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The Analysis and Configuration of a Control System for a Mars Propulsive Lander

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Abstract

Several control-system mechanizations for a planetary propulsive lander are presently being studied at the Laboratory. One of these, a soft-landing descent system that combines inertial and radar sensing techniques, controls the capsule from motor ignition to touchdown on the surface of a planet. A digital computer is the basic tool for analysis and simulation of the system and its environment.

The Analysis and Configuration of a Control System for a Mars Propulsive Lander

I. Introduction

Presently, there is a variety of methods being considered for injecting a capsule into a planetary impact trajectory. For this study, it was assumed that the planetary entry capsule is transferred from an orbital trajectory into a Mars impact trajectory under control of a pre-entry system. This system would contain complete three-axis attitude control (e.g., cold-gas reaction jets), and a thrust-vector control system for the deorbit maneuver. The system is inertially referenced, utilizing strap-down gyroscopes.

For the type of maneuver described above, it has been determined that at the completion of the deorbit phase, capsule velocity is nominally 15,000 ft/sec at an altitude of 800,000 ft. The entry corridor is constrained to path angles from 14 to 19 deg ± 1 deg.

The capsule now begins the ballistic entry portion of the descent. During this time, no control system is active, and the craft's aerodynamic properties dictate attitude and velocity. Approximately 14,000 ft/sec of velocity is removed during this period, due to the ablative heat shield. The aeroshell configuration is chosen to maximize the aerodynamic stability of the capsule during entry.

A soft-landing descent system must operate successfully over the range of possible ignition conditions. System design is somewhat complicated by the large range of possible initial path angles and velocities that may be encountered at motor ignition. This range of angles and velocities is the result of uncertainties in several areas including the atmospheric model, the nature of the winds, and the entry conditions. A soft vertical landing of a capsule will require continual in-flight adjustments of both the direction and magnitude of the velocity vector. A successful descent is defined in this Report as a final vertical velocity less than 25 ft/sec, a final horizontal velocity less than 5 ft/sec, and an attitude within 10 deg of the local vertical. In light atmospheres, a planetary entry capsule can be supersonic at ignition with a velocity greater than 1000 ft/sec. In denser atmospheres, the velocity is near terminal ($\sim$400 ft/sec). A major factor in determining the conditions at ignition is wind, which can increase the vehicle's ground speed by more than 20%. The problem is further complicated by random wind gusts and wind shear.

With these problems in mind, three digital computer simulation programs were developed as tools for the investigation and optimization of various control systems. They are as follows:

(1) A 3-deg-of-freedom model (motion-in-a-plane).

(2) A 6-deg-of-freedom program.

(3) A modification of the 6-deg program used to examine the effect of winds during ballistic flight.

Significant features of the simulation programs are given in Table 1. To date, results of the control system studies have indicated a single configuration that is worthy of further study and analysis.

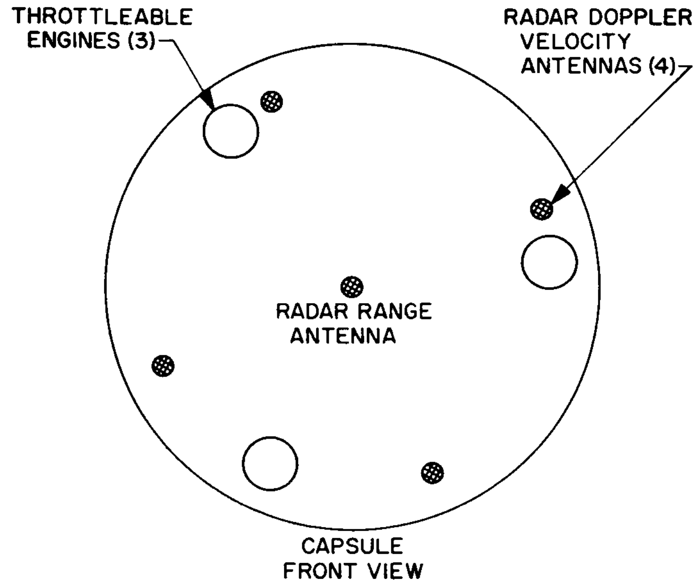
The capsule configuration, used for this study, and some of its parameters are shown in Fig. 1. Note that three bi-propellant throttleable engines are used. Pitch and yaw attitude control is accomplished by throttling these engines differentially. Roll control is provided by a single jet vane in the thrust chamber of each engine.

The radar configuration consists of a single, pencil-beam, slant range radar mounted along the roll axis. This system provides the ignition mark (at a slant range of 20,000 ft), and slant range information during the descent. The three components of velocity as well as slant range are obtained by using four symmetric doppler radar beams arranged on a cone about the roll axis, at an angle of 20 deg. Only three of the four beams are needed to derive all information.

Table 1. Features of digital simulation programs

Capsule simulation	Control system simulation	Radar system	Atmosphere
1. Mass depletion	1. Complete mode switching for automatic landing	1. Simulated noise on all signals	1. Wind: a. continuous b. shear c. gradient
2. C.g. offsets	2. Gyro dynamics and error terms	2. Terrain bias errors	d. random gusts e. surface
3. Boresight errors	3. Actuator dynamics and hysteresis	3. Acquisition logic	f. tailoff 2. Density computed
4. Fuel slosh			3. Mach number (acoustic velocity) computed
5. Complete aerodynamics			

Throughout the following discussion, four atmospheric models will be referenced: AM-10, AM-4, AM-7, and AM-8. The salient characteristics of each, as well as the nominal ignition conditions, are shown in Table 2. AM-10 represents the densest, AM-4 a nominal, and AM-7 and AM-8 the lightest atmospheres. The highest ignition velocities occur in AM-7; AM-8 is characterized by very shallow path angles at ignition. From the data in Table 2, the large range of initial conditions over which the control system must operate is apparent.



ENTRY WEIGHT: 2725 lb (EARTH)
DIAMETER: 16 ft
SHAPE: 60-deg SPHERE CONE
BALLISTIC COEFFICIENT: $M/(C_d \cdot A) = 0.27$
ENGINES: THREE BIPROPELLANT, THROTTLEABLE
HEAT SHIELD: ABLATIVE
ANTENNAS: FOUR DOPPLER VELOCITY (ANY THREE CAN BE USED TO SUPPLY ALL VELOCITY DATA)
ONE SLANT RANGE RADAR ANTENNA

Fig. 1. Capsule configuration and parameters

Table 2. Atmospheric characteristics

Atmosphere model	Density ($\times 10^{-5}$), slugs/ft^3	Pressure, millibars	Continuous surface wind, ft/sec	Nominal ignition conditions		
				Velocity, ft/sec	Altitude, ft	Path angle, deg (WRTH)
AM-4	4.98	10	155	587	17,400	60.9
AM-7	1.32	5	220	1075	14,600	46.8
AM-8	2.56	5	220	915	12,600	38.9
AM-10	7.44	20	110	443	19,800	81.9

II. A Nominal Landing

Four major assumptions in the analysis of the control system follow:

(1) Capsule configuration is that described in the Introduction.

(2) Ignition is started by a 20,000-ft slant range mark from the radar subsystem.

(3) At least three of the four radar beams remain locked throughout the flight.

(4) No staging of the aeroshell is performed.

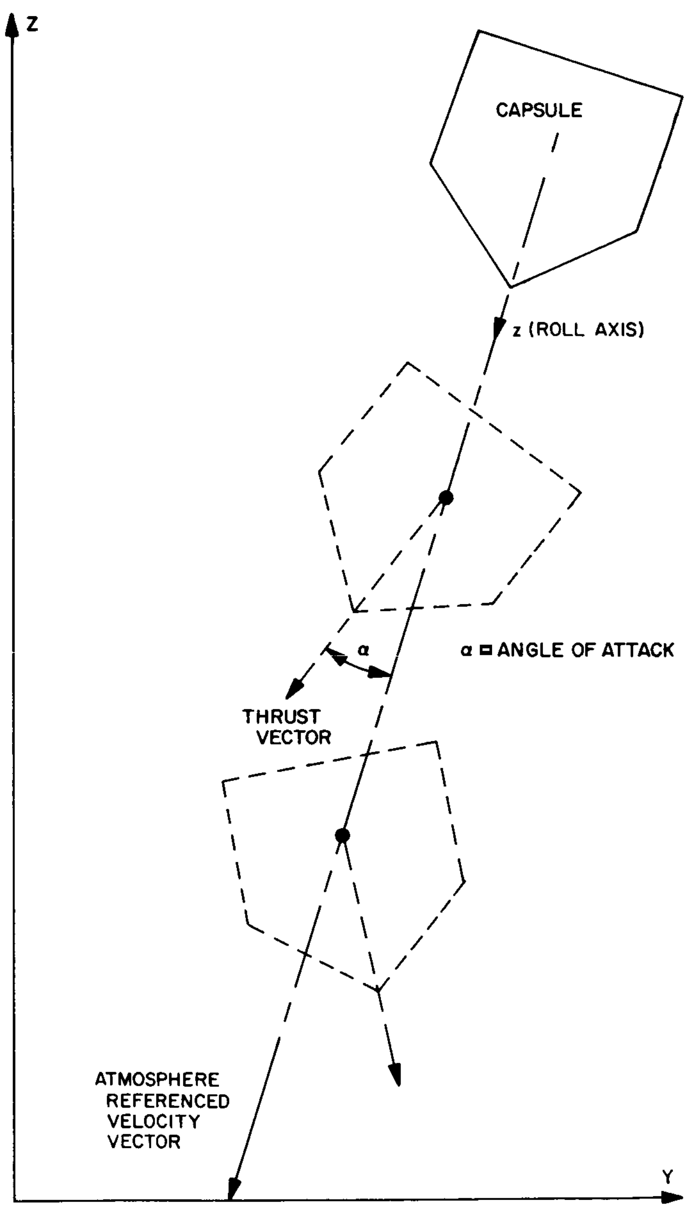


Fig. 2. Aerodynamic oscillations

The following nominal landing case uses the AM-8 atmospheric model with worst case continuous winds (no gusts), and an entry angle of 20 deg.

Prior to ignition, the vehicle experiences aerodynamic oscillations about a small attack angle (Fig. 2). These oscillations are on the order of 10 deg (Fig. 3), and are the result, primarily, of poor aerodynamic damping in the transonic region coupled with a c.g. offset of 0.1 ft.

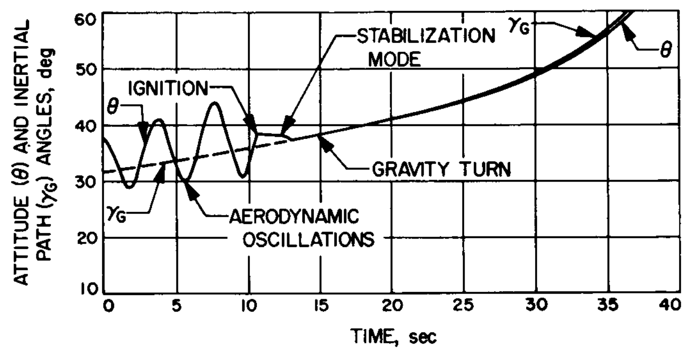


Fig. 3. Three-deg-of-freedom data

At a slant range mark of 20,000 ft as indicated by the radar system, the control system is activated. This occurs at an altitude of 11,000 ft and at an attitude angle (with respect to local horizontal) of 36 deg, which corresponds to the atmospheric path angle. Ground-referenced velocity is approximately 1000 ft/sec.

Ground speed is the velocity of the capsule with respect to the planet surface whether or not a steady-state wind is present. Aerodynamic velocity is the velocity of the capsule with respect to the planetary atmosphere.

For the first 2 sec after ignition, the system is in the stabilization mode. In this mode, the vehicle's attitude is inertially stabilized, to counteract any thrust offset or aerodynamic torques. This mode uses the wide-angle position gyros (uncaged at ignition) with rate gyros for compensation (Fig. 4). This rate-plus-position signal differentially throttles the engines to produce the necessary stabilizing torques. During stabilization, a constant axial acceleration is commanded, nominally 26 ft/sec^2. In Fig. 3, curve 1, stabilization of the vehicle's attitude with less than 0.1 deg change can be seen. This figure illustrates a typical 3-deg-of-freedom run.

Almost immediately after ignition, exhaust gases from the engines interfere with the normal aerodynamic flow,

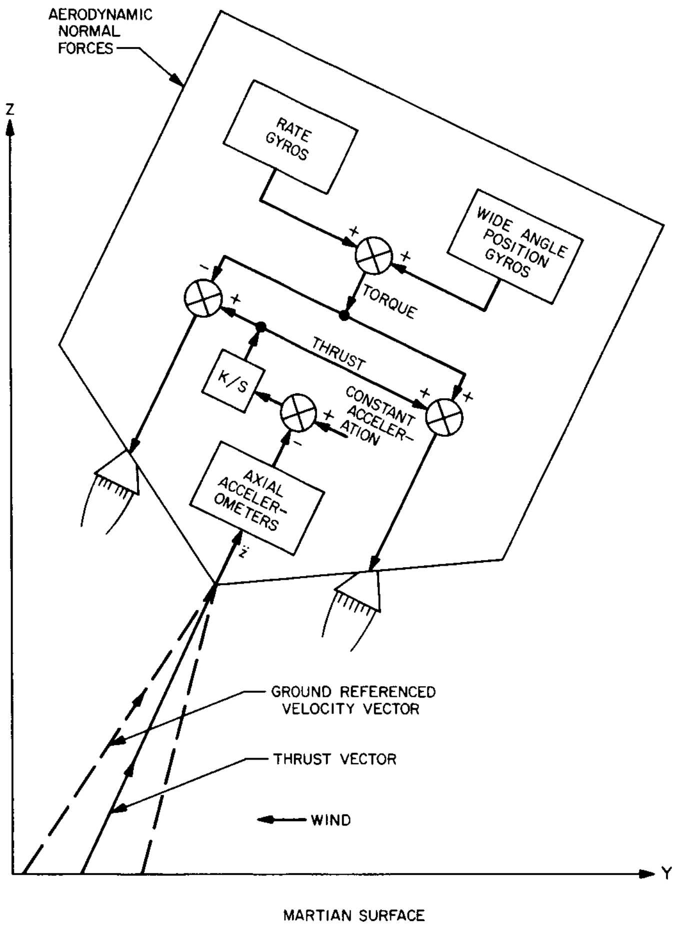


Fig. 4. Stabilization mode

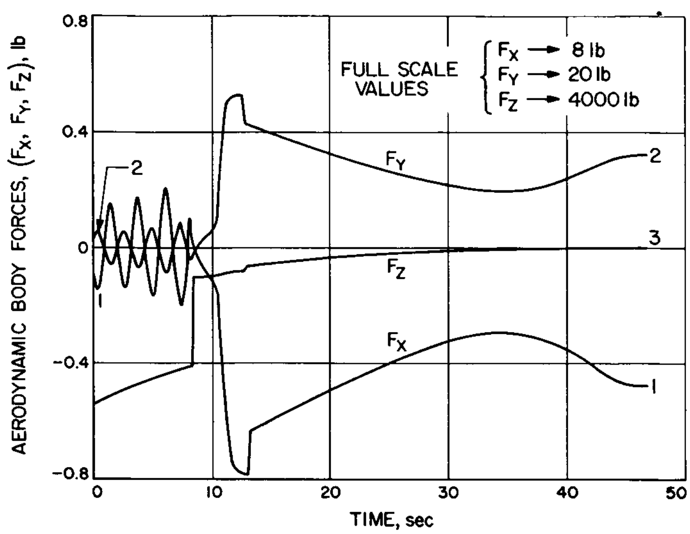


Fig. 5. AM-8 aerodynamic forces (6-deg-of-freedom)

which causes a general reduction in aerodynamic drag force. Figure 5 shows the components of the aerodynamic forces in a body-fixed coordinate system. Curve 3 of this figure is the drag plotted with a full scale value of 4000 lb. Just prior to ignition at 8.4 sec, the curve shows a vehicle velocity of over 1025 ft/sec, causing a drag force of 2000 lb. After ignition, the plume effect reduces drag to 500 lb. A second small reduction in drag occurs just before 13 sec when the capsule goes subsonic.

During the stabilization mode, the gyro-controlled attitude reference system forces the capsule to maintain its preignition attitude. At completion of the stabilization mode, the gravity-turn mode is initiated.

After the radar system has acquired the planet, a gravity turn is initiated and the capsule is commanded to zero angle with respect to the ground-referenced velocity, V_G. If a steady state wind is present, the attitude of the capsule will change when radar control is initiated. Thus, when a steady state wind is present, the capsule

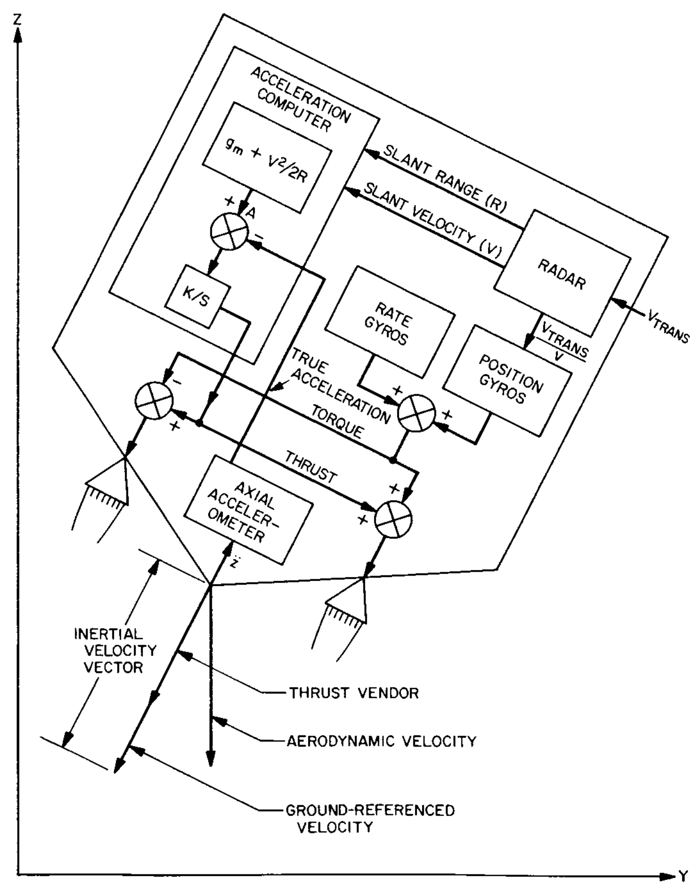


Fig. 6. Gravity turn mode

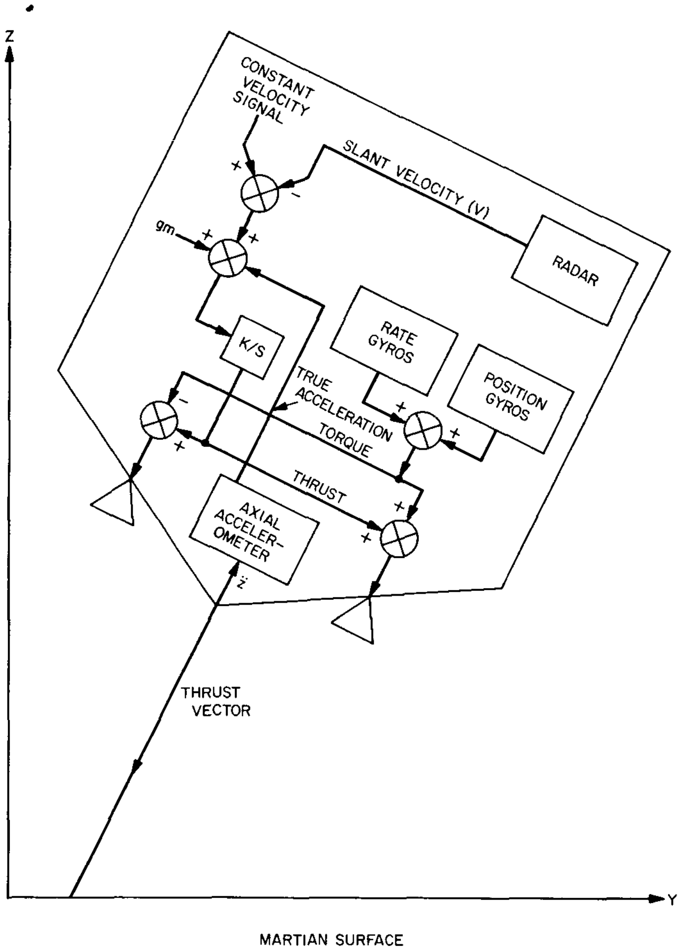


Fig. 7. Terminal mode

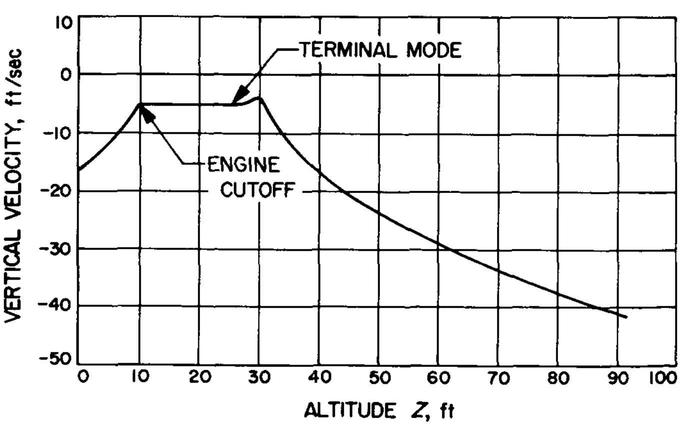


Fig. 8. Terminal mode velocity

will actually be flying at a non-zero angle of attack. During this mode, the vehicle's attitude must be such that the thrust vector is aligned with the ground-referenced velocity vector. Considering the worst-case wind shear and wind direction, the path angle of the ground-referenced velocity is 32 deg (Fig. 3). Following the gravity turn, the ratio of transverse to total ground-referenced velocity, as measured by radar, is used to torque the position gyros (Fig. 6). This velocity ratio is a measure of the gravity-turn angular error, which is reduced to near zero by the engine torquing system. As seen in Fig. 3, this gravity turn "acquisition" occurs from 12.6 to 13 sec, at which time the roll axis is aligned to the ground-referenced flight path angle. The vehicle maintains this attitude until the terminal mode is initiated.

During the gravity-turn mode, the instantaneous axial acceleration (A) required to land with zero vertical velocity is computed using range (R) and velocity (V) from the radar system:

$$A = g_m + \frac{V^2}{2R}$$

where g_m = Martian gravity. The actual capsule axial acceleration ($\ddot{z}$), as sensed by the accelerometer, is compared with (A), and the integral of the error is used to set the thrust level (Fig. 6). The above equation is the required acceleration for zero velocity at zero range. To permit the use of a terminal mode, the radar range signal is biased by 30 ft to force a 5 ft/sec velocity at this altitude. When this velocity is reached, the system is switched into the terminal mode.

At an altitude of 30 ft, the control system removes the torquing signals from the gyros, thus locking the craft's attitude in an inertial hold. The computed acceleration (A) signal is removed from the thrust loop, and a 5 ft/sec constant velocity is commanded (Fig. 7). This constant velocity can be seen in Fig. 8. When the craft reaches an altitude of 10 ft, the engines are shut off, and the capsule drops to the surface. Typical terminal conditions are:

(1) A vertical velocity of 18 ft/sec.

(2) A horizontal velocity of 2 ft/sec.

(3) An attitude angle of 85 deg with respect to horizontal.

A typical range curve, showing the capsule trajectory, is in Fig. 9, and a typical out-of-plane trajectory (no crosswind) is shown in Figure 10.

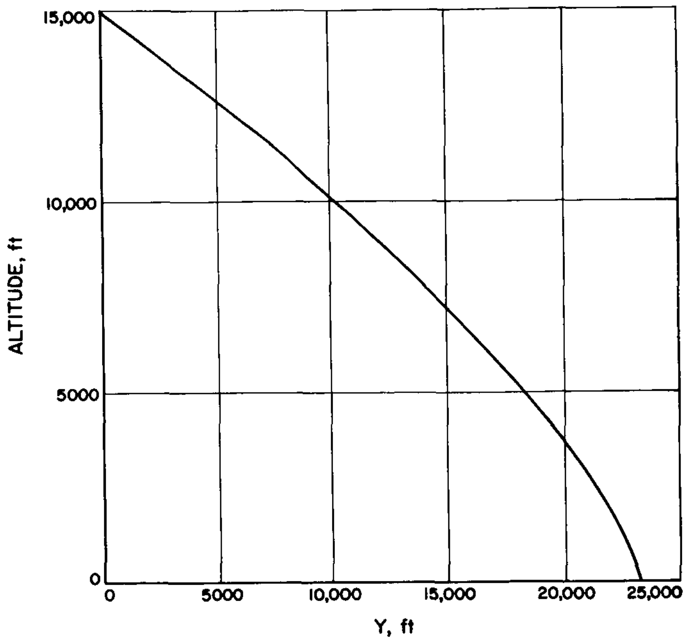


Fig. 9. AM-8 capsule trajectory

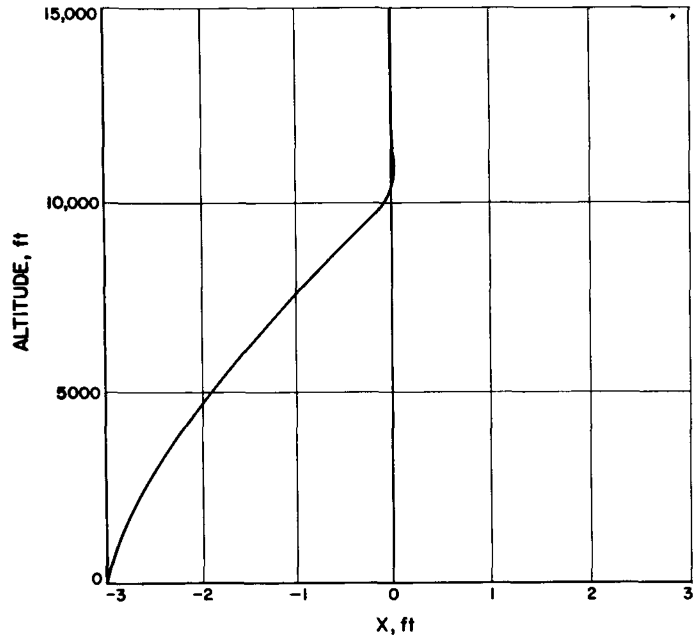


Fig. 10. AM-8 out-of-plane trajectory

III. Description of Control Modes

A detailed description of each of the control modes mentioned above follows.

A. Motor Ignition

The motor ignition method is a slant range mark, nominally 20,000 ft. This method is not entirely satisfactory from a control viewpoint, because, in the less dense atmospheres where the propulsion system must take out the most ΔV, the path angles are shallow and the actual altitude is much less than the slant range. In atmospheres of greater density, the path angles are steep and the altitude is almost equal to the slant range. This condition may mean that the densest atmosphere will require the maximum amount of fuel.

An omnidirectional altitude marking radar would be a better method of ignition, but such a system is difficult to implement and requires large amounts of power. A compromise could be effected if the doppler and range descent radar were turned on and ignition were initiated when the beam indicating minimum altitude matched a preset value. This method will always cause ignition at an altitude greater than the slant range method described above, and thus help minimize dispersions in ignition altitude.

B. Stabilization Mode

The purpose of the 2-sec stabilization period is inertial stabilization of the capsule to permit radar system lock. During this radar acquisition period, the rate of velocity change with respect to the planet must be less than 80 ft/sec^2, and the slant-range change rate must be less than 4000 ft/sec. Once radar system lock has occurred, ten times these values are allowable before tracking capability is lost.

Because of random wind gusts, the aerodynamic path angle will vary rapidly; if no stabilization mode were included, the capsule would attempt to null the aerodynamic angle-of-attack causing large random changes in capsule attitude. An inertial hold mode, however, will make the capsule insensitive to these atmospheric variations.

The stabilization mode torquing system is shown in Fig. 11. The wide-angle gyros are uncaged at ignition and their output (proportional to angular error) provides torquing signals to the engines. The rate gyros provide the necessary damping for the loop. The three kinds of torque that must be nulled by the pitch-yaw attitude control system are aerodynamic torques, thrust-offset torques, and fuel-slosh torques. In a heavy atmosphere (AM-10) with wind gusts, the aerodynamic torques can be on the order of 500 ft-lb. For an offset of 0.1 ft, a

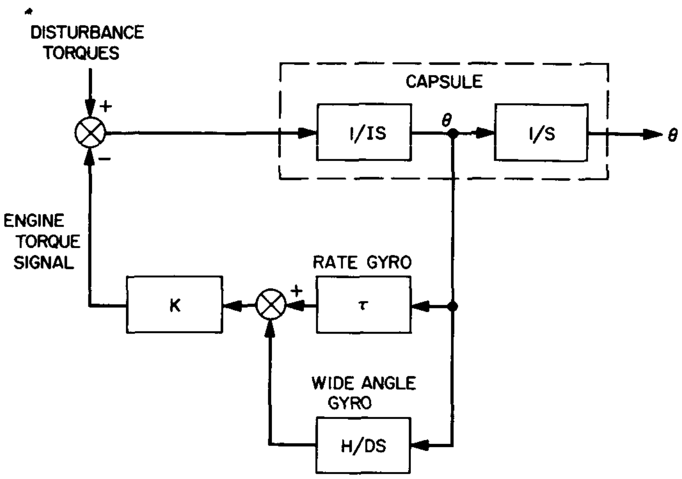


Fig. 11. Attitude stabilization control loop

torque as high as 300 ft-lb can occur when operating at maximum thrust. A third disturbing torque is created by fuel slosh. A slosh-torque profile for an AM-10 atmosphere case is shown in Fig. 12.

Roll control is maintained by jet vanes placed in each engine, the control loop of which (basically the same as the pitch-yaw configuration), requires wide-angle gyros and rate gyros. The major sources of roll disturbance torque are thrust misalignment and unsymmetrical aerodynamic forces. At present, little is known about the magnitude of the latter effect.

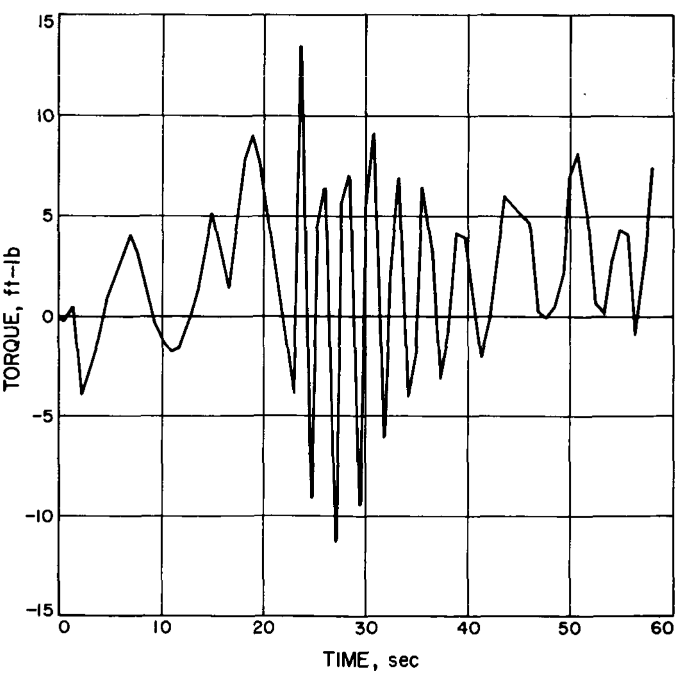


Fig. 12. AM-10 fuel slosh torque

During the stabilization mode, no radar information is available and a constant acceleration level is commanded. This acceleration level is chosen to provide nominally mid-range engine thrust, and to allow the use of large differential torques, if necessary. An axial accelerometer senses the true acceleration, including the drag effect, and closes the loop around the commanded acceleration. A 26-ft/sec^2 acceleration level results in approximately 2000 lb of total thrust during this mode.

C. Gravity Turn Mode

1. Computed thrust. After stabilization is complete and at least three radar beams have acquired the surface, the system begins a gravity turn using the law $A = V^2/2R + g_m$, to calculate the acceleration that controls the thrust loop (a simplified block diagram is Fig. 13). The total thrust can be slewed at 7000 lb/sec with this loop. The thrust saturation signal, representing feedback from throttle actuator positions, is used to prevent maximum or minimum thrust from all engines simultaneously. This is necessary to permit the engines to generate the corrective torques required. Thus, if maximum thrust and large torques are needed simultaneously, the acceleration requirement is sacrificed to attitude stability.

Alignment of the thrust exactly opposite the ground-referenced velocity vector is most important during this mode. Lack of this alignment results in degradation of the terminal conditions. During the stabilization period, the original atmosphere-referenced attitude is held. At the start of the gravity turn, however, the gyros must be torqued to make the vehicle attitude coincident with the ground-referenced velocity. The doppler radar resolves ground-referenced velocity into capsule body-fixed coordinates, and, if the attitude is correct, only the velocity component along the roll axis is nonzero. If, for example, the body-referenced V_x is positive, positive yaw torque should be applied; if the body-referenced V_y is positive, however, negative pitch torque should be applied. The pitch and yaw control loops for the gravity turn mode are shown in Fig. 14.

Referring back to Fig. 3, the gravity turn was initiated at 12.6 sec and the gyros were torqued for approximately 0.5 sec, at which time the vehicle attitude coincided with the ground-referenced velocity vector. Typical wide-angle gyro parameters assumed for the study were K_T/H equal to 360 deg/hr/ma, and H/D equal to 0.4. The gyro torque amplifiers saturate at 200 ma.

Figure 15 shows total thrust as a function of time. During motion burn, a variable I_{sp} ranging from 270 to 300 sec was programmed as a function of throttling ratio.

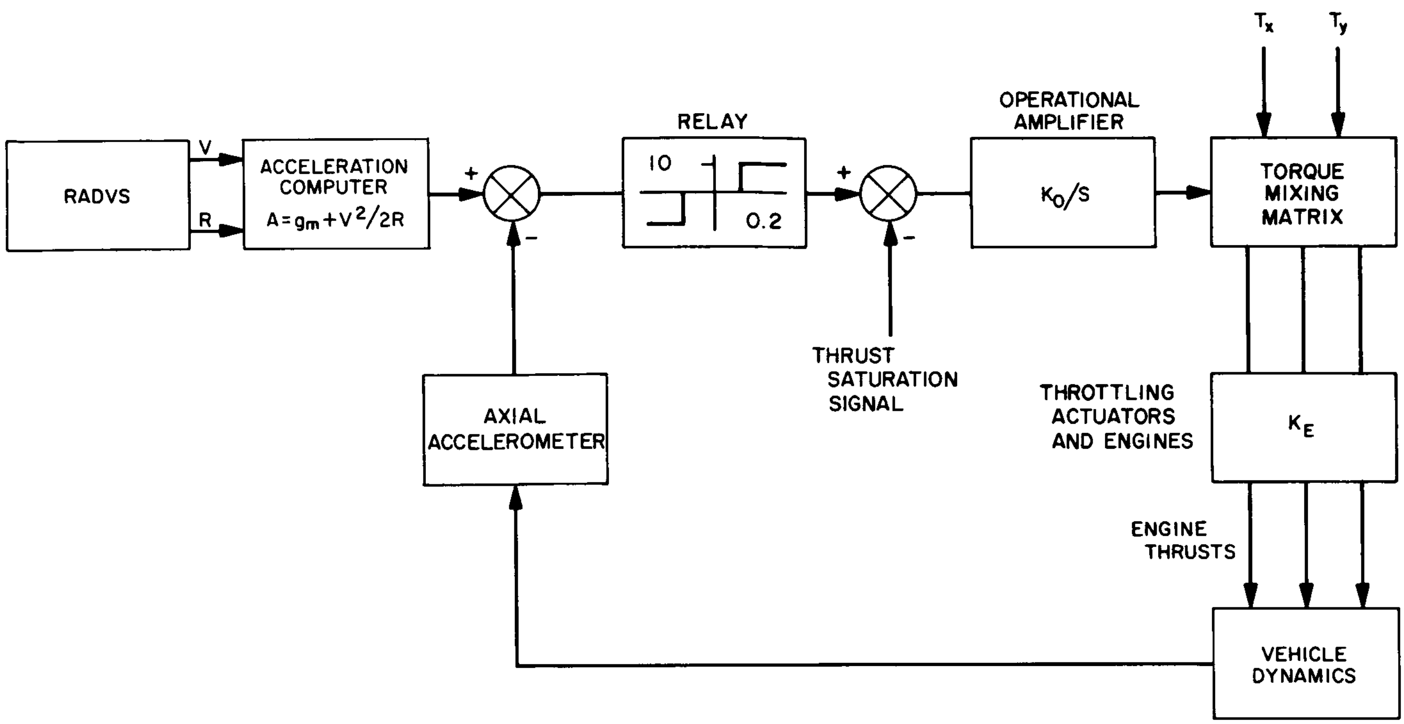


Fig. 13. Simplified gravity turn thrust control loop

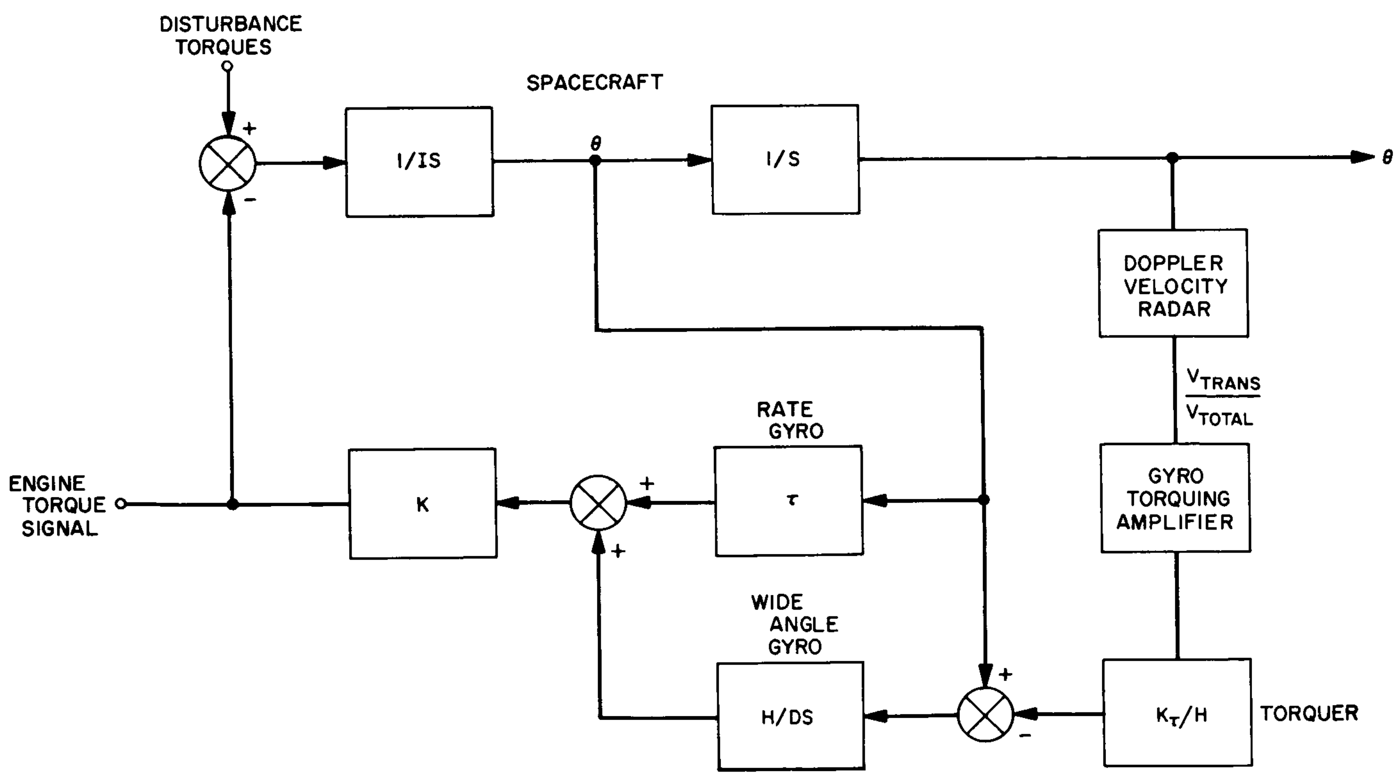


Fig. 14. Pitch and yaw control loop for gravity turn

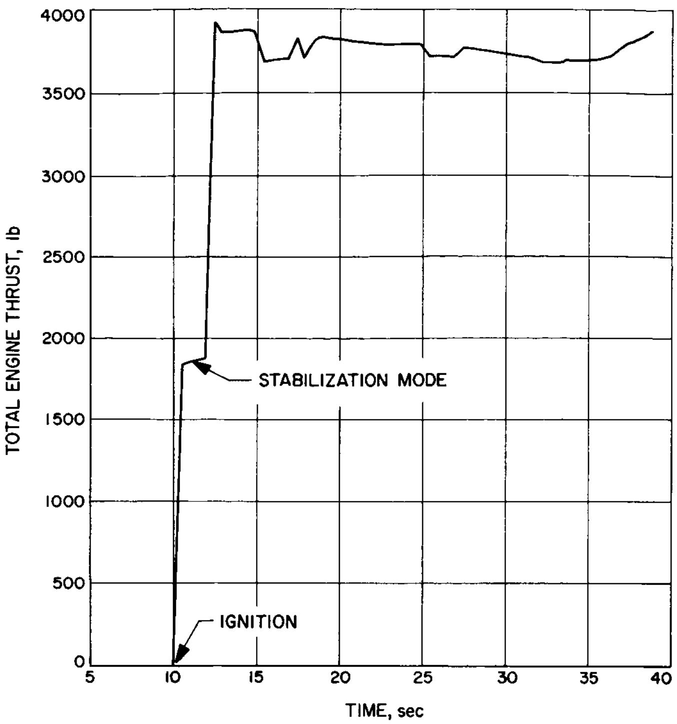


Fig. 15. Thrust profile

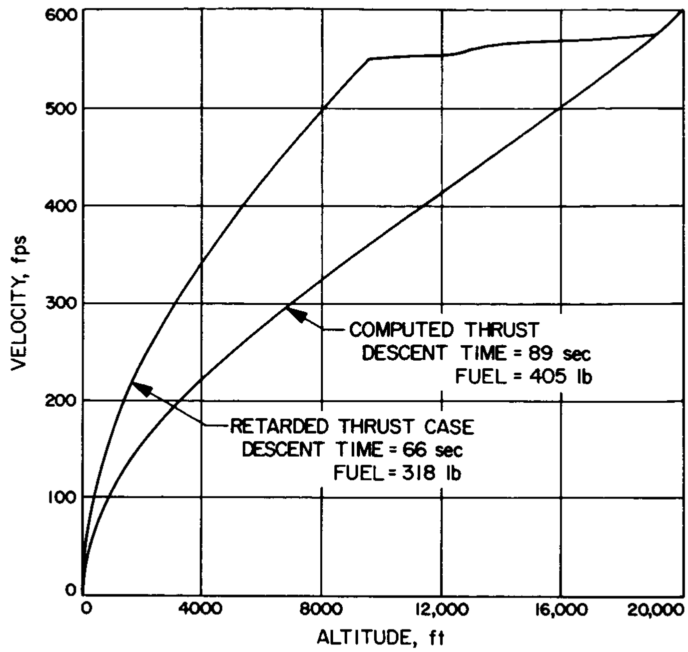


Fig. 16. AM-10 retarded thrust comparison

2. ***Retarded-thrust mode.*** The total fuel required during descent is determined by the total velocity to be removed. The total velocity is the sum of the ground-referenced velocity plus the velocity acquired due to gravity action during the propulsive descent. In dense atmosphere, the ground-referenced velocity is lower at a given slant range, but the altitude is higher because of the high flight-path angle. In thin atmosphere, the ground-referenced velocity is higher at a given slant range, but the altitude is lower because of the low flight-path angle. Therefore, although the ground-referenced velocity is less in dense atmosphere, the total velocity can be higher, raising fuel consumption. A retarded-thrust mode is introduced to avoid this situation.

The retarded-thrust mode begins after completion of the stabilization mode, and when the vehicle begins its gravity turn. If at this time the acceleration computer requests a thrust-to-weight ratio less than 2.1 (25.9 ft/sec^2), the engines are throttled to their minimum setting. Vehicle velocity will then increase or decrease to reach terminal velocity.

The system remains in this mode until the acceleration computer requests a thrust-to-weight ratio of at least 2.1. The acceleration computer output is then switched into the loop to provide thrust control. During the retarded-thrust mode, the thrust-control loop is identical to that used in the stabilization mode, except that the constant acceleration level commanded is minimum rather than nominal.

The velocity vs altitude curves for retarded thrust and those for a computed thrust in the heaviest (AM-10) atmosphere are compared in Fig. 16. As shown in the curves, the retarded thrust mode is used from stabilization to 9500 ft, where the computed-thrust mode is switched in. Figure 16 also indicates flight times and fuel consumed for these cases. As a second example, consider the lightest atmosphere (AM-4) in which the retarded-thrust mode would be used (i.e., the thrust-to-weight ratio initially commanded would be less than 2.1). The retarded-thrust mode begins at stabilization and ends at a slant range of 14,000 ft (Fig. 17). It is readily apparent from examination of Fig. 16 and 17 that considerable fuel can be saved by using this mode.

D. Terminal Modes

There are two candidate terminal modes that can be used after completion of the gravity turn mode at 30 ft. In both terminal modes, the vehicle attitude is inertially held by removing the gyro torquing signals at the end of the gravity turn. This retains the nearly vertical attitude alignment resulting from the gravity turn. The attitude control loop for both modes is identical to that shown in Fig. 11. The two modes differ, however, in the

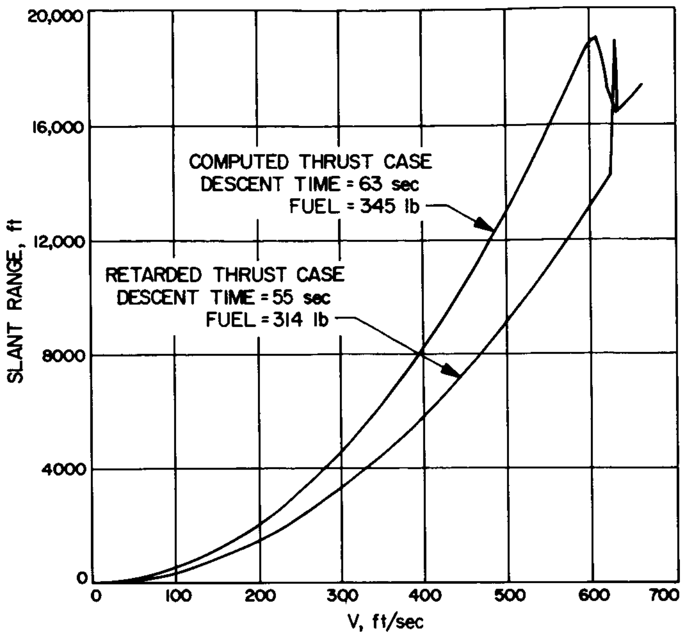


Fig. 17. AM-4 retarded thrust comparison

method used for axial thrust control. One mode commands a constant velocity descent of 5 ft/sec starting at an altitude of 30 ft. The thrust-control loop is shown in Fig. 18. This loop compares the radar-derived velocity with the 5 ft/sec commanded value and the error used to drive the standard acceleration loop. The other mode commands a constant acceleration level of less than one Martian *g* with a loop identical to the stabilization loop shown in Fig. 11. Before this mode is initiated, an axial velocity of less than 5 ft/sec must be achieved. This mode may produce slightly increased touchdown velocities, but it has the advantage of being independent of the radar signal after the 5 ft/sec axial velocity is achieved. In both modes, engine shutdown occurs at an altitude of 10 ft and the vehicle free-falls to the surface.

IV. Problem Areas

A. Pitch Turns

The required alignment of the capsule roll axis with the ground-referenced velocity vector during the gravity turn mode presents a problem: the resultant angles may be too shallow to permit acquisition of the surface by at least three radar beams. A pitch turn mode to circumvent the problem of shallow path angles is being considered. If the vehicle attitude angle is shallow enough to cause loss of radar lock, a pitch turn is initiated to orient the vehicle in a more horizontal attitude.

The direction of the initial pitch turn is derived from the radar beams by causing the vehicle to rotate in the

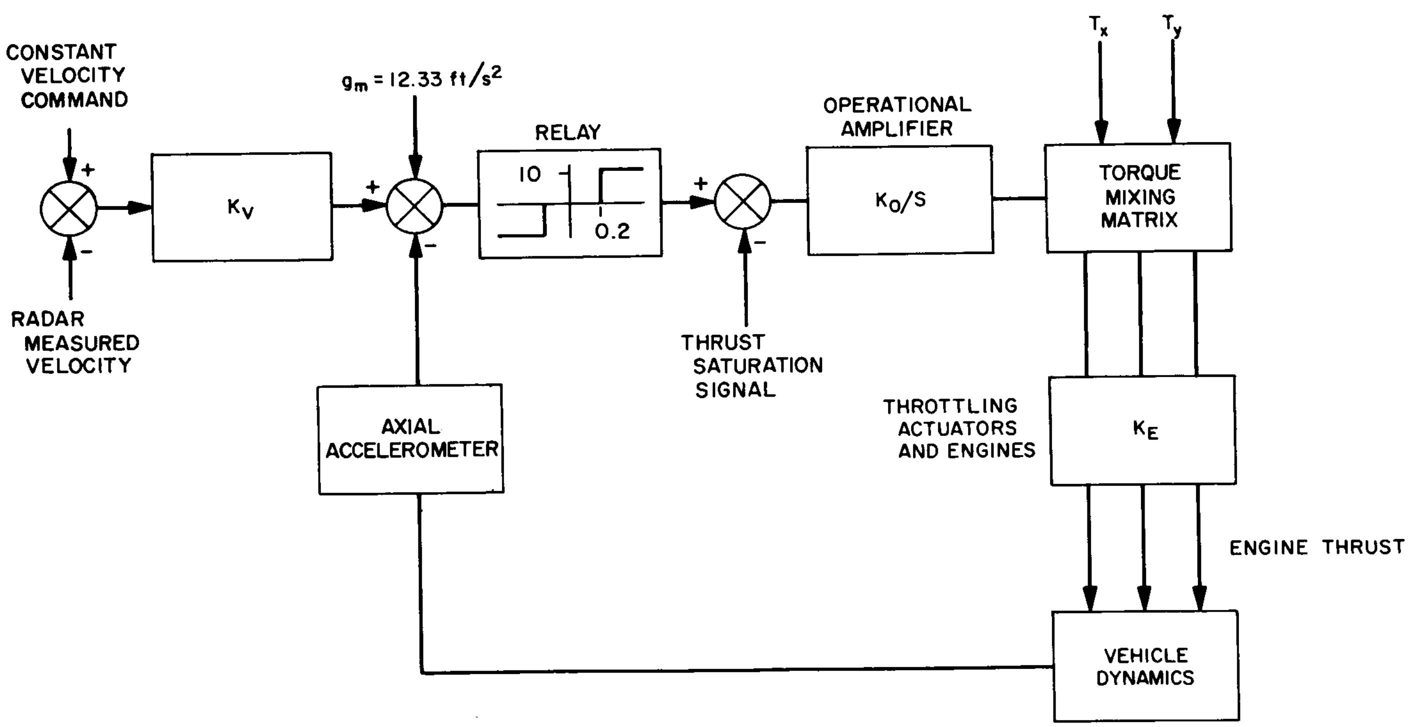


Fig. 18. Constant velocity loop for terminal mode

direction of the unacquired beams. The attitude loop for the pitch turn mode is shown in Fig. 19. The turn is performed by applying a command current to the pitch gyro torquer coil. Except for the startup and shutdown transients, this causes the vehicle to rotate at a constant angular rate. The time duration for which the command current is applied determines the magnitude of the pitch turn. The vehicle attitude angle (angle between the vehicle roll axis and the local horizon) and ground referenced path angle for a pitch turn in the AM-8 atmosphere is shown in Fig. 20.

This pitch-turn maneuver raises the thrust vector above the ground-referenced velocity vector. A subsequent command for high-thrust level causes the ground-referenced velocity vector to rotate towards the local vertical. After a high-thrust burn of predetermined duration, the vehicle is turned in pitch toward the local vertical until three beams acquire the surface. Assuming the radar beams are oriented 20 deg from the vehicle roll axis, and that a surface incidence angle of at least 25 deg is required for beam acquisition, a minimum attitude angle of 45 deg will be required for three-beam radar acquisition (Fig. 21). Referring to Fig. 20, the vehicle attitude angle at the end of the stabilization mode is 36 deg, and the ground-referenced path angle is 27 deg.

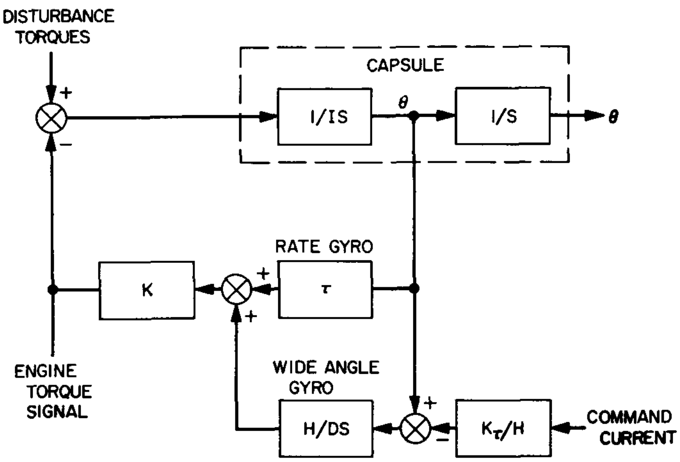


Fig. 19. Pitch turn loop

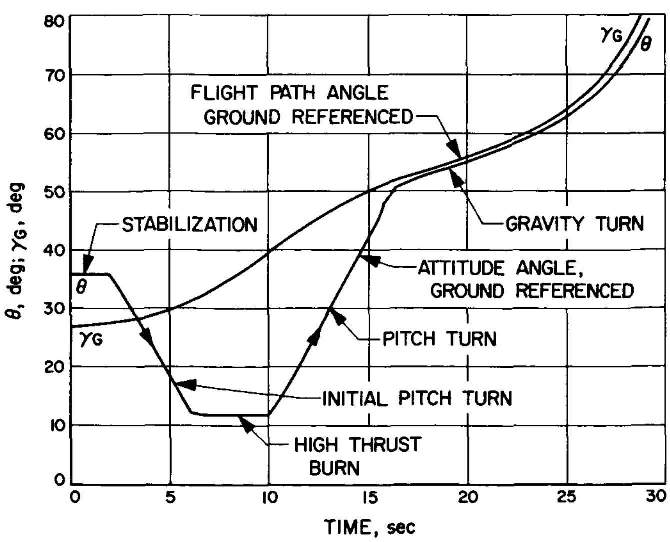
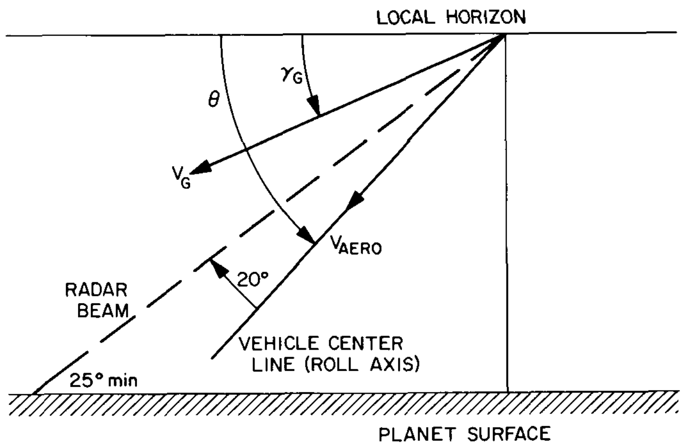


Fig. 20. AM-8 pitch turn mode



γ_G = PATH ANGLE FOR GROUND-REFERENCED VELOCITY (GROUND-REFERENCED PATH ANGLE)

θ = VEHICLE ATTITUDE ANGLE (NOMINALLY THE SAME AS THE AERODYNAMIC PATH ANGLE AT THE END OF THE STABILIZATION MODE)

V_G = VEHICLE GROUND-REFERENCED VELOCITY

V_{AERO} = VEHICLE AERODYNAMIC VELOCITY FOR RADAR ACQUISITION $\gamma_A \geq 45°$

Fig. 21. Radar reference angles for beam

Because the attitude angle is less than the required 45 deg, the radar will not be completely acquired. For complete acquisition, the system is switched into the pitch-turn mode described above, and the vehicle is rotated toward the horizontal at a commanded turn rate of 6 deg/sec for 4 sec. Note that at 3.4 sec the thrust vector has gone "above" the ground-referenced velocity vector, and that the latter is beginning to rotate toward the local vertical. At the completion of this pitch turn, the command current is removed and the vehicle remains in inertial hold while the engines burn for 4 sec at maximum thrust. After completion of the high-thrust burn, the pitch turn is reversed and the vehicle is rotated back towards the local vertical until radar acquisition occurs. With the thrust vector sufficiently rotated to permit complete radar acquisition, the system automatically goes into a gravity turn. Note that at this time (16 sec), the ground-referenced velocity vector has been rotated past the 45 deg necessary to ensure radar acquisition (Fig. 20). A flight log for a two-axis turn showing simultaneous turns in pitch and yaw is shown in Fig. 22.

B. Pluming Effects

Changes in the vehicle aerodynamic coefficients caused by exhaust plumes can create a difficult control problem. Because aerodynamic torque is proportional to the normal aerodynamic force coefficient (C_n), the attitude error during the gravity turn mode can become large if this coefficient does not reduce during motor burn as expected. Increasing the control loop gain can only compensate for a limited increase in (C_n) because the high torques will cause high angular rates that result in gyro saturation. This effect can be quite strong near touchdown where the capsule is at a nearly 90-deg angle-of-attack and (C_n) is maximum.

The degree to which the drag coefficient decreases is also important. If the reduction is large, it can cause instability in the thrust loop; if it is small, it makes the retarded-thrust mode less efficient by causing velocity reduction even at minimum thrust settings. In AM-10, for example, the terminal velocity is reached approximately at ignition. This means that the specific drag is one Martian g. If after ignition the drag drops to 25% and a ¾-g acceleration is commanded in the retarded mode, the velocity will remain constant. If, however, the drag does not drop sufficiently, an undesirable decrease in vehicle velocity will occur.

FLIGHT LOG

TIME	ALTITUDE	EVENT DESCRIPTION
8.391	10999.1	STABILIZATION BURN
8.410	10990.3	BEAM 1 LOCK
8.410	10990.3	BEAM 4 LOCK
10.389	10065.8	COMMANDED TURN* PITCH - YAW -
13.014	8890.8	LOCK LOST ON 1
14.320	8321.5	LOCK LOST ON 4
14.390	8290.7	HIGH THRUST BURN
19.391	5990.6	COMMANDED TURN* PITCH + YAW +
19.898	5745.0	BEAM 4 LOCK
21.172	5124.2	BEAM 1 LOCK
25.363	3192.9	BEAM 3 LOCK
25.363	3192.9	GRAVITY TURN INITIATED
26.145	2874.4	BEAM 2 LOCK
39.859	30.0	CONSTANT VELOCITY MODE
41.721	12.0	ENGINE CUTOFF

Fig. 22. Flight log for 2-deg-of-freedom pitch turn mode

For the purpose of this study, it was assumed that the plume effect reduces the aerodynamic force coefficients by a factor of 4. As further testing of the aerodynamic model continues, a more accurate evaluation of the plume effects can be made.

V. Control Loop Description

A. Thrust Control Loop

Operation of the thrust control system, which is shown in detail in Fig. 23, follows. When the acceleration error exceeds the relay deadspace, the thrust is increased at a linear rate corresponding to 7000 lb/sec. To prevent the actuators from hitting their stops and causing loss of throttling capability, the actuator position is fed back through a deadspace to the thrust integrator. Thus, when the actuators approach their mechanical limits, the nonlinear feedback prevents them from actually reaching their stops. Note that this provides a thrust signal override to permit throttling for the generation of attitude stabilizing torques, if required.

Because of the nonlinear nature of this thrust control loop, the parameters were initially chosen by simulating the actual system on a digital computer. A separate digital program is currently being developed for the purposes of loop stability evaluation and optimization.

B. Attitude Control Loops

The basic attitude control loop evolved from a configuration using wide-angle gyros followed by double lead-lag networks for rate damping. A maximum dc loop gain

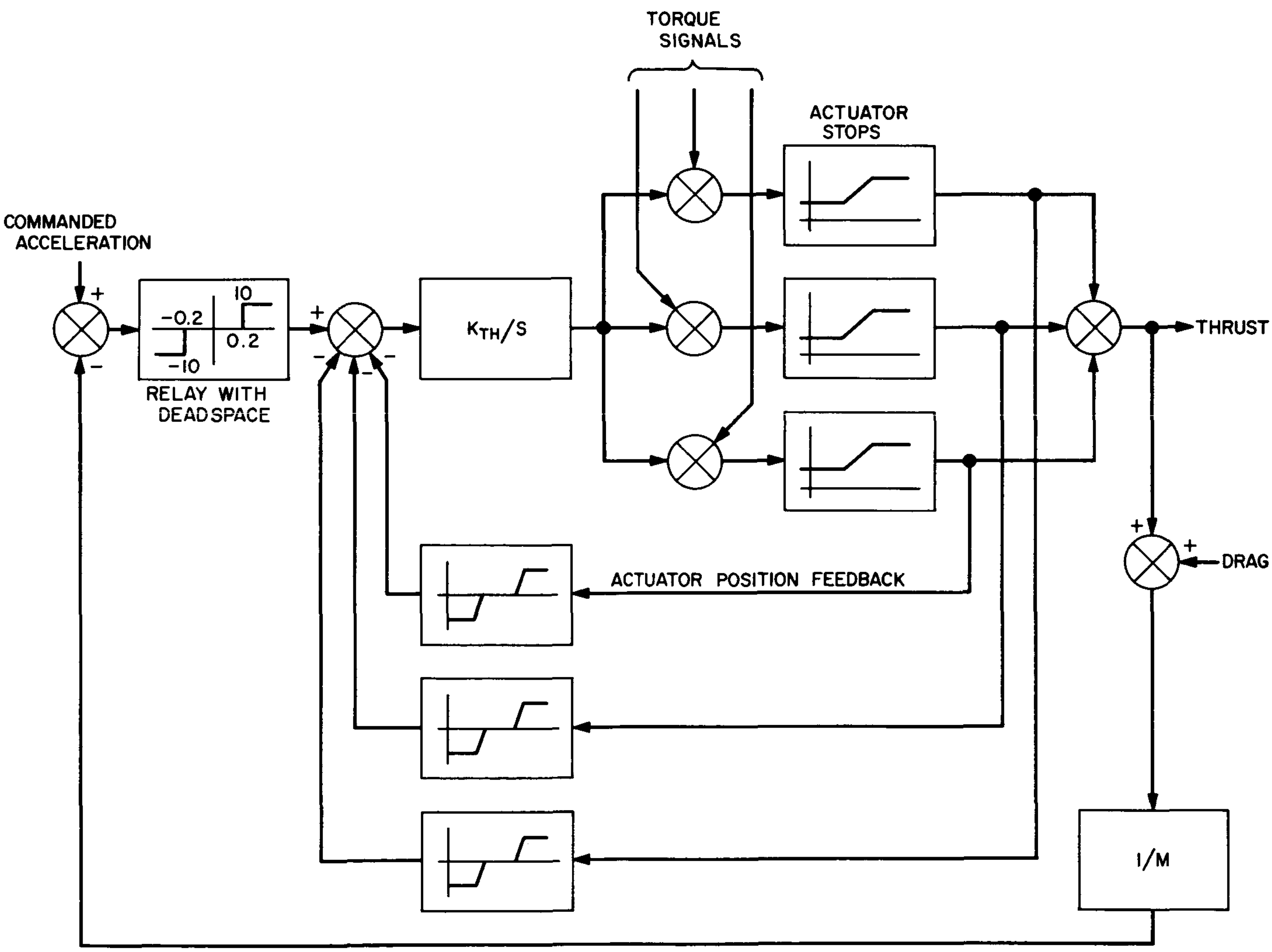


Fig. 23. Thrust control system

of 10 was found feasible for this configuration. For this gain setting, however, attitude errors during the stabilization mode were on the order of 10 deg. A decision was made to use rate gyros to compensate the loop. This configuration is easily stabilized and permits the use of much larger loop gains. A block diagram of the attitude loop in

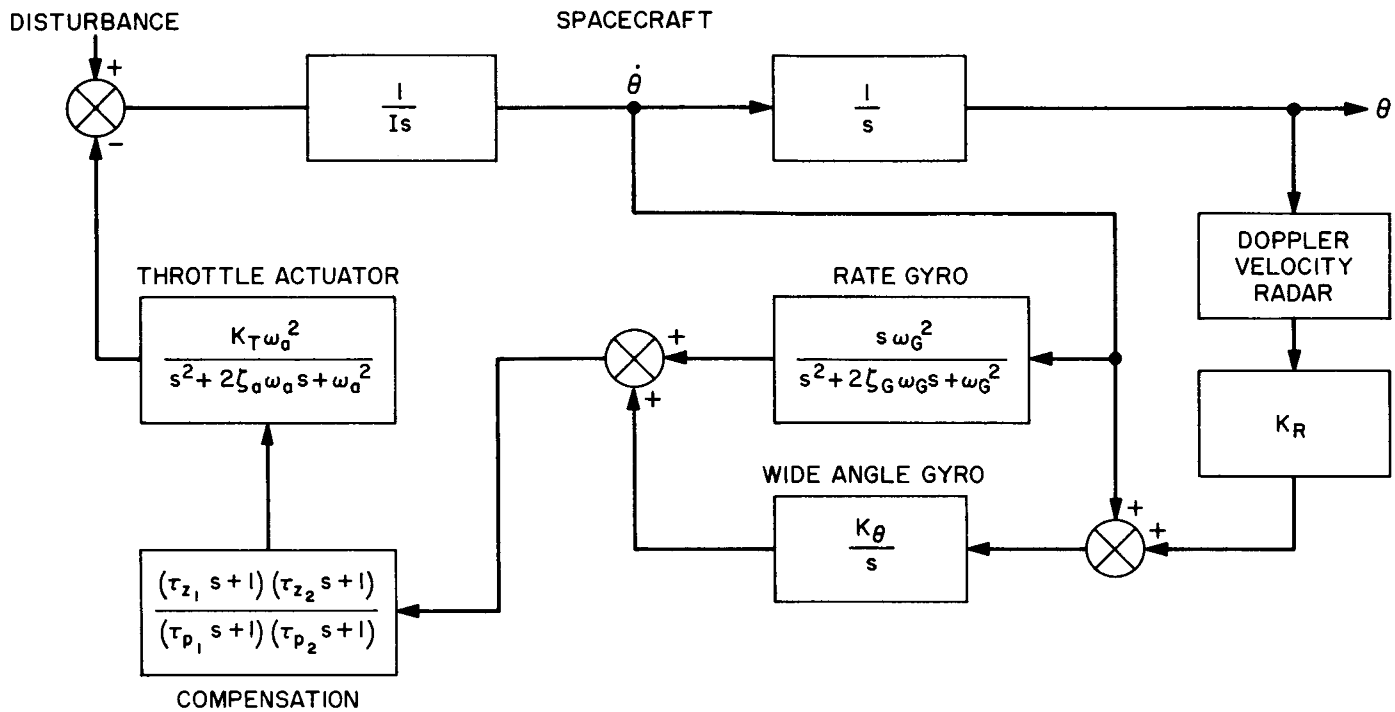


Fig. 24. Attitude-control with dynamics

Table 3. Simulation flight descriptions

AM	Control system	Descent time, sec	Final vertical velocity, ft/sec	Final horizontal velocity, ft/sec	Final WRTH attitude, deg	Fuel, lb
4	Gravity turn	63	15	0.2	91.5	346
4	Gravity turn and retarded thrust	55	14	0.2	91.8	314
4	Pitch turn	59	15	0.2	91.3	370
7	Gravity turn	42	14	10	82	404
7	Pitch turn	52	12	7	87	418
8	Gravity turn	37	13	9	80	345
8	Pitch turn	34	15	8	82	354
10	Gravity turn	89	14	2	96	404
10	Gravity turn and retarded thrust	66	15	2	92	318

Table 4. Powered descent control system weight and power estimates

Power descent control elements	Weight, lb	Input power average, w	Input power peak, w
3 wide-angle gyros with electronics and heaters	A/C[a]	20	42.5
1 axial accelerometer with electronics (no heater)	1	5	5
3 rate gyros with electronics	A/C[a]	10.5	10.5
1 thrust-level estimation and logic-control computer	13.0	70	70
1 autopilot and coordinate conversion electronics	10.0	10	35
3 roll-control jet vane actuators	9.0	9	11
wiring harness	0.5		
Total	33.5	124.5	174

[a]Weight is allocated to pre-entry control system.

the gravity turn mode that includes gyro and actuator dynamics, and the compensation networks, is shown in Fig. 24. The open-loop transfer function is a fifth-order numerator over a ninth-order denominator, as follows:

$$\frac{F}{\epsilon} = \frac{K_T\,(As^4 + Bs^3 + Cs^2 + Ds + E)\,(\tau_{Z_2}\,s + 1)}{s^3\,(\tau_{p_1}\,s + 1)\,(\tau_{p_2}\,s + 1)\,[s^2/\omega_G{}^2 + (2\zeta_G/\omega_G)\,s + 1]\,[s^2/\omega_a{}^2 + (2\zeta\,a/\omega_a)\,s + 1]\,I}$$

where

$$A = K_\theta\,\tau_{Z_1}$$

$$B = K_\theta\,(\tau_{Z_1}\,K_R + 2\zeta_G\,\omega_G\,\tau_{Z_1} + 1) + \omega_G{}^2\,\tau_{Z_1}$$

$$C = K_\theta\,(\tau_{Z_1}\,\omega_G{}^2 + 2\zeta_G\,\omega_G K_R \tau_{Z_1} + 2\zeta_G\,\omega_G + K_R) + \omega_G{}^2$$

$$D = K_\theta\,(\tau_{Z_1} K_R\,\omega_G{}^2 + \omega_G{}^2 + 2\zeta_G\,\omega_G\,K_R)$$

$$E = K_\theta\,K_R \omega_G{}^2$$

Note: K_R is the gain from the radar signal to the wide-angle gyro torquer, and is a function of axial velocity.

VI. Conclusions

Table 3 summarizes the simulated flight data obtained from the computer programs. Descent times, final velocity, final attitude and fuel consumption are shown for each case.

The basic control system equipment required, along with weight and power estimates for the powered descent phase of the mission, is given in Table 4.

From the results of the simulation studies of the system described above, it appears that this control system configuration can perform a successful planetary descent over the anticipated range of initial conditions.

Nomenclature

A	commanded axial acceleration
A	capsule area
a	angle of attack
A/C	attitude control
AM	atmosphere model number
C_d	drag force coefficient
c.g.	center of gravity
C_n	normal aerodynamic force coefficient
D	wide-angle gyro damping coefficient
F_X	aerodynamic force along body X axis
F_Y	aerodynamic force along body Y axis
F_Z	aerodynamic force along body Z axis
g_m	Martian gravity
γ_A	aerodynamic path angle
γ_G	ground-referenced path angle
H	wide-angle gyro angular momentum
I	capsule moment of inertia
I_{SP}	propellant specific impulse
K	generalized gain block
K_E	thrust loop engine actuator gain
K_R	radar transverse velocity gain
K_T	gyro to throttle actuator gain
K_τ	wide-angle gyro torquer scale factor
K_θ	torque loop position to rate gain
K_{TH}	overall thrust control loop gain
K_V	constant velocity loop gain
M	capsule mass
ω_a	throttle actuator natural frequency
ω_G	rate gyro natural frequency
R	slant range
RADVS	range and doppler velocity system
τ_P	pole time constants in torque loop compensator
τ_Z	zero time constants in torque loop compensator
θ	vehicle attitude angle

Nomenclature (contd)

V radar measured slant velocity

V_{AERO} vehicle aerodynamic velocity

V_G vehicle ground-referenced velocity

V_{TRANS} body measured transverse velocity

V_X inertial velocity along X axis

V_Y inertial velocity along Y axis

WRTH with respect to horizontal

X out-of-plane distance

Y down-range distance

Z altitude

$\ddot{z}$ body measured axial velocity (along roll axis)

ζ_A throttle actuator damping ratio

ζ_G rate gyro damping ratio

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