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Guidance and Targeting for the Strategic Target System

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Guidance algorithms and targeting procedures for the Strategic Target System (STARS) launch vehicle are described. The STARS vehicle is a three-stage booster, based partly on retired Polaris A3 missile assets, which is intended to support development and testing of the Strategic Defense Initiative by delivering target payloads to the vicinity of the Kwajalein Atoll. STARS will be launched from the Kauai Test Facility located on Kauai, Hawaii. The STARS guidance objective is to deliver payloads to a prescribed target location with intercontinental ballistic missile re-entry conditions. The guidance problem is complicated by the fact that all three stages lack thrust termination or other velocity control mechanisms, and by range safety requirements for one or more out-of-plane turns. Mission objectives are achieved with a combination of guidance algorithms. The original Polaris guidance is used during the atmospheric ascent phase. The powered explicit guidance used by the Space Shuttle is later employed to execute an out-of-plane turn and to place the third stage as closely as possible onto the desired coast trajectory. Third-stage guidance is a modified Lambert procedure formulated to eliminate the target miss due to off-nominal ascent phase performance.

I. Introduction

THIS paper describes the guidance algorithms and targeting procedures used in conjunction with the Strategic Target System (STARS). The STARS booster is being developed[1] to provide a low-cost launch system for delivering targets into the area of the U.S. Army Kwajalein Atoll (USAKA) in support of Strategic Defense Initiative Organization (SDIO) development activities. The STARS booster, depicted in Fig. 1, is a three-stage system with the first two stages composed largely of retired Polaris A3 missile assets. The third stage consists of a newly developed rocket motor and an attitude control system that has been adapted from salvaged Pershing II components. The Polaris A3 rocket motors and thrust vector control (TVC) actuation systems are generally in excess of 20 yrs old, and some refurbishment/retesting has been required[1] to ensure the suitability of this hardware for STARS purposes. All of the remaining guidance, navigation, and control components, as well as the supporting electrical/electronics system, have been replaced with state-of-the-art hardware to accommodate the requirements for mission flexibility, reliability, and low operational cost. The STARS booster is scheduled for its first test flight in early 1993. The launch site is located on the Kauai Test Facility (KTF) on the island of Kauai, Hawaii. This facility is located on the Pacific Missile Range Facility (PMRF) and is managed by the Department of Energy (DOE)/Sandia National Laboratories (SNL).

The guidance requirement of the STARS system is to deliver a payload to a target location in the vicinity of USAKA with re-entry conditions at or near those of an intercontinental ballistic missile. The program objectives of mission versatility and minimum operational cost have placed additional constraints on the choice of guidance hardware and algorithms.

The STARS guidance task can be discussed in terms of three flight phases. The first phase extends from liftoff until the dynamic pressure has decreased to a level acceptable for significant missile maneuvering. This typically occurs well into second-stage flight. The second phase begins at this point and continues until second-stage burnout. The third phase is initiated after second-stage burnout and is completed with burnout of the third stage.

The first flight phase is typical of most ballistic missiles with the primary objective of flying a pitch profile consistent with mission objectives that also minimizes vehicle angle of attack. A significant turn from the launch plane into the target plane is required during the second flight phase to avoid overflight of the populated island of Niihau, Hawaii (see Sec. IV). The second phase is further complicated by the lack of a thrust cutoff or other mechanism that would allow control of vehicle velocity. The resulting coast trajectory is therefore directly related to the actual performance of the missile relative to the pre-launch nominal. The third phase typically consists of a coast phase between second-stage burnout and third-stage ignition, with the third-stage burn occurring after apogee. The post apogee ignition is used to boost the velocity performance of the payload, rather than extend range or altitude as would be the case for a pre-apogee ignition time. A second out-of-plane turn is generally required for targets close in to USAKA to ensure that the instantaneous impact point (IIP) does not cross over any inhabited islands downrange (see Sec. VII). The third-stage burn must also eliminate the target miss due to off-nominal boost performance. Overall accuracy is limited by the lack of a velocity control capability for the third stage.

The guidance philosophy adopted for the STARS program is based on two underlying principles: 1) adapt proven algorithms to promote software reliability and 2) keep the algorithms as explicit as possible to maximize flexibility for achieving variable mission objectives and to reduce preflight engineering costs.

The organization of this paper is as follows: Sec. II discusses the STARS guidance, navigation, and control hardware. Section III describes the guidance algorithm used for the first phase, and Sec. IV gives details of the second-phase algorithm. Third-stage targeting and guidance are discussed, respectively, in Secs. V and VI. Section VII describes a typical STARS trajectory, with six-degree-of-freedom simulation results used to quantify system accuracy. Guidance system hardware in the loop tests are also discussed in Sec. VII. Conclusions are given in Sec. VIII.

II. Guidance System Hardware

The STARS guidance system hardware[1] has been chosen to provide 1) performance consistent with mission objectives, 2) flexibility for accommodating a variety of potential mission objectives, 3) reliability, 4) minimum operational constraints, and 5) low cost. The original Polaris A3 guidance hardware did not meet these requirements for several reasons: 1) hard-

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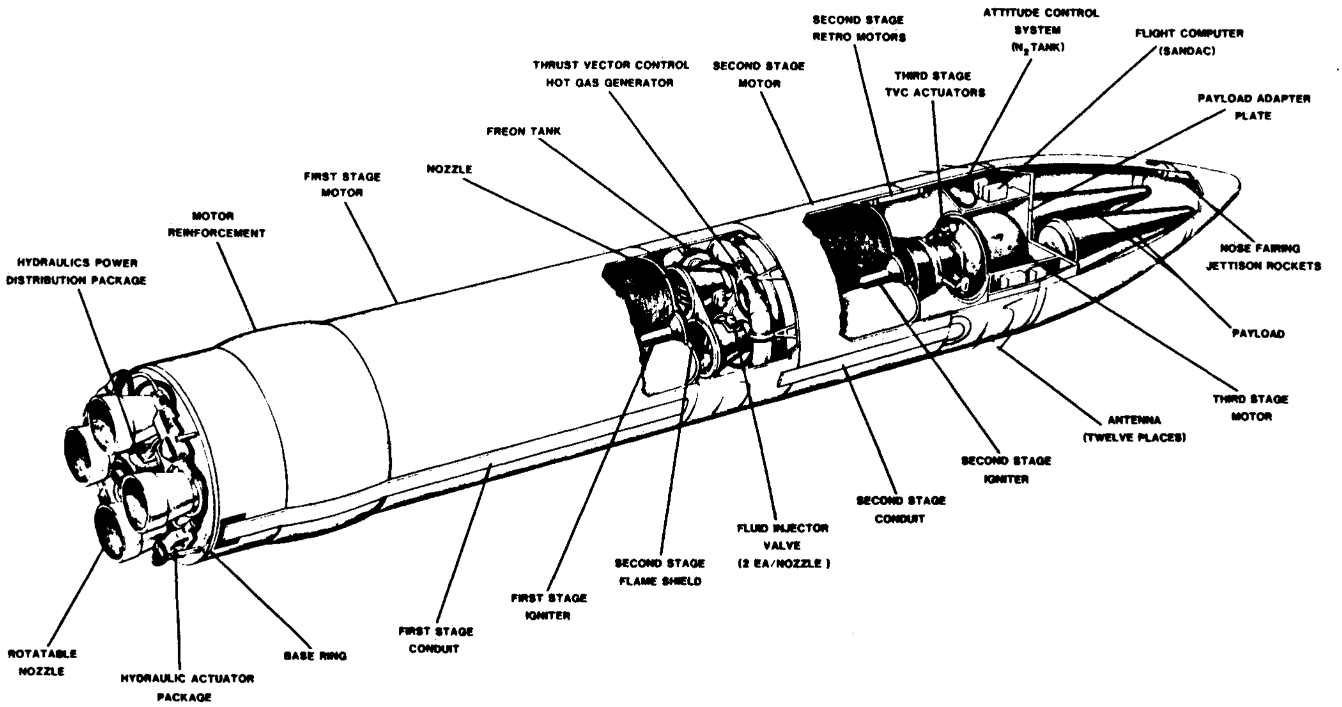

Fig. 1 STARS launch vehicle.

ware was specifically designed for the Polaris mission and did not allow mission flexibility; 2) hardware was developed in the 1960s and would have been difficult to support; and 3) acquisition and condition of the guidance sets was uncertain. The Polaris A3 TVC hardware has been retained for use by the first two stages of STARS, although the servo electronics have been updated.

The new STARS guidance, navigation, and control components are the inertial measurement unit (IMU), the flight computer, and the three-axis rate gyro package. The STARS IMU is based on an instrument sensor assembly (ISA) that is common to three military inertial navigation systems in production by the Honeywell Military Avionics Division. The ISA includes three Honeywell model 1342 ring laser gyros, three Sundstrand QA2000 accelerometers, and the associated support electronics. This ISA is used with two Air Force inertial navigation systems for fighter aircraft and the Army Modular Azimuth Position System (MAPS), which is used for artillery land navigation. The STARS IMU is procured as a standard Army MAPS system from Honeywell and is then slightly modified. The MAPS system was chosen because of its ready adaptability to the STARS application.

The current STARS design also includes a three-axis rate gyro package for measuring body rates. The rate gyro package will be eliminated on future flights, once the suitability of body rate data extracted from the laser gyros in the IMU is verified.

The STARS flight computer is a SANDAC V. This flight computer was developed at SNL to support a variety of research and development programs. Its primary features are high throughput, light weight, and expandability. Each SANDAC V processor module includes a Motorola MC68020 32-bit microprocessor, a 68881 floating-point coprocessor, and 128 KB of local RAM memory. The STARS configuration has four processor modules, although up to 15 modules are possible. The average computational throughput for each module is 2.5 million instructions per second (MIPS). The STARS flight computer also has a utility module with read-only memory for startup and power conditioning, a system input/output (I/O) module with interfaces for the gateway computer, parallel I/O that is used for the IMU, and a dual-channel Mil-Std-1553B bus interface that provides communication with the telemetry system, other subsystems, and payloads.

The STARS booster is equipped with a TVC capability for all three stages. The first stage has an electro-hydraulic TVC, the second stage uses an electro-mechanical fluid injectant TVC, and the third stage employs an electro-mechanical TVC. The TVC systems for the first two stages are essentially those used by the Polaris A3 missile and provide a three-axis control capability. The limited quantity of fluid injectant for the second-stage TVC imposes a significant constraint on the ascent trajectory. The third-stage TVC design provides for two-axis control during the third-stage burn. A cold gas attitude control system (ACS) is used to maintain roll attitude during the third-stage burn, and to achieve/maintain desired vehicle attitudes during the coast phases prior to and after the third-stage burn.

III. First-Phase Guidance and Steering

The first-phase guidance requirements for STARS are dominated by structural loading and control limitations. The principal guidance objective during this phase is to fly the missile along a gravity turn trajectory to minimize the angle of attack and therefore the aerodynamic loading of the missile. The loft on the trajectory must be consistent with mission range objectives, but should be chosen to minimize the use of second-stage fluid injectant when possible. First-phase guidance must also accommodate liftoff and staging.

The first-phase guidance and steering logic has been adapted directly from the original Polaris A3 missile since it provides a flight-proven atmospheric ascent procedure. This algorithm is based on a Q-system guidance[2,3] in conjunction with a Z-dot steering logic[4] to fly approximate-gravity turn ascent trajectories.

The Q-system guidance algorithm is based on a definition of the velocity-to-go, where the required or correlated velocity, v_C, is defined as the velocity that is required to place the vehicle on a ballistic trajectory to the target point from the current position with a fixed time of flight. If the missile velocity is given by v, the velocity-to-go is given by $v_{GO} = v_C - v$. A differential equation implicitly describing the velocity-to-go can be derived[2] as

$$\dot{v}_{GO} = -a_T - Qv_{GO} \tag{1}$$

where Q is a symmetric matrix of partials defined by

$$Q = \frac{\partial v_C}{\partial r} \Big|_{r_{\mathrm{TGT}}, t_{\mathrm{FLT}} = \mathrm{const}} \tag{2}$$

where r is the current missile position, r_{TGT} the target position, t_{FLT} the time of flight, and a_T the thrust acceleration. The initial conditions in Eq. (1) essentially determine the trajectory to be flown with the Qv_{GO} term serving as a corrective term to improve accuracy. The differential equation of Eq. (1) is integrated in real time with missile accelerometer feedbacks to define v_{GO}, with the usual guidance objective of placing the missile on a ballistic trajectory to the target by driving v_{GO} to zero. The Polaris inertial guidance frame is shown in Fig. 2. The initial condition on the x-axis velocity-to-go is primarily a function of the range to the target and is used only to determine the time for payload release. The initial condition on the y-axis velocity-to-go is zero for planar ascent trajectories. The initial condition on the z-axis velocity-to-go is used to determine the loft on the ascent trajectory.

The Z-dot steering law used to drive v_{GO} to zero is based on an exponential approximation of a gravity turn ascent trajectory.[4] Liftoff is accomplished by ignoring steering commands for a short time after launch and commanding a vertical attitude. A near-zero angle of attack can be maintained during the staging maneuver by biasing the z component of the velocity-to-go at a preset time after liftoff.

For planar trajectories the steering is implemented with nulling yaw rate commands proportional to

$$[\tau_Y a_{Ty} - v_{\mathrm{GO}y}] \tag{3}$$

where a_{Ty} is the y-axis accelerometer output, τ_Y is the yaw axis time constant, and $v_{\mathrm{GO}y}$ is the y component of the velocity-to-go. The pitch rate steering loop is implemented so as to emulate a gravity turn trajectory with a missile pitch rate command proportional to

$$[\tau_P a_{Ts} - v_{\mathrm{GO}z}] \tag{4}$$

where $a_{Ts} = a_{Tz} + (K_L)a_{Tx}$, τ_P is the pitch axis time constant, and v_{GO_z} is the z component of the velocity-to-go. The initial condition on $v_{\mathrm{GO}z}$ can be expressed as a linear function of K_L. The parameter K_L adjusts the loft of the ascent trajectory. The adjustment of K_L essentially controls the final attitude of the missile (e.g., the missile will have a final attitude of 45 deg relative to launch local level for $K_L = 0$).

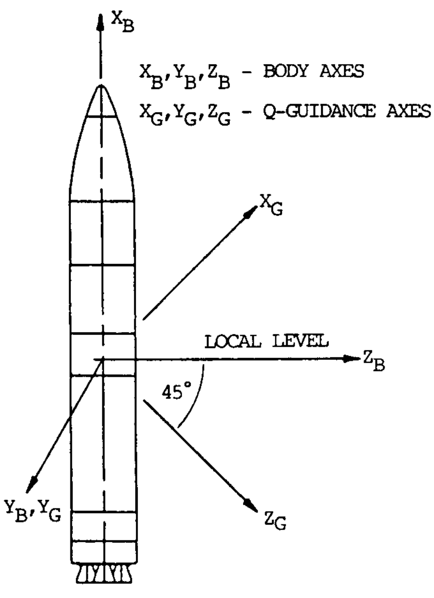

Fig. 2 Axes definitions at launch.

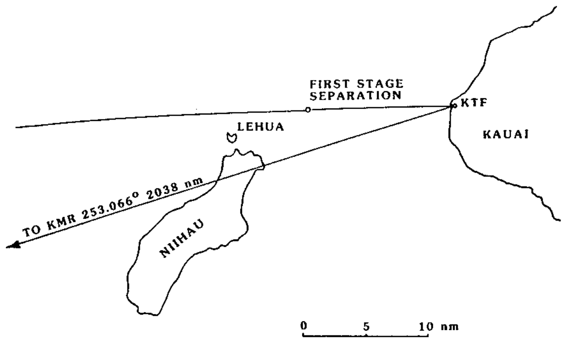

Fig. 3 Launch azimuth geometry.

The implementation of the Polaris guidance has been modified for the STARS application. Since the first-phase guidance does not extend to payload release, $v_{\mathrm{GO}x}$ does not need to be calculated. Since targeting accuracy is not critical during this phase, the Q-corrective term in Eq. (1) can be set to zero. Finally, an open-loop guidance strategy has been adopted during the staging event to improve autopilot margins. During this critical time guidance/steering commands are ignored, and vehicle attitude is controlled by an inertial angle-of-attack autopilot logic that is structured to minimize control effort. Overall guidance objectives are minimally affected by this approach, since first-phase objectives are achieved well before staging and second-phase guidance can correct for any resulting trajectory dispersions.

IV. Second-Phase Guidance and Steering

The second-phase guidance requirements are primarily dictated by mission and range safety constraints. The second phase algorithms must be flexible enough to accommodate a variety of potential mission objectives and must allow the execution of a considerable turn out of the launch plane into the target plane to satisfy range safety requirements. Figure 3 depicts the azimuth of a great circle path into the USAKA area. A due west or slightly northwest initial azimuth is required to avoid overflight of and appropriate separation from Niihau. The precision with which the second-phase guidance algorithms can place the third stage on a coast trajectory to the target is limited by the lack of second-stage velocity control. Limited thrust vector control fluid injectant availability imposes constraints on the amount of steering that is allowed. This limitation on TVC usage and the requirement for maximum velocity delivered to the target led to the early elimination of the generalized energy management system (GEMS)[5] as a potential second-phase guidance algorithm. Although GEMS explicitly constrains the target position as well as the time of arrival, the velocity control problem for the second stage is not eliminated, and significant maneuvering is generally a requirement. In addition, the GEMS procedure wastes some energy even for nominal performance to ensure that an under-performing missile can still reach the target. The approach taken for STARS is to allow a variable time of arrival and to apply all available energy to vehicle velocity.

The second-phase guidance algorithm selected for STARS is the ascent powered-explicit-guidance (PEG) used by the Space Shuttle.[6,7] PEG is a predictor-corrector algorithm that uses velocity-to-go as the independent variable, and determines the velocity-to-go required to place the vehicle on a coasting trajectory to the target that coincides with the powered trajectory at cutoff. The explicit nature of PEG makes it adaptable to a variety of mission objectives with cutoff constraints of altitude, velocity, flight-path angle, and trajectory plane. The downrange cutoff position is not explicitly constrained, nor is the time of flight. The PEG targeting for the STARS application attempts to fly the missile onto a trajectory that will take the third stage to its targeted position for third-stage ignition, based on the expected nominal ascent performance. Off-nominal ascent performance will produce off-nominal coast trajectories, since there is no mechanism for second-stage velocity control (i.e., cutoff is not possible). Guidance updates are terminated for under-performance when the thrust acceleration falls below a specified level and for over-performance when the guidance time-to-go reaches its minimum specified level.

The PEG algorithm is discussed in detail in Refs. 6 and 7. An overview is included here for completeness. The point mass equation of motion for a missile can be written as

$$\ddot{\boldsymbol{r}} = \boldsymbol{g} + \boldsymbol{a}_T \tag{5}$$

where $\boldsymbol{r}$ is the missile position, $\boldsymbol{g}$ is the acceleration due to gravity, and $\boldsymbol{a}_T$ is the thrust acceleration. The integration of Eq. (5) forward to the predicted time of cutoff, t_P, produces

$$\boldsymbol{v}_P = \boldsymbol{v} + \boldsymbol{v}_{\mathrm{GRAV}} + \boldsymbol{v}_{\mathrm{THR}} \tag{6}$$

$$\boldsymbol{r}_P = \boldsymbol{r} + \boldsymbol{v}(t_P - t) + \boldsymbol{r}_{\mathrm{GRAV}} + \boldsymbol{r}_{\mathrm{THR}} \tag{7}$$

where $\boldsymbol{r}_P$ and $\boldsymbol{v}_P$ are the predicted missile position and velocity at cutoff, respectively; $\boldsymbol{r}$ and $\boldsymbol{v}$ are the missile position and velocity at time t, respectively; $\boldsymbol{r}_{\mathrm{GRAV}}$ and $\boldsymbol{v}_{\mathrm{GRAV}}$ are the respective contributions in position and velocity due to gravity between the states of $(\boldsymbol{r}, \boldsymbol{v})$ and $(\boldsymbol{r}_P, \boldsymbol{v}_P)$; and $\boldsymbol{r}_{\mathrm{THR}}$ and $\boldsymbol{v}_{\mathrm{THR}}$ are similar contributions due to thrust. The velocity-to-go at t_P is defined by

$$\boldsymbol{v}_{\mathrm{GO}_{\mathrm{CO}}} = \boldsymbol{v}_C - \boldsymbol{v}_P = \boldsymbol{v}_C - \boldsymbol{v} - \boldsymbol{v}_{\mathrm{GRAV}} - \boldsymbol{v}_{\mathrm{THR}} \tag{8}$$

and the velocity-to-go at time t can be written as

$$\boldsymbol{v}_{\mathrm{GO}} \simeq \boldsymbol{v}_{\mathrm{THR}} = \boldsymbol{v}_C - \boldsymbol{v} - \boldsymbol{v}_{\mathrm{GRAV}} \tag{9}$$

with the position-to-go given by

$$\boldsymbol{r}_{\mathrm{GO}} \simeq \boldsymbol{r}_{\mathrm{THR}} = \boldsymbol{r}_C - \boldsymbol{r} - \boldsymbol{v}t_{\mathrm{GO}} + \boldsymbol{r}_{\mathrm{GRAV}} \tag{10}$$

The time-to-go is determined as

$$t_{\mathrm{GO}} = t_P - t \tag{11}$$

The missile state vector $(\boldsymbol{r}, \boldsymbol{v})$ is known from the onboard navigator, and the thrust integrals can be computed from a priori knowledge of $\boldsymbol{a}_T$. The gravity integrals in Eqs. (10) and (11) can be approximated from a coast trajectory that is near the powered flight trajectory. The predicted burnout state $(\boldsymbol{r}_P, \boldsymbol{v}_P)$ can then be calculated. Targeting logic is used to determine the $\boldsymbol{r}_C$ and $\boldsymbol{v}_C$ that will place the payload on the desired coast trajectory. The updated $\boldsymbol{v}_{\mathrm{GO}}$ is then determined[6,7] by adding $\boldsymbol{v}_C - \boldsymbol{v}_P$ to the current value.

The objective of the STARS second-phase targeting is to determine an appropriate cutoff state vector $(\boldsymbol{r}_C, \boldsymbol{v}_C)$ so that the turn out of the launch plane and into the target plane is accomplished, and second-stage burnout occurs with the third stage on the required trajectory to the third-stage ignition position. The latter requirement is accomplished with an initial target set that describes the desired state at third-stage ignition with a target altitude, velocity magnitude, and inertial flight-path angle. These quantities are used to define the semimajor axis, the angular momentum magnitude, and the eccentricity of the transfer ellipse. The true anomaly at the targeted position can also be calculated. The desired target plane is defined by a unit vector, $\hat{\boldsymbol{i}}_{\mathrm{TGT}}$, that is normal to the target plane containing the inertial third-stage ignition position vector, $\boldsymbol{r}_{\mathrm{TIG}}$, and a vector that lies at the intersection of the launch target planes, $\boldsymbol{r}_{\mathrm{INT}}$, from

$$\hat{\boldsymbol{i}}_{\mathrm{TGT}} = \hat{\boldsymbol{r}}_{\mathrm{TIG}} \times \hat{\boldsymbol{r}}_{\mathrm{INT}} \tag{12}$$

The determination of $\boldsymbol{r}_{\mathrm{INT}}$ is based on simulation data and is a function of the time at which the turn is initiated and the rate at which the vehicle is allowed to turn into the new plane. The predicted plane of motion is determined by the cross product

$$\hat{\boldsymbol{i}}_P = \hat{\boldsymbol{v}}_P \times \hat{\boldsymbol{r}}_P \tag{13}$$

where a circumflex denotes a unit vector. The predicted out-of-plane position component is computed from

$$r_{\mathrm{OP}} = \boldsymbol{r}_P \cdot \hat{\boldsymbol{i}}_P \tag{14}$$

and then $\boldsymbol{r}_C$ is approximated by

$$\boldsymbol{r}_C = \boldsymbol{r}_P - r_{\mathrm{OP}}\hat{\boldsymbol{i}}_P \tag{15}$$

The true anomaly at $\boldsymbol{r}_C$, the magnitude of $\boldsymbol{v}_C$, and the flight-path angle for the desired transfer trajectory can now be calculated from the conic section orbit equations. The required cutoff velocity vector is then determined as

$$\boldsymbol{v}_C = v_C[\sin(\gamma)\hat{\boldsymbol{r}}_C + \cos(\gamma)\hat{\boldsymbol{w}}] \tag{16}$$

where $\hat{\boldsymbol{r}}_C$ is the unit vector along $\boldsymbol{r}_C$, $\hat{\boldsymbol{w}}$ is the unit vector determined from the cross product of $\boldsymbol{r}_C$ and $\hat{\boldsymbol{i}}_P$, γ is the required cutoff flight-path angle, and v_C is the magnitude of the required velocity. Despite the plane control feature of this targeting, off-nominal ascent performance will result in some out-of-plane insertion error due to burnout state vector prediction inaccuracies. The out-of-plane errors are small for the expected three-sigma performance uncertainty and are not expected to pose any problems in achieving STARS mission objectives.

The second-phase guidance solution is updated at a 2-Hz rate for the STARS application. A cross-product steering law of the form $(\hat{\boldsymbol{a}}_T \times \hat{\boldsymbol{v}}_{\mathrm{GO}})$ is used to provide body rate commands that are input to the autopilot command calculation algorithm at 50 Hz. A wings-level roll command logic is similarly defined from the cross product of the commanded body y axis and the projection of the commanded body y axis onto the local level plane. This logic ensures proper antenna link margins during the out-of-plane turns. Identical steering logic is used during the third-stage burn.

V. Third-Stage Targeting

A targeting procedure based on the solution of Lambert's problem has been developed[8] for the STARS booster. This procedure is specifically designed to eliminate the target miss that results from the lack of velocity control in the first two stages. The procedure has been formulated to be as consistent as possible with the proven methods for solving the usual Lambert targeting problem. The core element in the STARS targeting algorithm is a solution routine for Lambert's problem that has been reproduced from Refs. 9 and 10. This code is valid for elliptical transfer trajectories with few restrictions.

Lambert's problem is that of determining the magnitude and direction of an impulsive delta-velocity maneuver that will place a spacecraft on a ballistic intercept trajectory between two inertial positions with a specified time of transfer:

Input: $\boldsymbol{r}_{\mathrm{TIG}}$, $\boldsymbol{r}_{\mathrm{TGT}}$, t_{TRF}

Output: $\boldsymbol{v}_C$ where $\boldsymbol{v}_{\mathrm{GO}} = \boldsymbol{v}_C - \boldsymbol{v}_{\mathrm{TIG}}$

The desired target position is represented here by $\boldsymbol{r}_{\mathrm{TGT}}$, the third-stage ignition position and velocity by $(\boldsymbol{r}_{\mathrm{TIG}}, \boldsymbol{v}_{\mathrm{TIG}})$, and the time-of-transfer by t_{TRF}. Lambert's problem[2] is formulated in terms of Keplerian trajectories, but a closed-form solution is not possible. The practical implementation of the Lambert solution typically requires adjustments to compensate for the idealized assumptions of the problem formulation. The two primary limitations on the accuracy of the solution result from the assumptions of an impulsive motor burn and a Keplerian gravity field. The accuracy of the solution can be greatly improved by using the position offset techniques described in Ref. 11. These corrective techniques compensate for

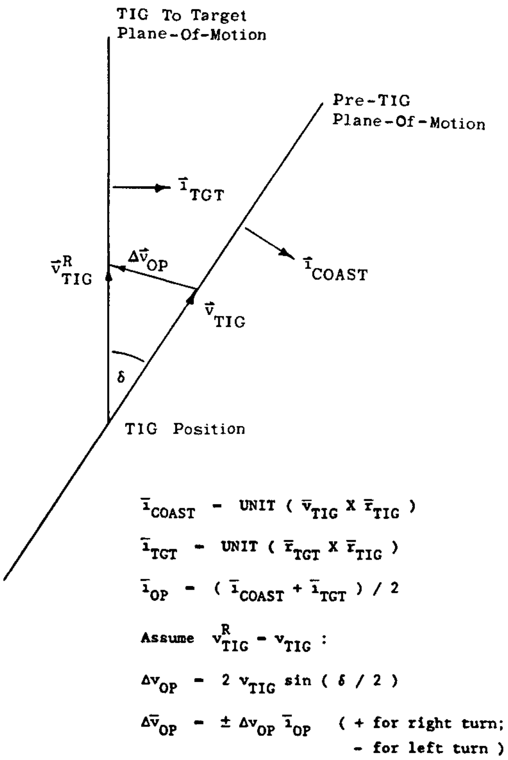

Fig. 4 Targeting plane control logic.

the finite motor burn with an ignition position offset, Δr_{TIG}, and for the nonspherical Earth gravity field with a target position offset, Δr_{TGT}. These offset positions are then used as inputs to the Lambert routine to produce an improved solution:

Input: $r_{\text{TIG}}^{\text{OFF}}$, $r_{\text{TGT}}^{\text{OFF}}$, t_{TRF}
Output: v_C

where

$$r_{\text{TIG}}^{\text{OFF}} = r_{\text{TIG}} + \Delta r_{\text{TIG}} \tag{17}$$

$$r_{\text{TGT}}^{\text{OFF}} = r_{\text{TGT}} + \Delta r_{\text{TGT}} \tag{18}$$

$$\Delta r_{\text{TIG}} = -(\|v_{\text{GO}}\|/2a_T)v_{\text{GO}} \tag{19}$$

$$\Delta r_{\text{TGT}} = r_{\text{IMP}} - r_{\text{TGT}} \tag{20}$$

The ignition position offset [Eq. (19)] is computed from terms related to the performance of the rocket motor, and the target offset [Eq. (20)] is the difference between the predicted impulsive impact position r_{IMP}, and the desired target position.

The usual Lambert case assumes that the performance capability of the individual stages can be varied to account for off-nominal situations, such that a fixed total time of flight is maintained. The performance of the individual STARS motors cannot be controlled, and the third-stage burn must eliminate target miss due to off-nominal ascent performance. The STARS targeting solution must also be consistent with the nominally fixed delta-velocity performance available from the third stage. The STARS modified Lambert targeting approach holds the time of ignition fixed and allows the total time of flight to vary for off-nominal ascent trajectories, while an outer-loop iterative algorithm ensures that the delta velocity, $\|v_{\text{GO}}\|$, required to intercept the target is identical with the nominal delta-velocity capability, Δv_{CAP}. This procedure produces a burn attitude that will eliminate the target miss due to off-nominal ascent performance. The solution obtained by this algorithm is not guaranteed to be unique. However, the flight time is smooth and monotonically increasing with the flight-path angle[5]; and this lack of uniqueness does not create operational difficulties given a reasonable initial guess of the time of transfer. This modified Lambert targeting procedure was motivated by a linearized targeting algorithm used successfully by the SNL sounding rocket program.

The Lambert solution also assumes that the transfer occurs in the original plane of motion. The STARS targeting problem requires the addition of a plane control logic. The launch plane and the target plane will not coincide, except for the nominal, because the targeting algorithm does not constrain the total time of flight. The plane control procedure assumes that the plane control delta velocity and the transfer delta velocity occur in separate components. The out-of-plane delta-velocity magnitude is determined as if the plane change occurs at an orbital node, and the direction is taken as the average of the unit vectors normal to the ascent and target planes. This plane control logic is depicted in Fig. 4. The delta-velocity capability of the third stage is reduced by the out-of-plane component, and the ignition velocity vector is rotated into the target plane prior to its use by the Lambert algorithm. A scale factor on the plane change angle is used to correct for the approximate nature of this logic.

Targeting computations must be completed prior to the time of ignition (TIG) of the third stage, so that the attitude control system will have time to orient the vehicle to the desired burn attitude. The inertial position vector of the third stage at ignition is required by the Lambert algorithm, and a post-second-stage state vector must be provided by the navigator and numerically integrated forward to TIG. The propagation routine for STARS has been reproduced from Ref. 12.

The inertial target position vector must be also determined as an input to the Lambert algorithm. The STARS target point can be defined by a fixed latitude, longitude, and altitude. For a known time of intercept, an inertial target position vector can be computed that compensates for the rotation of the Earth.

The STARS targeting algorithm initially uses the predicted TIG position vector, the target position vector, and the nominal time of transfer with the Lambert solution routine. The

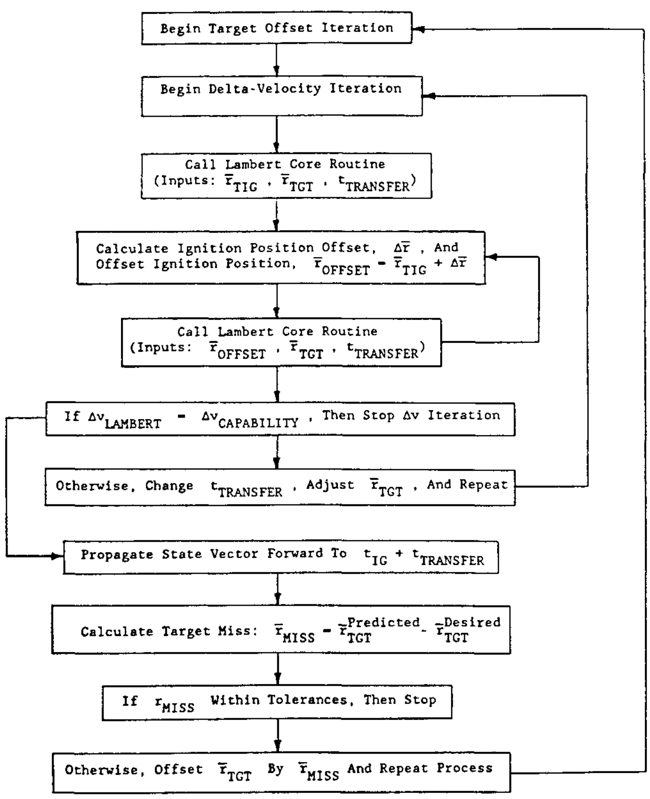

Fig. 5 Modified Lambert targeting.

initial position offset is then determined from this impulsive solution, and an improved solution is obtained with a second call to the Lambert core element. The second call to the Lambert solution routine produces a valid solution for the input problem, but the result is not generally useful for the case of a fixed delta-velocity capability. The useful solution must ensure that the required and available delta velocities are identical. This is accomplished with an outer-loop iteration that uses time of transfer t_{TRF} as the independent variable and the difference between the required and available delta velocities as the dependent variable, $\|v_{GO}\| - \Delta v_{CAP}$. A simple Newton-Raphson iterative algorithm, similar to that used by the Lambert core routine,[9,10] is used to achieve outer-loop convergence. Each iteration on the time of transfer requires two or more calls to the Lambert core routine: one with no offset on the TIG position vector and one or more with a TIG position offset applied. Numerical experience with the STARS trajectories suggest that two or three calls with an ignition position offset are required per iteration for maximum accuracy. The inertial target position vector is recalculated as the time of transfer is varied. Once a converged solution has been determined, the target position offset is calculated by numerically propagating the state vector forward to the latest time of intercept. A ballistic propagation of the Lambert solution produces a target offset that accounts for the non-Keplerian nature of the gravity field. The target miss vector is then used as a target offset on the original target vector to produce a pseudo-target position. This pseudo-target can be used as the next input target position, and the procedure can be repeated. This iteration can continue until adequate accuracy is achieved. Given a reasonable estimate of the time of transfer, this modified Lambert procedure provides a reliable, near-explicit targeting algorithm for the STARS third stage. The general targeting procedure is shown in Fig. 5.

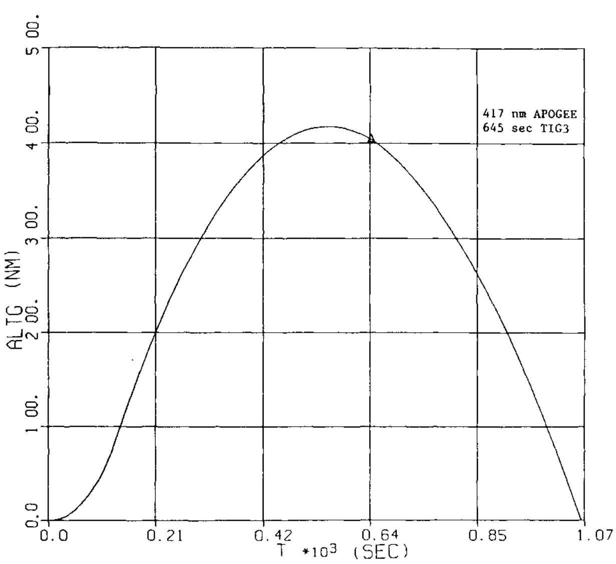

Fig. 6 Altitude vs time.

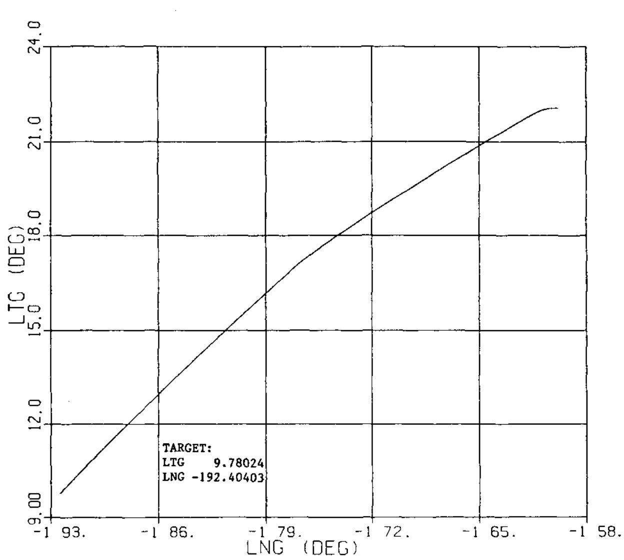

Fig. 7 Latitude vs longitude.

Table 1 Typical mission timeline

Time	Event description
−165.0	Begin IMU alignment
−45.0	Begin geographic navigation
−28.0	Begin 1st stage TVC test
−2.0	Begin inertial navigation, 1st-phase guidance
0.0	Ignition signal sent
0.067	Lift-off (1st motion)
2.18	Begin pitch program
40.18	Maximum dynamic pressure
60.90	Arm: 1st separation, 2nd-stage ignition (1.3 g's and decreasing)
61.20	1st Separation initiation, 2nd-stage ignition
61.229	Separation first relative motion
61.24	2nd-Stage autopilot on
61.264	2nd-Stage full thrust
71.30	Begin 2nd-phase guidance (begin Niihau turn)
140.18	2nd Separation initiation, 2nd retro ignition (0.1 g's and decreasing)
140.18	ACS (all thrusters) on
160.0	Ascent shroud jettison
170.0	Begin 3rd-stage targeting task
180.0	Begin ACS maneuver to orient 3rd stage
645.0	Pitch and yaw ACS thrusters off
645.0	3rd-Stage ignition; autopilot, guidance on
685.0	3rd-Stage burnout; pitch and yaw ACS on
690.0	Begin re-entry prediction task
700.0	Begin reorientation for payload release
750.0	Payload release
760.0	Begin reorientation for retro burn
800.0	3rd-Stage retro ignition
1027.9	Payload entry interface (300 kft)
1075.4	Payload impact

The overall targeting computation requirement on the SANDAC V is ~10 s with a 0.1-s integration step size. This time requirement can be reduced to <1 s without serious degradation of delivery accuracy, if the STARS six-degree-of-freedom simulation is used to determine the target offset data a priori.

VI. Third-Stage Guidance

The third-stage targeting algorithm has been formulated to generate a burn direction that will eliminate the target miss due to off-nominal ascent trajectories. However, the third stage also lacks velocity control, and a significant target miss can result from third-stage performance perturbations. The objective of third stage guidance is to reduce this target miss.

Several third-stage guidance strategies have been considered. The first approach is that of a Lambert guidance update that uses the time of transfer from the targeting solution as the fixed time of intercept. This approach is effective in eliminating boost phase target miss, but does not reduce the target miss due to off-nominal third-stage performance. The second approach employs the modified Lambert procedure used for targeting in an active guidance mode. This procedure produces a nearly constant attitude burn and does not waste energy. The delta-velocity loop in this algorithm, when used with integrated accelerometer feedback, forces the guidance solution to be consistent with the third-stage nominal delta-velocity capability. This eliminates target miss due to thrust perturbations (constant total impulse and variable burn time). Miss due to

Table 2 Accuracy data

Boost perturbation	Miss distance, ft			3rd-Stage perturbation	Miss distance, nm		
(3-sigma errors)	Lat.	Long.	Total	(3-sigma errors)	Lat	Long.	Total
1st-Stage impulse	631	460	781	3rd-Stage impulse	4.10	3.23	5.22
1st-Stage mass	750	246	789	3rd-Stage mass	2.83	2.19	3.58
1st-Stage thrust	556	445	712	3rd-Stage thrust	0.13	0.20	0.24
2nd-Stage impulse	1306	1133	1729	Total miss (rss)	4.98	3.91	6.33
2nd-Stage mass	987	871	1316				
2nd-Stage thrust	759	69	762				
Total miss (rss)	2127	1587	2654				

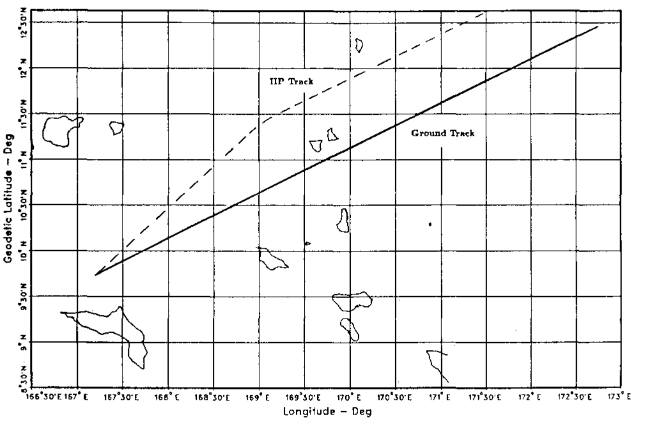

Fig. 8 Third-stage ground track and IIP.

mass and total impulse uncertainties remains unchanged. The third guidance method considered was GEMS. This algorithm has accuracy properties similar to the modified Lambert approach, but constrains time of flight by wasting energy in a manner that generally requires significant vehicle maneuvering. A fourth guidance approach (which is similar to GEMS) is based upon Ref. 13, where the objective is to guide the vehicle so that the Lambert velocity-to-go is nulled with the final thrust direction along a range insensitive direction. This technique has the potential to significantly reduce the target miss due to all performance variations. Simulation results indicate, however, that the achieved accuracy is dependent on guidance updates late in the burn.

The modified Lambert procedure has been adopted as the baseline third-stage guidance algorithm. Timing studies[8] on the SANDAC V have indicated that an update rate of at least 1 Hz is possible.

VII. Simulation Results

The STARS guidance algorithms have been tested on several proposed trajectories with a six-degree-of-freedom simulation. Three early trajectories are discussed in Ref. 8. A typical recent trajectory is depicted in Figs. 6 and 7. A typical timeline is shown in Table 1. Accuracy data for this trajectory are given in Table 2.

The trajectory shown in Figs. 6 and 7 is targeted to a point just north of the Kwajalein Atoll and includes an out-of-plane turn during the third-stage burn to ensure that the IIP does not traverse any populated area. This portion of the trajectory is depicted in Fig. 8. A simple IIP predictor is included in the flight software to adjust the time at which this turn is initiated, so that the IIP is safe for off-nominal situations. This simulation included a 500-lb payload and achieved a re-entry velocity of ~20,000 ft/s. The three-sigma target miss for this trajectory includes about ±5.74 nm along track, with about ±2.68 nm of cross-track error. Additionally, there are cross-track errors of approximately ±3.0 nm due to navigation errors. The total cross-track error will increase to about ±4.0 nm.

This trajectory has been generated with flight software that has been interfaced with a STARS six-degree-of-freedom simulation running on a mainframe computer. The flight code has also been integrated into the STARS SANDAC V flight computer. Extensive testing for the first flight test has been completed with the SANDAC interfaced to a STARS six-degree-of-freedom simulator that is capable of running in a real-time environment with hardware-in-the-loop.

VIII. Conclusions

The STARS guidance objectives have been achieved with the use of three flight-proven algorithms. The atmospheric first phase is flown with a Q-system guidance and Z-dot steering. The second phase uses the Space Shuttle ascent PEG algorithm to place the vehicle on a desired coast trajectory with an out-of-plane turn. Since the second stage lacks a velocity control capability, the coast trajectory will vary with off-nominal performance. A modified Lambert procedure is used for third-stage guidance to eliminate the boost phase and one source of the third stage contribution to target miss. The third-stage guidance is also capable of executing a second out-of-plane turn. Since the third stage also lacks velocity control, the target miss is a function of the remaining sources of third-stage performance uncertainty.

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