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SUMMARY

The feasibility of manually guiding the upper stages of a large launch vehicle by means of a minimum of on-board equipment has been investigated. A fixed cockpit analog simulation of the rigid body dynamics of the upper stages of the Saturn V (S-II and S-IVB) lunar mission vehicle was used. Five guidance schemes were studied and the results indicate that with manual guidance of the vehicle into earth orbit the injection errors can be less than ±1000 m in altitude, ±3 m/s in velocity, and ±0.1° in flight-path angle.

INTRODUCTION

There has been considerable speculation that pilot participation in the guidance and control of the Saturn V launch vehicle could increase the probability of overall mission success. Some study of piloted control of large launch vehicles has already been completed, most notably for the Titan III (refs. 1-4), and the somewhat earlier investigation by Holleman, Armstrong, and Andrews (ref. 5). These investigations have shown that the problem of controlling launch vehicles may be broken into two separate phases. The first is the atmospheric flight phase for which the guidance system requirements are relatively insignificant. The primary problems here are attitude stabilization and aerodynamic load reduction of a large, flexible, and usually aerodynamically unstable vehicle. These problems are further complicated by wind disturbances and in some instances by propellant-sloshing dynamics. During the second phase of flight, outside the sensible atmosphere, the problem areas reverse. Since the vehicle is usually much stiffer (second and third stages of flight) and there are no atmospheric disturbances, the attitude stabilization task is relieved. However, during this second phase of flight, the vehicle must be guided precisely into an earth orbit. The orbit injection requirement makes guidance the primary problem during the second or guidance phase of flight.

These previous investigations have demonstrated the feasibility of manually controlling both phases of flight for particular vehicles and particular mission profiles. While much can be learned from these studies, the large differences of the Saturn V vehicle and mission objectives make it difficult to extrapolate the results. Consequently, the Marshall Space Flight Center (MSFC) and Ames Research Center (ARC) conducted a joint feasibility study to determine the possible role of the pilot in the guidance and control of the Saturn V launch vehicle during both phases of flight, from lift-off through earth orbit injection. Reference 6 presented the results obtained during the study of the atmospheric flight phase. The present report deals with the study results obtained for the second or guidance phase of flight.

Large-booster "automatic" guidance systems typically use a fairly sophisticated on-board computer to guide the vehicle to a sub optimal trajectory. Thus complex guidance equations, such as the path adaptive, iterative guidance scheme, are used for the upper stages of Saturn V for precise orbital injection. A full description of the primary guidance system for Saturn V can be found in reference 7. While it may be possible to incorporate the pilot in such a primary guidance system, it is felt that the pilot's greatest potential contribution to mission success would be in an independent backup guidance and control system. If the primary guidance system in the launch vehicle should fail, this backup capability could (1) allow time for reactivating the primary system or switching to a secondary system (such as the spacecraft on-board computer), (2) provide alternate earth orbit mission capability, or (3) provide a large selection of abort sites.

The objective of the study, then, was to examine some simple manual guidance schemes which could be used as a backup system for flying a large multistage launch vehicle into a near circular earth orbit at 185 km altitude. Both of the previously mentioned manual boost guidance investigations (refs. 4 and 5) used a "guide to a nominal trajectory" technique with a stored nominal pitch program and examined various display combinations of altitude, altitude rate, angle of attack, velocity and/or flight-path angle, and a ground voice command system. Similar techniques were used in this study. Guidance parameters, altitude and altitude rate, plus voice communications were assumed to be available on board the launch vehicle. Five simple manual guidance schemes which used these parameters were proposed and evaluated.

Since the maximum accelerations for the upper stages of Saturn V are quite low (2 g at S-II burnout), a fixed cockpit simulation was considered adequate. The first stage study (ref. 6) showed little difference between fixed-cockpit and centrifuge data for a burnout thrust-to-weight ratio of 4.7. Also, during the latter portion of first stage flight, when dynamic pressure drops near zero, there was little effect on vehicle control due to bending or sloshing. Therefore, only rigid body dynamics were included for the upper stages guidance study.

NOTATION

a	gain coefficient in rate loop, s
b	gain coefficient in pilot command loop

c.g.	center of gravity
F_β, F_φ M_β, M_β'	time varying coefficients in the equations of motion
g	earth's gravitational acceleration, m/s^2
h	altitude, m
$\dot{h}$	altitude rate, m/s
Δh	altitude error, m
$\Delta\dot{h}$	altitude rate error, m/s
I	moment of inertia, $kg\text{-}m^2$
k	actual to nominal thrust ratio
K_1, K_2	pilot weighting gains
m	mass of the vehicle, kg
p	Laplace operator, 1/s
q	dynamic pressure, N/m^2
R	downrange distance, nautical miles
r	radius vector from earth center to c.g., m
r_β	roll thrust lever arm, m
T	total thrust of engines, N
T'	swiveled thrust, N
T^*	thrust about roll axis, N
t	mission time, s
V	nominal vehicle velocity, m/s
v_i	ith component of vehicle perturbation velocity, m/s
x_i	ith component of vehicle c.g. location with respect to nominal location, m
y_β	distance from c.g. to engine gimbal pivot, m

α	angle of attack, rad
β_i	ith component of engine gimbal angle, rad
γ	flight-path angle, rad
ΔV_{co}	vehicle velocity minus desired orbital velocity, m/s
$\Delta\theta_{3e}$	orbital range angle error, rad
$\Delta\varphi_i$	ith component of attitude angle error, rad
δ	pilot controller input, rad
θ_{3E}	nominal orbit range angle from space fixed launch point, rad
φ_i	ith component of total attitude angle with respect to a space fixed coordinate system, rad
χ_i	ith component of nominal vehicle attitude with respect to a space fixed coordinate system, rad
ω_i	ith component of attitude angle rate, rad/s

Subscripts

bo	burnout condition
co	cutoff condition
c	command
lateral	perpendicular to nominal trajectory plane
n	nominal flight-path conditions
o	sea-level value
s	space fixed reference

Axis Systems

P,Y,R	pitch, yaw, and effective roll of engines about engine gimbal point
1B,2B,3B	nominal body axes (fig. 18)
1b,2b,3b	body axes (fig. 18)

1E,2E,3E	nominal earth axes (fig. 18)
1e,2e,3e	earth axes (fig. 18)
1p,2p,3p	inertial platform space fixed axes (fig. 18)

DESCRIPTION OF VEHICLE AND MISSION

Vehicle Description

The Saturn V launch vehicle, as defined for the Apollo lunar landing mission, was simulated for this study. This vehicle (fig. 1) consists of three booster stages and the Apollo lunar mission payload. Its overall length is 110 m; its maximum diameter is 10 m (not including fins); and it has three liquid fueled stages. This investigation involved only the two upper stages, the S-II and the S-IVB. The second stage, the S-II, has five J-2 engines with a total thrust of 4.5×10^6 N. Attitude control and guidance maneuvers are accomplished by swiveling the four outer engines. The third stage, S-IVB, is powered by a single J-2 engine, which swivels for pitch and yaw control. Roll control is achieved by an auxiliary propulsion system which is also used during coasting periods. This auxiliary system is composed of six body-fixed control nozzles mounted on the stage structure (two for pitch, four for yaw and roll).

Trajectory Description

The trajectory parameters for a typical launch profile are shown in figure 2. This investigation is concerned with the trajectory from first stage burnout through S-II burn and staging, and first burn of the S-IVB to circular orbit insertion at an altitude of 185 km. The second stage, moving at 2700 m/s, ignites at an altitude of approximately 63 km, and burns for about 400 seconds with a maximum thrust-to-weight ratio of 2.2 and a velocity of 6700 m/s. The first burn of the third stage starts at an altitude of 184 km, and burns for about 160 seconds giving an orbital velocity of 7800 m/s with an acceleration at cutoff of 0.7 g.

LAUNCH VEHICLE GUIDANCE AND CONTROL

Primary Guidance and Control System

The Saturn V launch vehicle has a single inertial navigation and guidance system for all stages, located in the Instrument Unit (IU) on top of the S-IVB. This system is independent of the one contained in the Apollo spacecraft. The main components of the inertial navigation, guidance, and control system are the stabilized platform, the digital guidance computer and data adapter, and the control computer. The platform serves as an inertial reference frame for attitude control and acceleration measurements. The digital guidance computer performs navigation and guidance computations and generates control commands, including discrete sequencing signals (engine cut-off, stage separation, etc.). During flight, information can be inserted into the guidance computer from ground stations through a radio command link. The control computer accepts signals from the guidance computer and rate gyros to generate the proper actuator-engine control commands.

The launch vehicle closed loop guidance starts after the second stage ignites. The system uses a path adaptive guidance scheme to steer the vehicle on a "minimum propellant" trajectory to the aim conditions for orbital insertion, stored in the guidance computer. More complete details of the automatic guidance and control system and components for the Saturn V can be found in reference 7.

Vehicle Manual Control Dynamics

The vehicle dynamics system assumed in this rigid body simulation study was a rate augmented system with manual attitude control. The system transfer function

$$\frac{\varphi(p)}{\delta(p)} = \frac{bM_\beta}{p(p + aM_\beta)}$$

is derived from the equations of appendix A and relates the vehicle attitude and the pilot controller output. The rate feedback gain used in the pitch and yaw channels ($a = 0.75$) resulted in a system dynamic response time constant of 3 to 0.5 second during second stage flight and 2 seconds during the third stage. Gain in the roll channel rate feedback ($a = 0.1$) gave a time constant of 1 second for both stages. The controller gain used was $b \cong 0.2$ in pitch and yaw and 0.1 in roll. The maximum controller output was scaled to give $\pm 3^\circ$ of engine deflection in pitch or yaw and $1\text{-}1/2^\circ$ in roll. A unity gain transfer function was assumed for the engine actuator combination.

Manual Guidance Schemes

The objective of this study was to examine some simple manual guidance schemes that could be used as a backup system to guide the upper stages of a Saturn V type vehicle into a nearly circular earth orbit. These manual guidance schemes, the primary guidance system, and the vehicle system and sensors are illustrated in figure 3. The conditions assumed for the backup mode were: (a) that the guidance computer and data adapter of the primary system had failed before second stage ignition (the manner of failure, effect on vehicle control, method of failure detection, and switch over techniques were not included in this investigation), (b) that data were available directly from the space fixed platform instrumentation, including platform gimbal angles and space fixed velocity components obtained from integrating accelerometers on the platform, (c) that attitude rates were available from control rate gyros, and (d) that altitude and altitude rate were available (from on on-board radar altimeter). It should be noted that a radar altimeter has been successfully flight tested on some early Saturn I flights.

The manual guidance schemes used in this study drew upon the experience of the investigations of references 4 and 5, both of which used guidance to a "nominal trajectory" (i.e., a stored nominal pitch versus time program) and examined various display combinations of altitude, altitude rate, angle of attack, velocity, and flight-path angle. Ground voice command techniques were also studied. Reference 5 recommended using velocity versus altitude displayed on an oscilloscope containing a nominal trajectory reference curve, plus a nominal pitch program display. The boost glide study (ref. 4), which had a one earth orbit type of mission, recommended using nominal or command displays of pitch attitude, altitude, and altitude rate as functions of time.

Based on these schemes and the guidance and control information assumed to be available from the launch vehicle, five simple pitch plane guidance schemes were proposed and evaluated. The methods and the type of instrumentation used are outlined in table I. A signal flow diagram of the pitch plane guidance and control systems is shown in figure 3. A description of each method follows:

1. GCB method

   On-board guidance information was assumed to be unavailable. Ground controllers would then command pilot inputs based on ground tracking information in a manner similar to a "no-gyro" GCA aircraft approach. The pilot was commanded (at approximately 15-second intervals) to pitch and yaw at constant predetermined rates ($1^{\circ}/s$) to null ground observed trajectory deviations. For this scheme there was no pitch or yaw attitude information so that the operation of a minimum instrumentation guidance system without additional cues could be evaluated. Roll attitude information was available.

2. $\Delta h, \Delta \dot{h}$ error display method

   The guidance display for this scheme consisted of altitude minus nominal altitude (Δh), and altitude rate minus nominal altitude rate ($\Delta \dot{h}$), on the flight director needles. In addition, the pilot was presented attitude, attitude rates, time, nominal pitch program versus time around the clock, and lateral velocity. The pilot's task was to fly the nominal pitch program while maintaining null values of Δh and $\Delta \dot{h}$.

   A computer would be required for this scheme since the reference trajectory altitude and altitude rate would have to be stored and the errors Δh and $\Delta \dot{h}$ computed. This computer could be an on-board system or a ground based computer plus telemetry system.

3. h versus $\dot{h}$ reference plus $h - h_{co}$, $\dot{h} - \dot{h}_{co}$ method

With this method, the initial pilot's task was to fly the vehicle so that actual altitude and altitude rate coincided on an x-y display with the nominal reference curve. During the terminal portion (i.e., earth orbit injection) of the third stage burn, cutoff meters presented altitude minus the desired altitude at cutoff ($h - h_{co}$), and altitude rate minus the desired rate of cutoff ($\dot{h} - \dot{h}_{co}$). The desired altitude rate at cutoff is zero ($\dot{h}_{co} = 0$) for a circular orbit. In the latter phase of the launch the pilot's task was to fly to null the altitude cutoff meter and then maintain zero altitude rate until velocity cutoff. This method gave a high degree of resolution for the important terminal or injection phase of the flight. A simple on-board computation would be needed to subtract the preset cutoff altitude (a constant) from the actual altitude. The value of $h - h_{co}$ was presented on the vertical flight director needle and $\dot{h} - \dot{h}_{co}$ was presented on the horizontal flight director needle. The altitude cutoff meter was used during the last 1200 m of altitude change and the altitude rate meter was activated for altitude rates of less than 60 m/s. The additional items displayed were attitude, attitude rate, nominal pitch vs. time on the clock card, time, and lateral velocity.

4. h versus $\dot{h}$ reference method

As in the previous method, the pilot's task was to fly the vehicle so that the actual altitude and altitude rate coincided with the nominal reference trajectory altitude vs. altitude rate curve on the x-y display. No extra resolution was provided for cutoff conditions. Additional pilot displays were attitude, attitude rate, time, pitch program on the clock card, and lateral velocity.

5. t versus h reference method plus $h - h_{co}$

This method required the pilot to fly the vehicle so that the actual altitude and mission time coincided on an x-y presentation with the nominal reference trajectory altitude vs. time curve. For better resolution during the last moments of burn for earth orbit injection, altitude minus desired altitude at cutoff ($h - h_{co}$) was displayed on the horizontal flight director needle. As in method 3, a simple on-board computation would be required to generate $h - h_{co}$. The altitude cutoff meter was scaled for use during the last 1200 m of altitude change. The additional displays were the same as for method 3.

The yaw plane guidance and control and roll attitude control system for all of the aforementioned manual pitch guidance schemes consisted of a rate augmented vehicle with pilot inputs for attitude command. Except for the GCB method, the pilot's displays included attitude, attitude rate, and lateral velocity. Lateral displacement was assumed to be unavailable since the primary guidance system had failed. The lateral integrating accelerometer on the inertial platform was assumed to be alined perpendicular to the boost trajectory plane and thereby to provide a direct readout of lateral velocity. The pilot's task was to maintain zero values of roll attitude and lateral velocity.

In all of the guidance methods the pilot shut down the third stage in one of two ways: (a) directly reading the velocity change to go on a digital display, or (b) countdown from a ground station.

MANUAL GUIDANCE SIMULATOR

The equations of motion used to simulate the booster's orientation and trajectory are presented in appendix A. Perturbation techniques were used wherein only deviations from the nominal trajectory and attitude are calculated. Second and higher order terms were assumed to be small and were not included. The equations of appendix A were mechanized on an electronic analog computer.

The fixed-base cockpit which was used is shown in figure 4. It consisted of a display panel, a three-axis, side-arm controller, and the pilot chair. Figure 5 is a close up of the side-arm controller. It is designed for a high acceleration environment as forward and aft motion of the handle give no input signal. Pitch input is obtained by up or down motion of the handle, yaw by translating the handle left or right, and roll by rotating the handle left or right. Pivot points for pitch and yaw are at the wrist. This same controller was used for the first stage study, and the controller stick force and displacement characteristics can be found in reference 6.

The display panel is shown in figure 6. In the center is the primary control instrument, the all-axes attitude indicator which incorporates two ILS type flight director needles and two displacement meters on the periphery. Other meters include the pitch, yaw, and roll rate meters and a clock with a sweep second hand. A scale around the clock circumference shows the nominal pitch program schedule. The X-Y plotter is situated in the upper portion of the panel, and a digital voltmeter readout, used for velocity cutoff, is located below. A sequence light on the panel was used to indicate S-II/S-IVB staging.

GUIDANCE AND CONTROL PERFORMANCE CRITERIA AND PROCEDURES

Performance Criteria

The pilot guidance and control system performance was evaluated on the basis of orbit insertion accuracy. As stated previously, the manual guidance task was to guide to a preselected nominal trajectory which puts the payload in a 185 km circular orbit. Some arbitrary orbit insertion window tolerances were selected for this study. Setting the actual tolerances for a particular mission would have to be based on such things as trade-offs between orbit insertion accuracy, earth orbit correction maneuvers, and initial condition requirements for translunar injection. The tolerances used for this study were:

Altitude error	±1000 m
Velocity error	±3 m/s
Flight-path-angle error	±0.1°

Procedure

In general the pilot's task was to null any initial condition errors in altitude, altitude rate, or lateral velocity and then maintain near nominal flight conditions. He was given the additional task of cutting the thrust at the desired orbital velocity. Details of the thrust cutoff technique are discussed later in the report. For a multiman mission, this task could possibly be assigned to another crew member so that the pilot could concentrate on guidance and control only.

Six subjects were used for the simulated flights - three Ames Research pilots (subjects 1, 2, and 5) and three engineers (subjects 3, 4, and 6). All simulated flights were conducted as follows: Each subject flew practice trajectories using a particular guidance scheme until he felt he had become sufficiently familiar with the scheme. He then made simulated flights during which data were collected. At the end of each flight, injection error data were read and the pilot's comments were recorded.

Since the first stage has open loop guidance (preprogrammed pitch attitude control system), large burnout trajectory dispersions can be caused by atmospheric forces, off nominal thrust, or other factors. Digital computer runs were performed at MSFC to determine the envelope of these burnout dispersions. The worst combination of burnout conditions was chosen as the initial conditions for the majority of this study. The off-nominal trajectory error values were:

Downrange error	10,000 m downrange
Vertical error	2,300 m low
Lateral error	0
Downrange velocity error	85 m/s downrange
Vertical velocity error	85 m/s toward earth
Lateral velocity error	20 m/s to the right

Actual lateral error from the MSFC dispersion envelope study was about 1600 m. However, since prime interest was in pitch plane guidance and, secondly, only lateral velocity was assumed available for display to the pilot, an initial value of zero was used. An initial lateral offset could be corrected by ground voice commands as in the "GCB" scheme.

RESULTS AND DISCUSSION

As stated, the performance of the pilot guidance and control systems was evaluated on the basis of orbit injection errors in altitude and flight-path angle in the pitch plane, flight-path angle, and lateral displacement in the yaw plane and injection velocity. The study results are presented in the following order: pitch plane guidance, yaw plane guidance, thrust termination, and off-nominal and augmentation-failure conditions.

Pitch Plane Guidance

As would be expected, the pitch plane guidance data, altitude error, and altitude rate error at cutoff strongly reflect the display resolution available to the pilot. A comparison of altitude error at cutoff for the various guidance schemes is shown in figure 7. Also shown is the display resolution in kilometers of altitude error per millimeter of instrument needle or pen deflection. The dark center area on the resolution line shows the resolution at cutoff where the $h - h_{co}$ display meter becomes active. Figure 8 presents a comparison of flight-path-angle error at cutoff for the various guidance schemes. Also shown is the flight-path-angle resolution at thrust cutoff, which is directly related to the display resolution used for altitude rate. The flight-path-angle resolution was determined by dividing the altitude rate indicator sensitivity, in meters per second per millimeter of needle deflection, by the desired velocity at cutoff, approximately 7800 m/s. The data show the resolution effects with those methods which present computed error from nominal values (GCB and Δh, $\Delta \dot{h}$) giving the better performance. Presenting reduced resolution information (i.e., total actual values compared with total reference values) plus an alternate high resolution display for the final part or injection phase of the flight (h vs. $\dot{h}$ with $h - h_{co}$, $\dot{h} - \dot{h}_{co}$) resulted in performance similar to the direct error methods. The data for the first three methods fell well within the 1 km and 0.1° performance criteria shown by the dashed lines in figures 7 and 8. The data for the h vs. $\dot{h}$ method show the altitude criteria were met in 83 percent of the runs and the flight-path-angle criteria in all the runs. A typical display of h vs. $\dot{h}$ is shown in figure 9. The circle symbol shows the initial conditions, 2.3 km low in altitude and 0.085 km/s low in altitude rate. The orbit injection window is indicated by the small rectangle shown to scale in the lower right portion of the figure. The fifth method, t vs. h with $h - h_{co}$, shows the effect of no altitude rate or flight-path-angle information. The pilot could estimate the altitude rate from the altitude-time curve as it was plotted and from the $h - h_{co}$ signal during the injection phase. The performance for the t vs. h plus $h - h_{co}$ system, shown in figures 7 and 8, was better than anticipated with altitude error criteria met in 84 percent of the runs and flight-path angle within tolerance in 77 percent of the runs. The maximum flight-path-angle error was about three times the tolerance value, and the maximum altitude error was about two and a half times the tolerance chosen. Figure 10 shows a typical display plot of t vs. h. Again the circle symbol indicates the initial conditions and the flat rectangle at the right indicates the altitude error tolerance. The curve shows the pilot flying a slow oscillating path about the nominal reference curve, with a wave period of about 200 seconds. This low frequency guidance mode suggests that heavy filtering could be used to eliminate any high noise levels contained in the radar altimeter output.

The variation in performance of a single pilot and between pilots is indicated in figure 11. Here the three main performance parameters, velocity, altitude, and flight-path-angle errors at cutoff have been combined to form a performance factor (PF). The equation used was

$$PF = \frac{\Delta V_{co}(m/s)}{3} + \frac{\Delta h_{co}(km)}{1} + \frac{\Delta \gamma_{co}(deg)}{0.1}$$

Velocity cutoff error is discussed in detail later in the report, but is included in the performance factor to reflect the performance for the total task in the pitch plane. Figure 11 shows the range of data for a single subject with the data point indicating his average. The solid symbol is the average for all subjects for a particular scheme. The performance factor range for a single subject for the first four methods shows some variation. Of course, the t vs. h with $h - h_{co}$ method resulted in large range or spread because of the low display resolution and lack of altitude rate information. There is no obvious difference between performance factor for the trained pilots (subjects 1, 2, and 5) and that for the engineers (subjects 3, 4, and 6).

The average performance factor for all subjects (the solid symbol in fig. 11) indicates the relative performance between guidance schemes. The figure shows that performance with the first three schemes is quite similar with a performance factor of about 0.8. The performance factor for h vs. $\dot{h}$ method is about 125 percent of this value and the h vs. t method 300 percent. It appears that any one of the first three schemes (GCB; Δh, $\Delta \dot{h}$; h vs. $\dot{h}$ with $h - h_{co}$, $\dot{h} - \dot{h}_{co}$) could be used for a manual backup guidance system for a mission which has injection tolerances like those assumed in this study.

Yaw Plane Guidance

As discussed earlier, the pilot's task for yaw plane guidance was to maintain the lateral velocity at zero. The ability of the subject to do this while flying the various pitch plane guidance methods is indicated in figure 12. Here the integral of the lateral velocity (i.e., lateral displacement at cutoff) is plotted for each guidance scheme. Lateral displacement error information was available to the simulated ground controller in the GCB method, and the samll errors in displacement at cutoff shows the effect of closing this loop. This ground voice information was not used for the other four methods so that the performance with a completely on-board simple system could be evaluated. The error in the lateral position for these four schemes was 2 km or less for 95 percent of these runs. The penalty for a 2 km error would have to be determined for a particular mission, taking into account error variations in the other flight parameters at cutoff.

The lateral velocity at cutoff is shown in figure 13. The lateral velocity display meter resolution used with all pitch guidance schemes except GCB was 2.76 m/s per mm of needle deflection. The resolution for the GCB method was similar with a simulated ground monitor plot sensitivity of 2.5 m/s per mm. This resolution is indicated on the figure. The lateral velocity at cutoff was within 10 m/s for all guidance schemes corresponding to a yaw plane flight-path-angle error of about 0.07°. The low values and tight grouping of the data for the Δh, $\Delta \dot{h}$ scheme probably reflects the pilot's smaller scan pattern inherent for this method, although the integral of the velocity error, shown in figure 12, was not significantly less than the values for the other schemes except for subject 2.

Thrust Termination

The thrust termination of the third stage engine at the proper earth-orbit velocity was an additional pilot task in this simulation study. Two shut-down techniques were used. With the first technique the pilot was assumed to have a direct digital readout of the actual horizontal velocity minus the desired earth orbit horizontal velocity. For a circular orbit, the other two velocity components are essentially zero. The second technique was to use a voice command warning and countdown from a simulated ground controller.

Before detailed thrust termination procedures were established, a preliminary study was made to determine a good digital display system for the task. A single task study was conducted to evaluate the effect of the rate of change in digital display on the reaction time of the pilot. The procedure was to have the pilot hit the simulated thrust termination button when a value of zero was displayed. The task was started with 10 seconds to go. Twenty runs were made for each digital display rate condition. A simple light signal thrust termination command with a 10-second warning was also tried. Figure 14 shows the results of this study, with the average value of the absolute error (lead or lag) in cutoff time plotted versus digital display rate. Also shown is the equivalent error in time, 4.5 seconds, for an injection velocity error of ± 3 m/s at the third stage burnout acceleration of 0.69 g. It is seen that all the data fall well below this value. For a single choice task, the reaction time is about equal to the 0.20 to 0.25 second neuromuscular lag. This latter condition agrees with the warning light cutoff task data. With additional lead information in the form of a digital countdown, performance was much better. Digital display rates of 1 and 2 digits per second resulted in time errors on the order of 0.06 second.

From these results, it was decided that a good cutoff display would be obtained by displaying the velocity to go, in feet per second. At nominal third stage cutoff, the velocity change is about 22 fps^2. The rate of change in the tens' digit corresponds approximately to the 2 digits per second (near optimum) display rate and so a velocity error at cutoff as low as 1.3 fps (22×0.06) or 0.4 m/s is possible for this single task.

The velocity display discussed above was used in a side study to compare the two manual thrust cutoff techniques, ground voice command, and direct readout, in conjunction with a guidance task. The last 2 minutes of flight before third stage thrust termination was simulated with an initial altitude error of 2 km low. The pilot's task was to null this initial error and fly to cutoff conditions using the Δh, $\Delta\dot{h}$ pitch guidance scheme, and then terminate the thrust. The direct readout technique consisted in the pilot monitoring the velocity change to go on the digital display at the bottom of his display panel. With the ground voice technique the pilot had no direct readout and was given shut down commands by a ground controller. The pilot was given a 30-second warning and a 5-second countdown to thrust termination. These two techniques are compared in figure 15, where errors in velocity, altitude, and flight-path angle at cutoff for the two methods are plotted. The data show a slight deterioration in performance for the direct readout technique. This can be attributed to the pilot's larger workload with this additional display included in his visual scan pattern. This indicates that the ground voice command technique is desirable. Also, comparing these data with those for the single task in figure 14 shows that a further improvement should be possible by assigning the thrust cutoff task to one of the other astronauts in a multiman mission.

The velocity errors at cutoff for the guidance scheme comparison study are plotted in figure 16. For the majority of the thrust termination runs the ground voice command system was used. A few runs were made using the direct readout method and these are indicated by the solid symbols. The data generally show quite a bit of scatter but the 3 m/s velocity error criterion (dashed lines) was met for 90 percent of the runs. No particular correlation is evident between velocity error and guidance scheme.

Off-Nominal Conditions and Rate Augmentation Failure Condition

Nulling the initial condition trajectory errors used in this study was no problem for any of the flights. An additional off nominal condition was simulated wherein a 3-percent reduction in thrust was assumed. This condition had no noticeable effect on the manual guidance task or performance other than an increase in time to reach orbital velocity.

A failure condition was simulated wherein the rate augmentation loop in each of the three axes was assumed to be open at second stage ignition and throughout the rest of the flight. The performance data for the rate dampers out are compared with the normal condition in figure 17. The Δh, $\Delta\dot{h}$ guidance scheme was used for this comparison. The pilots could maintain control of the vehicle and the data show injection performance was fairly well within tolerances. However, the pilots considered the system with no rate damper to be unacceptable for normal operation and the system with normal rate dampers to be satisfactory, but to have some mildly unpleasant characteristics.

CONCLUSIONS

A preliminary study has been made of manual guidance techniques for the upper stages of the Saturn V launch vehicle. Five guidance schemes were studied:

(1) Ground control voice command
(2) Error null of altitude error and altitude rate error
(3) Altitude versus altitude rate on an X - Y presentation with a reference nominal flight curve plus display of altitude error from cutoff value and altitude rate error from cutoff value
(4) Altitude versus altitude rate on an X - Y presentation with a reference nominal flight curve
(5) Mission time versus altitude on an X - Y presentation with a reference nominal flight curve plus display of altitude error from cutoff value.

A nominal pitch altitude versus time schedule card on the mission time clock was provided for schemes (2) through (5). The ability of the pilot to use these schemes to guide and inject the payload into a 185-km (100 nautical mile) altitude circular orbit was investigated. His performance was indicated by altitude, flight-path angle, and velocity differences from nominal values at injection. The injection error "window" selected was:

Altitude error	±1000 m
Velocity error	±3 m/s
Flight-path-angle error	$\pm 0.1^{\circ}$

The study resulted in the following conclusions:

1. With good display resolution, methods (2) and (3) and the ground command scheme, performance was well within tolerances. Performance was fairly good with the total value display schemes, (4) and (5), where altitude error was less than ±1000 m for 84 percent of the runs for each scheme, and flight-path-angle errors were well within tolerance for scheme (4) and less than 0.1° for 75 percent of the runs of scheme (5).

2. The study indicates that the manual velocity cutoff task should be separated from the guidance and control task and assigned to another crew member on multi-manned missions.

3. Three off-nominal situations were studied - rate dampers out, thrust reduction, and initial trajectory velocity and displacement dispersions. (The latter was included for all flights.) The study results indicated that

a. Injection performance did not deteriorate appreciably with rate damping out on all axes; however, the pilots rated the system as unacceptable for normal operation.

b. The 3-percent reduction in thrust did not affect performance noticeably other than to increase time to reach orbital velocity.

c. The off-nominal initial trajectory velocity and displacement values used had no appreciable effect on the mission as the pilots could easily recover back to near-nominal conditions.

Ames Research Center
National Aeronautics and Space Administration
Moffett Field, Calif., Nov. 28, 1966
904-01-08-02

APPENDIX A

EQUATIONS OF MOTION

The following assumptions were made in deriving the equations of motion:

1. rigid body
2. no aerodynamic forces
3. no fuel sloshing
4. neglect effects of swiveling engine mass
5. planar nominal trajectory

The position and attitude terms were derived as error or perturbation quantities with respect to an assumed nominal trajectory and vehicle attitude program. Cosines and sines of small angles were approximated by first order series expansion terms. Geometrical representation of axes systems is shown in figure 18.

$$\dot{v}_{1p} = -kF_\beta\beta_P \cos \chi_{3B} - kF_\varphi\Delta\varphi_{3b} \cos \chi_{3B} + g\Delta\theta_{3e} \cos \theta_n - (k - 1)F_\varphi \sin \chi_{3B}$$

$$\dot{v}_{2p} = -kF_\beta\beta_P \sin \chi_{3B}[-kF_\varphi\Delta\varphi_{3b} \sin \chi_{3B}] + g\Delta\theta_3 \sin \theta_n + (k - 1)F_\varphi \cos \chi_{3B}$$

$$\dot{v}_{3p} = kF_\varphi\varphi_{1b} - kF_\beta\beta_Y$$

$$\dot{\omega}_{1b} = -kM_\beta\beta_Y$$

$$\dot{\omega}_{2b} = -M_\beta{}'\beta_R$$

$$\dot{\omega}_{3b} = -kM_\beta\beta_P$$

$$\dot{\varphi}_{1b} = \omega_{1b}$$

$$\dot{\varphi}_{2b} = \omega_{2b}$$

$$\dot{\varphi}_{3b} = \omega_{3b}$$

$$\Delta\dot{\varphi}_{3b} = \omega_{3b} - \dot{\chi}_{3B}$$

$$\Delta\dot{h} = -\dot{x}_{1p} \sin \theta_n + \dot{x}_{2p} \cos \theta_n -(x_{1p} \cos \theta_n + x_{2p} \sin \theta_n)\dot{\theta}_n = \dot{x}_{2e} - x_{1E}\dot{\theta}_n$$

$$\Delta h = -x_{1p} \sin \theta_n + x_{2p} \cos \theta_n = x_{2e}$$

$$\dot{x}_{1p} = v_{1p}$$

$$\dot{x}_{2p} = v_{2p}$$

$$\dot{x}_{3p} = v_{3p}$$

$$\Delta\theta_3 = - \frac{1}{r_n}(x_{1p} \cos \theta_n + x_{2p} \sin \theta_n)$$

Time varying coefficients

$F_\beta = T'/m$	$(m/s^2)/rad$
$F_\varphi = T/m$	$(m/s^2)/rad$
$M_\beta = y_\beta T'/I_{1b}$	$(rad/s^2)/rad$
$M_\beta' = r_\beta T^*/I_{2b}$	$(rad/s^2)/rad$

Display equations

Attitude

$$\varphi_{pitch} = 57.3\ \chi_{3B} + 90 + 57.3\ \varphi_{3b}$$

$$\varphi_{yaw} = 57.3\ \varphi_{1b}$$

$$\varphi_{roll} = 57.3\ \varphi_{2b}$$

Altitude

$$\dot{h} = \dot{h}_n + \Delta\dot{h}$$

$$h = h_n + \Delta h$$

Velocity

$$V_{1p} = V_{1p_n} + v_{1p}$$

$$V_{2p} = V_{2p_n} + v_{2p}$$

$$V_{3p} = v_{3p}$$

$$\Delta V_{co} = V - V_{co_n} = \dot{V}_{co}(t - t_{co}) + v_{1e}$$

$$= \dot{V}_{co}(t - t_{co}) + v_{1p} \cos \theta_n + v_{2p} \sin \theta_n$$

Control law

$$\beta_P = a_{3b}\dot{\phi}_{3b} - b_P\delta_P$$

$$\beta_Y = a_{1b}\dot{\phi}_{1b} - b_Y\delta_Y$$

$$\beta_R = a_{2b}\dot{\phi}_{2b} - b_R\delta_R$$

$$(\beta = \beta c)$$

Nominal time functions

$$\theta_n = -8.27\times10^{-7}t^2 - 3.35\times10^{-4}t - 0.01745 \text{ (rad)}$$

$$\dot{\theta}_n = -16.54\times10^{-7}t - 3.35\times10^{-4} \text{ (rad/s)}$$

$$\chi_{3B} = -0.001745\ t - 0.9860 \text{ (rad)}$$

$$\dot{\chi}_{3B} = -0.001745 \text{ (rad/s)}$$

$$(F_\varphi)_{S\text{-}II} = (0.14125 - 2.395\times10^{-4}t)^{-1} \text{ (m/s}^2\text{/rad)}$$

$$(F_\varphi)_{S\text{-}IVB} = (0.2259 - 2.395\times10^{-4}t)^{-1} \text{ (m/s}^2\text{/rad)}$$

$$(F_\beta)_{S\text{-}II} = 0.8(F_\varphi)_{S\text{-}II} \text{ (m/s}^2\text{/rad)}$$

$$(F_\beta)_{S\text{-}IVB} = (F_\varphi)_{S\text{-}IV} \text{ (m/s}^2\text{/rad)}$$

$$(M_\beta)_{S\text{-}II} = 7.18\times10^{-11}t^4 + 0.380 \text{ (rad/s}^2\text{)/rad}$$

$$(M_\beta)_{S\text{-}IVB} = 0.65 \text{ (rad/s}^2\text{)/rad}$$

$$(M_\beta')_{S\text{-}II} = 10.0 \text{ (rad/s}^2\text{)/rad}$$

$$(M_\beta')_{S\text{-}IVB} = 0.026 \text{ (rad/s}^2\text{)/rad}$$

$$V_s = 1.16\times10^{-2}t^2 + 6.36\ t + 2500\ (m/s)$$

$$\dot{h}_n = 1062\ e^{-0.0085t}\ (m/s)$$

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TABLE I.- PITCH PLANE TRAJECTORY GUIDANCE SCHEMES

	Trajectory variable						Control variable		
Trajectory guidance scheme	h	Δh	$\dot{h}$	$\Delta \dot{h}$	$h - h_{co}$	$\dot{h} - \dot{h}_{co}$	φ	$\dot{\varphi}$	φ_n
GCB								Meter	
$\Delta h, \Delta \dot{h}$		Meter		Meter			All axes attitude indicator	Meter	Clock
h vs. $\dot{h}$, $h - h_{co}$, $\dot{h} - \dot{h}_{co}$	X-Y plotter	X-Y plotter	X-Y plotter	X-Y plotter	Meter	Meter	All axes attitude indicator	Meter	Clock
h vs. $\dot{h}$	X-Y plotter	X-Y plotter	X-Y plotter	X-Y plotter			All axes attitude indicator	Meter	Clock
t vs. h, $h - h_{co}$	X-Y plotter	X-Y plotter			Meter		All axes attitude indicator	Meter	Clock

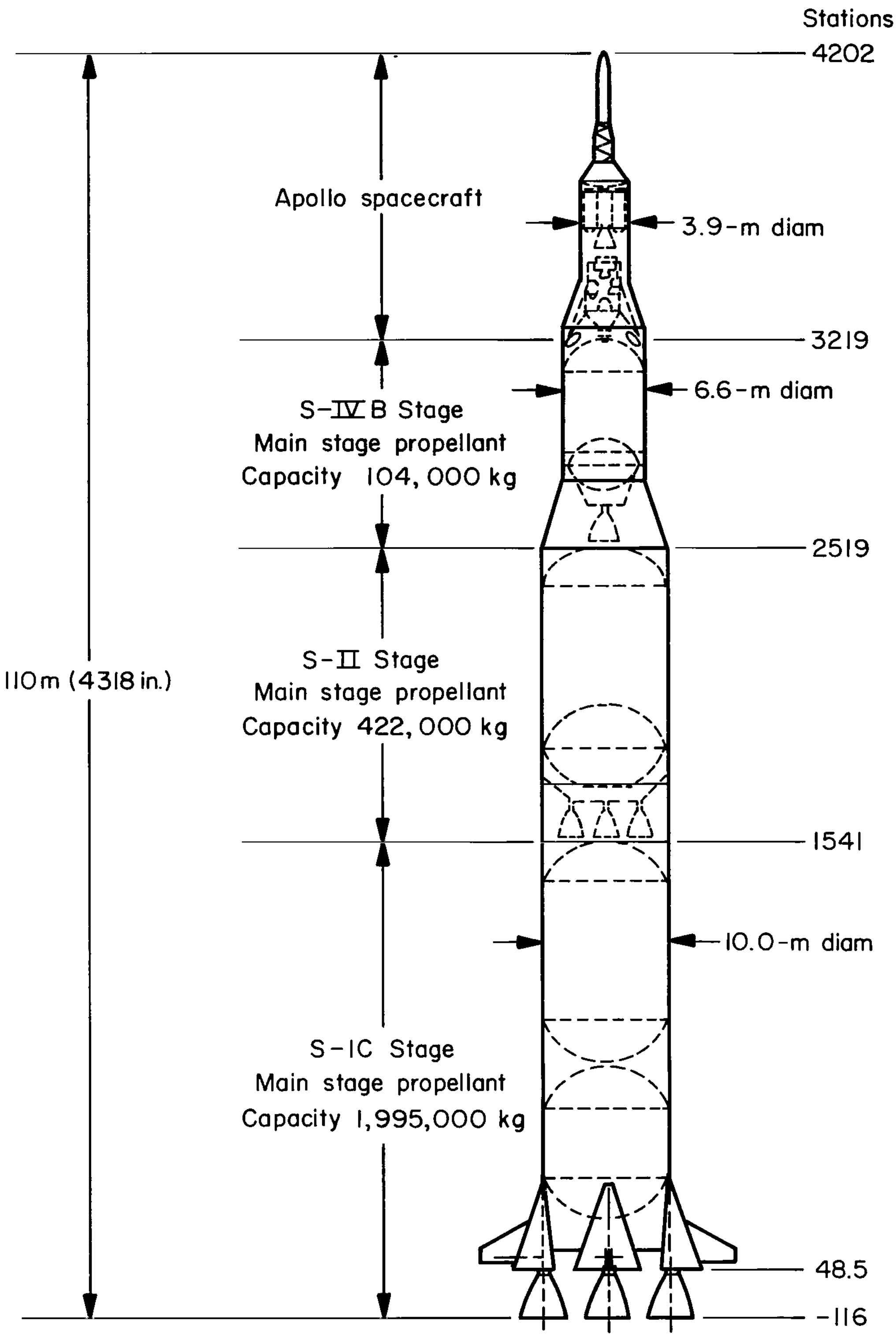

Figure 1.- Saturn V vehicle configuration.

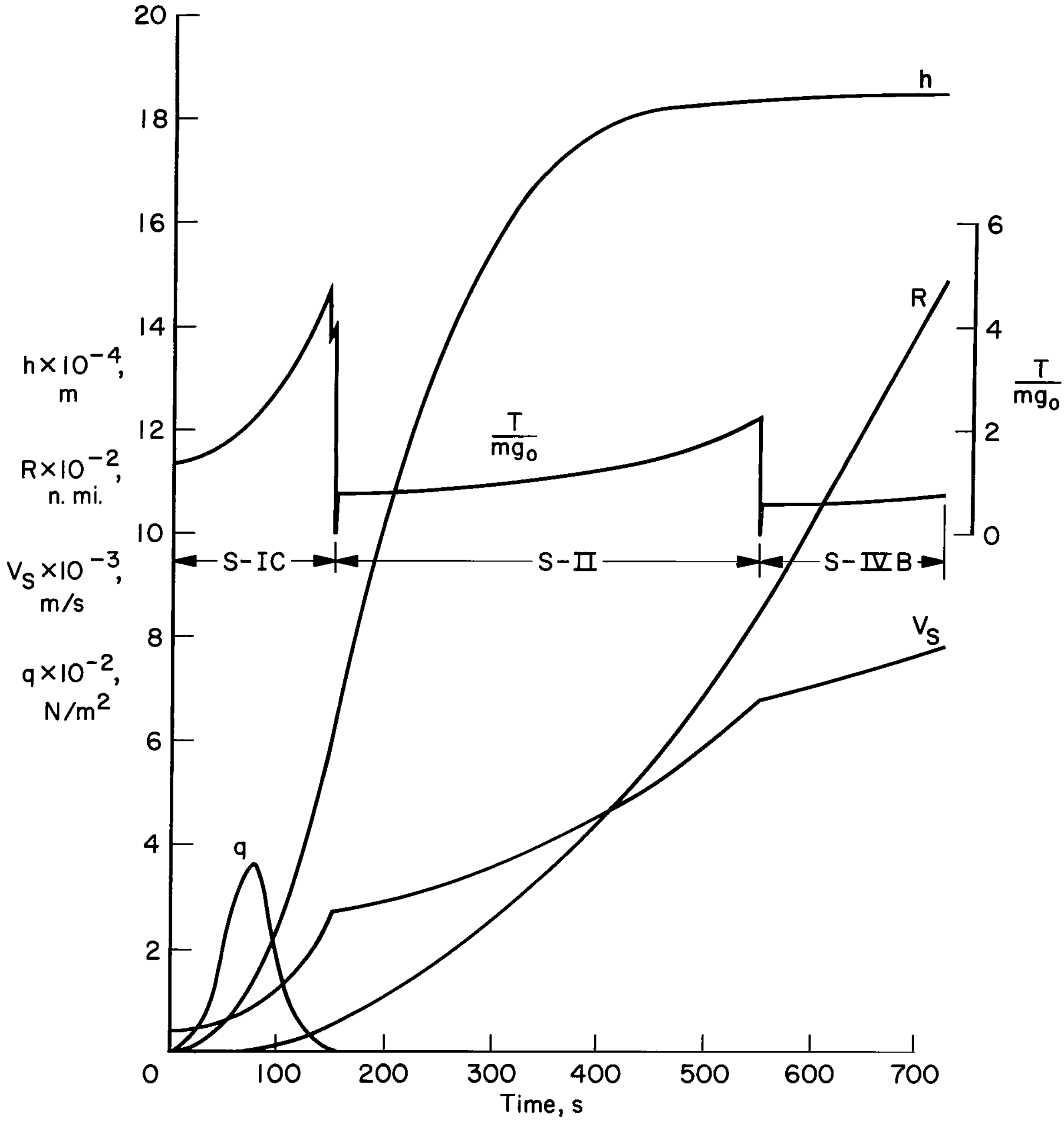

Figure 2.- Typical trajectory.

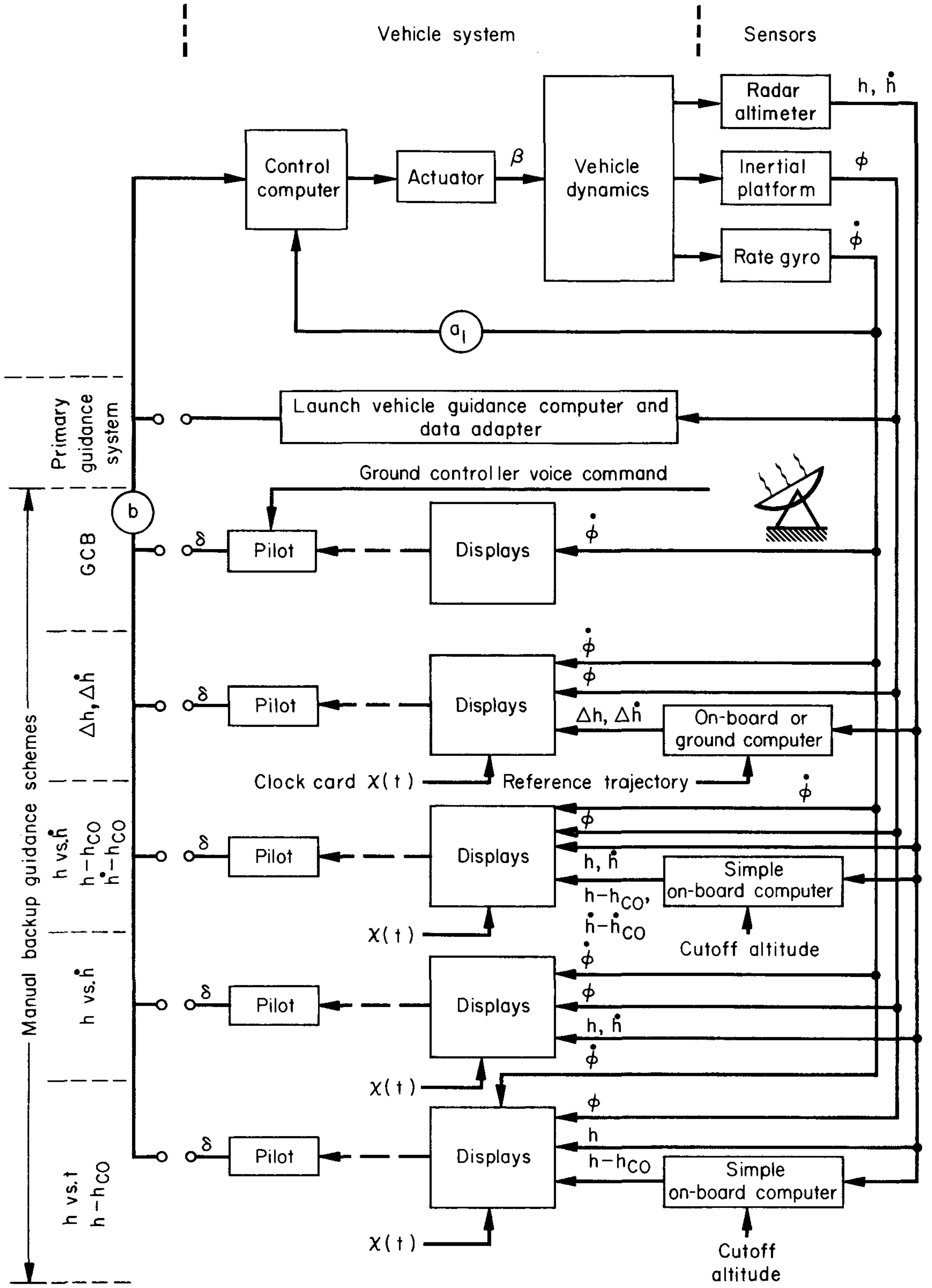

Figure 3.- Pitch plane guidance and control systems.

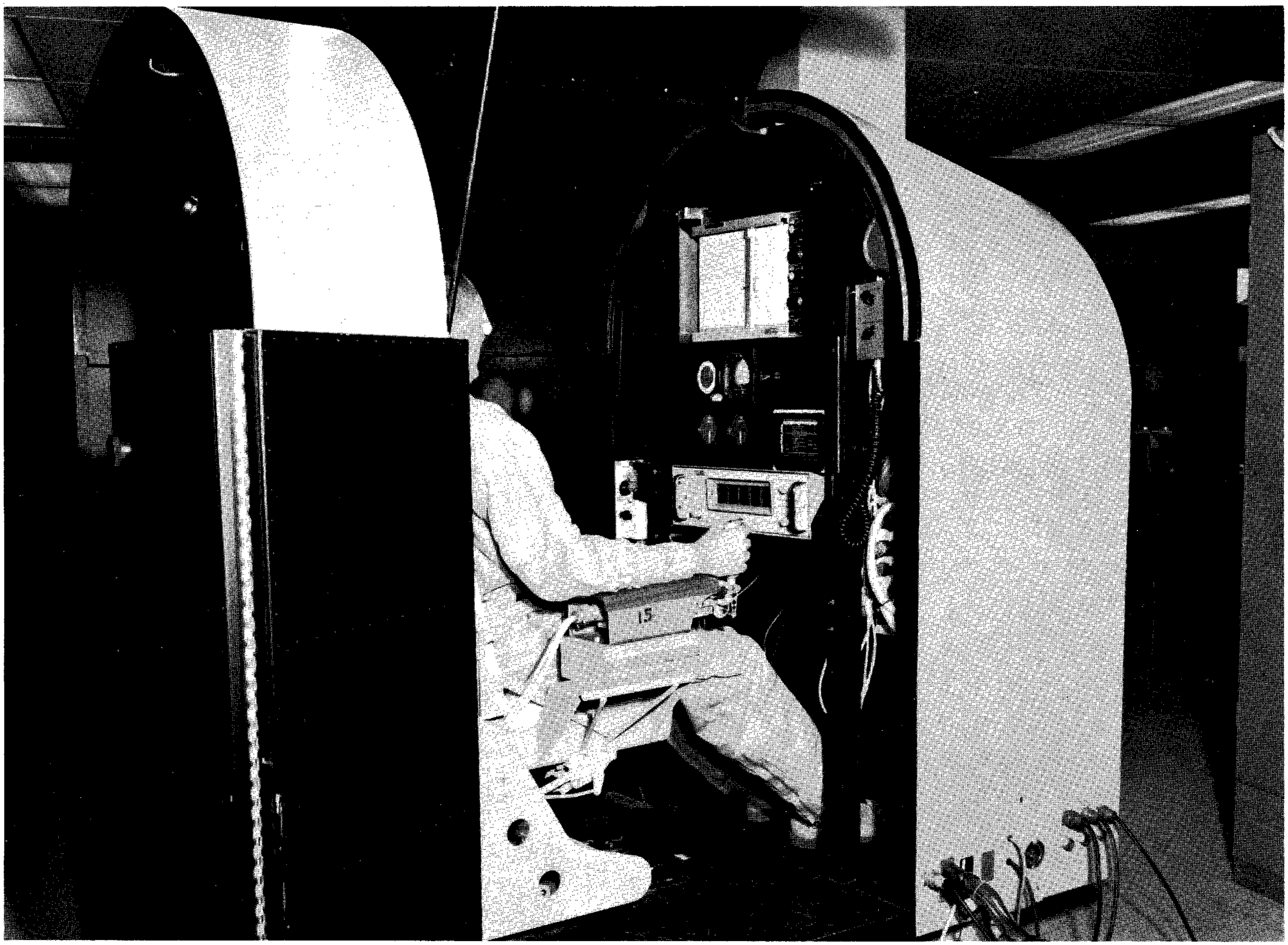

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Figure 4.- Manual guidance simulator.

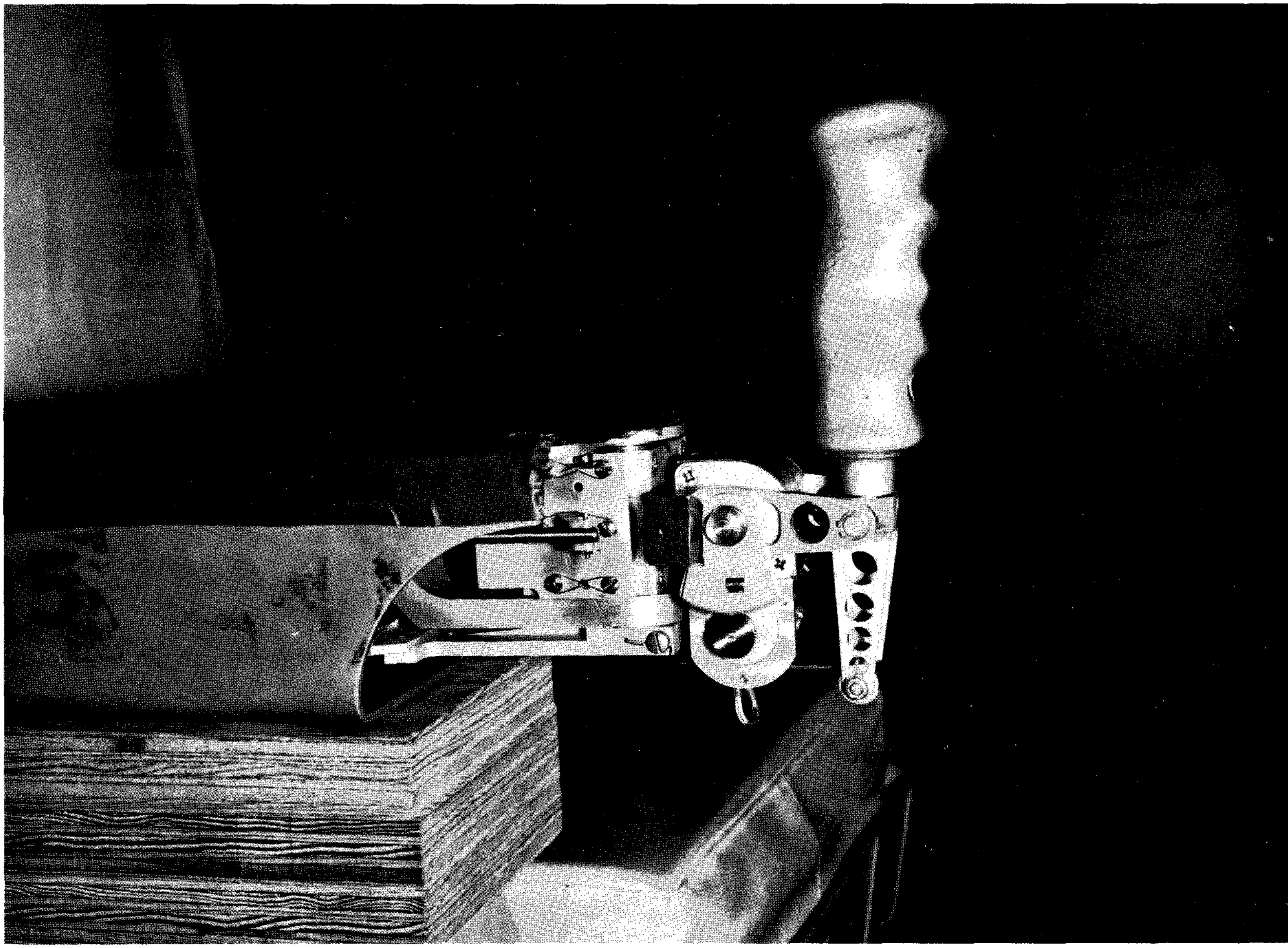

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Figure 5.- Three-axis side-arm controller.

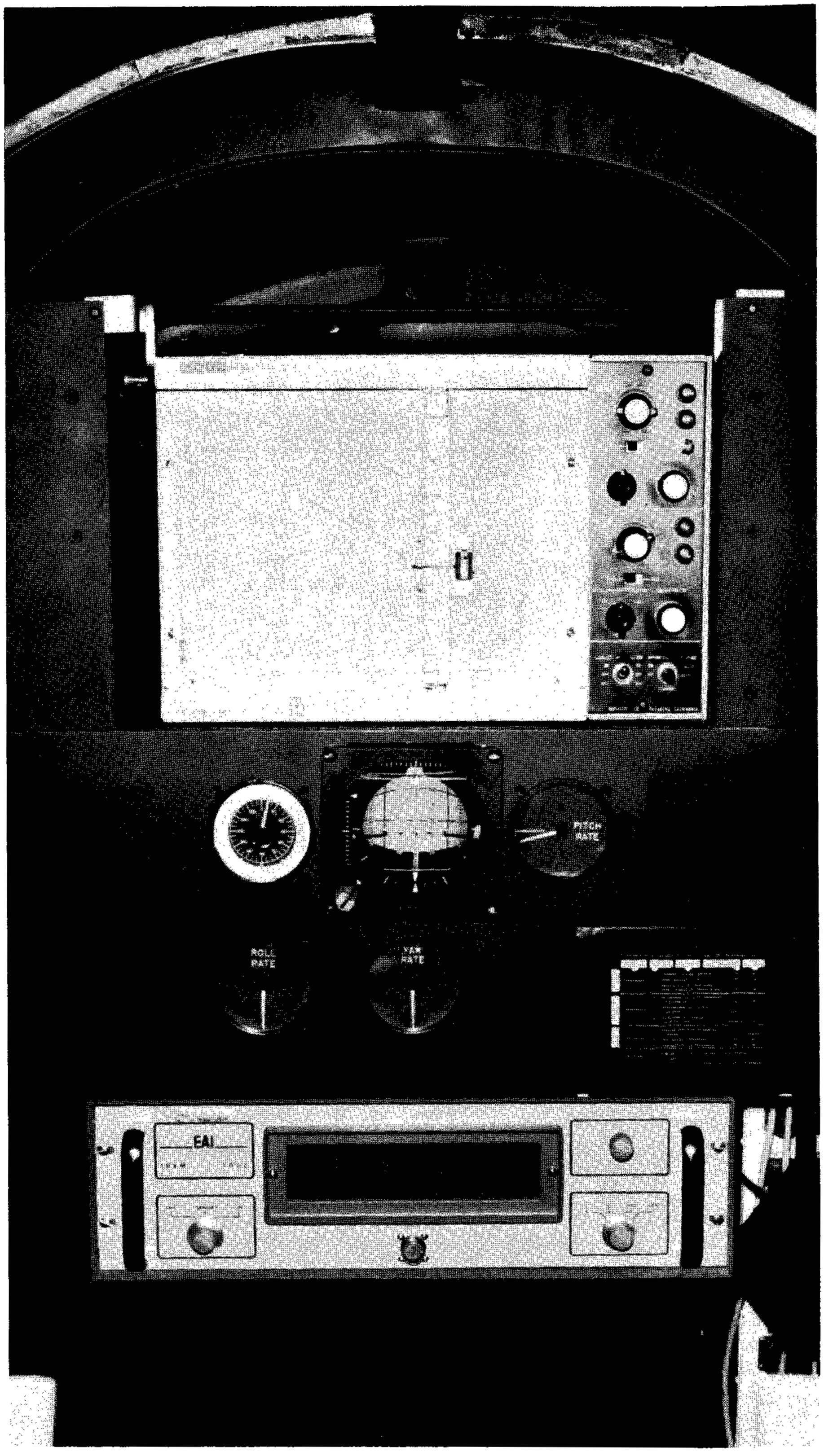


Figure 6.- Manual guidance display panel.   A-33190

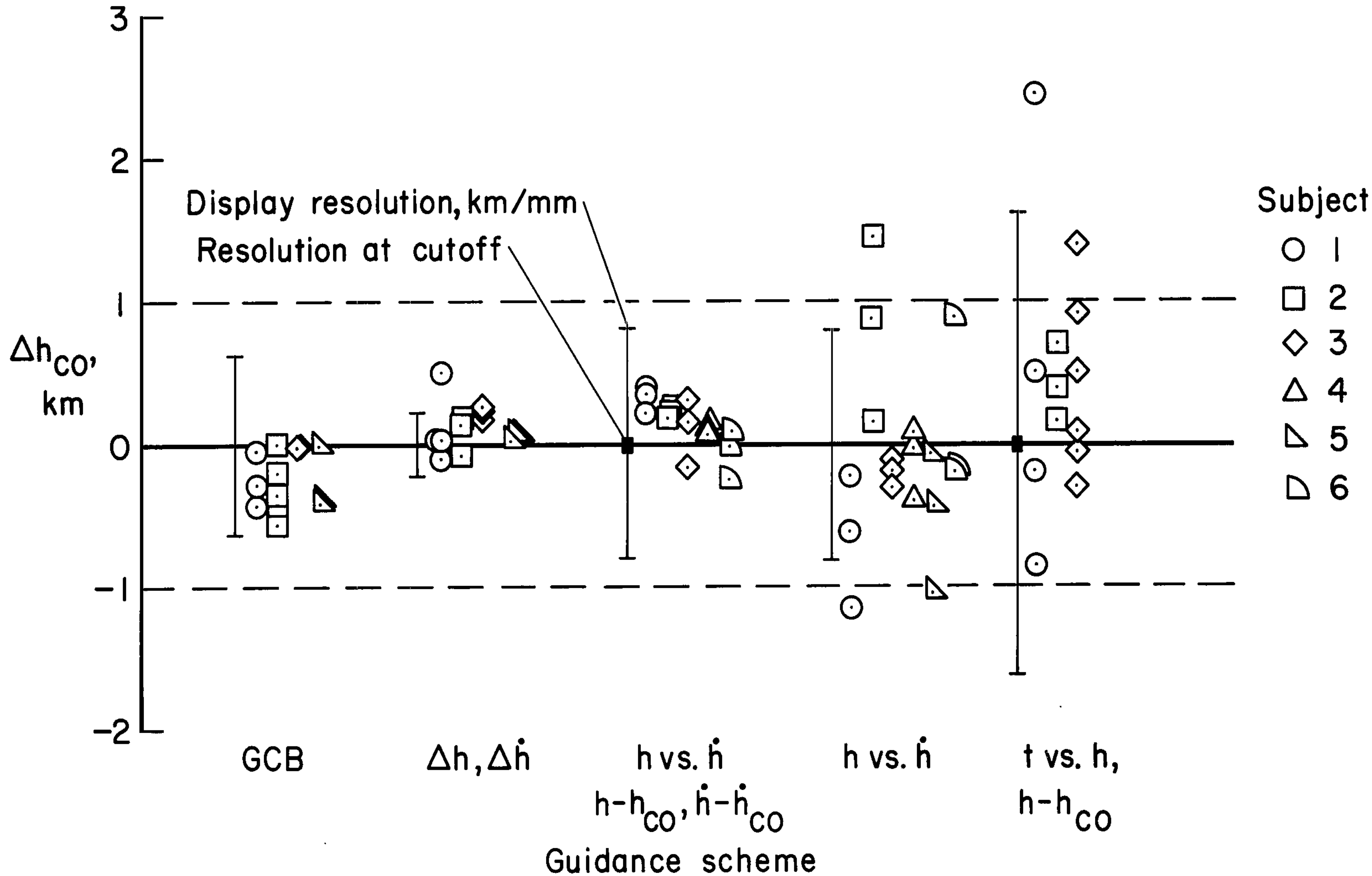


Figure 7.- Cutoff altitude performance.

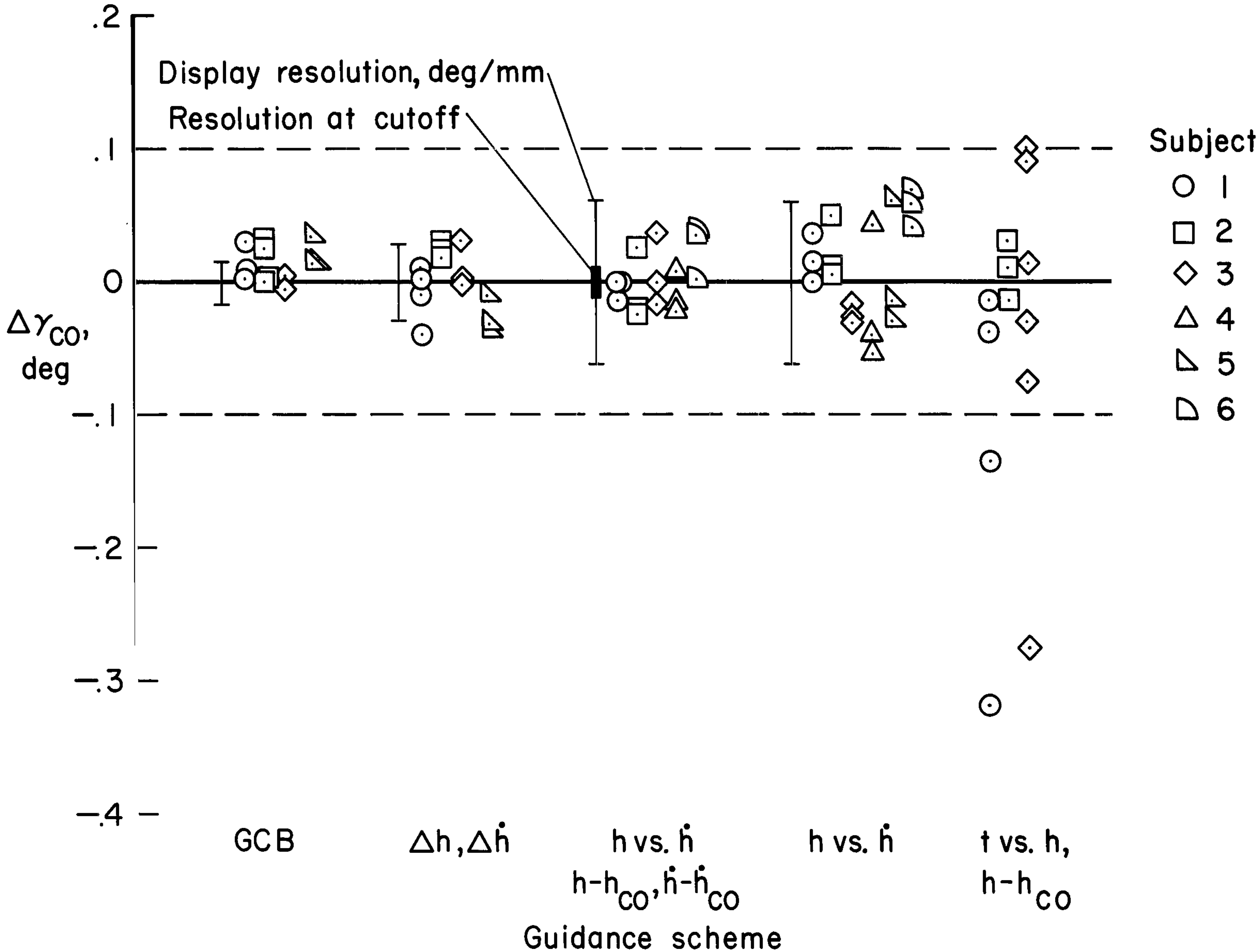


Figure 8.- Cutoff flight-path angle performance.

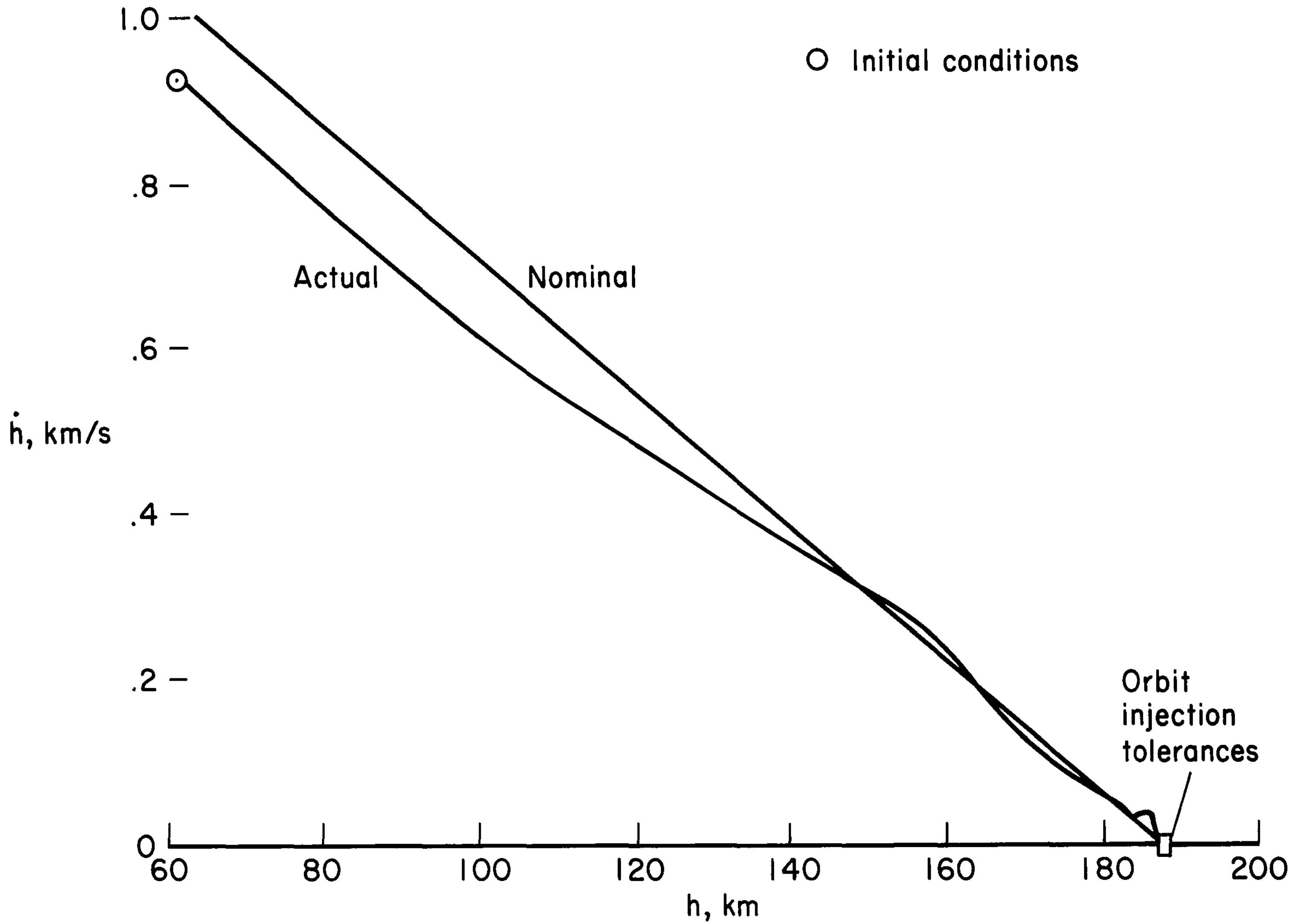


Figure 9.- Typical pilot guidance curve using the h vs. $\dot{h}$ guidance scheme.

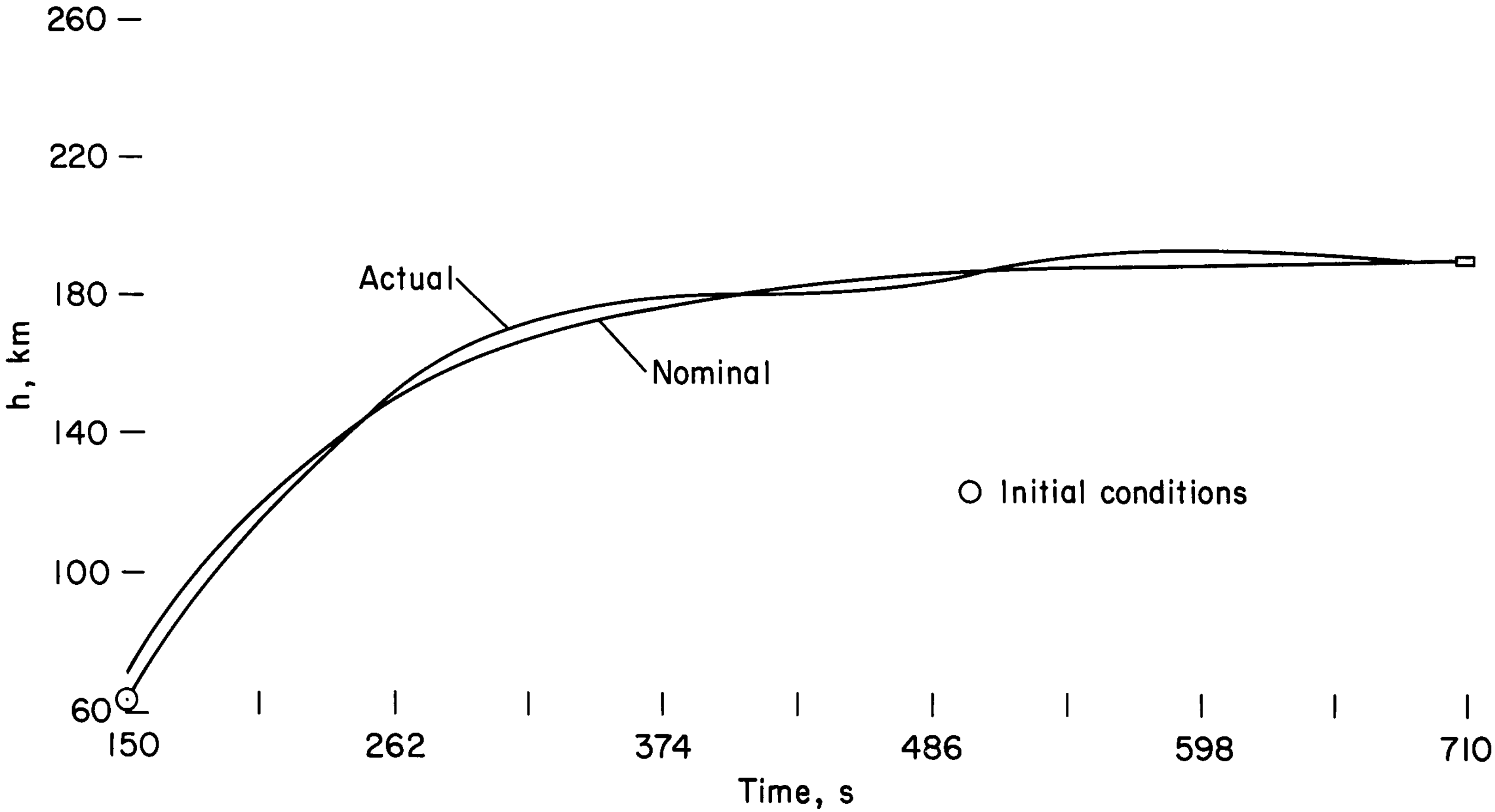


Figure 10.- Typical pilot guidance curve using the t vs. h guidance scheme.

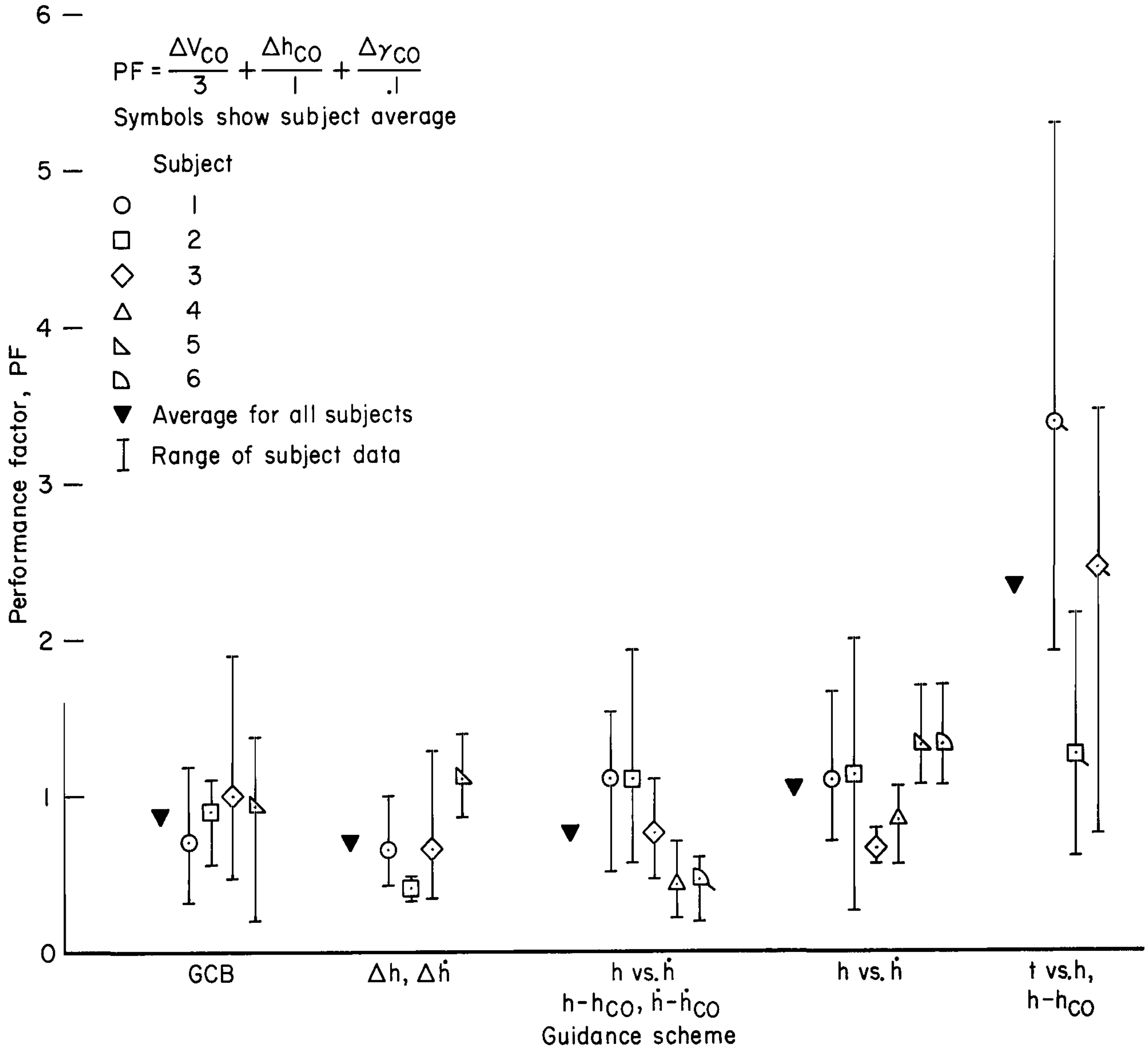


Figure 11.- Pitch plane guidance performance factor.

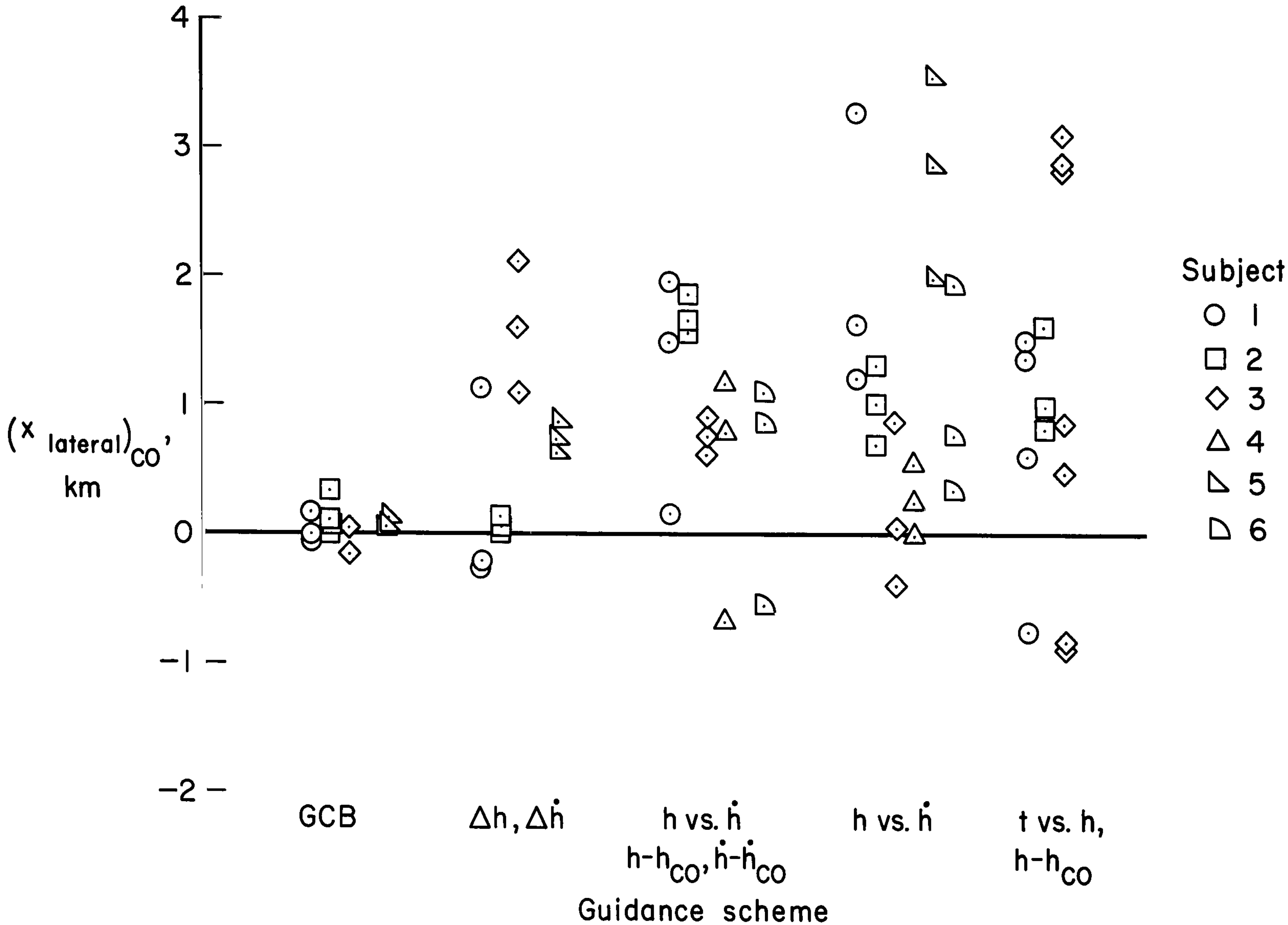


Figure 12.- Lateral displacement at cutoff.

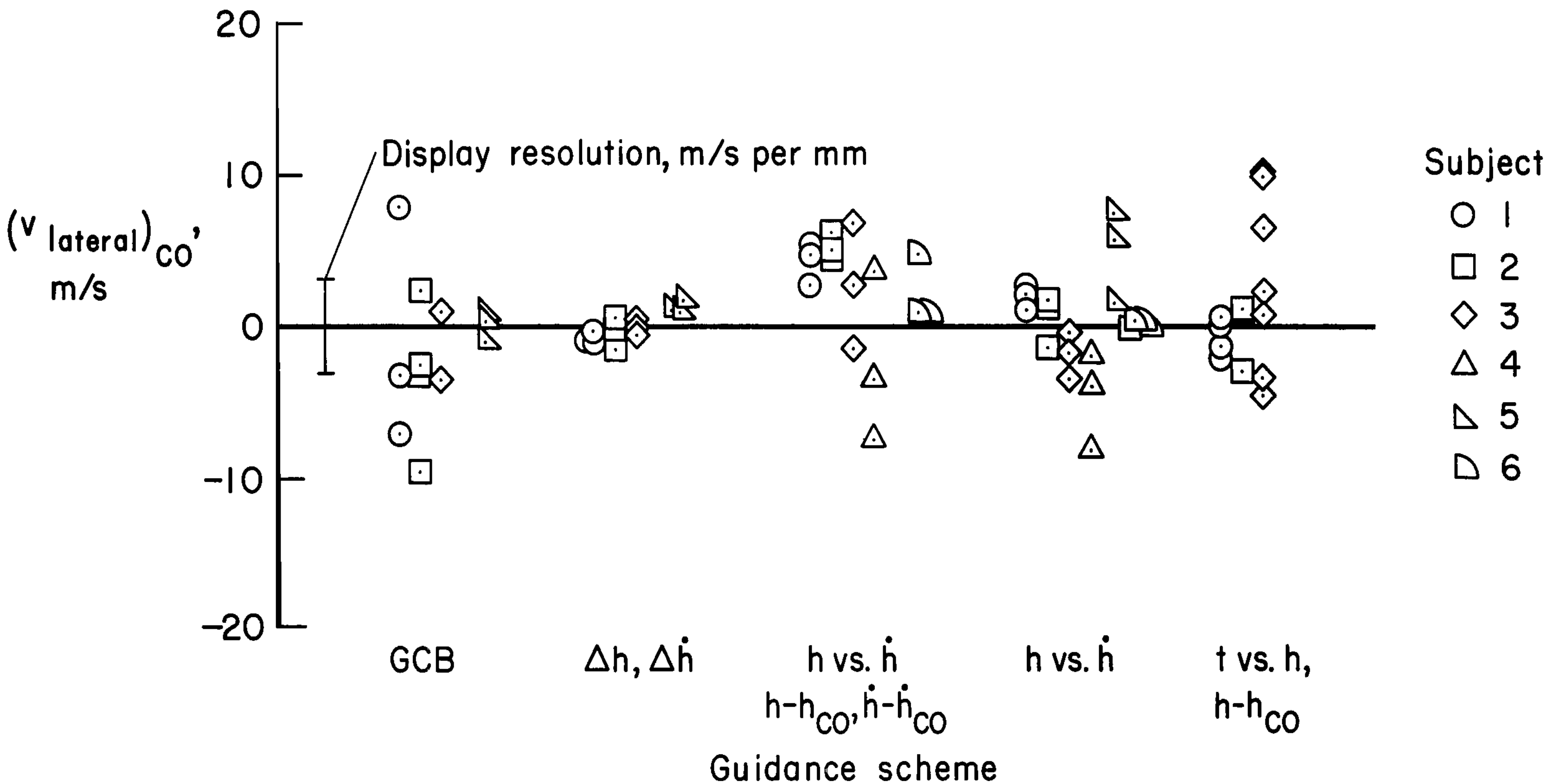


Figure 13.- Lateral velocity at cutoff.

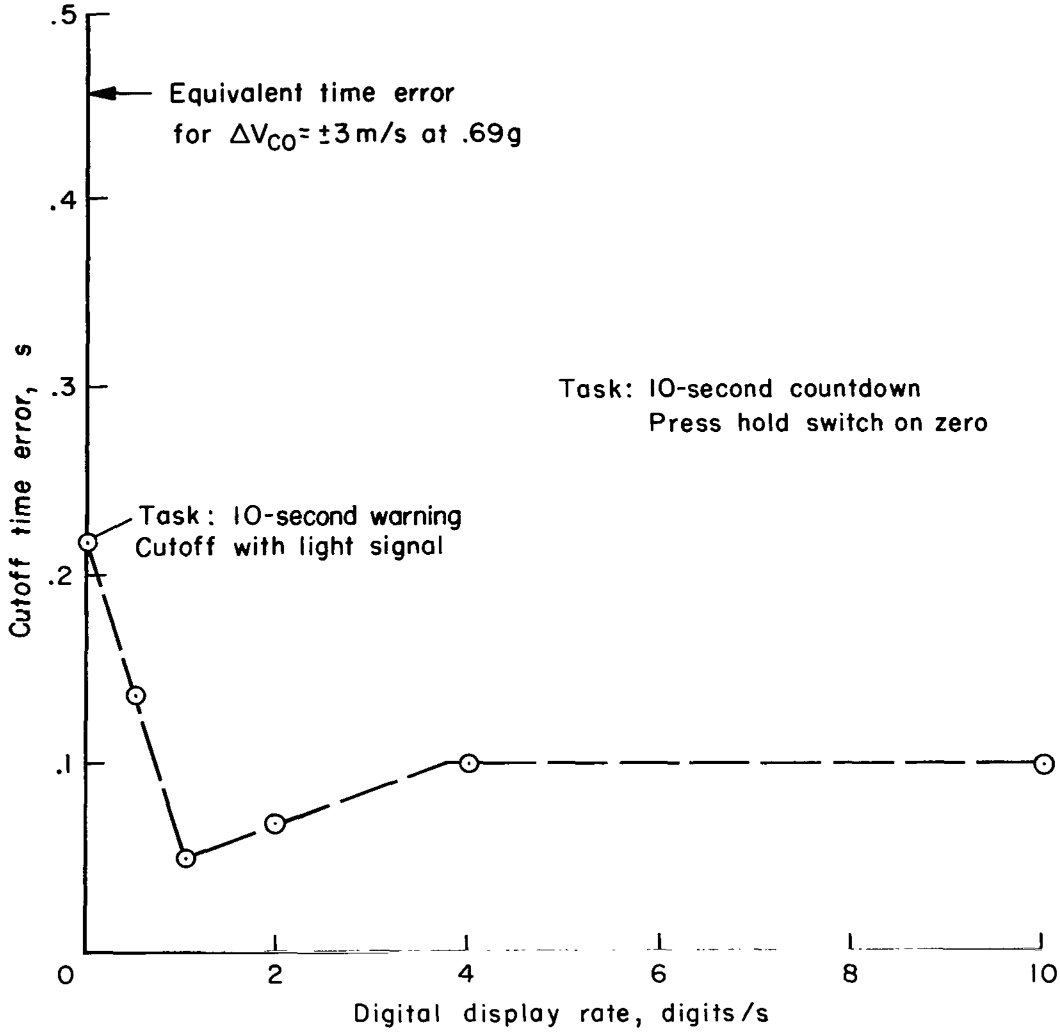
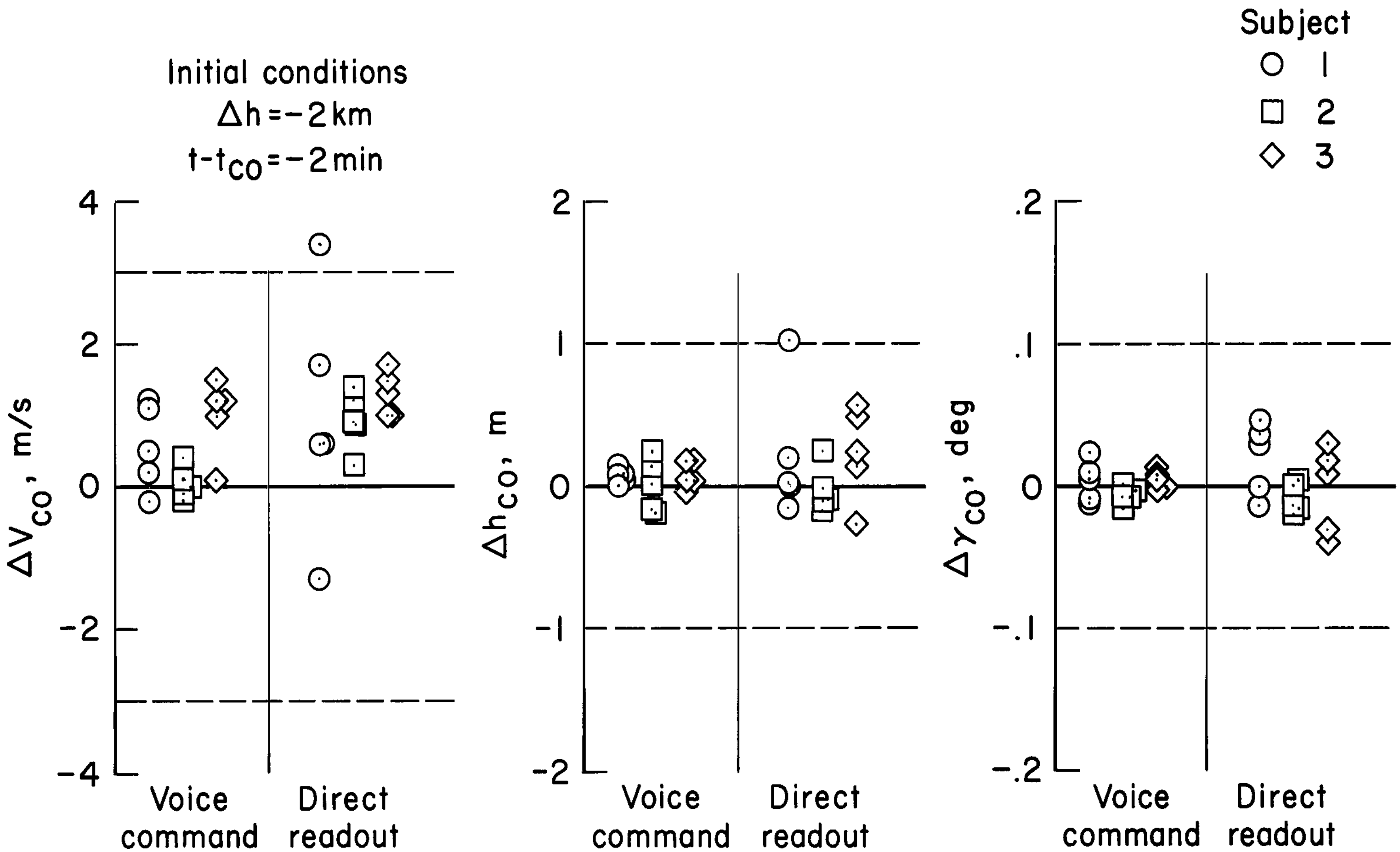


Figure 14.- Reaction time study.



Voice command: Pilot cutoff by ground voice command; 30-second warning 5-second countdown

Direct readout: Pilot cutoff by direct readout of fps to go; 20-fps per second countdown

Figure 15.- Thrust cutoff technique comparison; Δh, $\Delta\dot{h}$ guidance scheme.

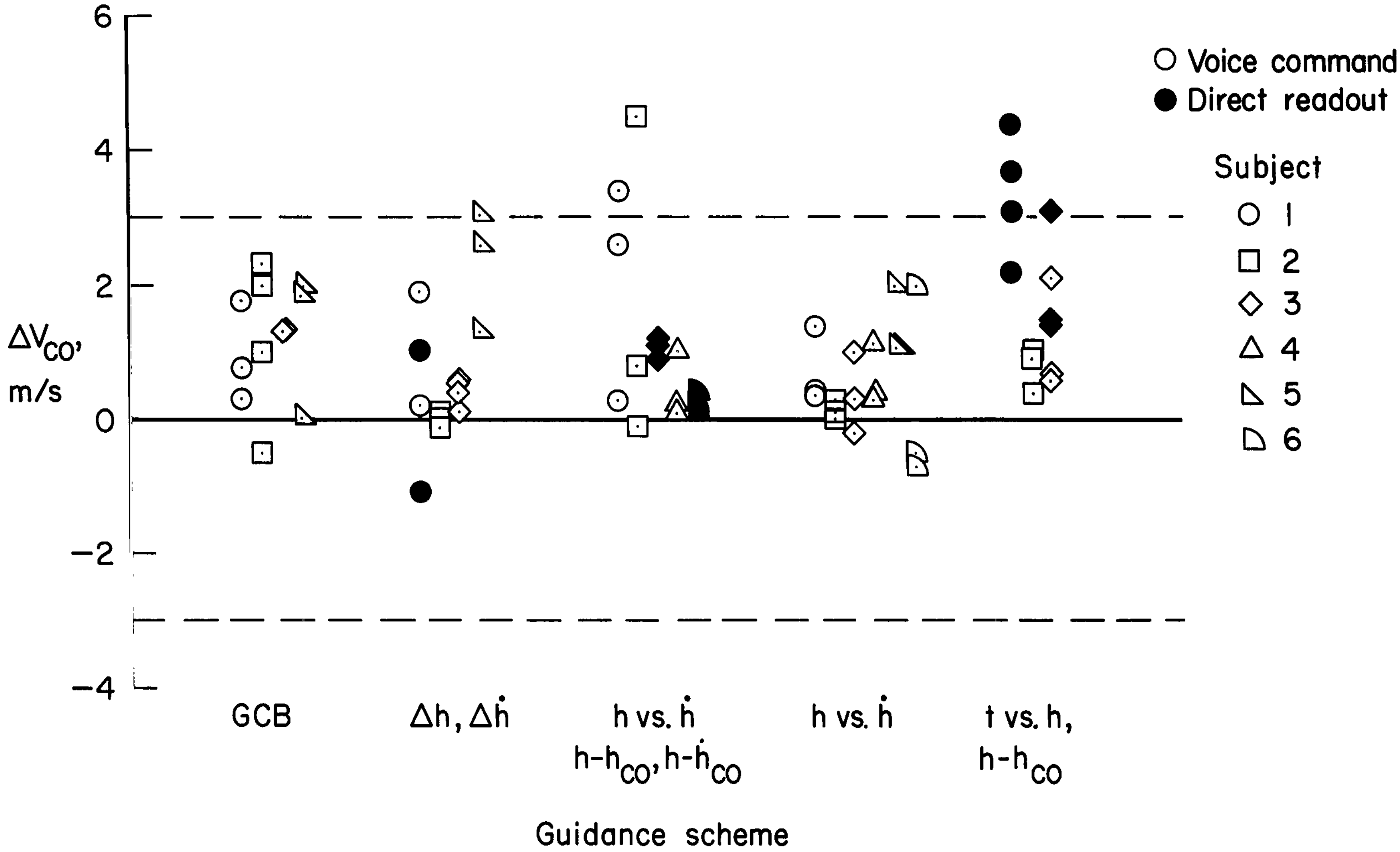


Figure 16.- Cutoff velocity performance.

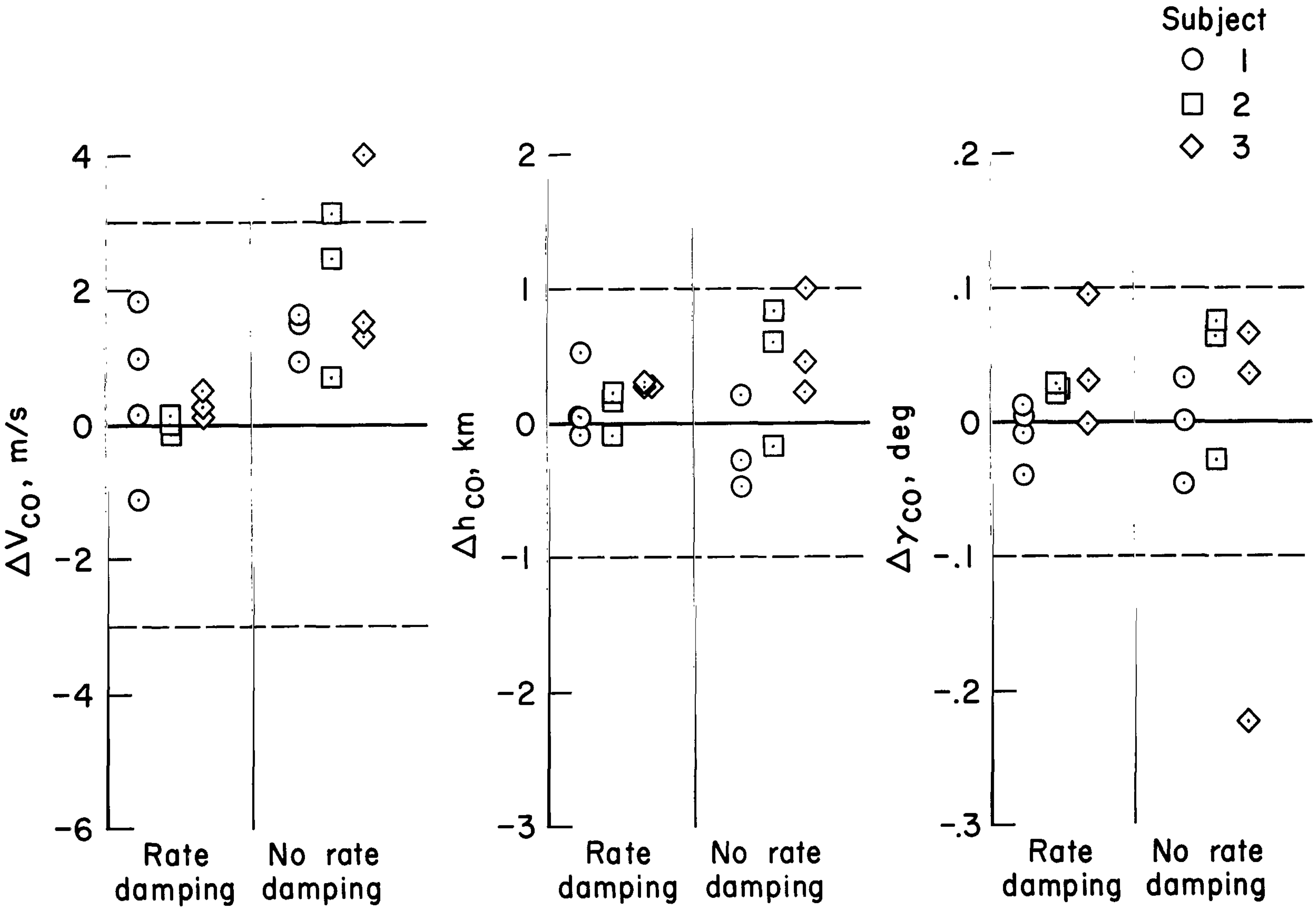


Figure 17.- Effect of rate damping; Δh, $\Delta \dot{h}$ guidance scheme, cutoff by direct readout.

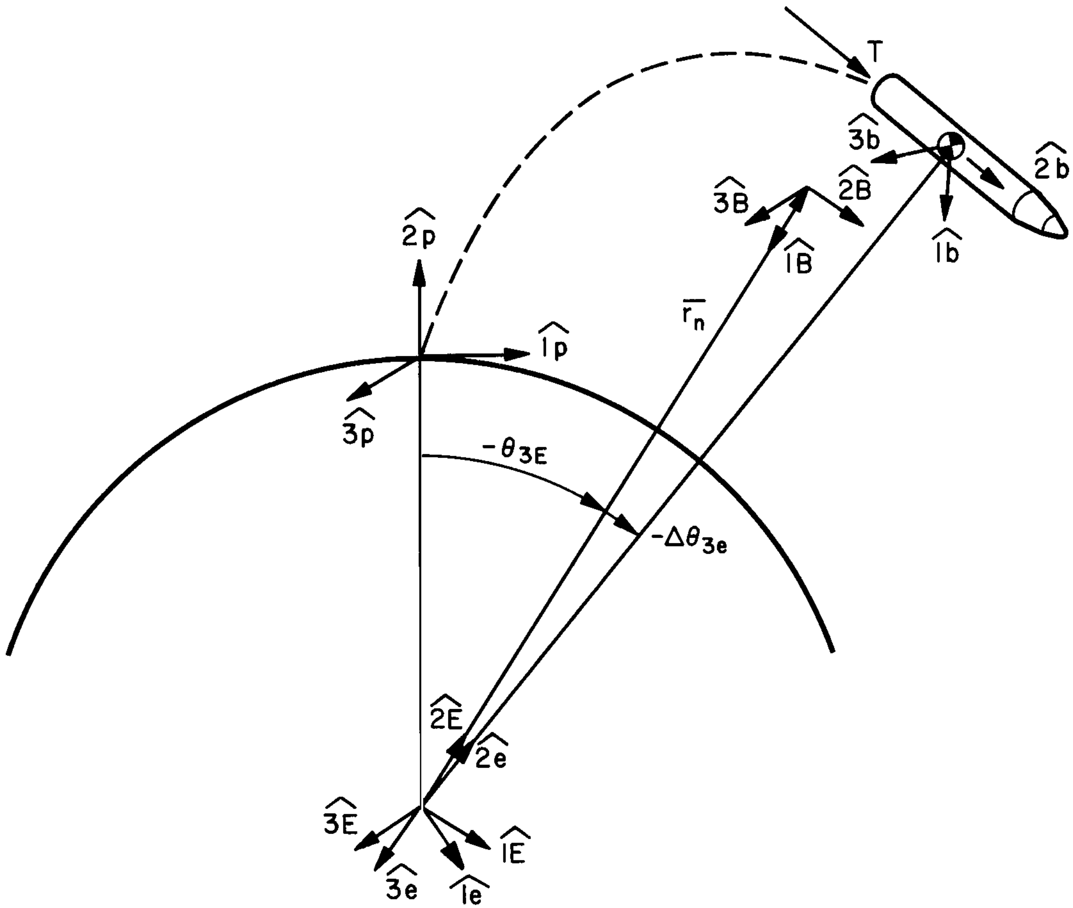


Figure 18.- Sketch of axes systems.

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—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

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