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AIMP MISSION PROFILE AND REAL TIME COMPUTING SYSTEM DESCRIPTION

BY
R. T. GROVES
A. J. FUCHS
R. J. SANDIFER

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NASA

GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GODDARD SPACE FLIGHT CENTER
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REAL TIME PROGRAM DESCRIPTION

By

R.T. Groves
A.J. Fuchs
R.J. Sandifer

APPROVED BY: C. A. Packard for
J. J. Donegan, Head
Data Operations Branch

F.O. Vonbun
Dr. F.O. Vonbun, Chief
Systems Analysis Office

June 22, 1965

SUMMARY

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This document describes the Anchored IMP (AIMP) satellite mission events and the Real Time Computing System to support the mission. Three broad categories are covered: (a) the general mission sequence of events, which gives launch and transfer trajectory events and required times involved; (b) a trajectory profile, which includes such critical parameters as flight path angle and burnout speed; and (c) a general description of the programming requirements to achieve lunar orbit or an alternate mission.

Author

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Section 1

INTRODUCTION

The primary objective of the AIMP mission is to measure interplanetary magnetic fields, solar plasma fluxes, and interplanetary dust distributions near the moon. Secondary objectives are to obtain information concerning the gravitational field of the moon by analysis of the spacecraft orbital motion and to study radio wave propagation in the vicinity of the moon and the lunar ionosphere. *

The nominal trajectory of AIMP is designed to achieve a lunar orbit in order to provide a more complete sampling of the environment at lunar distances than is possible with circumterrestial satellites. By utilizing the moon as an anchor, the spacecraft will be able to make approximately 13 complete circuits of the earth at lunar distances within a year's time.

The mission profile and the Real Time Program will be designed according to the following constraints and ground rules:

a. During the time period from 3rd stage burnout to as long as 20 minutes after burnout, the AIMP spacecraft will be transmitting optical aspect and spacecraft status telemetry data. Because range and range rate data and telemetry data cannot be transmitted simultaneously, no range and range rate tracking data can be obtained during this time period (up to 20 minutes after burnout), even though tracking station coverage may exist prior to that time.

b. Within 30 minutes after 3rd stage burnout, all optical aspect data received from the spacecraft by Kano, Johannesburg, Ascension, and Tananarive will be available at Goddard.

*References 1 & 2

c. Within approximately 70 minutes after 3rd stage burnout, sufficient tracking information will be available at Goddard from Tananarive and Carnarvon to determine the orbit to an accuracy of approximately ±16 kilometers in position and ±8 meters/second in velocity. *

d. With existing communications systems, range and range rate tracking data cannot be received at GSFC in real time. For example, it requires 10 minutes to receive two minutes of tracking data at GSFC for a sampling rate of one sample per second.

e. Although simultaneous coverage exists over a major portion of the transfer trajectory and the lunar orbit, only one station can receive range and range rate data at a given time.

f. There will be no midcourse correction during the transfer phase of the trajectory between 3rd and 4th stage burns.

g. The go/no-go decision for insertion into lunar orbit will be made approximately three to eight hours after 3rd stage burnout. The sole criterion for this decision will be the ability to insert the spacecraft into a lunar orbit with a six-month dynamical lifetime. ** If this criterion is not met, an alternate mission (circumterrestial orbit) will be flown.

h. Assuming that a six-month dynamical lifetime in lunar orbit can be achieved, the following criteria will be used to determine the time of 4th stage retrofire for insertion into a lunar orbit:

   1. Apocynthion distance to be minimized.
   2. Time in shadow to be minimized or less than two and one-half hours.

i. In the event of a no-go decision, an alternate mission will be chosen.

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*References 3 & 4

**Subject to modification by the AIMP Project Office. Refer to Figure 4-4 for Real Time Program logic concerning lifetime criteria.

Section 2

GENERAL MISSION SEQUENCE OF EVENTS

The Anchored Interplanetary Monitoring Platform (AIMP) spacecraft will be launched by a three-stage Improved Delta (DSV-3E) vehicle from Complex 17, Cape Kennedy, Florida. The flight will be controlled by means of on-board autopilot programs for 1st stage powered flight (one roll and four pitch rate programs) and for 2nd stage powered flight (two pitch rate programs). In addition, both 1st and 2nd stages will be actively controlled from Bell Telephone Laboratories' ground stations. Since the 2nd stage burnout (SECO) is below the horizon for the BTL guidance system, an integrating accelerometer is used to terminate 2nd stage thrust.

After SECO, the 2nd stage programmer initiates a coast phase pitch program to obtain the proper attitude for the spacecraft spin-up operation. The programmer also initiates the firing of the spin rockets, the separation of the 2nd stage, and the activation of the 3rd stage ignition timer. The 3rd stage burn injects the spacecraft into the transfer trajectory to the moon. The duration of burn is about 22.6 seconds; the impulsive velocity increment provided is about 2.68 km/sec. After 3rd stage burnout, the de-spin yo-yo mechanism is deployed, and the spacecraft booms and solar paddles are erected. The spent 3rd stage is then separated.

The 4th stage/spacecraft combination continues on its approximately three-day flight to the vicinity of the moon. Telemetry and tracking data during this period will be utilized to determine the attitude and trajectory so the 4th stage can be fired at the optimum time to obtain a stable lunar orbit. The actual firing of the 4th stage, whether performed by direct command or by an on-board timer, is the final control action on the AIMP trajectory; the spacecraft orbit is then fixed.

A time schedule of mission events from liftoff to injection is presented in Table 2-1, and from injection throught the early phase of transfer trajectory in Tables 2-2 and 2-3. No attempt has been made to detail the roll and pitch rate programs of the 1st and 2nd stages since no firm information is available on these programmed maneuvers at this time. A typical reference trajectory set of parameters is given in Table 2-4, and the launch azimuths and times for the 2nd and 3rd quarter, 1966, launch dates are given in Table 2-5.

Table 2-1. Sequence of Events (Liftoff to Injection)

Time from Liftoff (Seconds)	Event
0	Vehicle liftoff
0	Start 1st stage programmer
90	Start 1st stage active guidance (BTL)
148.62	Main engine cutoff
148.62	Start 2nd stage programmer
152.62	Separate 1st stage
152.62	Ignite 2nd stage
163.62	Start 2nd stage active guidance (BTL)
215.5	Jettison payload fairings
327	Stop 2nd stage active guidance
327	Start 2nd stage integrating accelerometer
542.36	2nd stage engine cutoff command from integrating accelerometer
542.36	Switch to coast phase control
542.7	2nd stage cutoff
562.7	Start coast phase pitch program
626.5	Stop coast phase pitch program
1081.7	Fire spin rockets
1083.7	Separate 2nd stage
1087.7	3rd stage ignition
1110.3	3rd stage burnout = Injection
1156.7	Deployment of de-spin yo-yo mechanism
1204.7	Erect solar paddles
1206.7	Erect spacecraft booms
1211.7	Separate 3rd stage

Table 2-2. Sequence of Optical Aspect Sensor and Associated Telemetry Events*

Event	Time from Injection (Minutes) for 2nd and 3rd Quarter, 1966, Launch Dates**								
	Day 0	+ 1	+ 2	+ 3	+ 4	+ 29	+ 30	+ 31	+ 32
Injection = 3rd stage burnout at 1110.3 sec. from liftoff	0	0	0	0	0	0	0	0	0
Optical aspect sensor operates for attitude measurements	0-14.6	0-12.1	0-9.5	0-7.0	0-4.6	0-15.5	0-13.2	0-10.8	0-8.5
Kano reception of optical aspect telemetry data	1.8-14.6	1.9-12.1	2.0-9.5	2.6-7.0	4.0-4.6	1.9-15.5	2.3-13.2	3.3-10.8	6.5-8.5
Ascension reception of optical aspect telemetry data	—	—	—	2.8-7.0	1.0-4.6	—	4.1-7.0	1.4-10.8	0.6-8.5
Johannesburg reception of optical aspect telemetry data	8.4-14.6	8.3-12.1	7.8-9.5	—	—	8.0-15.5	7.4-13.2	7.0-10.8	6.7-8.5
Tananarive reception of optical aspect telemetry data	9.5-14.6	9.5-12.1	9.4-9.5	—	—	9.4-15.5	9.4-13.2	9.4-10.8	—

NOTE: It is a project requirement that the telemetry data at the above times be transmitted from the above stations to Goddard as soon as received in order to ensure capability for an alternate mission decision.

*Reference 5

**Because launch date information is classified, the first launch day in the 2nd and 3rd quarter of 1966 is denoted as Day 0.

Table 2-3. Sequence of Tracking Station Coverage*

(Early phase of transfer trajectory—first eight hours)

Tracking Coverage (Elevation > 10°)	Time from Injection (Hours) for 2nd and 3rd Quarter, 1966, Launch Dates**								
	Day 0	+ 1	+ 2	+ 3	+ 4	+ 29	+ 30	+ 31	+ 32
Tananarive views AIMP spacecraft (Range & Range Rate)	.16-8+	.16-8+	.16-8+	.16-8+	.16-8+	.16-8+	.16-8+	.16-8+	.16-8+
Carnarvon views AIMP spacecraft (Range & Range Rate)	.50-6.4	.50-6.3	.50-6.3	.50-6.2	.50-6.1	.50-6.3	.50-6.2	.50-6.1	.50-6.0
Johannesburg views AIMP spacecraft (Minitrack)	.15-8+	.14-8+	.13-8+	.12-8+	.11-8+	.13-8+	.13-8+	.12-8+	.11-8+
Santiago views AIMP spacecraft (Range & Range Rate)	6.7-8+	6.7-8+	6.8-8+	6.8-8+	6.7-8+	6.8-8+	6.8-8+	6.9-8+	7.0-8+

NOTES: 1. It is a project requirement that the tracking data at the above times be transmitted from the above stations to Goddard as soon as received in order to ensure capability for alternate mission decisions.

2. Since only one range and range rate station can receive data at a given time, the station giving the most favorable geometry will be selected for the times during which overlapping coverage occurs.

3. The coverage data were obtained from a patched conic trajectory program. Thus, results may differ slightly from trajectories in Reference 7.

*Reference 6

**Because launch date information is classified, the first launch day in the 2nd and 3rd quarter of 1966 is denoted as Day 0.

Table 2-4. Reference Trajectory*

<u>Coordinate System</u>
Center: geocentric
x-y plane: mean ecliptic of date
z axis: north ecliptic pole

<u>Injection Conditions (3rd Stage Burnout)</u>
t = 0 at $16^{h}.40700$ GMT on launch day + 2

t (Hrs. past injection)	Position Vector (km)			Velocity Vector (km/sec)		
	x	y	z	$\dot{x}$	$\dot{y}$	$\dot{z}$
0	-5,503.009 1	+3,851.323 6	+436.628 45	-7.020 366 0	-8.209 601 8	-0.654 961 52
5	+15,270.797	-73,297.176	-6,987.592	+1.529 510	-2.612 597	-0.263 311
10	+40,762.179	-112,547.135	-10,979.296	+1.318 253	-1.868 828	-0.191 539
15	+63,151.200	-142,758.364	-14,090.578	+1.177 769	-1.520 428	-0.157 271
20	+83,385.250	-168,044.837	-16,714.519	+1.075 088	-1.303 332	-0.135 647
25	+101,985.412	-190,051.907	-19,010.069	+0.994 464	-1.149 611	-0.120 162
30	+119,270.585	-209,648.845	-21,061.501	+0.928 031	-1.032 626	-0.108 239
35	+135,452.930	-227,368.688	-22,920.374	+0.871 362	-0.939 523	-0.098 616
40	+150,681.626	-243,572.651	-24,621.244	+0.821 700	-0.863 337	-0.090 595
45	+165,064.878	-258,526.255	-26,188.875	+0.777 155	-0.800 115	-0.083 758
50	+178,681.114	-272,441.868	-27,642.046	+0.736 244	-0.747 795	-0.077 844
55	+191,582.825	-285,509.654	-28,995.903	+0.697 486	-0.706 020	-0.072 707
60	+203,790.413	-297,934.796	-30,263.910	+0.658 572	-0.677 159	-0.068 313
65	+215,253.849	-310,022.541	-31,460.933	+0.612 725	-0.671 785	-0.064 910
70	+225,563.710	-322,568.581	-32,617.067	+0.507 345	-0.751 048	-0.064 679
73.260 24**	+228,908.533	-332,399.553	-33,438.095	-0.223 595	-0.791 506	-0.079 352
73.261 26***	+228,907.700	-332,402.479	-33,438.386	-0.224 054	-0.791 227	-0.079 355

*Data in this table is from Reference 7. The trajectory data were converted from nautical miles and feet per second to kilometers and kilometers per second.

**Begin occultation by moon

***Closest approach to moon

Table 2-5. Launch Azimuths and Times for 2nd and 3rd Quarter, 1966, Launch Dates*

Launch Date	Launch Time	Launch Azimuth
0 Day	14.235 08 hours GMT	78.80 degrees
+1 "	15.179 58 " "	80.40 "
+2 "	16.098 58 " "	84.15 "
+3 "	17.006 58 " "	89.82 "
+4 "	17.941 58 " "	97.32 "
+29 "	13.991 58 " "	82.65 "
+30 "	14.889 58 " "	87.70 "
+31 "	15.806 78 " "	94.55 "
+32 "	16.837 58 " "	103.80 "

*Reference 7

Section 3

TRAJECTORY PROFILE

The current AIMP launch philosophy involves using a launch azimuth varying from day to day, but held constant during a daily launch window of three minutes duration, and using essentially a constant powered flight central angle from liftoff to 3rd stage burnout. For each launch date a new launch azimuth and launch time will be used in order to achieve a desired aim point in the vicinity of the moon.

Selection of possible launch dates is governed primarily by the following considerations:

a. The capability of the vehicle to produce a lunar transfer trajectory must exist using a constant powered flight central angle (liftoff to 3rd stage burnout) while keeping the launch azimuth within range safety limits (70^o - 110^o).

b. The spin axis-sun angle must be within 30^o to 150^o within two weeks of lunar orbit insertion, and must remain within these limits for at least a three-month time period.

Typical trajectory conditions (nearly constant for any launch date) are as follows:

a. Central angle (earth-fixed) from liftoff to 2nd stage burnout: 20.4^o
b. Second stage burnout altitude: 148 km
c. Second stage burnout speed: 8.373 km/sec
d. Second stage burnout flight path angle: 0^o
e. Central angle (earth-fixed) from liftoff to 3rd stage burnout: 59.4^o
f. Third stage burnout altitude: 355 km
g. Third stage burnout speed: 10.822 km/sec
h. Third stage burnout flight path angle: 5.3^o

Azimuth and sub-vehicle latitude and longitude at 2nd and 3rd stage burnouts are then functions of the respective central angles and the launch azimuth for a particular launch date.

The transfer trajectory will be nominally of 70 to 80 hours duration to pericynthion and will approach the moon within a few degrees of the ecliptic plane leading the moon.* The 4th stage will be fired at the best time to achieve a stable lunar orbit based on tracking and telemetry data from which the transfer orbit and the spacecraft spin-stabilized attitude have been determined.

*Reference 8

Section 4

GENERAL DESCRIPTION OF THE ANCHORED IMP REAL TIME COMPUTING SYSTEM

The Anchored IMP Real Time program will be modeled after the GSFC Real Time Programming System used in support of the Mercury and Gemini projects. This program will be designed to accommodate the asynchronous demands of incoming tracking data and control parameters and the transmission of output displays necessary to provide for continuing mission control.

The processors of the real time program will be divided into two major categories: monitor processors and computational processors. The monitor processors will contain all of the logic necessary to handle the program interrupts arising from real time data transfer and will assign priority to the computational processors of the system. The computational processors comprise the remainder of the routines in the system and are under the control of the real time monitor processors.

The computational processors will be designed to accommodate the requirements of the different mission phases. The programs for the launch phase of the mission will be able to accept IP 3600 vectors to drive launch displays and to obtain insertion vectors. The IP 3600 is the range safety computer at Cape Kennedy which processes powered flight data during the launch phase. The vectors which it computes are transmitted by high speed lines to GSFC where they are reformatted and used to generate additional data required for displays. The displays will include the current velocity, flight path angle, altitude, sub-vehicle position, and other parameters on the new multipurpose display unit. The launch phase programs will also accept high speed raw radar data from the C-band beacon on the 2nd stage of the Delta vehicle transmitted via Bermuda or via down-range ETR sites (eg., Antigua) in order to obtain insertion vectors and to predict 2nd stage impact. The launch programs will also include capability for variable azimuth from day to day for launch displays. The launch programs will accept telemetry signals for such event times as liftoff and 2nd stage cutoff. Manual intervention on telemetry signals will be included in case a telemetry signal is not received or is received garbled.

The real time computer program will be able to accept low-speed raw radar data in order to improve the estimate of the 2nd/3rd stage trajectory, and to use this orbit to obtain better predictions for 2nd stage impact. The pre-injection programs will use the corrected 2nd/3rd stage orbit and will add a nominal 3rd stage burn for injection determination. There will be a period of up to 20 minutes after injection into the transfer orbit during which aspect sensor telemetry data will be transmitted back to the tracking stations from the spacecraft. During this period, no range and range rate tracking data will be received at GSFC. Minitrack data may be used to obtain a fix on the orbit prior to range and range rate data transmission. (See Table 2-1 for the sequence of events during this period.)

The transfer phase will include routines to accept, validate, and edit the tracking data from the Goddard Range and Range Rate Network and from the Minitrack Network; to determine, predict, and correct the estimate for the transfer trajectory; to generate acquisition data for the supporting networks; and to generate displays of present trajectory conditions.

An 8th order central difference Gauss-Jackson numerical integrator will be used to generate the orbit. The equations of motion near the earth will include second, third, and fourth harmonic terms for the earth's potential, a drag deceleration term, a thrust term (to be used for the 3rd and 4th stage firings), and perturbation terms due to the sun and the moon. The moon will be treated as a triaxial ellipsoid. The integration will be done in an inertial frame referenced to the mean equator and equinox of 1950.0. All output during the transfer trajectory will be referenced to the true equator and equinox of date in a geocentric frame. The partial derivatives of the position and velocity vectors with respect to the initial position and velocity vectors which are needed for the differential correction processor will be obtained from the integration of the variational equations. Weighted least squares and/or Bayes techniques are to be included in the differential correction processor.*

As soon as a good determination of the transfer trajectory is achieved, the real time program will begin to compute possible 4th stage firing times. (See Figure 4-1.) First, if it is determined on the basis of the computed transfer trajectory that it is possible to achieve some lunar orbit, the computer programs will generate a series of plots and printouts to determine the best 4th stage firing time. These plots will include as a function of various firing times the following parameters:

a. Apocynthion
b. Pericynthion
c. Inclination to lunar equator
d. Orbital period
e. Shadow time
f. Lifetime
g. Apocynthion-moon-sun angle

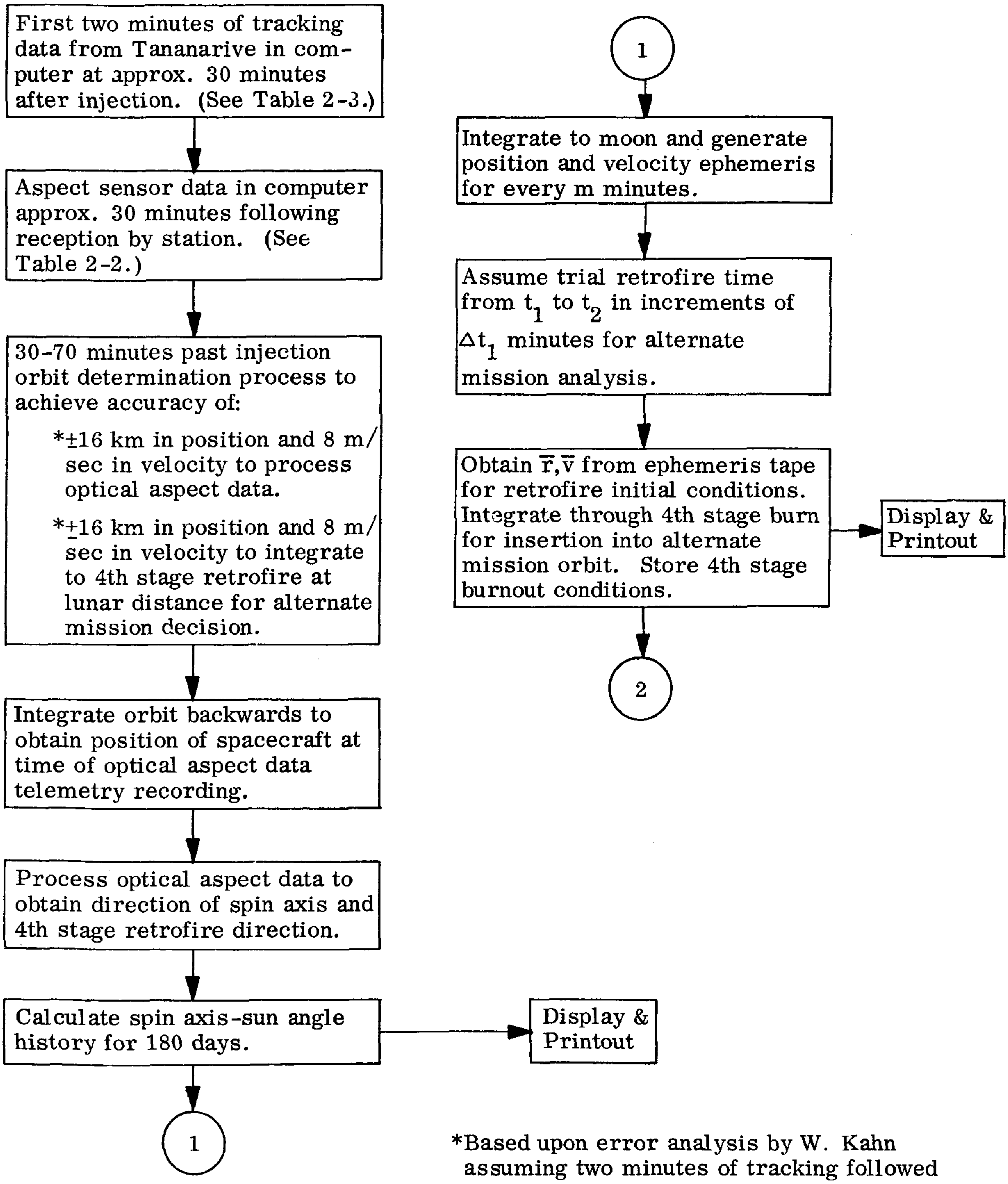
These parameters will be weighted and used by the computer to generate the optimum time of 4th stage firing, subject to manual override by the project officers. Second, if it is determined that a lunar orbit is not possible, the computer programs will go through an analysis to determine the time to fire the 4th stage to achieve the best possible alternate mission. Plots of the orbital parameters versus 4th stage firing times will be displayed.

Once the 4th stage has fired and the spacecraft has achieved its final orbit, the real time program will continue to process the orbit in a manner similar to that used for the transfer phase in that a continually updated estimate of the orbit will be maintained, associated displays will be driven, and acquisition data will be sent out to the supporting networks. After a satisfactory determination of the lunar orbit is achieved, the Data Operations Branch at GSFC will terminate continuous real time support, but will continue the orbit determination process

*Reference 9

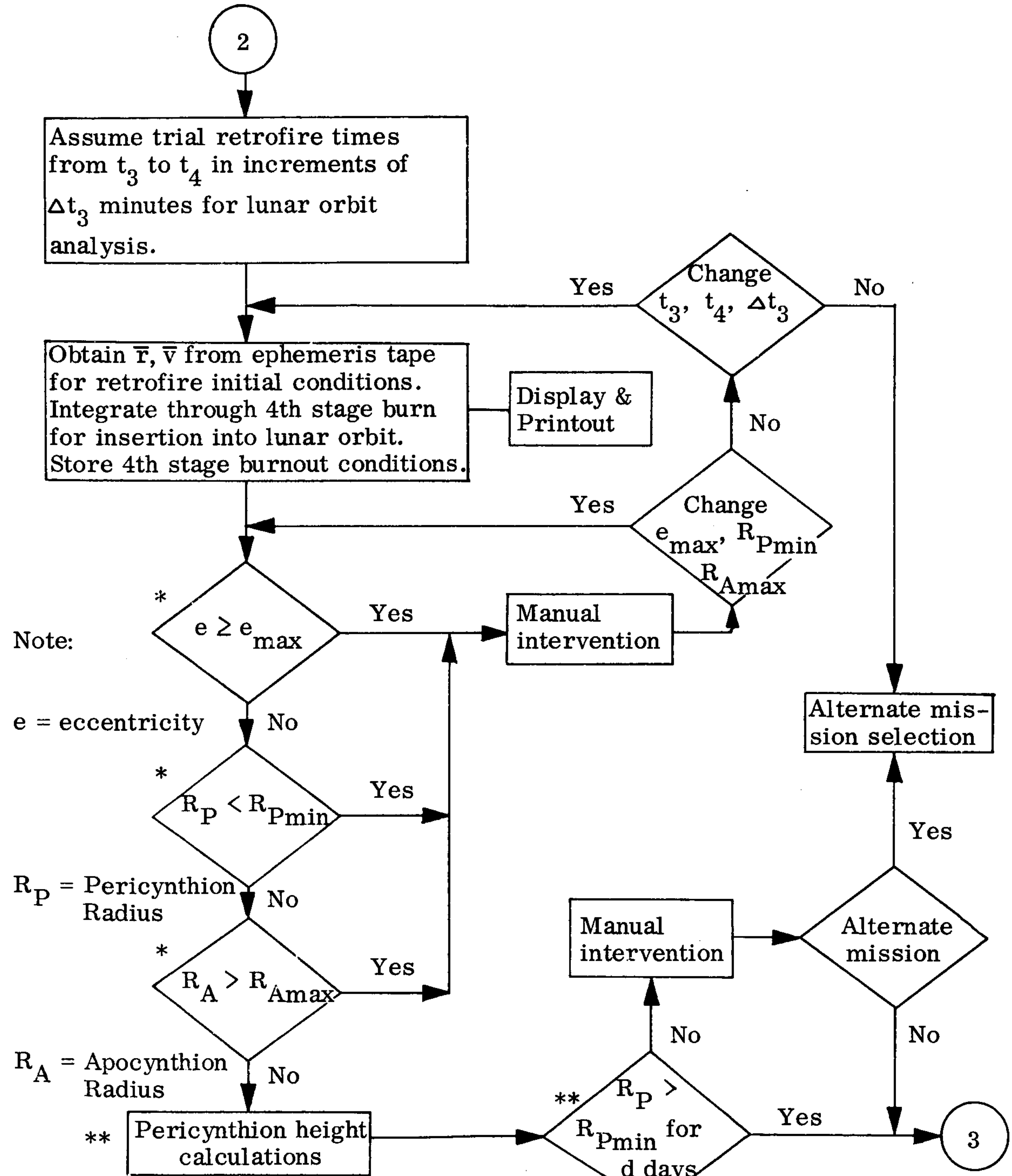
by periodically taking tracking data and updating the orbit. This process will continue for the lifetime of the AIMP mission. Orbit tapes for this phase of the mission will be produced and made available at periodic intervals to the experimenters in the various coordinate systems.

See Figure 4-1 for a pre-programming logic flow.



*Based upon error analysis by W. Kahn assuming two minutes of tracking followed by eight minutes of no track for a total of 30 minutes. (References 3 & 4)

Figure 4-1. Pre-Programming Logic Sequence (Sheet 1 of 3)



*Preliminary selection based upon initial conditions in lunar orbit, taking into account propagation of errors due to orbit determination and 4th stage firing errors. These criteria are tentative subject to further investigation.

**Intermediate selection based upon integration for 180 days in lunar orbit using simplified equations of motion.

Figure 4-1. Pre-Programming Logic Sequence (Sheet 2 of 3)

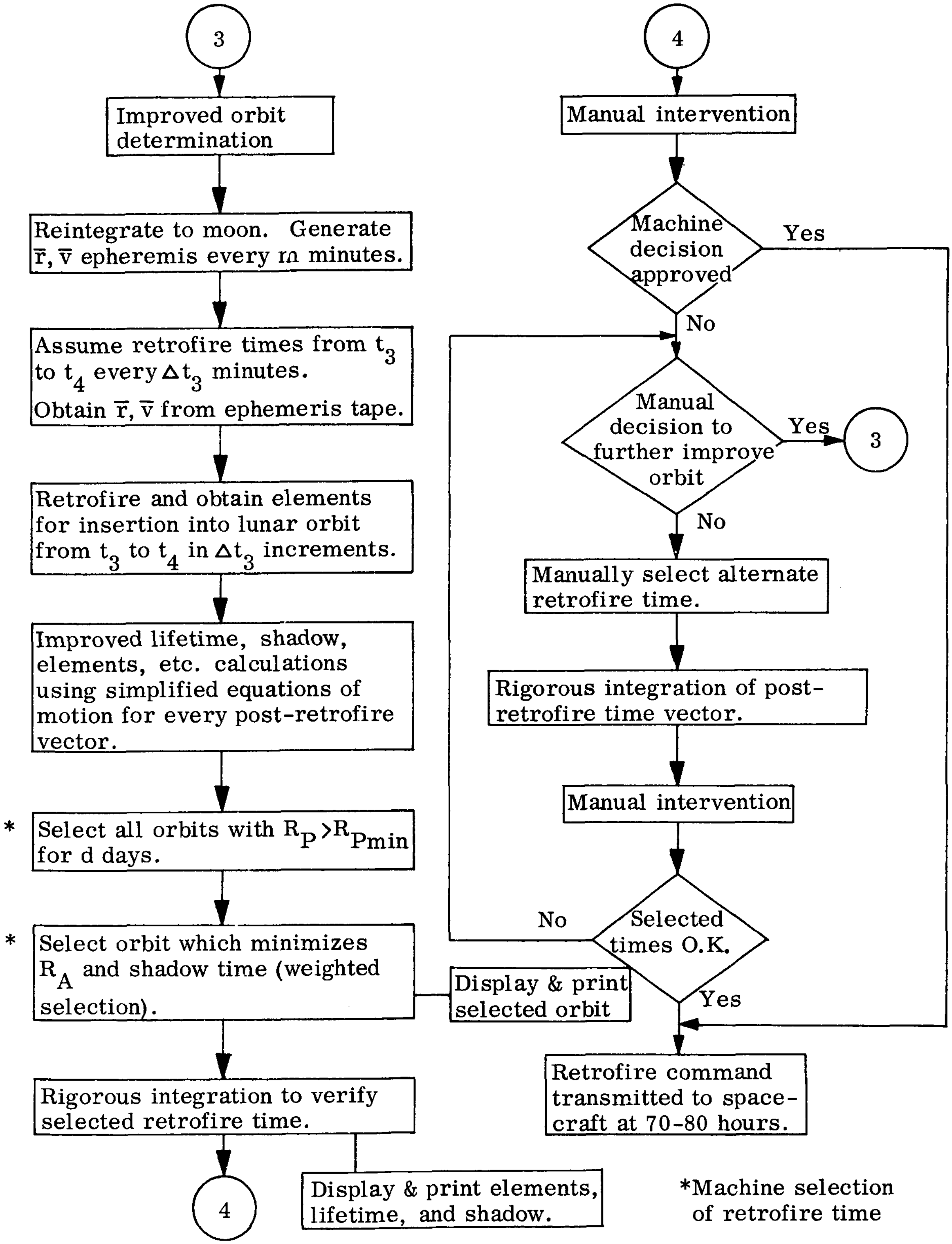


Figure 4-1. Pre-Programming Logic Sequence (Sheet 3 of 3)

Section 5

RESPONSIBILITIES

The Systems Analysis Office, as the cognizant representative of the Tracking and Data Systems Directorate in the computational area for the AIMP project, will provide the Data Operations Branch with all analysis and equations pertaining to lifetime and shadow calculations, alternate mission calculations, and 4th stage retrofire decisions in support of the AIMP mission.

The Data Operations Branch has the responsibility for analysis necessary for the orbit determination programs and supplemental analysis. The Data Operations Branch further has the responsibility for implementing and operating the Real Time Computer Program in support of the AIMP mission, including post-launch orbit determination for the duration of the AIMP mission.

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