005       3       0       5       -0.       RADAR       UPDATE         4000.0       1050980.0       120U       0433M       0.005       3       0       5       -0.       RADAR       RDAR       UPDATE         4000.0       117/33990.0       130U       0433M       0.005       3       0       5       -0.       RADAR       RDAR       UPDATE         4000.0       117/33990.0       130U       0433M       0.005       3       0       5       -0.       RADAR       RDAR       UPDATE         4000.0       117/33990.0       150U       0433M       0.005       3       0       5       -0.       RADAR       UPDATE         4000.0		8642100.0	10 UN 0H	35M 0.00S	¢	1	0	•	SUBSYSTEM ON/OFF
0.0         6.642100.0         100U         0H35M         0.005         6         4         -0         SUBSYSTEM         ON/OFF           3880.0         86.42100.0         100U         0H35M         0.005         6         5         -0         -0         SUBSYSTEM         ON/OFF           3880.0         9505980.0         110U         0H33M         0.005         3         0         5         -0         80SYSTEM         ON/OFF           4000.0         10750980.0         110U         0H33M         0.005         3         0         5         -0         RADAR         UPDATE           4000.0         10750980.0         120U         0H33M         0.005         3         0         5         -0         RADAR         UPDATE           4000.0         11733980.0         130U         0H33M         0.005         3         0         5         -0         RADAR         UPDATE           4000.0         17793980.0         14ñU         0H33M         0.005         3         0         5         -0         RADAR         UPDATE           4000.0         12791980.0         15nU         0H33M         0005         3         0         5         -0         <		8642100.0	1000 0H	35M 0.005	<b>9</b>	N	0 1	•0•	SUBSYSTEM ON/OFF
3880.0         9505980.0         100U 0H35M 0.005         6         5         -0         0.         SUBSYSTEM ON/OFF           4000.0         9505980.0         110U 0H33M 0.005         3         0         5         -0.         RADAR UPDATE           4000.0         10750980.0         120U 0H33M 0.005         3         0         5         -0.         RADAR UPDATE           4000.0         10750980.0         120U 0H33M 0.005         3         0         5         -0.         RADAR UPDATE           4000.0         11723980.0         130U 0H33M 0.005         3         0         5         -0.         RADAR UPDATE           4000.0         1777980.0         14ñU 0H33M 0.005         3         0         5         -0.         RADAR UPDATE           4000.0         17791980.0         15ñU 0H33M 0.005         3         0         5         -0.         RADAR UPDATE		8642100.0	1000 0H	35M 0.00S	Ð	4	0	•01	SUBSYSTEM ON/OFF
9505980.0       110U       0H33M       0.005       3       0       5       -0.       RADAR       UPDATE         4000.0       10567980.0       120U       0H33M       0.005       3       0       5       -0.       RADAR       UPDATE         4000.0       10567980.0       120U       0H33M       0.005       3       0       5       -0.       RADAR       UPDATE         4000.0       11233990.0       130U       0H33M       0.005       3       0       5       -0.       RADAR       UPDATE         4000.0       120Y7980.0       140U       0H33M       0.005       3       0       5       -0.       RADAR       UPDATE         4000.0       120Y1980.0       150U       0H33M       0.005       3       0       5       -0.       RADAR       UPDATE		8642100.0	1000 0H	35M 0.005	<b>y</b>	ມ	01	• 0 •	SUBSYSTEM ON/OFF
4000.0 4000.0 11743990.0 1200 0H33M 0.005 3 0 5 _0. RADAR UPDATE 4000.0 1207980.0 14ñD 0H33M 0.005 3 0 5 _0. RADAR UPDATE 4000.0 12051980.0 15ñD 0H33M 0.005 3 0 5 _0. RADAR UPDATE		9505980.0	11 AU 0H	33M 0.005	en	0	S	•0-	RADAR UPDATE
000000 11243990.0 1300 0433M 0.005 3 0 5 -0. RADAR UPDATE RADAR UPDATE 12077980.0 1460 0433M 0.005 3 0 5 -0. RADAR UPDATE 12000.0 12051980.0 1560 0433M 0.005 3 0 5 -0. RADAR UPDATE		10369980.0	120U 0H	33M 0.005	m	0	ŝ	•0•	RADAR UPDATE
0.000.0 12097980.0 1400 0H33M 0.005 3 0 5 .0. RADAR UPDATE 8400.0 12091980.0 1500 0H33M 0.005 3 0 5 .0. RADAR UPDATE		11233980.0	1300 0H	33M 0.005	Ð	0	ŝ	• <b>U</b> -	RADAR UPDATE
12951980.0 1500 0H33M 0.005 3 0 5 .0. RADAR UPDATE		12027980.0	140U 0H	33M 0:00S		0	ŝ	•0-	RADAR UPUATE
	0.000.40	12951980.0	1500 OH	330 0 MEE	сл С	0	ហ	-0-	RADAR UPDATE

FIGURE A-8c. MARS MISSION, DETAILED SCHEDULE OUTPUT

SCHEDULE DUMP CONTINUED

	13825980.0	160U 0H	33M 0.00S	<b>m</b>	0	ហ	• • • •	RADAR UPD
864000•0 864000•0	14699980•0	17AU OH.	33M 0.005		0	្រុ	•0•	RADAR UPD
	15553980.0	1800 OH	33M 0.005	<b>C</b>		ហ	-0-	RADAR UPD
0.000000	16417980.0	1900 OH.	33M 0.00S		0	ហ	• 0 •	RADAR UPD
	16934400.0	196U 0H	0M. 0.00S	<b>ч</b> р 			••••	SUBSYSTEM
	16934400.0	196U 0H	0M 0.005	<b>9</b>	N		• 0 •	SUBSYSTEM
	16934400+0	196U 0H	00 0°002		ស		• • • •	SUBSYSTEM
	16934400.0	195 <sup>0</sup> 0H	04 0.00S	φ	4		-0-	SUBSYSTEM
	16934400.0	1960 OH	00 0°008	γ <b>ς</b>	<b>m</b>	-	-0.	SUBSYSTEM
	16934400.0	1960 04	0M 0.00S	<b>.</b>	80	-	• • • • •	SUBSYSTEM
	16934200.0	196U 0H.	300 0 WUE	6	2	. <b>1</b>	• 0 •	STAR MANY
	16936260•0	196U 0H	31M 0.005	5		~	-0-	STAR MANV
	16936320.0	196U 0H	\$00°0 WZ2	6	N	60	• 0 •	STAR MANV
	16936320•0	196 <sup>0</sup> 0H	32M 0.00S	T	01	0	1.00005-02	CHANGE DE
	16936380.0	195U 0H	33M 0.005		0	ŝ	•0-	RADAR UPD
	16936380.0	1940 0H	33M 0.005	10	Î	0	• • •	HOR, SENSO
	16934440.0	1960 0H	34M 0.00S	ŝ	•	0	• • • • •	MIDCOURSE
	1693650000	1960 OH:	35M 0.00S	æ	01	0	2+0000E+01	CHANGE DE
	16936500•0	196U 0H.	35M_0•00S	£	8	0	• 0 1	SUBSYSTEM
	16936500•0	1960 0H.	35M 0.005	\$	m	0.	-0-	SUBSYSTEM
	16936500 <b>。</b> n	196U 0H.	35M 0.00S	<b>9</b>		0 	•0•	SUBSYSTEM
	16936500.0	196U 0H	35M 0,00S	ъ¢	N	0	•0 •	SUBSYSTEM
0.00			0 ° ° ° ° ° ° ° ° ° ° ° ° ° ° ° ° ° ° °					

RADAR UPDATE	RADAR UPDATE	SUBSYSTEM ON/OFF	SUBSYSTEM ON/OFF	SUBSYSTEM ON/OFF	SUBSYSTEM ON/OFF	SUBSYSTEM ON/OFF	SUBSYSTEM ON/OFF	STAR MANV/SEARCH	STAR MANV/SEARCH	STAR MANV/SEARCH	02 CHANGE DEADBAND	RADAR UPDATE	HOR, SENSOR UPDATE	MIDCOURSE CORRECTION	01 CHANGE DEADHAND	SUBSYSTEM ON/OFF	SUBSYSTEM ON/OFF	SUBSYSTEM ON/OFF	SUBSYSTEM ON/OFF	SUBSYSTEM ON/OFF	SUBSYSTEM ON/OFF
-0-	•0•	•0•	•0•	• 0 •	• 0 •	• 0 •	• • • • • • • • • • • • • • • • • • • •	-0-	• 0 •	• 0 •	1.00005-	•0•	• 0 •	• 0 •	2+000E+	• 0 1	• 0 •	• 0 •	• 0 •	• 0 =	• 0 •
ິມ ເ	ស				-	7	-		~~~	m	0	۲ ۲	0	0	0	0	0	0 	0	0 I	01
0	0		N	ស	4	n N	00	2	N	N	01	0	01	0	0 I	æ	m	7	N	4	ഹ
e	с С	<b>.</b>	9 9	<b>9</b>	ър Г	Ş	<b>\$</b>	6	5	6	£	n N	10	ۍ ا	x	Ŷ	\$	<b>9</b>	¢	Q.	¢
1800 0H33M 0.005	1900 0H33M 0.00S	196U 0H 0M. 0.00S	196U 0H 0M 0.005	196U 0H 0M 0,00S	1950 OH OM 0.00S	1960 0H 0M 0.005	196U 0H 0M 0.00S	1960 0H30M 0.005	196U 0H31M 0.00S	1960 0H32M 0.005	1969 M3EH0 0,00S	194U 0H33M 0.005	194U 0H33M 0.005	1950 0H34M 0.00S	1960 0H35M 0.00S	1960 0H35M 0.005	1960 0H35M 0.005	1964 0H35M 0.005	1960 0H35M 0.00S	196U 0H35M 0.00S	1940 0H35M 0.00S
15553980.0	16417980.0	16934400.0	16934400.0	16934400.0	16934400.0	16934400.0	16934400.0	16934200.0	16936260•0	16936320.0	16936320•0	16936380.0	16936380.0	16934440.0	16936500.0	16936500•0	16936500 <b>。</b> 0	16936500.0	16936500.0	16936500•0	16936500.0

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SUBSYSTEM ON/OFF

FIGURE A-8d. MARS MISSION, DETAILED SCHEDULE OUTPUT

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0.0

SCHEDULE DUMP CONTINUED

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17107200.0     1980     04 00       17107200.0     1980     04 00       17107200.0     1980     04 00       17107200.0     1980     04 00       17107200.0     1980     04 00       17109060.0     1980     04320       17109120.0     1980     04320       17109120.0     1980     04320       17109180.0     1980     04320       17109180.0     1980     0433       17109180.0     1980     0433       17109180.0     1980     0433       17109180.0     1980     0433       17109180.0     1980     0433
17107200.0     195       17107200.0     195       17107200.0     195       17107200.0     196       17109200.0     196       17109120.0     196       17109120.0     196       17109120.0     196       17109120.0     196       17109120.0     196       17109120.0     196       17109180.0     196       1710920.0     197       1710920.0     197

FIGURE A-8e. MARS MISSION, DETAILED SCHEDULE OUTPUT

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## Detailed Results of the Mars Mission Using the Optimal Midcourse Correction Strategy

The remainder of this appendix presents the error analysis and penalty evaluation results for the reference astrionics system on the optimal midcourse correction schedule. The reference system consists of three Arma D4-E accelerometers, three GG334A gyroscopes, a designed inertial sensing unit, the SRT Ruk-2 computer, the ITT Lunar Orbiter star tracker, the Adcole 1402 Sun sensor, the MCR503X receiver, a hypothetical approach radar named APP. RAD. X and a designed transmitter.

## <u>Error Analysis</u>

Detailed printing of the error analysis results as the schedule is evaluated are shown in Figures A-9a throught A-9g. The analysis begins at the top of Figure A-9a with the launch vehicle on the pad at time equal to zero. It should be noted that there are no initial errors in position, velocity, or attitude in this analysis. After the initial conditions, the next block of printing occurs 57 minutes, 27 seconds, into the mission. Between time equals zero and this point, the launch vehicle burns for 617.7 seconds, the errors for that burn are added, the last launch vehicle stage and spacecraft remains in the parking orbit for 41 minutes, 58 seconds, and a second burn of 310 seconds is initiated. The errors for the final burn are added and approximately one second of coasting occurs before the print point. The print point is approximately at final launch vehicle cutoff. Since the launch vehicle burns are assumed to be under perfect closed loop control, the computed deviations are zero. The deviations and errors are equal and are as shown. The strapdown error analysis program, SEAP, was not rerun for the Mars booster trajectory. Thus these injection errors are only approximate as the sensitivities used are those for the Jupiter mission.

The first decision point occurs immediately after booster cutoff and the schedule optimizing technique indicates a midcourse correction should be performed at the first decision point. The ISU and the computer are shown turned off but are turned back on immediately to initiate the midcourse correction subschedule. In actual flight operations, the ISU and computer would not be turned off and then turned back on again. The operations for a midcourse correction as indicated in Subschedule No. 1 are performed as shown. A midcourse correction with an expected value of 136 feet per second is indicated at the top of Figure A-9b. The subschedule concludes with the turning off of the various subsystems necessary to perform the midcourse correction. The analysis then continues with updates performed as indicated in Subschedule No. 2 for those decision points where the optimizing routine indicated only updates should be performed. The midcourse correction sequence is then repeated at the seventh and the eleventh decision points as specified by the midcourse optimization technique.

The final corrections within the Mars sphere of influence are made beginning at 196 days into the mission as shown in Figure A-9x. These corrections

.854E-01 0065-03 854E-01 8545-01 .425E=01 .854E-01 545-0 ć, ΰ 40-V A-OP 845 dC-V .005E \$ • • \$ XMITTER 7.095E-04 7.097E-04 ÷ \* 101 .854E-01 5 01 . 854E-0 • 854E-0 . 8545mn 0.000.0 1.0000 0.000.0 ٩ \$ PITCH A-CR 0 A-CR 2.284E • 854E \$ 133.7760 \$ \$ SUN SENSOR ISU/C.P.S. COM. RCVR. APP. RADAR ≑ \$ Q o 1.0000 0.0000 0.0000 ¢ ¢ 7.109E-04 7.112E-04 7.854E-01 7.854E-01 YAW 10-2.284E-07 .8545-01 7.854E-01 0 \$ ¢ A-UR A-DR ٥ A-UR 0 \*.854E LONG \$ 0000000 1.0000 0.000.0 \$ ROLL 0 ¢ ¢ 1.6165+01 1.6165+01 1.616E+01 1.616E+01 ¢ \$ -34.1414 1.416E+01 1.416E+01 \$ \$ ¢ 40**-**0 2 2 A A 40-V 40-N d0-1 ⊅ \$ \$ ò ٥ • .... •4168 •8507 \$ .2923 ± LAT= \$ ¢ 3.461E+01 3.461E+01 3.461E+01 3.461E+01 3.461E+01 3.461E+01 <u>а</u> \$ ÷ \$ ¢ \$ 2-08 4-CR a0+2 **V-C**R ¢ ¢ \$ 80.987 \$ -.3750 --7780 -.5040 • • • \$ ₽ .... \$ ž ¢ ¢ ⊅ 6.892E+00 6.892E+00 6.892E+00 6.892E+00 \$ 6.892E+00 6.892E\*00 \$ \$ ¢ \$ .0711 1619. -.470İ ANG= V-DR 80-7 0 V-DR 40°-7 \$ \$ ¢ 20 \$ ¢ \$ • ċ ċ • \* • • ATT. CONT. STAR THCKR \$ \$ ₽ NO PRPGTE \$ н г × ≻ \$ 3.6942076E+04 P=02 17 8.021E+03 8.021E+03 8.021E+03 8.021E+03 8.021E+03 8.021E+03 \$ \$ ٥ ¢ ¢ 40-d d0-d d0-d •2923 •4168 -- 8607 \$ \$ PITCH ; ; \$ • ٥ 20.00000 DEG. ZO.OVONU PEG. °. 20.00000 DEG. SCHEDULE \$ \$ \$ 1.176E+04 1.176E+04 1.1765+04 \$ 1.176E+04 1.1765+04 1.1705+04 \$ \$ ------.7780 -.3750 (506°0 WOEHO \$ 0H41M58.365) .945 \$ \$ 10 A 10 A 10 A 10 A но-ч YAW 0-C3 80**-**0 \$ t; \$ \$ ¢ \$ 00 \$ ¢ \$ N 0 1.•0406485E+06 •8797 1074-0711 s. U. ≑ \$ \$ 1 • 964F + 04 1 • 964F + 04 1 • 966F + 04 1 • 966F + 04 IC 1.9645+04 1.904F\*04 RULL 1 CHANGED TO 2 CHANGED TO 3 CHANGED TO \$ ≎ \$ P=0a aC-d ç ç S \$ \$ P=0a ac-d \$ ຄ ¢ \$ \$ •---• X 17 • • \$ ి \$ PENALTY (MODE ⇒ • •0 \$ \$ \$ SEC. 2518.36 SEC. 310.000 SEC. 94 SEC. COMPUTER CORP. DEV. Deviations Erhors TURNED OFF TURNED OFF \$ DEVIATIONS  $^{\rm Z}_{\rm C}$ ¢ \$ S COMP. DEV. DEVIATIONS COMP. DEV. Z 20 Z NO COMP, DEV. DEVIATIONS ATTITUDE COM. RCVR. TURNED ON 617.700 SEC. ☆ \$ \$ \$ ¢ ≈ 3447 ALT= ¢ TURNED TURMED TUPNED 02Nun1 TURNED TURNED ERRORS ERRORS FRRORS 1400.00 \$ ULXV ZO SIXV NO 00 0H57M27.005 DFADBAND ON AXTS ≎ \$ ٩ ¢ \$ SUHSYSTEM SUN SEVSOR STAR TROKE ATT. CONT. ů ů AUD PURN ERRORS STATUS ADD PURN ERRORS t. S. U. COMPUTER DEADEAND **UNAHOARO** XHITER COMPUTER . 5. U. TARGET EGCAPE FSCAPE a C L C0431 2000 COAST FOR RURN FOR BURN FOR a F P ∧ 5 K ⊥ V

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FIGURE A-9a.

. MARS MISSION, ERROR ANALYSIS OUTPUT

ψ 7.854E-01 17E-02 .854E-01 ≎ .854E-0] \$ \$ \$ A-0P A-0P o ¢ ¢ ¢ 10 ⊅ ΰ \$ XMITTER XMITTER 7.854E-01 7.854E-01 \$ 1.717E-02 •854E-01 ¢ 1.717E-02 7.854E-01 .3976 -.0000 • 0000 1.0000 -.9151 .0664 \$ \$ PITCH P I TCH A-CR A-CR \$ \$ -153,1083 LONG= -152.4907 \$ \$ ٩ ٥ RCVR. APP. RADAR . RADAR \$ ۵ ₽ ۵ - 1400 - 9902 -.3/18 -.5205 -.7687 ۵ \$ \$ \$ 7.854E=01 7.854E=01 1.717E-02 7.854E-01 7.854E-01 YAW YAW 1.717E-02 \$ \$ \$ ż Ŕ ¢ \$ ż A-0R 0 A-DR 0 APP. LONG= \$ ¢ ¢ -1560 -8513 ..1400 --5010 -9402 .0000 å ¢ ¢ ¢ ¢ RCVR. ROLL 60 SEC ROLL ۵ \$ ۵ -8092 V-02 1.318E+01 1.318E+01 1.318E+01 1.318E+01 ≑ \$ ψ 1.407] 90.09 \$ \$ ¢ ٥ COM. 800 40-V 8 8 8 8 6 0 8 . MOC \$ ¢ ٥ ¢ ۵ ¢ \$ ۰ • • .2923 .4168 .8607 -4168 -8607 SUN SENSUR ISU/C.P.S. SUN SENSUR ISU/C.P.S. \$ • 2923 \$ ¢ ₽ 78. 74. BETWEEN UPTICAL AIDS= LAT= \$ 21 \$ ¢ \$ LATE 4•084E+01 4•084E+01 4.084E+01 4.084E+01 60 90 ¢ \$ \$ ¢ -82.733DEG. V-CR V-CR \$ \$ \$ ٢ 3.769 3,323 \$ \$ \$ -.2482 -.8361 -.489) \$ --2482 --4891 -.8361 •0 • \$ \$ \$ \$ g g \$ ¢ ₽ ≄ 1.509E+01 \$ 1.5095+01 \$ ¢ ≉ ≎ ٩ \$ V-DR ROLL V=0R \$ \$ \$ ¢ ≂9NA HONA g č ٥ \$ \$ \$ ċ • THCKR TKCKG \$ \$ \$ ¢ PRPGTE -58.349046. \$ \$ \$ ≎ 3.2113354E+04 3.2263012E+04 SUN-CANOPUS ин ХХ 17 X 17 2.582E+04 2.582E+04 2.5825+04 2+5825+04 \$ \$ ≎ \$ ŧ ATT. CONT. STAR d0-d ATT. CONT. STAR \$ d0-d ¢ • 6510 • 2923 •4168 .6138 --8607 SUN-CANOPUS .4465 0z \$ \$ ≎ ¢ PITCH PITCH • ≄ \$ • BETWEEN CELESTIAL BUDIES= \$ \$ \$ \$ \$ \$ YAW 1.2092+05 1.2095+05 1.209E+05 1.209E+05 MANEUVERING, BEGIN SEARCH FUR \$ \$ ⊅ \$ -.7750 .3680 4153 -.5040 -- 1698 (\$00°0 ٥ \$ \$ \$ 20 10 10 10 ۸۸W vet= 00 10 Y A W 8 • 00 24 • 74 vel= MAX. 64+00 ¥ 4 \$ ¢ TO LOCATE PITCH-170.754DEG. \$ \$ ¢ • • Σ ţ ¢ \$ °6639 2.01216646+07 P-DR -- 1752 -. 7270 1.8920650E+07 1978. -04701 • 0711 Se U. ¢ \$ \$ 14.5 2.41 с, 2.441 S• U. 7-8455+04 7-8455+04 7.845E+04 BULL SET 1708 1.8455+04 \$ ٥ ¥ p-100 00 ≎ ۵ ¢ ÷ BEGIN MANEUVERING • ہ ۱-1 Ŷ \* ₹: \$ ин Х Х WAY 111 7 7 7 BANDS ROLL PITCH 12 ů •0 -٢ ¢ \$ \$ ¢ ¢ COMPUTER 60.00. SEC. COMPUTER CCMP. DEV. DEVIATIONS EPRORS ANGLE COMP. DEV. DEVIATIONS FRRORS \$ \$ \$ ψ ATTITUDE ATTITUDE DEAD MANEUVERS 5247 ALTE 5307 ALT= ٥ ٥ \$ ÷ t: ÷ ¢ ¢ 2H27N27.005 1H28N27.005 ţ ¢ 4 t E N D \$ ¢ ٩ \$ SUASYSTEM STATUS SUBSYSTEM STATUS FICAPE ESCAPE FSCAPE 334253 С С 6 a C L 11 |---С |---COAST -1 44

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MARS MISSION, ERROR ANALYSIS OUTPUT FIGURE A-9b.

1.7175-02 1.7185-02 7.1245-04 ₽ \$  $\mathbf{o}$ A-OP ċ ≑ ¢ \$ XMITTER --02593 0.0000.0 1.7175-02 1.7175-02 2.3765-04 ó ٥ .3976 -.9151 .0664 \$ \$ PITCH ۰ \$ -151,9224 ٥ \$ COM. RCVR. APP. HADAR \$ \$ .PITCH •PITCH -.3717 -.7687 \$ \$ 1.717E-02 1.717E-02 2.422E-04 ΥAW \$ \$ ¢ A-DR \$ ¢ ⊅ LONG= ψ \$ \$ -.5010 -1560 .8513 \$ \$ ¢ 60SEC. 60SEC. 0.0000.0 • 02593 ROLL \$ ۵ \$ 1.318E+01 1.318E+01 \$ \$ \$ 1.9706 6.A31E-03 90.00 00.06 \$ \$ MARS MISSION, ERROR ANALYSIS OUTPUT ¢ V=0P a a a a ≎ \$ \$ \$ \$ -.4168 -.8607 SUN SENSOR ISU/C.P.S. \$ «ΥΛΨ ٥ «ΥΔΨ \$ £592. AIDSH 78.74. BETWEEN UPTICAL AIDS= \$ \$ LAT= Z Z 1.9885-02 4.0845+01 4.0845+01 6 \$ ¢ --001DEG--.001DEG. ¥ 1.00000 PASSES 1.00000 PASSES ۵ AD-V 78.74. BETWEEN UPTICAL 3.154 ⊅ \$ --8361 --4891 \$ -.2482 0 • 0 0 0 0 0 .00000 ٥ \$ \$ g ¢ \$ ٥ ¢ -509E+01 \$ 7.9458-03 \$ ·1407 \$ \$ •9236 ••3566 \$ ROLL ROLL V=0R \$ ٢ **ANG** ¢ DЯ \$ \$ ¢ RAIES (RAD/SEC)=ROLL 29.7 SECONDS WITH RATES (RAU/SEC)=ROLL \$ THCKR ٦ \$ 30+3 SECONDS WITH NO PRPGTE •000DEG• PRPGTE •000DEG. \$ \$ ¢ \$ 3.19638265+04 1.281E+01 2.582E+04 2.582E+04 2.582E+04 ×≻ N \$ ٥ ₽ d0-d ¢ \$ ATT. CONT. STAR 1 1 • 4465 •6138 1159. g \$ ٥ ¢ PITCH ¢ ANGLE BETWEEN CELESTIAL BUDIES= ¢ .10000 DFG. ANGLE BETWEEN CELESTIAL BUDIES= \$ DEG. ,10000 DEG. FIGURE A-9c. \$ ¢ ⊅ YAW YAW 5.9405.01 1.2095.05 1.2095.05 \$ ٥ \$ ċ15ċ. .10000 .3680 ••7698 0.005) 00 0H 1M 0.00S) ≎ \$ ⇔ ≎ V51.a 5-CR 54.00 8.00 24.74 8.00 26.74 ¥Aw 64.00 MAX. MAX. ₽ ¢ ANGULAR ANGULAR .000DEG. •000DFG• \$ ₽ ¢ SUN-CANUPIJS 0H IM ≎ Z ۵ ¢ Z .7270 2.1335757E+07 --1752 ₽ •∩ •5 ≎ • 10 010 •10 2.41 ۵ 2.41 A.0455401 7.8455404 7.8455404 2.5 Sn ⊤ SET BULL ⇒ ¢ 101 \$ с Н 50 P=0R ۵ CHANGED 1 CHANGED 1 \$ CANOPUS \$ CHANGED ¢ \$ \$ • --RULL YAW HOLI PITCH ΥAW ч н × ≻ DEAU BANDS ROLL PI1CH PITCH 12 DEAU HANUS FOR ٥ ~ \$ -NUS \$ ٥ \$ \$ a. 60.00 SEC. 60.00 SEC. COMPUTER FOR а 0 1. DEVIATIONS ERRORS SEARCH \$ 4 \$ COMP. DEV. ATTTUDE **~**t Nm ANEUVERS MANEUVERS 00 1H29M27.005 = 5367 ALT= ۰ 3 \$ ~ ON AXIS ON AXIS SEARCH SEARCH \$ ON AXIS \$ \$ \$ \$ \$ ACOUIRED GNΒ ¢ ¢ \$ SUHSYSTEM STATUS DF A DHAND DEAUNAND DFADSAND FCCAPE ESCAPE ESCAPE COAST FOR COAST FOR ц Ц SUGONCO +- v

ή •00af-03 .179E-04 .1285-04 .00AE-03 .1255-04 .00RE-03 .128E-04 1355-04 0 40-V 0 ۵ dund 40-V c ψ ¢ ΰ ¢ XMITTER XMITTER 7.515E-04 2.390E-04 -04 7.1255-04 .511E-04 \$ \$ 7.125E-04 2.3855-04 2.3768-04 7.514E-04 .3976 .3976 -.9151 .0664 -.9151 .0664 \$ PITCH £ PITCH A-CR A-CR ψ \$ -151.3994 -150.9185 ٥ ¢ 7 SUN SENSOR ISU/C.P.S. COM. RCVR. APP. RADAR ¢ RADAR ۵ 7.1255-04 7.5305-04 7.5305-04 7.2.4365-04 2 -.3717 -.3717 -.7687 -.7687 ¢ ¢ 7.536E-04 2.455E-04 7.125E-04 7.530E-04 YAW ΥAW 7.125E-04 2.4348-04 ٥ APP. A-DR A-DR ⊅ A-UR C ¢ o =9N01 -SNO-¢ ₫ .8513 .8513 -.5010 -.1560 --5010 -.1560 \$ HANGE DOT 9.778823E+00 4.100000E-02 \$ ROLL RCVR. 60 SEC. ROLL \$ ò 6.8316-03 1.318E+01 1.318E+01 1.766E+01 ¢ \$ 3.447E+00 1.3185+01 3.447E+00 3.0039 .3186+01 2.5021 1.26655401 00.06 \$ \$ V-05 · MOC 40-A a a a a н с С С V-0P 80 \$ \$ SUN ANG. 172.462 \$ ż .2923 .4169 .8607 SUN SENSUR ISU/C.P.S. • 4168 • 8607 ¢ \$ • 2923 AIDS# \$ \$ LAT= Z LAT= 3•9958+01 4•0845+01 8•4438+00 1.9886-02 4.0846+01 3.995E+01 4.084E+01 90 a. 0 7.2496085-04 00+3E+00 4.084E+01 ¢ ¢ ٥ \$ 80-7 V-CR 20-2 ALIMUTH UPTICAL 3.253 3.559 16.689 \$ ŵ -.4361 -.4891 -2485 --2482 -836I -+4891 . Γ. ٥ ٥ a S g \$ \$ •0-10+3652+1 10+3652+1 7+9452-03 1+5092+01 1+5092+01 1.505E\*01 509E\*01 \$ ¢ 4.6155+00 4.615E\*U0 78.74. BETWEEN •1405 • 3366 -1403 \$ -.3566 • 5230 ٦ • \$236 ROLL V-DR X-0K <"---> ELEVATION 4.4229326-03 ÷ \$ -127-688 **FONA** ANGa ő M7V ã ¢ \* TROKE TRCKP ≎ ≑ PRPGTE. .000DEG. ¥ \$ ы. х. ж 17 3.18144205+04 3.1665163E+04 2.5825+04 2.0115+04 1.281E+01 2.5822+00 2.5825+00 ч н Х Ж 5 1.6195+04 •552E+04 2.0115+04 1.6195+0\* \$ ٥ SIAR ATT. CONT. SIAR 2.2080596E+**0**4 ₽ \$ d0-d d0-d ••• --d0-d • 6130 • 65130 • 46511 •6138 •6511 9944. 0 Z \$ ۵ PITCH PITCH RANGE UOT \$ \$ CELESTIAL BUDIES= ATT. CONT. 1.1110275004 5.000006+01 \$ \$ YAW 5.9408.01 1.2098.05 1.2098.05 1.180E+05 1.209E+05 -1805+05 . 20% + 05 \$ 2.6432+04 \$ 2.643E+04 -- 7699 -.7698 0665. 0665. 0.005) \$ \$ **BANGF** YAW NVX VELa 13V A.Ón P-02 20-0 MAX. 64.00 24.74 20-d \$ ₿ •000DEG• \$ ¢ 3.2575255E\*07 Σ \$ \$ 2+25619355+07 -01752 ... 1270 -.1752 •6638 \*\*7270 • ୧୦୬୫ 2.3792909E+07 •10 ¥ \$ •10 .10 s. U. 5. U. 4.0457\*01 7.0457\*04 7。6355\*04 7.8455\*04 1.8015\*04 л С SET 7.8455\*04 20LL 7.6555 \*04 7.5455+04 1.501F\*04 RULL MEASUREMENT ERRORS INERTIAL SYSTEM ٥ \$ RANGE من<u>،</u> م BETWEEN 20-d ĉ P-DA ..... \$ \$ • 0 5--4 \$ \$ YAW н н Х Унн Х≻ ROLL 17 BANDS PITCH PITCH N \$ ~ \$ 0515 ⊅ ٥ COMPUTER sec. COMPUTER DEVIATIONS Errors COMP, DEV. DEVIATIONS ۵ \$ ANGLE ATTIUUE ATTITUDE CONP. DEV. DEVIATIONS COMP. DEV. DEAU GOLDSTONE MANEUVERS KALMAN UPDATE 5427 ALT= 5487 ALT= ≎ \$ STATTON ERRONS 520423 \*\*\* 60.00 ≎ r, 17305487°005 OD IH3IM27.005 ¢ \$ ACOUTRED ACOUIRED : \$ SUNSYSTEM SURSYSTEM STATUS 5111105 ESCAPE RSCAPE AFTER C C α C L 11 1-3 11 1-SUGONAD CANOPUS 12000 + < 4

FIGURE A-9d. MARS MISSION, ERROR ANALYSIS OUTPUT

AFTER MIDCOURSE CORRECTION (IMC= n) + RNY= 1.3617E+U2 (FI/SEC) + DOF=1.002

TO ZERO COMPUTED DEVIATIONS AT 16934400 SECUNDS

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A**-**34

FIGURE A-9e. MARS MISSION, ERROR ANALYSIS OUTPUT

10-.854F-0] .854E-0] .854F-01 .854E-01 54E-01 5 . 854E-01 ¢ ċ ≎ 8565 \$ ≎ 154E d C L V ¢ \$ ٥ ż ŵ ¢ ۵ \$ ¢ XMITTER XMITTER \$ ¢ ≎ -0972 5 .8545-01 101 .854E-01 .854E-01 6 : 7.8546-01 • 9887 •854E-0 .2412 -.1145 .9637 -.1146 \$ ¢ ψ PITCH PITCH A-CR C .8545 0 •854E ¢ ¢ . 856 ÷ .854 -101.9129 \$ \$ ⊅ RAUAR \$ ¢ RADAR \$ •0230 •1913 • 9230 • 3338 -.9411 -.3338 \$ \$ ۵ 7.854E=01 7.854E=01 7.854E=01 7.854E=01 YAW ۸AY 7.854E-01 .854E-01 •854E=01 ..854E-01 7.8545-01 ÷ ⊅ \$ APP. RCVR. APP. A-DR ٥ A-UR \$ 0 ≎ A-DR 7.854E L0NG= \$ \$ \$ 1863 •1402 -3239 ,9356 2998 \$ 5.023960E+02 ⇒ \$ 4.100000E-02 1.71579AE+01 4.10000E-02 RCVR. POLL ROLL ¢ ۵ \$ RANGE DOT 2.330E+00 2.849E+00 2.044500 2.5555400 1.5325400 RANGE DOT ¢ \$ 1.640E+00 ٥ 23.4478 2.585E+00 2.585E+00 7.0405-03 ¢ ۵ ŵ COM. 40**-**1 SUN SENSOR ISU/C.P.S. COM. a a a 2200 40-N 40-V ٥ ٥ \$ SUN ANG. SUN ANG. 124.903 \$ ٥ \$ 136.616 •4238 •9055 SUN SENSOR ISU/C.P.S. •0227 •4236 •9056 • 0228 \$ \$ \$ \$ ¢ \$ LAT= 9.352E+00 9.451E+00 1.354E+00 90 AZIMUTH 7-5529185-04 8.9035+00 99 9.451E+00 -03 9.012E+00 1-033789E-04 1+392E+00 9.451E+00 \$ \$ \$ \$ C \$ α0<del>-</del>Λ c 20-2 ≎ V-CR 6+622F AZIMUTH \* -.3816 -.2119 72.038 ¢ 89.492 72.368 ٥ •8409 096\*\*-.9111 ELE. Е. Г. ≎ \$ \$ g g \$ \$ \$ ••• 2.350E+01 2.350E+01 1.603E+01 \$ \$ ⊅ 3.923E-02 2.3505401 2.3505+01 8.6295-04 2.170E+01 .1705+01 .1566 ≎ \$ •4116 e123. ÷ ·5407 •394b 0: **°** ELEVATION 3.2154105-04 -0. V-0R V-0R V-UR 119-455 \$ ٥ ANGU \$ 1.2303235-04 -0. 128.614 ŝ MZA ano AZM ≉ ¢ \$ ELEVATION TROKB ٦ ₽ TRCKR \$ H H H K K K K \* ò 1.5547183E+06 нн х ≻ 17 \$ 3.8225+06 4.7715+06 2.8565+06 .411E+05 4.771E+06 .916E+06 4.771E+00 1.4025+00 1.249E+04 \$ \$ \$ ATT. CONT. STAR ATT. CONT. SIAR ŝ 1,3227597E+04 ≎ \$ î ¢ 00-d д**0-**д 1-3492698E+04 d0-0 •6138 •6511 •6138 .4465 •6511 • 4465 \$ ¢ ₽ PITCH PITCH RANGE UDT RANGE UOT \$ ¢ ٥ RANGE 1.744744E+07 1.824746E+04 5.000000E+01 5.0000006+01 ¢ \$ \$ 1.5665+07 1.5825+07 2.4055+06 1.2145\*05 ¢ ¢ 1.1725+04 .8595.06 \$ 764€ + 06 1.5326+07 582E+07 •3690 •52153 •3490 ¢125. --7698 --7698 (500°0 0 • 0 0 5 ) \$ \$ RANGE \$ VEL= YAW 5-CS ×۵۲ C I 0 1 80-4 D=CH ≎ ¢ \$ **†** \$ ¢ 1.1555416E\*10 2.3269202E+10 N S ¢ No ¢ 4 . 6638 . --1752 -- 7270 2.3268221E+10 --1752 . 66.28 -- 7270 •∩• 3• ∩• \$ ۵ \$ S• U. 3 • 7545 • 07 3 • 7545 • 07 1 • 9485 • 05 Ц С 5 L 3705 3.0345+04 PULL 3.7545+07 3.7545+07 • 034540 1.3085+07 MEASUREMENT ERRORS INERTIAL SYSTEM 1°808E\*01 \$ ¥ RANGE MEASUREMENT ERRORS TNERTIAL SYSTEM \$ RANGE c I д0**-**9 \$ ( 100 ≎ 90-9 с 1 (105) 20-02 ٥ г• • •--4 \$ ⋨ ミュモンメン X \$ OSIF ≎ > SISO ≎ ¢ ≉ \$ COMPUTER ບ ພິຍ COMPUTER COMP. DEV. PEVIATIONS ERPORS 0 9 2 0 3 0 0 3 0 COMP. DEV. DEVIATIONS FRRORS DEVIATIONS EPROPS \$ \$ ≎ ATTTUDE COMP. DEV. ATTIUDE GOLDSTONE KALMAN UPDATE GOLDSTONE AFTER KALMAN UPDATE \$ A1, T= ¢ \$ °. STATION STATION 01 \$ 00°000490 \$ 864000.00 CH334 0.005 **冷・** ポ \$ \$ 1729980 \$ \$ ¢ SUNSYSTEM SURSYSTEM IOCENT. HFL TOCENT . HEL TOCENT. STATUS STATUS AFTER 000 0.0 0.0 FOR オウドウ COAST Δ7

FIGURE A-9f. MARS MISSION, ERROR ANALYSIS OUTPUT

A-0P 7.854E-01 7.854E-01		8		A-07 1 7.8546-01 1 7.8546-01 1 7.8546-01 1 7.8546-01		A+0P 7.854E+01 7.854E+01 7.854E+01		я В С		A=0P
8238 A-CR 7.854E-01 7.854E-01 7.854E-01	PITCH 9195 • .3760	AR XMITTE 0		A-CR 7.854E-01 7.854E-01 7.854E-01	* * * * * * * *	•5065 A=CR 7.854E≈0] 7.854E≈0] 7.854E≈0]	PITCH • 8589 • 1147	AR XMITTE 0	•	A-CR
LONG= -103. A-DR 7.854E-01 7.854E-01 7.854E-01 7.854E-01	НЦ YAW 2272 • 3208 2702 • 8866 9356 • 3339	R. APP. RAD	- 01 - 01	A-UR 7.8546-01 7.8546-01 7.8546-01	\$     \$       \$     \$       \$     \$       \$     \$       \$     \$       \$     \$       \$     \$       \$     \$       \$     \$       \$     \$       \$     \$       \$     \$	LONG= -105. A=DR 7.854E-01 7.854E-01 7.854E-01	LL YAW 2622 • 4399 2356 • 3337 9356 • 3337		-01 -02	A-DR
21.5480 2.5480 2.28255400 2.28255400 2.28255400 2.4245100	6 0 0 0 0 0 0 0 0 0 0 0 0 0 0 0 0 0 0 0	. COM. RCV	- ANG. 794 2.1047686 4.100006	V=0P 2.2822+00 2.2825+00 1.5205=03	<ul> <li>↓</li> /ul>	19.3976 V-0P 2.086E400 2.086E400 1.394E-03	66-4 80 90 90 90 90 90 90 90 90 90 90 90 90 90	. COM. RCV	1 ANG. 219 Range Dot 1.0567496 4.1000006	40 <b>-</b> 1
9.648 LAT# V-CR 9.691€+00 9.691€+00 9.691€+00	CR 0P •9604 •022 •2608 •423	UR ISU/C.P.S	ELE. SUN 2.067 147. ZIMUTH 557440E-07	V=CR 9.691E+00 9.691E+00 1.587E=03	* * * * * * * * * * * *	9.715 LATE V-CR 9.2006+00 9.2006+00 1.4576-03	CR 0P •9895 •022 •1387 •423 •0401 •905	0R ISU/C.P.S	ELE. SUN 1.131 158, ZIMUTH 418850E-08	CR
ANG= V+DR > 641E+01 7 661E+01 7 2552E+03	08 • 2776 • 8675 • 1514 •	KR SUN SENS	AZM 37.060 10N 10N 38E107 38E107	V+DR 2.6412+01 2.641E+01 3.511E+01	* * * * * * * *	ANGE 8 V=08 3.0295+01 3.0295+01 4.0702=04	0 8 8 9 8 9 8 9 8 9 8 9 8 9 8 9 8 9 8 9	KR SUN SENS	ALM 44-978 10N 33E-07 5-	V-08
<pre>&gt;06462888+06 P=0P 6*8638405 6*8638405 6*8638405 1*84084 1*84084</pre>	۲۲CH ۱۰6138 XI ۱۰6521 ۲۱ ۱۰۴465 XI	- STAR THC	UOT PERE+04 1: ELEVAT 32 6.1787	6•8638+06 6•8638+06 6•8638+00 4•071€+03	* * * * * *	3687410E+06 P-0P 8.729E+06 8.729E+06 5.328E+06	•ITCH •6138 XI •6511 YI •4465 ZI	I. STAR THC	UOT +24E+04 1' ELEVAT 01 1-5048	d0 <b>=</b> d
VEL= 2.4 p.cR 2.4138.07 2.4138.07 1.7318.09 1.7318.09	- 4 AW - 7598 - 3550 - 5519 - 5509 - 5509 - 5519 - 5509 - 5500 -	ATT. CONT	RANGE 10 1.44037 RANGE 2.92088244 5.00000444	2.4135407 2.4135407 2.4135407 4.3805403	* * * * * * * * * * * * * * * * * * *	VEL= 3. P=CR 3.2355+07 3.2355+07 5.5955+07 5.5955+03	- ≺ A - √ A -	ATT. CONT -0	RANGE 10 1.65114 RANGE R.0598595+0 5.00000240	۲. د د
5538209E+10 P-nR 5.902E+07 5.902E+07 5.952E+07 1.953E+07	RULL XI ••1752 YI •6638 ZI •7270	I. 5. U. -0	RANGE 3.55592224E* IT ERRORS IL SYSTEM IL	8-9025-07 5-9025-07 5-9025-07 5-4225-02	* * * * * * * * * * * * * * * * * * *	9069258E+10 P-02 8.3465+07 8.3465+07 8.5465+07 8.5065+02	R0LL XI1752 YI6638 ZI7270	• 0 • 0 •	RANGE 4-90703918* 17 ERRONS 14 SYSTEM 516	50 <b>-</b> 4
0.005 10 ALT= 3. OMP. DEV. FUIATIONS ERRORS	AITITUDE	COMPUTER	STATION SCOUSSTONE MEASUREMEN INERTIA DS N HDDATE	COMP, DEV. Eviations Errors	$\begin{array}{c} \bullet \\ \bullet $	0.005 10.01.T= 4. 10MP, DEV. ERFORS	ATTITUDE	A COMPUTER	STATION SOLOSTONE MEASURGMEN TNERTIA US	ATAGAN NO
300 04334 259334 259339 200		SUBSYSTEM STATUS	G G KALMA MA		HELIUCENT. FOR 8540 HELIOCENT.	400 04334 1≞ 345798 0		SURSYSTEM STATUS	G	ASTEN XALMA

FIGURE A-9g. MARS MISSION, ERROR ANALYSIS OUTPUT

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-1-26

\$ ¢ ¢ ψ 7.854E-01 .854F-01 7.854E-01 \$ ¢ .85AE+0] ទ .854E-0] 854E-01 .854E-0 8545-0 .854E-0 ⊅ ٥ 9 .854E-0 ψ ⊅ ⊅ 40-V . A54E dü=v 40-V .854F ⊅ c ψ ⊅ ٥ ٥ ÷ ΰ \$ \$ 4 \$ \$ \$ XNITTER XMITTEÀ \$ \$ ≎ A-CR 7.854E-01 ·854E-01 •854E-01 \$ .35¢E-01 .854E-01 7.854E-01 7.8545-01 7.854E-01 .854E-01 . 854E-01 •6089 E\$01. 7.3546-0 .7006 0 V810 -.1147 -.1147 ¢ v PITCH ٥ \$ PITCH A-CR 0 A-CR \$ ۵ \$ ۵ -106.7230 -98,9646 \$ \$ \$ \$ ۵ \$ RCVR. APP. RADAR \$ \$ RCVR. APP. RADAR - 7678 - 3339 •6405 -.6916 .3333 \$ ۵ ¢ ¢ YAW YAW 7.854E-01 7.8542-01 .8545-01 7.854E-01 7.854E-01 7.854E-01 7.854E-01 7.8546-01 7.854E-01 7.854E-01 .854E-01 7.854E-01 ٥ \$ \$ ۵ \$ \$ \$ A-0R 0 ¢ 0 A-0R A-DR LONG= LONGE \$ \$ \$ ≎ +195. ...1994 .9356 .3145 -1605 .9356 ¢ ¢ \$ 1.002674E-01 4.100000E-02 ROLL ROL \$ \$ ¢ ż 001 2.1175.00 2.1175.00 5.8305.04 2.4235400 2.4235400 1.887540 ÷ ¢ 2. JR6E+00 2.7865.00 17.1660 2.117E+00 2.117E+00 ٥ \$ 15.1528 2-1265-04 6.502E-04 \$ \$ \$ ψ 40-V COM. 40-A HANGE 2 2 0 0 0 0 200 • MOU -40-N ¢ ¢ \$ ۵ ANG. \$ \$ ŧ 165.823 \$ • 0227 • 4236 • 9056 SUN SENSOR ISU/C.P.S. \$ • 4236 SUN SENSOR ISU/C.P.S. \$ \$ \$ SUN .0227 -.9056 \$ \$ \$ LAT= n \$ 7.670E+00 7.670E+00 7.359E-04 5.005E+00 5.005E+00 3.294E+04 7.670E+00 7.670E+00 9.200E+00 9.200E+00 8.229F-04 LAT 40 aO AZIMUTH 1.396049E-08 3.698E-04 \$ \$ ≑ \$ V-CR V-CR V-CR ¢ ¢ \$ \$ С С \$150\* ≉ \$ .0164 69.669 \$ 89.768 ¢ 84.748 -.0185 +0705 1566. 1266. ь. Е С Е • \$ \$ \$ \$ č ž ź \$ \$ \$ 3•4735+01 3•4735+01 2•2046+04 3•9285+01 3•9285+01 7•0835=05 3.029E+U1 3,029£+01 ٥ \$ 3.473E+01 9.5282-05 ≎ \$ 7.284E=04 3.4735+01 . 9005 . 4183 . \$255 \$259 \$ \$ ٥ \$ 8600° --1181 V=0R ° V-DR 0 V-DR ANG \$ \$ ٥ ANG 151.075 \$ ELEVATION 7.5114795-09 MZM ă Ξ \$ ¢ Ķ ⇒ THCKR ¢ ≄ THCKR \$ ۵ ۰ ۵ 4.5035107E+06 ٥ \$ 5.861A356E+06 77 1.052E+07 1.052E+07 エエエメ 1.245E+01 1.745E+07 ын Х Х 1.3055+03 8.729E+05 2.4745+03 3.0045+03 1.1405+03 8.729E+06 1.0522+07 1.052E+0/ \$ şt. \$ ₽ STAR sive \$ 1.95872005+04 ¢ \$ d0-d •••• ≉ d0-d C I 0 1 d0-d .6138 .6511 .6138 .6511 .4405 • 440S ۵ \$ ≎ \$ PITCH PITCH 100 39NAA \$ \$ ¢. \$ ATT. CONT. CON1. 1.1451415+02 5.0000006+01 ¢ ¢ \$ ۵ 3.2355×07 3.2355×07 3.9728.07 4.526E+07 4.526F+07 3.9725+03 3.8735+03 ¢ \$ 3.972E+07 \$ 3.2015-03 ୍ ଅ ଅ ଅ ଅ ଅ ଅ ଅ .972E.07 \$ 2.18/E.U3 . 8885+03 .3699 ໍ່ປະເອີ -.7698 -.7695 0+005) 0.005) \$ \$ ¢ \$ RANGE ATTo н<mark>0-</mark>а ΥAw n\_1 5102 YAW 40-d VEL= с і 0 ÷ \$ ¢ \$ 5 ٢ \$ \$7 ⊅ 6.4778154E+10 ¢ ¢ ¢ \$ S 9023H27M 8.3486142E+10 .6638 ••7270 6.4777452E+10 --1752 •6639 -.7270 1.1752 \$ °• 0.• ≎ ٩ ٠ ∵ ° 1.1155\*08 1.1155\*08 6.9&77\*02 1.4345408 1.4345408 3.7965402 Н ROLL 1.1155\*08 RULL 8.3465+07 50-2810-4 1.1155\*08 0.3465+07 3+083F+02 MEASUREMENT ERROWS INERTIAL SYSTEM \$ ٥ RANGE φ ¢ P-Da aΩ⊸q C 0 1 an-q \$ \$ ŝ 4 ≉ с 1 \* •--4 • ¢ ≎ ٢ \$ и х ж х ж エエエズ -≎ ¢ -\$ 4 051F ¢ \* \* ۵ COVPUTER COMPUTER COMP, DEV. UEVIATIONS ERPORS CONP. DEV. DEVIATIONS ERRORS 0 5 5 5 5 SEC. \$ \$ 4 COMP. DEV. DEVINTIONS ٥ COMP, DEV. OFVIATIONS OFVIATIONS ATTITUDE AITITUDE GOLDSTONE A1. T= KALMAN UPDATE \$ ¢ STATION ¢ ≄ 5384000 ALT= c) I ERRURS 01 ERPORS ΰ ¢ \* **1**: 864000.00 0.005 862020.00 S00\*0 W0 H0 ≎ ¢ ¢ \$ 4321980 \$ ٥ **\$**7 ¢ SUASYSTEM MELSYSICS HFLIOCENT. Croox HFLIOCENT. JELIOCENT. HFLIOCENT. STATUS SUTATUS AFTER 004 500 COAST FOR 502 ۱۱ سر 12 1---00457 2 11

FIGURE A-9h. MARS MISSION, ERROR ANALYSIS OUTPUT

\$ ٥ .717E-02 7.854E-01 7.854E-01 854E-01 .717E-02 .854F=01 .425E-0] \$ \$ Ċ d∩⊷A ¢ ≎ ¢ du-V 40-V ŵ ۵ ⇒ Ŷ # Ó ŵ ¢ ₽ ٩ XMITTER 1.7175-02 7.854E-01 7.8545-01 201 \$ ⇒ 7.8545-01 -854E-01 .7006 •4255=01 .8545-01 --7043 --1147 \$ PI1CH ٥ Φ A-CR A-CR A-CR 1.71 /E ٥ \$ ≎ -106.7138 -106.4638 ٩ ۵ ۵ \$ \$ ¢ ¢ HADAR .6405 -.6916 -.3339 \$ \$ \$ \$ 1.717E-02 7.854E-01 7.854E-01 Y A W 1.7175-02 •854E-01 1.8542-01 E-01 7.854E-01 \$ \$ 9 ٥ ۵ A-0R RCVR. APP. A-0R \$ ≉ 0 \$ A-DR LONG **85%** LONG= .425 ۵ ¢ \$ ٥ .3145 -.1605 .9356 \$ φ \$ \$ 60SEC. ROLL ROLL \$ \$ ٥ \$ 2.4235400 2.4235400 1.8875400 2.6236+00 2.623E+00 \$ 15.1483 15.1484 V=0P ¢ ÷ o 4 1.987E-04 2+4235+00 1.887E-04 \*<235+0 00.06 \$ \$ \$ \$ COM. 40=/ 40-N ¢ \$ \$ \$ \$ ¢ ⊅ \$ .0227 .4236 .9056 \$ SUN SENSOR ISU/C.P.S. \* ٩ 4 AIDSa \$ Z ¢ LAT= ٩ LATa <u>a</u>0 5+0055+00 5+005E+00 3.294E-04 \$ ≉ ≉ ۵ -56.0370EG. ₽ ¢ \$ \$ 89.39, BETWEEN OPTICAL • 0705 • 0705 ۵ ≎ ٥ ¢ 89.768 897.98 .9927 ₽ \$ ٥ ¢ g \$ \$ \$ ۵ 3+9266+01 3+9285+01 7+0635+01 3.9285\*01 3.9285\*01 ŧ ⊅ 7-0836-05 ≉ \$ 7.0436-05 3-9205+01 3,9285401 • 9006 • 4183 ۵ \$ ⇒ ¢ --1181 ROLL 90-7 V-0R V-0R ----\$ ٥ \$ ¢ ANG ANG аd ¢ \$ ¢ ₽ ٥ \$ \$ **TKCKR** ¢ PRPG1E. PR96TE 11.4270EG. ¢ ¢ ٥ 5.8656004E+U6 \$ 5.8654789E+06 1.2455407 1.2455407 1.3055403 1.7455+07 1.3055+03 1.245E+01 1 1.245E+07 1.245E+07 1.305E+03 \$ ¢ ≎ \$ STAR \$ \$ 60-d \$ \$ d0-d P-07 •6138 .6511 .4465 SUJN-CANOPUS <u>0</u>2 \$ \$ \$ ∶ 0 Z \$ PITCH ⊅ \$ \$ ¢ =S∃ION8 ATT. CONT. ₽ \$ ¢ \$ YAW 4.520F007 2.19/203 ŵ \$ 4.5265+07 ٥ 4.5265+07 2.137E+03 ٥ 4.5265+07 .5266.07 4.5265.407 2.1875+03 • 3690 • 5215 (500-0 -- 7593 (500.0 \$ ¢ \$ \$ vrL≡ P=CR YAW MAX. 64.00 35+39 p=CR R.00 VEL= P-CR CELESTIAL \$ ≎ \$ \$ LOCATE -.771DEG. Ω, \$ \$ ¢ Σ Ļ 4 ٥ 0H30M ¢ ¢ 8.35363745+10 •6638 -7270 8.35380495+10 --1752 ۵ ġ 5 \$ 2.41 2.41 2.41 ٢ 1。4345+08 1.4345+08 3.7955+08 1.4345\*08 1.4345\*08 SF.T BULL r 3.7965+02 8042989. · 4345+08 3.7965 02 с Т \$ \$ ⊅ \$ ů, aù-d ¢ g P=0R \$ c BETWEEN \$ \$ 40+d MANEUVERING . . \$ ¢ \$ \$ RULL YAW и и и и и и и PITCH PIICH BANDS \$ \$ -¢ Φ -≎ \$ ¢ ¢ TURNED ON TURNED ON TURNED ON TURNED ON TURNED ON SEC. COMPUTER DEVIATIONS EPRORS CONP. DEV. DEVIATIONS EPAORS COMP. DEV. DEVIATIONS 60.00 SEC. ANGLE \$ ΰ ÷¢ COMP. DEV. ₽ Z ATTITUDE DEAD MANFUVERS ALTe ALT= ¢ ≎ Φ \* ERRORS TURNED ≎ \$ ≉ ⊅ CHEIM 0.005 1890.00 0°00S BEGIN ¥ \* \$ \* 5185360 ٦ \$ ≎ ≎ 5185800 **SUBSYSTEM** MOEHO STAP 19CKR ATT. CONT. HFL TOCENT. SUN SENSOR HFLIOCENT. STATUS HELIOCENT. HELIUCENT. XWITTER COMPUTER t. S. U. しつく 600 с 0 л COAST F05 11 |--ħ COAST ΥT 4 V

MARS MISSION, ERROR ANALYSIS OUTPUT

A-9i.

FIGURE

ź \$ 1.7175-02 1.7185-02 6.8235-04 \$ 0 A-OP \$ \$ \$ \$ \$ Φ ¢ \$ XMITTER XMITTER 00000-0 0.00000 1.717E-02 1.717E-02 \$ \$ 2.453E-04 .9522 •9522 •-2064 --2064 -.2252 -.2252 PITCH ٥ ¢ PITCH α \$ A-C -106,9637 \$ \$ RADAR +PITCH +PITCH SUN SENSUR ISU/C.P.S. COM. RCVR. APP. HADAR \$ ۵ -.9352 -.2597 --2597 \$ \$ 1.717E-02 1.717E-02 2.742E-04 YAW ΥAW ₽ \$ APP. A-UR 0 ÷ \$ 0 rong= \$ \$ .1609 --2877 6441 •1609 --2877 .9441 ٥ \$ 60SEC. RCVR. 0.00000 .02618 ROLL ROL \$ \$ 2.423E+00 2.423E+00 1.887E-04 15.1481 V-OP ⊅ \$ 00.06 ¢ \$ 8000 8000 å ч с С С · WOU ≎ ¢ ≎ ۰ .0227 •0227 •4236 •9056 SUN SENSOR ISU/C.P.S. \$ • YAW \$ WAY. 89.39. BETWEEN UPTICAL AIDS= ¢ \$ Z LAT= 5.0055+00 5.0055+00 3.2945-04 90 6 \$ ≎ -.001DEG. V-CR \$ 1.00000 PASSES 1.00000 PASSES \$ 89.768 •9927 .0705 •9927 • 07 05 • 07 05 • 0975 \$ \$ .02618 • 00000 \$ ≄ g g \$ \$ 3.9285401 3.9285401 7.0836-05 ¢ \$ 0000° 5318° 0000 • 4123 ٥ \$ -.1181 X-0X ROLL ----#UNG# ≉ \$ ao с С \* ¢ 30-0 SECONDS WITH RAIES (RAD/SEC)=ROLL 30.0 SECONDS WITH RATES (RAU/SEC)=ROLL TRCKR ¢ \$ TKCKR PRPGTE. •0000FG• \$ \$ 5.8657220E+06 SUN-CANOPUS X H N 1.245E+0/ 1.245E+0/ 1.305E+03 чь Х≻ 17 6 \$ \$ ATT. CONT. STAR \$ ATT. CONT. STAR 1 1 d0-d --3225 •7420 .5877 .5317 -.3225 g *τ*, \$ PI TCH PITCH ٥ \$ BETWEEN CELESTIAL RUDIES= 586. 086 020 \$ \$ YAW 4.5202+01 2.1875+03 MANEUVERING, SEGIN SEARCH FUR • \$ ۵ 4.5266+07 .10000 --9032 .10000 0424 ---4£90° 2602. --4270 ÷ 6 4 9 + (S00.0 \* ₽ v€**L**= л•00. 35•39 WVY 64.00 Y A W D-CP MAX. ¢ ₽ ANGULAR ANGULAR .000DEG. \$ ¢ FOR SUN-CANOPUS Σ ₽ Z \$ Z - H079 -•8079 8.35397255.10 --28332 .51.63 s• ∪. 4 10.5 \$ 1 % • 2.41 HO 1.434F+08 3.796E+02 ς. U. 20LL 57.T 1.4365403 2011 ¢ 0.2 \$ C F an-q ŝ CANOPUS ¢ ¢ CHANCED CHANGED CHANGED • •--1 e 1--; \$ ¢ HOLId エエエスメンズ RULL YAW PITCH цы Х У 14 DEAD BANDS \$ -SUN \$ ٩ \$ COMPUTER с. Сы С COMPUTER ANGLE 205 702 SEARCH ή \$ CO4P. DEV. ATTITUDE DEVIATIONS ERRORS ATTITUDE N m MANEUVERS ٩ \$ 5185920 ALT= -60.00 ŵ SEARCH SEARCH \$ 500""U WZEHO \$ ACQUIRED 0 Z Z U ÚN3 ¢ \$ suusystem s SUBSYSTEM STATUS STATUS HFLIOCENT. HEL LOCENT DTADRAND READRAND DIVENCIAND 605 a0u n F CANOPUS COAST 1-2-

FIGURE A-9j. MARS MISSION, ERROR ANALYSIS OUTPUT

**.869E-04** .869E-04 .124E-04 6.82AE-04 8285-04 00 •125F. XMITTER 7.540E-04 2.466E-04 40-■540E=04 2.466E=04 •125E-04 .2252 .9522 PI1CH -107.2137 • SUN SENSUR ISU/C.P.S. COM. RCVR. APP. RADAR -.2407 -.2597 \$ 7.125E-04 7.639E-04 2.754E-04 ¥Δ¥ 7.639E-04 40-2.7545-04 đ A-DR ż 0 LONG .125 \$ .1609 -- 2877 1446\* \$ ٥ \$ 8-622030E-02 4.100000E-02 60SEC. ROLL 60 SEC \$ \$ 2.423E+00 8.429E=05 2.423E+00 2.423E+00 HANGE DO \$ 2+4235+00 ¢ ¢ 15.1480 1.887E-04 00.06 90.06 \$ ¢ ٥ g аO 40-V ЧO d0=/ \$ \$ ٩ SUN ANG. ۵ \$ 165.904 ≑ \$ •4236 •9056 + 0227 ΰ \$ 89.39, BETWEEN UPTICAL AIDS= AIDS= Z ٥ \$ LAT= ψ Z 5.005E+00 5.005E+00 3.294E+04 60 6.423449E=09 5,005E+00 1.742E-04 5.005E+00 \$ ٩ -.001DEG. -.001DEG. \$ \$ \$ 20-N V-CR 89.39. BETWEEN UPTICAL ALIMUTH 1266. •0975 •0705 61.993 \$ \$ 89.768 \$ \$ Е. Е. \$ \$ \$ 8 0 \$ ₽ ٥ •••• 3.928E+01 7.083E-05 ٥ \$ \$ 2.930E-U5 \$ 3.9285+01 3.9206+01 3.9205+01 • 9000 • 4183 ٩ \$ ٥ ۵ 1911.-ROLL V-0R X-0R ROLL ≎ ÷ \$ 3.0114985-08-08-08 \$ =9NG 154.450 AZM č \$ \$ ₽ ¢ ELEVATION TRCKR ₽ \$ \$ ¢ NO PRPGTE PRPGTE. •0000t6. .0000EG. ۵ \$ \$ \$ 5.8658435E+06 1。245E+0/ 1。245E+0/ 1。305E+03 ž 7 •245E+07 5.104E+02 \$ .2456+0 ¢ ≎ ¢ ATT. CUNT. STAR \$ \$ \$ \$ 2,3570856E+04 40**-**d p-0p -7420 -5877 ••3225 ò ≎ \$ \$ \$ PITCH 001 \$ ANGLE BETWEEN CELESTIAL BUDIES= \$ \$ ¢ CELESTIAL BUDJES= 1.091234E+02 5.000000E+01 \$ \$ \$ \$ RANGE YAW YAW 4.5262+07 2.1875+03 ₽ 4.525E.07 1.0705+03 \$ \$ 4.5208+07 \$ 4.526E+07 -+9032 -.4270 •0434 (\$00+0 0.005) ≄ \$ ¢ ΰ RANGE バマム R•00 v⊑L≕ 60.44 ал•СО 00.80 00 P-CR a)-d MAX. 60.09 39.39 MAX. ≎ \$ \$ ¢ -000DEG-.000DFG. \$ \$ \$ \$ 8.3542923E+10 Σ ≎ \$ Σ \$ 8.3541401E+10 -.2833 .5168 -•RU79 s. U. 01. .10 •10 \$ .10 ٥ \$ •10 ٥ .10 1.434F\*08 3.796E\*02 r c H<sub>0</sub> SET 55.4 1.4345+08 BULL 80\*19E7° . 434E\*08 1-6145+02 \$ ÷ ≎ ERROHS TNERTIAL SYSTEM \$ RANGE BETWEEN 80**-**-d \$ S ¢ 00-d ¢ ç ₽ ⊅ -----¢ φ t PITCH YAW ROLL ヨマン Z 12 DEAU RANUS 2104 PI TCH DEAD RANUS P11CH ΥT PITCH DSIF \$ -¢ \$ Ċ MEASUREMENT \$ \$ \$ SEC. COMPUTER 60.00 SEC. ANGLE ⇒ ¢ DEVIATIONS ERRORS COMP. DEV. ψ COMP. DEV. ATTITUDE DEVIATIÓNS FERDES GOLDSTONE KALMAN UPDATE MANEUVERS MANEUVERS ≎ 5185980 ∧LT= \$ ≎ ≎ STATION 60.00 \$ ΰ \$ ٥ 0.005 \$ ٥ ACQUIRED \$ 4 \$ ۵ ٥ SUBSYSTEM Mesh0 STATUS HFLIUCENT . HELL TOCENT . HEL JUCENT . HELIOCENT. AFTER C C C V V яOл COAST FOR 11 F-CANOPUS COAST Δ-1 Δ

FIGURE A-9k. MARS MISSION, ERROR ANALYSIS OUTPUT

	•		-								<b>*</b> .	•	
А-ОР 7.1255104 9.8745104 6.83651104	•	0			A-OP 7.1255.04 9.8745.04 6.8365.04	* * * *		* * * * *	A-0P 7.1255-04 9.8575-04 4.8275-04		0		A-OP 7.8545-01 7.8546-01
-++637 A-CR 7.1255-04 7.5475-04 2.4885-04	PITCH * \$522 7 * 2065	AR XMITTER 1	•		A+CR 7.125E+04 7.547E+04 2.488E+04	* * * * *		* * * * *	7136 7.1256-04 7.556-04 7.558-04	PITCH PITCH 2 9522 2 1 9522 2 1 22552 7 1 22552	AR XMITTER 1		A-CR 7.854E-01 7.854E-01
LONG= -107 A-DR 7.1255-04 7.6465-04 2.7735-04	LL YAW 1609 = 259 877 = 935 9441 = 240	RAD. RAD.			A+DR 7.1255-04 7.6455-04 2.1735-04	* * * * *		4 4 4 4	LONG= -107 A-DR 7.1255-04 7.5555-04 2.8005-04	L YAW 1609 - 259 2877 - 935 3441 - 240	а. АРР. КАО, 0		A-DR 7.8546-01 7.8546-01
15.1478 15.1478 2.423540 2.4235400 2.4235400 8.5235400	27 DR 10	5. COM. RCVI			V-0P 7.661E+00 7.661E+00 9.275E-03	+ + + + + + + + +		* * *	15.1477 V-0P 7.461E+00 7.461E+00 9.275F-00	200 L01 L01 L01 L01 L01 L01 L01 L01 L01 L	5. COM. RCVI		V-OP 7.6612400 7.6612400
J.763 LAT≡ V-CR 5.0055+00 5.0055+00 1.7428-00	00 0927 0975 0975 0705	JR ISU/C.P.	JOF=1.026		<pre></pre>	* * * * *		* * * *	4.768 LAT= V-CR LAT= 6.88849 6.888490 6.331287-00 5.31287-00	00 0927 002 0975 422	JR ISU/C.P.S		<pre> &lt; - Cq</pre>
ANG= ANG= V-70= 3 9 2 2 2 2 2 2 2 2 2 2 2 2 2 2 2 2 2 2	1919 1918 1918 1918 1918	KR SUN SENS	(FI/SEC) . 1		V-DR 4.2265401 4.2265401 1.2375101	* * * * * *	e	* * * *	ANG= 4+22655401 4+22555401 4+22555401	1811. 181. 1811. 1	CR SUN SENS		V=DR 4-2265+01 4-2265+01
3659650E+06 P+0P 1+245E+07 1+245E+07 1+245E+07 5+104E+02	•ITCH •3226 XI •7420 YI •5877 ZI	. STAR THCK 1	1.3317E+01	SONDS	P-0P 1.245E+07 1.245E+07 5.104E+02	<b>☆</b> ☆ ☆ ☆ ☆	NO PRPGTE	\$ \$ \$ \$ \$	16608666406 P-0P 1.24556407 1.24556407 5.10456407	.11CH .3226 XI .7420 YI .5877 ZI	• SIAR THCK 6. 1 6.		р-Ор 1.2456+07 1.2456+07
<ul> <li>VELa 5.</li> <li>VELa 5.</li> <li>P+62</li> <li>A-5266+07</li> <li>A-5266+07</li> <li>A-5766+07</li> <li>A-93</li> </ul>	NAY NG020 NAY NG2700 NAY NAY NAY NAY NAY	ATT. CONT	= / WH + ( U	6934400 SEC	₽+C8 ♦,5269 ♦ 07 ♦,5269 ♦ 07 1.0768 ♦ 03	* * * * *	M 0+005)	₽ ₽ ₽ ₽ ₽ ₽ ₽ ₽ ₽ ₽ ₽ ₽ ₽ ₽ ₽ ₽ ₽ ₽ ₽		1 1 5 0 0 0 0 5 0 0 0 1 1	ATT. CONT 20.00000 DF 20.00000 DF 20.00000 DF		р-С <u>я</u> 6.5262+07 6.5205.07
35%3076F*10 アーリア 1.4%36F*08 1.4%36F*08 1.4%36F*08 1.4536F*08	RULL XI2832 XI28326 XI28079	- - -	TION (IMC=	TIONS AT 1	P-DR 1.4348408 1.4368408 1.6162408	* * * * *	( 00 0H 1	* * * *	35¢¢7526+10 35¢¢7526+10 1.¢3¢8*08 1.¢3¢8*08 1.¢3¢8*08	жоцс 25832 - 12 7108 - 12	I. S. U. ANGED IO ANGED IO ANGED IO		P+02 + \$350 - \$350 - \$350 + 03 - \$350 - \$03 - \$350 - \$03 - \$350 - \$03 - \$00 - \$03 - \$00 -
DHZ&M G.005 Sigguad Alt= 8. Comp. DEV. Deviations Errors	ATTITUDE Acquired	SYSTEM COMPUTER TATUS 1	MIDCOURSE CORREC	RO COMPUTED DEVIA	COMP. DEV. Deviations Errors	CENT: * * * * * *	60.00 SEC.	CENT。 > > > > + +	0H35M 0.005 5186100 ALTE 8. Comp. DEV. Ufviations Errors	A1111UUE	SYSTEM COMPUTER TATUS 1 AND ON AXIS 1 CH AND ON AXIS 2 CH AND ON AXIS 2 CH	TER TURNED OFF CONT. TURNED OFF TER TURNED OFF U. TURNED OFF TREKR TURNED OFF	ENSON TURNED OFF COMP. DEV. DEVIATIONS
АТ 500 Т=	CANOPUS	ร หกัง	AFTER	32 01		01"ILH	COAST FOR	HELTO	AT 40D T=		0 0 0 0 0 0 0 0 0 0 0 0 0 0 0 0 0 0 0		9 2 0 0

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FIGURE A-91. MARS MISSION, ERROR ANALYSIS OUTPUT

۵ Ф ٩ ٢ .854E-01 854E-01 .854E-01 8545-01 \$ ¢ 3546-0 \$ 854E-0 ٩ 2 01 8545-0 .854E-0 40-A 8548-¢ A-OP \$ \* .85AE A-O-V ψ \$ ¢ ÷ ⇒ \$ ψ 2 ¢ ¢ \$ ۵ XMITTER XMITTER \$ ¢ ≄ 7.854E-01 ۵ •9704 ••0870 .854E=01 .854E=01 0 7.854E-0] •9739 •0293 7.8546-0 .854E-0 7.854E-0 -,2252 •854E=0 - 2255 \$ \$ ٠ ٠ PI1CH PITCH A-CR A-CR c 3428. 7.854E A-CR ⇒ \$ ¢ \$ -105,6352 -106.8491 ٥ ¢ ¢ \$ RCVR. APP. RADAR \$ \$ ø \$ RCVR. APP. HADAR --1419 --9607 -.9102 -.0265 -.2407 -.2407 ۵ \$ \$ ¢ 7.854E-01 7.854E-01 7.854E-01 7.854E-01 XAW X A W 7.854E-01 7.854E-01 7.854E-01 .8545-01 .8545-01 7.d54E-01 \$ \$ \$ ź A-UR \$ \$ \$ ¢ C C A-0R A-DR LONG\* LONG ≎ \$ \$ \$ •1953 .2256 1446. --2404 6441 KANGE DOT 3.897716E+01 4.100000E+02 ¢ \$ ¢ \$ ROLL ROLL ċ ¢ ٥ ¢ 9.4825.00 9.4825.00 9.4825.00 9.6015.03 1.205E+01 1.206E+01 2.496E+03 9.481E+00 9.482E+00 13.5876 V=0P ٥ \$ 12.6072 2+275E-03 \$ ¢ 2.791E-03 \$ ¢ OUTPUT ¢ ¢ . MO7 COM. 40-A 8 8 A A 800 d0=1 \$ \$ \$ \$ SUN ANG. ANG. 159.483 ٥ ¢ \$ ¢ .0227 .4236 -.9056 .0227 .4236 .9056 SUN SENSOR ISU/C.P.S. \$ <sup>.</sup> \$ \$ ISU/C.P.S. SUN MARS MISSION, ERROR ANALYSIS \$ ¢ LATS \$ \$ LAT= 1.929E+01 1.929E+01 2.552E-03 1.220E+01 1.220E+01 4.775E-03 1.220E+01 1.220E+01 3.157E+03 5.3125-03 90 6.838069F-08 **a**0 \$ \$ \$ \$ A-CR V-CR \$ \$ ٥ \$ C С V-CR ALIMUTH •1693 •9349 •3118 •2083 •2083 \$ 797.48 ċ 89.783 66.410 \$ ź . ۲ ELE. \$ ¢ \$ ≎ SUN SENSOR ĩ g \$ ¢ \$ \$ 4.789E+01 4.789E+01 4.789E+01 1.705E-03 5+204E+01 5+204E+01 ₽ \$ \$ 1.010E-03 1.2375-02 1.4486-02 4.7895+01 ¢028° .23399 ¢ ٩ \$ -.3541 V-0R V-DR 01 Ŷ V=0R ANGu \$ ANGa ¥ \$ ELEVATION 4.8910742-08 \$ 155.046 MZ.A g MZA ř ٥ \$. \$ THCKR \$ \$ \$ THCKR ¢ \$ 9.4120675E+06 \$ ¢ 7.4950784E+06 1.7825+07 7.9035+03 エエエメン ч Т Т 17 2.599E+07 2.599E+07 1.782E+07 2.186E+03 4.485E+03 5.104E+02 1.7826+07 1.7825+07 \$ ¢ \$ \$ STAR STAR p=0p \$ \$ \$ 2.81652916+04 î ¢ P-0P 0 •01 d0-d - 3226 - 7420 - 5877 --3226 -7420 .5877 \$ \$ PITCH \$ PITCH \$ RANGE UOT 101 ٥ \$ \$ \$ ATT. CONT. 8.086810E+03 CONT. 5.0000006401 \$ \$ ٥ \$ RANGE 2.5205.07 2.5205.07 5.8156.03 3.7645+07 3.7645+07 3.764E+07 4.122E+03 3.7625+07 3.7645+07 3.7646+07 3.275E+03 \$ \$ .076E+03 ¢ \$ ∿£%0° - 5032 - 5032 - 6536 2508\*-..... 0.00S) (500-0 ٥ \$ ATI. ( \$ \$ RANGE VEL= ΥAW VELa YAW 20-0 с ; ۰ ډ p-CR anon anon ≑ \$ \$ \$ \$ ٠ \$ ⊅ •0603598E •1 νo \$ \$ ≎ 9023H58M \$ •51.68 -.2832 9700 ·-1.0603424E+11 6203 --1.3266059E+11 ----s, U. \$ \$ \$ \$ 1.8388\*08 1.8398\*08 •∩ •5 1.8385\*08 1.9385\*08 3.7355\*02 2.23%F\*08 2.23%F\*08 5.104F\*03 301L 8.083F+03 H O ງີເດສ 1.6145\*02 MEASUREMENT ERRORS INERTIAL SYSTEM \$ ¢ \$ RANGE \$ RANGE р**-**Пр 90**-**9 du-d \$ \$ C S ≉ ( 100 ≎ ç • \* \$ ٥ ≎ ٩ н х ≻ 17 1 H H X X Z ۵ \$ \$ \$ PIS0 \$ ¢ ¢ \$ SEC. SEC. COMPUTER COMPUTER ≎ \$ DEVIATIONS ERRORS \$ COMP. DEV. DEVIATIONS \$ COMP. DEV. DEVIATIONS ERRORS COMP. DEV. ATTITUDE ATTITUDE BOLDSTONE KALMAN HPDATE 6049980 ALT= ٥ \$ \$ 6913980 ALT= ٩ STATION STATION 02 01 ERRORS FRRORS ₽ \$ ¢ 864000.00 SOO°U WEEHO 863880.00 ٩ 0.005 ≎ \$ \$ \$ ٠ \$ ψ ≎ SUBSYSTEM SUBSYSTEM MEGHO HELIOCENT. HFLIOCENT. HELIOCENT. STATUS HELIOCENT. STATUS AFTER 700 с С Cod 20 H 11 |-11 |---COAST C0A57 - -۸T

FIGURE A-9m.

ŵ ΰ ¢ 7.8545-01 7.8545-01 7.854E-01 7.854E-01 854E-01 7.8545-01 7.8545-01 7.8545-01 .8545-0 \$ .8545+0 ≎ .854E-0 9 ÷ 7 BEAF 40-V \$ ψ ac \$ A-OP AU-A ψ φ ⊅ \$ ż \$ ⊅ ¢ \$ ¢ ۵ XMITTER A-CR 7.054E-01 7.854E-01 7.8545-01 7.0545-01 \$ ٦ 7-8540-01 7-8540-01 101 .854E-01 \$ ۵ .8545-01 7.854E-01 .854E-01 7.854E-01 0496. •1414 -.2252 ¢ ₽ ¢ PITCH ż PITCH A-CR A-CR 0 •854E ¢ \$ ≎ -103,6603 -92.7982 ٥ \$ ٥ ¢ ٥ ¢ \$ RADAR \$ .0855 .9668 -.2407 ≉ ¢ ۵ \$ 7.854E+01 7.854E+01 7.854E+01 7.854E-01 7.854E-01 7.854E-01 7.854E-01 7.854E-01 7.854E-01 7.854E-01 7.854E-01 7.854E-01 ΥAW YAW 7.854E-01 \$ ٢ \$ ŵ АРР. 0 A-UR A-DR A-0R \$ \$ \$ ¥ A-UR L0NG= LONG= \$ \$ \$ ¢ .2518 1446. RANGE DOT 3.074993E-01 4.100000E-02 RANGE DAT 7.4763656-02 4.1000006-02 \$ \$ \$ \$ RCVR. ROLL ROLL \$ \$ ¢ \$ 1111 1.5365+01 1.5365+01 1.3585+01 1.536E+01 1.536E+01 6.501E-04 1.9125.01 1.9125.01 6.3465.04 1.206E+01 12-3708 V-0P 12.2202 V-0P \$ ¢ \$ \$ 1.4246-03 ¢ \$ ۶ \$ 40**-**7 200 .MOC d0-1 \$ \$ ¢ \$ SUN ANG. 151.512 ≎ \$ \$ \$ 143.651 •0227 •4236 •9056 SENSOR ISU/C.P.S. ٥ ¢ ☆ ۵ \* \$ LAT= \$ ۵ n 2.7995+01 2.7995+01 2.2635-04 1.929E+01 1.929E+01 1.964E-04 2.7995+01 2.7995+01 1.3555-04 3.8245+01 3.8245+01 1.458913E-08 å 5.433642E-09 LAT 1.643E-04 0. 0 \$ \$ \$ \$ V=CR ⊅ ₽ ٢ C \$ 4-CR 20-2 ALIMUTH AZIMUTH ELE. 63.941 \$ \$ 89.811 .8879 .4078 65.067 •2130 ¢ \$ 89.825 ۵ \$ \$ \$ g g \$ \$ \$ ¢ •0-5.7276+01 5.7276+01 1.0736+04 5.3045.01 5.3045.01 8.3785.01 д.727£+01 5.727£+01 3.647£−05 \$ ≄ ٩ ¢ 5003. 3668 ⊅ \$ \* ≎ î ELEVATION 5.7357976-08 -0. V=0R V~DR SUN <u>v-09</u> ANGS ang= ¢ ¥ \$ \$ 153.238 149.490 2.1256195-08 -0. MZA C H 20 \$ \$ ≎ ۵ ELEVATION THOKR \$ ٥ \$ ≎ \$ \$ \$ ≎ 1.16194445+07 1.40971706+07 5。1455407 5。1455407 2.1625403 чн Х≻ 3.7075.07 3.7075.07 3.3992.03 2+5995+07 2+5995+07 7 3.707E+07 3.707E+07 2.1365+03 1-609E+03 ٥ ۰ ٥ \$ STAR d0-d ≎ \$ 3.30523276+04 \$ 00-0 3.79617285+04 ¢ d0-d 0 1 d0-d ••3226 •7420 •5877 \$ \$ \$ \$7 PITCH HOLICH 001 ¢ ¢ \$ Ý ATT. CONT. 4.201949E+03 5.000000E+01 1.648351E+02 5.000000E+01 ¢ \$ ٦ \$ 2.8325+07 2.8325+07 5.7425+02 RANGE 2.520E+07 2.520E+07 1.1502007 1.1502007 1.1502003 ٦ 1.1495,07 \$ £, 1.1036+03 \$ 1.1406+07 5.4415+02 N FU S \* 1 ·0434 0.005) 0.005) ¢ \$ ≎ \$ PANGE RANGE ч Ч П С П С П С П 80-a YAW p=CH YAW 0 1 z. ≎ ≎ ¢ \$ \$ \$ \* 5+11 1.3266254E+11 \$ ¢ ž ž \$ \* 1.6356564E\*11 P-DR 02080-1.98556112.11 ີ ວີ. ຍິ ŧ \$ ⇔ 4 91123H2 2.7235+03 2.7235+03 3.1965+03 2.7235+08 2.7235+08 3.2615\*08 3.2615\*08 1.7105\*02 2.234F+08 2.234F+08 HO 1.613F\*02 JUCA 1.200E\*02 1704 MEASURE 1.6556877 MEASUREMENT ERRORS INCRITAL SYSTEM ټ MEASURGMENT ERRORS INERTIAL SYSTEM ---ţ. ٦ RANGE 001 P...D2 aù-a P-DD ¢ ٥ \$ C I \$ • ≎ \$ ŵ \$ ⊷-->< Υ HZ. 9120 ۵ -\* 0515 \$ \$ \$ \$ \$ ÷ SEC. SEC. COMPUTER COMP. DEV. Deviations Errors COMP. DEV. DEVIATIONS \$ \$ \$ ÷ COMP. DEV. NEVIATIONS ATTTUDE COMP. DEV. DEVIATIONS ZONLIIV V GOLPSTONE GOLDSTONE KALMAN UPDATE KALMAN UPDATE ALT= \$ \$ 2777580 ALT= STATTON \$ **:**? 0 1 RADRS ERRORS **FRORS** \$ \$ ٩ \$ 864090.00 0.003 862020-00 00 0°002 4 ¢ \$ ⊅ 8640000 ٥ ≎ \$ ٩ ũ SU9SYSTEM MEEHO HEL LOCENT. TOCENTS HFLIOCENT. HELIOCENT. STATUS Чo AFTER AFTER GUO с О R 0000 FOR Г. Бл 1) |---11 5----COAST 00451 7 7 14 A-43

OUTPUT

ERROR ANALYSIS

MARS MISSION,

A-9n.

FIGURE

5-175-02 .85¢E-01 ý ΰ 0 . 854E=0 " B54E. 0 ¢, ≎ \$ du-d d C = V 3345 \$ ¢ ų ó ≎ \$ ≎ XMITTER XMITTER ø \* 7.854E-01 .2479 .2479 2246 .9422 î 0 ĉ -.2252 -,2252 ¢ ¢ PITCH .8545 .8545 •8545 ..717E ≎ \$ -100.2913 \$ \$ \$ APP. RADAR \$ APP. RADAR .1927 -.9513 .1927 .9513 -.2407 -.2407 \$ \$ 1.717E-02 7.854E-01 7.854E-01 YAW E-01 .854E-01 \$ ≎ \$ \$ A-UR C 0 ż .8545 LONG \$ \$ × .1834 •2739 -.1834 1446. .9441 ¢ \$ \$ Ó RCVR. RCVR. 60SEC ROLL \$ \$ \$ φ 12.3720 V-0P 1.912E+01 1.912E+01 \$ \$ \$ ۵ 6.346E-04 6.346E-04 •912E+01 90.00 ≄ ۵ \$ \$ COM. a c COM. 2008 ΩВ ٥ Ş \$ ۵ \$ ٥ ٥ \$ • 4236 SUN SENSUR ISU/C.P.S. v \$ .0227 •4236 SUN SENSOR ISU/C.P.S. --95056 \$ \$ AIDS= \$ ≉ LA7= Z \$ \$ 3.824E+01 3.824E+01 3.824E+01 3.824E+01 90 1.643E-04 1.6435-04 \$ ψ ¢ \$ -21.715056. 0 \$ ¢ \$ ¢ UPTICAL •4954 • 2525 .8312 +956\*\* \$ \$ 89.825 \$ .8312 .2525 ٥ \$ \$ \$ \$ ž \$ \$ \$ ¢ с. 998₽ 01 5. 993₽ 01 5.719₽ 05 \$ \$ 5.7198-05 \$ ÷ - 0 + ÷0] UETNEEN ••5556 •7584 • 4556 • 7584 . 340 H ٥ .340§ ¢ \$ ¢ и, 5986 1, 5986 ROLL Ŷ X-08 X+08 ANG \$ \$ \$ \$ č ¢ ¢ ¢ \$ ¢ \$ \$ \$ TROKE TXOXN PRPGTE. 95+70+ PRPGTE ٩ ¢ ¢ ٩ -5+238046. 1.4102964E+07 X 17 X 17 5.145E+07 5.145E+07 >--+04 2.1625+03 2.162E+03 .145E+07 ¢ \$ \$ ≎ ATI. CONT. SIAR \$ \$ \$ STAR \$ î 0010 d0-d .5877 •7420 •5877 .7420 5.145 -.3226 -.3226 SUN-CANOPUS \$ g \$ ŝ \$ ٥ PITCH \$ \$ BETWEEN CELESTIAL RUDIES= ۵ \$ ATT. CONT. ¢ \$ \$ ¢ YAW 2.8325.07 2.8325.07 5.7425.02 ٦ \$ \$ \* E+07 10 \* 3 5.742E+02 5502. + C 4 7 4 -- 4270 •0\*34 ..9032 --4270 0.005) 0.005) ¢ ψ £, ¢ VEL B-CR • ΥA₩ 64.00 50.30 30.30 0-CR MAX. . 832E ¢ \$ ¢ \$ TO LOCATE -23.555DEG. \* ŵ \$ \$ ż ¢ ٥ Σ 4 MOCHO .5168 - 8079 -+ HU79 .5168 1.9867048E+11 P-DA -25832 --2832 °. \$ ¢ \$ \$1 2.41 2.41 5 د م 3.2417+08 3.2417+08 1.710502 E 4 0 ¥ RULL SET 1.710502 .2415+08 \$ ۵ \$ ŝ •. •. \$ àù-d ¢ ŝ 5 ¢ ç \$ \$ MANEUVERING • \$ • ⋨ \$ \$ HANDS PITCH 80LL 117 7 X X PITCH их Х Z \$ **ņ** ÷ \$ -\$ \$ \$ ¢ sec. COMPUTER COMPUTER 60.00 SEC. COMP, DEV. DEVIATIONS Errors **NGLE** \$ \$ ¢ \$ <u>z</u> 20 20 ZC NO NO DEVIATIONS COMP. DEV. ATTITUDE DEAU MANEUVERS ¢ ≎ Ì \$ ≎ î TURNED TURNED TURNED TURNED TURNEO ERRORS TURNED 9641800 ALT \$ \$ ≎ ¢ 1800.00 0H30M 0.00SH0 BEGIN ≎ \$ ¢ ≎ \$ ☆ ¢ Ŷ SURSYSTEM SUHSYSTEM STAR TRCKR SUN SENSOR ATT. CONT. HFLIOCENT. HFL JUCENT. HEL TOCENT. HELLOCENT. STATUS STATUS XHITTER COMPUTER • S• U. 0001 COAST FOR F 0.0 <u>n</u> COAST 14

A-44

FIGURE A-90. MARS MISSION, ERROR ANALYSIS OUTPUT

становли и на станова и на станов

A-00 1.7176-02 7.8546-01 7.8546-01		0		*					- - - - -		\$ \$ \$ \$ \$ \$ \$ \$ \$ \$ \$ \$ \$ \$ \$ \$ \$ \$ \$		A-OP 1.7176-02 1.7196-02 6.8536-02	•	
6411 A-CR • 71 (E-02 • 85%E-01	PITCH 9497 11 28897 11 28897	K XMITTER 1		\$ \$ \$		:			0 • 0 0 0 0	0 • 0 0 0 0	* * * *	606	A-CR •71/E-02 •717E-02 •653E+04	PITCH •9497 •2382	XMITTER
6= -100.5 -08 176-02 546-01 7546-01	7 44 1 • 1 757 1 • 9427 • 2838	PP. RADAF		≠ \$ \$ \$					•PIÌCH	• PITCH	₽ ₽ ₽ + + + + + + + + + + + + +	G= =100,7	-DR 175-02 1 175-02 1 755-04 2	YAW •1757 •2438	PP. RADAF
720 LON +01 1.7 +01 7.8 -04 7.8	ROLL • 2594 • 2338	RCVR. A		. ☆ . ↓ . ↓		) o		60SEC.	.0000	02612	•	120 LON	01 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1	ROLL • 2594 • 2338 • 9371	RCVR. A
н н н н н н н н н н н н н н	227 DR 236 CR 056 OP	•\$• COM.	1	*		5# 90 •			XAW O	WAY	¢ ¢ ¢		× 0 1 0 1 0 1 0 0 0 0 0 0 0 0 0 0 0 0 0	227 236 05 09	.S. COM.
. 825 LAT V-CR 3.8246+01 3.8246+01 3.8246+01 1.6436-04	18 8312 4954 2525 - 9	R ISU/C.P		\$ \$ \$		TICAL AID		000DEG. I	PASSES	PASSES 000	¢ ¢	.825 LAT	V=CR 3.8245+01 3.8245+01 1.6435-04	R 8312 4958 2525 • 9	H ISU/C.P
	•.• •.• •.• •.• •.• •.• •.• •.•	r SUN SENSO				BETWEEN OF		BOLL -	1.00000	1.0000		ANG= 89	<pre></pre> <pre><td>108 008 008 008 008 108 108 108 108 108</td><td>SUN SENSC</td></pre>	108 008 008 008 008 108 108 108 108 108	SUN SENSC
	ТСН 7302 XI 5161 YI ¢477 XI	STAR THCKP 1	N-CANOPUS	÷ ↔ ↔ ↔	NO PRPGTE.	= 95.70.		-•0000¥6	CONDS WITH RAD/SEC)=RO	CONDS WITH RAD/SEC)=HC	* * * * * *	03351E+07		TCH 7302 XI 5161 YI 4477 ZI	STAR THCKE
1925 1925 1925 1925 1925 1925 1925 1925	та ма сто сто сто сто сто сто сто сто сто сто	ATT. CONT.	RCH FOR SU	\$ \$ \$ \$ \$	(\$00.0	TAL DODIES	м м м м м м м м м м м м м м м м м м м	G. YAW	AR RATES (	29+9 SE AR RAJES (	* * * *	VEL= 1.41	Р-СК ВЗ26407 5 2422407 5 7422407 5	YAW 900000 900000 900000 900000	ATT. CONT.
418408 2 418408 2 418408 2 104402 5	FULL • 0715 • 7094	• 5• U.	BEGIN SEA	\$ \$ \$ \$ \$	00 0H 1M	EEN CELEST	S N N N E N N N E N N N E N N E N N E N E	•0000	IN ANGUL	PUS IN	\$ \$ \$ \$ \$	84195+11	*18*08 2. 418*08 2. 418*08 2.	POLL • 0715 • 7094	ຳ ດໍະ <b>ງ</b>
AMP DEV. 3.2 VIATIONS 3.2 TROODS 1.2	ATTTTUDE X X Z Z	COMPUTER I	ND MANEUVERING	* * * * * *	0.00 SEC. (	ANGLE BETW	DEAU BANDS ROLL ROLL YAW PITCH	HOLIG .	LARCH FOR SUN	LARCH FOR CANO	* * * * * *	).005 ) ALT≕ 1.985	MMP. DEV. 3.2 S.S. 2.2 S.C.S. 2.2 S.C.S. 2.2 S.C.S.S.S.S.S.S.S.S.S.S.S.S.S.S.S.S.S.S	ATTITUUE XED XI XI XI XI XI XI	COMPUTER I
		SUASYSTEM STATUS	Ц Ц	¢LIOCENT∘ ∻	F.O.R 6			2	S S	S N	ELIOCENT. #	000 0H32M 0 = 8641920		S ACQUIS	SUSSYSTEM
				ĭ	COAST !						ñ	1 L L		CANDPUL	

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FIGURE A-9p. MARS MISSION, ERROR ANALYSIS OUTPUT

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ŵ 9.889E-04 7.1255-04 .889E-04 8555-04 6 858E-04 ٥ 7.125E-¢ 0 A C M ٥ ٥ ż \$ ¢ XMITTER 7.1258-04 7.6078-04 2.6658-04 \$ \$ 7.607E-04 7.125E-04 2.6655-04 -.2382 1676\* -.2034 \$ \$ PITCH A-CR \$ ¢ LONG= -101.0406 \$ ≎ \$ RCVR. APP. RADAR \$ ø ٠ \$ \$ 7.581E-04 2.588E-04 YAW 7.125E-04 7.125E-04 7.581E-04 2.588E-04 ¢ \$ \$ \$ ٠ ٥ A-UR C \$ \$ ⊅ .2594 .2338 1750. RANGE DOT 5.813816E-02 4.100000E-02 \$ ٩ \$ 60SEC. ROLL 4 ¢ \$ \$ \$ \$ 12.3721 6.346E-04 1.912E+01 1.012E+01 1.912E+01 - 981€-04 1.012540 90.00 00.06 ٥ \$ ٥ SUN SENSOR ISU/C.P.S. COM. 00-A 200 40= ^ ≎ \$ ۵ SUN ANG. 136.253 \$ ¢ ¢ • 0227 • 4236 • 9056 ¢ \$ ≄ 95.70. BETWEEN UPTICAL AIDS= A105= Z \$ ¥ 4 LAT= 3.824E+01 3.824E+01 2.832549E-09 **n** 0 1.643E-04 3.8245+01 1.002E-04 3.824E+01 ₽ ٥ \* -•0000EG. V-CR \$ \$ ٠ V=CR UPTICAL AZIMUTH 89.825 •8312 •4994 •2525 ≎ ¢ ¢ 62.908 ELE. \$ \$ \$ g ¢ \$ \$ 6.998E\*01 5.998E\*01 5.9985.01 5.9985.01 \$ \$ \$ 2.9098-05 5.1192-05 BETWEEN • 5555 • 7584 90%50 \$ ۵ \$ V-DR ROLL V--0R \$ ¢ ≎ =9NA ELEVATION 1.0916336-08 144.241 ž A.Z.M \$ ¢ ¢ ATT. CONT. STAR THCKR ≎ ¢ ٥ PRPGTE. PRPGTE. 95.70+ -.0000EG. ٦ ۵ ¢ 1.41035445+07 5.145E+07 5.145E+07 2.162E+03 エエエス 5.145E+07 5.145E+07 1.3205+03 ÷ ۵ ۵ d0-d \$ ₽ 4.2087921E+04 •0• ۵ d0-d --7302 -5161 -477 g 02 ¢ ۵ \$ PITCH 001 ₽ BETWEEN CELESTIAL BUDIES= \$ BETWEEN CELESTIAL BUDIES= . ອີມີດ ອີມີດ ¢ DEG. 1.157263E+02 5.000000E+01 \$ ψ ۵ YAW RANGE 2.8325+07 2.5325+07 5.7426+02 \$ ¢ 2.8325×07 2. P32E4.07 \$ 3.3568402 .10000 .10000 .10000 (SOO .0 ML HO IM 0.005) ¢ \$ \$ VEL= DANGE a-CR 00\*\*5 2\*00 64 • 00 8 • 00 р=СR YAW .XAM -30.30 MAX. \$ ₽ ۵ +01005G+ \$ \$ ¢ SUN-CANOPIJS DLDSTONE 1.9968923E+11 MEASUREMENT ERRORS F INERTIAL SYSTEM 1. ₽ ¢ ¢ \$502° 1.9863694E\*11 -.7012 00 .10 ≎ \$ \$ • 10 • 1 o 3.2018408 3.2018408 1.7108402 с. С. 3.2415\*08 3.2415\*08 т Г 581 SET 7708 1043214-6 ≎ φ 222 ÷ RANGE 00 a0-d ŝ \$ \$ ¢ CHANGED CHANGED -, D-Da CHANGED \$ \$ \$ PITCH 722 YAW I TCH エ エ エ て て て て ROLL DEAD BANDS ROLL HANDS F08 -≎ ≎ \$ 9180 \$ ¢ ٩ sec. SEC. a COMPUTER COMP. DEV. DEVIATIONS ERRORS ANGLE ANGLE SEARCH \* \$ COMP. DEV. DEVIATIONS :: ATTTTUDE •---• N m DEAD GOLDSTONE MANEUVERS KALMAN UPDATE ţ, ٥ 8641980 ALT= \$ STATION ERRORS ~ ¢ 60°00 ¢ 00°09 DEADMAND ON AXIS DEADMAND ON AXIS DEADMAND ON AXIS φ 500.00 ÷ \$ ¢ ACOUIRED END ND \$ **:**1 ¢ SUHSYSTEM NECHO HELIOCENT. STATUS HFLIOCENT. HELIOCENT AF 15R н С Ц 1000 COAST FOR # |--CANDPUS COAST 44

MARS MISSION, ERROR ANALYSIS OUTPUT

FIGURE A-9q.

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PITCH .10 30.30

٥ \$ .1255-04 .895E-04 .902E-04 .125E-04 6.865E-04 7.1255-04 8648-04 876E-04 ٩ ₽ .895E-0. ٥ 4-0P ψ A-O-A A-O-A ¢ ¢ \$ ¢ ż ź s S σ σ ¢ \$ XMITTER XNITTER 7.125E-04 7.6245-04 2.713E-04 7.1258-04 7.1258-04 7.58885-04 7.6168-04 2.6095-04 2.6858-04 \$ \$ 255-04 .6145-04 2.685E-04 1040. -.2385 • • 2362 --2034 --2034 1942 C PITCH ٩ ₽ PITCH A-CR A-CR ⊅ ۵ -101-2904 -101-5402 ≎ \$ 7.1 ₽ RADAR \$ RCVR. APP. RADAR ٥ -.1757 -.9426 -.1757 -.9426 •283g •283g ¢ \$ 4 7.125E-04 7.598E-04 2.637E-04 ۸d۲ YAW 7.588E-04 7.125E-04 2.609E-04 \$ ٩ ≎ A-UR COM. RCVR. APP. \$ \$ \$ 0 A-DR A-0R C LONG= LONG= ≎ \$ \$ .2594 .2338 .9371 .2594 .2338 1759. ¢ ¢ \$ ROLL 60SEC ROLL \$ \$ \$ 12.3722 V=0P 1.912E+01 1.912E+01 12.3721 V-0P ŧ 1.912E+01 1.012E+01 \$ \$ 3.881E-04 3.8806-04 3.8805-04 1.912E.01 1.0125+01 ¢ \$ \$ • WOU 8 8 8 0 0 0 a a a 0-0 \$ \$ \$ \$ \$ \$ .0227 4236 .9056 •0227 •4236 SUN SENSOR ISU/C.P.S. \$ SUN SENSOR ISU/C.P.S. \$ \$ -.9056 \$ \$ ¢ 89.825 LAT= Z LAT= 3.8245+01 3.8245+01 1.0055+04 3.824E+01 3.824E+01 3.824E+01 3.924E+01 a å 1.002E-04 1.005E-04 ٥ ٥ \$ (FI/SEC) . UOF=1.767 -•000DEG. \$ 40-7 \$ **V-CR** \$ 4-05 • 4954 • 2525 .8312 .4954 ≉ 89.825 ¢ ٥ 8312 • 2525 ≎ ¢ \$ g g \$ ⊅ \$ 5.9985401 5.9985401 2.9085401 д•9982\*01 д•9932\*01 д•17¢5\*05 ¢ \$ \$ 3.1746-05 5.993E+01 **ч**•У98Е+01 .340H 99995 9995 9995 ≎ .7584 3405 ¥ \$ -.5556 ROLL V+0K V+0.8 V-DR ANGE \$ \$ \$ ANG α a ٥ \$ \$ THCKR \$ 1XOXE \$ ۵ PRUGTE --000DEG-\$ 177 1.4348E-02 ٥ \$ 17 1.4103737E+07 ž VEL= 1.4103930E+07 5。1456407 5。1456407 1.3202403 нн Х≻ 5.1455407 5.1455407 1.3265403 5.1452+07 1.320E+03 ⊅ 6.145E+0 ¢ \$ STAR STAR д0**-**д dn-d ٩ ¢ 9-0P --7302 -5161 -4477 .5161 .4477 -.7302 ₽ ٥ 0 Z ≎ PITCH PITCH SECONDS 29,00000 056. 20,00000 056. 20,00000 056. \$ \$ \$ ATT. CONT. ATT. CONT. ΰ ≎ \$ YAW 2.8325.07 2.8325.07 3.3562.02 2.8327407 2.8327407 3.3557402 \$ □) + RMV= \$ ٥ 2.8325+07 3.3565+02 2. A32F.+ U7 0000° 100° 100° 00540--,55549 4670.-+010°-0.00S) 16934400 \$ ¢ \$ р-С2 p-CP Y AW WAY W \$ ¢ \$ •000DE6• ¢ \$ \$ \$ Σ \$ ¢ --7012 \$170. 2170. 1.98692455411 -,7012 1.98689689891 •0715 • 7094 \$ \$ \$ CORRECTION (IMC= S. U. 3.2417+08 3.2417+08 9.4127+01 ς. υ. 3•2415•08 3•2415\*08 9•4128\*01 3.2415+08 3.2415\*08 r I CHANGED TO 2 CHANGED TO 3 CHANGED TO BULL 9.412E\*01 POL \$ **\***5 ٥ DEVIATIONS AT P-0R b-De 9–72 \$ ĉ ¢ 4 • ¢ \$ ۲. ۲ \$ н X X N X ž PITCH ۵ ≄ N ٩ -\$ 4 \$ 350 0 07.5 COMPUTER sec. COMPUTER COMP. DEV. Deviations Errors COMP. DEV. DEVIATIONS ERRORS COMP. DEV. DEVIATIONS \$ \$ ¢ ATTTUDE 3CUTITIA SCUTITIA MANEUVERS ¢ ≎ 3642100 ALT= \$ 8542040 ALT= TURNED GEREAR. ERRORS \$ COMPUTED ¢ STADEAND ON AXIS DEADRAND ON AXTS DEADRAND ON AXTS \$ 500°0 MyCH0 60.00 200° U MSEH0 AFTER WIDCOURSE \$ ≎ ٢ ACOUIRED \$ \$ ¢ NO GNURGULA SUBSYSTEM MBTRYSHUR FELIOCENT. HELICCENT. STATUS HEL TOCENT. STATUS 1200 8211400 TO ZERO 0008 18 1000 COAST COR 11 1--11 }--SUGONVO A 1

FIGURE A-9r. MARS MISSION, ERROR ANALYSIS OUTPUT

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¢ \$ 854F-01 .8545-01 854E-01 854E-01 . 854E-0] 7.854E-0 ÷ .854E-0 .854E.0 ¢ .854E-0 B54E-0 \$ •854E=0 Φ \$ ¢ ŵ A-OP 40-V au-v ¢ ⊅ ٥ ₽ \$ \$ 1 5 ₽ Ŷ ٠ ۵ XMITTER 7.854E-01 7.854E-01 7.8545\*01 7.8545\*01 7.8545\*01 7.854E-01 7.854E-01 7.854E-01 \$ ₽ \$ ¢ 5 5 -.1335 -.2034 7.8545-01 .0201 -.0201 .854E-01 0025. PITCH ₽ ٩ \$ PITCH ≎ A-CR 0 A-CR .854F 0554F \$ \$ \$ \$ -94.3070 -97.8913 Φ ¢ ≉ \$ RADAR \$ ¢ \$ ¢ --0721 --9562 --2838 .0295 .5584 ≉ \$ \$ \$ 7.8545-01 7.8545-01 7.8545-01 7.8542-01 7.8542-01 7.8542-01 YAW YAW .8545-01 7.854E-01 7.854E-01 ទី 7.8545-01 7.854E-01 ≎ \$ \$ ¢ APP. A-UR A-UR A-DR \$ \$ \$ \$ C A-DR 7.854E LONG# LONG= \$ \$ \$ \$ -2524 -2506 -3371 .2035 \$ ¢ ۵ 4.368537E-02 4.100000E-02 φ RCVR. ROLL ROLL ¢ ⇒ ≑ \$ RANGE DOT 2.8255401 2.82555401 2.0005-04 12-9775 V-0P 2.3605+01 2.3605+01 3.7565-04 2.350E+01 2.350E+01 2.953E-04 \$ \$ \$ \$ 10+3 3.880E-04 13.5451 1-9125+01 \$ . ¢ \$ 40-N • MOC d0-1 880 а С З 40-V \$ ANG. \$ 1.012 \$ \$ \$ \$ 129.347 ٥ ¢ . 0227 . 4236 . 9056 SENSUR ISU/C.P.S. • 0227 • 4236 \$ \$ ¢ • sun ¢ LATE ¢ ۰ ± LAT= 4.9476+01 4.9476+01 1.2486-04 6.1067+01 6.1067+01 1.3625-04 3.824E+01 3.824E+01 dO 2.108715E-09 -0. 60 1.005E-04 4.9475+01 4.9475+01 \$ ¢ 1.0248-01 \$ \$ V-CR ¢ ₽ \$ 20-2 ٥ c V-CR AZIMUTH .7658 .5750 .2881 •6934 •6458 ¢ \$ ¢ ¢ 89.838 61.810 89.851 ELE. ¢ \$ \$ \$ ð g \$ ŧ \$ \$ 6.062E+01 6.062E+01 3.741E-05 д. - 9 4 3 9 д. - 9 4 3 9 д. - 9 4 3 9 д. - 9 0 6 - 9 0 - 9 0 - 9 0 - 9 0 5.9988401 5.9998401 3.1748405 6.062E+01 6.062E+01 \$ \$ \$ 3.1436-05 \$ 5057. 5055: - 6421 - 7000 ¢ ٢ ⊅ \$ V-0R V-0R ¥0~∧ • ELEVATION 6.7097935-09 -0. SUN V-0R \$ \$ ANG 137.915 ⊅ \$ ANGS AZM č ся \$ ۰ پ \$. \$ 13 ¢ THOKR \$ \$ \$ \$. ⊅ ≄ 1.6437919E+07 エエエス 1.9785063E+07 ч н х У 6.975E+07 6.975E+07 1.639E+03 Р-ОР 9.2036+07 9.2036+07 1.4476+03 5.145E+07 5.145E+07 6.975E+07 6.975E+07 1.3205+03 1.2505+03 ¢ ¢ ' \$ \$ SIAR d0-d d0-d ¢ \$ î 4.7082053E+04 ≎ \$ d0-d --7302 -5161 -4477 .5161 PITCH PITCH ≎ \$ **‡** [ 001 ATT. CONT. S ÷ ۵ ۵ ۵ 1.065850E+02 5.00000E+01 ¢ ≎ \$ \$ RANGE 2.8325.07 2.8325.07 3.3565.02 1.1195+08 1.1195+08 2.3955+02 6.468E+07 3.410E+02 \$ \$ 6.4685+07 6.4686+07 2.2156+02 ۴ \$ 6.468E+07 -•6794 -•4000 1:00:0 --6794 (\$00.00) \$ \$ \$ RANGE V5L= MVX YAW V£L≓ 5**~**0 p-CR P-CR p-CR ¢ ÷ ٥ \$ ۵ ٩ ⊅ \$ 2.3731092E+11 Σ ÷ 9023H58M ٥ ≎ ≎ •0115 •7094 7094 2.8059857E+11 •0715 2.3/808445+11 P-DA s• ∪. \$ \$ \$ \$ 3 - 7695 + 08 3 - 7695 + 08 8 - 9455 + 01 3.7695\*08 3.7695\*08 7.3796\*01 6.2785\*08 4.2785\*08 7.5095\*08 3.2415+08 3.2415+08 J\_lOa g ROLL 9.412F\*01 \$ ¢ SUREMENT ERRORS ÷ ٥ RANGE P-0R 101 ΰ ¢ C t \$ \$ P-02 20-÷ \$ \$ \$ \$ I X X Z ~ DSIF ÷ \$ \$ -\$ MEASUREMENT ¢ ¢. UEV. DEVIATIONS FRRDRS \$ ¢ SEC. 075 055 970 SEC. сс Ш COMP. DEV. Deviations Egrors COMP. DEV. Deviations Errors COMP. DEV. DEVIATIONS ERRORS t: ≎ \$ \$ ATTITUDE ATTITUDE COMPUT GOLDSTONE KALMAN UPDATE \$ 9505980 ALT= STATION **\$**\$ \$ AL T= 2 TURNED 01 TURNED ¢. \$ 864000.00 ¢. ¢ 863830.00 0.005 0.005 ٦ \$ \$ ≑ \$ ¢ \$ ₽ 103699801 I. S. U. Star Irckr Sun Sensor SURSYSTEM 0H33M M88H0 HEL IOCENT. HFLIOCENT. HELLIOCENT . STATUS IOCENT. α ω 1100 F 0 P 1200 а С 1 111 11 1-" COAST 72400 ∆ † 77

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FIGURE A-9s. MARS MISSION, ERROR ANALYSIS

OUTPUT

ψ \$ .854E-01 •854E-01 .854E~01 .854E-01 854E-01 \$ ÷ \$ 854F-0 A-OP .854E. ¢ ΰ \$ .854E  $\mathbf{c}$ C .854F A-OP ⊅ ⊅ ¢ ¢ ¢ r \$ ¢ ¢ 10 XMITTER XMITTER å 10-¢ ¢ ¢ .8546-01 7.8545-01 .8545-01 7.8545-01 •9765 •0714 6 7.8545-01 7.8545-01 7.854E-0 -.2034 -.2034 \$ \$ \$ ٥ PITCH A-CR 0 0 A-CR 7.854E 7.854E ≎ \$ \$ ٥ -86.1070 -90.3611 ≎ ٥ ¢ \$ \$ ¢ ₽ ¢ RCVR. APP. RADAR RCVR. APP. RADAR .1286 .2839 2096\*-•283g ¢ \$ ¢ \$ 7.854E-01 7.854E-01 7.854E-01 YAW 5 7.854E-01 7.854E-01 .854E-01 7.854E-01 ۵ \$ \$ ٥ .854E-0 \$ ۰ A-DR ٦ A-DR \$ 0 A-0R 7.8545 0 =9N0T Long= ≉ \$ \$ ¢ 1730 .9371 \$ ¢ \$ \$ 3.444212E-02 4.10000E-02 3.036059E-02 4.100000E-02 ROLL ≎ ≎ ÷ \$ HANGE DOT HANGE DOT 3.315E+01 3.315E+01 2.081E-04 3.315E+01 3.315E+01 0 \$ ₽ \$ 2.825E+01 2.825E+01 2.1245-04 15.1868 1.4655-04 16.6235 ≎ \$ ¢ \$ DUTPUT SUN SENSOR ISU/C.P.S. COM. 40-V .MOC 40-1 8000 40-N å ₽ \$ \$ ۵ SUN ANG. SUN ANG. \$ \$ 116.781 ₽ \$ 122.879 .0227 .4236 .9056 \$ \$ \$ \$ SUN SENSOR ISU/C.P.S. -.9056 \$ \$ LAT= ¢ \$ LAT= 7.310E+01 7.310E+01 1.286E-04 1.739178E-09 2-170693E-09 0 1.0135-04 •310F+01 .310E+01 •000E=04 6.106E+01 6.106E+01 \$ \$ ₽ ٥ ۵ ≎ \$ \$ \$ c c X-0R V-CR X-CR AZIMUTH AZIMUTH .7081 ¢ \$ 89.862 58.859 ¢ .3466 89.873 •3195 104.00 н. Г. 512. ≎ \$ \$ \$ g \$ ≑ \$ \$ •0• •01 5.5965.01 5.5965.01 5.9815-05 \$ ۵ 5.943E+01 5.943E+01 \$ ٥ 4.031E-05 <-648E-05 5-596E+01 5.5966+01 • 5650 \$ ÷ Û \$ -2445 16220 V-0R V=0R î V-DR Ŷ =9NA \$ \$ ٥ ELEVATION 4.905964E-09 123-655 ELEVATION 3.3625885-09 130.923 MZM M24 ANO α ٥ \$ \$ 3 ATT. CONT. STAR THCKR -0 -0 \* THCKR ≎ ¢ \$ ۵ \$ H N ≉ ¢ 2.2900461E+07 エエエメン 2.6135191E+07 9.2035+07 9.2035+07 1.0475+03 1.1906.08 1.1906.08 1.2206.03 1.1902+08 1.1905+05 4.117E+02 4 ¢ \$ \$ STAR 5.1037946E+04 d0-d 5.4479938E+04 ≎ • \$ ¢ d0-d • 0 • 90-9 \$ î --7302 -5161 -4477 • 4477 • ≄ ٥ **‡**` ⊅ PITCH 001 HAWGE UOT ≎ \$ ATT. CONT. \$ \$ 8.8987155401 9.673461E+01 5.0000002+01 5.000000E+01 \$ ¢ ⊅ ¢ NANGE NANGE 1.1195+08 1.1195+08 1.7025.08 1.7025.08 ٩ \$ 2.0855+02 ■ 702E+08 1.702E+08 1.5105.02 \$ ٥ 1.7405402 +619.-00800--.5549 -.5549 (200-0 ч П – п Р – С Р 0.0053 ¢ \$ ¢ \* RANGE RANGE 9-0A ¥≙₩ n p=CR FIGURE \$ \$ \$ ٩ VEL ≎ ¢ \$ ≄ LDSTONE 3.2665J19E+11 MEASURENT ERRORS 5 MERTIAL SYSTEM 9.0 2-80601286411 Σ 0 ₽ <u>Σ</u> \* \* \$ 0715 3.754925665412 --7012 3.26650185+11 \$601° -.7012 \$ \$ \$ \$ I. S. U. 4.7775\*08 4.7775\*08 8.3205\*01 ົ ц С 4.7775+08 E 80+344200 4+2785+08 80 + 401 6.4295401 ROL L 6.7795401 ≎ 4 ≎ φ MEASUREMENT ERROKS TNERTIAL SYSTEM RANGE RANGE ٥ ₽ с0-d ad = d P-Da 007 0 с 1 \$ ç \$ î ≎ ÷ \$ ţ -4.27 12 H H H X X N DSIF \$ \$ ٩ ¢ 91SC ₽ \* \$ a SEC. COMPUTER COMPUTER ů COMP. DEV. DEVIATIONS ERRORS \$ ņ \$ COMP. DEV. DEVIATIONS \$ UEVIATIONS ERRORS ATTITUDE COMP. DEV. GOLDSTONE 3NJ770109 KALMAN UPDATE ω ທ KALMAN UPDATE 4 \$ \$ 11233980 ALT= NOLIVIS \$ AL T= STATTON 0 RP085 c 1 ≉ ₽ r; ≎ 864000**.00** LOU O NOORO LOY! P64000.00 0H33M 0 005 \*\* \*\* \$ ≄ \$ 120973386 φ φ : SUBSYSTEM SUUSYSTEM RELICCENT. HELJOCENT. HFL TUCENT. STATUS STATUS HELIOCENT AFTER AFTER 1300 COAST FOR CCAST FOR 11 |---.∺ }--{ A 7 ş. - 5

A-49

ERROR ANALYSIS MARS MISSION, A-9t.

A-0P 2.854E-01 7.854E-01 7.854E-01		Ð		A=0P 7.854E=01 7.854E=01 7.854E=01	\$ \$ \$ \$ \$	¢ ¢ ¢ ¢		A=0P 7.854F=01 7.854F=01 7.854E=01		0		A=00 7.85401 7.8540101 7.8540101 7.8540101
А-СК 7.8546-01 7.8546-01 7.8546-01 7.8546-01	P11CH 9562 9562 82234	R XMITTER 0		A=CR 7.8548-01 7.8548-01 7.8548-01	• • • • • • • • •	* * * *	5812	A-CR 7.8546-01 7.8546-01 7.8546-01	PITCH 9424 1.2654	R XMITTER 0		A-CR 7.854E-01 7.854E-01 7.654E-01
A=DR 7.854E-01 7.854E-01 7.854E-01	LL YAW 1414 .2243 3193 -9323 9371 .2838	R. APP. RAD.	N N 00	А-DR 7.8546-01 7.8546-01 7.8546-01 7.8546-01	¢ ¢ ¢ ¢	* * * *	LONG= -81,	A-DR 7.854E-01 7.854E-01 7.854E-01	LL YAW 1088 .3163 3318 .9055 9371 .2838	R. АРР. RAD, 0	000	A.DR 7.854E-01 7.854E-01 7.854E-01 7.854E-01
V = 0P 3.813€ + 01 2.813€ + 01 1.469€ = 04	27 DR 236 CR 02	S. COM. RCV 1	N ANG. • 990 RANGE DAT 2.772365E 4.100006	V-0P 3. R136+01 3. R136+01 1. SR26+01	\$ \$ \$	\$ \$ \$	18.1652	V=0P 4+295E=01 4+2955E+01 1+553E=04	27 DR 36 CR	S. COM. RCV 1	N ANG. •453 •453 •453 •4506 2.5526606 4.1000006	<pre>&lt; &lt; &lt; OP</pre>
V-CR 8.5576+01 8.5576+01 1.1636+04	CH 0P •5321 •02 •7618 •42 •3696 ••90	OR ISU/C.P. N	ELE. SU 6.832 110 ZIMUTH 238743E-09	V=CR 8.5577+01 8.5577+01 1.0887-04	¢ ¢ ¢ ¢	\$ \$ \$ \$	9.883 ⊾AT=	V-CR 9.739E+01 9.739E+01 1.203E-04	CR 0P •4450 •02 •8069 •42 •3885 •90	OR ISU/C.P.	ELE• SU 4.400 105 4.1MUTH 123980F-09	V-CR 9.7396+01 9.7396+01 1.1796+0¢
V+DR 4.0285*01 5.0285*01 9.7325-05	UR • \$\$46\$ • \$902	CKR SUN SENS +0	AZM 16.446 5 100 4 100 4 100 10	V • ОК 5 • 0206 • 01 4 • 0266 • 01 8 • 291 • 05	≎ ¢ ¢	⇒ ⇒ · ⇒ ·	ANG= 8	V-DR 4.2525401 4.25525401 1.13955401	UR • 8053 • 4118	CKR SUN SENS	ALM ALM 242 242 242 101 101 101 101 101 101 101 101 101 10	<ul> <li>V = 0R</li> <li>4 • NSSRE + 01</li> </ul>
0-00 1.4955408 1.49556408 9.24455408	РІТСН 7302 XI 7302 XI 7302 XI 7302 XI	JT. STAR THO -0	E DOT 5006E+04 1 5006E+04 1 ELEVAT 02 2-1995 01 -0.	□	¢ ¢ ¢ ¢ *		0439392E+07	P-OP 1.3396.08 1.8396.08 7.3206.02	PITCH -7302 XI -5161 YI	UT. STAR THO	2 UOT 32255404 ] 22255404 ] 52255404 ] 5255404 ] 52555404 ] 52555404 ] 52555404 ] 52555404 ]	1.83398.02 1.83398.08 1.83398.08 6.1098.02
P+C9 2.3829+08 2.3828+08 2.3828+08 1.03%6+02	5 - 4AW 4 - 6794 5569 2 - 5569	ATT. CON	*11 5.7356 *11 5.7355 RANGT 1.0538834 5.00000E	5.3827.08 2.3827.08 2.5827.09 1.6746.08	* * * *	0M ∩•005) + + + +	1 VEL= 2.	D-CR B.167E.08 3.167E.08 2.157€.08 2.424€.08	5 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1 1	ATT. CON -0	+11 5.9628 RANGE 7.509201E. 5.09000E.	3.1677.00 3.1677.00 3.1677.00 2.1677.00 2.1677.00 2.1677.00 2.1677.00 2.1677.00
6+03 5.2515+08 5.2515+08 1.2525-02	ROLL XI •071 YI •709 ZI •709	I. 5. U.	RANGE 3.7550285 NT EREARS AL SYSTEM SIF	P-02 5.2017403 5.2017403 9.7037403	¢ ≎ ≎ ≎ ≎	+ + + + + + + + + + + + + + + + + + +	• 2663300E+1	5.6337 5.6337 5.6337 5.6337 5.337 5.0337 5.03 5.03 5.03 5.03 5.03 5.03 5.03 5.03	ROLL XI • 071 YI • 709 ZI • 709	I. 5. U.	RANGE 4.26636916 NT ERROKS AL System SIF	8+03 8+03 6+03 6+03 6+03 6+03 4+03 1+03 1+03 1+03 1+03 1+03 1+03 1+03 1
COWP. DEV. UEVIATIONS ERRORS	ATTTUDE	SUBSYSTEM COMPUTER STATUS -0	STATTON STATTONE GOLDSTONE REASURSME TNERTI TNERTI O AFTER KALMAN UPDATE O	COMP. DEV. DEVIATIONS Errors	HellIOCENT。 * * * *	COAST FOR 854000.00 SEC. HFLIOCENT. * * * * *	AT 1560 0H33M ŕ00S T= 12961980 ALT= 4	COMP. DEV. DEVIATIONS Errors	AITIUDE	SUBSYSTEM COMPUTER STATUS =0	STATTON STATTON GOLUSTONE MEASUREME TNEASUREME TNEASURAN D	COMP. DEV. COMP. DEV. ERRORS

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FIGURE A-9u. MARS MISSION, ERROR ANALYSIS OUTPUT

ź .854E-01 854E-01 ¢ 854E-0 • 854E-0 .854E-0 .854F-0 ψ 4-0P 0-0-V 54 8545 ΰ ¢ ŵ ó ψ \$ ٥ XMITTER XNITTER ¢ ŵ 7.854E-01 7.854E-01 101 •854E=01 7.854E-01 .4436 7.854E-01 .8728 .9118 .3567 --2034 .8545-0 •85¢E=0 --2034 ¢ PI1CH ₽ PI1CH 0  $\circ$ A-CR \$ \$ -71,8044 -76.8084 \$ \$ 4 \* RCVR. APP. RADAR έ RADAR ¢ \$ ⊅ - 4862 - 8265 •4036 •283A .2833 -. 8698 ¢ \$ ¢ \$ 7.854E-01 7.854E-01 7.854E-01 7.854E-01 YAW ¥ΔY 101 7.854E-01 7.854E-01 7.854E-01 ¢ \$ \$ LONG= A=DR RCVR. APP. \$ A-DR \$ 0 ¢ • 85¢ LONG= \$ ⇒ ¢ ò .3466 6075. .9371 .0423 . 0757 1759. \$ ó ¢ ð 4.10000E-02 ROLL ROLL \$ ¢ ≎ ٥ 001 RANGE DOT 2.639520F ò 4.811E+01 4.811E+01 ۵ 21.4418 1. n18E-04 ¢ ¢ 9-591E-05 19.8097 1.0145-04 5.4625+01 4.A11E+01 4.AllE401 5•462E+01 ¢ \$ RANGE \$ E C C C SUN SENSUR ISU/C.P.S. COM. a a SUN SENSOR ISU/C.P.S. COM. 40-A d0-1 V-0P aO \$ \$ ۵ ¢ SUN ANG. SUN ANG. იემ აბე 100.120 ¢ \$ \$ ۵ • 0227 • 0227 • 4236 • 9056 \$ -.9056 ¢ ¢ ø \$ LAT= ≄ ٥ LATE 1.0795+02 1.0795+02 1.2505-04 1-1505+02 1.150F+02 0 0 ٥ 0 1.304391E-09 -079E+02 •079E+02 1.0995-04 1.181E-04 \$ \$ \$ ٥ \* \$ 40-7 C Ŷ 20=2 4-CR ALIMUTH AZINUTH 006\*68 •2624 29140 48°403 •3548 ₽ ÷ .8715 89.892 51.578 \$ • 8434 • 4034 ELE. ELE. ₽ ٦ \$ ٥ g g \$ \$ ۵ •••• ≎ ۰ •912E+01 ۵ 3.2245401 1.6816-04 .240E-04 .012E+01 245+01 .2245+01 ,1311 -- \$5647 ¢3305 ٠ ÷ .00° ÷ 14860 ---V-DR î V-0R î <u>v = 0</u>R 7 ۵ 97+163 \$ ANG= 1.172889E-09 ANG= 103.093 M24 чO с С AZM 3.0 ٥ \$ \$ ELEVATION ELEVATION THCKR TROKE \$ ≄ \$ ≉ ¢ 17 \$ 3.6071473E+07 ž 2.681E+08 7.712E+02 t 3.27655365+07 2.2495+08 1.401E+02 X Υ 2.249E\*08 5.941E\*02 1 2.681E+08 5 2.2495408 2.2495+08 ≎ \* ¢ 4 STAR ATT. CONT. STAR d0-d •••• ≎ \$ 01 6.2300616E\*04 6.1279121E+04 ¢ \$ d0-d 0 d0-d --7302 -5161 -4477 +4477 \$ \$ PITCH \$ ¢ PITCH RANGE UOT RANGE UOT ATT CONT. S ¢ ۵. \$ \$ 6.5262092+01 5.0000002+01 \$ ¢ \$ ¢ 5.0256+08 5.0255+08 4.000000000 3.2010402 \$ 3.8382.02 4.0692.05 4.0655.08 ¢ ٥ \$ 4.0665.003 2,4905,02 --4R00 6759.-+6290-00800---5549 +010--0.005) \$ ٥ 0-005) ¢ ≎ VEL= RANGS RANGE 5-CU p=CR XXX MVX VEL= 20-0 î \$ ≑ ≎ \$ \$ \$ ¢ ¢ 5.3366666665 4.79532926411 ⊅ δ ¢ δ \$ ·0715 \$601. 2170. -17012 \$501° -•7012 5.3363737E+11 4 • 79528395 • 11 P=04 5• U. \$ \$ ¢ φ 5.9545\*08 5.9545\*08 2.2015\*02 6.198c\*08 3.546c\*02 I 0 r 0 6.198F\*n8 201 ≉ MEASUPEMENT ERRORS \$ ERROYS TNERTIAL SYSTEM RANGE :: Φ RANGE 00-0 9-05 ¢ 0 1 101 (100 ۰C ۲ ÷ \$ ۵ • •--٥ \$ ٢ ÷ X 14 17 031F \$ ٥ × > \$ ź. MENT \$ ≉ ¢ \$ sec. COMPUTER COMPUTER COMP. DEV. \$ SEC ¢ \$ \$ DEVIATIONS COMP. DEV. DEVIATIONS ERPORS ATTTUDE COMP, DEV. ATTITUDE GOLDSTONE 340494109 MEASURE KALMAN UPDATE \$ \$ 14689980 ALT= 13325980 ALT= ¢ ٥ NOITATO STATION ERRORS Ŷ ERRORS 01 \$ 864000.00 \$ ţ, 364000.00 AT 1700 0H33M 0.005 ٥ AT 140D 0H33M 0.005 ÷ ÷ ÷ ¢ ≎ ≎ ٥ ¢ SUBSYSTEM SUBSYSTEM HEL TOCENT. HELIOCENT. HELIOCENT. STATUS STATUS HFL IOCENT AFTER COAST FOR COAST FOR н 1-11

FIGURE A-9v. MARS MISSION, ERROR ANALYSIS OUTPUT

٩ ٩ 7.854E-01 7.854E-01 7.854E-01 7.854E-01 7.854E-01 7 854E-01 7.854E-0] .854E-0 .854E-0 ≉ ¢ \$ \$ ٥ 40-A A-OP 8545 ٠ ٥ A-OD 00-V \$ ¢ \$ ¢ ¢ ₽ \$ \$ ٥ XMITTER 7.854E-01 7.854E-01 7.854E-01 7.854E-01 7.854E-01 7.854E-01 \$ ¢ ¢ 7.85¢E+01 .8545-01 101 . 8545-01 7-854E-01 7.8545-01 • 8275 • 5232 •7735 •6002 7.8546-0 -,2036 PITCH φ \$ ¢ \$ PITCH A-CR A-CR A-CR 0 A-CR •854E \$ \$ ŧ \$ -66,8078 -61.2921 \$ \$ ۵ ¢ ė ≉ ۵ RCVR. APP. HADAR ¢ .5614 -.7774 .2837 •6334 \$ \$ ¢ \$ 7.854E-01 7.854E-01 7.854E-01 7.854E-01 7.854E-01 7.854E-01 7.854E-01 7.854E-01 7.854E+01 7.854E-01 YAW 7.854E-01 YAW .854E-01 7.854E-01 \$ ¢ ٥ ۵ A-DR \$ \$ \$ A-DR \$ A-UR o A-DR -ong= LONG= ≉ \$ ¢ \$ • 0038 .3490 •3483 --0235 ⊅ \$ 6.626939E-02 4.100000E-02 \$ ٦ 2.844509E-02 4.100000E-02 ROLL ROLL 4 \$ ¢ ⊅ RANGE DOT 6.626939E-6.0695+01 6.0695+01 6.4668.401 6.4668.401 2.3038.404 22.9513 V-0P 24.4379 \$ •. 769E+01 \$ ≎ LC. \$ 1.953E-04 1-500E-04 5.4625+01 • 462E+0 • 963E=0 \$ \$ ≎ \$ 800 SUN SENSOR ISU/C.P.S. COM. 8 8 8 8 do-n 40=A 40-V \$ \$ \$ \$ SUN ANG. ۵ 89.783 \$ \$ ⊅ • 0229 • 4235 • • • 2331 • • • 236 ະກ 0 ¢ \$ s S ٢ \$ ¢ \$ LAT= ¢ \$ LAT= 3.5076+01 3.5076+01 2.4976+04 4.708E+01 4.708F+01 1.150E+02 1.150E+02 4.7085+01 4.7085+01 2.415F-04 c O 9.629840E-10 <u>a</u>0 1.472058E-09 •023E-04 7.061E-05 \$ ۵ ٥ \$ V=CR V-CR \$ \$ \$ α \* c 20-2 AZIMUTH 1711 - 8908 - 4210 45.067 •0758 •9025 \$ \$ \$ 89.914 ¢ 106.48 • ພາງ ພ ₽ \$ ¢ \$ č g \$ \$ \$ ٥ •••• 01 9.7836+00 9.7836+00 2.0255+04 1.9125+01 1.9125+01 1.6255-04 4.1045401 4.1045401 2.2155401 9.783E+00 9.783E+00 \$ \$ ≎ ۵ 1.785E-04 • 1645 • 0521 •0769 •0744 \$ ۰. پ .\$850 ۵ ۵ V=DR V-DR ? V-0R V-0R ELEVATION 2.3137345-09 -0. ANGH ٥ \$ 9.2872835-10 ¢ \$ 065.16 ANGU сu AZM с С ۵ ¢ \$ \$ STAR THCKR \$ ۵ ¢ \$ \$ : ۵ \$ \$ 4.2370763E+07 6。1335407 6。1335407 5。4635408 нн X ≻ X 17 3.9169204E+07 1.774E+08 1.774E+08 <.681E+08 7.656E+02 1. 774E+08 1. 774E+08 1.317E+03 2.681E+08 559E+02 ¢ ' ¢ \$ \$ d0-d 0-≎ d0-d \$ ٠ ٥ d0-d 01 6.2174187E+04 60-d 1915--5161 .4477 -.7302 \$. \$ \$ \$ PITCH PITCH 001 ¢ ≎ \$ ≎ ATT. CONT. 1.888839E+03 5.000000E+01 -000000E+01 7.0799935+01 \$ \$ \$ \$ RANGE 1.2149.03 1.2149.03 5.0309.02 5,0257+08 5,0252+08 3,3665202 3.3687+08 3.3687+08 8.6657+08 .364E+U2 .768E.08 - 368E + 03 \$ ٢ \$ ≄ .1735+02 +510.-0.005) 0.00S) \$ \$ r. \$ RANGE VEL= 6 YAW WAY 9-CR VEL= p-CR n-C2 9-0F ⊅ \$ ¢ ٥ ¢ \$ ₽ ≉ LDSTONE 5.85775182011 MEASUREMENT ERROMS 1 INERTIAL SYSTÉM 1.8 X O 5 \$ ≎ mm \$ 20 ¢ 2102.-•7094 •7094 6.41196556+11 5.8576907E+11 •0715 ≉ ≑ \$ ۶ ະບ ເ 6 • 103 = 408 6 • 193 = 408 2 • 99 = 408 2 • 99 = 408 4.050F\*08 4.050F\*09 1.799F\*03 1。45777408 1。4577408 4.8777408 PULL PULL r. o RULL I O 4.050509 4.0500008 20\*2651\*1 \$ ۵ RANGE ٢ \$ SYSTEM c0-d 101 100 C F ¢ ٢ aŭ-d \$ ۰ P-02 P-0a • φ ۵ \$ ⊅ 저 거 ~ -\$ ٥ DSIF ٥ \$ 7120 TNERTIAL ¢ ٥ \$ ¢ SEC. SPIC. COMPUTER COMP. DEV. UEVIATIONS ERRORS COMP. DEV. DEVIATIONS ERRODE \$ COMP. DEV. DEVIATIONS \$ ÷ COMP. DEV. DEVIATIONS ATTITUDE \$ ATTITUDE SNOTSCIONE UPDATE KALMAN UPDATE ALT= ≎ ٥ 15553980 ALT= ÷ ٩ STATION ERRORS 0 1 ERRORS 4 864000.00 \$ ≎ 864000.00 \$ 500°0 00°0 \* ≄ ≎ ≎ \$ KALMAN ₽ ≎ \$ 16417980 SUBSYSTEM WEEHO QUAL 100D 0H33M HFL IOCENT. HELIOCENT. HFLIDCENT. PRUIOCENT. STATUS AFTER AFTCR а Сы а Сь 11 }--19 1--15400 COAST 1-1-4-7

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NULUSIW SAW WERE WISCLE

OUT

3XT YNY

AC - CE

ŵ 0. ð .854E-0 ٩ 01 ⊅ 8545-0 \* 85¢F d0⊷A 854E. . 854E. \$ ≉ c ,854E φ 854.9 A-OP DO D 854E ⊅ \$ \$ \$ ۵ ≎ \$ \$ \$ ø ¢ XMITTER XMITTER ÷ ₽ ¢ \$ .854E-01 7.8545-01 7.8546-01 .8545-0. 01 -.2400 .7544 .6110 \$ \$ \$ \$ PI1CH A-CR A-CR 0 C <u>ل</u>د خ 7.854E 7.8542 ۵ 4 \$ \$ \$ -57.0149 **\*495**267 ŧ ۵ \$ ¢ \$ \$ ⊅ ¢ RCVR. APP. HADAR RCVR. APP. RADAR .3851 .2835 \$ ٥ \$ \$ 7.854E=01 7.854E=01 YAW 7.854E-01 .854E-01 ..425E-01 •854E-01 .8545-01 \$ ¢ \$ \$ ٥ 4 \$ A-UR ۵ មុំ ភូមិ A-DR 0 A-0R -9NO-LON6= ≎ ٩ \$ \$ \$6253 •3613 •6917 æ. .9371 ⊅ \$ ¢ \$ 4.906602E-02 4.100000E-02 POLL \$ ≎ \$ \$ RANGE DOT 25.3802 V=0P 1 • 048E + 02 1 • 048E + 02 25.3930 V=0P ÷ ¢ • 6485+02 \$ ¢ 2.813E-04 · · · 48E+05 \*<28E-04 2.4135-04 6.466E+0] ≎ \$ ≎ \$ • wo0 · MOU V-0P å ¢ \$ \$ ٥ SUN ANG. ≉ ≎ \$ 85.025 \$ -.6584 .6212 -.4251 SUN SENSUR ISU/C.P.S. \$ ¢ SUN SENSOR ISU/C.P.S. \$ ٥ ----٩ \$ \$ Ð ¢ LAT 1.259E+02 1.259E+02 3.355E+04 1.259E+02 1.259E+02 1.552E-04 LAT 6-964174E-10 d O 3+3555-04 3.507E+01 3.507E+01 \$ ⊅ ۵ \$ \$ 20-7 \$ V-CR \$ \$ c c V-CR 616**.**64 20-V AZIMUTH \$ \$ ⊅ 616.28 \$ • 4240 41.272 •2044 -.8952 -.3961 : 13 3 3 3 ₽ ≎ \$ \$ č \$ ¢ ٥ ≎ 8.9135+01 8.9135+01 \$0**1** ≎ \$ \$ ψ 2.3096-04 8.9135+01 2.309E-04 0+3213e+01 4.104E+01 -+ •0113 ふかごと。 .6762 \$ ٤ \$ ۵ •1338 X-0R 2 01 1.6140 V-0R V-0R ANGS ¢ ¥ ANGa ۵ ELEVATION 1.1809895-09 \$ 86.957 AZM Ч ٥ \* \$ \$ 4 e 1 THCKR \$ \$ THCKH \$ ⊅ PRPGTE, \$ ٢ \$ 4.4372603E+UT 17 ٥ 4.4366198E+07 エエエスメイン 6.665E+03 4.765E+02 6.6285+03 6.623E+03 6.6455+03 4.765E\*02 4.680E+02 6,1335+07 ۵ ÷ ¢ \$ ¢ STAR SIAR \$î d0-d ۵ ≎ \$ 6.25656/3E+04 d0-a d0-d c t 40-4 0 1915. 1915. • 4477 11.04. \$ 0 Z \$ \$ \$ 0 -PITCH 001 ≎ \$ \$ \$ ATT. CONT. ATT. CONT. 1.27938555402 5.000000E+01 \$ ≎ \$ \$ RANGE 5.0%20+03 5.10%7\*03 4.5%57\*02 \$ 1.2145.08 .2145.08 ţ ¢ \* 5.0835+03 5.1045+03 4.6855+02 .0855+02 0+005) 0.004) ⊅ \* ⊅ ≎ RANGE vel. Prox MVL p-CR 01 01 VEL= P-CR 4 t \$ ٦ ¢ \$ ₽ ¢ 6.4120367E+11 ¢ ₽ ΰ 5023H27M -7 моено •0715 •7094 6.76406605.411 --7012 6.7629291E\*11 P-D9 -.7012 5. U. 17 \$ \$ \* 5 1。1635\*04 1。1685\*04 4。5345\*02 1。1635\*04 1。1645\*04 2012 1-4525+08 • 4525+08 3.2655\*02 4.5345002 φ RANGE MEASUREMENT ERRORS INERTIAL SYSTEM ≎ \$ \$ F-102 c ŧ P\_79 \$ ¢ 2 2 1 \$ 60 \$ ç • •--1 • \$ :7 ₽ ≎ 12 1 H H Z \$ USIF 4 \$ ~ \$ ¢ \$ ¢ ¢ SEC. COMPUTER COMPUTER SEC. COMP. DEV. DEVIATIONS ERROAS APT. CONT. TURNED ON APP. RADAR TURNED ON ≄ \$ \$ ٥ DEVIATIONS ATTITUDE 20 20 NO OBNALL TURNED ON COMP. UEV. COMP. DEV. DEVIATIONS COLDSTONE KALMAN UPDATE \$ \$ \$ R ¢ 16934400 ALT= STATION TURNED DINANTO AL T: ERRORS o t 0 F ERRORS ÷ ≎ ¢ ¢ 516420.00 500000 1800.00 1960 CHOOM C.005 :: \$ ٥ \$ ٥ ÷ ₽ ٩ 662-26 SUBSYSTEM SUBSYSTEM δ с**т**. STAR TROKE AORAND MIL C1. 8 HELIOCENT. STATUS STATUS 5 COMPUTER ຳລ • s • . ro 100 13997 138651 TARGET AFTER しょうし C0357 F09 COAST FOR 11 1-,t: ,---Š A 7

### A-53

ERROR ANALYS IS OUTPUT MARS MISSION, FIGURE A-9x.

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include the use of the approach radar in the measurement sequence and zero the computed deviations at periapsis of the swingby trajectory. The first approach radar measurement occurs in Figure A-9aa. At this time the inertial system errors, after updating with the DSIF, show a predicted range error of 310 feet and range rate error of approximately .0001 feet per second. The onboard approach radar errors are 300 feet and 40 feet per second respectively. Thus the measurement of range to an accuracy of 300 feet, comparable to the inertial system accuracy, should produce some improvement in overall system estimate errors. However, it is doubtful that any benefit is obtained by the measurement of range rate with an uncertainty of 40 feet per second. The midcourse correction is the first approach correction subschedule operation which occurs at the top of Figure A-9bb, with an expected value of 186 feet per second. This correction has a large expected value because it must remove the velocity deviations imparted by earlier midcourse corrections to bring the spacecraft back to the nominal trajectory at the Mars sphere of influence. A second approach update is performed approximately 198 days into the mission to remove errors generated in performing the large first approach update correction sequence. This midcourse correction, printed in the center of Figure A-9ff has an expected value of 12 feet per second.

The nominal trajectory periapsis is indicated in Figure A-9gg. This is the point in the mission for orbiter and landers where a retroburn would be performed to obtain an elliptical orbit about Mars. The cross range position error at the nominal periapsis is  $1.6 \times 10^5$  feet. This is the error in the periapsis magnitude. The down range and out of plane components represent errors in the periapsis position vector. They are approximately  $3 \times 10^6$  feet. At this time the nominal retro  $\Delta V$  is added. This burn may be altered in direction and magnitude to remove errors in velocity at the nominal periapsis point. An impulsive burn is used in this analysis. However, for the Mars capture phase of the Viking mission, a forty-five minute retro burn has been proposed. To study the effect of varying burn time, a short error analysis program was written and the results are presented in Figure A-10. The errors shown are those that would exist 45 minutes after retro ignition for a burn time varying from 3 to 45 minutes, with a  $\Delta V$  of 3800 ft/sec. The error sources considered are:

- (1) Initial Conditions
  - (a) Position
  - (b) Velocity
  - (c) Attitude
- (2) Accelerometers
  - (a) Bias, K
  - (b) Scale factor, K
  - (c) Nonlinearity,  $K_2^{\dagger}$
  - (d) Cross axis sensítivity, M
- (3) Gyroscopes
  - (a) Fixed drift, D<sub>FR</sub>
  - (b) G sensitive terms, D<sub>UI</sub>,D<sub>US</sub>

1.475E-01 7.854E-01 7.854E-01 .854E-0 8546-0 C AC.A 3995 XMITTER XMITTER 7.854E-01 7.854E-01 7.4065-02 1.425E-01 7.854E-01 7.854E-01 -.1275 .5800 -.2400 PITCH -.7544 .6110 0 A-CR PITCH 0 -57.2645 RCVR. APP. RADAR .2070 RCVR. APP. RADAR .2970 .1843. -.7425 .3851 1.371E-01 7.854E-01 7.854E-01 ۸AY 7.854E-01 7.854E-01 XΔW \$ 1.425E-01 ۵ A-0R ¢ ò LONG= ٥ .5143 .3388 \$ 7879 .3613 .6253 \$ .6917 \$ 60 SEC • PULL ¢ ROLL \$ \$ 1.048E+02 1.048E+02 2.813E+04 25.3831 V-0P \$ \$ 1.0485+02 1.0485+02 2.81355-04 ٥ 00.06 00.02 \$ \$ • WOU \$ 6 C G SUN SENSOR ISU/C.P.S. COM. ≎ 2 2 A A \$ ≄ \$ \$ \$ SUN SENSUR ISU/C.P.S. \$ .6212 ۵ -.6584 -. 4251 .6212 ŧ AIDS= -.4251 -.6584 AIDS= LAT= ۰ Z \$ 1.259F+02 1.259F+02 \$ 3.355E-04 ð 1.259E+02 3.355E-04 \$ d O 1.2595+02 \$ \$ -49.680DEG. \$ 102.63, BETWEEN OPTICAL \$ 2-02 \$ 89.919 UPTICAL ⊅ \$ •2044 -.3961 -.8952 \$ -.3961 •2044 2508.-\$ \$ \$ č ¢ č \$ ₽ 8.8136.01 8.9136.01 7.3096.04 \$ 8.9136+01 8.9136+01 7.2096-04 ٥ 102.63. BETWEEN • 7244 • 6762 \$ -.133d ¢ ⊅ ,725. 4057. -.1334 V-0R ROLL \$ ANG \$ \$ а \$ 80 \$ \$ TROKE ٥ \$ \$ TXOXX PRPUTE ٥ NO PRPGTE. -5.875DEG. SUN-CANOPUS \$ ٥ 4.4372817E+07 6.6232+03 6.645E+03 ×× 4.7655+02 . 1 6.645E\*03 4.765E+02 17 ₽ × 6.6285+03 ٥ ۵ ≎ ATT. CONT. STAR STAR \$ ≎ d0-d -.1333 1.920.-22 ≎ -5161 -4477 SUJN-CANOPUS --7302 ¢ ≎ PITCH PITCH BETWEEN CELESTIAL RUDIES= ≎ \$ ANGLE BETWEEN CELESTIAL RUDIES-\$ ATT. CONT. ٥ ¢ 4 YAW FUR \$ 5.1045+03 4.4855+02 5.1042403 5.4452403 4.4452402 \$ 5.0825403 \$ •1110 ••6080 1901 --5.0835403 0,005) ≉ 64.00 4455.-(500+0 3010---.4800 \$ NAX. ¢ YAW 64.00 VEL# 75.54 p-0R 8±00 SEARCH ≎ WAY MAX. ≎ ₽ TO LOCATE 0.186DEG. \$ ¥ ٩ Σ. ۵ \$ 0H 1M ≎ 02010 --0769 -- 7073 6.76410255411 \* 00°0¢ HC 0115 -- 7012 \$602\* 20.00 ≉ s. ∪. s. U. **8,00** 20.00 1 • 1635 \* 04 1 • 1645 \* 04 4 • 5345 \* 02 MANEUVERING, BEGIN \* 201L Set 1 1 - 166#\*0% \$ 536#\*02 ≑ ასის \$ 4042091.1 \$ S ¢ ¢ au+d S \$ ~ BEGIN MANEUVERING \$ • ¢ ROLL DIAD RANUS ¢ 17 ž YAW BANDS ROLL EOLId \$ PITCH エアン ы Х \$ \$ -≄ ₽ 60.00 SEC. t: COMPUTER ANGLE SEC. \$ DEVIATIONS ERRORS COMPUTER ¢ DEVIATIONS ERRORS COMP. DEV. ATTITUDE \* ATTITUDE DEAU COMP. DEV. MANEUVERS ÷ 16936260 ALT= \$ \$ t, ¢ 0H31M 0.005 60.00 ≎ ٥ ¢ CN1 4 z, \$ \$ SUBSYSTEM C1° SUBSYSTEM STATUS TARGET CT. STATUS 5 TAUGET TADGET 1960 COAST FOR COAST FOR 11 |---7 T A-55

DUTPUT

MARS MISSION, ERROR ANALYSIS

A-9y.

FIGURE

÷ \$ \* 1.399E=01 4.311E=04 7.125E-05 4.377E-04 .399E-01 ₽ \$ ٥ ¢ 0 A--OP A-OP ¢ • \$ \$ \$ ÷ \$ \$ ¢ \$ \$ XNITTER i 0.0000.0 0.0000.0 7.125E-05 6.026E-04 7.406E-02 5.979E-04 ź \$ 7-4065-02 ¢ .5800 -1275 \$ PITCH \$ ¢ A-CR 0 A-CR \$ \$ \$ -57,5141 -57.7637 \$ \$ \$ THCKR SUN SENSOR ISU/C.P.S. COM. RCVR. APP. RADAR •PITCH +PITCH \$ \$ ¢ \$ \$ 1.371E=01 1.371E=01 2.917E=04 7.12555-05 3.0135-04 YAW \$ \$ \$ A-DR A-DR \$ \$ \$ =9N01 LONG= • ≉ \$ .5143 .7879 \$ ¢ \$ 60SEC. 60SEC. POLL 00000000 • 02586 \$ \$ ¢ 1.048E+02 1.048E+02 2.813E-04 25•3832 V=0P 25+3933 V=0P ≉ ÷ ≉ 00.06 \$ \$ \$ MARS MISSION, FRROR ANALYSIS, OUTPUT a a d \$ \$ ¢ # ¢ \$ -4251 •YAW \$ ¢ ' • YAW \$ -.6584 102.63. BETWEEN OPTICAL AIDS= \$ LATE Z 89.919 LAT= \$ \$ Z 1.259E+02 1.259E+02 3.355E-04 ð 1.259F+02 1.259E+02 ¢ \$ \$ -0000EG. -•000DEG+ \$ \$ 1.00000 PASSES 40-7 2-CR PASSES \$ 610.98 ¢ •2044 •3961 \$ \$ --8952 .02586 • 00000 \$ \$ \$ g \$ \$ \$ 1.00000 8.9135.01 8.0135.01 8.0135.01 2.4095.01 V-0R R.913E+01 8.913E+01 \$ ٥ \$ •7244 •6762 \$ ≎ **••1**338 \$ ROLL ROLL \$ ANGH ٩ \$ ANGa g ¢ \$ \$ (RAD/SEC)=ROLL RAIES (RAU/SEC)=HOLL \$ \$ \$ PRPGTE. 29.6 SECONDS WITH RAIES (RAU/SEC)=RC --0000EG. 30.4 SECONDS WITH \$` 17 \$ \$ 4.4373244E+07 P-0P 4.43730305+07 6.645E+03 4.765E+02 T X X 6.6285+03 0.045E+03 6.6285+03 **\*** . ≉ ¢ ¢ ATT. CONT. STAR ₽ \$ d0**-**d ••1333 **••**9008 \$ 9 Z ۰ \$. PITCH ٥ 05G. \$ ¢ DEG. BETWEEN CELESTIAL BODIES= .01000 DEG. FIGURE A-9z. \$ \$ ⊅ YAW YAW 5.08%5.03 5.1047.03 4.4857.02 5.0825\*03 5.104F+03 \$ ¢ \$ .01000 .01000 1401 --0.005) ¢ ¢ \$ VELa 23.37 v5L= p=CR YAW 64 • 90 00.0 MAX. 75.65 20-0 00.8 \* \* \$ ≎ ANGULAR ANGULAR •000DEG• .000DFG. \$ \$ END SEARCH FOR SUN-CANOPUS 0H IM \$ \$ \$ \$ 21 Z 6.7641403E+11 --0769 0207° -.7071 6.76417R2E+11 \$ s. U. 8.00 20.00 \$ •01 1.1637+04 1.1665\*04 4.5345\*02 ].1634\*04 1.1665\*04 •01 .0. RULL Set. \$ Ŷ 200 \$ ŧ 00 an-q CC-d CANOPUS \$ \$ CHANGED CHANGED CHANGED • \$ \$ \$ PITCH PITCH MAY PITCH I X X ROLL 717 BANDS PITCH SUN \$ ¢ ~ ٥ \$ \$ \* SEC. COMPUTER FOR FOR \* \$ \$ ANGLE 07V. DEVIATIONS ERRORS DEVIATIONS COMP. DEV. ATTITUDE DEAD N m MANEUVERS MANEUVERS ∧LT= \$ \$ \$ 16936320 ALT= ON AXTS ON AXTS SEARCH 60.00 ¢ AXIS SEARCH ¢ ₽ 1950 0H32M 0.005 AT 1960 0H33M 0.005 CONP. \$ \$ ACQUIRED \$ 16926380 \$ \$ \$ zo SUBSYSTEM . TARGET CT. STATUS TARGET CT. 5 **DFADHAND** DEADHAND DEAUNAND TARGET α C L :: \_ CANOPUS COAST AT

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5.984E-04	PITCH 1.80%6 1.2275 .5800	R XMITTER 0		A - CR 7.12555 6.02651:05 5.98451:04		A-CR 7.1255-05 6.0265-04 5.9845-04	* * * * *				¢ * * *	0133 A-CR 7.12557-05 6.93557-05 5.99357-04	PITCH • 8046 • 1275 • 5800
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2+3098+04	02 • 72%4 • 72%4 • • 1338 • • 1338	R SUN SENSO	AZM E 4.054 38 0N AZ 3510 7.4	х • • • • • • • • • • • • • • • • • • •	3.104371 3.00000	<pre>∀*DR α•913E+01 8•913E+01 1.655E*01 1.655E*04</pre>	* * * * *	•	HETWEEN OP		\$ \$ \$ \$ \$ \$	ANG= 89 V=0R R=V156+01 R=V156+01 R=V156+01 1=+5586+01	UR • 7244 • 6762 • 1338
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4 • 685E • 02	YAW 9 • 1116 9 • 5986 1 • 7057	ATT. CON	RANGE *11 6.2126 RANGE 8.52235615+ 5.0000005+	9.4CR 5.096E403 5.100€E403 7.300€E403	CRS (RANGE. DRS (RANGE.	5-10-03 10-03 5-10-4-03 7-01-04 7-02 8-02 8-02 8-02 8-02 8-02 8-02 8-02 8	* * * * *	1M 0.005)	ESTIAL RUDI	MAX. 64.00 23.37 23.37 2056. YAW	<b>♀</b> <b>◇</b> <b>◇</b> <b>◇</b>	1 VEL= 4. P+CR 5.1008+03 5.1008+03 5.1008+03 2.0108+02	YAW 9 .1116 9
4.5345+02	XI = 076 XI = 076 ZI = 707	I. 5. U.	RANGE 6-766262E NT ERRORS AL SYSTEM SIF	P+Dq 1.1637+04 1.1667+04 3.0567+04	KADAR UCDAT SYSTEM ERR RADAR ERR	Р=П? 1•1665°04 1•1665°04 2•12655*08 2•12655*08	* * * * *	H0 (10 )	BETWEEN CEL	ANDS SET Rould •01 Yaw •01 ITCH •01 ITCH •01	* * * * * * *	•7642160E+1 p=00 1+1642404 1-1642404 1-1662404 2-1265404	XI -076 XI -076 702
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	SUGUNDUS.	ទ	یں بل ح	U 	AF 75		TARG	COAST FOR			TARG	AT 1965 T=	CANOPUS

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FIGURE A-9aa. MARS MISSION, ERROR ANALYSIS OUTPUT

\$ \$ \$ 7.125E-05 4.406E-04 \$ 4 3395-04 4 3315-04 7.8545+01 7.8545+01 7.8545+01 7.1255-05 4\_348E-04 7.854E-01 7.854E-01 7.854E-01 ≎ \$ ¢ ۵ A...A ¢ \$ o ₽ ٠ dů-v o 40-V A-0P ⊅ \$ \$ \$ \$ ⋬ ٠ \$ ≉ \$ ¢ ¢ XMITTER XMITTER 7.1255-05 ¢ 7.854E-01 7.854E-01 7.1256-05 \$ \$ ÷ 6.0355-04 5.993E-04 6.005E-04 7.8545-01 7.854E-01 10-3458. 1.8545-01 ••8046 .5800 -.1275 ≑ ¢ ٥ \$ PI1CH A-CR 0 A-CR A-CR 0 A-CR \$ \$ \$ \$ -58,2629 STORESSON. **\*48,3920** \$ \$ \$ \$ \$ \$ \$ \$ RCVR. APP. RADAR APP. RADAR .2322 -.2070 \$ \$ \* \$ 7.1255.05 3.0315-04 2.9465-04 7.125E-05 3.055E-04 2.971E-04 A-DR 7.854E-01 7.854E-01 7.854E-01 7.854E-01 YAW 7.054E-01 000000 ¢ ₽ ٠ 7.854E-0 \* A-UR \$ A-0R A-DR \$ ¢ \$ Long= LONG= ÷ ¢ \$ ¢ .5143 •5388 •7879 ٠ \$ \$ ¢ ROLL RCVR. • \$ ¢ \$ ۰ ý. 1.711E-04 5.0555-02 5.0546-02 1.711E-04 5.055E-02 5.054E-02 1.711E-04 5.055E-02 5.054E-02 9.8275-03 5.557E+00 5.557E+00 \$ \$ 25+3835 \$ \$ 25.6480 OUTPUT \$ \$ \$ \$ ۷**-**0P 200 COM. 40=V ISU/C.P.S. COM. 40-A 40=N ---ŧ ٥ \$ \$ ۵ ¢ ۰ \$ -.6584 .6212 SUN SENSOR ISU/C.P.S. \$ \$ \$ **‡** ] ANALYSIS -.4251 ¢ ' \$ 1.8105103 4.2047102 4.1995102 1.810F-03 4.204F-02 4.199E-02 \$ ¢ LAT= V-CR 8.1388103 7.0588400 7.0588400 LATE 1.810F-03 4.2045-02 4.199E-02 a \$ ¢ \$ \$ =1.000 V-CR V-CR \$ \$ <u>x-C</u>R ٦ \$ •2044 •3961 \$ \$ 89.919 ¢ \$ 610.68 -.8952 ERROR 200 \$ \$ ٥ \$ SENSOR č ₽ \$ \$ \$ 6.613E=04 4.572E=02 4.577E=02 4.577E=04 4.577E=02 4.577E=02 6.6136-04 4.5786-02 4.5776-02 т. е / е н = 03 7. 68,65 + 00 7. 68,65 + 00 (FT/SEC) + \$ ¢ \$ \$ .7244 \$ **\$**` ••1338 ¢ . WISSIN, V-0R V=0R V-0R Y0-7 sun ANG \$ \$ **UNG** 4 ٥ e O ≎ \$ ۵ ¢ STAR TRCKP ≎ \$ TRCKR \$ \$ PRPGTE. ٥ \$ 17 \$ \$, 1.8651E+02 4.4373671E+07 н Х 4.49805656+07 6.630E+03 6.665E+03 4.565E+02 6.4055403 5.7005404 9.7505404 6.630E+03 5.645E+03 6.630E+03 6.6%5E+03 4.565E+02 4,565E+02 MARS \$ \$ ≎ \$ · STAR \$ d0-d \$ 0-0p \$ ⊅ 00-3 d0-d --1333 --0248 -.9908 NO 4 \$ \$ \$` PITCH ₽ ¢ DFG. 056**.** 056. \$ \$ CONT. CONT. A-9bb. \$ ۵ \$ \$ 5.500500 1.2005505 1.2005505 5.1005+03 5.1005+03 2.0105+03 5.104E+03 2.010E+02 5.1008.03 5.1048.03 \$ \$ 20.00000 1 20.000000 1 20.000000 1 \$ \$ 5.100F+03 O) + RMV= .010E+U2 •1116 •6080 1901 --(\$00:0 \$ ¢ ٥ 0.005) ŵ ATT. ATT. p=CR YAW p-CR 13V 2010 VEL= 0<u>0</u>-0 ¢ \* ጶ ٥ FIGURE ≎ \$ \$ \$ \$ Σ **In**ZH25M ≎ \$ 2 \$ 1107. . . 6.7642539E+11 6.8719560E+11 \$ ≎ •∩•5 \$ ۰ (IMC= ື່ 1。1645°04 1。1645°04 1.1645\*04 1.1645\*04 1.35\*7\*05 1.3487\*05 H O RULL 1.1645+04 1.1645+04 2.126E+02 2.1265+02 1.1905404 2.1265002 \$ ¢ 222 \$ \$ ŝ P\_DO 9-0R ≎ ŝ ¢ Р<u>-</u>Ор dG-d CHANGED CHANGED CHANGED \$ ÷ CORRECTION • •--\$ ۵ \$ ŝ \$ 17 ž 000000 \$ \$ ٩ \$ \$ \$ TURNED OFF TURNED OFF TURNED OFF TURNED OFF ¢ \$ 0750 7750 sec. COMPUTER SEC. COMP. DEV. DEVIATIONS EREDRS COMPUTER COMP. DEV. USVIATIONS ERRORS COMP. DEV. DEVIATIONS ERRORS COMP. DEV. DEVIATIONS ¢ ≉ ATTITUDE ÷ \$ and the second s - 20 AL T= ≎ ۴ ۵ 4 17107200 ALT= TURNED APP. RADAR TURNED ERHORS 500-0 \$ ¢ DEADBAND ON AXIS DEADBAND ON AXIS DEADBAND ON AXIS 16936500 AL ¢ 170700.00 \* 60.00 MIDCOURSE ¢ ≎ ٦ \$ ≎ ÷ \$ I. S. U. T STAR TRCKR 1 SUN SENSOR 7 SUBSYSTEM SUBSYSTEM ATT. CONT. 5 ct. ct. 0H35M c1° ct. STATUS STATUS COMPUTER в TARGET TAPGET TARGET TARGET AF TER 1960 laan FOR F03 ţŧ ų H COAST COAST 1-1 Δ 7

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\$ ¢ \$ 7.854E=01 7.854E=01 7.854E=01 .4255-01 \$ \$ .425E=0] \$ \$ 0 ٠ o dù-V A-OP \$ \$ ≑ \$ ۵ ٥ \$ \$ \$ ł ⊅ XMITTER XMIITER 1.425E-01 7.854E-01 7-854E-01 7-854E-01 \$ \$ \$ 7.8545-01 1.425E-01 ·1330 5792 -.8043 -1330 .5792 \$ \$ PITCH ¢ PITCH A-CR c A-CR ٩ \$ \$ -55,8801 \$ \$ \$ 1 RADAR \$ \$ . RADAR ≎ . 2995 9326 . 2016 \$ \$ \$ 1.425E=01 7.854E=01 1.425E=01 7.854E=01 7.854E=01 YAW ۷AW •854E-01 ≉ \$ ٥ COM. RCVR. APP. \$ A-DR A-DR ٥ C \$ LONG= RCVR. APP \$ ¢ \$ 5133 3356 7899 .5133 .3356 \$ \$ \$ 60SEC. ROLL אטרר \$ ÷ \$ 9.427E-03 5.457E+00 9.827E-03 5.557E+00 5.557E+00 \* \$ ¢ 5+557E+00 25.6507 90.00 \$ \$ \$ • ₩ ₩00 ۷<del>۳</del>0P 880 40-1 a a e \$ \$ ¢ ¢ ¢ \$ -- 6571 -- 6189 -- 4303 SUN SENSUR ISU/C.P.S. ≎ SUN SENSUR ISU/C.P.S. ¢ \$ .6189 --6571 E0E\* --102.68. BETWEEN OPTICAL AIDS= **₽** ₽ \$ LATE \$ 8.1385-03. 7.6685+00 --882DEG. IN å 8.138F-03 7.668F+00 7,669E+00 7.6695+00 **a** 0 ¢ \$ ≉ C V-CR \$ ac-A \$ 610.68 \$ CH -2101 -3979 \$ ¢ --8931 \$ \$ \$ g \$ \$ \* V-DR 4.873E-03 5.878E-03 7.586E\*00 7.686E\*00 7.686E\*00 ≎ ¢ ¢ ٥ \$ 7.6865\*00 1.13140 •7240 •6772 ٥ \$ 01 ROLL V-DR -ANG ≉ ¢ č ä ₽ ÷ \$ THCKR THOKR \$ \$ \$ PRPGTE, PRPGTE --119066. ≎ ¢ ≎ ¢ ٩ ~~~ ~~ 17 4.4986992E+07 9.750E+04 6.405E+03 9.760E+04 6.405E+03 9.750E+04 \$ \$ \$ \$ \$ ≄ ATT. CONT. STAR -0 -0 ATT. CONT. STAR ¢ d0-d 0-d -.1333 -.0248 -.9908 8066 .--SUJN-CANOPUS ò <u>0</u>2 \$ ÷\$ PITCH PITCH \$ ≉ BETWEEN CELESTIAL BUDIESH \$ ¢ ¢ YAW 5.6045+03 1.7045+05 5.6048.03 1.2048.00 1.2048.00 1.2038.00 ۴ \$ \$ 2007 • 1 9007 • 1 L.203E+05 •1116 •••0986 -.7061 0.00S) 0.005) \* \* \* \* ٠ ≑ p=0R VEL= YAW Y A W P-CR 60°37 8.00 22.52 MAX. ŵ ٥ TO LOCATE • 025056. \$ \$ Σ MOCHO \$ ¥ .7029 69200-6.8730954E+11 -+0769 .7029 1.1905+04 1.2535+04 1.3685+05 1202 --3• 0. 3 20.00 8.00 ¢ ⊅ \$7 20.00 1.1908 404 1.3338 405 1.3468 405 s. U. нo ROLL ROLL 551 ⊅ ٥ ≎ P-02 S S ≎ 01 00-d ¢ \$ • •-1 MANEUVERING ¢ **t**i \$ PIICH ΥNW хх ROLL エエエメイズ BANDS 17 PITCH \* -\$ \$ ~ ş \$ \$ SUN SENSOR TURNED ON STAR FRCKR TURNED ON ATT. CONT. TURNED ON APP. RADAR TURNED ON SEC. COMPUTER 60.00 SEC. COMPUTER COMP. DEV. UEVIATIONS ERRORS ANGLE 2 2 0 0 UEVIATIONS ¢ ¢ \$ ¢ \$ \$ ATTITUDE COMP. DEV. ATTITUDE I. S. C. SUN SENSOR TURNED C. STAR FRCKR TURNED OF STAR FURNED OF DEAU MANEUVERS ŵ 17109000 ALT= 0 TURNED TURNED ERRORS ≎ ≎ 1800.00 ⊅ AT 14RD 0H30M 0.005 BEGIN \$ \$ \$ TARGET CT. \* ₽ SUBSYSTEM SUBSYSTEM с<u>1</u>. **C1**• STATUS STATUS COMPUTER 1. 5. 1. TARGET TARGET COAST FOR COAST FOR H

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FIGURE A-9cc. MARS MISSION, ERROR ANALYSIS OUTPUT

.398F.**~**0 .854E-0 854E-0 A-OP 40-A 525 500 ۵ Φ XMITTER 0000000 0000000 XMITTER 7.359E-02 7.359E-02 5.977E-04 \$ E416. 7.854Em0 -.1194 6508°**-**6603 ----.5743 -.1194 \$ PITCH PITCH C A-CR 7.359E •854E \$ -56.3793 -56.1297 \$ +PITCH RCVR. APP. RADAR + DITCH \$ RCVR. APP. RADAR •-2872 •9344 --2872 -.2109 -.2109 •9344 ¢ 1.375E-01 2.926E-04 YAW YAW 7.8545-01 1.3755-01 1.375E-01 .854E-01 ۵ ŵ ٦ \$ A-UR A-0R LONG= LONG= ¢ ≎ \$ .7910 .73357 .5115 .5115 \$ \$ \$ 60SEC. 0.00000 .02586 POL-ROLL ¢ \$ \$ 9.827E+03 5.557E+00 5.557E+00 9.8275-03 5.5575+00 5.5575+00 \$ \$ 25.6508 ٥ 25.6509 TUTTUO 00\*06 ۵ ¢ . MOC ¢ TPIL 40••V 40**-**۸ \* HOC a a a а 0 α <u>ο</u> ¢ \$ ¢ \$ \$ ۵ SUN SENSOR ISU/C.P.S. --6571 -6189 ٩ -.4303 SUN SENSUR ISU/C.P.S. \$ \$ • Y A W ,ΥAW --4303 .6189 -.6571 MARS MISSION, ERROR ANALYSIS AKS. SSIL LERF LAN& SIS AIDS= LATS Z ¢ LAT= ٥ 8+1388-03 7+6688+00 7-6698+00 8.1385-03 7.6685+00 å 7.669E+00 å ¢ \$ \$ -+000DEG+ 1.00000 PASSES 1.00000 PASSES ٥ \$ X-CR \$ V-CR 616.98 102.68. BETWEEN OPTICAL 616.68 1268.-.3979 ¢ .2101 \$ \$ -,8931 .00000 .02586 ¢ ¢ ¢ ŝ č ¢ \$ \$ 7+636E+00 7+686E+00 \$ 7-5865+00 7.686E+00 \$ \$ G.8785-03 5.0706-03 .13140 •7240 •6772 •161<u>•</u> ¢ \$ φ. V-05 V-0R ROLL ₽NG¤ ¢ =ONA \$ \$ ğ č \$ \$ \$ RATES (RAD/SEC)=ROLL 29.6 SECONDS WITH RAIES (RAD/SEC)=HOLL TRCKP \$ \$ TROKE \$ NO PRPGTE. SECONDS WITH SECONDS WITH --000DEG. \$ \$ ۵ 4.4487207E+07 4.760E+04 4.750E+04 SUN-CANOPUS 4.4487422E+07 てて 17 N 9.750E+04 6.4055+03 6.405E+03 ≄ \$ \$ ATT. CONT. STAR p...o ₽ ۵ STAR ٥ -05 -.1455 -.9858 --4688 -.1455 --0342 \$ \$ ٥ P11CH PITCH . bbd-A ANGLE BETWEEN CELESTIAL RUDIES-≉ \$ \$ ATT CONT. \$ ≎ ٩ YAW 30.4 1.2048+05 1.203E+05 F0R 5. KOAR+03 1. ROAR+05 ٥ ٩ . 2035. 45 5.4045403 \$ • 1269 -.7059 .1269 -.6949 -. (059 (\$00+0 \$ ¢ \$ FIGURE 1 ¥Δw VELa dOrd YAW 00°8 23.32 V5La 80 - d MAX. 60.49 MANEUVERING. REGIN SEARCH \* ¢ ¢ ANGULAR ÅR .000DEG. ¢ ¢ ٩ ANGUL ٦ ¢ MI HO Z ⊅ \$201°-•707• -- 07A9 --0789 6.8731335E.11 6.87317155+11 5. U. 8.09 ۵ \$ ٩ 20.00 20.00 ⊃. 1.1905404 1.3535405 1.3485405 1 • 1907 • 09 1 • 3537 • 05 1 • 3487 • 05 SET SET RULL n L L L L ٦ ۵ \$ î 80-d g CANOPUS 001 ۴ ٥ \$ • \$ ⊅ \$ e ---: ままメン 12 ROLL YAW PITCH 17 BANDS PITCH H X X X \$ ≎ -2 SUN ¢ ⊅ ŵ COMPUTER 60.00 SEC. COMPUTER COMP. DEV. DEVIATIONS ERRORS F 0R SEARCH FOR \$ DEVIATIONS ERRORS \$ ΰ COMP, DEV. ATTITUDE ATTITUDE DEAU MANEUVERS ¢ 17109060 ALT= ⊅ \$ 17109120 ALT= SEARCH ٥ ¢ 500°0 ¢ 500°0 W2EH0 \$ \$ ¢ ACQUIRED GNG ٥ \$ \$ SUHSYSTEM SUNSYSTEM ct. MICHO 0-1-0 с<u>1</u>. STATUS TARGET TARGET TARGET նթել AT 1980 0. C L 11 |-ti らいPOPUS COAST 4 7 A-60

7.1255-05 4.3765-04 4.3176-04 7.1255-05 4.3765-04 4.3765-04 4.3176-04 ·175E-05 4.317E-04 ψ \$ ¢ \$ C A-OP CCIV c A-O-A ο Ŷ \$ ۵ ø ¢ XMITTER 6.0255-04 5.9825-04 \$ \$ 7.1255-05 105 5.9525-04 ß 6.025E-04 .0255-04 5.982E-04 6608.--,1194 .5743 7.125E-0 ¢ ¢ PITCH A-CR 0 A-CR \$ \$ -56.6289 \$ \$ ¢ 4 SUN SENSUR ISU/C.P.S. COM. RCVR. APP. RADAR 9 -2872 9344 -.2109 \$ ¢ 7.1255-05 3.0226-04 2.9376-04 7.125E+05 3.022E+04 2.937E+04 7.1256-05 3.0226-04 2.9376-04 YAW ≏ ۵ \$ ۵ A-DR A-DR A-DR LONG= ż \$ .5115 .3357 0162\* ٥ \$ 1.105845E+01 4.100000E+02 60SEC. ROLL \$ \$ RANGE DOT 9 • 827E • 03 5 • 457E • 03 5 • 457E • 00 5 • 457E • 00 5•4238+00 5•538400 1•2155+00 5.4507+00 5.4572+00 2.9622:01 ¢ \$ 25.6510 3.3248A1E-01 4.000000E+01 00.06 \$ \$ V=0P 40-7 0-N ¢ \$ SUN ANG. 81.409 ¢ ¢ \$ \$ .6189 -.4303 -.6571 UPTICAL AIDS= \$ \$ Z 89.919 LAT= 8.138E-03 7.668E+00 7.669E+00 7.668E+00 7.668E+00 1.150E-01 7.668E+00 7.668E+00 1.794E-02 <u>a</u> AZIMUTH 1.374944E-07 \$ ≄ -.000DEG. \$ ٥ V-CR V-CR 4-CR ¢ 38,000 2.693598E+03 3.000000E+02 ٠ -.8931 ELE. \$ \$ g \$ \$ 7.6502+00 7.6502+00 3.1744-01 5.8785-03 7.8065400 7.6065400 7.686E+00 7.686E+00 4.4625-02 \$ \$ 102.68. BETWEEN •7240 •6772 •1314 \$ \$ ROLL V-CR V-0R 40-2 ELEVATION 1.4892976-07 -0. 83+172 \$ ۵ ANGH A2M č \$ ¢ ATI. CONT. STAR THCKR \$ \$ NO PRPGTE. INCRITIAL SYSTEM ERRORS (RANGE, PANGE RATE) = --000DEG-\$ ¢ 4.4987637E+07 17 9.5465×04 9.7605×04 9.7697.06 4.7607.06 4.7612.00 9.750E+04 6.405E+03 6.010E\*04 ¢ \$ ¢ \$ 0-0 - 0 6.2374201E\*04 p-0P 90-9 -1455 4886.---0362 \$ ÷ PITCH RANGE DOT .01000 DEG. BETWEEN CELESTIAL BUDIES= Φ ۵ 1.507297E+05 5.000000E+01 010 ŧ ۵ YAW 1.2045×05 1.2045×05 5 • ×0 ∻n + 00 1 • 20 ∻n + 00 1 • 20 ÷n + 05 ¢ ¢ 1.2045+05 . 204F.05 2,3535+03 2.4502402 •1269 ••0068 .01000 - 1059 0.005) RANGE ٩ \$ 9-CP 60°\$3 MVX в. 00 23. 32 VEL= p=CR 5-CR MAX. ٩ \$ .000DEG. \$ \$ SUNTCANOPUS HLDSTOVE 6.873289956\*11 MEASUREMENT ERACHS E TNERTIAL SYSTEM 1.5 MI HO ٥ ¢ 4201° APPROACH RAUAR NPDATE -- 0789 11+30002210+9 s: U. 10. 10. ≎ 1.00001405 1.00001405 2.70001405 \* 2.4975463 1.1905+04 1.3535+05 1.3483405 SE1 RULL RANGE 010 2 ţ. \$ P~nn P-0R 5 ÷ ¢ acird CHANGED CHANGED CHANGED • •--1 ⊅ à XAN н н н х <del>х</del> х RULL HO1Id PITCH BANDS DSIF SEARCH FOR φ ~ ⋧ ¢ ٩ COMP. 03V. DEVIATIONS JRFARD COMP. DEV. DEVIATIONS Erroqs COMPUTER 60.00 SEC. \$ ٩ COMP. DEV. DEVIATIONS ANGLE ATTITUDE ~ ~ ? 0520 GOLDGTONE KALMAN UPOATE MANEUVERS KALMAN UPDATE ≎ ≎ 17109180 ALT= STATION 520683 DEADMAND ON AXIS DEADMAND ON AXIS DEADMAND ON AXIS φ \$ 1940 0H33M 9.005 ¢ ≎ ACQUIRED ٢ ٥ SUBSYSTEM ; ; Ĵ STATUS STATUS TARGET **TARGET** APTER AFTER FOR 11 1--SUGONVO COAST ⊷ ₹

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FIGURE A-9ee. MARS MISSION, ERROR ANALYSIS OUTPUT

٩ 255-05 3885-04 .1255-05 4.388E-04 255-05 3305-04 .330E-04 .4055-04 4.347E-04 ⊅ \$ ≎ ń A-0P A-OP A-OP \$ ⊅ ٥ 4 ¢ ¢ XMITTER 7.1255-05 6.0335-04 7.1255-05 6.046E-04 50-1E-04 5.991E-04 ¢ ٥ 6.004E-04 5.991E-04 5743 • • 6034 -1193 PITCH ₽ \$ PITCH c 7.1256 A-CR \$ ¢ 6.033 -56.8785 -57.1281 \$ \$ \$ \$ RCVR. APP. HADAR ۵ - 2872 - 9344 - 2109 .9344 --2872 -.2109 \$ \$ \$ 7.125E-05 3.064E-04 2.980E-04 7.125E-05 3.040E-04 2.955E-04 7.125E-05 3.040E-04 YAW ۸AY 2.955E-04 ٥ \$ ۵ A-UR A-DR \$ ¢ ۵ A-1)R LONG= LONG= \$ \$ ¢ .5115 .3357 .7910 .5115 .3357 .7910 ٩ \$ \$ 60 SEC. ROL ⊅ \$ ¢ ROL ۵ 7.029E-02 3.041E-01 2.960E-01 5.505+00 5.5575+00 2.9627501 25.6511 V=0P 25.6512 \$ ۵ \$ 7.029E-02 3. n41E-01 2.960E-01 90.00 ¢ \$ \$ ¢ • WOU d0-1 a no o no 40-V 000 000 000 -\$ \$ ≄ ¢ ≎ ۵ \$ \$ •6189 SUN SENSOR ISU/C.P.S. -. 6571 -. 6303 \$ \$ \$ --6571 -.4303 AIDS= \$ \$ ٥ Z 89.919 LAT= LATS 7.6685+00 7.6685+00 1.7945-02 1.5756-01 1.5916-01 1.8266-02 ٥. 0 90 .575E-01 1.826E-02 .591E-01 \$ \$ \$ UOF=1.541 \$ \$ 2-CR 40-7 \$ V-CR OPTICAL 616E.-1012.1012.1 ⊅ ₽ 89.919 ٥ --893I ۶ \$ \$ č g \$ \$ \$ 7.686F\*00 7.686F\*00 5.462F\*00 1.901E=01 1.934E=01 5.450E=02 (FI/SEC) + \$ ¢ ٥ 5.4505-02 •966E-01 RETWEEN 1-901E-01 • 7240 • 6772 •7240 •6772 •1314 \$ \$ \$ ROLL V-08 X-98 -V-0R ⊅ ≉ ANG \$ ¢, ANGS ñ c ₽ \$ ¢ Ŷ ٥ THCKR \$ PRPGTE 102.68, PRPGTE --000DEG. ¢ \$ \$ 1.2089E+01 4.4988067E+07 9.7435.404 9.7605404 4.7612403 エエエン・シ 4.49878525+07 4 • 7400+04 9 • 7600+04 4 • 7610+04 I I I X X X 90+3892+04 9.750E+04 4.761E+03 \$ ţ ٠ STAR d0-d \$ ¢ d0-d \$ d0-d --9888 --1455 --1455 --0342 -.9888 2450\*oz ٥ 02 ٩ \$ HOTIGH \$ PITCH ≎ \$ Û BODIESE \$ ATT. CONT. ÷ ¢ ¢ ¢ YAW 1.2045+05 1.2045+05 1.2065.05 1.2065.05 2.4505.02 \$ \$ \$ INA .(U 2.4595402 •1269 •1269 •1764 ••5568 -- (053 -- 1059 (200-0 (500-0 \$ ¢ ¢ ¢ VEL= P=CR YAW v5ta b Cu MMY 64.00 8.00 23.32 p...CR MAX. \$ \$ \$ CELESTIAL -000DEG-≎ \$ ¢ \$ Σ Σ ¢ \$ \$ \$∠02• \$∠02• • 707 •-6820... --07R9 6.8732857E+11 P-DA 6.8732477E+11 \$ CORRECTION (IMC= \$ \$ •01 10 s. U. 1.3535405 1.3535405 2.7655405 1.3537\*05 1.3537\*05 2.7657\*05 2.7657\*02 1.3535\*05 1.3535\*05 2.7655\*02 I H O SE 1 ROLL າງດະ ¢ ¢ ٩ \$ BETWEEN 20-d 6 ç p-nn ¢ ¢ ¢ 4 ŵ • •--¢ PITCH ΥN PITCH 12 77 2708 てよく BANDS エエメン ¢ \$ --٥ \$ ٦ ≎ \$ COMP. DEV. Deviations Errors SEC. 60.00 SEC. COMP. DEV. DEVIATIONS ERRORS COMPUTER COMP. DEV. OSVIATIONS ERPORS ANGLE \$ ≎ \* \$ ATTITUDE ATTTTUDE DEAD MANEUVERS ALT= ÷ 17109300 ALT= \$ \$ ٥ ~ ¢ ≎ ¢ \$ 00.09 000°0 000°0 MIDCOURSE ₽ \$ \$ ₽ ACQUIRED ÷ 17109240 ٥ Φ \$ SUBSYSTEM о и З 5 И MASH0 6 ů, c1° STATUS 5 5 TARGET TAPGET TARGET 710657 ARTCH CIRP[ TA AT 1990 20 С С С COAST FOR U F 11 CANAPUR C0A57

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FIGURE A-9ff. MARS MISSION, ERR

MARS MISSION, ERROR ANALYSIS OUTPUT

20 8545-01 ₽ .854E-0 .854F-0 545-0 8545-0 .8545-0 ≎ A0-A C du-v ∌ ٥ \$ ¢ \$ \$ XMITTER XMITTER \$ \$ .854E-01 .8545-01 .854E-01 7.8545-0 •5743 •854E=0 \$ đ PITCH 0 A-CR ម៉េ ទ ហ 3 ź ⊅ 179.5295 \$ RADAR \$ \$ RADAR .6030 ..7694 -.2108 \$ \$ ۸AY 7.8545-01 7.854E-01 7.8546-01 101 7.854E-01 \$ \$ ATT. CONT. STAR THCKR SUN SENSOR ISU/C.P.S. COM. RCVR. APP. COM. RCVR. APP. A-0R \$ \$ \$ c A-UR .8541 .85% LONG= \$ .5677 .7910 \$ \$ PULL ≎ ¢ 3.124E+03 3.113E+03 ¢ \$ 25.6958 2.4585+02 1013 2.960E-0 ☆ ≎ ¢ V-0P 880 3142. \$ \$ \$ \$ .6189 SUN SENSOR ISU/C.P.S. ¢ --4304 \$ \$ \$ LA7a 8.457E+01 3.551E+03 3.550E+03 9 -01 •826E-02 1 ٥ \$ X-CR С \$ 20-2 - 2 1.5915 1.575 606.68 ¢ \$ •5860 -.8081 •0601 ¢ ¢ g \$ ≑ 2.008E+02 2.054E+03 2.041E+03 \$ ¥ 01 10-\*\*50E-U2 1.600937285+05+ DUF=1.000 \$ \$ .4743 • 4022 .7831 104 V..DR ? 3080. ٩ ⊅ aNG: 106.1 č ¢ \$ ATT. CONT. STAR THCKP \$ \$ \$ ¢ 4.5094146E+07 7 2 × 3,4395+06 -6.R11E+02 -2.500E+01 3.439E+05 60E+04 4.761E+03 485+04 ≎ ≎ 2.4045E+02 (FT/SEC)+00F=1.000 \$ \$ p-0p î SEC. ບໍ່ ພູດ ວ ວະເ ວະ 540. ----------.1455 -.0342 \$ \$ PITCH . 20,00000 956. 20,00000 956. 20,00000 956. ¢ \$ 00.74051 10500-00 13947.00 00-00-00 3947.00 17138900.00 <200,00 00.0020 6300.00 \$ \$ 1.6105405 \$ \$ 1.610E+05 2.4595402 .204E+05 .204E.05 •1264 650] --6900 .-\* \* \* (500-6 ≎ ハロしょ WVX ĉ 2-CR 510 10 ≎ TARGET MISS, RMS= \$ ¢ \$ MAI48 6.89240255+11 4.7035+06 4.7035+06 ч. С. ¢ ¢ 3. U. 1.353F+05 1.353F+05 -6.1245-01 2015 2.7655.02 1 CHANGED TO 2 CHANGED TO 3 CHANGED TO \$ ¢ ad-d ⇒ င့် \$ с † P-02 • н \$ 8 |--1 \$ ин Х≻ IΖ ¢ ٢ \$ PETRO CORRECTION, MMR \$ 5. 5. 0 OFF 910 COMPUTER COMPUTER 055 TURNED OFF TURNED OFF 29649.00 SEC. ATT. COMT. STAR YRCKR \$ \$ COMP. DEV. DEVIATIONS COMP. DEV. DEVIATIONS ATTTUDE RUSHIS NUS surc, P.s. CON. NOVR. 1011 TONR \* \* \* \* ¢ T= 17133949 ALT= , S, U, CONPUTER X-111122 TURNED APP. RADAR TURNED TURNED ERRORS 0 \$ ERFORG DEADHAND ON AXIS DEADHAND ON AXIS SIXV NO GNUHUVZU ¢ ≎ SH49M 9.005 \$ SUBSYSTEM SUUSYSTEM I. S. U. STAR TRCKR SUN SENGOR c1° c1. ATT. CONT. COMPUTER STATUS STATUS T A R G E T TARGET A7 1990 COAST FOR

FIGURE A-9gg. MARS MISSION, ERROR ANALYSIS OUTPUT

45 42 39 .92 LOUISOL SME 33 -continue Super-30 27 18 21 24 Burn Time, minutes 15 2 თ φ М 0 ft/sec ft 3.57 4600 4200-3900-4500-4400-4300-4100-4000 м. -3.0 2.9-2.8-3.4 -3.3 -3.2rad x 10<sup>4</sup> 14 ح 0 5 ы 12--8 2 0

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FICURE A-10. RMS ERRORS vs. BURN TIME

Initial conditions are the position, velocity, and attitude errors at periapsis from the computer runs shown in Figure A-9gg. The attitude errors are reduced to those that would exist if a star tracker and Sun sensor update were performed prior to ignition. The calculations are direct integrations of the acceleration uncertainties,

 $\underline{\Delta a} = \underline{\Delta \Phi} \times \underline{a} + \underline{\Delta a}_{a}$   $\underline{\Delta \Phi} = \underline{\Delta w}_{g}$ where  $\underline{\Delta a} = \text{sensed acceleration error}$   $\underline{a} = \text{nominal acceleration}$   $\underline{\Delta a}_{a} = \text{accelerometer errors}$   $\underline{\Delta \Phi} = \text{attitude error}$   $\underline{\Delta \psi}_{g} = \text{gyro errors.}$ 

Neglecting gravity feedback (Schuler effect),

 $\underline{\Delta a} = \underline{\Delta \dot{v}} = \underline{\Delta \dot{r}}$ 

Increasing the burn time from 3 to 45 minutes increases position error 13 percent and velocity error 25 percent. This relatively small increase is due to the large initial condition values. Burn times of less than 3 minutes were not considered. As the burn time is shortened to approach an impulsive delta-V, the nonlinearity term will cause infinite errors. The 3 minute burn requires a  $21.1 \text{ ft/sec}^2$  acceleration or 4800 lbs thrust for a 7553 lb spacecraft. This is felt to be a reasonable upper limit on acceleration.

More significant effects on the penalty will be due to the reliability requirements for long burn times. The effect on the penalty calculations of including the reliability effects for a 3 and 45 minute burn are shown in Table A-I.

The results assume the propulsion system constant (nozle weight, etc.) and coefficient (tankage factor, etc.) would be the same for the two extreme cases.

In summary, the choice of a retro burn thrust level and resulting burn time could affect position and velocity errors by 10 to 30 percent and the penalty by 20 to 40 lbs. Modifications to the penalty calculations could be made to reflect carrying several (3 to 5) identical nozzles with a common tankage system. This would provide accurate midcourse correction control by using only one engine, yet enable shorter retro burn times to lower the astrionics reliability effects. As in the case with all results shown in this report, these values demonstrate the use of the penalty evaluation technique and are not to be interpreted as conclusive

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	Reti	co Burn Duration	
Parameter	Not Considered In Penalty	3 Minute	45 Minute
P <sub>FA</sub> (Specified)	.150	.150	,150
P <sub>FR</sub>	.111	.119	.125
P <sub>FV</sub>	.043	.035	.029
Ψv	1.71	1.81	1.88
∆V capability (ft/sec)	4373	4405	4428
W <sub>DV</sub> (1bs)	3120	3140	3160
Penalty (lbs)	3499	3519	3539

## TABLE A-I. EFFECT OF RETRO BURN TIME ON THE PENALTY

for the Viking or any other mission. Changes in the direction of the retro  $\Delta V$  do not affect the penalty. Changes in the magnitude do. Changes in the magnitude of the retro  $\Delta V$  necessitate burning more fuel and thus require either carrying more fuel for a given probability of having sufficient  $\Delta V$  capability or, as an alternative accepting a greater probability of failure if the amount of fuel carried is held fixed. Since the onboard system uses computed deviations as a basis for making corrections, the downrange component of computed deviations represents the standard deviation of the uncertainty of the retro  $\Delta V$ . Thus, the retro burn standard deviation, indicated as RMR, is 240 feet per second. For the analysis shown, the target miss requirement was assumed to be the magnitude of the periapsis vector 1.6 x  $10^5$  feet. This value is used as the target miss parameter.

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#### Penalty Evaluation

The evaluation of the astrionics effectiveness using penalty Mode 3 is shown in Figure A-11. The first line of the penalty evaluation shows the three accelerometers (Arma D4E) and three gyroscopes (GG334-A) selected. The next block of information is the results of the ISU design calculations based on the specified hardware components. The ISU design routine selected horizontal design No. 4 as the optimum. The various designs available for selection are discussed

	334-A		.ENERGY= 168.526 POWER 43.500 P.FAIL= 07037 WEIGHT= 33.463	ENERGY= 271.940 Power = 141.375	WEIGHT= 5.608	ENERGY= 17084.659 Power = 130.519 P.FAIL= 0.04310 Weight= 64.530			ENERGY=. 29.167 Power =. 10.000	P•FA1L≈ .00112 WEIGHY≖ 54.189	WEIGHT= 110.454	WEIGHT# 110.000	WEIGH1# 3120.273	
	99		EXCIT EXCIT EXCIT TOTAL TOTAL	TOTAL	TOTAL	101AL 101AL 101AL 101AL			TOTAL	TOTAL	TOTAL	TOTAL	TOTAL	
	66 334-4	3.874	•	2.1750 1.0875	61	XMITTER DESIGNED 1.750 24.534 24.534 24.019 0.0000 1.430	2.917	SEC) PITCH	7.7066 251.9980 13.4314 273.1360	1301	·			
	= 66 334-A	N TIME (HR) -	1.345 10.000 6.000	L COND.=	DEG.)= 2.41	APP. KANAR APP. KANAR 1.167 1.167 11.667 10.000 15.000 15.000	N TIME (HR)=	UMPTION (LH- Yaw	37.5446 259.2120 8.5985 305.3751	IGHT= 21.234	766			WEIGHT 3498+517 3836+483
	GYROSCOPES	0 9 8	WEIGHT LATION= RONICS= VNEVIS=	MAX . THERMAI MIN . THERMAI	NIING TOL. (	04. RCVR. CH+503X 47-0.819 47-0.819 3.500 3.100	ō	FUEL CONSI	= 13.2344 = 588.7063 = 8.691 = 610.6098	FUEL WE	Ч6Ү≕ 17386 <b>.</b>		4372+689	ASTHLONICS= SPACECRAFT=
	4 D-4E	CYCLES	ELECTI	75 00	•26 P0I	N N N N N N N N N N N N N N	ES= 5		SEAPCHING DEAD BAND MANEUVERS TOTAL IMP	.1208	TOTAL ENE		ABILITY#	
	-4E ARM	(WUM)	7 8.540 2.86	97.8 .97	(K¥)#	100 100 100 100 100 100 100 100	CYCL	c	ວວຕ.ຄ.ຕ	LSE= 1189.	468.[9		=1.001 CAF	ES • 0434] • 0000(
	ARMA D	4 0PT	желоски 2000 Соски Сосски Сосски Сосски Сосски Сосски Сосски	на РОМЕК На РОМЕК	7 ERP			HOLFd	00000000000000000000000000000000000000	TAL IMPUI	*ER= 21		157 DOF	SAPILITI EL.FUEL=
	ARMA U-4E	GN NUMBER	510NS 650 650	WAK HEATI MIN HEATI	(DB)= 16.	SIAR ITT-LUN-00 ITT-LUN-00 23-33 5.0000 5.0000 5.0000 5.0000 5.0000 5.0000 5.0000 5.0000 5.00000 5.0000 5.0000 5.00000 5.00000000	ANAL YSIS	YAW	11 20 20 20 20 20 20 20 20 20 20 20 20 20	154 TU	TOTAL PO			PROU F,DELTA-VI ET MISS (
	LEROM.= A	ONTAL DESI	THE UIMENS	VLYSIS	GAIN	MRTRRS MRTRRS T RUTRRS 34 8 74 34 8 75 36 90 9 00 36 90 00 36 00 00 36 00 00	OL SYSTEM	TZING (LB) ROLL	0000 0000 0000 0000 0000 0000 0000 0000 0000	S= 97	DATA		ENGINE Dev. delta	10N INSU TARG
STNBNONENTS	ACCE	SH UATA (HURLZ	CUTS LENG MID HEIG	SU THERMAL AN	NFENNA DESIGN	URSYSTEM PARA CO CY CYCLESS FNRES FN	TILIUDE CONTR	THRUST S	ADLER PROVE ACT = IMPACT = ADLEUVIRS = MAX+THRUST = MAX+THRUST =	NO. OF FIRING	NERGY SOURCE	DNIBL	ELTA-VELOCITY STD-	ENALTY SUMMAT

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FIGURE A-11. MARS MISSION, PENALTY EVALUATION

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in detail in Reference 4. The ISU was cycled on and off six times and had a total operating time of 3.874 hours. The designed ISU is 9.35 inches long, 10.45 inches wide, and 5.45 inches high. A breakdown of the weights for the block, base, cover, insulation, electronics and the sensor components is shown. On the extreme right of this block of information are the subsystem parameters necessary for penalty calculation. These are the total excitation energy, 168.5 watt hours, the excitation power, 43.5 watts, the total probability of failure for the ISU, .07037, and the total ISU weight of 33.463 pounds.

The ISU is assumed to be mounted on a variable thermal conductance which requires a maximum heater power of 97.875 watts and a minimum heater power of zero watts. To maintain these conditions, a maximum thermal conductance of 2.175 and a minimum thermal conductance of 1.0875 is required. The heater power and energy necessary to maintain the ISU at the desired temperature are added to the excitation energy and power to determine the total energy and power required for the ISU. These are 271.94 watt hours and 141.375 watts respectively. The transmitter and antenna system is designed as discussed earlier in this report. Since a transmitter was not specified, a tradeoff optimization between antenna size and transmitter size is performed and the resulting antenna is shown to be optimum with a gain of 16.7 db, to give an effective radiated power of .26 kilowatts. This allows a pointing tolerance of 2.409 degrees. This antenna is estimated to weigh 5.608 pounds.

The next block of information shows the operating time in hours, the number of on-off cycles, the excitation energy in watt hours, the power in watts, the probability of failure, and the weight in pounds for the computer, star tracker, Sun sensor, communication receiver, approach radar, and transmitter. It should be noted that no ISU/CPS power supply is carried. It is assumed that the ISU and computer contain their own power supplies. As mentioned above, the transmitter is designed rather than specified and the resultant design uses 24.53 watt hours at 14 watts with essentially no probability of failure and a weight of 1.43 pounds. The total energy, power, probability of failure, and weight for these subsystems are shown at the right hand side of the report.

The attitude control system analysis indicates the system is cycled on and off five times with a total operating time of 2.9 hours. The thrust sizing requirements are shown at the left side of the report. The thrusts required on all three axes for sizing conditions of solar pressure, micro-metrorite impact, trajectory maneuvers, and steering during midcourse corrections are shown. The maximum thrust required on each axis is used as the design criteria. The maximum thrust is 6.095 pounds and is determined by the maneuvering requirements for all three axes. Fuel consumption due to searching, dead band operation, and maneuvering is calculated with the thrusts set at the values specified above. The total impulse in pound~seconds required for thrusting about each axis is shown at the bottom of this block of information. It should be noted that dead band operation is dominant in use of fuel about all three axes. On the following line, the total number of thruster firings required and the total impulse requirement in pound-seconds are shown. From this information, the total fuel weight, 21.23 pounds, is calculated as well as the energy, power, probability of failure, and total attitude control system weight.

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The energy source subsystem is sized using the total power and the total energy requirements for all of the above subsystems. Since it is assumed that radioisotope thermo-electric generators (RTG's) are used for production of power, the power rather than the energy becomes the important sizing parameter. For the total power requirement of 281.894 watts it is estimated that RTG's totaling 110.454 pounds will be required. The weight analysis is then concluded with the addition of 110 pounds for intersubsystem wiring.

Under Penalty Mode 3, the  $\Delta V$  engine is sized, based on the probability of failure due to insufficient fuel which can be tolerated within the limit of specified overall astrionics total probability of failure. This calculation is shown under the heading Penalty Summation. The total astrionics probability of failure is specified as .15. With failure due to lack of reliability equal to .1143 and probability of target miss essentially zero, the  $\Delta V$  propulsion subsystem must have a probability of insufficient fuel of .0434 or less. The  $\Delta V$  calculations from the error analysis indicate a standard deviation in  $\Delta V$  of 334.157 feet per second with one degree of freedom. This standard deviation in conjunction with the mean specified in the data bank, 3800 feet per second, necessitates carrying a capability of 4372.689 feet per second to meet the requirement of a probability of failure of .04341 or less. Thus, it is estimated that the total  $\Delta V$  propulsion subsystem will weigh 3120.273 pounds. Adding the  $\Delta V$  propulsion subsystem to the weights discussed above yields the total penalty (Mode 3) of 3498.5167 pounds. As a matter of information, when this weight is subtracted from the total spacecraft weight of 7335 pounds, a spacecraft structure and payload weight of 3836.483 remains.

When comparing the results discussed above to those obtained from the Jupiter flyby mission, it must be remembered that the retro  $\Delta V$  fuel requirements are included along with midcourse correction  $\Delta V$  fuel requirements when sizing the engine. If separate engines are carried for midcourse correction and retro  $\Delta V$ , the penalty analysis would be somewhat different. In the latter case, the retro  $\Delta V$ , the dominant use of fuel, would not be reflected in the penalty analysis and a much smaller propulsion system would result. In the Jupiter mission, there was no retro  $\Delta V$  so that the engine is sized only to perform the midcourse corrections.

APPENDIX B

THE CUMULATIVE PROBABILITY FUNCTION FOR THE MAGNITUDE OF THE TOTAL  $\triangle V$  REQUIREMENT FROM THE MEAN AND COVARIANCE OF RETRO BURN  $\triangle V$ 

#### APPENDIX B

# THE CUMULATIVE PROBABILITY FUNCTION FOR THE MAGNITUDE OF THE TOTAL $\Delta V$ REQUIREMENT FROM THE MEAN AND COVARIANCE OF RETRO BURN $\Delta V$

The magnitude of the retro burn  $\Delta V$  is found by

$$\Delta V = \sqrt{(M_v + v_x)^2 + v_y^2 + v_z^2}$$
(B-1)

where  $v_x$ ,  $v_y$ , and  $v_z$  are the zero mean random components of covariance  $C_v$ , and  $M_v$  is the mean or nominal  $\Delta V$  and is assumed to be in the x direction. The two extreme cases ( $M_v \approx 0$ , and  $M_v \approx \infty$ ) are easily solved. If  $M_v$  is nearly zero ( $M_v \ll \sigma v_x$ ), the problem reduces to finding the distribution of the magnitude of a vector with zero mean, as discussed in Reference 1, 4, and 8. If  $M_v$  is very large, that is  $M_v \gg \sigma_v$ ,  $\sigma_v$ , or  $\sigma_v$ , the approximation  $v_x = v_z$ 

$$\Delta V \approx M_{v} + v_{x}$$
 (B-2)

is valid and  $\Delta V$  is a normally distributed variable with mean  $M_V$  and standard deviation,  $\sigma_{V_X}$ . The general problem where  $M_V$  is neither very large nor very small cannot be solved in closed form, but is easily handled using Monte Carlo techniques. For typical interplanetary mission results, however, the assumption of large  $M_V$  is reasonable. The runs of Appendix A indicate standard deviations an order of magnitude less than the mean, 3800 ft/sec. The discussion below is based on large  $M_V$ .

PRM ( $\Delta V$ ,  $M_v$ ,  $R_v$ ,  $D_v$ ) is the probability that the magnitude of a vector with mean  $M_v$ , square root of the trace of the covariance  $R_v$ , and degree of freedom,  $D_v$ , exceeds  $\Delta V$ . Since only one component of the vector is significant, this is equal to

$$PRM = P_{r} \left( \frac{\Delta V - M_{v}}{R_{v}} \right)$$
(B-3)

where  $P_r$  is the cumulative normal distribution of a zero mean, unity variance random variable.

REQ (P, M<sub>v</sub>, R<sub>v</sub>, D<sub>v</sub>) is the inverse function used when the probability, P, is known and the required  $\Delta V$  is to be found.

$$REQ = M_{v} + R_{v}P_{r}^{-1} (P)$$
 (B-4)

where  $P_r^{-1}(P)$  is the inverse function of  $P_r(X)$ , the normal, zero mean, unity variance cumulative distribution.

It should be noted that the argument D is unused since, under the assumption of large  $M_V$ , the magnitude is approximately distributed with one degree of freedom. Further study could result in an appropriate analytical or Monte Carlo solution to the general case and then  $D_v$  would be needed.

APPENDIX C

NEW TECHNOLOGY

## APPENDIX C

## NEW TECHNOLOGY

During the period covered, no new concepts were conceived or first reduced to practice.