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AN ADAPTIVE NUMERIC PREDICTOR-CORRECTOR GUIDANCE ALGORITHM FOR ATMOSPHERIC ENTRY VEHICLES

by

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AN ADAPTIVE NUMERIC PREDICTOR-CORRECTOR
GUIDANCE ALGORITHM FOR ATMOSPHERIC
ENTRY VEHICLES

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B.A.E., Georgia Institute of Technology (1985)

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ABSTRACT

An adaptive numeric predictor-corrector guidance algorithm is developed for atmospheric entry vehicles which utilize lift to achieve maximum footprint capability. Applicability of the guidance design to vehicles with a wide range of performance capabilities is desired so as to reduce the need for algorithm redesign with each new vehicle. Adaptability is desired to minimize mission-specific analysis and planning. The guidance algorithm motivation and design are presented.

Performance is assessed for application of the algorithm to the NASA Entry Research Vehicle (ERV). The dispersions the guidance must be designed to handle are presented. The achievable operational footprint for expected worst-case dispersions is presented. The algorithm performs excellently for the expected dispersions and captures most of the achievable footprint.

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SYMBOLS

\( a \) = acceleration magnitude
\( \vec{a} \) = acceleration vector
\( \vec{a}_i \) = inertial acceleration measured by the inertial measurement unit
\( \vec{A}_i \) = total inertial acceleration vector
\( AOTV \) = Aerobraking Orbital Transfer Vehicle
\( BTU \) = British Thermal Unit
\( \bar{C} \) = mean aerodynamic chord
\( \cos(\Delta \phi) \) = cosine of incremental lift for heat rate control
\( C' \) = proportionality factor for the linear viscosity-temperature relationship
\( C_0 \) = aerodynamic drag coefficient
\( C_L \) = aerodynamic lift coefficient
\( C_s \) = speed of sound
\( CPU \) = central processing unit
\( CR \) = crossrange
\( CSDL \) = The Charles Stark Draper Laboratory, Inc.
\( \text{det} \) = determinant of sensitivity matrix
\( DR \) = downrange
\( DOF \) = degree of freedom
\( ERV \) = Entry Research Vehicle
\( f_{\text{earth}} \) = flattening of oblate Earth
\( \vec{F}_i \) = inertial force vector
\( g \) = gravitational acceleration magnitude
\( \vec{g}_i \) = gravitational acceleration vector
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<td>Global Positioning System</td>
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<tr>
<td>$h$</td>
<td>altitude</td>
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<tr>
<td>$\dot{h}$</td>
<td>derivative of altitude with time</td>
</tr>
<tr>
<td>$\ddot{h}$</td>
<td>second-derivative of altitude with time</td>
</tr>
<tr>
<td>$h_s$</td>
<td>scale height for exponential atmosphere model</td>
</tr>
<tr>
<td>$\vec{i}$</td>
<td>unit vector</td>
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<tr>
<td>$J_2$</td>
<td>second zonal harmonic coefficient</td>
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<td>JSC</td>
<td>Lyndon B. Johnson Space Center</td>
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<tr>
<td>$k$</td>
<td>term in geodetic to geocentric latitude conversion</td>
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<td>term in heat rate control equation</td>
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<td>velocity vector term in the integration algorithm</td>
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<td>$K_{L/D}$</td>
<td>multiplicative scale factor on the nominal L/D</td>
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<td>$K_\theta$</td>
<td>gain on heat rate error in heat rate control equation</td>
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<td>gain on rate of change of heat rate in heat rate control equation</td>
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<td>$K_\rho$</td>
<td>multiplicative scale factor on the standard density</td>
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<td>gain in first-order filter for L/D smoothing</td>
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<tr>
<td>L/D</td>
<td>lift-to-drag ratio</td>
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<td>LaRC</td>
<td>Langley Research Center</td>
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<tr>
<td>$m$</td>
<td>mass</td>
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<tr>
<td>$M$</td>
<td>Mach Number</td>
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<tr>
<td>$\tilde{M}_f^p$</td>
<td>inertial-to-Earth-fixed transformation matrix</td>
</tr>
<tr>
<td>$M_0$</td>
<td>mean molecular weight of air at sea level</td>
</tr>
<tr>
<td>n.m.</td>
<td>nautical mile</td>
</tr>
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<td>NASA</td>
<td>National Aeronautics and Space Administration</td>
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\textit{POST} = Program to Optimize Simulated Trajectories
\( \bar{q} \) = dynamic pressure
\( Q \) = heat load
\( \dot{Q} \) = heat rate
\( \ddot{Q} \) = time rate of change of heat rate
\( R \) = position vector magnitude
\( R_{\text{equator}} \) = radius of oblate Earth at equator
\( \vec{R}_i \) = inertial position vector
\( R_{\text{pole}} \) = radius of oblate Earth at pole
\( Re \) = Reynolds Number
\( S \) = Sutherland’s constant in viscosity equation
\( S \) = aerodynamic reference area
\textit{SEADS} = Shuttle Entry Air Data System
\( t_{\text{GMT}} \) = Greenwich mean time
\( T' \) = reference temperature
\( T_m \) = molecular scale temperature
\( T_{\text{static}} \) = freestream static temperature
\( T_{\text{wall}} \) = wall temperature
\textit{TAEM} = Terminal Area Energy Management
\( \bar{V} \) = Viscous Interaction Parameter
\( V_i \) = magnitude of inertial velocity
\( \vec{V}_i \) = inertial velocity vector
\( V_R \) = magnitude of Earth-relative velocity
\( \vec{V}_R \) = Earth-relative velocity vector
\( z \) = geocentric colatitude of the position vector
\( \alpha \) = angle of attack
\( \beta \) = angle of sideslip
\( \delta \) = small incremental change
\( \Delta \) = incremental change
\( \gamma \) = ratio of specific heats for air
\( \lambda \) = longitude
\( \mu \) = coefficient of viscosity for air
\( \mu \) = Earth gravitational constant
\( \omega_{\text{earn}} \) = Earth's rotation vector
\( \omega_n \) = natural frequency of heat rate control response
\( \phi \) = bank angle
\( \mathcal{R} \) = universal gas constant
\( \rho \) = atmospheric density
\( \sigma \) = standard deviation
\( \tau \) = time constant of filter
\( \zeta \) = damping ratio of heat rate control response

**SUBSCRIPTS**

aero = aerodynamic
c = geocentric
cmd = command
d = desired
des = desired
drag = drag term
\begin{align*}
e &= \text{error} \\
El &= \text{entry interface} \\
f &= \text{final} \\
g &= \text{geodetic} \\
imu &= \text{inertial measurement unit} \\
inplane &= \text{projection of target unit vector into plane formed by the position vector and the relative velocity vector} \\
lat &= \text{direction perpendicular to the plane formed by the position vector and the relative velocity vector} \\
lift &= \text{lift term} \\
lim &= \text{limiting value or boundary} \\
L/D &= \text{lift-to-drag ratio} \\
max &= \text{maximum} \\
min &= \text{minimum} \\
nom &= \text{nominal} \\
perpen &= \text{direction perpendicular to the plane formed by the position vector and the relative velocity vector} \\
pole &= \text{direction of the north pole} \\
R &= \text{direction of the position vector} \\
sl &= \text{sea level} \\
std &= \text{standard value} \\
t &= \text{target aim point} \\
\rho &= \text{atmospheric density}
\end{align*}
SUPERSCRIPTS

\( EF \) = coordinatized in Earth-fixed coordinates

\( imu \) = value measured by inertial measurement unit on current cycle

\( imu \text{ past} \) = value measured by inertial measurement unit on past cycle

\( \wedge \) = estimated or measured

\( \cdot \) = value from previous guidance cycle
1.0 INTRODUCTION

Routine access to space and the maintenance of a Space Station will increasingly require greater flexibility in mission planning and the requirement for lower system maintenance costs. The launch and recovery phases of space flight have historically been the most demanding phases of space flight and therefore require the most development effort and investment. Mission flexibility requires more frequent launch and deorbit opportunities. For the case of re-entry vehicles, deorbit opportunities are defined by the ranging capability of the vehicle. A high L/D vehicle increases the available deorbit opportunities increasing mission flexibility. High L/D vehicles also are of interest for over-flight missions for the purpose of reconnaissance.

Entry guidance algorithms developed to date have been highly vehicle-specific and required great development and maintenance efforts over the life of the vehicle. These algorithms were not applicable to other vehicles without extensive modification.

This study seeks to design an adaptive entry guidance algorithm that maximizes the usable footprint by making full use of the available vehicle capability. This algorithm should also be easy to maintain throughout the vehicle definition phase and operational life. Minimizing the number of mission-dependent input parameters (I-loads) is desirable. The algorithm should also be easily transported to other vehicles to minimize development cost. Transportability is accomplished by minimizing vehicle-specific features of the algorithm. Explicit heat rate control should be provided to allow full use of the entry corridor up to the heat rate limits.
This study seeks to design such an algorithm. A candidate entry guidance algorithm is defined for the NASA Entry Research Vehicle (ERV), but is easily adapted to other vehicles with minimal modification. The proposed algorithm attains almost complete coverage of the achievable footprint, while employing a simple one-phase entry algorithm with explicit heat rate control. Vehicle-specific features and I-loads are minimized, reducing algorithm development and maintenance costs.

The ERV [1] is a proposed high-performance entry vehicle designed as a test bed for future technology development in the areas of:

1. Maneuvering entry/synergetic plane change
2. Atmospheric uncertainties
3. Advanced thermal protection systems
4. Aerodynamic/aeroheating prediction
5. Adaptive guidance and navigation
6. Load-bearing thermostructures

The ERV is designed for deployment from the Space Shuttle, after which the ERV enters the atmosphere for demonstration of the synergetic plane change, over-flight, and entry missions. Figure 1 on page 78 shows a three-view drawing of the ERV and the surface areas of the aerodynamic control surfaces. Also seen is the size of the ERV in relation to the diameter of the Shuttle payload bay in which the ERV must fit.
2.0 MOTIVATION

2.1 INTRODUCTION

The goal of any entry guidance algorithm is to successfully guide the vehicle to the desired final state for the largest range of dispersions possible without violating any vehicle constraints while also maximizing the achievable footprint. It is also desirable to minimize the mission and vehicle-specific aspects of the guidance algorithm so as to minimize pre-mission analysis and planning. Transportability of the algorithm from one vehicle to another significantly reduces guidance algorithm development effort and cost.

To maximize the footprint attainable, the guidance algorithm must follow the optimal path to any particular point in the footprint. The algorithms developed to date for such vehicles as the Apollo capsule [2] and the Space Shuttle [3] have attempted to do this by fitting the optimal trajectory with phases that follow important parameters (reference profiles) over some range of conditions. These guidance algorithms were required to be computationally efficient because of the limited on-board computer resources available. Analytic expressions for the reference profiles allowed for low execution time and tailoring of the trajectory for vehicle-specific constraints. For example, trajectories for these vehicles had to be shaped to reduce and control the maximum heat rate experienced below that allowed for the available thermal protection system materials.
The Space Shuttle entry guidance system employs three major modes with seven phases:

1. Entry
   a. Pre-entry
   b. Temperature control
   c. Equilibrium glide
   d. Constant drag
   e. Transition
2. Terminal Area Energy Management
3. Approach and Landing

Except for the pre-entry phase which is open-loop, each phase is described by an analytic expression relating the desired drag and altitude rate (the measured feedback terms used) to the desired profile. Because the algorithms are tailored for a particular vehicle and the reference profiles do not follow the optimal profile to all points in the footprint, guidance algorithms developed to date cannot be easily adapted to other vehicles or provide full coverage of the theoretically achievable footprint.

The next generation of entry vehicles will not be so constrained due to advances in thermal protection system materials and computer technology. For example, flight computers are now capable of supercomputer speeds on the order of 40 million instructions per second utilizing parallel processing architecture [4]. A different approach to guidance that attempts to follow an optimal profile to maximize footprint capability is therefore possible.

The proposed approach is a predictor-corrector algorithm that numerically predicts the final state for a particular control variable history and then corrects the control variable history to satisfy the specified final state constraints. This approach, proposed previously for
various guidance problems, has most often been impractical because of the long trajectories that must be predicted and the slow computer speeds.

Such an approach has been employed for the Space Shuttle Powered Explicit Guidance (PEG) [5] used for second stage ascent and orbit insertion burns where the trajectory is short enough to be predicted with the available computer resources. The Shuttle algorithm numerically predicts the gravitational effects during the powered flight phase with a 10 step integration of the 500 second trajectory.

A predictor-corrector has also been proposed for Aerobraking Orbital Transfer Vehicles (AOTV) [6] which would utilize more advanced computers. This algorithm numerically integrates the equations of motion along a skimming trajectory through the upper atmosphere that is approximately 500 seconds long and requires about 100 integration steps.

The trajectories flown by the ERV or any high L/D entry vehicle are typically from 30 to 100 minutes long from entry interface (400K feet) to landing, so the computational demand for such an algorithm is very great early in the entry when the time to landing is long. However, because the entry is long and the vehicle has excess ranging capability for all but a small region along the edge of the footprint, the accuracy of the early predictions need not be as high as for the later predictions. Hence, large time steps can be used early in the predictor algorithm. Later, when the vehicle nears the landing site, the time remaining is short, and hence, the prediction is short. This allows the predictor-corrector to be executed more often near landing just like the current analytic algorithms. Throughout the entry, vehicles using an analytic guidance algorithm with reference profiles must closely follow the reference profile if the assumed reference profile is to guide the vehicle to the correct final state. A predictor-corrector effectively recomputes a new reference profile each time it is executed, so the guidance execution rate can be much lower than that for analytic algorithms.
2.2 DISPERSIONS

Before the guidance algorithm can be designed, the possible dispersions that may affect the trajectory must be considered. The Shuttle entry guidance system is required to reach the Terminal Area Energy Management (TAEM) interface with less than a 2.5 nautical mile position error from the target aim point. The dispersions of significance to an entry vehicle trajectory include:

1. Vehicle characteristics
   a. Mass
   b. Aerodynamics
   c. Maneuver rates

2. Environment characteristics
   a. Atmospheric density
   b. Atmospheric winds
   c. Atmospheric properties influencing aerodynamic flow regimes (temperature, mean free path, etc.)

3. Initial entry state vector
   a. Velocity
   b. Flight path angle
   c. Heading

4. Propagation errors in navigation state vector

Of these potential dispersion sources, only the vehicle mass, aerodynamics, and the atmospheric density and winds will be significant. By the early 1990's, almost perfect navigation can be expected through use of the Global Positioning System (GPS). If the deorbit burn guidance and control systems are assumed to correctly guide to the navigated state
and there are no navigation errors, then the dispersions in the initial entry state vector are negligible.

The vehicle mass should be known accurately, so for this study, a 3σ error of ±5% is assumed. Experience from the Space Shuttle program shows that the vehicle aerodynamics should be known to within ±5% for the force coefficients on the first flight. Only the stability derivatives and control effectiveness were missed significantly [7]. Even though the force coefficients may be known to excellent accuracy, reduced control effectiveness can reduce the possible trim angle of attack range reducing the maximum L/D achievable. Therefore, for this study, a ±10% dispersion in the lift and drag coefficients is considered. It should be noted that the first few flights of a new vehicle are usually targeted to the middle of the footprint to maximize margin and allow for accurate determination of the vehicle characteristics before the full ranging capability of the vehicle is used. After the first few flights, the aerodynamic characteristics should be known to within a few percent, so only about a ±3% dispersion must be considered.

The atmospheric dispersions were obtained from two sources. Reference [8] specifies the atmospheric dispersions to which aerospace vehicles must be designed. The average of the steady state winds at four geographic locations is shown in Figure 2 on page 79. This model was incorporated into the simulator environment with a magnitude scale factor to simulate less than worst-case winds. The wind direction was selected for each run made with winds and held constant throughout the trajectory. Reference [8] specifies Reference [9] as the source for atmospheric density dispersions. However, the recent Shuttle flights have provided estimated density data of a quality never before available. Atmospheric density profiles derived from Shuttle accelerometer measurements of the normal force acceleration and the estimated normal force coefficient and relative velocity vector are presented in Reference [10]. Figure 3 on page 80, taken from that report, shows the envelope of the
derived density profiles for the first 12 Shuttle flights. Of particular interest is the range of dispersions seen: -47% to +12%. Figures 4 on page 81 and 5 on page 82 show the density profiles for the STS-1 and STS-9 Shuttle flights. High frequency density shear components and constant density biases from the standard atmosphere are seen. For this study, constant density biases of ±30% and the Shuttle derived density profiles from Reference [10] were used.

2.3 REFERENCE TRAJECTORIES

The size of the footprint for a particular vehicle is determined by the range in vehicle L/D and the constraints placed on the trajectory such as heat rate limits. The edges of the footprint correspond to the use of maximum or minimum L/D. Maximum downrange or crossrange, for example, requires maximum L/D, while minimum downrange requires minimum L/D.

The determination of the optimal angle of attack and bank angle control histories for maximum crossrange and downrange has been the topic of many papers [11] [12] [13]. Wagner [12] used several optimization techniques to evaluate the maximum crossrange achievable for a multiphase bank angle history flown at maximum L/D. The multiphase bank profiles considered are shown in Figure 6 on page 83. It is seen that as the number of phases increases, the multiphase profile approaches the optimal continuous profile also shown in this figure. It was determined that a three-phase bank angle profile as illustrated in Figure 6 achieved almost the same crossrange as a continuous bank profile. This is shown in Figure 7 on page 84 reproduced here from that paper. Further, as the number of phases
increases, the optimum bank angle profile approaches a continuous profile that is almost linear with velocity as shown in Figure 8 on page 85. It was also shown that flying at the maximum L/D maximizes the crossrange attained.

This result is confirmed in Reference [13] which utilized a nonlinear programming technique to optimize the Space Shuttle trajectory for the maximum downrange and maximum crossrange cases. The maximum downrange trajectory requires flying at zero bank angle and at the angle of attack corresponding to maximum L/D as shown in Figure 9 on page 86. The control histories for the maximum crossrange case are shown in Figures 10 on page 87 and 11 on page 88. Again, the optimal control history is the angle of attack corresponding to maximum L/D and an almost linear bank angle profile with velocity.

Optimized trajectories for the ERV were reported in Reference [14]. These trajectories were determined using the Program to Optimize Simulated Trajectories (POST) [15] and imposed the following constraints on the trajectories:

1. Maximum heat rate of 125 BTU/sq ft/sec
2. Maximum heat load of 150K BTU/sq ft

The achievable footprint with these constraints, reported in Reference [14], is shown here in Figure 12 on page 89. Subsequently, the heat load limit was increased to 175K BTU/sq ft resulting in the larger footprint shown in Figure 12. As will be seen, these footprints omit a large area in the minimum downrange region that is achievable within the heating constraints. Also shown is the footprint of the Space Shuttle which has a maximum hypersonic L/D of 1.2 as compared with 1.8 for the ERV.

Figure 13 on page 90 shows the altitude history for the maximum downrange, maximum crossrange, and minimum downrange cases. Figures 14 on page 91 and 15 on page 92
show the bank angle and angle of attack histories for these trajectories. Figures 16 on page 93 and 17 on page 94 show the heat rate and heat load histories for these cases.

Figure 15 shows that the constant angle of attack corresponding to maximum L/D is flown for the edge of the footprint except for the minimum downrange case. For the minimum downrange case, the angle of attack corresponding to the minimum L/D on the back side of the L/D curve (high drag coefficient) is flown early, followed by a ramp in angle of attack starting at 1500 seconds after entry interface. This ramp corresponds to the vehicle actually turning around and flying slightly back uprange, so maximum L/D is desired later to maximize the distance flown uprange. The angle of attack for the maximum downrange case is slightly greater than that for maximum L/D because this trajectory exceeds the heat load limit if flown at maximum L/D. The maximum downrange region of the footprint is therefore limited by the heat load limit set for the ERV. If the limit were relaxed, flight at maximum L/D would allow a longer downrange trajectory.

Figure 14 shows that the bank angle profile for maximum crossrange is approximately linear with time which is almost linear with velocity, which suggests that a linear bank angle profile with velocity is sufficient. The maximum downrange case has a constant bank angle of zero which is again linear with velocity. The minimum downrange case does not have a linear bank profile. As was mentioned previously, for this case, the vehicle turns around and flies back uprange.

The results of these studies suggest that use of a constant angle of attack profile and a linear bank with velocity profile will capture a large portion of the achievable footprint. As will be seen in the results, these profiles suffice to capture most of the footprint reported in Reference [14] and additionally reach a large area in the minimum downrange region out-
side the reported footprint. Only a small area of the reported footprint in the minimum downrange region is unachievable.

Also of interest are the peaks in heat rate seen in Figure 16. Because the peaks in heat rate are very short, explicit control of the heat rate should be possible in the maximum heat rate regions without significantly impacting the guidance.

2.4 GUIDANCE APPROACH

The guidance design will attempt to maximize the size of the footprint while flying a constant angle of attack profile and a linear bank angle with velocity profile. The predictor algorithm integrates the equations of motion forward in time using the assumed control profile and the necessary environment and vehicle models. The corrector then determines (using multiple predicted trajectories with various control histories) the sensitivities of the final state constraints to the control variables. The sensitivities are then used to compute the required control variable values to reach the desired final state conditions. Heat rate control is provided locally during the regions of maximum heating without significantly affecting the assumed control histories. Also, in-flight measurements are utilized to increase the accuracy of the predicted trajectories by compensating for off-nominal conditions.

Such a simple profile for the maximum downrange and crossrange cases simplifies the modeling of the control histories in the predictor. The only remaining question is how much of the footprint this profile will capture. As will be seen in Subsection "4.2 Open-Loop
Footprint” on page 57, such a profile achieves almost complete coverage of the achievable footprint.

Also of concern is the linearity and convergence properties of the final state constraints with the control variables. As will be seen, over almost all of the footprint except near the edges, the constraints are highly linear and convergent with the control variables. Operationally, only about 75% of the achievable footprint is used to ensure guidance margin. Thus, the question of nonconvergence near the edges is avoided.
3.0 GUIDANCE DESIGN

3.1 INTRODUCTION

This section describes the implementational details of the guidance scheme described in the previous section. The equations of motion and environment and vehicle characteristics modeled in the predictor algorithm are described. The corrector algorithm to control the final state constraints with the two available control variables is derived. Also derived are the heat rate control and in-flight measurement algorithms. The heat rate control algorithm provides control of the peaks in stagnation heat rate during the early portion of entry. The in-flight measurement algorithm utilizes accelerations measured by the navigation system to more accurately model the expected environment and vehicle characteristics in the predictor algorithm. Because the predictor-corrector algorithm is computationally intensive, areas where significant execution time savings have been or can be realized are indicated. Program listings of the algorithm coded in the HAL/S computer language are presented in "Appendix B. ALGORITHM PROGRAM LISTINGS" on page 135.

As will be seen, the only inputs to the guidance system are the environment and vehicle models, the assumed control profiles, and the navigated state vector. The state vector is an input to any guidance system. The other inputs are developed for the analysis of any new vehicle. Therefore, the guidance system is highly transportable between vehicles because only the vehicle characteristics and aerodynamics model must be changed for a new vehicle.
3.2 UNIT TARGET VECTOR

The target aim point to which the vehicle is to be guided is specified by the longitude and geodetic latitude of the Terminal Area Energy Management (TAEM) interface point which occurs at 80K feet for the Shuttle. This point is selected based on the guidance algorithm employed during the TAEM guidance phase. TAEM guidance provides precise control of vehicle energy during the final stages of entry to guide to a specified runway with acceptable energy. For computational ease, the longitude and geodetic latitude are converted to a target unit vector in Earth-fixed coordinates by first computing the geocentric latitude from,

\[ \phi_c = \tan^{-1}\left( \frac{\tan(\phi_g)}{k} \right) \]  

where,

\[ k = \left( \frac{R_{\text{equator}}}{R_{\text{pole}}} \right)^2 = \left( \frac{1}{1 - f_{\text{earth}}} \right)^2 \]  

The unit target vector is then computed from,

\[ \vec{i}_t^e = \begin{bmatrix} \cos(\phi_c) \cos(\lambda) \\ \cos(\phi_c) \sin(\lambda) \\ \sin(\phi_c) \end{bmatrix} \]  

Alternatively,
\[ \vec{i}^{EF} = \begin{bmatrix} i_x \\ i_y \\ i_z \end{bmatrix} \]

where,

\[ i_z = \sin(\phi_c) \]  
(5)

\[ i_x = \cos(\lambda) \sqrt{1 - i_z^2} \]  
(6)

\[ i_y = \text{sign}(\lambda) \sqrt{1 - i_z^2 - i_x^2} \]  
(7)

### 3.3 COMMANDED ATTITUDE COMPUTATION

Because the predictor can not be executed as frequently as analytic guidance algorithms early in the entry, and because it in fact does not have to be executed as frequently, it is necessary to update the commands sent to the vehicle autopilot more frequently than the predictor-corrector execution rate. Typically, this would be done at the rate of current analytic guidance algorithms, e.g., the Space Shuttle rate of .52 hz. The commanded bank angle, \( \phi_{cmd} \), is computed for the linear bank with velocity profile as shown in Figure 18 on page 95 from the desired bank angle, \( \phi_d \), and the current navigated inertial velocity magnitude, \( V_i \).

\[ \phi_{cmd} = \phi_d \frac{V_f - V_i}{V_{mi} - V_f} \]  
(8)
to yield the near-optimal linear bank with velocity profile. The desired angle of attack control history is a constant angle of attack, and therefore,

$$\alpha_{\text{cmd}} = \alpha_d$$  \hspace{1cm} (9)

As implemented in the current design, the guidance algorithm executive is executed at 1.0 hz. The attitude commands are updated at this frequency using Eq. (8) and (9). The predictor-corrector algorithm is executed at .02 hz. during the entire entry phase, although it is practical to run it much more frequently late in the trajectory when the length of the trajectory to be predicted is short. The possible execution rate of the predictor-corrector for a typical flight computer is addressed in Subsection “4.8 Algorithm Execution Time” on page 66.

3.4 CORRECTOR ALGORITHM

The corrector algorithm is executed to update the commanded attitude control history to be flown. The guidance algorithm controls to two final state constraints, downrange error and crossrange error, using two control variables, a constant angle of attack and the intercept of the bank profile at the entry interface velocity as shown in Figure 18 on page 95 and expressed in Eq. (8).

Expanding the downrange and crossrange errors in a Taylor series expansion of the control variables and neglecting the second-order and higher terms yields,

$$\Delta DR_e = \frac{\partial DR_e}{\partial \alpha_d} \Delta \alpha_d + \frac{\partial DR_e}{\partial \phi_d} \Delta \phi_d + \ldots$$  \hspace{1cm} (10)
\[ \Delta CR_e = \frac{\partial CR_e}{\partial \alpha_d} \Delta \alpha_d + \frac{\partial CR_e}{\partial \phi_d} \Delta \phi_d + \ldots \] \hspace{1cm} (11)

To intercept the target, the change in the constraint errors must null the predicted errors, or,

\[ \Delta DR_e = -DR_e \] \hspace{1cm} (12)

\[ \Delta CR_e = -CR_e \] \hspace{1cm} (13)

Equations (10) through (13) provide a set of two simultaneous equations in two unknowns,

\[
\begin{bmatrix}
\frac{\partial DR_e}{\partial \alpha_d} & \frac{\partial DR_e}{\partial \phi_d} \\
\frac{\partial CR_e}{\partial \alpha_d} & \frac{\partial CR_e}{\partial \phi_d}
\end{bmatrix}
\begin{bmatrix}
\Delta \alpha_d \\
\Delta \phi_d
\end{bmatrix}
= 
\begin{bmatrix}
-DR_e \\
-CR_e
\end{bmatrix}
\] \hspace{1cm} (14)

which are solved for the control variable changes required,

\[ \Delta \alpha_d = \left( \frac{\partial DR_e}{\partial \phi_d} CR_e - \frac{\partial CR_e}{\partial \phi_d} DR_e \right) / \text{det} \] \hspace{1cm} (15)

\[ \Delta \phi_d = \left( \frac{\partial CR_e}{\partial \alpha_d} DR_e - \frac{\partial DR_e}{\partial \alpha_d} CR_e \right) / \text{det} \] \hspace{1cm} (16)

where det is the determinant of the matrix in Eq. (14). The partial derivatives are approximated by finite difference equations of the form,

\[ \frac{\partial DR_e}{\partial \phi_d} = \frac{DR_e(\phi_d = \phi_3) - DR_e(\phi_d = \phi_1)}{\phi_3 - \phi_1} \] \hspace{1cm} (17)

There are four partial derivatives that must be evaluated. They can be evaluated from three predicted trajectories with control histories selected as:
1. $\alpha_1 = \alpha'_d, \quad \phi_1 = \phi'_d$

2. $\alpha_2 = \alpha'_d + \delta x_d, \quad \phi_2 = \phi'_d$

3. $\alpha_3 = \alpha'_d, \quad \phi_3 = \phi'_d + \delta \phi_d$

where the primes denote the control variables from the previous guidance solution. The new guidance commands are then,

$$\alpha_d = \alpha'_d + \Delta \alpha_d \quad (18)$$

$$\phi_d = \phi'_d + \Delta \phi_d \quad (19)$$

Protection must be provided for the case where the determinant in Eq. (15) and (16) is small or identically zero which corresponds to a loss of control authority of the control variables over the control constraints. In this case, no change is made to the control variables, and the guidance command from the previous cycle is used. As the vehicle approaches the TAEM interface altitude, the control authority decreases. Large control variable changes become necessary to null the constraint errors in the short flight time remaining. This problem can be avoided in one of two ways. First, the guidance commands can be frozen at a selected point before the termination altitude. For entry guidance, this approach is not preferred because the vehicle still has not landed. Alternatively, the target aim point can be lowered below the TAEM interface altitude point at which TAEM guidance is activated. The decreasing control authority problem is therefore reduced.

For the simulated trajectories in this report, the first approach is employed because it is desired to evaluate guidance performance by considering the dispersions in the final state at the TAEM interface altitude. Because the guidance algorithm controls only the final state and not the intermediate states, it is necessary to target for the point at which the guidance is terminated.
3.5 PREDICTOR ALGORITHM

3.5.1 Introduction

The predictor algorithm is a simplified three-degree-of-freedom (3-DOF) trajectory simulator complete with models for those environment and vehicle characteristics necessary to model the translational equations of motion of the vehicle. Because the predictor is computationally intensive, the algorithm must be carefully designed to minimize computation, and the coding of the algorithm in a particular computer language should make use of any language-specific features to reduce computational requirements. Also, because the corrector only utilizes the final state vector errors to correct the control variables, only the accuracy of the predicted final state vector need be considered in selecting those effects to be modeled.

The environmental effects of concern for the long trajectories flown by entry vehicles over large altitude and velocity ranges are:

1. Variation of atmospheric properties with altitude
2. Earth oblateness effect on gravity vector
3. Effect of atmospheric rotation with Earth on relative velocity vector
4. Movement of runway due to Earth rotation

The vehicle characteristics of importance are:

1. Vehicle mass
2. Aerodynamic coefficient variation with flight regime
3. Aerodynamic coefficient variation with angle of attack
4. Control history during trajectory

Dispersions to be considered are:
1. Vehicle mass variation from nominal
2. Winds
3. Atmospheric density variation from nominal atmosphere
4. Aerodynamic coefficient variation from nominal

These dispersions can be measured in-flight because they affect the sensed acceleration measured by the vehicle’s inertial navigation system. The estimation of these dispersions is discussed in Subsection “3.6 Estimators” on page 48.

The predictor performs the following computations upon being called by the corrector with a desired control variable history:

1. Initialize the predictor state to the navigated state vector
2. Compute any ancillary parameters from the state vector
3. Compute the total acceleration vector from the predictor state vector and the environment and vehicle models using the control variable profiles specified by the corrector
4. Integrate the equations of motion forward in time one time step
5. Check the predictor termination conditions
   a. Repeat steps 3 and 4 if the conditions are not met
   b. Continue on to step 6 if the conditions are met
6. Compute and return to the corrector the final predicted state errors from the target state vector and the predicted final state vector

3.5.2 Equations of Motion

The corrector provides a time-homogeneous navigated state vector comprised of,

1. The GMT time tag of the state vector, \( t_{\text{GMT}} \)
2. The inertial position vector, \( \vec{R} \)
3. The inertial velocity vector, \( \vec{V} \)
Also provided is the control variable history to be followed for the prediction. The equations of motion to be integrated are,

\[
\frac{d R_i}{dt} = V_i, \qquad (20)
\]

\[
\frac{d V_i}{dt} = A_i, \qquad (21)
\]

The acceleration is computed from the atmosphere and vehicle models as follows,

\[
A_i = \frac{F_i}{m} = \ddot{g}_i + \ddot{a}_{aero}, \qquad (22)
\]

The gravitational acceleration, \( \ddot{g}_i \), is computed including the \( J_2 \) term as,

\[
\ddot{g}_i = -\frac{\mu}{|R_i|^2} i_g
\]

where,

\[
i_g = \dot{R}_i + \frac{3}{2} J_2 \frac{R_{\text{equator}}^2}{|R_i|^2} ((1 - 5 z^2) R_i + 2 z i_{\text{pole}})
\]

and,

\[
z = \dot{R}_i \cdot i_{\text{pole}}
\]

The aerodynamic acceleration, \( \ddot{a}_{aero} \), is computed from,

\[
\ddot{a}_{aero} = a_{\text{lift}} \dot{v}_{\text{lift}} + a_{\text{drag}} \dot{v}_{\text{drag}}
\]

where,

\[
a_{\text{lift}} = \frac{C_L \bar{q} S}{m}
\]
\[ a_{\text{drag}} = \frac{C_D \, \bar{q} \, S}{m} \]  

(28)

\[ \bar{q} = \frac{1}{2} \rho \, V_R^2 \]  

(29)

\[ V_R^2 = \vec{V}_R \cdot \vec{V}_R \]  

(30)

\[ \vec{V}_R = \vec{V}_I - \vec{\omega}_{\text{earth}} \times \vec{R}_I \]  

(31)

\[ \vec{i}_{\text{drag}} = -\frac{\vec{V}_R}{|\vec{V}_R|} \]  

(32)

\[ \vec{i}_{\text{lift}} = (\vec{i}_{\text{drag}} \times \vec{i}_{\text{lat}}) \cos(\phi) + \vec{i}_{\text{lat}} \sin(\phi) \]  

(33)

\[ \vec{i}_{\text{lat}} = \frac{\vec{i}_R \times \vec{i}_{\text{drag}}}{|\vec{i}_R \times \vec{i}_{\text{drag}}|} \]  

(34)

\[ \vec{i}_R = \frac{\vec{R}_I}{|\vec{R}_I|} \]  

(35)

The acceleration due to lift, \( a_{\text{lift}} \), is more easily computed from,

\[ a_{\text{lift}} = \frac{L}{D} \, a_{\text{drag}} \]  

(36)

since the nominal lift-to-drag ratio, \( L/D \), is corrected using in-flight accelerometer measurements of the actual vehicle sensed aerodynamic accelerations.

The atmospheric density, \( \rho \), is computed by the atmosphere model using the position vector, \( \vec{R}_I \). The 1962 U.S. Standard Atmosphere model is employed and is described in Reference [16]. If another atmosphere model is selected as being a more accurate estimate of the day-of-flight atmosphere, this model would replace the 1962 U.S. Standard Atmosphere model. An operational vehicle might employ monthly or seasonal atmospheres from such
sources as the GRAM Atmosphere [9] or even day-of-flight measurements to more accurately model the expected atmosphere in the predictions. The level of accuracy required in the atmosphere model will depend on the vehicle ranging capability and the amount of that capability to be used for a particular entry. Entries to the edges of the footprint will demand a very accurate atmosphere model.

The aerodynamic coefficients are highly vehicle dependent. To minimize computational requirements, they should be updated during the prediction as infrequently as possible. Of course, the update frequency required depends on the trajectory flown and the rate of change of the aerodynamic coefficients with flight regime change. The aerodynamic coefficient model for the ERV is presented in “Appendix A. ERV AERODYNAMICS MODEL” on page 133.

The density, \( \rho \), from the atmosphere model and the lift-to-drag ratio, \( L/D \), from the aerodynamic model are both corrected by in-flight measurements as covered in Subsection “3.6 Estimators” on page 48. The estimated dispersions are compensated for using the following equations,

\[
\rho = K_\rho \rho_{std} \tag{37}
\]

\[
\frac{L}{D} = K_{\frac{L}{D}} \left( \frac{C_L}{C_D} \right)_{nom} \tag{38}
\]

where the density and lift-to-drag ratio scale factors, \( K_\rho \) and \( K_{\frac{L}{D}} \), are provided by the estimator and are held constant throughout the prediction being made.

The control history to be followed is the constant angle of attack, \( \alpha_c \), and the linear bank angle with velocity, \( \phi \). The latter is computed from,
\[ \phi = \phi_d \frac{V_i - V_f}{V_{bi} - V_f} \]  \hspace{1cm} (39)

where \( \phi_d \) is the intercept of the linear bank angle profile at the entry interface velocity, \( V_{bi} \). Because the entry interface and final velocities are not known a priori, and because small variations in them have little effect on the predicted trajectory compared with the selected control variables' values, the velocities are selected as constant values that cover all expected dispersions in the entry and final velocities. These values are,

\[
V_{bi} = 26,000 \text{ ft/sec}
\]

\[
V_f = 1,000 \text{ ft/sec}
\]

3.5.3 Integration of the Equations of Motion

The equations of motion are integrated using the 4th order Runge-Kutta algorithm with a variable time step to minimize the number of time steps required to integrate the trajectory to the final state. The 4th order Runge-Kutta algorithm requires four evaluations of the acceleration per time step, but permits a time step more than four times as large as an algorithm requiring only one acceleration evaluation per time step. The Runge-Kutta solution \([17]\) for the differential equations of motion of the form,

\[
\frac{d \vec{R}_i}{dt} = \vec{V}_i \]

\[
\frac{d \vec{V}_i}{dt} = f(t, \vec{R}_i, \vec{V}_i) \]  \hspace{1cm} (40)

\[
\frac{d \vec{V}_i}{dt} = f(t, \vec{R}_i, \vec{V}_i) \]  \hspace{1cm} (41)

is,
\[ \bar{R}_i(t+\Delta t) = \bar{R}_i(t) + \frac{\Delta t}{6} (K_0 + 2K_1 + 2K_2 + K_3) \]  \hspace{1cm} (42)

\[ \bar{V}_i(t+\Delta t) = \bar{V}_i(t) + \frac{\Delta t}{6} (K'_0 + 2K'_1 + 2K'_2 + K'_3) \]  \hspace{1cm} (43)

where,

\[ K_0 = \bar{V}_i \]  \hspace{1cm} (44)

\[ K_1 = (\bar{V}_i + \frac{K'_0}{2}) \]  \hspace{1cm} (45)

\[ K_2 = (\bar{V}_i + \frac{K'_1}{2}) \]  \hspace{1cm} (46)

\[ K_3 = (\bar{V}_i + K'_2) \]  \hspace{1cm} (47)

\[ K'_0 = f(t, \bar{R}_i, \bar{V}_i) \]  \hspace{1cm} (48)

\[ K'_1 = f(t + \frac{\Delta t}{2}, \bar{R}_i + \Delta t \frac{K_0}{2}, \bar{V}_i + \Delta t \frac{K'_0}{2}) \]  \hspace{1cm} (49)

\[ K'_2 = f(t + \frac{\Delta t}{2}, \bar{R}_i + \Delta t \frac{K_1}{2}, \bar{V}_i + \Delta t \frac{K'_1}{2}) \]  \hspace{1cm} (50)

\[ K'_3 = f(t + \Delta t, \bar{R}_i + \Delta t \bar{K}_2, \bar{V}_i + \Delta t \bar{K}'_2) \]  \hspace{1cm} (51)

The time step is varied inversely with the total acceleration on the vehicle. This method of time step control was selected because of its simplicity. The time step control equation is of the form,

\[ \Delta t = \frac{K_{\Delta t}}{|A_i|} \]  \hspace{1cm} (52)
and the time step is limited between a minimum and maximum value,

$$\Delta t = \text{midval}(\Delta t_{\text{min}}, \Delta t, \Delta t_{\text{max}})$$  \hfill (53)

The optimization of the integration algorithm is important in developing a flight quality algorithm, but is beyond the scope of this study. Higher-order integration algorithms with time step control methods [17] may yield significant reductions in the required computation time.

3.5.4 Termination Conditions for the Predictor

After each integration time step, the predicted state is compared with the termination condition. The termination condition is defined by the altitude of TAEM interface (80K feet). Because the predicted state at the TAEM interface altitude may have a relatively large altitude rate and range rate, the predictor must be terminated accurately to provide an altitude-homogeneous set of predicted state errors. Also, the variable time step control may allow large integration time steps if the acceleration is low near the final state, further complicating the task of terminating accurately. Reasonable altitude homogeneity is ensured by forcing use of the minimum integration time step starting some safe altitude above the termination altitude.

3.5.5 Final State Error Computation

The final state errors are computed from the unit target vector and the predicted final state vector. Because the target is fixed to the Earth and moves a significant distance dur-
ing the long entry trajectory, the rotation of the Earth must be considered. This is done by transforming the final state vector from inertial to Earth-fixed coordinates with the rotation matrix $M^E_F$ which is computed from the predicted termination time, the known orientation of the Earth at some epoch time, and the known rotation rate of the Earth. This computation is performed in the Earth-Fixed-From-Reference subroutine of the predictor-corrector which may actually be a GN&C utility function also employed by the navigation principal function.

The downrange and crossrange errors are defined as shown in Figure 19 on page 96. The errors are computed by first computing the downrange (in-plane) and crossrange (perpendicular) directions as follows,

\begin{align}
\vec{R}_{i}^{E} &= M_{i}^{E} \vec{R}_{i} \\
\vec{V}_{i}^{E} &= M_{i}^{E} \vec{V}_{i} \\
\vec{V}_{p_{\text{perpen}}}^{E} &= \frac{\vec{i}_{R}^{E} \times \vec{V}_{i}^{E}}{|\vec{i}_{R}^{E} \times \vec{V}_{i}^{E}|} \\
\vec{V}_{p_{\text{inplane}}}^{E} &= \frac{\vec{i}_{R}^{E} - (\vec{i}_{E}^{E} \cdot \vec{i}_{p_{\text{perpen}}}^{E}) \vec{i}_{p_{\text{perpen}}}^{E}}{|\vec{i}_{R}^{E} - (\vec{i}_{E}^{E} \cdot \vec{i}_{p_{\text{perpen}}}^{E}) \vec{i}_{p_{\text{perpen}}}^{E}|}
\end{align}

The downrange and crossrange errors are then,

\begin{align}
DR_{e} &= R_{\text{equator}} \cos^{-1}(\vec{i}_{R}^{E} \times \vec{i}_{p_{\text{inplane}}}^{E}) \text{sign}((\vec{i}_{R}^{E} \times \vec{i}_{p_{\text{inplane}}}^{E}) \cdot \vec{i}_{p_{\text{perpen}}}^{E}) \\
CR_{e} &= R_{\text{equator}} \cos^{-1}(\vec{i}_{p_{\text{inplane}}}^{E} \cdot \vec{i}_{R}^{E}) \text{sign}((\vec{i}_{p_{\text{inplane}}}^{E} \times \vec{i}_{R}^{E}) \cdot (\vec{i}_{p_{\text{perpen}}}^{E} \times \vec{i}_{p_{\text{perpen}}}^{E}))
\end{align}
These errors have the dimensions of $R_{\text{equator}}$ and are converted to nautical miles for ease of interpretation.

### 3.5.6 Algorithm Coding

A few comments regarding implementation of the predictor are appropriate. The computations required to update the aerodynamic coefficients are the major computational load for the predictor. It was found that it is not necessary to update the aerodynamics on each of the four acceleration evaluations of the 4th order Runge-Kutta algorithm. They are therefore only evaluated once each integration time step. The computational load could be reduced further if they are only updated when the independent variables (altitude, viscous interaction parameter, and Mach Number) change by a significant amount from the previous update. Also, although not done in this implementation, the aerodynamic coefficients should be curve-fit if possible to avoid a table lookup and interpolation implementation. It is noted in Figures 20 on page 97 and 21 on page 98 that the aerodynamic coefficients do not change very much below 300K feet until the Mach Number decreases below 2, so perhaps, two tables or curve-fits would suffice instead of the thirty tables currently used.

### 3.6 ESTIMATORS

The final state predicted by the predictor algorithm for a particular control history is a function of the assumed environment and vehicle characteristics. The accuracy of the predicted final state can be increased, and hence, the guidance margin increased, if in-flight
measurements are utilized to make the assumed models more accurately reflect the conditions actually experienced by the vehicle.

The accelerations modeled in the predictor are due to gravity and the aerodynamic forces. The gravity acceleration can be modeled to sufficient accuracy using standard gravity models. However, the aerodynamic accelerations are subject to significant variations due to uncertainties in the atmospheric density, atmospheric winds, vehicle aerodynamics, and vehicle mass. These uncertainties can be compensated for in the predictor by applying a multiplicative scale factor to the lift and drag accelerations modeled in the predictor that is equal to the ratio of the actual accelerations experienced to the predicted accelerations at any point in the trajectory.

The measured lift and drag accelerations are derived from the inertial measurement system sensed acceleration assuming a zero sideslip angle as follows,

\[
\hat{\Delta}_{\text{drag}} = -\Delta \cdot \frac{V_R}{|V_R|} \quad (61)
\]

\[
\hat{\Delta}_{\text{lift}} = \sqrt{\Delta \cdot \Delta - \Delta_{\text{drag}}} \quad (62)
\]

where the inertial acceleration, \(\Delta\), is computed by back-differencing the accumulated sensed velocity counts from the inertial measurement unit,

\[
\Delta = \frac{V_{imu} - V_{imu \text{ past}}}{\Delta t_{imu}} \quad (63)
\]

In the predictor, the aerodynamic accelerations are,

\[
\Delta_{\text{drag}} = \frac{C_D S}{m} \frac{1}{2} \rho V_R^2 \quad (64)
\]
Data from the Shuttle program [10] shows that the primary dispersion affecting the aerodynamic acceleration is in the atmospheric density. Further, over large altitude ranges, this dispersion can be modeled to an accuracy sufficient for the prediction process as a constant multiplicative bias. Therefore, for implementational purposes, the dispersion in the aerodynamic accelerations due to the atmospheric uncertainties will be lumped into a density scale factor as follows,

\[ \hat{K}_p = \frac{\hat{\rho}}{\rho_{std}} \]  

where,

\[ \hat{\rho} = \frac{2 \hat{a}_{drag}}{V_\alpha^2} \left( \frac{m}{C_D S} \right)_{nom} \]  

and the values for the nominal vehicle characteristics and the nominal atmospheric density are determined using the predictor models for the vehicle state at the time of the measurement. Because the nominal ballistic coefficient is assumed in deriving the measured density, and the measured acceleration is due to the actual ballistic coefficient, uncertainties in the ballistic coefficient will be reflected in the measured density. The equation for the drag acceleration in the predictor is then,

\[ a_{drag} = \left( \frac{C_D S}{m} \right)_{nom} \frac{1}{2} V_\alpha^2 K_p \rho_{std} \]  

or substituting for \( K_p \) from Eq. (66) yields,

\[ a_{drag} = \left( \frac{C_D S}{m} \right)_{nom} \frac{1}{2} V_\alpha^2 \hat{\rho} \]  

Substituting for \( \hat{\rho} \) from Eq. (67) then yields,
so the modeled drag is corrected for the dispersed drag coefficient, density, relative velocity, and vehicle mass.

In general, the measured drag acceleration is a noisy signal and will exhibit short term variations due to short lived local atmospheric dispersions [10]. Filtering of the density scale factor is therefore necessary and is implemented using a first-order filter,

\[ K_p = (1 - K_i) K_p + K_1 \frac{\hat{p}}{\rho_{\text{std}}} \]  

which has a time constant, \( \tau_p \), of,

\[ \tau_p = \frac{\Delta t}{\ln(1 - K_i)} \]  

where \( \Delta t \) is the sample rate of the measured drag acceleration, and \( K_i \) is the filter gain. A similar lift-to-drag ratio scale factor is derived and applied to the lift acceleration,

\[ a_{\text{lift}} = K_{L/D} \left( \frac{L}{D} \right)_{\text{nom}} a_{\text{drag}} \]  

where,

\[ K_{L/D} = \frac{\left( \frac{L}{D} \right)}{\left( \frac{L}{D} \right)_{\text{nom}}} \]
and,

\[
\left( \frac{L}{D} \right) = \frac{\hat{a}_{lin}}{\hat{a}_{drag}}
\]  

(75)

Again, filtering is necessary,

\[
K_L = (1 - K_2) K_{L_{\text{nom}}} + K_2 \frac{\left( \frac{L}{D} \right)}{\left( \frac{L}{D} \right)_{\text{nom}}}
\]

(76)

yielding a time constant, \( \tau_{L_{\text{nom}}} \), of,

\[
\tau_{L_{\text{nom}}} = -\frac{\Delta t}{\ln(1 - K_2)}
\]

(77)

A time constant of 25 seconds was selected for both the density and L/D filters. This value filtered out the high frequency density shear components seen in the Shuttle profiles while still providing adequate response to long term disturbances.

3.7 HEAT RATE CONTROL

The primary trajectory constraint on entry vehicles is the maximum heat rate the vehicle can withstand. In general, the thermal protection system material is selected to withstand
the maximum local heat rate on any particular portion of the vehicle, and the material thickness is selected to withstand the total integrated heat load over the trajectory. Accurate pre-flight predictions of the expected heat rate during entry can significantly reduce the thermal protection system weight, yielding significant performance increases for an entire mission.

Inspecting the reference trajectories in Figure 16 on page 93 shows that sharp peaks in the heat rate occur. If these peaks are accurately controlled, and this control can be accomplished using only short term departures from the predictor assumed control history, no significant departure will occur from the desired trajectory.

Heat rate control can be accomplished using either angle of attack, bank angle, or a combination of both. Of these, bank angle alone is preferred because a constant angle of attack trajectory is assumed and because angle of attack changes the vehicle drag coefficient resulting in a rapid change in energy rate and a rapid departure from the desired trajectory. Also, most entry vehicles restrict the angle of attack range during maximum heat rate regions to reduce the area on the vehicle that must be protected from the high heat rate. Although the ERV does not need to restrict the angle of attack range, and hence, the guidance does not provide for such a capability, the restriction can be handled by replacing the constant angle of attack control history by a reference angle of attack control history about which a constant angle of attack bias is applied for control.

Heat rate control is accomplished by computing the incremental bank angle required to fly along the specified heat rate boundary (assumed to be a constant heat rate for any flight regime) and then modulating bank angle according to the guidance value or the guidance value plus the incremental lift for heat rate control, whichever requires more lift up. Hence, no effort is made to pull the vehicle down into the atmosphere to follow the heat rate bound-
ary; instead, lift up is applied if the vehicle is flying “too low”. The incremental lift for heat rate control is computed to provide a second-order control response as follows,

\[
\cos(\Delta \phi) = \frac{K_0}{q} (\ddot{q} - \ddot{q}_{\text{des}}) + \frac{K_5}{q} (\dot{q} - \dot{q}_{\text{lim}})
\]  

(78)

To fly along a constant heat rate boundary,

\[
\dot{q}_{\text{lim}} = \text{constant}
\]  

(79)

and the desired rate of change of heat rate, \( \ddot{q}_{\text{des}} \), is,

\[
\ddot{q}_{\text{des}} = 0
\]  

(80)

so,

\[
\cos(\Delta \phi) = \frac{K_0}{q} \ddot{q} + \frac{K_5}{q} (\dot{q} - \dot{q}_{\text{lim}})
\]  

(81)

The stagnation heat rate is determined using the Engineering Correlation Formula [18] for a one foot radius reference sphere as,

\[
\dot{Q} = 17700 \sqrt{\rho} \left( \frac{V_R}{10000} \right)^{3.05}
\]  

(82)

The time rate of change of heat rate, \( \ddot{Q} \), is determined by back-differencing the heat rate between guidance cycles,

\[
\ddot{Q} = \frac{\dot{Q} - \dot{Q}_{\text{last}}}{\Delta t}
\]  

(83)

The equations of motion assuming small flight path angle yield,

\[
\ddot{h} = \frac{c_L \bar{q} \bar{S}}{m} \cos(\phi) - g
\]  

(84)
Considering only the perturbations due to the incremental lift, \( \cos(\Delta \phi) \), from Eq. (81) yields,

\[
\ddot{h} - \frac{C_l S}{m} K_\Delta \dot{h} - K_\Delta (\dot{h} - \dot{h}_{\text{lim}}) = 0
\]

Proper selection of the gains \( K_\Delta \) and \( K_\Delta \) is accomplished by linearizing Eq. (85) in altitude and assuming that the time rate of change of \( V_R \) is small compared to the change in \( \sqrt{\rho} \).

With these assumptions,

\[
\dot{h} = 17700 \left( \frac{V_R}{10000} \right)^{3/5} \frac{d\sqrt{\rho}}{dh} h
\]

and,

\[
\ddot{h} = 17700 \left( \frac{V_R}{10000} \right)^{3/5} \frac{d\sqrt{\rho}}{dh} \frac{dh}{dt}
\]

Therefore, the homogeneous second-order differential equation in altitude is,

\[
\ddot{h} + K K_\Delta \dot{h} + K K_\Delta h = 0
\]

where,

\[
K = -\frac{C_l S}{m} 17700 \left( \frac{V_R}{10000} \right)^{3/5} \frac{d\sqrt{\rho}}{dh}
\]

The natural frequency and damping ratio of the second-order differential equation are,

\[
\omega_n = \sqrt{K K_\Delta}
\]

\[
\zeta = \frac{K K_\Delta}{2 \omega_n}
\]

or alternatively, for a desired natural frequency and damping ratio, \( K_\Delta \) and \( K_\Delta \) are selected as,
\[ K_{\dot{\delta}} = \frac{\omega_n^2}{K} \]  
(92)

\[ K_{\dot{\phi}} = \frac{2 \zeta \omega_n}{K} \]  
(93)

The derivative in Eq. (89) can be evaluated assuming an exponential atmosphere of the form,

\[ \rho = \rho_0 e^{-\left(\frac{h}{h_s}\right)} \]  
(94)

yielding,

\[ \frac{d\sqrt{\rho}}{dh} = \frac{\sqrt{\rho_0}}{2h_s} e^{-\left(\frac{h}{h_s}\right)} \]  
(95)

This logic is contained in the guidance algorithm in the Heat Rate Control subroutine. The incremental lift required for heat rate control is provided to the Attitude Command subroutine which adds it into the guidance command if it requires more lift up than the guidance command. This occurs when the incremental lift given by Eq. (81) is greater than zero,

\[ \cos(\Delta \phi) > 0 \]  
(96)

Appropriate values of the natural frequency and damping ratio were determined parametrically as,

\[ \omega_n = 0.10 \text{ rad/sec} \]  
(97)

\[ \zeta = 1.00 \]  
(98)
4.0 PERFORMANCE

4.1 SIMULATOR

Open-loop and closed-loop entry trajectories were simulated for the Entry Research Vehicle (ERV) using a derivative of the 6-DOF Aeroassist Flight Experiment Simulator (AFESIM) [19] developed at The Charles Stark Draper Laboratory which is coded in the HAL computer language. For this study, the aerodynamic model described in "Appendix A. ERV AERODYNAMICS MODEL" on page 133 and the wind model shown in Figure 2 on page 79 were incorporated into the AFESIM. The characteristics of the ERV [14] are listed in Table 1 on page 73. The entry conditions with which all trajectories were initialized are also listed in Table 1 on page 73. Because only the performance characteristics of the guidance were being evaluated, the simulator was operated in the 3-DOF mode.

4.2 OPEN-LOOP FOOTPRINT

Open-loop trajectories were run using the constant angle of attack and linear bank with velocity profiles to determine the portion of the footprint achievable. All trajectories were terminated at the TAEM interface altitude of 80K feet, so the footprint can be increased about 100 nautical miles in all directions due to the range flown below 80K feet.
Figure 22 on page 99 shows the lift-to-drag ratio, \( L/D \), for the ERV at Mach 10 versus angle of attack, \( \alpha \). It is seen that maximum \( L/D \) is obtained at an angle of attack of 15 degrees. It is desirable to fly on the back side of the \( L/D \) curve (angle of attack greater than 15 degrees) so as to maximize the drag coefficient for a given \( L/D \). This reduces heating by causing a quicker loss of velocity early in entry than flying at the same \( L/D \) on the front side of the \( L/D \) curve. The \( L/D \) versus angle of attack curve shows the same shape with the maximum \( L/D \) at 15 degrees for all flight regimes with only a variation in the magnitude of \( L/D \) across the angle of attack range. Therefore, angle of attack is modulated between 15 and 50 degrees for the footprint with 15 degrees corresponding to maximum \( L/D \) and 50 degrees corresponding to minimum \( L/D \).

The open-loop footprint is shown in Figure 23 on page 100. Also shown for comparison is the reported footprint for a heat load limit of 175K BTU/sq ft (shown earlier in Figure 12 on page 89). That footprint included the range flown below 80K feet, hence the slight differences. It is seen that almost the entire reportedly achievable footprint is captured with the assumed control profile. Most importantly, all of the maximum crossrange region is reached when the range flown below 80K feet is included. Also, most of the minimum downrange region of the footprint was captured even though the control profiles used do not correspond to the optimal profiles determined using POST and shown in Figures 14 on page 91 and 15 on page 92. Additionally, the footprint reported in Reference [14] does not include the large area in the minimum downrange region that the open-loop trajectories reached. The small area not reached in the minimum downrange region by the control profiles is relatively unimportant because the downrange ranging capability of the vehicle can be adjusted by changing the deorbit time. A vehicle in low earth orbit travels at about four nautical miles per second, so downrange is easily adjusted while on-orbit.
Because the predictor-corrector guidance algorithm will follow the same control histories as used to generate the open-loop footprint for a nominal trajectory, the guidance algorithm can reach all of the open-loop footprint for nominal conditions. It is seen that the achievable footprint is bounded by the heat rate and heat load limits imposed on the ERV. At least an additional 2000 nautical miles of ranging capability in the downrange direction exists if the heat limits are relaxed.

4.3 EFFECT OF DISPERSIONS ON FOOTPRINT

The effect of dispersions on the achievable footprint was determined by repeating the open-loop trajectories with the dispersions discussed in Subsection "2.2 Dispersions" on page 26. The worst-case (3σ) dispersions are summarized in Table 2 on page 73. Table 3 on page 74 shows the dispersions in downrange and crossrange for three of the control histories in the maximum downrange region of the footprint. It is seen that only variations in the lift and drag coefficients cause significant dispersions in the final state. Also, it is seen that the effect of a +10% $C_l$ dispersion is the same as that of a -10% $C_o$ dispersion. This is expected because both dispersions cause the same increase in the vehicle L/D. The same occurs for a -10% $C_l$ dispersion and a +10% $C_o$ dispersion, both of which decrease the vehicle L/D.

The effects of the dispersions on trajectories to the maximum crossrange region of the footprint are seen in Table 4 on page 75. Again, it is seen that aerodynamic dispersions have the greatest effect. A dispersion that increases L/D increases the range, while a dispersion that decreases L/D decreases the range.
Table 5 on page 76 shows the effects of the dispersions on the minimum downrange region of the footprint. The worst-case range dispersions again occur for the aerodynamic dispersions.

4.4 ESTIMATOR PERFORMANCE

Figure 24 on page 101 shows the time response of the density filter with a 25 second time constant for the STS-9 atmosphere. This trajectory also has dispersions of $+1.9\%$ in $C_D$, $-3.2\%$ in mass, and a $63.8\%$ crosswind. Therefore, the filter output does not follow the actual density dispersion also shown in the figure. When the acceleration level is below 0.07 g’s, the measurements are not incorporated, so the filter is inactive before 300 seconds and from 600 to 850 seconds. As the velocity drops, the wind becomes a greater contributor to the measured density error, hence the divergence in the measured density ratio starting at 1000 seconds. Figure 25 on page 102 shows the response of the L/D filter with a 25 second time constant for a $-1.9\%$ $C_L$ and a $+1.9\%$ $C_D$ dispersion. Again, the winds affect the measurement by creating errors in the navigated angle of attack, so the estimated L/D ratio is slightly in error.

The use of an air data system like the Shuttle Entry Air Data System (SEADS) could significantly improve the estimation process by providing accurate estimates of the angle of attack, atmospheric density, and wind magnitude and direction. More accurate estimates will increase the guidance margin, thereby increasing the achievable footprint for dispersed trajectories.
4.5 CLOSED-LOOP PERFORMANCE

Based on the results of the open-loop trajectories with dispersions, worst-case dispersions were selected for each of three regions of the footprint: maximum downrange, maximum crossrange, and minimum downrange. Closed-loop trajectories with the predictor-corrector guidance algorithm were then run to the three regions of the footprint. The three target points selected for the closed-loop performance evaluation are shown in Figure 23 on page 100. The $3\sigma$ errors defined in Table 2 on page 73 were scaled such that the total error due to multiple error sources would still represent a $3\sigma$ dispersion so as to test the guidance system for reasonably probable dispersion cases [20]. To run all dispersions at their $3\sigma$ levels would be unrealistic.

The nominal and dispersed results for trajectories to each of the three regions are listed in Tables 6 on page 77 through 8 on page 77. Plots of selected parameters from these cases are included. Figures 26 on page 103 through 31 on page 108 present the altitude, velocity, heat rate, heat load, downrange, and crossrange time histories for the nominal maximum downrange trajectory. Figures 32 on page 109 through 37 on page 114 present the altitude, velocity, heat rate, heat load, downrange, and crossrange time histories for the nominal maximum crossrange trajectory. Figures 38 on page 115 through 43 on page 120 present the altitude, velocity, heat rate, heat load, downrange, and crossrange time histories for the nominal minimum downrange trajectory. In each of these cases, it is seen that the heat rate does not approach the heat rate limit, so no incremental bank angle is needed for heat rate control.

The control histories for the nominal maximum downrange trajectory and the dispersed case listed second in Table 6 on page 77 are presented in Figure 44 on page 121 and
Figure 45 on page 122. Figure 44 shows the angle of attack histories for the nominal and dispersed maximum downrange cases. It is seen that a two degree change in angle of attack is required early in the trajectory increasing to four degrees by the end of the trajectory. Figure 45 shows the bank angle histories for the nominal and dispersed maximum downrange cases. The bank angle required shows no change from zero degrees for this case.

The control histories for the nominal maximum crossrange trajectory and the dispersed case listed second in Table 7 on page 77 are presented in Figure 46 on page 123 and Figure 47 on page 124. Again, it is seen that a four degree change in angle of attack is required for the dispersed case. The bank angle history shows no change for the dispersed case from that of the nominal case.

The control histories for the nominal minimum downrange trajectory and the dispersed case listed second in Table 8 on page 77 are presented in Figure 48 on page 125 and Figure 49 on page 126. For this case, approximately a one degree change in angle of attack is required. No change is required in the bank angle profile.

The required change in angle of attack for each of the dispersed cases shown was primarily due to the change in the vehicle L/D as this was shown to be the primary dispersion source in Subsection “4.3 Effect of Dispersions on Footprint” on page 59. The breaking point of the guidance occurs when the vehicle does not have enough L/D range to overcome the loss in L/D due to aerodynamic dispersions. As mentioned previously, the Shuttle entry guidance algorithm was required to guide to the TAEM interface aim point to within 2.5 nautical miles of position. The results presented for the nominal and dispersed cases show that this requirement is met with the predictor-corrector guidance algorithm. Also, the trajectory plots show that the algorithm achieves this performance with very infrequent guidance.
updates (.02 hz.) and with very small control variable changes from the nominal constant angle of attack and linear bank with velocity profiles. Most importantly, almost all of the achievable footprint is captured using the predictor-corrector algorithm.

4.6 HEAT RATE CONTROL PERFORMANCE

The closed-loop trajectories shown previously did not require heat rate control because the maximum heat rate experienced was significantly lower than the limit imposed on the ERV. The time responses for bank angle, angle of attack, and heat rate for the beginning of a typical trajectory with and without heat rate control are shown in Figures 50 on page 127 through 52 on page 129. The resulting bank angle versus velocity profile is shown in Figure 53 on page 130. These trajectories are for the middle of the footprint where the peak heat rate does not exceed the limit for the ERV. Therefore, for illustrative purposes, the heat rate limit was reduced to 100 BTU/sq ft/sec. Comparing the trajectories with and without heat rate control, it is seen that the heat rate control takes place over a fairly long time range, but requires a significant departure from the linear bank profile over only a very short velocity range. The impact on the trajectory is therefore small, and the predictor-corrector stays converged on almost the same control history even though the vehicle does not follow the assumed control profile during the heat rate control area.
4.7 OVERCONTROL

For those trajectories not at the edge of the footprint, excess vehicle capability exists that can be utilized to increase guidance margin for dispersions that may occur later in the trajectory. For example, a 13,800 nautical mile downrange trajectory for the ERV only requires flying at 20 degrees angle of attack instead of 15 degrees for the nominal trajectory. The ERV can modulate angle of attack between 15 degrees (maximum L/D) and 50 degrees (minimum L/D) on the back side of the L/D curve, so the modulation capability is not equally centered about the commanded angle of attack if flying at 20 degrees. By flying at 15 degrees (maximum L/D) early in the trajectory, guidance can center the remaining guidance capability equally about the aim point to cover dispersions in all directions, not just those that require less L/D to reach the target point. This approach is referred to here as overcontrol or command biasing.

Overcontrol can be implemented in several ways. First, the command can be biased from the desired command when that command is not in the center of the modulation range. As the vehicle flies a biased angle of attack, for example, the predicted final state will differ from that for the unbiased command in such a direction that the next guidance command will be moved in the direction opposite to the bias. By biasing in the proper direction, the command can be driven toward the center of the modulation range. If the guidance requires an L/D higher than that in the middle of the L/D range, flying at an even higher L/D will drive the required L/D toward the middle. Secondly, the target aim point can be moved from the nominal aim point early in the entry. For example, for a trajectory to the maximum downrange region of the footprint, the target aim point can be moved even farther downrange. Of course, at some point in the trajectory, the aim point must be moved back to the desired point.
The first approach was implemented in the predictor-corrector algorithm by biasing the angle of attack by five degrees when it was more than two degrees away from 30 degrees. The biasing was terminated at an inertial velocity of 13,500 feet per second so as to allow the guidance to fly the proper control history near the end of the trajectory to reach the target aim point.

Figure 54 on page 131 compares the angle of attack control history for a 13,760 nautical mile downrange trajectory with the dispersions used for the closed-loop trajectories shown earlier. Without overcontrol, the vehicle misses the target aim point by 19.20 nautical miles. This occurs because the wind contribution to the dispersion increases as the vehicle velocity drops, so the multiplicative scale factor on density does not properly model this dispersion. As the wind contribution increases, a higher L/D is required, and the angle of attack is driven to 15 degrees or maximum L/D. Because maximum L/D was not utilized earlier in the trajectory, the vehicle did not reach the target. Late in the trajectory, the predictor-corrector goes unconverged as control authority is exhausted, causing the angle of attack to jump between 15 and 30 degrees. By this point, the target aim point was unreachable anyway due to the dispersions.

With command biasing, the commanded angle of attack early in the trajectory is that corresponding to maximum L/D or 15 degrees. It is seen that biasing drives the commanded angle of attack to 25 degrees once the biasing is terminated at a velocity of 13,500 feet per second or a time of 3,700 seconds. Later, when the wind dispersion drives the angle of attack toward 15 degrees, there is significant margin remaining, and the angle of attack is only driven to 24.5 degrees by the dispersion. With command biasing, the miss distance at TAEM interface is only 0.27 nautical miles. Therefore, guidance margin is increased by using overcontrol. More of the theoretically achievable footprint is attainable for dispersed
cases. An even larger magnitude dispersion could have been handled late in the trajectory since the angle of attack was not driven to that for maximum L/D.

Further work is needed in this area to determine the proper way to utilize overcontrol to maximize guidance margin for the expected dispersions. The probability of the various dispersions occurring and the histories of those dispersions along a trajectory must be considered. For example, if a "thick" atmosphere is encountered early in the trajectory equal to the worst-case expected dispersion, it is highly unlikely that the atmosphere will get "thicker" later in the trajectory. Therefore, it is unnecessary to preserve guidance margin in the direction needed to cover a "thicker" atmosphere beyond that already required for the expected worst-case atmosphere. Such considerations should be taken into account in the design of the overcontrol algorithm.

4.8 ALGORITHM EXECUTION TIME

An estimate of the execution time required for the predictor-corrector algorithm was made using the execution time estimate feature of the HAL compiler. The estimate is for the AP101 Shuttle flight computer. Figure 55 on page 132 shows the execution time required in seconds as a function of the time to the TAEM interface point for a maximum downrange trajectory. It is seen that early in the entry when the trajectory to be predicted is long, the predictor requires 43.7 seconds of CPU time. When only 500 seconds to the TAEM interface point remains, the required time drops to 4.5 seconds. This figure can also be interpreted as the minimum update interval for the predictor-corrector. Also, the guidance command will be computed and sent to the vehicle autopilot a period of time after the start of the guidance.
cycle equal to the required execution time. It is seen that early in the entry, a significant delay occurs between the start of the guidance cycle and the computation of the guidance command. This delay was not simulated in the closed-loop trajectories, but will have a minimal effect on the guidance margin because the guidance is not trying to fly a reference trajectory like the analytic guidance algorithms. The predictor-corrector is numerically computing a trajectory that will fly directly to the target aim point. Any error that builds up between the start of the guidance cycle and the issuing of the guidance command can be nulled easily since the entry is long, and the error will shrink as the delay decreases with decreasing time to the TAEM interface point.

The Shuttle AP101 CPU is the product of early 1970’s technology and is significantly slower than flight computers that might be employed in future entry vehicles. The 80C86 CPU for example is two to five times faster than the AP101 CPU, so the execution time required shown in Figure 55 on page 132 can be scaled down by a factor of two to five. Computers utilizing parallel processing architecture could predict the three required trajectories simultaneously in three CPUs, cutting the required execution time by a factor of three. If scaled by a factor of four due to the faster CPU and a factor of three due to parallel processing architecture, the maximum time required drops to 3.6 seconds, and the time with 500 seconds remaining to the TAEM interface point drops to 0.4 seconds. The predictor-corrector is therefore a viable guidance scheme for future entry vehicles.
5.0 FUTURE RESEARCH TOPICS AND CONCLUSIONS

5.1 FUTURE RESEARCH TOPICS

Several topics for further algorithm development and optimization are discussed. These are:

1. Further reductions in CPU execution time
2. Use of an air data system for in-flight measurements
3. Use of overcontrol to increase guidance margin
4. Control of more than two state constraints

Optimization of the predictor algorithm and the integration scheme can yield significant reductions in execution time beyond that already attained. Simplifying the aerodynamic model can yield a great reduction in execution time and an equally important reduction in the computer core required. The current model has 30 tables, each with 51 breakpoints over the angle of attack range. A curve fit of the aerodynamic coefficients over the angle of attack range and the flow regimes would reduce the core required to store the model data and the computations required for each lookup.

The estimator algorithm was shown to be effective in determining the dispersions from in-flight measurements. However, the estimator is unable to differentiate between density dispersions and atmospheric winds. Figure 24 on page 101 showed that the multiplicative density scale factor did not accurately model the wind contribution to the drag acceleration because the relative contribution of the wind to the dispersion increases as the vehicle...
velocity drops late in the trajectory. An air data system could provide an independent measurement of the atmospheric winds, improving the estimation process and increasing guidance margin by increasing the accuracy of the predicted trajectories.

The concept of overcontrol was introduced and shown to be effective for at least one dispersed case. Further investigations should be made to determine how much overcontrol is optimal for the expected dispersions. It may be possible to use the sensitivities of the constraints to the control variables to determine a proper amount of command biasing for any particular dispersion at any point in the trajectory.

Only the downrange error and crossrange error at TAEM interface are controlled in the current design. The vehicle energy is not controlled which can allow significant dispersions in the ranging capability during the TAEM phase of entry. Approaches include redefining the TAEM aim point in terms of a desired energy level or utilizing a third control variable to provide control over an energy level constraint. The Space Shuttle makes use of a split rudder as a speedbrake to provide a large energy control capability. Such an approach could be utilized with the predictor-corrector by computing the sensitivity of the three constraints to the three control variables. This would require four predictions instead of the three currently needed, but the fourth prediction could be made only during the latter part of entry to clean up any dispersions in energy level that occur during the entry due to dispersions. The CPU execution time would then increase by one-third over that currently projected when the third constraint is controlled.
5.2 CONCLUSIONS

A predictor-corrector entry guidance algorithm has been demonstrated that exhibits excellent performance and almost complete coverage of the achievable footprint. This algorithm employs a simple control variable history to achieve near-optimal guidance for the maximum downrange and maximum crossrange trajectories. Explicit heat rate control is employed without significantly impacting the achievable footprint. This is achieved because unlike previous guidance algorithms that included a long heat rate control phase with no active targeting, the proposed algorithm always actively targets to the aim point and only controls heat rate in the short high heat rate regions as required.

The algorithm has been demonstrated to handle atmospheric and aerodynamic dispersions within the capability of the vehicle. The required computer execution time is shown to be within the capability of new flight computers.

Algorithm adaptability is provided through the utilization of in-flight measurements to improve the accuracy of the predicted trajectory. Algorithm maintenance is simplified because there are no reference trajectories used, and there are a minimum of vehicle/mission-specific input parameters (I-loads). Transportability of the algorithm between different entry vehicles is provided by eliminating vehicle-specific entry phases other than the heat rate control phase which only requires the input of a heat rate limit. The guidance algorithm does require a vehicle aerodynamic model, but this is developed in the normal vehicle definition phase anyway.

In summary, an entry guidance algorithm has been developed that achieves near-optimal performance while maximizing flexibility, adaptability, and transportability. Although
more computationally intensive than analytic algorithms, execution of the predictor-corrector is within the capability of current flight computers. It is hoped that this guidance approach will significantly reduce the development and maintenance costs for new entry guidance systems.
Table 1. Characteristics of the ERV and Trajectory Entry Conditions

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Table 5. Dispersed Cases for Minimum Downrange Region

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<td>$\rho^-$</td>
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<td>1069</td>
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Table 6. Maximum Downrange Region Closed-Loop Results

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<td>CL (%)</td>
<td>CD (%)</td>
<td>Mass (%)</td>
<td>(\phi) (%)</td>
<td>Wind (%)</td>
<td>(\phi_g) (deg)</td>
<td>(\lambda) (deg)</td>
<td>Error (n.m.)</td>
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<tr>
<td>TARGET POINT</td>
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<td>144.700</td>
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<td>63.8 HW</td>
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Table 7. Maximum Crossrange Region Closed-Loop Results

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<td>Mass (%)</td>
<td>(\phi) (%)</td>
<td>Wind (%)</td>
<td>(\phi_g) (deg)</td>
<td>(\lambda) (deg)</td>
<td>Error (n.m.)</td>
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<td>63.8 CW</td>
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<td>+1.9</td>
<td>-3.2</td>
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Table 8. Minimum Downrange Region Closed-Loop Results

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<td>CL (%)</td>
<td>CD (%)</td>
<td>Mass (%)</td>
<td>(\phi) (%)</td>
<td>Wind (%)</td>
<td>(\phi_g) (deg)</td>
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<td>STS11</td>
<td>63.8 TW</td>
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</table>
Theoretical wing area = 177.40 ft$^2$
Body flap area = 7.50 ft$^2$
Elevon area per side = 6.07 ft$^2$
Aileron area per side = 0.88 ft$^2$
Tip fin area per side = 4.16 ft$^2$

Figure 1. Three-View Drawing of the ERV
SCALAR WIND SPEED V (m/sec) STEADY-STATE ENVELOPES AS FUNCTIONS OF ALTITUDE H (km) FOR TWO PROBABILITIES P (%) ENCOMPASSING ALL FOUR LOCATIONS

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P = 95

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P = 99

Figure 2. Atmospheric Wind Profile
Figure 3. Envelope of Density Profiles Derived from Shuttle Flights
Figure 4. STS-1 Density Profile Comparison

Reproduced from Reference (10)
Figure 5. STS-9 Density Profile Comparison

Reproduced from Reference (10)
Figure 6. Multiphase Bank Angle Program for L/D = 1.5

Reproduced from Reference (12)
Figure 7. Crossrange Versus Number of Bank Steps
Figure 8. Comparison of Optimum Bank Angle Programs
Figure 9. Optimum Shuttle Angle of Attack Profile for Maximum Downrange
Figure 10. Optimum Shuttle Bank Angle Profile for Maximum Crossrange
Figure 11. Optimum Shuttle Angle of Attack Profile for Maximum Crossrange
Figure 12. Landing Footprint for the ERV
Figure 13. Altitude Histories for the Entry Missions of the ERV

Reproduced from Reference (14)
Figure 14. Bank Angle Histories for the Entry Missions of the ERV

Reproduced from Reference (14)
Figure 15. Angle of Attack Histories for the Entry Missions of the ERV
Figure 16. Heat Rate Histories for the Entry Missions of the ERV
Figure 17. Heat Load Histories for the Entry Missions of the ERV
Figure 18. Bank Angle Versus Velocity Profile
Figure 19. Definitions of Downrange and Crossrange Errors
Figure 20. Predicted Lift Coefficient Profile for the ERV
Figure 21. Predicted L/D Profile for the ERV

Reproduced from Reference (14)
Figure 22. Predicted L/D versus Angle of Attack Profile for the ERV
Figure 23. ERV Open-Loop Footprint with the Control Profile
Figure 24. Time Response of the Density Filter
Figure 25. Time Response of the L/D Filter
Figure 26. Closed-Loop Altitude History for the Maximum Downrange Case
Figure 27. Closed-loop Velocity History for the Maximum Downrange Case.
Figure 28. Closed-Loop Heat Rate History for the Maximum Downrange Case
Figure 29. Closed-Loop Heat Load History for the Maximum Downrange Case
Figure 31. Closed-Loop Crossrange History for the Maximum Downrange Case
Figure 32. Closed-Loop Altitude History for the Maximum Crossrange Case
Figure 33. Closed-Loop Velocity History for the Maximum Crossrange Case
Figure 34. Closed-Loop Heat Rate History for the Maximum Crossrange Case
Figure 35. Closed-Loop Heat Load History for the Maximum Crossrange Case
Figure 36. Closed-Loop Downrange History for the Maximum Crossrange Case
Figure 37. Closed-Loop Crossrange History for the Maximum Crossrange Case
Figure 38. Closed-Loop Altitude History for the Minimum Downrange Case
Figure 39. Closed-Loop Velocity History for the Minimum Downrange Case
Figure 40. Closed-Loop Heat Rate History for the Minimum Downrange Case
Figure 41. Closed-Loop Heat Load History for the Minimum Downrange Case
Figure 42. Closed-Loop Downrange History for the Minimum Downrange Case
Figure 43. Closed-Loop Crossrange History for the Minimum Downrange Case
Figure 44. Angle of Attack Comparison for the Maximum Downrange Case
Figure 45. Bank Angle Comparison for the Maximum Downrange Case
Figure 46. Angle of Attack Comparison for the Maximum Crossrange Case
Figure 47. Bank Angle Comparison for the Maximum Crossrange Case
Figure 48. Angle of Attack Comparison for the Minimum Downrange Case
Figure 49. Bank Angle Comparison for the Minimum Downrange Case
Figure 50. Bank Angle Versus Time Comparison for Heat Rate Control
Figure 51. Angle of Attack Versus Time Comparison for Heat Rate Control
Figure 52. Heat Rate Versus Time Comparison for Heat Rate Control
Figure 53. Bank Angle Versus Velocity Comparison for Heat Rate Control
Figure 54: Angle of Attack Versus Time Comparison with Overcontrol
Figure 55. Required Execution Time for the Predictor-Corrector
APPENDIX A. ERV AERODYNAMICS MODEL

The aerodynamics of the ERV were reported in Reference [14], and the longitudinal performance coefficients, $C_L$ and $L/D$, are shown in Figures 20 on page 97 and 21 on page 98. Figure 22 on page 99 shows a typical $L/D$ versus angle of attack profile. This profile is for a Mach Number of 10, but across the flow regimes, the maximum $L/D$ always occurs at an angle of attack of approximately 15 degrees. This data was incorporated into the aerodynamic model of the simulator and into the aerodynamic model of the predictor. It is seen that the aerodynamic flow regimes are a function of:

1. Mach Number, $M$
2. Viscous Interaction Parameter, $\bar{V}$
3. Altitude, $h$

The Mach Number, $M$, is computed from,

$$M = \frac{V_r}{C_s} \quad (99)$$

where the speed of sound, $C_s$, is computed from,

$$C_s = \sqrt{\gamma \frac{R}{M_0} T_m} \quad (100)$$

The viscous interaction parameter, $\bar{V}$, is computed from,

$$\bar{V} = M \sqrt{\frac{C'}{Re}} \quad (101)$$

where,
\[ C' = \left( \frac{T'}{T_{\text{static}}} \right)^{0.5} \left[ \frac{T_{\text{static}} + 122.1 \times 10^{-5} T_{\text{static}}}{T' + 122.1 \times 10^{-5} T'} \right]^{10} \]  

(102)

and,

\[ \frac{T'}{T_{\text{static}}} = 0.468 + 0.532 \frac{T_{\text{wall}}}{T_{\text{static}}} + 0.195 \frac{\gamma - 1}{2} M^2 \]  

(103)

The Reynolds Number, \( Re \), is calculated from,

\[ Re = \frac{\rho Vr \bar{c}}{\mu} \]  

(104)

where the coefficient of viscosity for air, \( \mu \), is given by,

\[ \mu = \frac{\beta T_{\text{static}}^{3/2}}{S + T_{\text{static}}} \]  

(105)
Compiled listings of the flight software principal functions for the predictor-corrector guidance algorithm as coded for use in the 6-DOF Aeroassist Flight Experiment Simulator (AFESIM) follow. The algorithms are coded in the HAL/S computer language. The principal functions are:

1. IL_LOAD - Values for all constants and I-loads
2. FSW_SEQ - Flight Software Sequencer
3. ORB_NAV - Orbit Navigation Algorithm
4. AERO_GUID - Predictor-Corrector Guidance Algorithm

At the beginning of each principal function is a description of the function and the input/output parameters. At the end of each principal function is a cross reference table listing the program line at which each variable is referenced or computed.
STIFF SOURCE

FUNCTION: VALUES OF I-LOADS AND CONSTANTS

INPUTS: NONE

OUTPUTS: ALL I-LOADS AND CONSTANTS LISTED

CONVENTS: NONE

MATH CONSTANTS AND CONVERSION FACTORS

DECLARE PI SCALAR DOUBLE CONSTANT(3.1415926535897932385)
DECLARE DEG_TO_SEC SCALAR DOUBLE CONSTANT(3600)
DECLARE SEC_TO_DEG SCALAR DOUBLE CONSTANT(1/3600)
DECLARE DEG_TO_RAD SCALAR DOUBLE CONSTANT(PI / 180)
DECLARE RAD_TO_DEG SCALAR DOUBLE CONSTANT(180 / PI)
DECLARE SEC_TO_RAO SCALAR DOUBLE CONSTANT(1/3600)
DECLARE RAO_TO_SEC SCALAR DOUBLE CONSTANT(1/SEC_TO_RAO)
DECLARE FI_TO_H SCALAR DOUBLE CONSTANT(0.3048)
DECLARE H_TO_FT SCALAR DOUBLE CONSTANT(1/30.48)
DECLARE FT_TO_H SCALAR DOUBLE CONSTANT(1/0.3048)
DECLARE G_TO_FPS2 SCALAR DOUBLE CONSTANT(9.80665)
DECLARE FPS2_TO_G SCALAR DOUBLE CONSTANT(1/G_TO_FPS2)
DECLARE LBH_TO_KG SCALAR DOUBLE CONSTANT(4.4482216152605)
DECLARE KG_TO_LBF SCALAR DOUBLE CONSTANT(1/LBF_TO_KG)
DECLARE LBF_TO_N SCALAR DOUBLE CONSTANT(4.4482216152605)
DECLARE N_TO_LBF SCALAR DOUBLE CONSTANT(1/LBF_TO_N)
HAL/S STB 360-24.20
INTERNETICS, INC.
APRIL 27, 1987 15:04:40.70

STMT

SOURCE

CURRENT SCOPE

C| ------------------
C| FSN SEQ VARIABLES
C| ------------------
231 M| AERO_OAP_CNT = 3;
232 M| AERO_OAP_PHS = 0;
233 M| AERO_OAP_CNT = 3;
234 M| AERO_OAP_PHS = 0;
235 M| ORB_NAV_CNT = 5;
236 M| ORB_NAV_PHS = 0;

C| ---------
C| EPOCH DATA
C| ---------

E| EF_TO_REF_AT_EPOCH = MATRIX (+1.0, +0.0, +0.0, +0.0, +1.0, +0.0, +0.0, +0.0, +1.0)
S| DOUBLE_5,5

E| T_EPOCH = 0;

C| ------------------
C| EARTH PHYSICAL PARAMETERS
C| ------------------

239 M| EARTH_FLAT = 1 / 298.3;
240 M| EARTH_J2 = 1082.7E-6;

E| EARTH_MJ = (5.97605E+24) / (1.3048);
S| $$

E|$$

E| EARTH_POLE = VECTOR (2.89919495404719E-5, -8.11590555678430E-5, 9.30057959848170E-11);
S| DOUBLE_5

E| EARTH_R = (6378166.0) / .3048;
E| EARTH_RATE = 7.29211488321524E-5;

E| ME_NAV = EARTH_RATE EARTH_POLE;
HAL/S STD 360-20,0  1 N

STMT  SOURCE  CURRENT SCOPE
C  PREDICTOR-CORRECTOR I-LOADS (MISSION-SPECIFIC)  I I_LOAD
C
C  TARGET AIM POINT FOR MAXIMUM DOWNRANGE CASE
C  GONRANGE = 15049 N.M.
C  CROSSRANGE = 477 N.M.
C
C  LATITUDE -28.071 DEG  I I_LOAD
C  LONGITUDE -69.313 DEG  I I_LOAD
C  INCLINATION 28.50 DEG  I I_LOAD
C  FLIGHT PATH ANGLE -0.996 DEG  I I_LOAD
C  INERTIAL VELOCITY 25798.803 FT/SEC  I I_LOAD
C  ALTITUDE 4620000.0 FT  I I_LOAD
C
C  GEODETIC LATITUDE OF TAEM INTERFACE AIM POINT  I I_LOAD
C  LAT_TARGET = 3.0841
C
C  GEODETIC LONGITUDE OF TAEM INTERFACE AIM POINT  I I_LOAD
C  LONG_TARGET = 157.6591
C
C  PHI_EI = 0.0  I I_LOAD
C  ALPHA_EI = 20.0  I I_LOAD
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C  ESTIMATOR FILTER GAINS (TAU = 25.0 SECONDS)  I I_LOAD
C
C  K_RHO_FILTER_GAIN = 0.05921  I I_LOAD
C  I_OVER_R_FILTER_GAIN = 0.05921  I I_LOAD
C
C  VEHICLE MASS (SLUGS)  I I_LOAD
C
C  MASS_NAV = 186.0  I I_LOAD
HAL/S 360-24.20  INTERMETRICS, INC.  APRIL 27, 1987  15:048.60

**SIMS**  
**SOURCE**  
**CURRENT SCOPE**

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<td>V_INITIAL_MAG = 2500.0</td>
<td>IL_LOAD</td>
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<tr>
<td>266 MI</td>
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<td>IL_LOAD</td>
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C| CONSTANT ANGLE OF ATTACK PROFILE CONSTANTS | IL_LOAD |
| 267 MI| ALPHA_MAX = 45.0 | IL_LOAD |
| 268 MI| ALPHA_MIN = 15.0 | IL_LOAD |

C| VARIABLE TIME STEP CONTROL CONSTANTS | IL_LOAD |
| 269 MI| DELTA_T_PRED_GAIN = 200.0 | IL_LOAD |
| 270 MI| DELTA_T_PRED_MAX = 20.0 | IL_LOAD |
| 271 MI| DELTA_T_PRED_MIN = 2.0 | IL_LOAD |

C| HEAT RATE CONTROL CONSTANTS | IL_LOAD |
| 272 MI| HS = 3500.0 | IL_LOAD |
| 273 MI| OMEGA_DOT = 0.10 | IL_LOAD |
| 274 MI| QDOT_LIMIT = 125.0 | IL_LOAD |
| 275 MI| RHO_SL = 0.002378 | IL_LOAD |
| 276 MI| MU_DOT = 1.00 | IL_LOAD |
| 277 MI| CLOSE IL_LOAD | IL_LOAD |

**** BLOCK SUMMARY ****

COMPOOL VARIABLES USED
HAL/S STD 160-26.20

INTERNETRICS, INC.

APRIL 27, 1987 15:01:48.60

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IL_LOAD: PROCEDURE
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(CROSS REFERENCE FLAG KEY: 4 = ASSIGNMENT, 2 = REFERENCE, 1 = SUBSCRIPT USE, 0 = DEFINITION)
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FUNCTION: EXECUTE FLIGHT SOFTWARE PRINCIPAL FUNCTIONS AT
PROPER RATE AND IN PROPER ORDER WHEN FUNCTIONS ACTIVE

INPUTS:
- ORB_NAV_ACT - ORBIT NAVIGATION ACTIVE FLAG
- AERO_GUID_ACT - PREDICTOR-CORRECTOR ACTIVE FLAG
- AERO_DAP_ACT - DIGITAL AUTOPILOT ACTIVE FLAG

OUTPUTS: NONE

DECLARE FSM_PASS INTEGER DOUBLE INITIAL(1);
IF ORB_NAV.ACT = ON AND MOD(FSH_PASS, ORB_NAV_CNT) = ORB_NAV_PHS THEN
CALL ORB_NAV;

IF AERO_GUID.ACT = ON AND MOD(FSH_PASS, AERO_GUID_CNT) = AERO_GUID_PHS THEN
CALL AERO_GUID;

IF AERO_DAP.ACT = ON AND MOD(FSH_PASS, AERO_DAP_CNT) = AERO_DAP_PHS THEN
CALL AERO_DAP;

FSH_PASS = FSH_PASS + 1;
CLOSE FSH_SEQ;

*** BLOCK SUMMARY ***
EXTERNAL PROCEDURES CALLED
ORB_NAV, AERO_GUID, AERO_DAP
COMMON VARIABLES USED
ORB_NAV.ACT, ORB_NAV_CNT, ORB_NAV_PHS, AERO_GUID.ACT, AERO_GUID_CNT, AERO_GUID_PHS, AERO_DAP.ACT, AERO_DAP_CNT, AERO_DAP_PHS
HAL/S STD 340-84.20

INTERNETRICS, INC.

APRIL 27, 1987 15:11:9.00

COMPILATION LAYOUT

IL_POOL: EXTERNAL COMPOOL;
FSM_POOL: EXTERNAL COMPOOL;
ORB_NAV: EXTERNAL PROCEDURE;
AERO_GUID: EXTERNAL PROCEDURE;
AERO_DAP: EXTERNAL PROCEDURE;
FSM_SEQ: PROCEDURE;
### Symbol & Cross Reference Table Listing

**(Cross Reference Flag Key: 4 = Assignment, 2 = Reference, 1 = Subscript Use, 0 = Definition)**

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C FUNCTION: MAINTAIN ESTIMATE OF VEHICLE STATE VECTOR AND COMPUTE
C STATE VECTOR DERIVED PARAMETERS
C
C INPUTS:
C T_ATTITUDE - TIME TAG OF STATE VECTOR
C R - POSITION VECTOR
C V - VELOCITY VECTOR
C A - ACCELERATION VECTOR
C Q - ATTITUDE QUATERNION
C PHI - BANK ANGLE
C
C OUTPUTS:
C T_NAV - TIME TAG OF STATE VECTOR
C R_NAV - POSITION VECTOR
C V_NAV - VELOCITY VECTOR
C A_NAV - ACCELERATION VECTOR
C Q_B_TO_T - ATTITUDE QUATERNION
C R_MAG - MAGNITUDE OF POSITION VECTOR
C UNIT_R - UNIT VECTOR IN DIRECTION OF POSITION VECTOR
C ALT_NAV - ALTITUDE ABOVE ELLIPSOID
C V_MAG - MAGNITUDE OF VELOCITY VECTOR
C V_REL - RELATIVE VELOCITY VECTOR
C V_REL_MAG - MAGNITUDE OF RELATIVE VELOCITY VECTOR
C RGT_NAV - RADIAL VELOCITY MAGNITUDE
C G LOAD - SIGNED ACCELERATION MAGNITUDE IN G'S
C ALPHA_NAV - ANGLE OF ATTACK
C BETA_NAV - SIDESLIP ANGLE
C PHI_NAV - BANK ANGLE
C ENTRY_COMPLETE - TASK INTERFACE FLAG
C
C COMMENTS: PERFECT NAVIGATION IS ASSUMED, SO THE STATE VECTOR FROM
C THE ENVIRONMENT MODEL IS COPIED.
C
C LOCAL VARIABLES
C
C DECLARE VREL_BODY VECTOR(3) SINGLE INITIAL(3)

STMT | SOURCE | CURRENT SCOPE
---|---|---
C |  | ORB-nav
C | COMPUTE STATE VECTOR DERIVED PARAMETERS | ORB-nav
C |  | ORB-nav
531 M | T_NAV = T_INI_NAV | ORB-nav
E |  | ORB-nav
532 M | R_NAV_MAG = ABSVAL(R_NAV) | ORB-nav
E |  | ORB-nav
533 M | UNIT_R = R_NAV / R_NAV_MAG | ORB-nav
E |  | ORB-nav
534 M | ALT_NAV = R_NAV_MAG * (1 - EARTH_FLAT) EARTH_R / SQRT((1 - EARTH_FLAT) - 1) (1 - UNIT_R) | ORB-nav
E |  | ORB-nav
535 M | H_NAV = UNIT_R * EARTH_POLE | ORB-nav
E |  | ORB-nav
536 M | V_NAV_MAG = ABSVAL(V_NAV) | ORB-nav
E |  | ORB-nav
537 M | V_REL_NAV = V_NAV - (H_NAV * R_NAV) | ORB-nav
E |  | ORB-nav
538 M | HDOT_NAV = V_REL_NAV * UNIT_R | ORB-nav
E |  | ORB-nav
539 M | G_LOAD = ABSVAL(G) FPS2_TO_D | ORB-nav
C | ANGLE OF ATTACK AND SIDESLIP ANGLE | ORB-nav
C |  | ORB-nav
E |  | ORB-nav
540 M | VREL_BODY = SQFORM(SQPOS([R_DOT_T, V_REL_NAV])) | ORB-nav
541 M | ALPHA_NAV = SARCTAN2(VREL_BODY, VREL_BODY) RAD_TO_DEG | ORB-nav
E |  | ORB-nav
542 M | BETA_NAV = ARCSIN(VREL_BODY / V_REL_MAG) RAD_TO_DEG | ORB-nav
E |  | ORB-nav
C |  | ORB-nav
C | FLAG SIGNALING TAEN INTERFACE OR SKIP OUT | ORB-nav
C |  | ORB-nav
543 M | IF (ALT_NAV > ALT_EXIT) AND (HDOT_NAV > 0) OR (ALT_NAV < ALT_MIN) THEN | ORB-nav
E |  | ORB-nav
544 M | AERO_BRAKE_COMPLETE = TRUE | ORB-nav
HAL/S STD 340-26.20
INTERNETICS, INC.
APRIL 27, 1987 11:19:28.56

STATE
SOURCE
5/5 Ml CLOSE ORB_NAV;

**** BLOCK SUMMARY ****
EXTERNAL FUNCTIONS INVOKER
SUFFIX, SUFFIX, SUFFIX

COMPOOL VARIABLES USED
V_REL_NAV, ROOT_NAV*, G_LOAD*, A_NAV, FPSZ_TO_G, Q_B_TO_I, ALPHA_NAV*, RAD_TO_DEG, BETA_NAV*, V_REL_MAG, ALT_NAV, ALT_EXIT
ROOT_NAV, ALT_TASK, AERO_BRAKE_COMPLETE*
HAL/S STD 560-24.20
INTERNETRICS, INC.

APRIL 27, 1987
11:19:28.56

**** COMPI LATION LAYOUT ****

ENV_POOL: EXTERNAL COMPOOL;
FSH_POOL: EXTERNAL COMPOOL;
IL_POOL: EXTERNAL COMPOOL;
DOFORM: EXTERNAL FUNCTION;
Q_ERR_ANG: EXTERNAL FUNCTION;
SARCTAN: EXTERNAL FUNCTION;
SARCTAN: EXTERNAL FUNCTION;
SQMULT: EXTERNAL FUNCTION;
SQPOSE: EXTERNAL FUNCTION;
SRV_TO_GIL: EXTERNAL FUNCTION;
ORD_NAV: PROCEDURE;
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<td>510 SARTANZ</td>
<td>SCALAR FUNCTION</td>
<td>SINGLE, INITIAL, EXTERNAL, VERSION=2 XREF: 0 0510 2 0541</td>
</tr>
<tr>
<td>518 SARG</td>
<td>SCALAR</td>
<td>SINGLE, ALIGNED, INPUT-PARM XREF: 0 0518</td>
</tr>
<tr>
<td>510 SFORM</td>
<td>3 - VECTOR FUNCTION</td>
<td>SINGLE, INITIAL, EXTERNAL, VERSION=3 XREF: 0 0519 2 0540</td>
</tr>
<tr>
<td>520 SQZ</td>
<td>4 - VECTOR FUNCTION</td>
<td>SINGLE, INITIAL, EXTERNAL, VERSION=2 XREF: 0 0520 NOT REFERENCED</td>
</tr>
<tr>
<td>521 SQZ</td>
<td>4 - VECTOR FUNCTION</td>
<td>SINGLE, INITIAL, EXTERNAL, VERSION=2 XREF: 0 0522 NOT REFERENCED</td>
</tr>
<tr>
<td>521 SQZ</td>
<td>4 - VECTOR FUNCTION</td>
<td>SINGLE, INITIAL, EXTERNAL, VERSION=2 XREF: 0 0522 NOT REFERENCED</td>
</tr>
<tr>
<td>1 T_ATTITUDE</td>
<td>SCALAR</td>
<td>DOUBLE, ALIGNED, INITIAL XREF: 0 0001 2 0539</td>
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<tr>
<td>67 T_CTL_TAG</td>
<td>SCALAR</td>
<td>DOUBLE, ALIGNED, INITIAL XREF: 0 0525 2 0531</td>
</tr>
<tr>
<td>92 T_NAV</td>
<td>SCALAR</td>
<td>DOUBLE, ALIGNED, INITIAL XREF: 0 0525 2 0531</td>
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<td>514 T</td>
<td>3 - VECTOR</td>
<td>SINGLE, ALIGNED, INPUT-PARM XREF: 0 0519</td>
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<tr>
<td>519 U</td>
<td>3 - VECTOR</td>
<td>DOUBLE, ALIGNED, INITIAL XREF: 0 0001 2 0539</td>
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<tr>
<td>93 UNIT_R</td>
<td>3 - VECTOR</td>
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</tr>
<tr>
<td>DCL NAME</td>
<td>TYPE</td>
<td>ATTRIBUTES &amp; CROSS REFERENCE</td>
</tr>
<tr>
<td>----------</td>
<td>--------</td>
<td>-----------------------------</td>
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<tr>
<td>522 V</td>
<td>3 - VECTOR</td>
<td>SINGLE, ALIGNED, INPUT-PARM XREF: 0 0522</td>
</tr>
<tr>
<td>94 V_NAV</td>
<td>5 - VECTOR</td>
<td>DOUBLE, ALIGNED, INITIAL XREF: 0 0094 4 0527 2 0535 2 0536 2 0550</td>
</tr>
<tr>
<td>95 V_MAG</td>
<td>SCALAR</td>
<td>DOUBLE, ALIGNED, INITIAL XREF: 0 0095 4 0535 NOT REFERENCED</td>
</tr>
<tr>
<td>94 V_REL</td>
<td>SCALAR</td>
<td>DOUBLE, ALIGNED, INITIAL XREF: 0 0096 4 0537 2 0542</td>
</tr>
<tr>
<td>97 V_REL</td>
<td>5 - VECTOR</td>
<td>DOUBLE, ALIGNED, INITIAL XREF: 0 0097 4 0536 2 0537 2 0540 2 0542</td>
</tr>
<tr>
<td>524 VREL</td>
<td>3 - VECTOR</td>
<td>SINGLE, ALIGNED, STATIC, INITIAL XREF: 0 0524 4 0540 2 0541 2 0542</td>
</tr>
<tr>
<td>401 HE_N</td>
<td>3 - VECTOR</td>
<td>SINGLE, ALIGNED, INITIAL XREF: 0 0401 2 0536</td>
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</tbody>
</table>
FUNCTION: NUMERIC PREDICTOR/CORRECTOR ENTRY GUIDANCE ALGORITHM

FOR THE ENTRY RESEARCH VEHICLE (ERV).

INPUTS:
- A_NAV - SENSED INERTIAL ACCELERATION VECTOR
- ALPHA_NAV - ANGLE OF ATTACK
- ALT_NAV - ALTITUDE ABOVE FISHER ELLIPSOID
- G_LOAD - SENSED ACCELERATION MAGNITUDE IN G'S
- R_NAV - INERTIAL POSITION VECTOR
- T_GHT - GREENWICH MEAN TIME
- V_NAV - INERTIAL VELOCITY VECTOR
- V_REL - RELATIVE VELOCITY VECTOR
- V_REL_MAG - RELATIVE VELOCITY MAGNITUDE
- V_NAV_MAG - INERTIAL VELOCITY MAGNITUDE

OUTPUTS:
- ALPHA_CMD - COMMAND ANGLE OF ATTACK
- PHI_CMD - COMMAND BANK ANGLE

DESIGNED BY: K. SPRATLIN
C.S. DRAPER LABORATORY, INC.
555 TECHNOLOGY SQUARE
CAMBRIDGE, MA 02139
(617) 258-2643
C| LOCAL VARIABLES
C| -------------------
516 M| DECLARE ALPHA_DES SCALAR SINGLE;
517 M| DECLARE CL_EST SCALAR SINGLE;
518 M| DECLARE COSPHI_DOT SCALAR SINGLE INITIAL(-1.0);
519 M| DECLARE DELTA_T_PRED SCALAR SINGLE;
520 M| DECLARE EF_FROM_REF_AT_EPOCH_MATRIX(3, 3) DOUBLE;
521 M| DECLARE GUID_PASS INTEGER SINGLE;
522 M| DECLARE I_TARGET_EF_VECTOR(3) DOUBLE;
523 M| DECLARE INITIALIZE_GUIDANCE BOOLEAN INITIAL(TRUE);
524 M| DECLARE PHI_DES SCALAR SINGLE;
525 M| DECLARE RHO_NAV SCALAR SINGLE;
C| --------------------------
C| ATMOSPHERIC PROPERTIES STRUCTURE
C| --------------------------
526 M| STRUCTURE ATMOSPHER;
526 M| 1 H SCALAR SINGLE;
526 M| 1 RHO SCALAR SINGLE;
526 M| 1 TS SCALAR SINGLE;
526 M| 1 TM SCALAR SINGLE;
CALL HEAT_RATE_CONTROL;

CALL ATTITUDE_COMMAND;

END}
CO
HAL/S 5TD
$60-Z_.20 Z
N T E R H E T £ C S
_I
N C o APRZL
27, 1987 $TMT
366x347
SOURCE CURRENT
SCOPE
OPENPOOL VARIABLES USED
EF_TO_REF_AT_EPOCH, ALPHA_EI, PHI_EI, ALPHA_CHD, ALPHA_MIN, PHI_CHD, PHI_MAX, LONG_TARGET, DEG_TO_RAD, LAT_TARGET
EARTH_FLAT, K_RHONAV, K_LOD_NAV
OUTER VARIABLES USED
EF_FROM_REF_AT_EPOCH, QUID_PASS, ALPHA_DES, PHI_DES, I_TARGET_EF
DECLARE A_DRAG_MAG SCALAR SINGLE;
DECLARE A_LIFT_MAG SCALAR SINGLE;
DECLARE ATMOS ATMOSPROP-STRUCTURE;
DECLARE CD_NOM SCALAR SINGLE;
DECLARE CL_NOM SCALAR SINGLE;
DECLARE LOD_MEAS SCALAR SINGLE;
DECLARE LOD_NOM SCALAR SINGLE;
DECLARE RHO_MEAS SCALAR SINGLE;
DECLARE V_BAR SCALAR SINGLE;

LOOK UP OF NOMINAL DENSITY AND L/D

CALL USATHOS6ZI(NAV, EARTH_POLE, ATMOS)

CALL AERO_PARAMETERS(V_REL_MAG, ATMOS) ASSIGN(V_BAR, MACH)

CALL LOOKUP(ALPHA_NAV, ATMOS, V_BAR, MACH) ASSIGN(CL_NOM, CD_NOM)

LOD_NOM = CL_NOM / CD_NOM

COMPUTE DRAG AND LIFT ACCELERATION

A_DRAG_MAG = -A_NAV * V_REL_NAV / V_REL_MAG;
E1 587 H1 A_LIFT_MAG = SQRT(A_NAV * A_NAV - A_DRAG_HAG * A_DRAG_HAG)  
C | Compute measured L/D and density ratios  
C |  
588 H1 L/D_MEAS = A_LIFT_MAG / A_DRAG_HAG;  
E1 589 H1 RHO_MEAS = 2 * A_DRAG_HAG * MASS_NAV / (CD_NM * REF * V_REL_HAG * VREL_HAG);  
C | Compute filtered density ratios  
C |  
590 H1 K_LOD_NAV = (1 - L_OVER_D_FILTER_GAIN) * K_LOD_NAV + L_OVER_D_FILTER_GAIN * LOD_MEAS / LOD_NM;  
591 H1 K_RHO_NAV = (1 - K_RHO_FILTER_GAIN) * K_RHO_NAV + K_RHO_FILTER_GAIN * RHO_MEAS / ATMOS.RHO;  
C | Compute filtered estimated CL  
C |  
592 H1 RHO_NAV = K_RHO_NAV * ATMOS.RHO;  
C | Filtered density ratios  
C |  
593 H1 CL_EST = K_LOD_NAV * CL_NM;  
594 H1 CLOSE_FILTERS;  
C | Filtered density ratios  
C |  
*** BLOCK SUMMARY ***  
OUTER PROCEDURES CALLED  
LOOKUP, AERO_PARAMETERS_USATHOS62  
COMPOO VARIABLES USED  
R_NAV, EARTH_POLE, V_REL_HAG, ALPHA_NAV, A_NAV, V_REL_NAV, MASS_NAV, S_REF, K_LOD_NAV, L_OVER_D_FILTER_GAIN, K_LOD_NAV  
K_RHO_NAV, K_RHO_FILTER_GAIN, K_RHO_NAV  
OUTER VARIABLES USED  
RHO_NAV, CL_EST  
OUTER STRUCTURE TEMPLATES USED  
USATHOS62
595 | PROCEDURE;
595 | | ---------------
595 | | FUNCTION: CONTROL PEAK HEAT RATE
595 | | ---------------
595 | | LOCAL VARIABLES
595 | | ---------------
596 | DECLARE COSPHI_I SCALAR SINGLE;
597 | DECLARE COSPHI_2 SCALAR SINGLE;
598 | DECLARE FIRST_PASS BOOLEAN INITIAL TRUE;
599 | DECLARE HS_2 SCALAR SINGLE;
600 | DECLARE QDOT SCALAR SINGLE;
601 | DECLARE QDOT_RATE SCALAR SINGLE;
602 | DECLARE KL_GAIN SCALAR SINGLE;
603 | DECLARE KL_QDOT SCALAR SINGLE;
604 | DECLARE QBAR SCALAR SINGLE;
605 | DECLARE _DOT SCALAR SINGLE;
606 | DECLARE QDOT_PAST SCALAR SINGLE;
607 | DECLARE QDOT_RATE SCALAR SINGLE;
608 | DECLARE TWO_ZETA_OMEGA SCALAR SINGLE;
609 | ---------------
610 | IF (FIRST_PASS = TRUE) THEN
611 | DO;
612 | | ---------------
613 | | HEAT RATE CONTROL CONSTANTS
614 | | ---------------
615 | | ---------------
616 | GDOT = 17700.0 SQRT(RHO_NV) (V_REL_NV / 10000)
HS_2 = 2 * HS;
K1_GAIN = S_REF_SORT(RH0,SL) / (HS_2 * MASS_NAV);
OMEGA_QDOT_SQUARED = OMEGA_QDOT * OMEGA_QDOT;
THO_ZETA_OMEGA = 2 * ZETA_QDOT * OMEGA_QDOT;
QDOT_RATE = 0;
QDOT_RATE = (QDOT - QDOT_PAST) / DT_AERODYN;
QDOT_PAST = QDOT;
IF (QDOT > 1.5 QDOT_LIMIT) THEN
DO;
ESTIMATED DYNAMIC PRESSURE
QBAR = 1.5 * RH0 * NAV * REL_MAG * REL_MAG;
HEAT RATE CONTROL GAINS
K3_GDOT = K1_GAIN * CL_EST * (V_REL_MAG / 20000.0) * EXP(-(ALT_NAV / HS_2));
K_GDOT = OMEGA_QDOT_SQUARED / K1_GDOT;
K_GDOT RATE = THO_ZETA_OMEGA / K1_GDOT;
COSPHI_1 = K_GDOT RATE * QDOT RATE / QBAR;
COSPHI_2 = K_GDOT (QDOT - QDOT_LIMIT) / QBAR;
COSPHI_QDOT = COSPHI_1 + COSPHI_2;
HAL/S STB 360-24.20

INTERNETICS, INC.

APRIL 27, 1967

STMT  SOURCE  CURRENT SCOPE

631 M1 END;
632 M1 ELSE
632 M1 COSPHI_QDOT = -1.0;
633 M1 CLOSE HEAT_RATE_CONTROL;

**** BLOCK SUMMARY ****

COMMON VARIABLES USED
V_REL_HAG, HS, S_REF, RHO_SL_HASSNAV, OMEGA_QDOT, ZETA_QDOT, DT_AEROGUID, QDOT_LIMIT, ALT_NAV

OUTER VARIABLES USED
RHO_NAV, CL_EST, COSPHI_QDOT*
HAL/S STD 340-24.20 INTERMETRICS, INC.

ATTITUDE COMMAND PROCEDURE

LOAD ALPHA COMMAND

ALPHA_CMD = ALPHA_DES;

PHI_CMD = PHI_DES ABS(V_NAV_MAG - V_FINAL_MAG) / V_MAG_CHANGE;

PHI_CMD = MIDVAL(-PHI_MAX, PHI_CMD, PHI_MAX);

PHI_CMD = SIGN(PHI_CMD) ARCCOS(COSPHI_CMD) RAD TO_DEG;

IF (COSPHI_QDOT > 0) THEN

DO;

TEMPORARY COSPHI_CMD SCALAR SINGLE

COSPHI_CMD = COS(PHI_CMD DEG_TO_RAD) + COSPHI_QDOT;

COSPHI_CMD = MIDVAL(-1.0, COSPHI_CMD, +1.0);

PHI_CMD = SIGN(PHI_CMD) ARCCOS(COSPHI_CMD) RAD_TO_DEG;

END;

LIMIT ALPHA AND PHI COMMANDS

ALPHA_CMD = MIDVAL(ALPHA_MIN, ALPHA_CMD, ALPHA_MAX);

PHI_CMD = MIDVAL(-PHI_MAX, PHI_CMD, PHI_MAX);

CLOSE ATTITUDE_COMMAND;

COMPOOL VARIABLES USED
ALPHA_CMD, PHI_CMD, V_NAV_MAG, V_FINAL_MAG, V_MAG_CHANGE, PHI_MAX, PHI_HAX, DEG_TO_RAD, RAD_TO_DEG, ALPHA_MIN, ALPHA_MAX

OUTER VARIABLES USED
ALPHA_DES, PHI_DES, COSPHI_QDOT

BLOCK SUMMARY

169
PROCEDURE ...........................................
FUNCTION: PREDICTOR/CORRECTOR SEQUENCER ...........................................

LOCAL VARIABLES
DECLARE ALPHA_TRY SCALAR SINGLE
DECLARE DELTAALPHA SCALAR SINGLE CONSTANT(3)
DECLARE DELTA_PHI SCALAR SINGLE CONSTANT(3)
DECLARE DETERH SCALAR SINGLE INITIAL(0)
DECLARE DDREDA SCALAR SINGLE
DECLARE DDRE_DP SCALAR SINGLE
DECLARE DCRE_DA SCALAR SINGLE
DECLARE DCRE_DP SCALAR SINGLE
DECLARE CRE SCALAR SINGLE
DECLARE CRERROR ARRAY(3) SCALAR SINGLE
DECLARE CR_ERR SCALAR SINGLE
DECLARE PHI_TRY SCALAR SINGLE
DECLARE PRED_EXET SCALAR SINGLE
DECLARE DR_ERROR ARRAY(3) SCALAR SINGLE
DECLARE DR_ERR SCALAR SINGLE
DECLARE DRE SCALAR SINGLE

DO FOR TEMPORARY I = 1 TO 3;
670 MI 2 END;
671 MI 2 DO;
672 MI 3 ALPHA_TRY = ALPHA_DES + DELTA_ALPHA;
673 MI 3 PHI_TRY = PHI_DES;
674 MI 2 END;
675 MI 2 DO;
676 MI 3 ALPHA_TRY = ALPHA_DES;
677 MI 3 PHI_TRY = PHI_DES + DELTA_PHI;
678 MI 2 END;
679 MI 1 END;
       CALL PREDICTOR WITH DESIRED CONTROL HISTORY
       CALL PREDICTOR;
680 MI 1 CALL PREDICTOR;
       STORE FINAL STATE ERRORS
       CALL PREDICTOR;
       DR_ERROR = DR_ERR;
681 MI 1 CR_ERROR = CR_ERR;
       END;
682 MI 1 END;
       COMPUTE PARTIALS
       DCAE_DA = (DR_ERROR - DR_ERROR) / DELTA_ALPHA;
683 MI 1 DCAE_DP = (DR_ERROR - DR_ERROR) / DELTA_PHI;
       DCRE_DA = (CR_ERROR - CR_ERROR) / DELTA_ALPHA;
684 MI 1 DCRE_DP = (CR_ERROR - CR_ERROR) / DELTA_PHI;
       ...
HALS STD 360-24.20
INTERNETRICS, INC.

SHTT

<table>
<thead>
<tr>
<th>LINE</th>
<th>CODE</th>
</tr>
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<tbody>
<tr>
<td>608</td>
<td>DRE = OE_ERROR;</td>
</tr>
<tr>
<td>609</td>
<td>CRE = OR_ERROR;</td>
</tr>
<tr>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>SOLVE SET OF 2 SIMULTANEOUS EQUATIONS</td>
</tr>
<tr>
<td></td>
<td></td>
</tr>
<tr>
<td>689</td>
<td>DETERM = DCRE_DP DCRE_DA - DCRE_DP DCRE_DA;</td>
</tr>
<tr>
<td>690</td>
<td>IF (DETERM = 0) THEN</td>
</tr>
<tr>
<td>691</td>
<td>DO;</td>
</tr>
<tr>
<td>692</td>
<td></td>
</tr>
<tr>
<td>693</td>
<td>ALPHA_DES = ALPHA_DES + (DCRE_DP CRE - DCRE_DP DRE) / DETERM;</td>
</tr>
<tr>
<td>694</td>
<td>PHII_DES = PHII_DES - (DCRE_DA CRE - DCRE_DA DRE) / DETERM;</td>
</tr>
<tr>
<td>695</td>
<td>PHII_DES = PHII_DES - (DCRE_DA CRE - DCRE_DA DRE) / DETERM;</td>
</tr>
<tr>
<td>696</td>
<td>PHII_DES = PHII_DES - (DCRE_DA CRE - DCRE_DA DRE) / DETERM;</td>
</tr>
<tr>
<td>697</td>
<td>END;</td>
</tr>
</tbody>
</table>

CURRENT SCOPE
| CORRECTOR |
698 | PREDICTOR:
699 | PROCEDURE:
   | -------------------------------
   | FUNCTION: NUMERIC PREDICTOR ALGORITHM
   | -------------------------------
   | LOCAL FUNCTION
   | -------------------------------
   | DECLARE EARTH_FIXED_FROM_REFERENCE_FUNCTION_MATRIX(3, 3) DOUBLE;
   | -------------------------------
   | DECLARE TOTAL_TIME_STEPS INTEGER SINGLE;
   | DECLARE A_PRED VECTOR(3) DOUBLE;
   | DECLARE A_PRED_MAG SCALAR SINGLE;
   | DECLARE ALPHA_PRED SCALAR SINGLE;
   | DECLARE ATN spheresphere STRUCTURE;
   | DECLARE CD_PRED SCALAR SINGLE;
   | DECLARE CL_PRED SCALAR SINGLE;
   | DECLARE DOT SCALAR DOUBLE;
   | DECLARE EF_FROM_REF_MATRIX(3, 3) DOUBLE;
   | DECLARE I_INPLANE VECTOR(3) DOUBLE;
   | DECLARE I_NORMAL VECTOR(3) DOUBLE;
   | DECLARE INTEGRATION LOOP SCALAR SINGLE;
   | DECLARE IR_E VECTOR(3) DOUBLE;
   | DECLARE IOD_PRED SCALAR SINGLE;
   | DECLARE MACH_PRED SCALAR SINGLE;
   | DECLARE PHI_PRED SCALAR SINGLE;
   | DECLARE R_PRED VECTOR(3) DOUBLE;
   | DECLARE R_MAG_PRED SCALAR DOUBLE;
   | DECLARE R_DOT_PRED SCALAR SINGLE;
DECLARE T_PRED SCALAR DOUBLE;
DECLARE V_BAR_PRED SCALAR SINGLE;
DECLARE V_MAG_PRED SCALAR DOUBLE;
DECLARE V_PRED VECTOR(3) DOUBLE;
DECLARE V_REL_MAG_PRED SCALAR SINGLE;
DECLARE V_REL_PRED VECTOR(3) SINGLE;
DECLARE V_EL VECTOR(3) DOUBLE;

INITIALIZE PREDICTOR STATE VECTOR

T_PRED = T_0;
R_PRED = R_NAV;
R_HAG_PRED = ABVAL(R_PRED);
V_PRED = V_NAV;
V_HAG_PRED = ABVAL(V_PRED);
V_REL_PRED = V_PRED - (R_PRED - R_NAV);
V_REL_HAG_PRED = ABVAL(V_REL_PRED);

ANGLE OF ATTACK FOR PREDICTION

ALPHA_PRED = ALPHA_TRY;

CALL UGATHM62(R_PRED, EARTH_POLE) ASSIGN(ATHOS);

FORCE 1ST TIME STEP TO BE MINIMUM TIME STEP
AERODYNAMIC PROPERTIES LOOP

1 CALL AERO_PARAMETERS(V_REL_HAG_PRED, A_11VLOS) ASSIGN(V_BAR_PRED, MACH_PRED)

1 CALL LOOKUP(ALPHA_PRED, ATMOS.H, V_BAR_PRED, MACH_PRED) ASSIGN(CL_PRED, CD_PRED)

1 LOD_PRED = K_LOD_NAV CL_PRED / CD_PRED
C EQUATIONS OF MOTION FOR ENV ENTRY
C
750 M 1 DO FOR INTEG LOOP = 1 TO 45
751 M 2 TEMPORARY AERO ACCEL VECTOR(3) SINGLE
752 M 2 TEMPORARY CPHI SCALAR SINGLE
753 M 2 TEMPORARY DRAG ACCEL SCALAR SINGLE
754 M 2 TEMPORARY GRAY ACCEL VECTOR(3) DOUBLE
755 M 2 TEMPORARY HS_NORM_PRED SCALAR SINGLE
756 M 2 TEMPORARY ILAT VECTOR(3) SINGLE
757 M 2 TEMPORARY ILIFT VECTOR(3) SINGLE
758 M 2 TEMPORARY IVEL VECTOR(3) SINGLE
759 M 2 TEMPORARY LIFT ACCEL SCALAR SINGLE
760 M 2 TEMPORARY RHO PRED SCALAR SINGLE
761 M 2 TEMPORARY SPHI SCALAR SINGLE
762 M 2 TEMPORARY U PRED VECTOR(3) DOUBLE
763 M 2 TEMPORARY Z PRED SCALAR DOUBLE

C
C ESTIMATED DENSITY
C
764 M 2 RHO_PRED = K_RHO_NAV ATMOS.RHO
C
C BANK ANGLE MODEL
C
765 M 2 PHD_PRED = PHI_TRY ABS(V MAG_PRED - V FINAL MAG) / V MAG_CHANGE
766 M 2 PHD_PRED = MIDVAL(-PHI_MAX, PHI_PRED, PHI_MAX) DEG_TO_RAD
767 M 2 CPHI = COS(PHI_PRED)
768 M 2 SPHI = SING(PHI_PRED)
C
C AERODYNAMIC ACCELERATION
C
```
769 MI 2 DRAG_ACCEL = .5 * RHO_PRED * V_REL * MAG_PRED / MASS_NAV;
770 MI 2 LIFT_ACCEL = LOG_PRED * DRAG_ACCEL;
771 MI 2 IVEL = V_REL / V_REL_PRED;
772 MI 2 I_LAT = UNIT(IVEL * R_PRED);
773 MI 2 I_LIFT = UNIT(I_LAT * IVEL) * PHI * I_LAT * PHI;
774 MI 2 AERO_ACCEL = LIFT_ACCEL * I_LIFT - DRAG_ACCEL * I_VEL;
775 MI 2 UPRED = R_PRED / R_MAG_PRED;
776 MI 2 Z_PRED = UPRED * EARTH_POLE;
777 MI 2 UPRED = UPRED + (EARTH_R / R_MAG_PRED) *(1 - S_PRED) U_PRED + Z;
778 MI 2 Z_PRED = EARTH_POLE * Z;
779 MI 2 GRAV_ACCEL = -(EARTH_R / R_MAG_PRED) * U_PRED;
780 MI 2 CALL RUNGA-KUTTA INTEGRATOR;
781 MI 2 R_MAG_PRED = ABVAL(R_PRED);
```

612 M 2 V_MAG_PRED = ABVAL(V_PRED);
C|-----------------------------
C| RELATIVE VELOCITY
C|-----------------------------

613 M 2 V_REL_PRED = V_PRED - (HE_NAV X R_PRED);

614 M 2 V_REL_MAG_PRED = ABVAL(V_REL_PRED);
C|-----------------------------
C| 1962 U.S. STANDARD ATMOSPHERE
C|-----------------------------

615 M 2 CALL USATMOS42(R_PRED, EARTH_POLE) ASSIGN(ATMOS);

616 M 1 END;
C|-----------------------------
C| STATE PARAMETERS
C|-----------------------------

617 M 1 T_PRED = T_PRED + DELTA_T_PRED;

618 M 1 RDOT_PRED = V_PRED - R_PRED / R_MAG_PRED;
C|-----------------------------
C| CHECK FOR ATMOSPHERIC EXIT
C|-----------------------------

619 M 1 IF (IATMOS.N > 400000) AND (RDOT_PRED > 0)) THEN

620 M 1 PRE_EXIT = TRUE;

621 M 1 IF (PRE_EXIT = TRUE) THEN

622 M 1 EXIT;
C|-----------------------------
C| CHECK FOR TAEH INTERFACE
C|-----------------------------

623 M 1 IF (IATMOS.N <= ALT_TAEH) THEN

624 M 1 EXIT;

625 M 1 END;
E1  796  EF_FROM_REF = EARTH_FIXED_FROM_REFERENCE(T_PRED)
E1  797  IN_E = UNIT(EF_FROM_REF R_PRED)
E1  798  VR_E = EF_FROM_REF (V_PRED - HE_NAV * R_PRED)
E1  799  I_NORMAL = UNIT(IN_E * VR_E)
E1  800  I_INPLANE = UNIT(I_TARGET_EF * (I_TARGET_EF . I_NORMAL) I_NORMAL)
E1  801  DOT = I_INPLANE . I_TARGET_EF
E1  802  IF (ABS(DOT) > 1) THEN
E1  803  DOT = SIGN(DOT)
E1  804  CR_ERR = EARTH_R FT_TO_HR_ARCCOS(DOT) SIGN(I_INPLANE * I_TARGET_EF . (I_NORMAL * I_INPLANE))
E1  805  DOT = VR_E . I_INPLANE
E1  806  IF (ABS(DOT) > 1) THEN
E1  807  DOT = SIGN(DOT)
E1  808  DR_ERR = EARTH_R FT_TO_HR_ARCCOS(DOT) SIGN(IN_E = I_INPLANE) . I_NORMAL)
PROCEDURE CI

FUNCTION: 4TH ORDER RUNGE-KUTTA INTEGRATOR ALGORITHM

LOCAL VARIABLES

810 MI DECLARE ACCUM_ACCEL VECTOR(3) DOUBLE;
811 MI DECLARE ACCUM_VEL VECTOR(3) DOUBLE;
812 MI DECLARE ORIG_POS VECTOR(3) DOUBLE;
813 MI DECLARE ORIG_VEL VECTOR(3) DOUBLE;

814 MI DO CASE INTEG_LOOP;
815 MI 1 DO;
816 MI 2 ORIG_POS = R_PRED;
817 MI 2 ORIG_VEL = V_PRED;
818 MI 2 ACCUM_VEL = V_PRED;
819 MI 2 ACCUM_ACCEL = A_PRED;
820 MI 2 R_PRED = ORIG_POS + .5 DELTA_T_PRED V_PRED;
821 MI 2 V_PRED = ORIG_VEL + .5 DELTA_T_PRED A_PRED;
822 MI 1 END;
823 MI 1 DO;
824 MI 2 ACCUM_VEL = ACCUM_VEL + 2 V_PRED;
825 MI 2 ACCUM_ACCEL = ACCUM_ACCEL + 2 A_PRED;
HAL/2 STD 260-24.20

INTERMETRICS, INC.

STMT

SOURCE

026 M 2 R_PRED = ORIG_POS + .5 DELTA_T_PRED V_PRED;
027 M 2 V_PRED = ORIG_VEL + .5 DELTA_T_PRED A_PRED;
028 H 1 END;
029 H 1 DO;

030 M 2 ACCUM_VEL = ACCUM_VEL + 2 V_PRED;
031 M 2 ACCUM_ACCEL = ACCUM_ACCEL + 2 A_PRED;
032 M 2 R_PRED = ORIG_POS + DELTA_T_PRED V_PRED;
033 M 2 V_PRED = ORIG_VEL + DELTA_T_PRED A_PRED;
034 M 1 END;
035 H 1 DO;

036 M 2 R_PRED = ORIG_POS + (ACCUM_VEL + V_PRED) DELTA_T_PRED / 64;
037 M 2 V_PRED = ORIG_VEL + (ACCUM_ACCEL + A_PRED) DELTA_T_PRED / 64;
038 H 1 END;
039 H END;
040 H CLOSE INTEGRATOR;

**** BLOCK SUMMARY ****

OUTER VARIABLES USED
INTEG_LOOP, R_PRED, V_PRED, A_PRED, R_PRED*, DELTA_T_PRED, V_PRED*
DECLARE T SCALAR DOUBLE;
DECLARE CLAMBDA SCALAR DOUBLE;
DECLARE SLAMBDA SCALAR DOUBLE;
CLAMBDA = COS(T - T_EPOCH) EARTH_RATE;
SLAMBDA = SIN(T - T_EPOCH) EARTH_RATE;
RETURN (MATRIX (CLAMBDA, SLAMBDA, 0, -SLAMBDA, CLAMBDA, 0, 0, 0, 1)
DOUBLE,3,3)
EF
EF_FROM_REF_AT_EPOCH1
CLOSE EARTH_FIXED_FROM_REFERENCE;

**** BLOCK SUMMARY ****
COMMON VARIABLES USED
  T_EPOCH, EARTH_RATE

OUTER VARIABLES USED
  EF_FROM_REF_AT_EPOCH
HAL/S STD 360-24.20
INTERNETRICS, INC.


STMF

SOURCE

CURRENT SCOPE

B49 M CLOSE PREDICTOR;

***** BLOCK SUMMARY *****

OUTER PROCEDURES CALLED

USARTHOS42, AERO_PARAMETERS, LOOKUP

COMPOOL VARIABLES USED

F东海, RNAV, VNAV, DNAV, EARTH_RADIUS, ALT_TANGENT, DELTA_T_PRED_MIN, DELTA_T_PRED_MAX, T_4400, NAV

OUTER VARIABLES USED

ALPHA_TRY, DELTA_T_PRED*, DELTA_T_PRED, PHI_TRY, PRED.Exit*, PRED.Exit, I_TARGET Hạ, CR_ERR*, DR_ERR*

OUTER STRUCTURE TEMPLATES USED

ATHOSPROP
HAL/S STD 360-24.20  INTERMETRICS, INC.

STMT 850 MJ CLOSE CORRECTOR:

850 MJ CLOSE CORRECTOR:

**** BLOCK SUMMARY *****

COMPOUT VARIABLES USED
ALPHA_HOH, ALPHA_MAX, PHI_DES_MAX

OUTER VARIABLES USED
ALPHA_DES, PHI_DES, ALPHA_DES*, PHI_DES*
651 H| AERO_PARAMETERS;
651 H| PROCEDURE(V_REL_HAG, ATNOS) ASSIGN(V_BAR, MACH);
652 H| DECLARE V_REL_HAG SCALAR SINGLE;
653 H| DECLARE ATNOS ATMOSP-STRUCTURE;
654 H| DECLARE MACH SCALAR SINGLE;
655 H| DECLARE V_BAR SCALAR SINGLE;
656 H| DECLARE C_PRIME SCALAR SINGLE;
657 H| DECLARE GAMMA_VBAR SCALAR SINGLE;
658 H| DECLARE REYNOLDS_NUMBER_SCALAR;
659 H| DECLARE SPEED_OF_SOUND SCALAR SINGLE;
660 H| DECLARE T_PRIME SCALAR SINGLE;
661 H| DECLARE T_MACH SCALAR SINGLE;
662 H| DECLARE VISOSITY SINGLE SCALAR;
663 H| DECLARE C_BAR SCALAR SINGLE CONSTANT(25.01);
664 H| DECLARE DEG_R_TO_DEG_K SCALAR SINGLE CONSTANT(9 / 5);
665 H| DECLARE GAMMA SCALAR SINGLE CONSTANT(1.4);
666 H| DECLARE UNIV_GAS_CONST SCALAR SINGLE CONSTANT(8.31452 / 10);
667 H| DECLARE MOLE_MT_ZER SCALAR SINGLE CONSTANT(28.9664);
668 H| DECLARE SPEED_OF_SOUND_CONST SCALAR SINGLE CONSTANT(SQRT(GAMMA UNIV_GAS_CONST / MOLE_MT_ZER));
IF SPEED_OF_SOUND = 0 THEN
MACH = 0;
ELSE
MACH = V_REL_MAG / SPEED_OF_SOUND;

VISCOUS PARAMETER
IF REYNOLDS_NUMBER = 0

VBAR = 0
ELSE
REYNOLDS_NUMBER = ATMOS.RHO V_REL_MAG C_BAR / VISCOSS_PARAM;

IF REYNOLDS_NUMBER = 0
V_BAR = 0;
ELSE

STHY SOURCE

879 H1 DO1
880 H1 1 C1 T_HALL FOR VISCOUS PARAMETER
881 H1 1 IF (ATMOS.H < 240000) THEN
882 H1 1 IF HALL = 2178.0 DEG R TO DEG K1
883 H1 1 ELSE IF (ATMOS.H >= 260000) AND (ATMOS.H < 360000) THEN
884 H1 1 T_HALL = (5931.0 - 0.015 ATMOS.H) DEG R TO DEG K1
885 H1 1 ELSE IF (ATMOS.H >= 360000) THEN
886 H1 1 T_HALL = 504.0 DEG R TO DEG K1
887 H1 1 ELSE IF (ATMOS.H < 100000) THEN
888 H1 1 GAMMA_VBAR = 1.4;
889 H1 1 ELSE IF (ATMOS.H >= 100000) AND (ATMOS.H < 170000) THEN
890 H1 1 GAMMA_VBAR = 1.7 - 3.00E-6 ATMOS.H;
891 H1 1 ELSE IF (ATMOS.H >= 170000) AND (ATMOS.H < 215000) THEN
892 H1 1 GAMMA_VBAR = 1.375 - 1.09E-6 ATMOS.H;
893 H1 1 ELSE IF (ATMOS.H >= 215000) AND (ATMOS.H < 300000) THEN
894 H1 1 GAMMA_VBAR = 1.220 - 4.00E-7 ATMOS.H;
895 H1 1 ELSE IF (ATMOS.H >= 300000) THEN
896 H1 1 GAMMA_VBAR = 1.11
897 H1 1 T_PRIME AND C_PRIME FOR VISCOUS PARAMETER
898 H1 1 T_PRIME = (4.648 + 0.532 T_HALL / ATMOS.TS - 0.195 (GAMMA_VBAR - 1) MACH / 2) ATMOS.TS;
899 H1 1 C_PRIME = (T_PRIME / ATMOS.TS) (ATMOS.TS + 122.1 10) / T_PRIME + 122.1 10;
900 H1 1 (1/T_PRIME) 1;
901 H1 1
HAL/S  STD 360-24.20  INTERMETRICS, INC.  APRIL 27, 1987  14:15:28.05

```
E |  090 H | 1  V_BAR  =  MACH  (C_PRIME  /  REYNOLDS_NUMBER)  ;
   |  099 H |  END;
   |  000 H |  CLOSE  AERO_PARAMETERS;

**** BLOCK SUMMARY ****

COMPOOL  VARIABLES  USED
   M_TO_FT,  KG_TO_SLUG

OUTER  STRUCTURE  TEMPLATES  USED
   ATHCOMP

```

SOURCE  CURRENT  SCOPE

I  AERO_PARAMETERS
I  AERO_PARAMETERS
I  AERO_PARAMETERS
I  AERO_PARAMETERS
PROCEDURE(ALPHA, ALT, V_BAR, MACH) ASSIGNICL, CD)

FUNCTION: LOOK-UP OF CL AND CD VERSUS ALPHA, ALTITUDE, VISCOS
PARAMETER, AND MACH NUMBER.
INPUTS: ALPHA - ANGLE OF ATTACK (DEG)
ALTITUDE - ALTITUDE ABOVE FISHER ELLIPSOID (FT)
MACH - MACH NUMBER
V_BAR - VISCOS PARAMETER
CL - LIFT COEFFICIENT
CD - DRAG COEFFICIENT
FLON_REGIME: 1 = USE ALTITUDE DATA
2 = USE V_BAR DATA
3 = USE MACH DATA
REFERENCE: NOT DOCUMENTED

DECLARE ALPHA SCALAR SINGLE
DECLARE ALT SCALAR SINGLE
DECLARE CD SCALAR SINGLE
DECLARE CL SCALAR SINGLE
DECLARE MACH SCALAR SINGLE
DECLARE V_BAR SCALAR SINGLE

LOCAL VARIABLES
DECLARE ALPHA FRACT SCALAR SINGLE
DECLARE ALPHA HAX SCALAR SINGLE CONSTANT(50)
DECLARE ALPHA_MIN SCALAR SINGLE CONSTANT(0)
DECLARE ALT_SCALAR SINGLE
DECLARE ALT_MAX SCALAR SINGLE CONSTANT(557000)
DECLARE ALT_MIN SCALAR SINGLE CONSTANT(550000)
DECLARE ALT_RUN SCALAR SINGLE CONSTANT(ALT_MAX - ALT_MIN)
DECLARE CD_1 SCALAR SINGLE;
<table>
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<tr>
<th>HEL'S STD 540-29.10</th>
<th>INTERMIXITORS</th>
<th>INCHE</th>
<th>CURRENT SCOPE</th>
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<tr>
<td>955 ML 2</td>
<td>956 ML 2</td>
<td>957 ML 3</td>
<td>958 ML 3</td>
</tr>
<tr>
<td>VVAR_L = MEDVAL(VBAR_HH, VBAR_MCH, VBAR_MCH), (1)</td>
<td>DO FOR TEMPORARY K = 7 TO 1 BY -1</td>
<td></td>
<td></td>
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<table>
<thead>
<tr>
<th>CASE 1</th>
<th>CASE 2</th>
<th>CASE 3</th>
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<tr>
<td>946 ML 2</td>
<td>946 ML 3</td>
<td>EXPI</td>
</tr>
<tr>
<td>CLJ = CL_VISC 1, 2 + CL_VISC 3, 4 - CL_VISC 1, 4 + ALPHA_FACT</td>
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<tr>
<td>946 ML 2</td>
<td>946 ML 3</td>
<td>946 ML 5</td>
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<tr>
<td>CLJ = CL_VISC 1, 2 + CL_VISC 3, 4 - CL_VISC 1, 4 + ALPHA_FACT</td>
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</tr>
<tr>
<td>946 ML 2</td>
<td>946 ML 3</td>
<td>946 ML 5</td>
</tr>
<tr>
<td>PRCT = V_VAR_L - V_VAR_TABLE 1 / V_VAR_TABLE 1</td>
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</tr>
<tr>
<td>947 ML 1</td>
<td>947 ML 1</td>
<td>ENDJ</td>
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ORIGINAL PAGE IS OF POOR QUALITY
HAL/SSTD 340-24.20
INTERNETICS, INC.

STMT SOURCE CURRENT SCOPE
974 M1 Z CD_2 = CD_MACH 1+1,J (CD_MACH I+1,J + CD_MACH I+1,J - CD_MACH I+1,J 1+1,J) ALPHA_FRACT
I LOOKUP
975 M1 Z FRACT = (MACH_1 / Z - 1)
I LOOKUP
976 M1 1 END
I LOOKUP
977 M1 END
I LOOKUP CASE END
C INTERPOLATE BETWEEN TABLES
C -------------------------------
978 M1 CL = CL_1 + (CL_2 - CL_1) FRACT
I LOOKUP
979 M1 CD = CD_1 + (CD_2 - CD_1) FRACT
I LOOKUP
400 M1 CLOSE LOOKUP
I LOOKUP
PROCEDURE(R, POLE) ASSIGN(ATHOS)

DATE: 12/28/81

FUNCTION: COMPUTES THE DENSITY OF THE ATMOSPHERE (0-150KH) AT A SPECIFIED ALTITUDE USING THE 1962 U.S. STANDARD ATMOSPHERE MODEL.

INPUTS:
- R - INERTIAL POSITION VECTOR (FT)
- POLE - INERTIAL NORTH POLE (ND)

OUTPUTS:
- ATHOS.RH - DENSITY (LB/FT^3)
- ATHOS.RHU - DENSITY (LB/FT^3)
- ATHOS.TS - STATIC TEMPERATURE OF AIR (DEG K)
- ATMOS.TH - MOLECULAR TEMPERATURE OF AIR (DEG K)

NOÆMCNATURÆ: A - EARTH EQUATORIAL RADIUS (M)
- F - EARTH FLATTENING (ND)
- NO - EARTH RADIUS (Ft) + 45 DEGS (M)
- PHI - GEOGRAPHIC LATITUDE (DEGS)
- PI - GEOCENTRIC LATITUDE (DEGS)
- GO - SEA-LEVEL GRAVITY (M/S^2)
- NO - MEAN MOLECULAR WEIGHT OF AIR (ND)
- HR - UNIVERSAL GAS CONSTANT

DECLARE ATMOS ATMOSPHERE-STRUCTURE;
DECLARE R VECTOR[3] DOUBLE;
DECLARE POLE VECTOR[3] DOUBLE;
DECLARE GO SCALAR SINGLE CONSTANT(9.80665);
DECLARE NO SCALAR SINGLE CONSTANT(6.9646)
DECLARE RR SCALAR SINGLE CONSTANT(GO NO / RR);
DECLARE N_TO_FT SCALAR DOUBLE CONSTANT(1 / .3048);
DECLARE KG_TO_LBH SCALAR DOUBLE CONSTANT(1 / .6559237);
DECLARE A SCALAR DOUBLE CONSTANT(9.78176);
DECLARE F SCALAR DOUBLE CONSTANT(1 / 298.32)

APRIL 27, 1987
DECLARE RHO_BASE ARRAY(NSEGS + 1) SCALAR SINGLE CONSTANT(1.2250, 0.36392, 0.088035, 0.013125, 0.001275, 0.0007593, 0.0002510, 0.0000100, 0.000003170, 0.00000009829, 0.000000001836); 

DECLARE DTHOH ARRAY(NSEGS) SCALAR SINGLE INITIAL(0.065, 0.0000, 0.0100, 0.0000, -0.065, -0.0000, -0.0100, -0.0000); 

DECLARE R_MAG_H_SPSI, DH, EXPO, ALTADJ, ALT_RATIO, GRAV_RATIO, TEMPRATIO;

DETERMINE ALTITUDE ABOVE FISHER ELLIPSOID
1025 M1 2  IF (DTNOW = 0) THEN
1025 M1 2  I
1026 M1 2  ATMOS.RHO = RHO_BASE * EXP(-K2 * DH / TM_BASE)
1027 M1 2  ELSE
1027 M1 2  DO;
1028 M1 3  EXPO = 1 + K2 / DTNOW
1029 M1 3  ATMOS.RHO = RHO_BASE筹TM_BASE / ATMOS.TH)
1030 M1 2  END;
1031 M1 1  END;
1032 M1 1  ELSE
1032 M1 1  DO;
1033 M1 2  ALTADJ = R0 + H_BASE / DTNOW
1034 M1 2  ALTRATIO = (R0 + M) / (R0 + H_BASE)
1035 M1 2  TEMPRATIO = TM_BASE / ATMOS.TH;
1036 M1 2  EXP0 = (K2 / DTNOW) • (R0 / ALTADJ)
1037 M1 2  ATMOS.RHO = RHO_BASE筹TEMPRATIO筹EXP0筹(1 / (ALTRATIO + 2))
1038 M1 2  END;
1039 M1 1  EXPO
1040 M1 2  TEMPRATIO筹ALTRATIO
1041 M1 1  END;
1042 M1 1  ATMOS.RHO = ATMOS.RHO • (KG_TO_LBM / (M_TO_FT • G_TO_FPS2))
1043 M1 1  END;
1044 M1 1  CLOSE
**BLOCK SUMMARY**

**COOPOL VARIABLES USED**
G_TO_FPSZ

**OTHER STRUCTURE TEMPLATES USED**
ATHSPROP
1042: M | CLOSE AERO_GUID;

**** COMPI LATION LAYOUT ****

FSH_POOL: EXTERNAL CPOOL;
IL_POOL: EXTERNAL CPOOL;
AERQ GUID: PROCEDURE;
   INITIAL GUID: PROCEDURE;
   FILTERS: PROCEDURE;
   HEAT RATE CONTROL: PROCEDURE;
   ATTITUDE COMMAND: PROCEDURE;
   CORRECTOR: PROCEDURE;
   PREDICTOR: PROCEDURE;
   INTEGRATOR: PROCEDURE;
   EARTH FIXED FROM REFERENCE: FUNCTION;
AERO PARAMETERS: PROCEDURE;
LOOKUP: PROCEDURE;
USAMOS62: PROCEDURE;
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<td>574 NAV</td>
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