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NASA SPACE SHUTTLE PROGRAM WORKING PAPER

SPACE SHUTTLE GUIDANCE, NAVIGATION
AND CONTROL
DESIGN EQUATIONS

VOLUME II
PREFLIGHT THROUGH ORBIT INSERTION

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PURPOSE

The purpose of this document is to specify the equations necessary to perform the guidance, navigation and control onboard computation functions for the space shuttle orbiter vehicle. This equations document will: (1) establish more specifically, than on a functional level, the GN&C computational requirements for computer sizing, (2) provide GN&C design equations specification to develop demonstration software for hardware feasibility testing, and (3) define the hardware interface requirements with the GN&C subsystem software. The document will provide a standard of communication of information concerning the GN&C equations, and will provide a means of coordination of GN&C equation development.

SCOPE

This is Volume II of the document which defines the Guidance, Navigation and Control (GN&C) design equations sequencing and interfaces for the computations required in the GN&C Subsystem for all mission phases of the Shuttle Orbiter flight. The equations are intended to satisfy the functional requirements specified in Reference a. This document will describe in mathematical, logical, and operational language all the details necessary to initiate and carry out the design of the required computer modules (subprograms) for the GN&C functions.

The document will be organized into six volumes. Volume I contains Sections 1 through 8, which provide introductory information for the document. Volume II contains the current detailed equations for the preflight, boost, separation, orbit insertion and ascent abort phases of the Orbiter operation. Detailed equations for orbital operations of the Orbiter, which include the orbital coast, orbital powered flight, rendezvous, station keeping, docking and undocking, and docked operations phases, are presented in Volume III. Volume IV contains the current detailed equations for the deorbit and entry, transition, cruise and ferry, approach and landing, and horizontal takeoff phases of the Orbiter. Also, this volume will contain the equations for communications and pointing functions and the failure detection function. For this issue of the document, only Volumes I through IV are being published. In future issues, Volume V will contain the detailed flow diagrams for the equations. For the initial issue, the flow diagrams for the approved equations are included with the equations in Volumes II, III or IV. For future issues, the constants used in the equations will be summarized in Section 11 and the GN&C parameters and variables which can be entered or called via the keyboard will be enumerated in Section 12. These two sections will be contained in Volume VI of the document.

APPLICABILITY

This document is applicable to the Guidance, Navigation, and Control (GN&C) Subsystem of the Electronics System of the Space Shuttle Orbiter Vehicle. It is applicable to the definition of the shuttle computational requirements for the subsystem listed above. It is applicable to the Phase B and Phase C subsystem development. It defines the Manned Spacecraft Center Guidance and Control Division inhouse study baseline equations design.

9. DESCRIPTIONS OF EQUATIONS

The detailed equations for the GN&C functions are defined in this section. The organization of this section is tentative and will be modified so as to present the equations as they are designed in as clear a fashion as possible. As an introduction to each major subsection (usually a mission phase), the general GN&C software functions to be implemented will be identified and, where appropriate, a conceptual discussion and top level flow of the computations, inputs and outputs will be included in order to understand and summarize what is to be covered. This should be an order of magnitude less detailed than the flow diagrams of the equations which come later.

A GN&C Equation Submittal sheet will introduce each of the GN&C equation submittals and summarize the GN&C functions, and identify the source and NASA contact for each.

The detailed data to be presented for each GN&C function within each of the major subsections (usually a mission phase) is summarized below. Although items 6 through 10 are to be referenced only in the equations document, they are required submittals before the equations can be approved and finalized for flight software development.

1. Functional Requirements

The specific functional requirements (from the GN&C Software Functional Requirements Document) which are satisfied by the equations should be identified.

2. Functional Diagram

A brief functional explanation and description of the overall concept and approach. A functional block diagram should be used where clarity is enhanced. Inputs, outputs, and interfaces will be provided.

3. Equations and Flows

Detailed equations and a descriptive text which guides the reader through the flows of Section 10 should be provided. The minimum frequency of the computations shall be specified and rationale given or referenced.

4. Coordinate System

The coordinate systems used shall be defined.

5. Constants/Variables Summary

Constants and variables shall be summarized in tabular form with the following information:

- a. Variables/constants symbols and definitions
- b. Units
- c. Allowable quantization
- d. Range of values

6. FORTRAN Coding

The FORTRAN coding of the function for verification using the Space Shuttle Flight Simulation (SSFS) will be referenced.

7. Simulation

The SSFS specifications, description and user's guide used to verify each GN&C function will be referenced.

8. Testing

Test plans and test results will be referenced.

9. Derivation

The mathematical derivation of the equations including all mathematical assumptions shall be referenced.

10. Assumptions

The following will be referenced:

- a. Avionics baseline system assumed
- b. Reference missions assumed
- c. Vehicle mass properties assumed
- d. Propulsion models assumed
- e. Environment models assumed
- f. Error models assumed

The major subsections of this section are identified and partially expanded in the following.

9.1 PREFLIGHT

The following GN&C software functions are envisioned for this mission phase:

- 1) Generate a reference mission profile. Specifically, perform targeting for entire mission during the prelaunch phase to insure that mission objectives will be met. Target the launch phase to insure that launch abort targeting, rendezvous targeting, and deorbit targeting solutions are consistent with vehicle consumables, characteristics, etc., and ground related activities and locations.
- 2) Schedule sensor calibrations (if required in flight) consistent with sensor status and mission plan.
- 3) Schedule sensor alignments consistent with sensor status and mission plan .
- 4) Calculate initial inertial state prior to launch.
- 5) Transform launch guidance target parameters to proper time dependent form for launch.
- 6) Navigate vehicle and target to launch time.
- 7) Perform prelaunch checkout of all control hardware using autopilot and manual controller inputs.
- 8) Perform automatic calibration of sensors (i.e., accelerometers and gyros) or use predicted values based on time histories and compute compensation values prior to launch.
- 9) Provide automatic inertial reference alignment prior to launch.

9.1.1 GN&C System Initialization, Monitor, Test and Checkout (TBD)

9.1.2 Prelaunch Targeting (TBD)

9.1.3 Sensor Calibration and Alignment (TBD)

9.2 BOOST PHASE PROGRAM

The Orbiter Boost Phase Program will monitor the Booster performance during the first-stage boost period. State vector and attitude data will be maintained and updated periodically so that the Orbiter will be ready for staging at any time. Figure 1 represents a functional flow of guidance, navigation and control computations during Boost. Redundancy internal to the orbiter is undefined at this time and is not shown. Navigation and Guidance computations will be performed throughout the Boost Phase in order to monitor the behavior of the vehicle, provide data to the abort decision process, and to prepare the orbiter for staging and orbit insertion.

The guidance computations will include steering, throttle and engine cut-off time computations, although no takeover of Booster control is currently anticipated. The equations used for the guidance functions would be used only to detect malfunctions and would necessarily be the same equations used by the Booster.

The Boost Phase Program would be terminated at staging and replaced by the Orbit Insertion Program or, following an abort decision, by the Ascent Abort Program.

9.2 BOOST PHASE PROGRAM (continued)

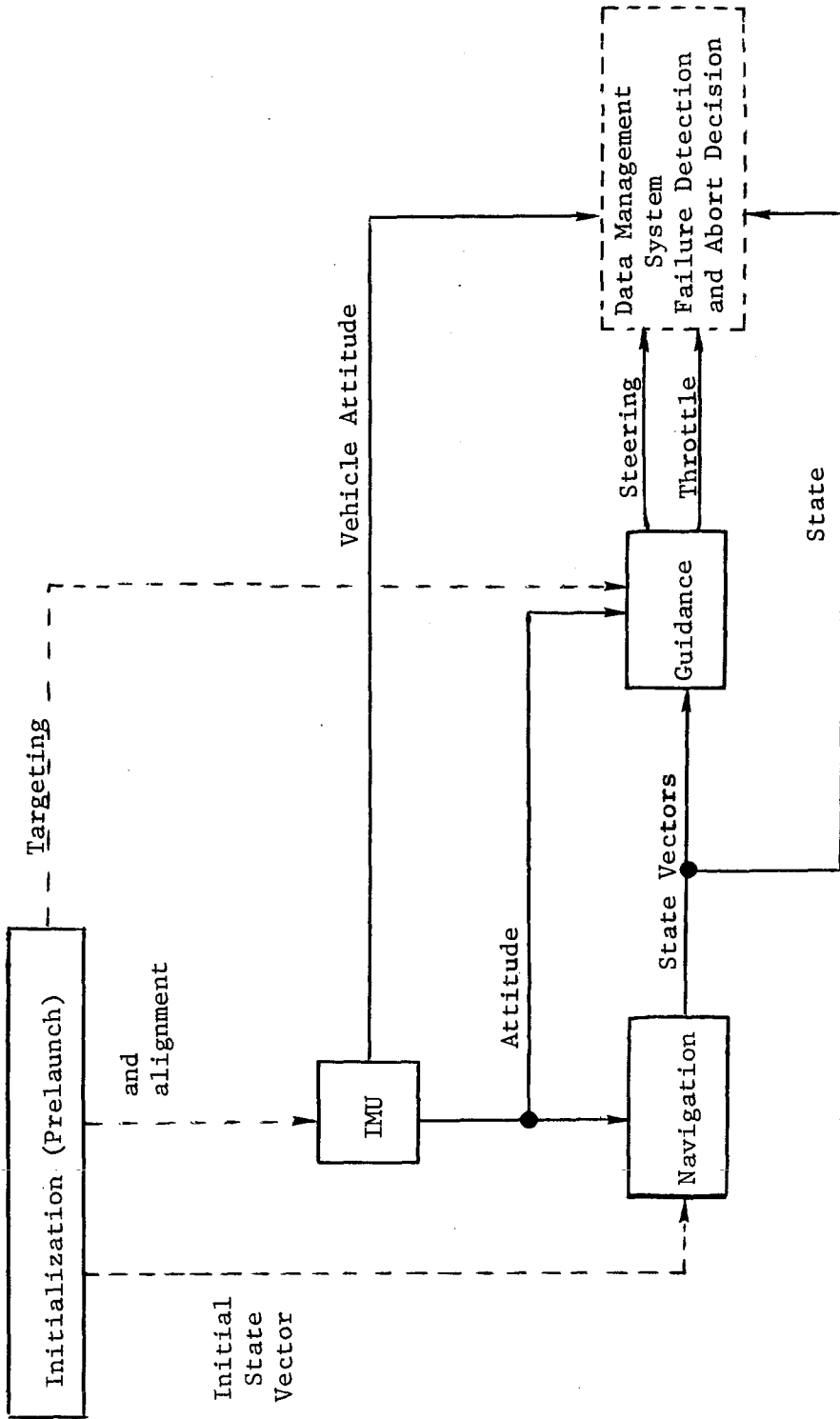


Figure 1 - Functional Flow - Boost Phase

9.2 BOOST PHASE PROGRAM (continued)

SPACE SHUTTLE

GN&C SOFTWARE EQUATION SUBMITTAL

Software Equation Section: Rapid, Real-Time State Advancement During
Specific Force Sensing

Submittal No. 19

Function: Powered Flight Navigation

Module No. ON1 Function No. 1 (MSC 03690)

Submitted by: W. M. Robertson Co. MIT No. 5-71

Date: Feb 1971

NASA Contact: J. Suddath Organization: GCD

Approved by Panel III: K.J. Col Date: 3/10/71

Summary Description: Provides the capability to quickly and continuously
update the spacecraft state vector in real time, incorporating into the
state update the real-time measurements from the accelerometers.

Shuttle Configuration: Not provided explicitly. However, ΔV 's from
inertially stable accelerometers are implied.

Comments:

(Design Status) _____

(Verification Status) _____

Panel Comments:

9.2.1 Rapid Real-Time State Advancement During Specific Force Sensing

1. INTRODUCTION

The routine for Rapid Real-Time State Advancement during Specific Force Sensing provides the capability to quickly and continuously update the spacecraft state vector in real-time (especially during periods of high computer activity), incorporating into the state update the real-time measurements which are being provided by the spacecraft's accelerometers relative to the specific (non-free-fall) forces to which the spacecraft is being subjected, such as thrust and aerodynamic effects.

The routine is merely a low order numerical integration of the differential equation satisfied by the spacecraft state vector in which only the central force field term and the largest earth oblateness term (J_2) are included, but in which provision is made to incorporate the specific force data from the accelerometers.

9.2.1 Rapid Real-Time State Advancement During Specific Force Sensing (continued)

NOMENCLATURE

$\cos \phi$	Cosine of colatitude of spacecraft position
J_2	Earth Oblateness Constant
$\underline{g}(t)$	Gravity vector at time t
r	Magnitude of \underline{r}
$\underline{r}(t)$	Spacecraft position vector at time t
r_E	Earth's mean equatorial radius
$\underline{v}(t)$	Spacecraft velocity vector at time t
Δt	Time interval
$\Delta \underline{v}_{\text{meas}}(\Delta t)$	Velocity vector increment measured by the integrating accelerometers during the time-interval Δt
μ_E	Earth's gravitational parameter
$\underline{1}_r$	Unit vector in the direction of \underline{r}
$\underline{1}_z$	Unit vector in direction of earth's north polar axis expressed in reference coordinates

9.2.1. Rapid Real-Time State Advancement During Specific Force Sensing (continued)

2. FUNCTIONAL FLOW DIAGRAM

The general functions performed by the routine are outlined in Fig. 1.

The routine is called repetitively during specific force sensing periods once every Δt seconds, where the time-interval Δt is chosen so that the procedure will be compatible with the basic guidance and control computation cycles.

Because the accelerometers are of the integrating type, the accelerometer data is output in the form of velocity increments (Δv_{meas}) over the repetitive computation time interval Δt . The computations of this routine are therefore in terms of increments of velocity rather than instantaneous accelerations. The increments are initially sensed in Inertial Measurement Unit or Stable Member Coordinates but are transformed into reference coordinates before use in this routine.

9.2.1 Rapid Real-Time State Advancement During
Specific Force Sensing (continued)

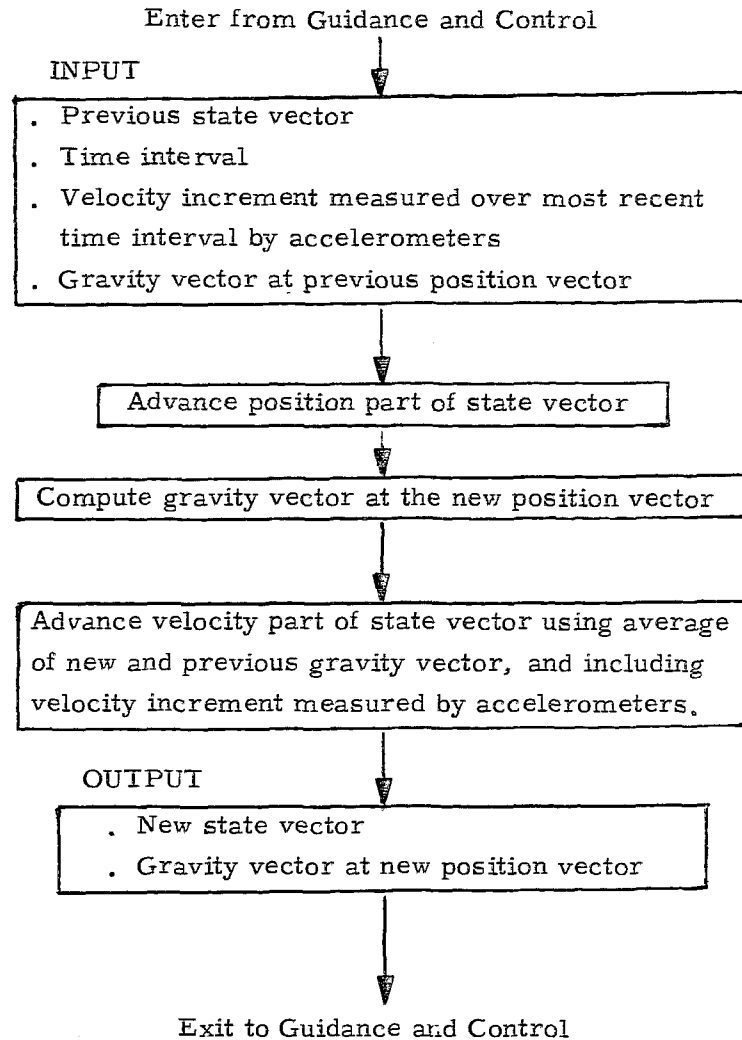


Figure 1 FUNCTIONAL FLOW DIAGRAM

9.2.1 Rapid Real-Time State Advancement During Specific Force Sensing (continued)

3. PROGRAM INPUT-OUTPUT

The required inputs to the routine consist of the most recent state vector, the time interval through which the advancement of that state vector is desired, the velocity increment due to the specific forces which was measured by the accelerometers over this time interval, and the gravity vector at the most recent position vector. The outputs of the routine are the new (i. e. advanced) state vector and the gravity vector at the new position vector, which will be used as inputs to the next computation performed by the routine.

Input Parameters

System

μ_E	:	Earth's gravitational parameter
r_E	:	Earth's mean equatorial radius
J_2	:	Earth oblateness constant
$\underline{1}_z$:	Unit vector in direction of earth's north polar axis expressed in reference coordinates

Real-Time

$\left[\underline{r}(t), \underline{v}(t) \right]$:	Most recent spacecraft state vector (at time t)
Δt	:	Time interval through which the most recent spacecraft state vector is to be advanced
$\Delta \underline{v}_{\text{meas}}(\Delta t)$:	Velocity vector increment which was measured by the integrating accelerometers during the time interval from t to t + Δt

9.2.1 Rapid Real-Time State Advancement During
Specific Force Sensing (continued)

$\underline{g}(t)$: Gravity vector at the most recent position
vector $\underline{r}(t)$.

Output Parameters

$\left[\begin{array}{l} \underline{r}(t+\Delta t), \\ \underline{v}(t+\Delta t) \end{array} \right]$: New (advanced) spacecraft state vector
(at time $t + \Delta t$)

$\underline{g}(t + \Delta t)$: Gravity vector at the new position vector
 $\underline{r}(t + \Delta t)$.

9.2.1 Rapid Real-Time State Advancement During Specific Force Sensing (continued)

4. DESCRIPTION OF EQUATIONS

The position and velocity vectors are advanced through a time interval Δt by the formulae:

$$\underline{r}(t + \Delta t) = \underline{r}(t) + \Delta t \left[\underline{v}(t) + \frac{\Delta t}{2} \underline{g}(t) + \frac{\Delta \underline{v}_{\text{meas}}(\Delta t)}{2} \right]$$
$$\underline{v}(t + \Delta t) = \underline{v}(t) + \frac{\Delta t}{2} \left[\underline{g}(t + \Delta t) + \underline{g}(t) \right] + \Delta \underline{v}_{\text{meas}}(\Delta t)$$

where $\Delta \underline{v}_{\text{meas}}(\Delta t)$ represents the velocity vector increment measured by the integrating accelerometers during the time interval Δt . The formulae represent a second-order technique of numerically integrating the differential equations of motion

$$\frac{d^2}{dt^2} \underline{r}(t) = \underline{g}$$

where \underline{g} is a simplified form of the gravity vector at the position \underline{r} . The technique requires only one evaluation of \underline{g} for each time-step Δt , which is of course a definite advantage in reducing overall computation time.

The gravity vector \underline{g} at the position \underline{r} is calculated from the expression:

$$\underline{g} = -\frac{\mu E}{r^2} \left\{ \underline{1}_r + \frac{3}{2} J_2 \left(\frac{r_E}{r} \right)^2 \left[(1 - 5 \cos^2 \phi) \underline{1}_r + 2 \cos \phi \underline{1}_z \right] \right\}$$

where

9.2.1 Rapid Real-Time State Advancement During Specific Force Sensing (continued)

$\underline{1}_r$ = unit vector in the direction of \underline{r}

$\underline{1}_z$ = unit vector in the direction of the earth's polar axis expressed in reference coordinates

$$\cos \phi = \underline{1}_r \cdot \underline{1}_z$$

and

r_E = the earth's mean equatorial radius.

9.2.1 Rapid Real-Time State Advancement During
Specific Force Sensing (continued)

5. DETAILED FLOW DIAGRAM

This section contains a detailed flow diagram of the routine for Rapid Real-Time State Advancement during Specific Force Sensing.

9.2.1 Rapid Real-Time State Advancement During Specific Force Sensing (continued)

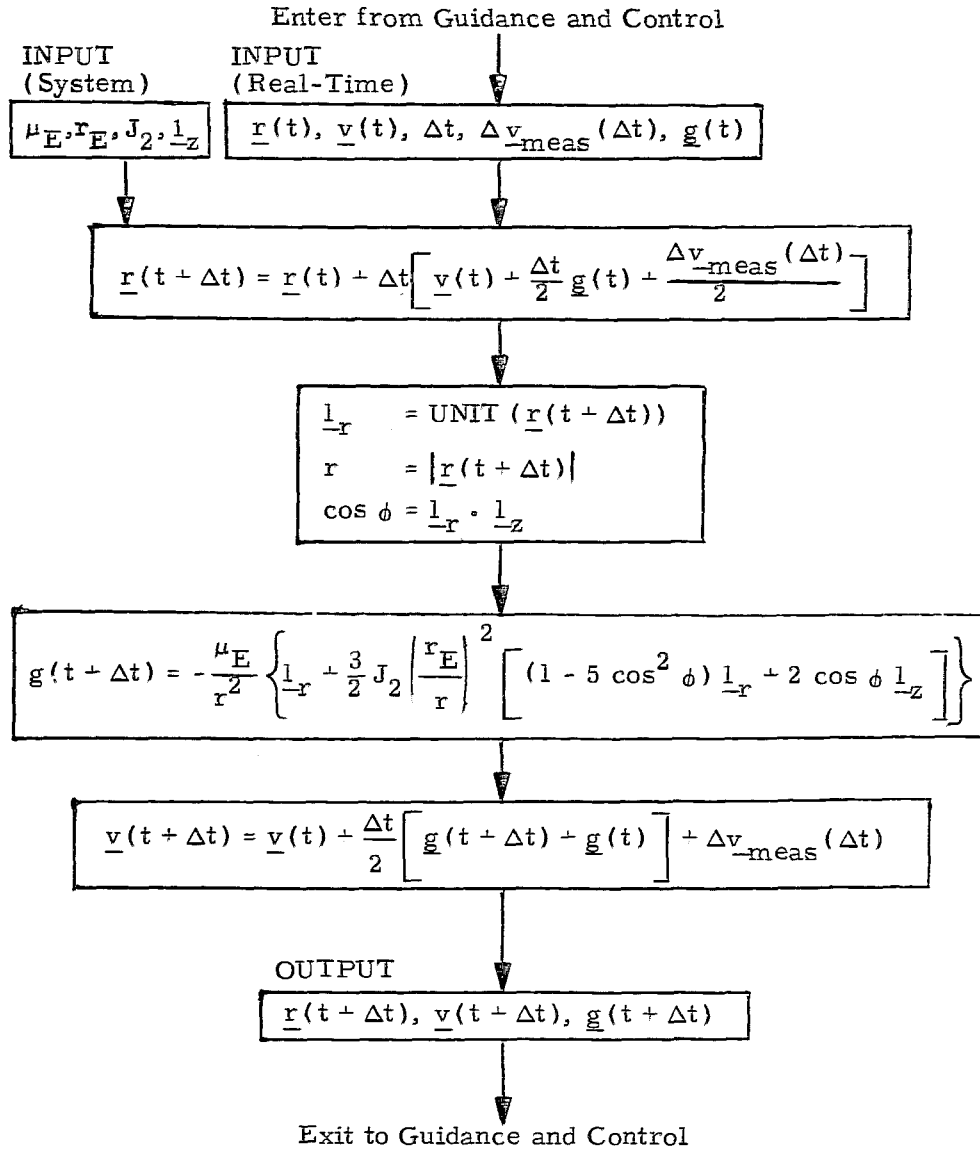


Figure 2 DETAILED FLOW DIAGRAM

9.2.1 Rapid Real-Time State Advancement During Specific Force Sensing (continued)

6. SUPPLEMENTARY INFORMATION

The formulation was originally developed for the Apollo Guidance Computer by Bairnsfather (1963), who also gave an error analysis of the computation when performed in eight, nine, and ten decimal digit precision (with addition instructions both truncated and rounded). It should be noted that Bairnsfather's analysis shows that a time-step of two seconds yields the most accurate results in terms of both position error magnitude and velocity error magnitude for eight decimal digit precision* with rounding over a thrity-five minute integration interval (100 ft. and 0.2 ft/sec, respectively).

References

1. Bairnsfather, R., 1963, Navigation Algorithms for Non-Freefall Flight, MIT/IL Space Guidance Analysis Memo #47, June 1963.

*The Apollo Guidance Computer has somewhat over eight decimal digit precision. The guidance programs written for it have had (for a variety of reasons) a guidance cycle period of two seconds.

9.2.2 Boost Monitor (TBD)

9.3 SEPARATION

The Separation Phase covers the portion of the orbital mission between the preparation for staging (during Boost) and the ignition of the Orbiter main engines. The maneuvers required during separation will be highly dependent on the mechanics of separation and the restrictions on relative vehicle motion due to impingement problems. Control functions during separation are greatly dependent on such open items as the vehicle configurations before and after separation, the separation mechanism employed, the plume impingement criteria, and the coast time limits.

The general GN&C software functions to be performed during the Separation are the following:

1. Continue powered flight navigation.
2. Command Staging sequence.
3. Compute and command RCS control during separation.
4. Perform attitude control during coast.

The following provides a preliminary description of the Shuttle dynamics during nominal and abort staging maneuvers.

"Preliminary Shuttle Dynamics Analysis," MSC
Internal Note No. 70-FM-143, 23 September 1970.

9.3.1 Attitude Control (TBD)

9.4 ORBIT INSERTION PROGRAM

The Orbit Insertion Phase extends from separation from the Booster through the second-stage boost until the desired orbital velocity has been achieved or an abort decision has been made. Presumably, data for initialization has been computed previously during the Boost and PreLaunch Phase. The Orbit Insertion Program is terminated by the start of the Orbital Coast Program or the Ascent Abort Program.

9.4.1 Navigation (Same as Rapid Real-Time State Advancement During Specific Force Sensing)

SPACE SHUTTLE

GN&C SOFTWARE EQUATION SUBMITTAL

Software Equation Section: Powered Ascent Guidance Submittal No. 4

Function: Provide Guidance Commands During Orbit Insertion

Module No. OG1 Function No. 1 (MSC 03690)

Submitted by: J. P. Higgins Co. MIT No. 1-71

Date: Jan 1971

NASA Contact: W. Peters Organization: GCD

Approved by Panel III: K.J. Cox Date: 2/24/71

Summary Description: The orbit insertion maneuver is controlled so as to achieve a specific orbit. To control the ascent to the position and velocity constraints, an explicit guidance law employing linear control is used. The radial and cross-range position and velocity constraints are controlled by their acceleration components. Downrange is controlled by time of thrust termination.

Shuttle Configuration: Equations are presented symbolically so that the configuration need not be specified.

Comments:

(Design Status) _____

(Verification Status) _____

Panel Comments:

9.4.2 Powered Ascent Guidance

1 INTRODUCTION

The objective of the Ascent Guidance Program is to provide guidance during the orbit insertion maneuver. This maneuver is controlled so as to achieve a specified orbit, which can be defined in terms of a desired velocity vector, a desired radius and a desired orbital plane at injection. To control the ascent maneuver to these position and velocity constraints, an explicit guidance law employing linear control is used. The radial and cross-range position and velocity constraints are explicitly controlled by two acceleration components, (a_{T_r}) and (a_{T_y}) , that vary linearly with time in the radial and cross-range directions. The third component of acceleration is determinate. The horizontal, down-range velocity constraint is controlled by terminating thrust at the proper time.

9.4.2 Powered Ascent Guidance (continued)

NOMENCLATURE

A	linear-control coefficient
$A_{11}, A_{12}, A_{21}, A_{22}$	parameters needed to compute linear-control coefficients
a_T	estimated thrust acceleration
a_{TL}	thrust-acceleration limit
a_{Tr}	required radial acceleration
a_{Ty}	required cross-range acceleration
a_{Tz}	required down-range acceleration
B	} linear-control coefficients
C	
D	
g_{eff}	effective gravity
\underline{i}_d	unit vector in the required thrust direction
\underline{i}_q	unit vector normal to the desired orbital plane
\underline{i}_r	unit local-vertical vector
\underline{i}_y	unit cross-range vector
\underline{i}_z	unit down-range vector
\underline{r}	vehicle-position vector
r	radius
\dot{r}	radial-velocity magnitude
r_d	desired radius
\dot{r}_d	desired radial-velocity magnitude
t_0	current time associated with \underline{r} and \underline{v}
t	time at which the thrust direction command is issued

9.4.2 Powered Ascent Guidance (continued)

Δt	guidance-cycle time step
t_{ig}	ignition time
t_{go}	time-to-go until thrust termination
t_L	time-to-go until acceleration limit is reached
\underline{v}	vehicle velocity vector
v_d	desired velocity magnitude
v_e	effective exhaust velocity
\underline{v}_g	velocity-to-be-gained vector
y	cross-range distance measured from the desired orbital plane
\dot{y}	cross-range velocity magnitude
\dot{z}	down-range velocity magnitude
\dot{z}_d	desired down-range velocity magnitude
γ_d	desired flight path angle
ι_d	desired inclination angle
λ	throttle setting - fraction of full throttle
μ	earth's gravitational constant
Ω_d	desired longitude of the descending node

9.4.2 Powered Ascent Guidance (continued)

2 FUNCTIONAL FLOW DIAGRAM

The chronological sequencing of functions performed by the Ascent Guidance Program is described in this section and illustrated by the functional flow diagram in Figure 1.

The Ascent Guidance Program is entered directly from the Boost Monitor Program when separation of the orbiter stage from the booster stage is sensed. The first function performed is to verify ignition time and to adjust it, if needed. Next, the desired target parameters are initialized and the initial thrust direction is computed. After these initialization functions are completed, the program cycles through the powered flight navigation routine until ignition occurs. Once ignition occurs, steering is initiated and the program cycles through the navigation routine and the steering routine every time step until thrusting is terminated.

9.4.2 Powered Ascent Guidance (continued)

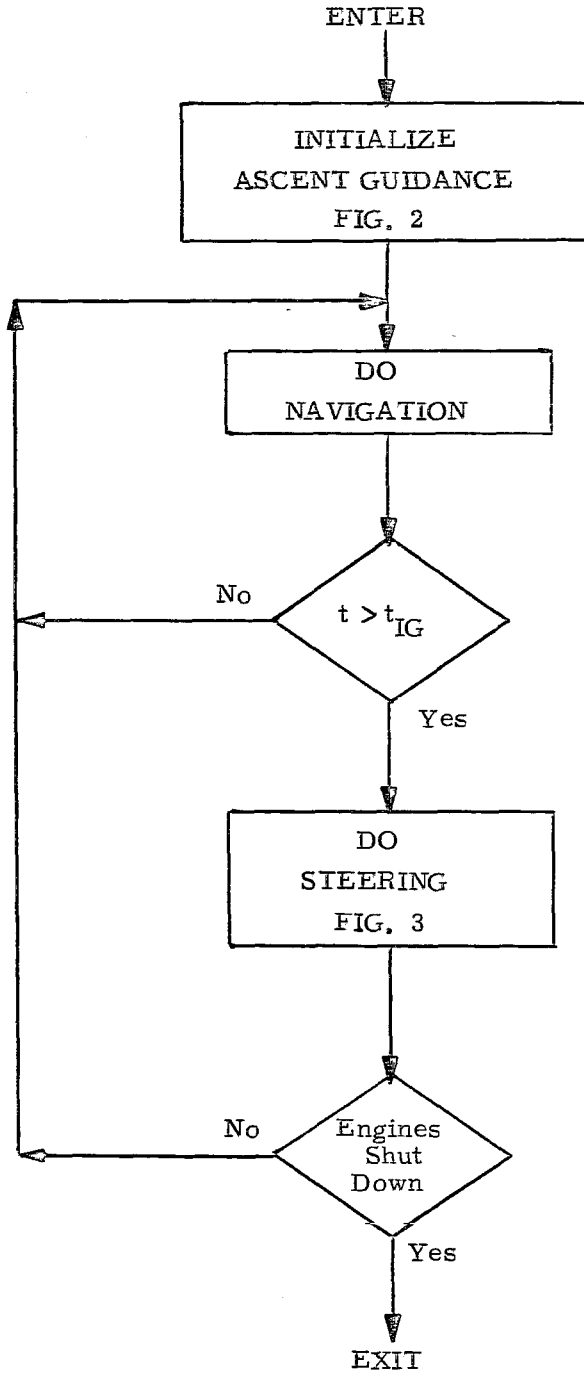


Figure 1 FUNCTIONAL FLOW DIAGRAM

9.4.2 Powered Ascent Guidance (continued)

3 PROGRAM INPUT-OUTPUT

The required inputs to the Ascent Guidance Program are the ignition time, an estimate of time-to-go, and a set of parameters defining the injection velocity vector, radius magnitude, and orbital plane. The output of this program is the thrust direction required at the present time to achieve the desired orbit and the throttle setting required to avoid exceeding the acceleration limit.

Input Parameters

r_d	desired radius
t_{go}	estimate of the time-to-go until thrust termination
t_{ig}	ignition time
v_d	desired velocity magnitude
γ_d	desired flight path angle
ι_d	desired inclination angle
Ω_d	desired longitude of the descending node

Output Parameters

\underline{i}_d	required thrust direction
λ	required throttle setting

9.4.2 Powered Ascent Guidance (continued)

4 DESCRIPTION OF EQUATIONS

The Ascent Guidance Program has two major phases, the pre-ignition phase and the powered ascent phase. Section 4.1 describes the equations required for the pre-ignition phase and Section 4.2 describes the equations required for the powered ascent phase. The computational sequences are illustrated in Figs. 2 and 3.

4.1 Pre-Ignition Phase

Figure 2 outlines the computational sequence of the pre-ignition phase. After ignition time (t_{ig}) has been verified, the set of parameters defining the desired injection velocity vector, radius magnitude, and orbital plane are input. Then the following thrust parameters are initialized:

$$a_T = (\text{TBD})$$

$$\tau = (\text{TBD})$$

$$a_{TL} = (\text{TBD})$$

$$v_e = (\text{TBD})$$

Initial thrust filter parameters = (TBD)

Next, the time-to-go until thrust termination (t_{go}) is set to the nominal value and the desired injection target parameters are initialized as follows.

$$t_{go} = (\text{TBD})$$

$$\dot{r}_d = v_d \sin(\gamma_d)$$

$$\dot{z}_d = v_d \cos(\gamma_d)$$

$$\underline{i}_q = \begin{bmatrix} -\sin(\iota_d) \sin(\psi_d) \\ \sin(\iota_d) \cos(\psi_d) \\ \cos(\iota_d) \end{bmatrix}$$

9.4.2 Powered Ascent Guidance (continued)

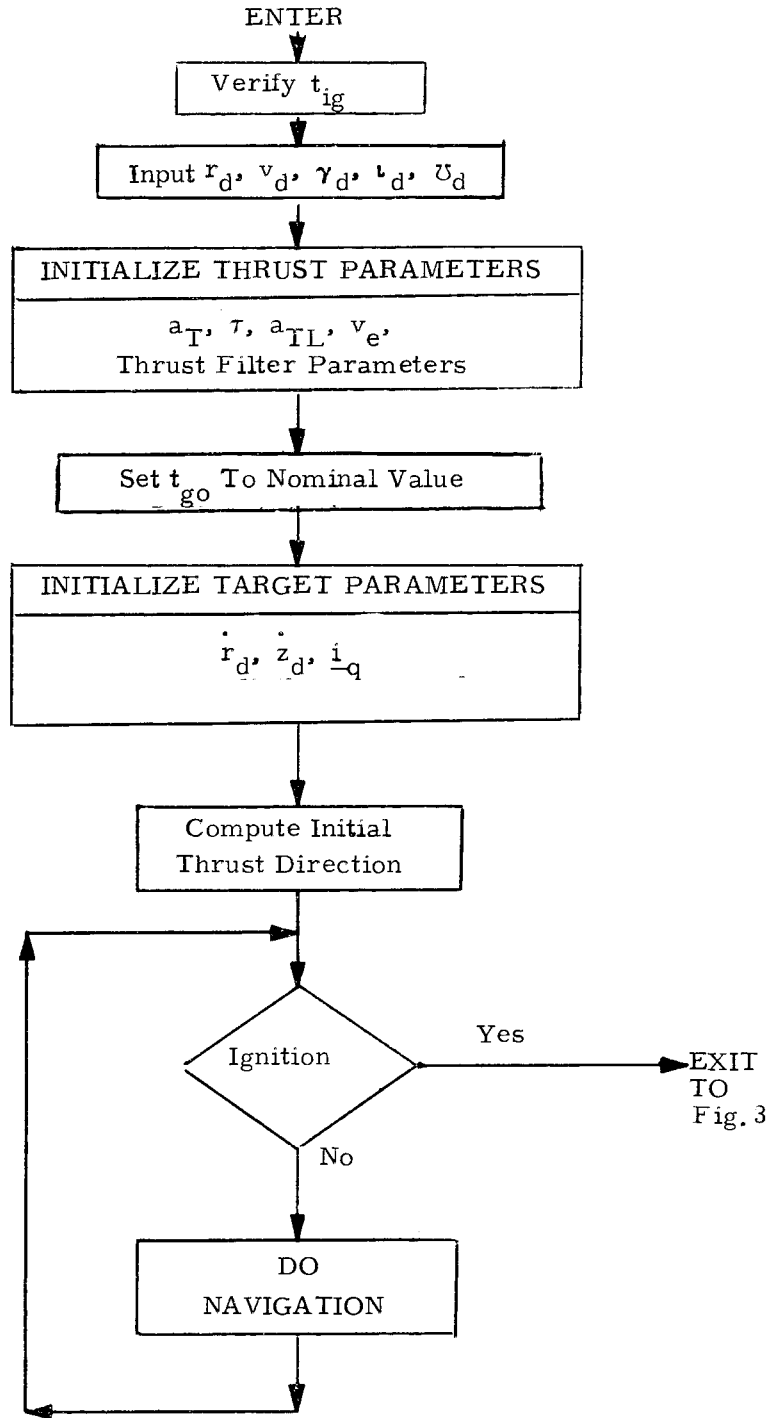


Figure 2 Pre-ignition Phase Computational Sequence

9.4.2 Powered Ascent Guidance (continued)

Finally, the initial required thrust direction is computed by cycling to point (A) of Fig. 3. After these pre-ignition computations are completed, the program cycles through the navigation subroutine until ignition is sensed. At this time, the program enters the powered ascent phase.

4.2 Powered Ascent Phase

The computational sequence during the powered ascent phase is outlined in functional form in Fig. 3 and the related equations are described below. These equations are recomputed at each time step during the powered maneuver unless otherwise stated.

Thrust Filter and Throttle Control (TBD)

The purpose of the thrust filter computations is twofold. First, the incremental velocity changes measured by the accelerometers include noise. As a result, the thrust filter is necessary to accurately estimate the present thrust acceleration (a_T) and the mass to mass-flow-rate ratio (τ). Secondly, since thrust acceleration is limited, it is necessary to predict the thrust acceleration over the next time step in order to determine if the engines need to be throttled down so as not to exceed this limit. The output of the thrust computations will be the following.

- (1) a_T = estimated thrust acceleration
- (2) τ = mass to mass-flow-rate ratio
- (3) λ = throttle setting

Local Position and Velocity Computations

After the thrust filter computations are completed the following parameters are computed:

9.4.2 Powered Ascent Guidance (continued)

$$\begin{aligned}\underline{i}_r &= \text{unit}(\underline{r}) \\ \underline{i}_z &= \text{unit}(\underline{i}_r \times \underline{i}_q) \\ \underline{i}_y &= \underline{i}_z \times \underline{i}_r \\ r &= |\underline{r}| \\ y &= r \sin^{-1}(\underline{i}_r \cdot \underline{i}_q) \\ \dot{r} &= \underline{v} \cdot \underline{i}_r \\ \dot{y} &= \underline{v} \cdot \underline{i}_y \\ \dot{z} &= \underline{v} \cdot \underline{i}_z\end{aligned}$$

Effective Gravity

Next, the effective gravity is computed for use in the velocity-to-be-gained computation.

$$g_{\text{eff}} = -\mu/r^2 + |\underline{r} \times \underline{v}|^2 / r^3$$

Velocity-to-be-Gained

The most recent estimate of the time-to-go is decremented by the guidance-cycle time step.

$$t_{go} = t_{go} - \Delta t$$

The velocity-to-be-gained is then computed by differencing the present velocity components with the desired velocity components and correcting for the average effect of gravity over the time-to-go.

$$\underline{v}_g = (\dot{r}_d - \dot{r}) \underline{i}_r - \dot{y} \underline{i}_y + (\dot{z}_d - \dot{z}) \underline{i}_z$$

$$\underline{v}_g = \underline{v} - \frac{1}{2} g_{\text{eff}} t_{go} \underline{i}_r$$

$$v_g = |\underline{v}_g|$$

Time-To-Go Prediction

The usual exponential equation

9.4.2 Powered Ascent Guidance (continued)

$$t_{go} = \tau (1 - e^{-v_g/v_e})$$

valid for a fixed-thrust rocket must be modified to compensate for g-limiting. First a t_{go} computation which neglects g-limiting is computed from a truncated series expansion of the previous equation

$$t_{go} = \tau \frac{v_g}{v_e} \left(1 - \frac{1}{2} \frac{v_g}{v_e}\right)$$

Then the time-to-go until the acceleration limit is reached (t_L) is computed from the following equation

$$t_L = \tau - v_e / a_{TL}$$

If t_{go} is greater than t_L , then t_{go} is recomputed to account for g-limiting by the following equation

$$t_{go} = t_L + \left[v_g + v_e \log(1 - t_L / \tau) \right] / a_{TL}$$

Guidance Parameters

The Ascent Guidance Program uses a linear control law which can be expressed by the following equations.

$$a_{Tr} = a_T \left[A + B(t - t_0) \right] - g_{eff}$$

$$a_{Ty} = a_T \left[C + D(t - t_0) \right]$$

a_{Tr} and a_{Ty} are the required radial and cross-range accelerations respectively, a_T is the filtered thrust acceleration, and $(t - t_0)$ is the difference between the time when the linear coefficients are computed and the time when the thrust direction command is issued. When t_{go} becomes less than Δt seconds, the linear coefficients are not recomputed; instead, the previous values are used to compute a_{Tr} and a_{Ty} . The equations required to compute the linear coefficients are as follows:

9.4.2 Powered Ascent Guidance (continued)

$$A = \left[-A_{12} (r_d - r - \dot{r} t_{go}) + A_{22} (\dot{r}_d - \dot{r}) \right] J$$

$$B = \left[A_{11} (r_d - r - \dot{r} t_{go}) - A_{21} (\dot{r}_d - \dot{r}) \right] J$$

$$C = \left[A_{12} (y + \dot{y} t_{go}) - A_{22} \dot{y} \right] J$$

$$D = \left[-A_{11} (y + \dot{y} t_{go}) + A_{21} \dot{y} \right] J$$

where

$$J = 1 / (A_{11} A_{22} - A_{12} A_{21})$$

The parameters A_{11} , A_{12} , A_{21} , and A_{22} are the coefficients of the simultaneous equations used to solve for A, B, C, and D. These parameters are computed in one of three ways, depending on the current values of t_{go} and t_L .

- 1) $t_{go} < t_L$ (acceleration limit will not be reached)

$$A_{11} = -v_e \log(1 - t_{go} / \tau)$$

$$A_{12} = -v_e \tau \log(1 - t_{go} / \tau) - v_e t_{go}$$

$$A_{21} = v_e (\tau - t_{go}) \log(1 - t_{go} / \tau) + v_e t_{go}$$

$$A_{22} = v_e \tau (\tau - t_{go}) \log(1 - t_{go} / \tau) + v_e t_{go} (\tau - \frac{1}{2} t_{go})$$

9.4.2 Powered Ascent Guidance (continued)

- 2) $0 < t_L < t_{go}$ (acceleration limit will be reached)

$$A_{11} = -v_e \log(1 - t_L/\tau) + a_{TL} (t_{go} - t_L)$$

$$A_{12} = -v_e \tau \log(1 - t_L/\tau) - v_e t_L + \frac{1}{2} a_{TL} (t_{go}^2 - t_L^2)$$

$$A_{21} = v_e (\tau - t_{go}) \log(1 - t_L/\tau) + v_e t_L + \frac{1}{2} a_{TL} (t_{go} - t_L)^2$$

$$A_{22} = v_e \tau (\tau - t_{go}) \log(1 - t_L/\tau) + v_e t_L (\tau + \frac{1}{2} t_L - t_{go}) + \frac{1}{6} a_{TL} (t_{go}^3 - 3 t_{go} t_L^2 + 2 t_L^3)$$

- 3) $t_L \leq 0$ (acceleration limit has been reached)

$$A_{11} = a_{TL} t_{go}$$

$$A_{12} = \frac{1}{2} a_{TL} t_{go}^2$$

$$A_{21} = \frac{1}{2} a_{TL} t_{go}^2$$

$$A_{22} = \frac{1}{6} a_{TL} t_{go}^3$$

Required Thrust Direction

The final computation performed by the Ascent Guidance Program is the required thrust direction. The radial and cross-range accelerations required to satisfy the linear control law are combined as follows:

$$\underline{a}_{Try} = a_{Tr} \underline{i}_r + a_{Ty} \underline{i}_y$$

$$a_{Try} = | \underline{a}_{Try} |$$

9.4.2 Powered Ascent Guidance (continued)

$a_{T_{Try}}$ is then compared with a_T . If a_T is less than $a_{T_{Try}}$, the acceleration vector $\underline{a}_{T_{Try}}$ is reduced by the ratio $a_T/a_{T_{Try}}$.

$$\underline{a}_{T_{Try}} = \underline{a}_{T_{Try}} a_T / a_{T_{Try}}$$

If a_T is greater than $a_{T_{Try}}$, the additional available acceleration is applied in the down-range direction.

$$a_{Tz} = (a_T^2 - a_{T_{Try}}^2)^{1/2} \text{sign}(\dot{z}_d - z)$$

The total thrust acceleration vector then becomes

$$\underline{a}_T = \underline{a}_{T_{Try}} + a_{Tz} \underline{i}_z$$

and the required unit thrust vector is

$$\underline{i}_d = \text{unit}(\underline{a}_T)$$

9.4.2 Powered Ascent Guidance (continued)

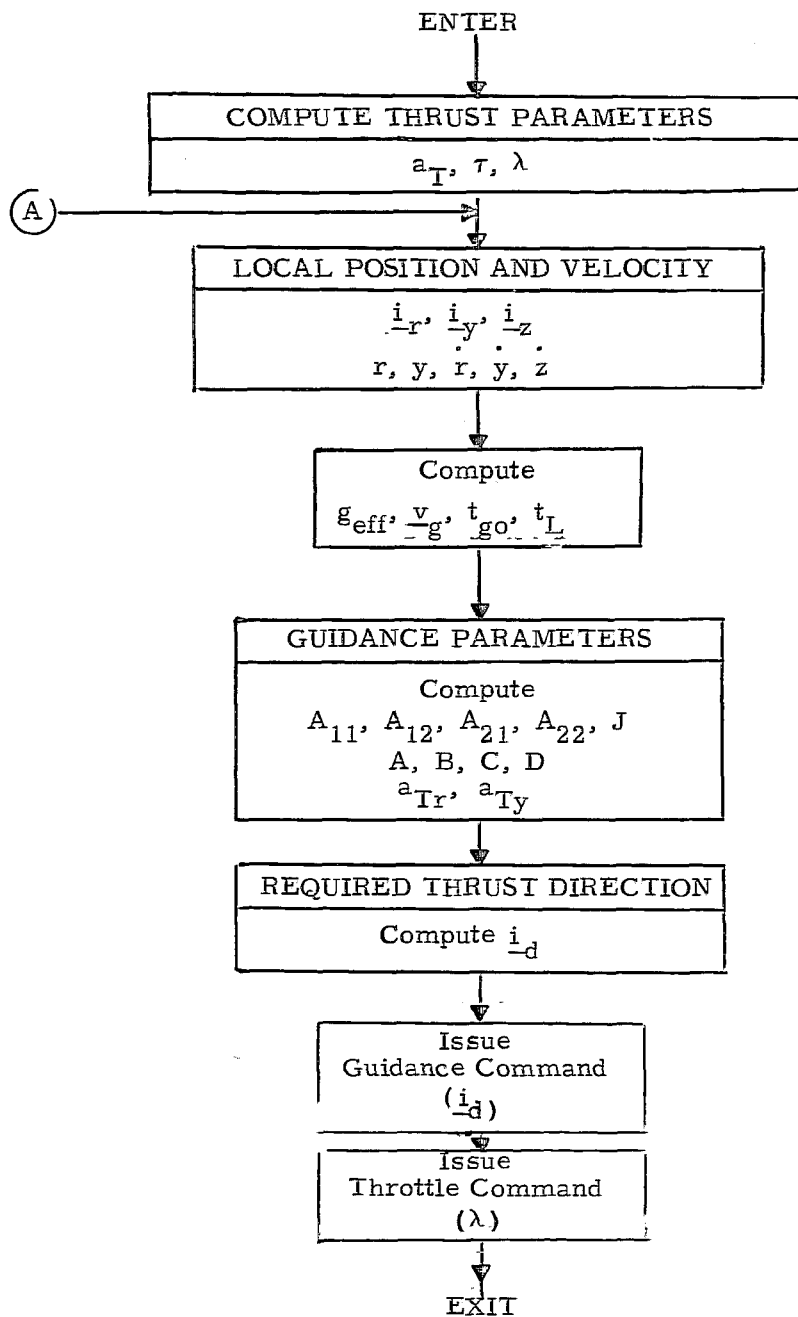


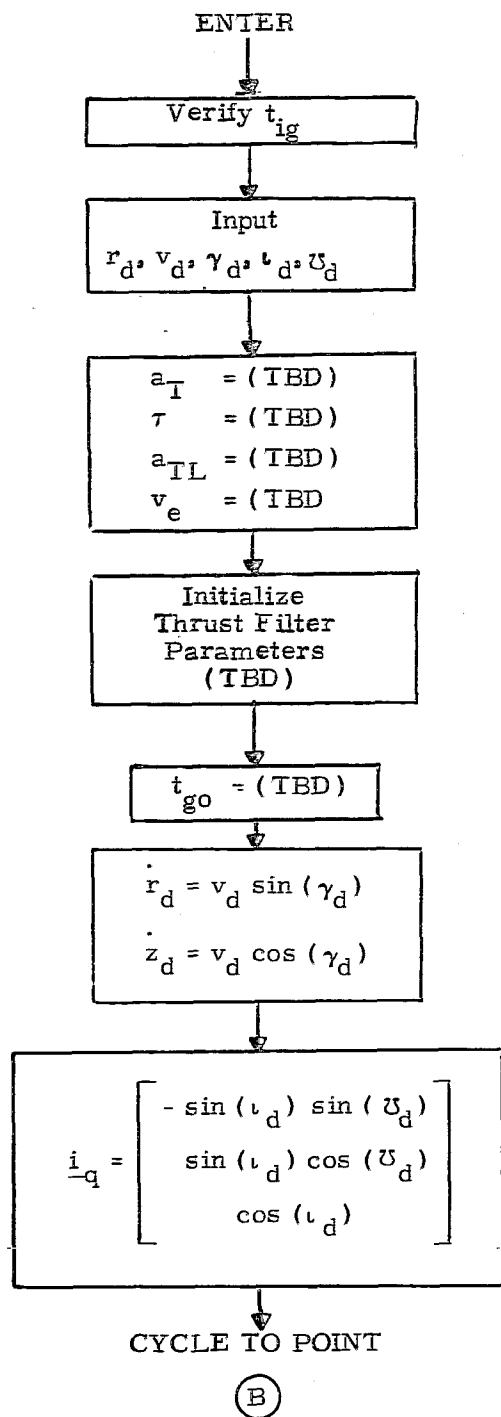
Figure 3 POWERED ASCENT PHASE COMPUTATIONAL SEQUENCE

9.4.2 Powered Ascent Guidance (continued)

5 DETAILED FLOW DIAGRAMS

This section contains detailed flow diagrams of the Ascent Guidance Program.

9.4.2 Powered Ascent Guidance (continued)



of Fig. 5a

Figure 4 DETAILED FLOW DIAGRAM, PRE-IGNITION PHASE

9.4.2 Powered Ascent Guidance (continued)

Nominal Entry from
Navigation (Every Guidance Cycle)

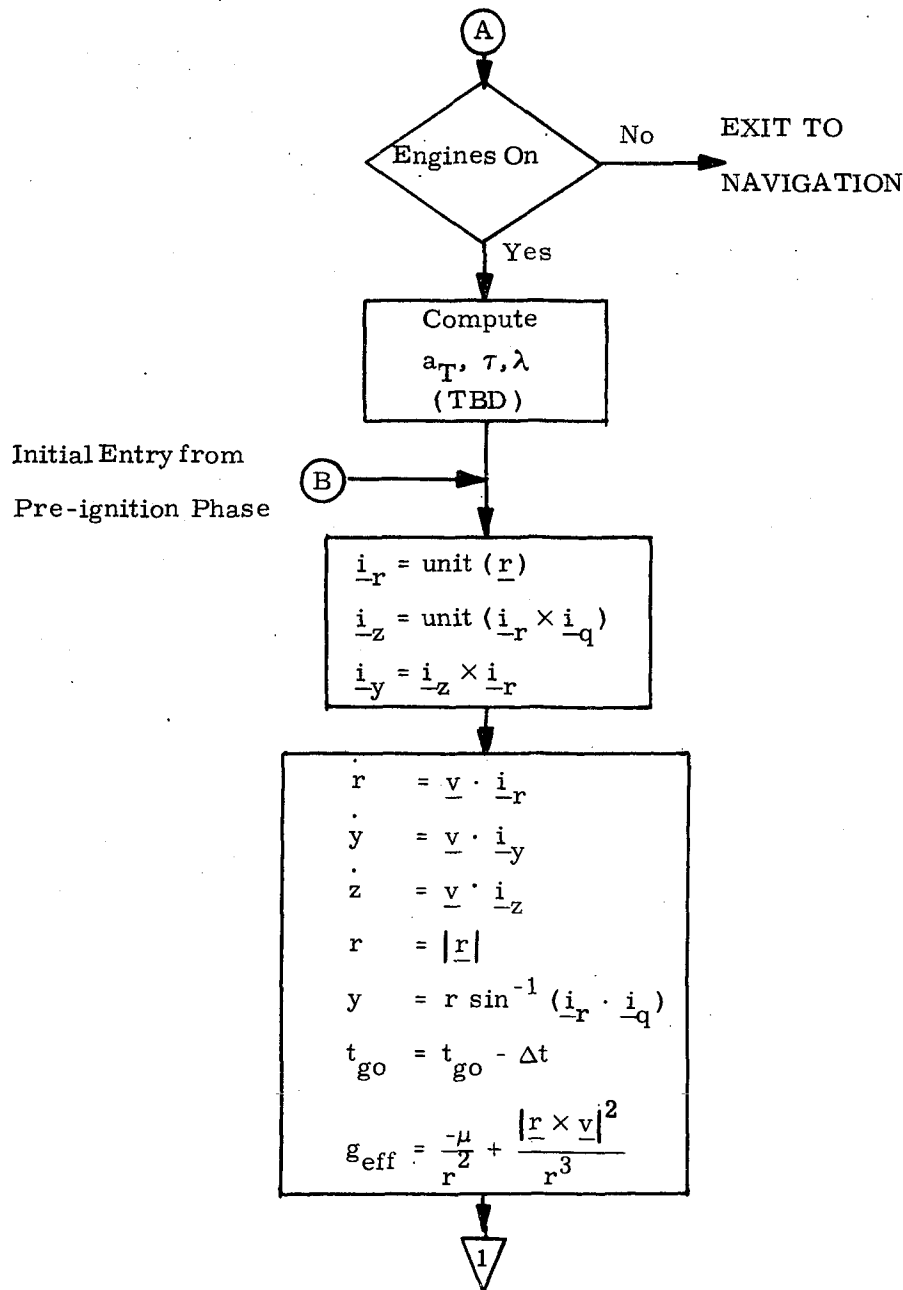


Figure 5a DETAILED FLOW DIAGRAM, POWERED ASCENT PHASE

9.4.2 Powered Ascent Guidance (continued)

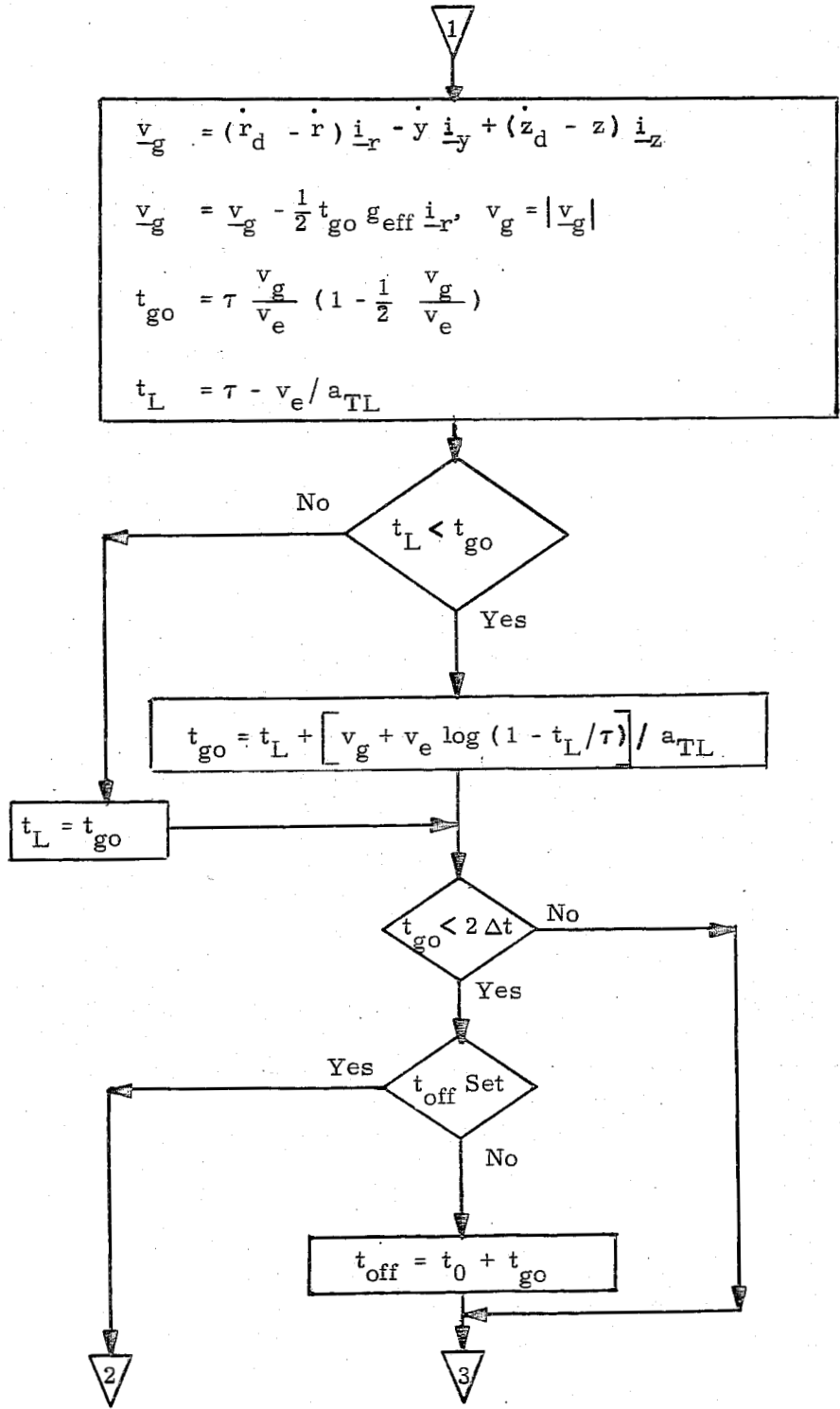


Figure 5b DETAILED FLOW DIAGRAM, POWERED ASCENT PHASE

9.4.2 Powered Ascent Guidance (continued)

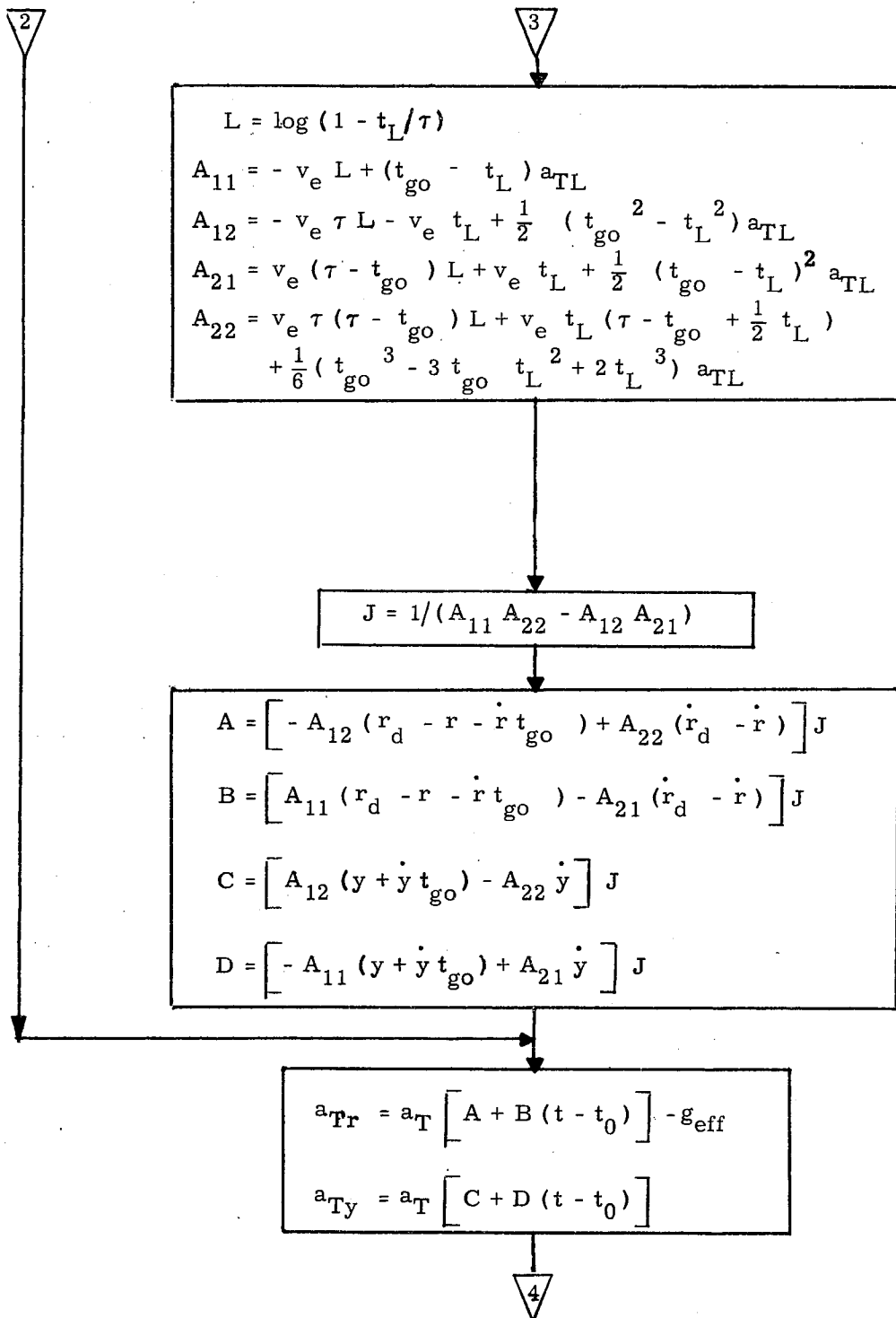


Figure 5c DETAILED FLOW DIAGRAM, POWERED ASCENT PHASE

9.4.2 Powered Ascent Guidance (continued)

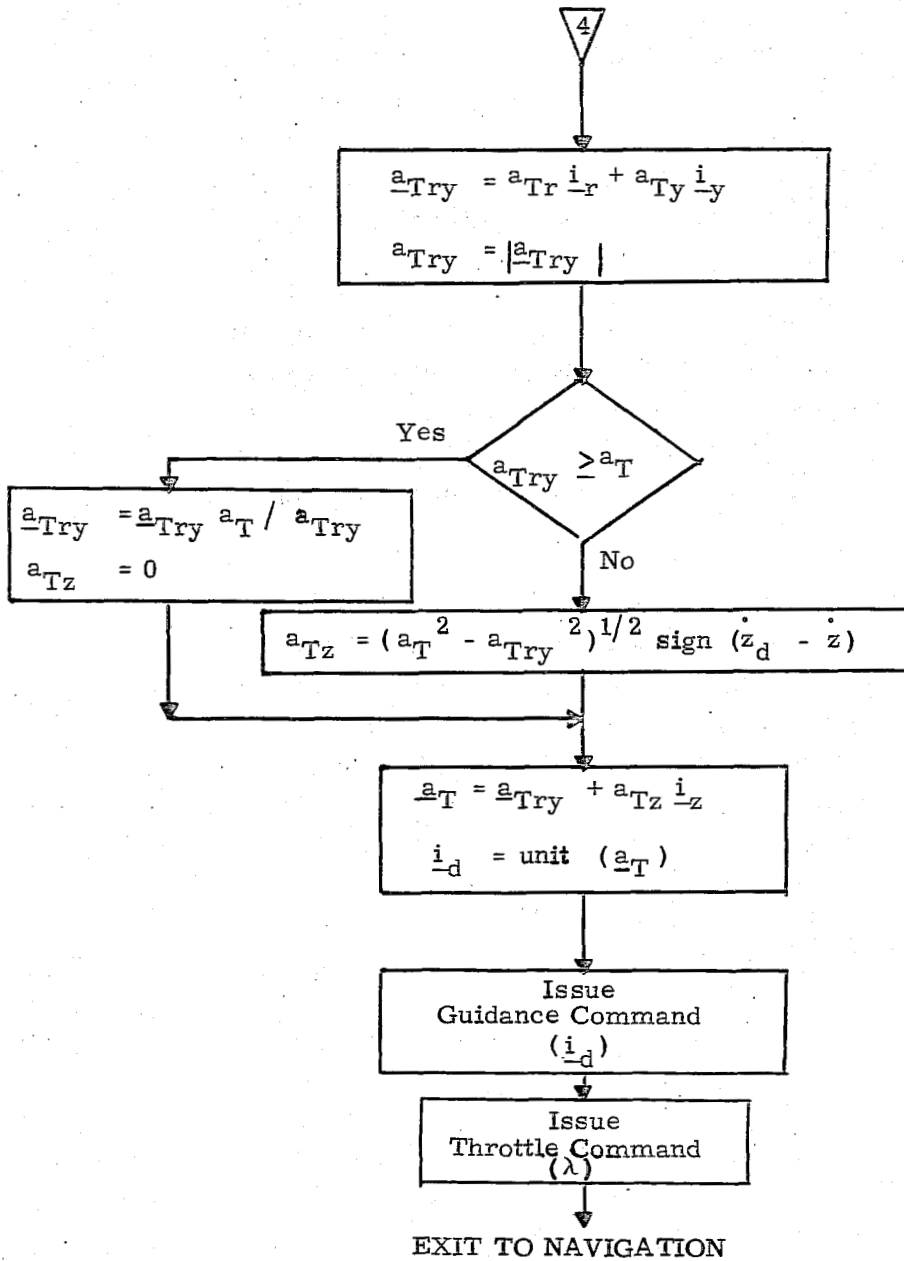


Figure 5d DETAILED FLOW DIAGRAM, POWERED ASCENT PHASE

9.4.2 Powered Ascent Guidance (continued)

6 SUPPLEMENTARY INFORMATION

The guidance equations presented here were derived from explicit guidance laws developed by G. Cherry¹. They were previously used in the Apollo Lunar Ascent Guidance Program². Since the space shuttle acceleration is limited, however, a major modification of the equations was required. The derivation of this modification can be found in Appendix A of Reference 3.

These equations have been used satisfactorily in the generation of various launch trajectories required for IMU performance analysis. In addition, a comparison was made with an optimal trajectory in one case. The results of this case showed the ascent-guidance equations to be near fuel optimal. Details of this comparison can be found in Reference 4.

Future work should include development of a thrust estimation filter, throttling logic, and thrust tailoff compensation. Since they are dependent on detailed characteristics of the ascent engines, they have been omitted from this document. In addition, a comparison of this scheme with other candidates, in particular the Saturn IGM equations, should be made.

9.4.2 Powered Ascent Guidance (con't)

References

1. Cherry, G. W., "A Class of Unified Explicit Methods for Steering Throttleable and Fixed-Thrust Rockets", R-417, MIT Instrumentation Laboratory, Revised January, 1964.
2. "Guidance System Operation Plan", R-567, Rev. 4, Section 5, MIT Instrumentation Laboratory, June 1969.
3. Higgins, J. P., Brand, T., "IMU Performance Analysis of STS Launch Trajectories", 23A STS Memo No. 5-70, MIT C.S. Draper Laboratory, April 11, 1970.
4. Pu, Chung L. "SSV Orbiter Launch Trajectory", 23A STS Memo No. 3-71, MIT C.S. Draper Laboratory January 18, 1971.

9.4.3 Attitude Control (TBD)

9.5 ASCENT ABORTS

Neither equations nor functions have been defined for Aborts during ascent. It is assumed here that the Abort Program will employ routines from other programs for most GN&C activities. The targeting would be performed by the Abort Program until the Orbiter has achieved a flight condition suitable for normal program sequencing for the return to Earth. The following functions are postulated for the Abort Program:

1. Target the Orbiter to a parking orbit, a suitable entry condition, or an acceptable cruise condition.
2. Sequence other GN&C programs to perform maneuvers required during abort operations.

9.5.1 Abort Decision and Mode Determination (TBD)

9.5.2 Abort Targeting (TBD)

9.5.3 Abort Guidance (TBD)

9.5.4 Navigation (TBD)

9.5.5 Attitude Control (TBD)

