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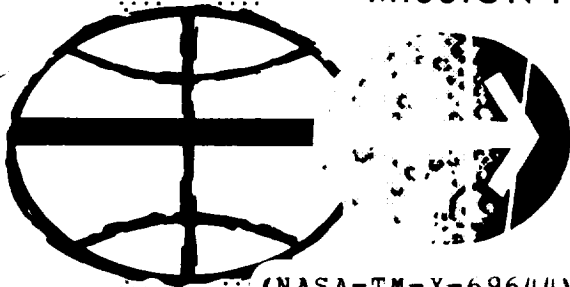
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LOGIC AND EQUATIONS FOR THE REAL-TIME CALCULATIONS OF THE POWERED DESCENT ABORT CONSTANTS

Orbital Mission Analysis Branch

MISSION PLANNING AND ANALYSIS DIVISION



MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

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PROJECT APOLLO

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By Ernest M. Fridge
Orbital Mission Analysis Branch

August 13, 1969

MISSION PLANNING AND ANALYSIS DIVISION
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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CONTENTS

Section		Page
1.0	SUMMARY AND INTRODUCTION	1
2.0	DEFINITIONS	2
2.1	Input	2
2.2	Output	3
2.3	Internal	3
3.0	GUIDANCE EQUATION CHANGES	5
4.0	USE OF EXISTING RTCC PROGRAMS	5
5.0	DESCRIPTION OF VARIABLE INSERTION TARGETING	6
6.0	METHOD OF COMPUTATION	7
	APPENDIX - POWERED DESCENT ABORT PROGRAM (PDAP)	9
	REFERENCES	21



LOGIC AND EQUATIONS FOR THE REAL-TIME CALCULATIONS OF THE
POWERED DESCENT ABORT CONSTANTS

By Ernest M. Fridge

1.0 SUMMARY AND INTRODUCTION

Changes in both the primary guidance and navigation control system (PGNCS) and the abort guidance system (AGS) logic for aborts from powered descent have established new requirements for Real-Time Computer Complex (RTCC) for Mission H and subsequent Apollo flights. Based on conditions at abort time and at the time of insertion into orbit, both guidance systems will generate the desired down-range component of the insertion velocity vector (\dot{Z}_D) for the powered ascent portion of the abort. This variable insertion target scheme results in a rendezvous with the command service module (CSM) which eliminates unnecessary phasing maneuvers, conserves ΔV , and approximates the nominal ccelliptic sequence time line defined for rendezvous from a nominal ascent. The calculation of \dot{Z}_D is a function of the desired semimajor axis (A_{ins}) of the insertion orbit. The A_{ins} is calculated empirically as a linear function of the phase angle θ between the vehicles. A minimum apolune radius restriction is required for the insertion orbit. The range of time for which this variable insertion target scheme is valid has been increased by addition of the capability to change the coelliptic sequence time line [i.e., slip terminal phase initiation (TPI) time approximately one revolution later] for those later aborts which produce a phase angle at abort time greater than the limit (θ_{LIM}). The RTCC requirements are to define the θ_{LIM} , the minimum apolune radius required (R_{amin}), and the two sets of coefficients required to generate A_{ins} . The first set of coefficients is used before TPI is slipped by θ_{LIM} considerations; the second set is used after TPI is slipped.

2.0 DEFINITIONS

2.1 INPUT

State vectors for both vehicles must be input. The LM state vector must be defined after the DOI maneuver.

All ascent guidance targets except \dot{Z}_D must be input.

GUIDDRN	Δt referenced from T_{LAND} used to start engine-ON-algorithm iteration for PDI; i.e., first guess at $PDI = T_{LAND} - GUIDDRN$
h_{amin}	minimum apogee altitude limit for the insertion orbit; referenced from the landing site radius
Δh_D	desired altitude differential between the LM and CSM orbits at CDH
h_{DINS}	desired insertion altitude referenced from landing site radius
h_{LS}	landing site altitude referenced from the mean lunar radius
λ	longitude of the landing site
μ	lunar gravitational constant
ϕ	latitude of the landing site
R_{MOON}	mean lunar radius
t_{LAND}	nominal landing time
t_{UL}	Δt between ullage ignition and PDI
Δt_{CAN}	Δt between orbit insertion and the canned maneuver
Δt_{CSI}	Δt between orbit insertion and CSI
Δt_{STAGE}	Δt between PDI and staging time
Δt_{STEP}	Δt between successive abort points
Δt_{2CSI}	Δt added to Δt_{CSI} for generation of the second set of targeting coefficients

Δt_{2TPI} Δt added to t_{TPI} for generating of the second set of targeting coefficients
 ΔV_{CAN} ΔV of the canned maneuver
 W_{TAPS} LM vehicle weight immediately after staging
 W_{TDRY} LM weight representative of DPS fuel depletion

2.2 OUTPUT

J } coefficients of the equation $A_{ins} = f(\theta)$
 K }
 θ_{LIM} phase angle at the orbit time which results in insertion into the minimum apolune orbit; the phase angle at each abort time is tested against θ_{LIM} to determine which target coefficients will be used
 R_{amin} minimum apolune radius permitted in the insertion orbit

2.3 INTERNAL

A_{ins} semimajor axis of the insertion orbit
 A_T values of A_{ins} stored in the A_{ins}, θ table
 ha apogee altitude referenced from the landing site
 Δh differential altitude between LM and CSM orbits at CDH
 $\Delta h'$ value of Δh on previous iteration
 I_s counter that identifies the number of entries in the A_{ins}, θ table
 K_3 flag that indicates which set of J,K is being determined; $K_3 \leq 0$ indicates that the first set is being determined
 K_{LOOP} iteration counter
 K_{STAGE} if value is ≤ 0 , indicates staging has not occurred; if value is >0 , indicates staging has occurred

Ra	apolune radius magnitude
Ra'	apolune radius magnitude at previous abort point
R _D	desired insertion radius
R _{ins}	actual insertion radius
\bar{R}_{LM}	LM state vector
RM _{LS}	landing site radius
\bar{R}_{LSSG}	landing site vector in moon-fixed coordinates
SLOPE	slope of the desired down-range component of insertion velocity vector as a function of Δh
t	state vector time
T	thrust magnitude
t _{CSI}	time of the CSI maneuver
t _D	state vector time during the powered descent maneuver
T	thrust level
T _{DPS}	commanded DPS thrust level
θ	phase angle
θ_D	phase angle at the current abort point
θ'	phase angle at the previous abort point
θ_T	phase angle angles stored in the A _{ins} , θ table
t _{GO}	time - to - go until orbit insertion if ascent guidance is being used; time - to - go until end of current descent phase when descent guidance is being used
t _{ins}	time of insertion into orbit after the abort maneuver
t _{STAGE}	time at which staging occurs
t _{TPI}	time of the TPI maneuver
Δt_{ABORT}	Δt from PDI to the current abort point
\bar{U}_T	current unit thrust vector

\bar{U}_T'	unit thrust vector calculated on previous guidance cycle
ΔV_{INC}	ΔV increment calculated to improve the value of \dot{Z}_D at each abort point
W_{TA}	LM weight during ascent
W_{TD}	LM weight during descent
\dot{Z}_D	desired down-range component of the insertion velocity vector

3.0 GUIDANCE EQUATION CHANGES

Recent guidance requirements for the PGNCS update the equation

$$\dot{Z}_D = \sqrt{u \left[\frac{2R_a}{R_p (R_a + R_p)} \right]}$$

continuously every 2 seconds during the powered ascent (ref. 1).

For this equation it is assumed that the predicted insertion radius (R_p) will be at perilune. However, perilune radius actually will be slightly less because the velocity target at insertion has a component along the radius vector. This r component is established by input and is not calculated by the variable insertion scheme. R_p is updated every 2 seconds, as shown in reference 1. The insertion orbit apolune radius (R_a) is calculated from the equation $R_a = 2A_{ins} - R_p$; R_a limited to be greater than or equal to R_{amin} where $A_{ins} = J + K\theta$; θ is positive when the CSM is ahead of the lunar module (LM); J and K are derived from a least squares curve fit of the table of A_{ins} and θ .

4.0 USE OF EXISTING RTCC PROGRAMS

Most of the logic presented in this paper and in the appendix makes use of computational schemes which have been previously documented or are well known. Thus, although most of the computations required are performed by the descent guidance (ref. 2), the ascent guidance (ref. 1),

and the coelliptic sequence (ref. 3), no attempt is made to explain these computations because they are well documented. The integration to be used should be one that propagates by use of the R2 potential model. A least squares curve fit routine is required for generation of the coefficients to calculate A_{ins} from a table of A_{ins} and θ defined at various abort points. Thus, the logic described in the appendix is mainly an iteration scheme which defines the interface needed between existant RTCC programs.

5.0 DESCRIPTION OF VARIABLE INSERTION TARGETING

The coelliptic sequence rendezvous time line used for aborts is defined and explained in reference 4. It follows very closely the nominal coelliptic sequence time line. The coelliptic sequence initiation (CSI) maneuver occurs at fixed Δt (currently 50 min) after orbit insertion; the constant differential height (CDH) maneuver occurs $\frac{N}{2}$ orbits after CSI; and TPI occurs either at a specified time or at a specified Δt before the sunrise which occurs after one revolution of the LM in the insertion orbit. The Δh desired at CDH and the elevation angle desired at TPI are entered as input. These conditions largely determine the phase angle required at CSI. If TPI is slipped forward one revolution, the phase angle required at CSI can be increased.

Variable insertion targeting sets up the desired phasing at CSI. Thus, the larger the phase angle (positive when the CSM is ahead) at insertion, the smaller the semimajor axis of the insertion orbit must be to allow a faster catchup rate. Because the phase angle increases as the LM is slowed down by the powered descent maneuver and because the LM targets for the same insertion altitude regardless of the abort time, the insertion orbit apolune decreases the longer the abort is from powered descent initiation (PDI). Thus, if abort occurs at approximately PDI plus 600 seconds, the desired insertion orbit will have the minimum allowable apolune radius. The phase angle at the abort time which produces this minimum orbit appears in the output as θ_{LIM} .

To increase the time span over which variable insertion targeting is valid, aborts which occur after this time will target for a coelliptic sequence during which TPI is performed approximately one revolution later. Because this new targeting increases the phase angle desired at CSI, the insertion orbit can once again be large to allow for the slower catchup requirements. However, when the apolune of the insertion orbit once again reaches the minimum, the process is stopped.

The output of this abort program will be the two sets of target parameters and the limiting values. One set of coefficients (J,K) for the A_{ins} equation will be generated for those early aborts whose phase angle at abort time is less than θ_{LIM} . Because the phase angle decreases continuously after PDI, θ_{LIM} is tested rather than abort time. The second set of coefficients is used for aborts after θ_{LIM} is passed. R_{amin} is the other limiting value; it is specified by input.

6.0 METHOD OF COMPUTATION

The procedure used to generate θ_{LIM} and the coefficients is to compile two tables of A_{ins} and θ and then to find a least squares curve fit for each table which solves for A_{ins} as a linear function of phase angle.

Abort points are taken every Δt seconds (currently 20 sec) along the powered descent trajectory. At these points, an abort maneuver is performed.

Two options exist for performance of aborts. Nominally, the ascent trajectory is flown as far as possible with the DPS engine. Vehicle weight is tested every 2 seconds; and when it falls below a specified value, which is the weight that represents DPS fuel depletion (i.e., total LM weight minus DPS usable propellant and consumables used prior to PDI), staging is performed, and the abort is completed with the APS engine. However, the option exists to perform abort stage (stage at the time of abort) and to perform the entire ascent portion of the abort with the APS engine. The choice of options affects the targets generated because the nominal route which stages according to vehicle weight criterion does not insert at the desired altitude because of jerk (the derivative of acceleration) limitations in the ascent guidance equations.

After insertion, the coelliptic sequence logic is processed to determine the Δh at CDH. An iteration loop, which includes the ascent trajectory and the coelliptic sequence, is cycled until the insertion conditions that produce the desired Δh at CDH are found. The A_{ins} and θ for this point are stored in the table. Each abort point along the descent trajectory is processed in this manner until the point is reached which produces an orbit at insertion whose R_a is below R_{amin} . All points stored in the table prior to this point are processed by the least squares

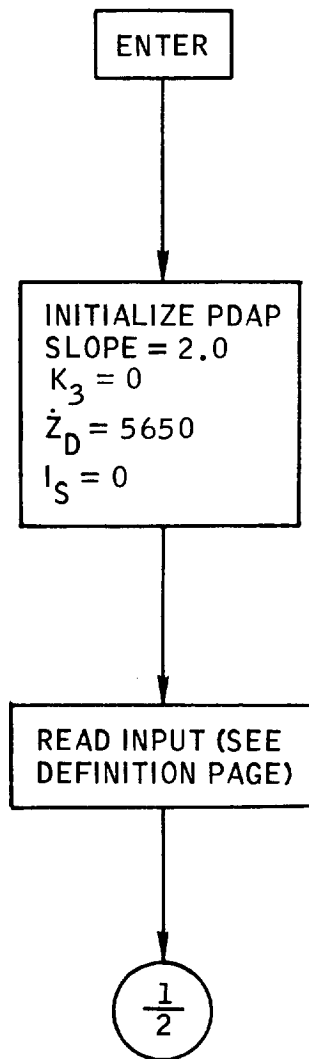
curve fit to determine J_1 and K_1 . Also, a linear interpolation is performed to determine θ_{LIM} . It will occur between the last point in the table and this last processed point.

The abort point mentioned above which produced an insertion orbit whose R_a is below R_{amin} is used as the initial point for the calculation of J_2 and K_2 . The coelliptic sequence is adjusted for this set by a forward slippage of both TPI and CSI. The amount of time slippage for both maneuvers is determined by input. The option exists to perform a canned horizontal, inplane maneuver between insertion and CSI if desired. The J_2 and K_2 are then generated in a manner analogous to generation of the first set. However, the second set will cover a large period of time during which the LM is on the lunar surface. During this period, therefore, the descent guidance will not be used, and the vehicle will be staged at abort time.

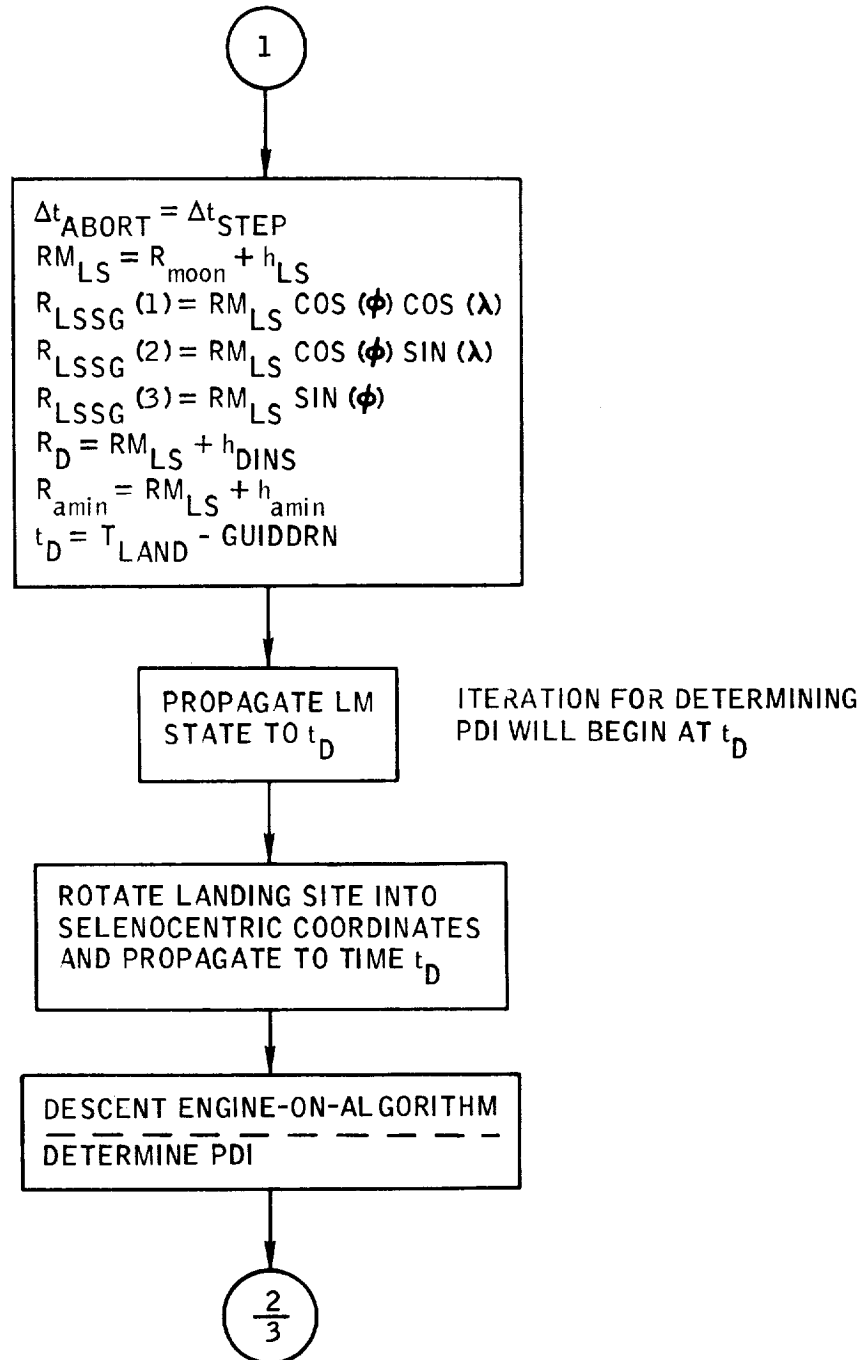
APPENDIX

POWERED DESCENT ABORT PROGRAM (PDAP)

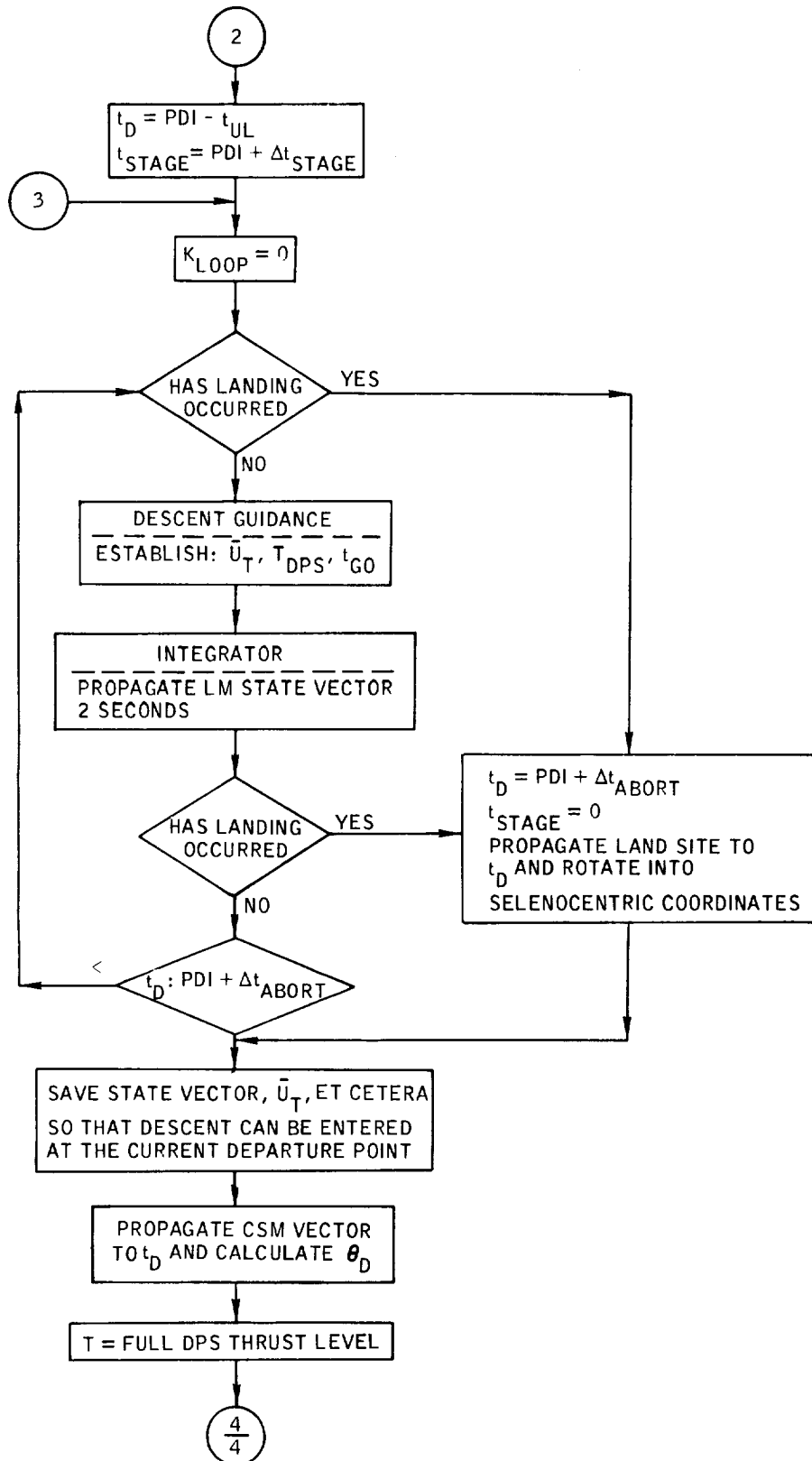




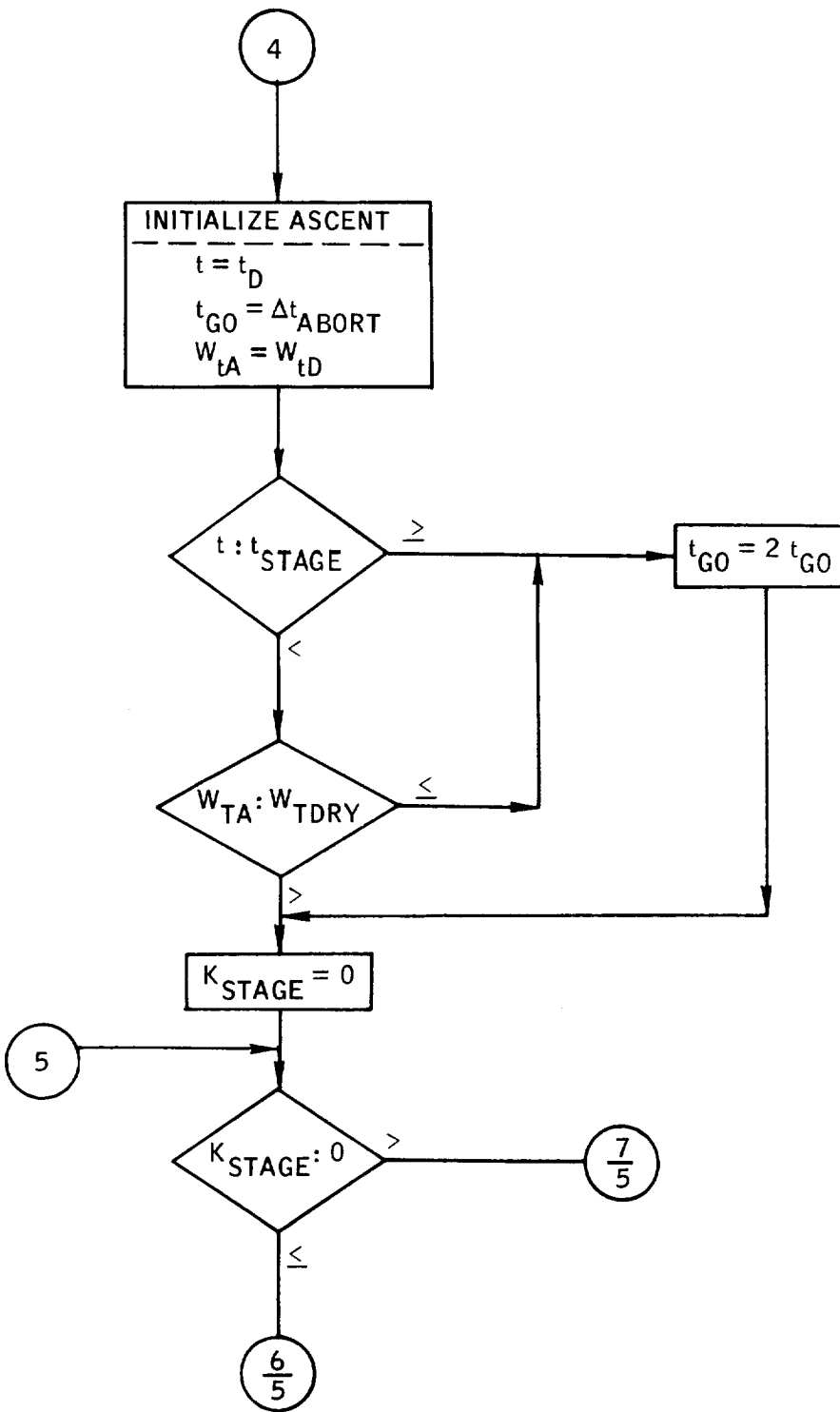
Flow chart 1.- Powered descent abort program.



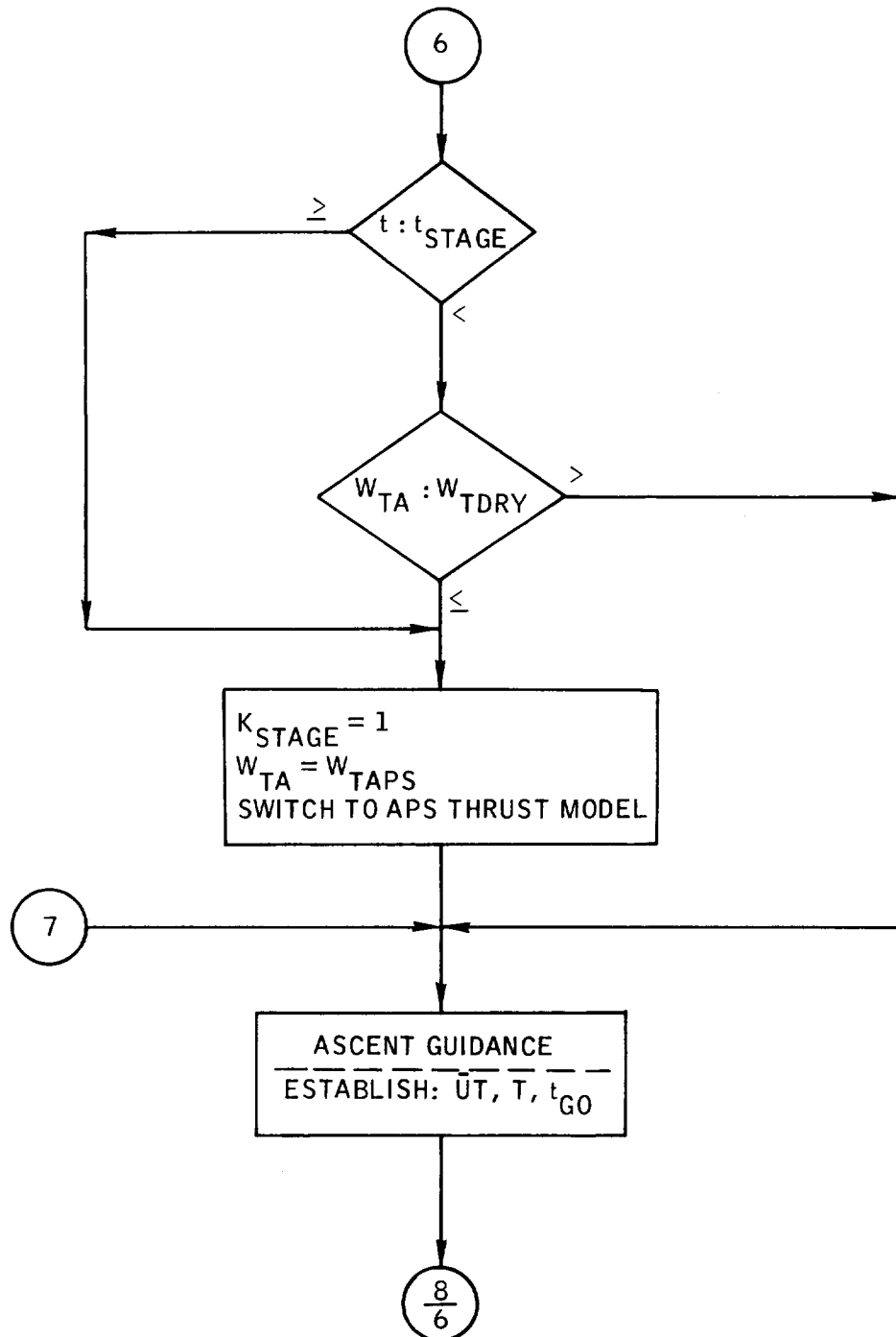
Flow chart 1. - Powered descent abort program - Continued.



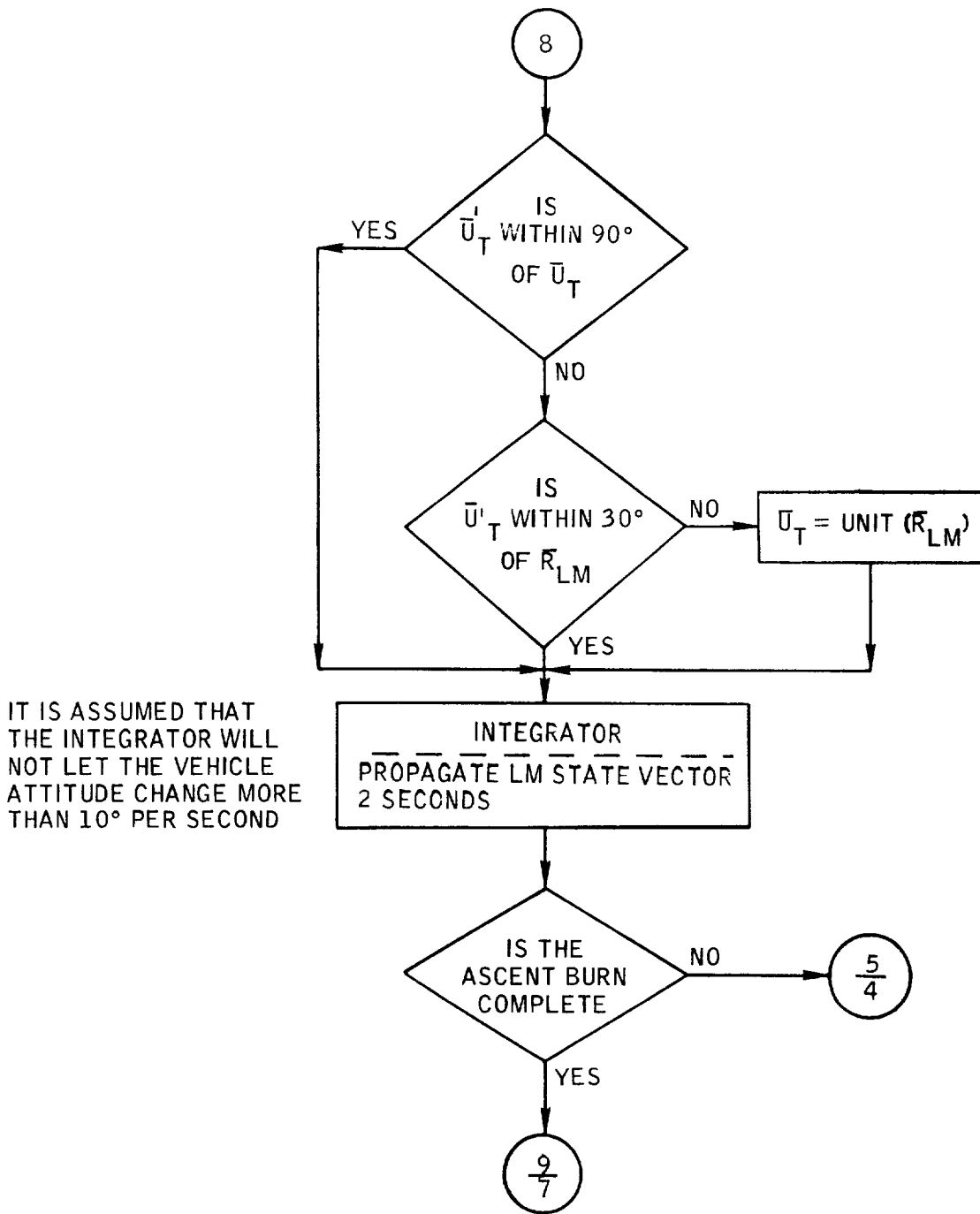
Flow chart 1. - Powered descent abort program - Continued.



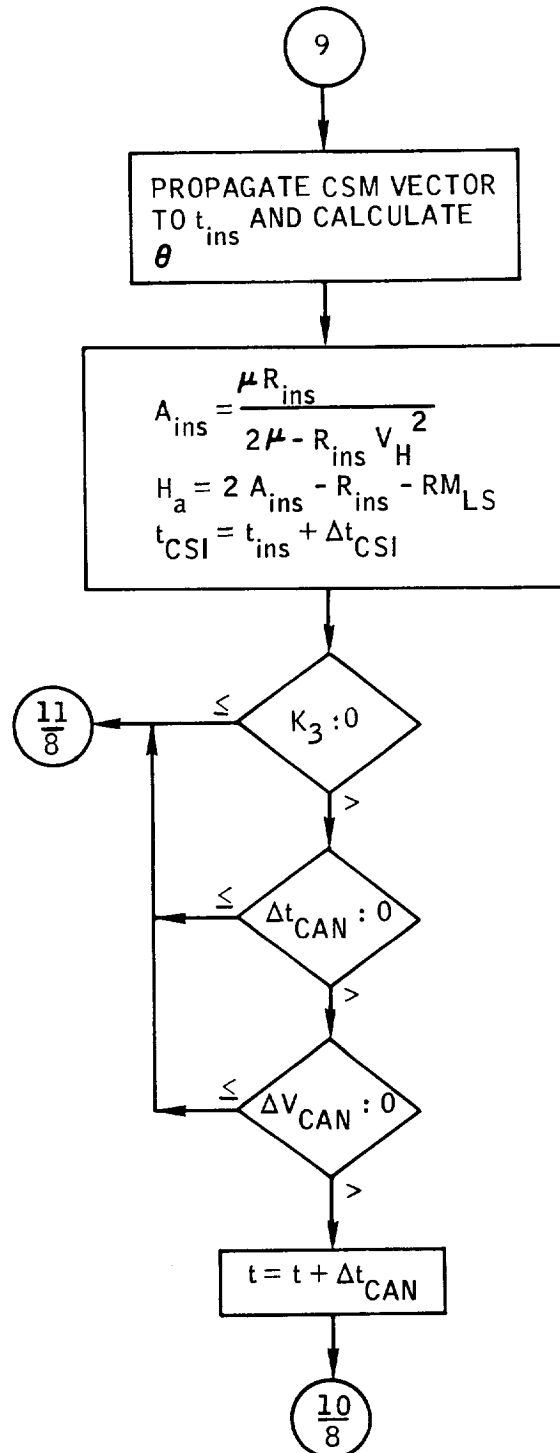
Flow chart 1.- Powered descent abort program - Continued.



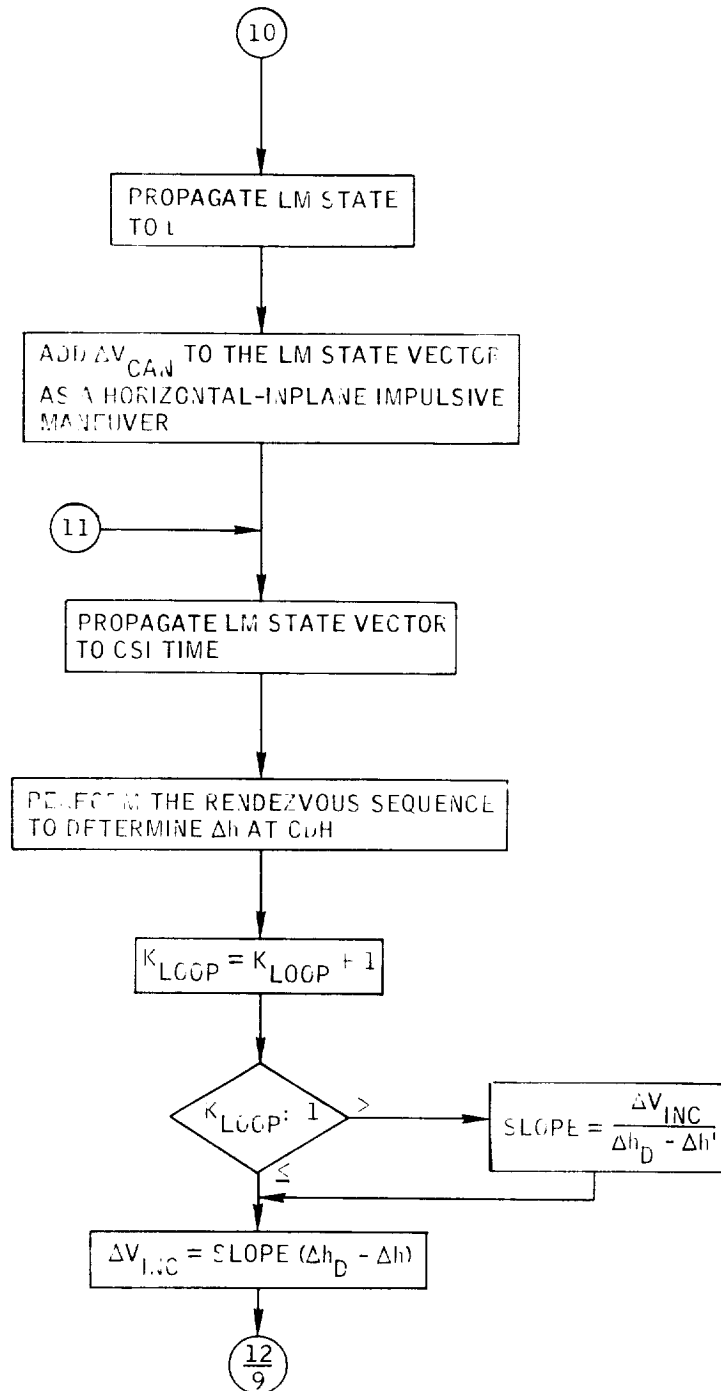
Flow chart 1.- Powered descent abort program - Continued.



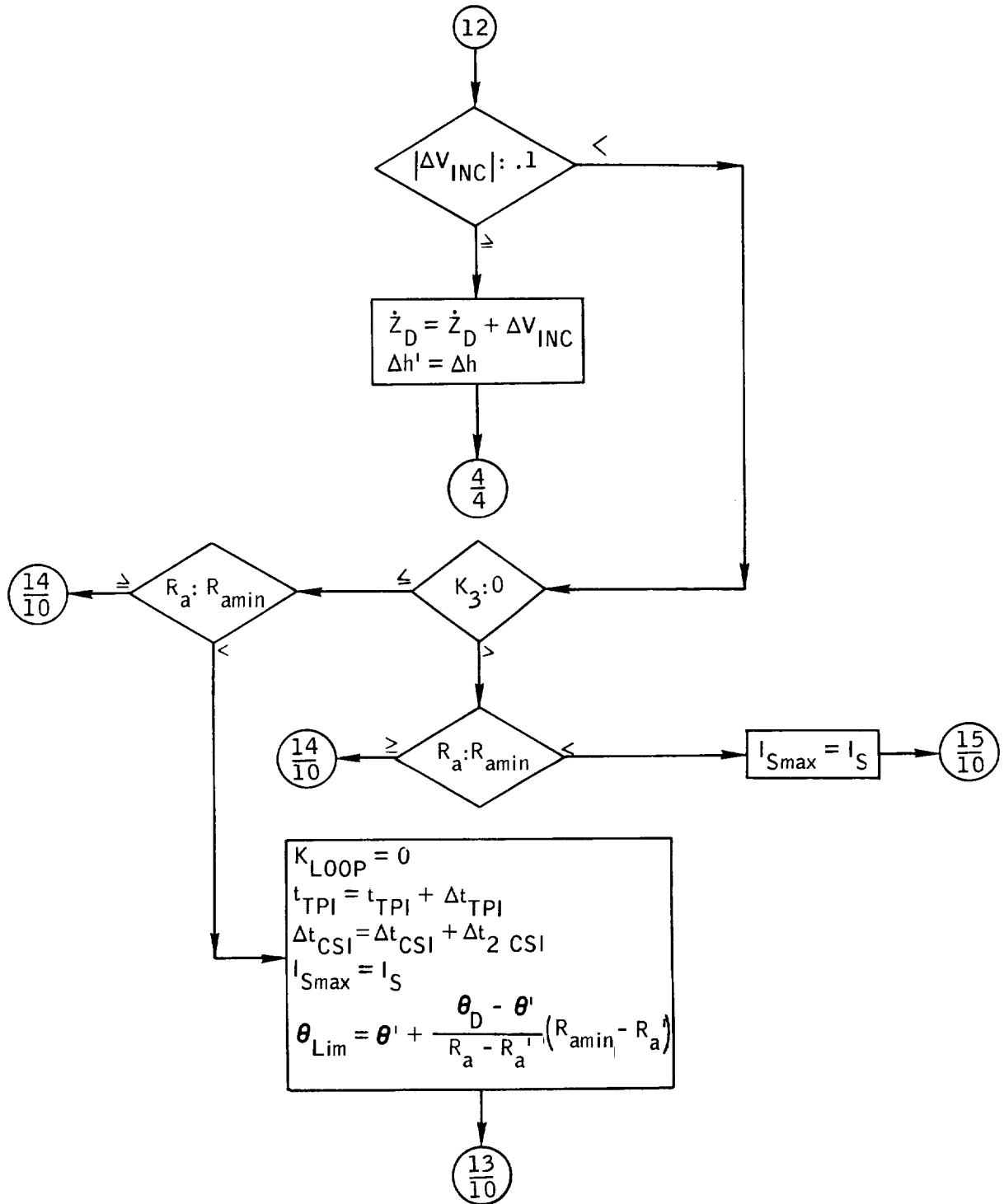
Flow chart 1.- Powered descent abort program - Continued.



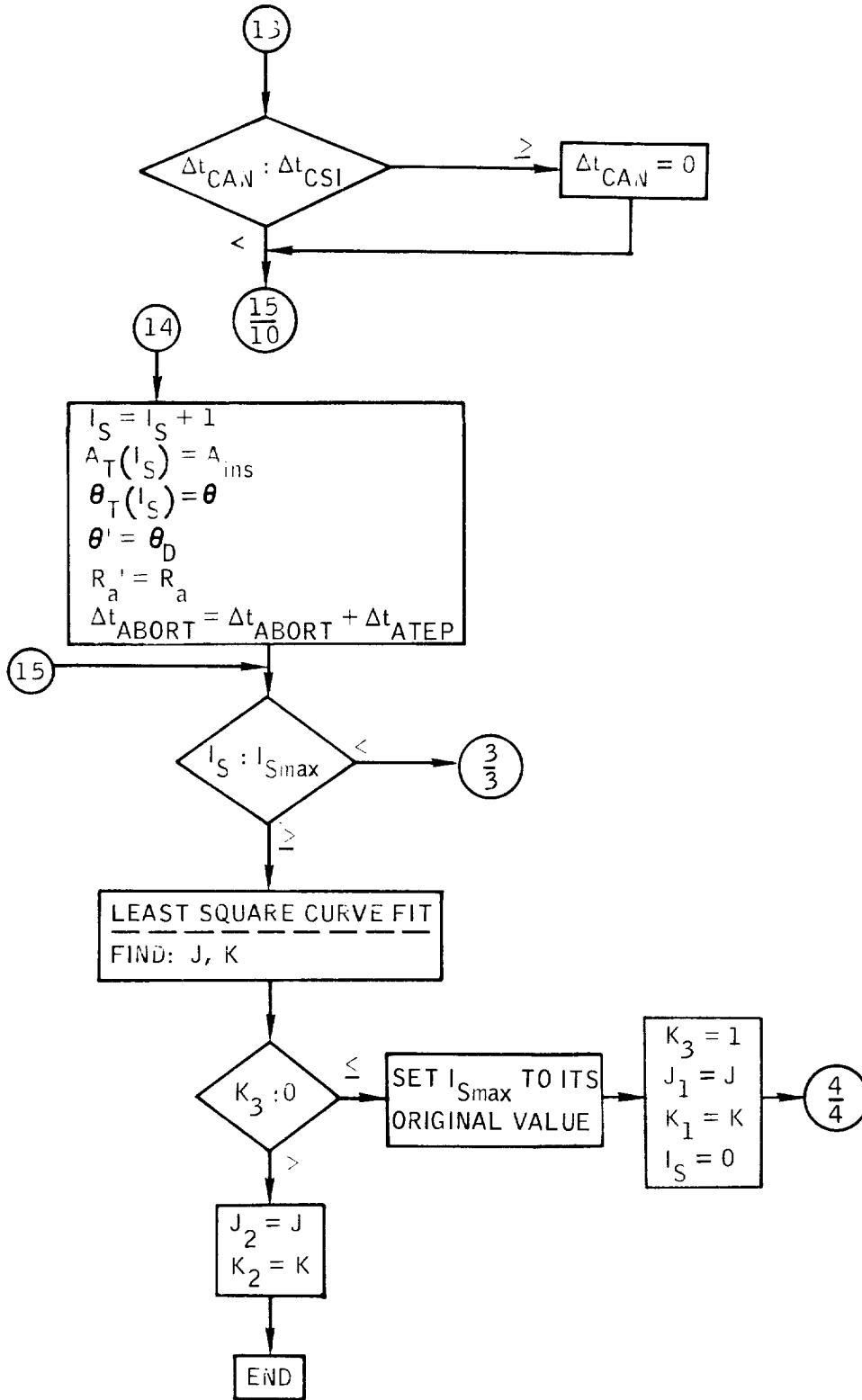
Flow chart 1. - Powered descent abort program - Continued.



Flow chart 1. - Powered descent abort program - Continued.



Flow chart 1.- Powered descent abort program - Continued.



Flow chart 1. - Powered descent abort program - Concluded.

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