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FLIGHT INVESTIGATION OF MANUAL AND AUTOMATIC VTOL DECELERATING INSTRUMENT APPROACHES AND LANDINGS

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16. Abstract

A flight investigation was undertaken to study the problems associated with manual and automatic control of steep, decelerating instrument approaches and landings under simulated instrument conditions. The study was conducted with a research helicopter equipped with a three-cue flight-director indicator. The scope of the investigation included variations in the flight-director control laws, glide-path angle, deceleration profile, and control response characteristics.

Investigation of the automatic-control problem resulted in the first automated approach and landing to a predetermined spot ever accomplished with a helicopter. Although well-controlled approaches and landings could be performed manually with the flight-director concept, pilot comments indicated the need for a better display which would more effectively integrate command and situation information.

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SUMMARY

The operation of vertical take-off and landing (VTOL) aircraft in the city-center environment will require complex landing-approach trajectories that insure adequate clearance from other traffic and obstructions and provide the most direct routing for efficient operations. As part of a larger program directed toward developing the technology base needed in establishing system design concepts and operating procedures for such VTOL transportation systems, a flight investigation was undertaken to study the problems associated with manual and automatic control of steep, decelerating instrument approaches and landings under simulated instrument conditions. The study was conducted with a research helicopter which was equipped with a three-cue flight director driven by control laws developed and refined during the course of the program.

The scope of the investigation included variations in glide-path angle, deceleration profile, and control response characteristics. In addition, three different methods for obtaining position and velocity information were investigated.

The automatic-control investigation resulted in the first automated approach and landing to a predetermined spot ever accomplished with a helicopter. Although well-controlled approaches and landings could be performed manually with the flight-director concept, pilot comments indicated the need for a better display which would more effectively integrate command and situation information.

INTRODUCTION

The operation of vertical take-off and landing (VTOL) aircraft in the city-center environment will require complex landing approach trajectories that insure adequate clearance from other traffic and obstructions and provide the most direct routing for efficient operations. It has become increasingly evident from previous research that coping with trajectories of this type will require both significant improvements in handling qualities through the development and application of advanced control and display technology and operating techniques that rely heavily on automation.

Many years of intensive research and development have led to certification of several conventional take-off and landing (CTOL) aircraft to perform automatic landings. Therefore, it is reasonable to attempt to apply CTOL automatic-landing technology to the VTOL problem. There are, however, many fundamental differences in both the vehicles and their respective terminal environments that preclude direct application of CTOL technology to the VTOL control and display areas. These differences include the VTOL requirement to fly steep, decelerating approaches to a hover landing and to control a number of degrees of freedom that are more numerous and more complex than those for CTOL, or even short take-off and landing (STOL), aircraft. Furthermore, the unique characteristics of the VTOL aircraft present opportunities for utilizing yet-undeveloped operating techniques for effectively dealing with adverse wind conditions and interfacing with the air-traffic control system.

In order to develop the navigation, guidance, control, and flight-management technology base needed in establishing system design concepts and operating procedures for VTOL short-haul transportation systems, the Langley Research Center has initiated a VTOL Approach and Landing Technology (VALT) Program. As part of this effort, a flight investigation of the steep, decelerating, instrument approach task was undertaken which included both manually controlled and automatically controlled approaches. The manual-control problem was investigated first, because of uncertainties in hardware signal characteristics, lack of experience in hardware failure modes, and the need for experimental development of the guidance and control laws. Additional rationale for this approach was based on the belief that the research to develop the flight-director control laws would be directly applicable to automation of the approach and landing. Subsequent investigation of the automatic-control problem resulted in the first automated approach and landing to a predetermined spot ever accomplished with a helicopter.

The scope of the investigation included variations in glide-path angle, deceleration profile, flight-director control laws, and control-response characteristics. In addition, three different methods for obtaining position and velocity information were investigated. The report will discuss the effect of these variations on the precision with which the approaches could be flown and, in the case of manually controlled approaches, their effect on pilot workload. Deficiencies encountered in the developed system will be discussed and areas requiring additional research will be identified.

SYMBOLS

Values are given in both SI and U.S. Customary Units. The measurements and calculations were made in U.S. Customary Units.

a_Z output of the accelerometer along the body Z-axis, m/sec^2 (ft/sec^2)

g gravity constant, 9.8 m/sec^2 (32.2 ft/sec^2)

$K_1, K_2, \ldots, K_{13}$ system gains

s Laplacian variable

X, Y, Z aircraft body axes

x range, m (ft)

x_o range at which glide-path capture is initiated, m (ft)

y cross range, m (ft)

z altitude, m (ft)

γ glide-path angle, deg

δ_p power (collective) control deflection, cm (in.)

δ_X roll control deflection, cm (in.)

δ_Y pitch control deflection, cm (in.)

δ_Z yaw control deflection, cm (in.)

θ pitch attitude, positive nose upward, deg

τ time constant, sec

ϕ roll attitude, positive right wing down, deg

Subscripts:

c commanded parameters

e difference between actual value and desired value of parameter (i.e., error)

A dot over a symbol indicates a derivative with respect to time.

DESCRIPTION OF EQUIPMENT

General Description

Figure 1 shows the major elements of the research system. Briefly, the tracking radar determined the position of the aircraft relative to the desired touchdown point in Cartesian coordinates. This information was telemetered to the helicopter and fed to the navigation computer where it was mixed with onboard data to derive a "best" estimate of position and rate of change of position. The position information was displayed in the cockpit and also routed to the guidance computer along with the position rate information. The guidance computer calculated deviations from a preprogramed approach path and velocity profile, generated the flight-director commands, and, in the case of automatic approaches, generated the control-system inputs. The control computer accepted inputs from either the guidance computer or the evaluation pilot's controls and, as described in a subsequent section, forced the helicopter to respond in a prescribed manner by means of a high-gain-model following technique.

Test Helicopter

The research helicopter shown in figure 2 served as the test vehicle. The physical characteristics of the helicopter are given in reference 1. The control system of this aircraft was modified by removing the mechanical linkages connecting the right-hand set of controls (center stick, pedals, and collective stick) to the basic-ship system. The position of each control was sensed electrically and routed to onboard computers for processing. A trimmable force-feel system provides linear force gradients of 1.8 N/cm (1 lb/in.) in pitch and roll and 8.8 N/cm (5 lb/in.) in yaw. Breakout and friction were negligible. The power (collective) control is equipped with an adjustable friction device. A full-authority electrohydraulic actuator is installed in each controlled degree of freedom (pitch, roll, yaw, and vertical). The actuators are installed in parallel with the safety pilot's controls, which are unaltered, so that his controls follow the control-surface motions resulting from electrical inputs. Either pilot may disengage the system which returns control to the safety pilot. In the present investigation the onboard computers were used to implement a high-gain attitude-command control system in pitch and roll which yielded a second-order attitude response to pilot-control imputs. The pitch- and roll-response parameters, which were altered somewhat during the tests, are given in table I.

For yaw control the pilot could select either a turn-following or heading-hold mode. The turn-following mode provided automatic coordination for roll-initiated turns (pedal fixed). The rudder pedals could be utilized to produce intentional sideslips. The heading-hold mode forced the aircraft to maintain a fixed magnetic heading. In this mode the rudder pedals could be used to change the reference heading according to the first-order response characteristics given in table I. A ±0.63 cm (±0.25 in.) deadband was employed in this mode to prevent inadvertent rudder-pedal inputs.

The manner in which this system was implemented, described in reference 2, provided a constant control-response characteristic throughout the flight regime and heavily suppressed the angular response of the vehicle to external disturbances and trim changes.

Stability augmentation was not provided for the vertical translational degree of freedom. The unaugmented aircraft has a moderate amount of normal-velocity damping, however, and pilot-control inputs result in a steady-state vertical rate within a reasonably short time. The main drawback associated with the unaugmented vertical degree-of-freedom characteristics was that the pilot had to compensate for the power-required curve of the vehicle. The approximate first-order response characteristics of the vertical degree of freedom are given in table I.

Cockpit Displays

Figure 3 is a photograph of the evaluation pilot's instrument panel. The engine instruments, pitch- and roll-attitude indicator, needle-ball, airspeed, barometric altitude, and instantaneous vertical-speed indicator (IVSI) were all driven from standard sources. The remaining indicators, the command needles and deviation needles on the attitude-director indicator (ADI), moving map, radar altimeter, power-control position indicator and heading-hold light, were driven by the onboard computer system. Details of these displays are given in the following sections.

Attitude-director indicator.- Figure 4 is a closeup view of the ADI which was set up to display the following information:

Pitch and roll attitude
Turn rate and lateral acceleration (needle and ball)
Three flight-director commands
Glide-path deviation
Cross-range deviation
Altitude (rising runway)

As shown in figure 4, the display provides pitch, roll, and power commands to the pilot. The control laws are designed so that the aircraft will maintain or return to the desired approach trajectory if the command needles are kept centered (zeroed). Unlike the typical flight-director systems for fixed-wing aircraft, the VTOL flight-director system requires a third cue, the power-command cue, for height control. The pitch-command cue is used for control of speed, as pitch attitude in this case corresponds to thrust-vector tilt. For the VTOL task, which includes deceleration to a precision hover over the landing spot, inertial rather than airmass velocity information must be used for control of speed during the final approach phase. Through the roll-command bar, the aircraft is commanded to bank which results in either a turn or a sideslip, depending on the heading mode selected, to maintain the desired ground track. The control laws, employed to generate the flight-director commands, are discussed in a subsequent section.

The glide-path and cross-range deviation indicators presented vertical (altitude) and lateral displacement errors from the desired trajectory. The errors were generated within the guidance computer in terms of actual displacement, as opposed to angular error; and, hence, the sensitivity of the needles remained constant throughout the entire approach. Full-scale deflections for the glide-path and cross-range deviation indicators were ±15.2 m (±50 ft) and ±45.7 m (±150 ft), respectively.

The final 30.5 m (100 ft) of altitude was displayed on a "rising runway" indicator which was biased down out of view at higher altitudes. Touchdown, or zero altitude, was indicated when the "rising runway" just touched the bottom of the aircraft-shaped reference symbol.

Moving map.- The moving-map indicator shown in figure 5 provided heading, range, and cross-range information. Although two heading modes were available, a north-up mode and a heading-up mode, only the heading-up mode was employed during the present investigation. In this mode the runway diagram and compass rose rotate relative to the fixed reference at the top of the display. The magnetic heading, therefore, appeared at the top of the display, whereas the crab angle was indicated by the angle between the runway heading and the lubber line extending through the fixed aircraft symbol at the center of the viewing screen.

Range and cross-range information was indicated by the position of the moving runway diagram relative to the fixed aircraft symbol. Three maps were employed to provide runway diagrams at three scales, 480 m/cm (4000 ft/in.), 36 m/cm (300 ft/in.), and 12 m/cm (100 ft/in.). Automatic switching between maps occurred at ranges of 2438 m (8000 ft) and 610 m (2000 ft).

Radar altimeter.- The radar altimeter displayed altitude from 0 to 366 m (0 to 1200 ft) relative to the landing pad. A nonlinear scale was employed to permit good resolution at low altitudes and coverage up to 366 m (1200 ft). The instrument was driven by the altitude signal generated in the navigation computer.

Data System

Data were recorded at the radar ground station and onboard the aircraft. The ground base data consisted of plots of altitude against range, cross range against range, and range rate (i.e., x-component of range rate) against range. Onboard the aircraft, data were recorded on magnetic tape in either a continuous or sampled (20 samples per second) format as shown in table II.

Photographic coverage was obtained for selected approaches during the research flights. Three cameras were utilized, a forward-looking nose camera, a cockpit-mounted camera, and a ground-based camera. The photographic coverage, particularly the onboard cameras, provided an independent source from which both the actual and displayed touchdown point could be obtained. These data were used, when available, to correlate onboard data with the ground-base data.

TASK DESCRIPTION

The primary task for the investigation was a steep, decelerating, simulated instrument approach to a touchdown as illustrated in figure 6. Instrument flight rule (IFR) conditions were simulated by means of a conventional, helmet-mounted IFR hood and by covering the lower windows in the cockpit with curtains. For manually controlled approaches, the evaluation pilot was given control of the aircraft on the downwind leg at an altitude of 229 m (750 ft), an indicated airspeed (KIAS) of 45 knots, and at a range of about 4572 m (15 000 ft). At this point, the power command on the flight director was activated which commanded a constant 229 m (750 ft) altitude until glide-path intercept. The pilot then executed an inbound turn by using situation information until his heading was within 90^{o} of the final approach heading. The roll command then became usable and the pilot completed capturing the approach center line by following the flight-director command. Following this maneuver, the pitch command was activated to command the pilot to slow down (or speed up, as the case may be) to a range rate of 45 knots. From this point on, the flight director provided continuous commands all the way to touchdown. For automatic approaches, the coupler was engaged at a range of about 3048 m (10 000 ft), an altitude of 228 m (750 ft), an airspeed of 45 knots, and with the heading within $\pm 20^{o}$ of the approach center line. The task terminated when the main landing gear made ground contact. A variation on the primary task for both manual and automatic approaches was to omit the vertical letdown and terminate the task in the hover.

A constant-speed, manual approach task was employed during the preliminary evaluation of glide-path angle effects. For this task, combinations of glide-path angle and rate of descent were investigated as shown in figure 7. For these tests, the flight director was programed to command a specific range rate (ground speed) commensurate with the combination of glide-path angle and rate of descent being investigated. The task commenced on center line about 30 seconds prior to glide-path intercept and terminated at an altitude of 30.5 to 45.7 m (100 to 150 ft) where the safety pilot took control.

EVOLUTION OF THE SYSTEM

Control Characteristics

As noted in reference 3, the original control characteristics in pitch and roll (see table I) were selected on the basis of motion base simulator results. Although these characteristics were judged satisfactory for the decelerating approach task, the pilots objected to the initial response to control inputs and stated that it was much too abrupt. A brief visual flight investigation was conducted to explore the problem and resulted in the modified set of attitude system gains given in table I. Subsequent tests indicated that the modified control-response characteristics did not appear to have any significant effect on either pilot workload or performance during the decelerating approach task. There was a noticeable improvement in the ride qualities, however, due to the absence of abrupt attitude changes.

Whether attitude command is the most suitable control concept for the task at hand is open to question. The primary advantage of the concept, according to pilot comments, is that the pilot can establish a stick trim position corresponding to the nominal pitch and roll attitudes required for a steady-state condition such as hovering flight. Then, the pilot can control the aircraft about this point with the knowledge that he can reestablish the nominal trim attitudes by simply relaxing any stick forces he is holding. Unfortunately, this same feature becomes a disadvantage during other phases of the approach. In capturing the approach center line, for example, constant bank turns are generally flown which require a constant roll-control input. The pilots object to having to hold the control displaced against the stick centering force, but at the same time they are reluctant to retrim the control since they would lose their wings-level reference.

The yaw-response characteristics were satisfactory from the beginning and, consequently, were not altered. The pilots developed a technique of switching to the heading-hold mode shortly after capturing the approach center line and made subsequent lateral corrections by sideslipping the aircraft. (This technique was also preferred by one pilot in the study reported in ref. 4.) As noted in reference 2, one of the pilots, who has a broad base of experience in display research for low-speed instrument flight, commented that the cross-range tracking performance and the associated level of pilot effort required with this system were superior to any system he had flown.

The response in the vertical degree of freedom, which was unaugmented, did not pose any handling-qualities problems in terms of the pilot's high-frequency control task. However, the trim shift associated with the aircraft power-required curve was the source of a major problem throughout the investigation. Early in the program a "fall out" tendency was encountered at a range of about 152.4 m (500 ft) (during the deceleration) wherein the aircraft invariably ended up low on the glide path. It was determined that the power control law did not provide sufficient lead and also tended to mask the fall out. The control law was modified and subsequent tests showed that the fall out had been reduced to a satisfactory level provided the command needle was kept centered. The pilot still had to compensate for the trim shift, however, which was approximately 5 cm (2 in.) during the deceleration. Unfortunately, this situation occurred at the most critical stage of the approach, just prior to the hover, when the pilot's workload was at its peak. It appears, therefore, that some form of augmentation is warranted in the vertical degree of freedom to remove or reduce the trim requirement in this axis.

Navigation System

Over the course of the flight-test program, three different airborne navigation-system configurations were employed to obtain aircraft positions and velocities in the runway coordinate frame.

Each configuration relied on the ground-based precision radar to provide position information to the aircraft via telemetry. The airborne navigation system, which receives this position information, must ultimately provide both position and velocity information for guidance. The airborne-system configurations which have been used include the following:

(1) Analog-computer differentiation and filtering

(2) An inertial navigation system with periodic radar-position updates

(3) Inertial smoothing system whereby radar-position information is continuously mixed with acceleration information obtained from low-cost airborne sensors

Precision radar.- Aircraft position was sensed by a precision tracking-radar system, located at the Wallops Flight Test Center, where the flight tests were performed. The position of the aircraft is sensed directly in terms of slant range and azimuth and elevation angles of the radar antenna. This information is transformed into rectangular coordinates in the runway reference frame and transmitted to the aircraft by means of telemetry.

The radar is K-band, with an antenna beamwidth of approximately 0.5°. A passive reflector has been mounted on the nose of the aircraft providing a specific point of high energy return. The limits of the radar tracking antenna are 30° in elevation and $\pm 45^{\circ}$ in azimuth. The accuracy of the radar as specified by the manufacturer is 0.02° for the azimuth and elevation angles and 3 m (10 ft) or 1 percent (whichever is greater) for slant range.

Airborne navigation equipment.- As reported in reference 2, the first series of tests were conducted with a 10-volt analog computer used to filter the radar-position information because of radar and telemetry noise. The telemetry unit was an FM narrow-band unit with ±10-volt discriminator outputs. Velocity information was obtained by approximate differentiation of the position signals. This approach, even with maximum tolerable filtering, resulted in considerable noise from velocity signals which, in turn, caused the flight-director command needles to "jitter." In the pilot evaluation, as discussed in reference 2, the random needle fluctuations were found to be quite objectionable.

An inertial navigation system with periodic radar-position updates was used during most of the steep-angle approach work discussed in this report. The system (a modified Gemini spacecraft system) was modified so that the digital computer could update the navigation outputs using GSN-5 radar-position data. The GSN-5 signals were sampled and digitized at a ground station at 1-second intervals and transmitted to the aircraft by means of a digital telemetry link. The position and velocity outputs from the digital computer were converted to analog form, and then routed to the onboard analog computers. In general, the signal outputs were of sufficiently high quality in terms of both accuracy and noise level to be used for the flight-director display application.

More recently, a system developed at the Langley Research Center has been used to provide accurate position and velocity outputs with low-noise content by using an inertial smoothing technique. A detailed description of this system and its development are presented in reference 5. This approach involved a second-order complementary filter with inertial-acceleration inputs as well as the radar-position inputs. The inertial-acceleration inputs provided high-frequency position and velocity information, and the radar-position inputs provided low-frequency, long-term position and velocity information. Aircraft instrumentation, including relatively low-cost attitude reference gyros and body-mounted accelerometers, were used to provide the inertial-acceleration information. Since only the high-frequency content of the onboard inertial-acceleration information was relied on in this application, the relatively low-quality inertial sensors were considered adequate. This system has also provided high-quality signals and has been used for the constant-attitude deceleration profile tests and the automatic approach tests, described herein.

Flight-Director System

General background.- The VTOL flight-director system was developed at the Langley Research Center through analysis, fixed-base piloted simulation, and flight tests. A more detailed presentation of the analytical approach used in optimizing the flight-director control laws is given in reference 6. It was found that development of a flight-director command was more difficult than development of an automatic control law, because, not only must a desirable aircraft response be achieved when the command bar is kept centered, but also the dynamic response of the command bar to the pilot's input must be such that the pilot can keep the bar centered with a minimum of effort and attention. Briefly, the design of the flight-director control law was considered in two parts. The main part was the fundamental automatic control law, which was selected to yield a critically damped system response with an analytical pilot model which assumed an acceptable flight-director response to pilot input. The second part was to shape the automatic control law with the additional quickening terms so that the resultant flight-director command bar would, in fact, provide a satisfactory command for the pilot to follow.

In developing the flight-director control laws, it was assumed that the angle between the aircraft longitudinal axis and the approach center line (that is, the crab angle) would be small. This assumption resulted in prohibiting the use of crab angles in excess of about 15°.

Reference 2 suggests several modifications to the flight director which were subsequently incorporated. The refined flight-director control laws for each of the pitch, roll, and power-command cues are presented in figures 8, 9, and 10, respectively.

Pitch command.- The pitch-command implementation, shown in figure 8, utilized a function generator (see fig. 8) to provide a range-rate command $\dot{x}_c$ as a function of range x. In general, all approaches have been flown from an initial constant speed of 45 knots. The deceleration portion of the profile of range rate against range would commence within 610 m (2000 ft) of the landing spot. The deceleration profile itself was found to be a significant factor and the various profiles which were flown will be discussed in detail in a later section.

A feed-forward attitude command $\theta_{c,lead}$, generated as a function of range, was found to be necessary for the deceleration task. This lead term was constructed by taking the nominal pitch attitude that would be required for the maneuver and shifting the nominal-pitch-attitude curve, point by point, by an incremental value of range equivalent to 2 seconds of lead. The feed-forward attitude command resulted in smooth transitions into the deceleration and into the hover while helping to keep speed deviations small during this dynamic maneuver.

Both proportional and integral control are used in nulling errors in speed $\dot{x}_e$. The integrator is needed to eliminate steady-state error which would occur if wind conditions demanded a pitch attitude other than the nominal attitude to hold the range rate. To prevent saturation, this integration was limited to $\pm 2.5^{\circ}$ which was considered adequate for wind conditions up to 15 knots. An absolute pitch-attitude command limit was incorporated to keep the pitch command no more than 5° greater than the hover attitude nor less than 5° below the nominal pitch attitude for 45 knots.

Roll command.- The roll-command implementation is shown in figure 9. The cross-range error y is limited to command a maximum closure rate of 9.6 m/sec (31.4 ft/sec) so that prior to acquiring the approach center line, the ground track intercept angle will be 22.5°, assuming that the range rate is 45 knots, as commanded. In the heading-hold mode, where a bank angle is commanded to counteract a crosswind, integral control is implemented to prevent steady-state errors. The integration is limited to prevent saturation. An integration limit of $\pm 5^{\circ}$ was selected as adequate to hold position without any standoff in crosswinds up to 15 knots. Also, from pilot comments, a steady bank angle of 5° appeared to be an acceptable upper limit. In turn-following, integral compensation is not needed as the aircraft will establish a new heading through bank-angle inputs and establish the proper crab angle to hold ground track along the approach center line. In switching from heading hold to turn-following, a first-order circuit is used to wash out the integral term gradually to zero, and thereby eliminate a possible abrupt change in the roll command. The roll command was limited to $\pm 10^{\circ}$.

Power command.- The power-command logic resulted in an altitude profile as a function of range. The profile included a constant-altitude segment (where $x > x_o$ in fig. 10), an asymptotic glide-path capture ($x \leqq x_o$), and a constant glide-path angle γ with an altitude offset causing the glide path to pass through a point 15.2 m (50 ft) above the landing spot (i.e., $z_{bias} = 15.2$ m (50 ft)). Glide-path capture was initiated at a point in range x_o where the distance below the glide path would nominally command zero rate of climb according to the asymptotic control law. The glide-path logic would be engaged at that time, whereupon a gradual reduction in power would be commanded to maintain the asymptotic closure to the glide path. This method resulted in smooth glide-path captures without any abrupt power commands and without any overshooting of the glide path. As indicated in figure 10, the range switching point was a function of several approach variables. The vertical rate which could be commanded was limited to a maximum rate of climb of 2.54 m/sec (500 ft/min) and a maximum rate of descent of 2.54 m/sec (500 ft/min) greater than the nominal rate of descent required to remain on the profile. A shaped normal accelerometer signal was used to provide the pilot with quickened information concerning the effects of his own control inputs as well as the effects of disturbances.

Vertical letdown.- The vertical letdown logic shown in figure 11 was implemented to allow a vertical descent from a 15.2-m (50 ft) hover to touchdown. This logic is activated by manually closing the vertical letdown logic switches. From a 15.2-m (50 ft) hover, a vertical-descent rate is commanded as a function of altitude resulting in a commanded touchdown velocity of 0.46 m/sec (1.5 ft/sec). The rate-of-descent profile is based on several factors. First, the time to make the letdown is kept as short as possible, with the constraint that the pilot has enough time to be aware of his status during the letdown. The primary concern is not only with maintaining the proper rate of descent with altitude but also with keeping longitudinal and lateral velocities near zero, especially at touchdown. This velocity profile has two advantages. First, it allows a higher rate of descent during the initial portion of the letdown, which minimizes the total time to make the letdown. Second, the reduction in the descent rate demands a gradual increase in power as altitude diminishes. This was considered beneficial as the pilot had indicated that he would be reluctant to respond to any reduce power commands during the final stage of the descent. Lastly, the vertical velocity at touchdown is such that it is gentle, yet firm enough to be a positive indication that touchdown has occurred. The first rate-of-descent profile, which was attempted, began at 1.2 m/sec (4 ft/sec) at 15.2 m (50 ft) and resulted in a commanded touchdown velocity of 0.3 m/sec (1 ft/sec). This profile was judged to be too slow overall and touchdown occurred so gently that it was difficult to tell when it had occurred. The rate-of-descent profile shown in figure 11 was found to be more acceptable.

Principal refinements.- The flight-director control laws used previously, reported in references 2 and 3, were based on the concept of commanding pilot control position. This concept required the feedback of the pilot's control position through the flight director. The feedback of the pilot's control position resulted in an immediate response of the flight director to the pilot input which was judged to be unnecessarily sensitive. The pilots commented that the command was usable, but a less sensitive command was preferred. It was found that the pilots preferred an integrator response to their input rather than a gain or amplifier response. Therefore, the concept of commanding pilot control position was abandoned, and, instead, the concept of commanding the basic inner-loop variables, pitch and roll attitude, and rate of descent was adopted. In order to achieve a desirable integrator response, the commands were quickened by adding pitch and roll rate and filtered normal-acceleration terms.

During the course of the investigation, some of the test pilots complained about the absence of flight-director command limiting. That is, large errors in flight path would result in very large pitch, roll, and rate-of-descent commands, and the pilots would have to supply their own limiting by not exceeding what they judged to be a large enough attitude or rate of descent. Pitch-attitude limiting was introduced so that the maximum pitch-up command was 5° above the hover attitude and the maximum pitch-down command was 9° below the pitch attitude for hover. The pitch attitude for an airspeed of 45 knots was approximately 4° nose-down from the attitude for hover. In roll, command limits of $\pm 10^{\circ}$ were introduced. For the power command, variable limits were used which restricted the resultant vertical velocity to a rate of climb of 2.54 m/sec (500 ft/min) and to a rate of descent 2.54 m/sec (500 ft/min) greater than the nominal rate of descent. Command limiting resulted in less pilot apprehension when moderately large errors existed since the flight-director commands could be centered without excessively large attitudes or rates of descent. However, on the whole, no great improvement in the manual-approach tracking performance resulted from the introduction of command limiting.

Automatic Approach and Landing Coupler

The automatic approach and landing coupler was achieved by mating the basic flight-director control laws, including limiting, with the control augmentation system. The overall system configuration by which either manual or automatic approaches could be selected was shown in figure 1. As pointed out previously, the pitch, roll, and yaw degrees of freedom were augmented by using a high-gain, model-following control technique. As indicated in figure 12, for an automatic approach, pitch and roll command signals in the guidance computer replaced the pilot-control input signals. The heading system, even for manual approaches, could be considered to be an automatic system, not requiring pedal inputs except to change heading in the heading-hold mode. Therefore, no changes were necessary in the yaw control system for automatic approaches. For the vertical degree of freedom, the $\dot{z}_e$ signal from the power-command control law was used to drive the basic aircraft power (collective) control.

Integral compensation was employed to eliminate steady-state errors in glide path. The gains for automatic control of power shown in figure 12 were obtained through closed-loop analysis and were selected so that the resulting bandwidth of the automatic system in power control was 50 percent higher than that assumed for manual control. This was done to minimize transient effects resulting from rapid changes in speed which created rapid trim changes in power. Because of the additional quickening term provided to the pilot through the flight-director display, the performance of the automatic or the manual system was approximately the same throughout the deceleration phase.

Glide-Path Angle

Early in the program, a series of constant-speed, steep-angle approaches were flown to determine the maximum feasible angles which could be employed during subsequent decelerating approach studies. The evaluation was restricted to only the steady-state (on glide-path) portion of the approach; it did not include a low-altitude "breakout" phase. The investigation was conducted with the early versions of the control and display systems utilizing the inertial navigation system.

The test conditions, shown in figure 7, covered glide-path angles from 9° to 60° and, in most cases, three nominal rates of descent: 2.5 m/sec (500 ft/min), 3.8 m/sec (750 ft/min), and 5.1 m/sec (1000 ft/min). The approach speeds corresponding to these conditions ranged from 5 to 60 knots. A minimum of 6 approaches were flown at each glide-path angle up to and including 25° and one approach at each of the higher angles. It should be emphasized that the tests were conducted only in favorable wind conditions. Specifically, the winds ranged from calm to quartering headwinds ($\pm 40^{\circ}$) up to 10 knots.

Approach tracks for several representative runs are presented in figure 13 on a plot of altitude as a function of range. The approach tracks shown for each angle include various combinations of rate of descent and approach speed. The number and letter shown by each angle indicates the number of runs at the designated rate of descent included in the plot.

The pilots indicated that variations in the nominal rate of descent (and, hence, approach speed) at a given angle had no significant effect on the control task or approach performance. The results of the glide-path-angle investigation indicated that angles up to 25° would have been completely acceptable had not the aircraft vibrated excessively. The shaking (vibration), which is usually encountered in helicopters flying low-speed approaches, was present for all angles over 6° and increased as speed was reduced (steeper angles). In some cases, the vibration level was so high it caused premature termination of the run.

The evaluation pilots all agreed that there were no significant differences in the control task for angles up to 25°. In addition, the accuracy of the approaches appears to be independent of the approach angle if one neglects the errors associated with capturing the glide path.

The 30° approach angle was found to be an upper limit during these tests. Although there were indications that the control task was more difficult but still acceptable, the aircraft vibration was too severe to obtain a reasonable evaluation and testing was terminated after one approach.

Although it would have been possible to utilize a higher nominal rate of descent and hence a higher approach speed to circumvent the vibration problem, previous studies have shown that nominal rates of descent much in excess of 5.1 m/sec (1000 ft/min) are unacceptable at low altitudes from the pilot's standpoint. The investigation reported in reference 4, for example, concluded that based on rate of descent and glide-path intercept considerations, 30° was an upper limit for steep angle approaches.

Only one approach was attempted at each of the steeper angles (45° and 60° at a rate of descent of 5.1 m/sec (1000 ft/min)), and neither was successful. In addition to the vibration, the pilots could not control the rate of descent. This was probably caused by the helicopter entering the vortex ring state, at least intermittently, during the approaches.

Overall, the pilots indicated that a nominal rate of descent of 5.1 m/sec (1000 ft/min) was acceptable provided the altitude was above 61 to 91 m (200 to 300 ft). They indicated that having low power (high rate of descent) near the ground was very uncomfortable.

Furthermore, they found it more difficult to correct down to the glide path than up to it; there was a definite reluctance to follow reduce power commands below 122 to 152 m (400 to 500 ft).

Deceleration Profile

The first decelerating approaches employed a linear deceleration profile (i.e., the range rate was a linear function of range) as shown in figure 14. This profile was selected primarily because of its simplicity and ease of mechanization. During the initial flight tests, however, it became readily apparent that the deceleration requirements with respect to the time domain were totally unacceptable. Following a large initial deceleration, the level drops too low and, as a result, a great amount of time is spent at low speed approaching the intended touchdown point. This condition gives the pilot the impression that he is being commanded to hover well short of the touchdown point.

The next profile investigated was the constant-deceleration profile illustrated in figure 14. It should be noted that this and the subsequent profile terminated in a linear deceleration at the very end. This linear segment was required to provide a smooth transition to the hover. The range at which the profile became linear depended on the deceleration level programed (0.06g, 0.08g, etc.) and occurred between a range of 15.2 and 61 m (50 and 200 ft). The constant-deceleration profile was a vast improvement over the linear profile and served as the basis for the investigation reported in reference 2. The constant-deceleration profile had one drawback, however, which was inherent in the design of the profile. In order to maintain a constant deceleration level the pitch attitude of the aircraft had to be continuously increased during the deceleration to compensate for the loss of aerodynamic drag. The pilots objected to this characteristic in that it left them in a nose-high—low-power condition when coming into the hover.

The most recent profile, the constant-attitude deceleration profile, shown in figure 14 was designed to eliminate the problem noted previously. To date, only one version of this profile has been tested which uses a nose-up attitude 2° above the hover angle. This attitude yields a deceleration level varying from 0.1g at range to 0.04g at the transition to the linear hover segment.

Flight-test results indicated that the constant-attitude profile provided an easier transition to the hover than the constant-deceleration profile.

REFINED-SYSTEM CONFIGURATION

Manual-Approach Results

The refined system included inertially smoothed radar-position information, the modified (milder) attitude command-system response, and the updated version of the flight-director control laws utilizing the constant-attitude deceleration profile. Overall, more than 20 manual approaches were performed with the refined system using glide-path angles of 6° and 15°. Wind conditions ranged from light headwinds to direct crosswinds gusting up to 18 knots.

Tracking performance.- The tracking performance obtained under favorable wind conditions (headwinds or light crosswinds) is illustrated in figures 15 and 16 for glide-path angles of 6° and 15°, respectively. Figures 15(a) and 16 present approach tracks on plots of range rate against range, cross range against range, and altitude against range for the final 1524 m (5000 ft) of the approach. The dashed lines represent the desired level of the parameters as a function of range. In addition, wind direction and magnitude are indicated by a "bull's-eye" symbol such as the one shown in figure 15(a). In this case, the winds were from the right (variable) at about 12 to 14 knots. Each ring of the "bull's-eye" represents a windspeed of 5 knots.

The five 6° approaches shown in figure 15(a) illustrate the consistency and accuracy which could be obtained when the pilots kept the flight-director commands reasonably centered. A comparison between these approach tracks and those of reference 2 indicates that the effect of the system refinements were not evident in the glide-path and cross-range tracking performance. On the other hand, the range-rate tracking performance showed a definite improvement. This improvement was attributed primarily to refinements in both the flight-director control laws and the navigation system.

The time history shown in figure 15(b) illustrates the control activity, flight-director commands, aircraft motions, and deviations of the tracked parameters associated with one of the 6° approaches shown in figure 15(a). The time history shown in figure 15(b) is similar to the one presented in figure 9 of reference 2 with two notable exceptions. First, the pitch and roll control inputs are about four to five times larger than the inputs shown in the previous time history. This difference is attributed primarily to the fact that the control sensitivities employed during the present tests are only one-third those of the previous study. In addition, the attitude sensitivities were also somewhat lower. It is interesting to note that despite having to use larger control inputs to null the flight-director commands, the pilots did not object to the reduced control sensitivities. The second exception is that the flight-director power-command variations are considerably larger than those shown in reference 2. The reason for this is that the power command logic associated with the time history reported earlier did not include a normal-acceleration term. As pointed out in reference 2, a normal-accelerometer signal was employed to cure a glide-path dropout problem, which it did, but at the expense of incurring more frequent flight-director commands. In fact, the pilots claimed they could not keep the command centered without a considerable effort so they applied control only to keep the variations about zero. As indicated in figure 15(b), this results in approximately the same control activity as experienced with the previous control law.

Figure 16 illustrates two typical 15° approaches. The tracking performance for the 15° approaches was, in general, similar to the 6° case with the exception of a slight tendency for the pilot to remain on the high side of the glide path during the 15° approaches. The pilots indicated that, because the nominal rate of descent of 6.2 m/sec (1222 ft/min) was high down to a 137-m (450 ft) altitude, they were very reluctant to reduce power, especially at the lower altitudes. Therefore, once they got above the glide path they tended to stay there. Also, for the approaches shown in figure 16, this situation was aggravated because the aircraft speed was higher than commanded which required a rate of descent even higher than nominal to remain on the glide path.

Due to component limitations, the small-angle assumption used in the flight-director control laws (differences between aircraft heading and the approach center-line heading) was not modified during the course of the investigation. This limited the crab angles the pilot could employ to counteract crosswinds to about 15°. The effect of this restriction was discussed briefly in references 2 and 3. However, subsequent data illustrate the complexity of the problem. Figure 17 shows four 6° approaches flown in direct crosswinds of 10 to 13 knots. During these approaches, the pilot established a crab angle of from 15° to 20° during the initial part of the approach to compensate for the crosswind. As he entered the deceleration, the effectiveness of his correction was reduced, and the aircraft was blown off course. The pilot, therefore, had to compensate by banking the aircraft and sideslipping back to the center line. The maneuver required a maximum effort from the pilot and in one of the cases shown he failed to compensate for the drift because of a minor distraction and an aborted approach resulted. An obvious cure to this problem would be to make all approaches into the wind. However, this solution is not too practical from an operational standpoint, since usually there will be obstructions or other restrictions which will limit the usable approach directions. One alternate solution might be to command the aircraft to turn into the wind upon entering the deceleration phase.

Pilot acceptance of flight-director system.- In order to perform the instrument deceleration to a hover successfully, the pilot had to devote full attention to the flight-director commands. Even with the most refined flight-director control laws tested, the pilots spent most of their time trying to keep the three needles centered. This resulted in not having enough time to scan the other displays to obtain situation information. Thus, the pilot's role became essentially that of a servo during the more critical phases of the decelerating approach. The pilots objected to having to follow the commands without adequately knowing the approach status. In addition to this, the flight-director commands did not provide sufficient anticipation for the deceleration to a hover. The pilots commented that they thought the flight-director system was about as good as it could be; yet, for the task of decelerating to a hover, it was not adequate for operational use. It should be pointed out that, for the constant-speed portion of the task, the flight-director display was considered satisfactory. It appears that for the more demanding tasks, the basic limitations of the flight-director concept have been reached. As suggested in reference 2, a display which better integrates the situation and the command information is needed for such a task.

Automatic-Approach Results

Systems performance.- The performance of the automatic system is illustrated in figures 18 and 19 for glide-path angles of 6° and 15°, respectively. Part (a) of each figure presents the track of a typical approach on plots of range rate against range, cross range against range, and altitude against range for the final 1520 m (5000 ft) of the approach. The dashed lines represent the desired level of the parameters as a function of range. Part (b) of each figure shows the corresponding time histories of the flight-director commands, aircraft motion, range-rate error, and flight-path deviations.

As expected, the automatic system produced more accurate approaches than were obtained during the manual-control tests. The capability of the system to track the decelerating range-rate profile is indicated in figures 18 and 19. The maximum range-rate errors shown are small, about ± 3 knots. In addition, the tracking capability is, as expected, unaffected by the approach angle.

In both cases, range-rate errors are seen to occur about 46 m (150 ft) from the pad and are the result of the aircraft deceleration being somewhat high. The higher than normal deceleration may be attributed, in part, to the pitch-attitude lead compensation employed in the pitch-control loop. Since the lead term is based on a no-wind condition (i.e., airspeed equals range rate (ground speed)), there will be a tendency for the aircraft to decelerate too rapidly during approaches involving a headwind component such as for the two cases illustrated.

The flight-director deviations in pitch, which are indicative of how well the automatic system was compensating for deviations from the desired speed profile, seldom exceed ± 10 percent of full scale during either of the two approaches shown. In all cases, the deviations were only transient, this indicated that the automatic control loop was performing properly.

Pitching motions of the aircraft were similar for both approach angles. The angular rates were low: a maximum of $\pm 4^{\circ}$ per second, even during the pitch-up at the initiation of the deceleration. The maximum nose-up attitude encountered was 12° which was less than the 13° limit programed into the pitch coupler.

From referring to figures 18 and 19 it can be seen that the cross-range deviations about the desired track were within ±9 m (±30 ft) throughout the approach. As indicated in the figures, the crosswind components during each approach were 10 and 0 to 8 knots for the 6° and 15° glide-path angles, respectively. Roll-attitude variations were very low, only 2° to 3° during the entire approach. Since the approaches were flown in the heading-hold mode, yaw rates were insignificant and, consequently, are not shown.

When compared with pitch and roll, the flight-director deviations in the vertical degree of freedom are much higher in amplitude. This is due, in part, to the fact that the normal-acceleration term, which provides the nulling signal for the deviation and deviation-rate terms, was not included because of hardware limitations. In addition, the power command was more sensitive to deviation and deviation-rate errors than were the pitch and roll commands.

For both the 6° and 15° glide-path cases, there was a tendency for the aircraft to fall gradually below the glide path starting at a range of about 305 m (1000 ft). This dropout was caused by the aircraft power-required characteristics which necessitated approximately a 5-cm (2 in.) power-control (collective) trim change during the deceleration to a hover. As noted under the description of the approach and landing coupler, an integrator had been incorporated in the power-control loop to account for trim changes. However, the rate of the trim change together with the gains employed in the loop closure resulted in a dropout of 9.1 to 12.2 m (30 to 40 ft) which is considered unacceptable since it occurs close to the ground. Several solutions appear possible such as providing open-loop compensation, and/or increasing the gains employed in the loop closure. Aside from this transient condition, the maximum glide-path deviations shown never exceeded 6.1 m (20 ft) for either the 6° or 15° approach angles.

Automatic-approach monitoring.- During automatic approaches, the pilots were able to judge the performance of the automatic system by noting how well it was keeping the flight-director commands centered. Also, the pilots, relieved of the control task, were able to scan the situation displays and be more aware of the approach status throughout the entire approach. However, the pilot commented that with the display he would still have had difficulty knowing when to take control, if and when that became necessary.

CONCLUSIONS

Flight tests were conducted with a research helicopter to study the problems associated with manual and automatic control of steep, decelerating approaches to a hover and/or landing under simulated instrument conditions. Based on the results obtained, the following conclusions are drawn:

1. Refinements to the deceleration profile, navigation system, and flight-director control laws resulted in improved approach performance but did not significantly reduce the attendant pilot workload which is considered too high for operational use.

2. The navigation system, employing an inertial smoothing technique, provided accurate, low-noise position and velocity signals suitable for both manual and automatic approaches and landings.

3. Under the conditions flown, there were no significant differences in the control task or approach performance for glide-path angles up to and including 25°. Steeper angles were unflyable because of aircraft vibration and problems in controlling rate of descent.

4. The constant-attitude deceleration profile was the most acceptable profile tested. The linear-deceleration profile was unacceptable.

5. The flight-director display was considered satisfactory for the constant-speed portion of the task.

6. An improved display which provides integrated command and situation information is required for pilot acceptance of the manual decelerating approach and landing task.

7. Within the range tested, lower control sensitivities which required larger control inputs to null the flight-director commands did not adversely affect pilot workload.

8. The display concept was not satisfactory for monitoring fully automatic approaches.

Langley Research Center,
National Aeronautics and Space Administration,
Hampton, Va., March 6, 1974.

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4. Adams, Glen D.: V/STOL Approach System Steep Angle Flight Tests. Rep. No. RD-66-68, FAA, Jan. 1967. (Available from DDC as AD 646 236.)
5. Niessen, Frank R.: A Low-Cost Inertial Smoothing System for Landing Approach Guidance. NASA TN D-7271, 1973.
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TABLE I.- CONTROL-SYSTEM CHARACTERISTICS

	Original gains	Modified gains
Pitch and roll:		
Control power, rad/sec^2	2	1
Control sensitivity, $\frac{\text{rad/sec}^2}{\text{cm}}$ $\left(\frac{\text{rad/sec}^2}{\text{in.}}\right)$	0.24 (0.6)	0.08 (0.2)
Damping ratio	0.75	0.75
Undamped natural frequency, rad/sec	2.0	1.43
Attitude sensitivity, rad/cm (rad/in.)	0.06 (0.15)	0.04 (0.10)
Yaw:		
Heading-hold mode:		
Undamped natural frequency, rad/sec		2.0
Damping ratio		0.7
Maximum heading-rate capability,[a] rad/sec		0.8
Heading-rate control sensitivity,[a] $\frac{\text{rad/sec}}{\text{cm}}$ $\left(\frac{\text{rad/sec}}{\text{in.}}\right)$		0.14 (0.35)
Turn-following mode:		
Control power, rad/sec^2		0.25
Control sensitivity, $\frac{\text{rad/sec}^2}{\text{cm}}$ $\left(\frac{\text{rad/sec}^2}{\text{in.}}\right)$		0.08 (0.2)
Directional stability, $\frac{\text{rad/sec}^2}{\text{m/sec}}$ $\left(\frac{\text{rad/sec}^2}{\text{ft/sec}}\right)$		0.013 (0.004)
Damping-to-inertia ratio, sec^{-1}		0.7
Yaw due to lateral control (early version), $\frac{\text{rad/sec}^2}{\text{cm}}$ $\left(\frac{\text{rad/sec}^2}{\text{in.}}\right)$		0.026 (0.065)
Yaw due to roll angle (later version), $\frac{\text{rad/sec}^2}{\text{rad}}$		0.30
Vertical (approximate):		
Maximum thrust-weight ratio		>1.1
Power (collective) control sensitivity, g/cm (g/in.)		0.08 (0.2)
Normal-velocity damping-to-mass ratio, sec^{-1}		0.4

[a]Outside a ±0.64-cm (±0.25 in.) deadband.

TABLE II.- MAGNETIC-TAPE PARAMETERS RECORDED ONBOARD THE AIRCRAFT

Continuous data
Lateral acceleration
Normal acceleration
Range
Range rate
Cross range
Cross-range rate
Altitude
Altitude error
Vertical (altitude) rate
Time

Sampled data (20 samples per second)
Control positions – pitch, roll, yaw, and power
Flight-director commands – pitch, roll, and power
Attitudes – pitch and roll
Magnetic heading
Angular rates – pitch, roll, and yaw
Range-rate error

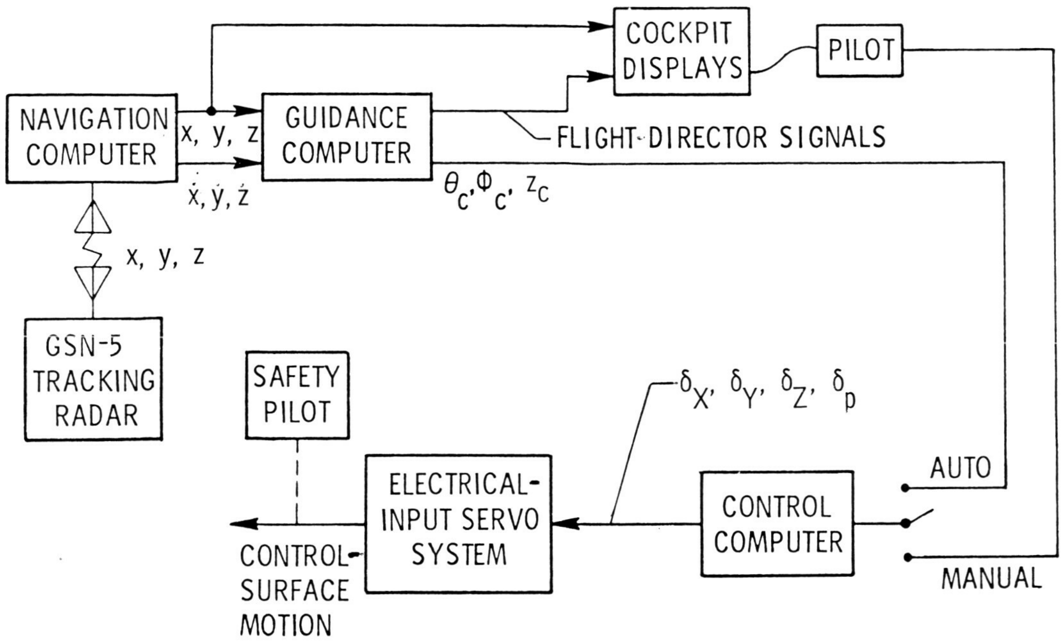

Figure 1.- Research-system elements.

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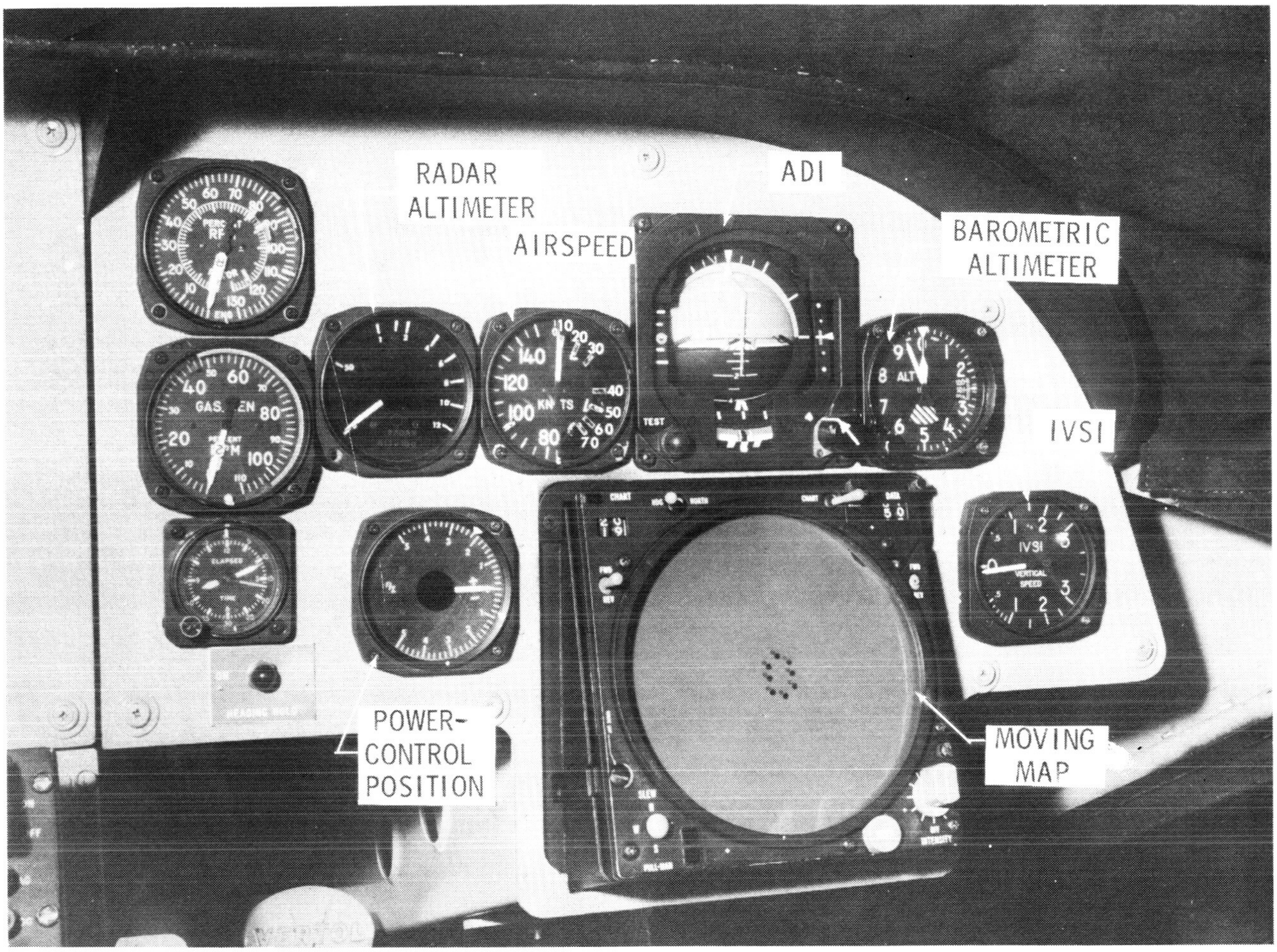
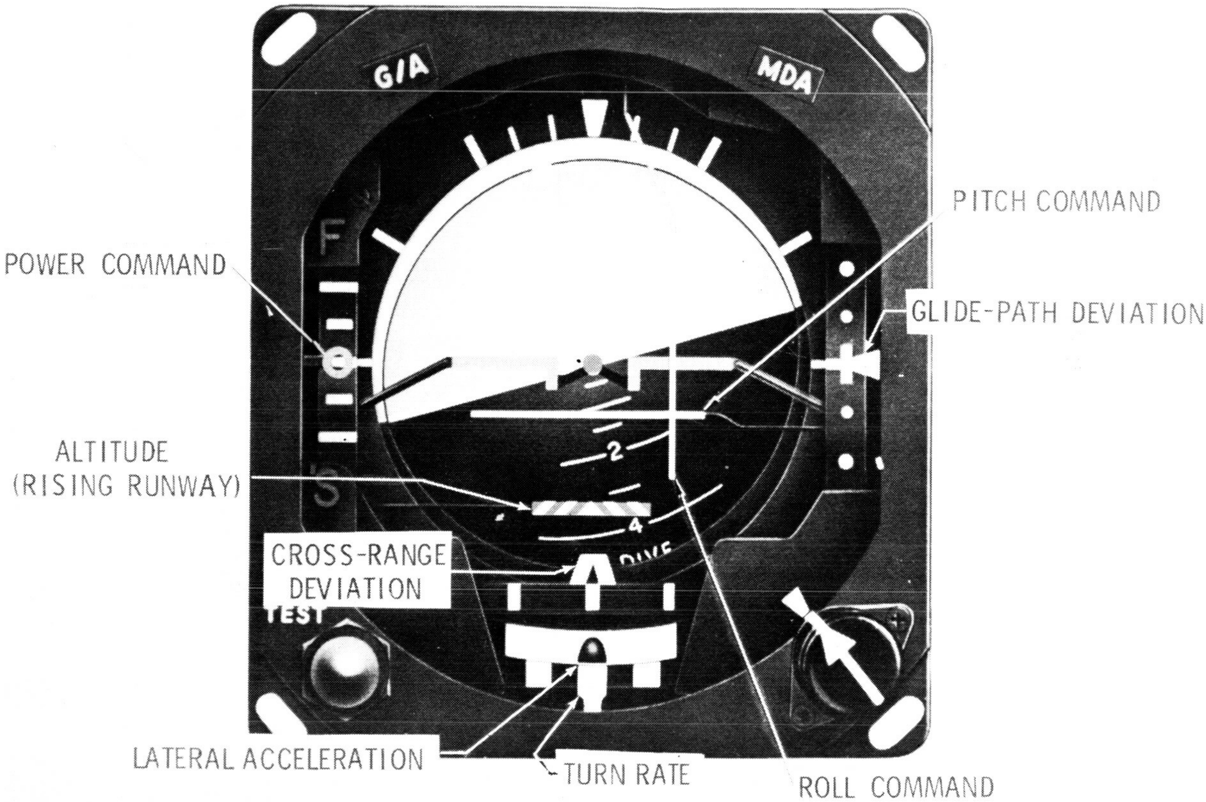
Figure 2.- Research helicopter.



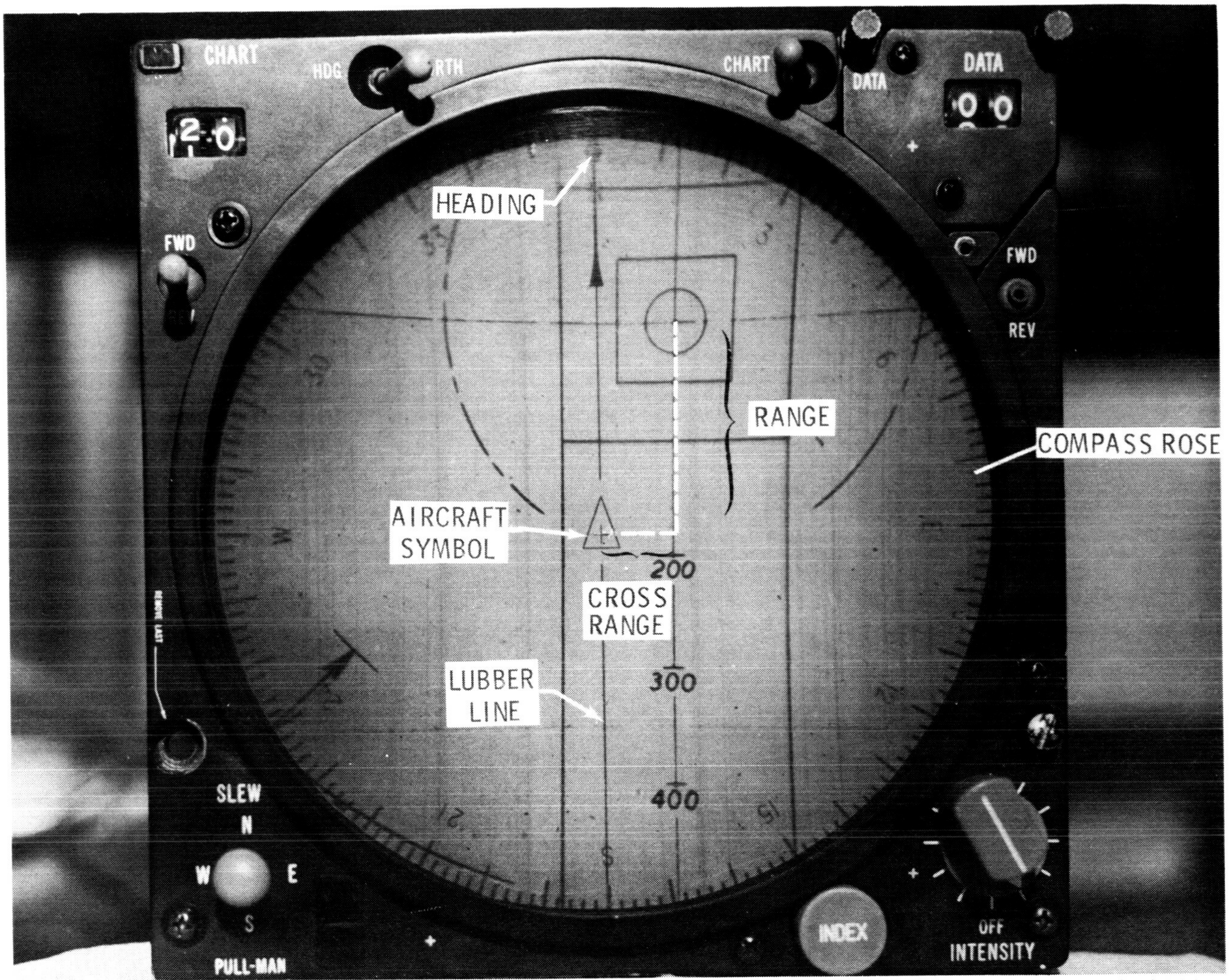
L-74-1048

Figure 3.- Evaluation pilot's panel.



L-74-1049

Figure 4.- Attitude-director indicator.



L-72-1443.1

Figure 5.- Moving-map display.

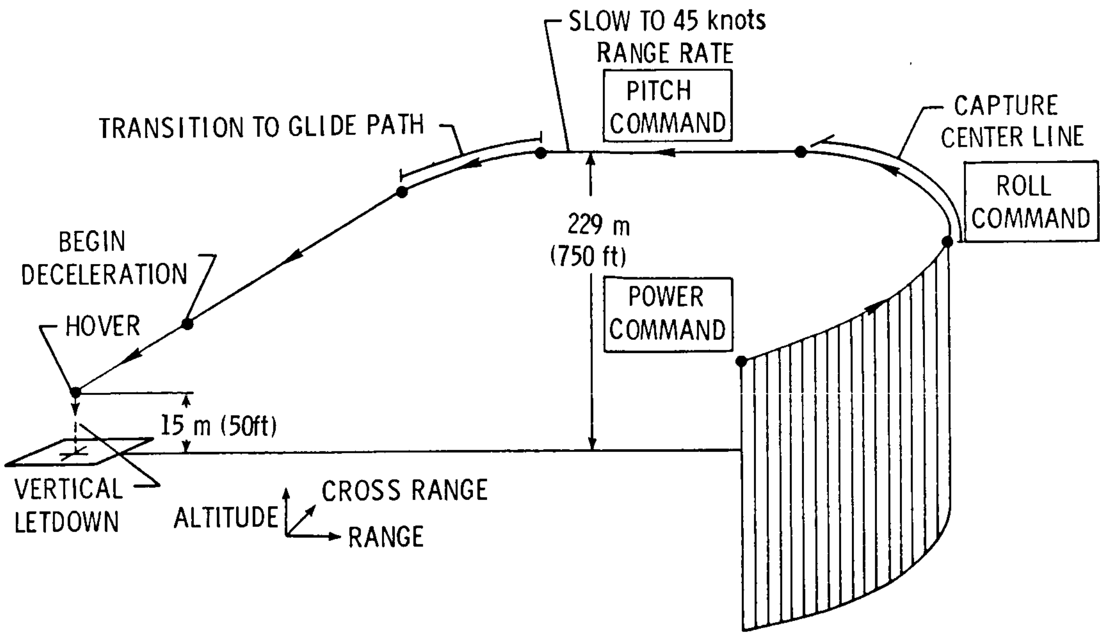


Figure 6.- Decelerating approach task.

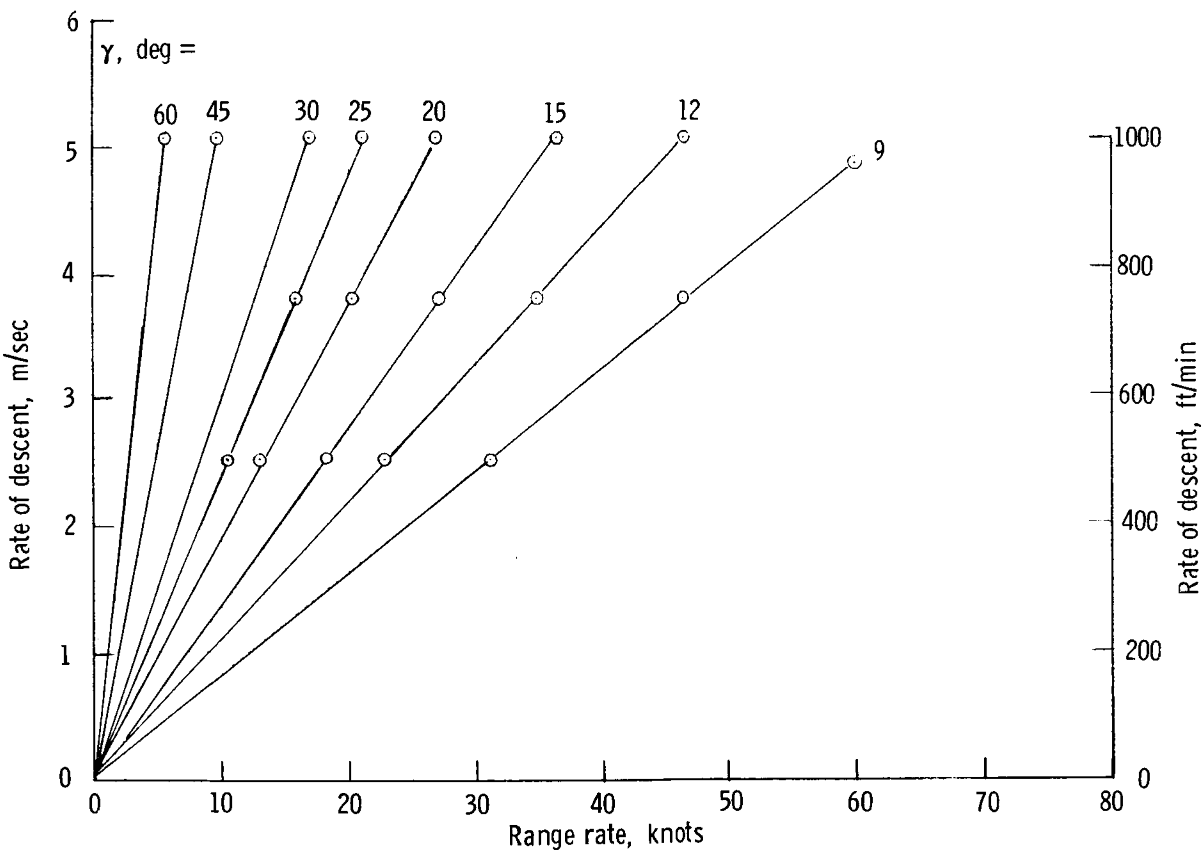


Figure 7.- Constant-speed test points.

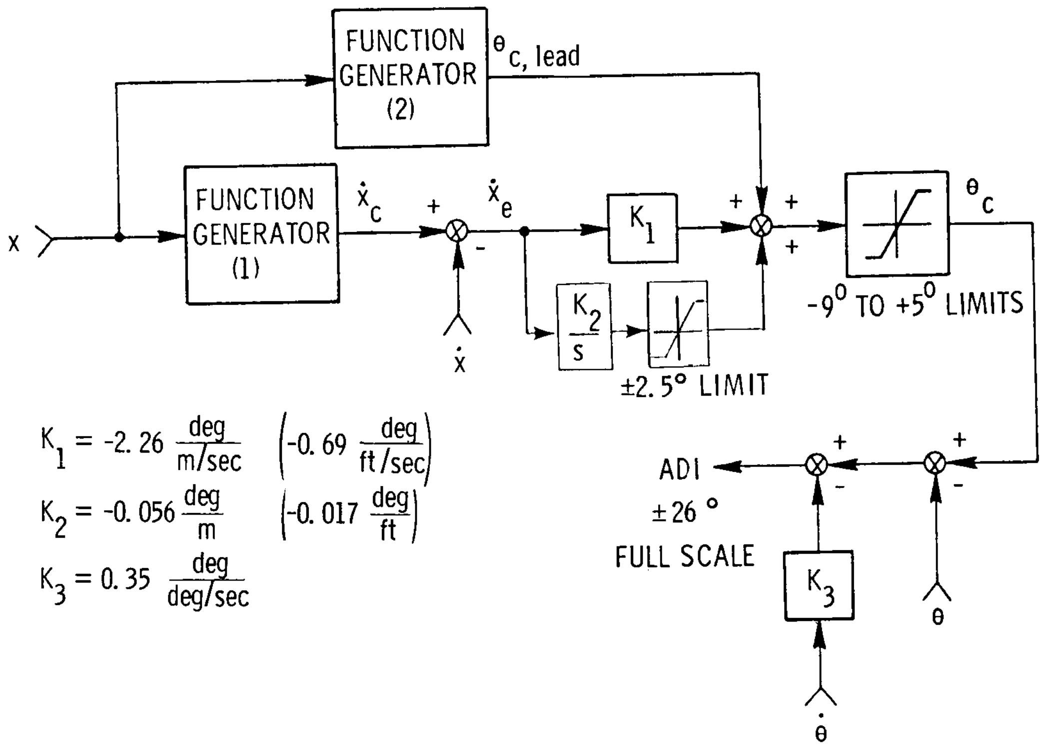
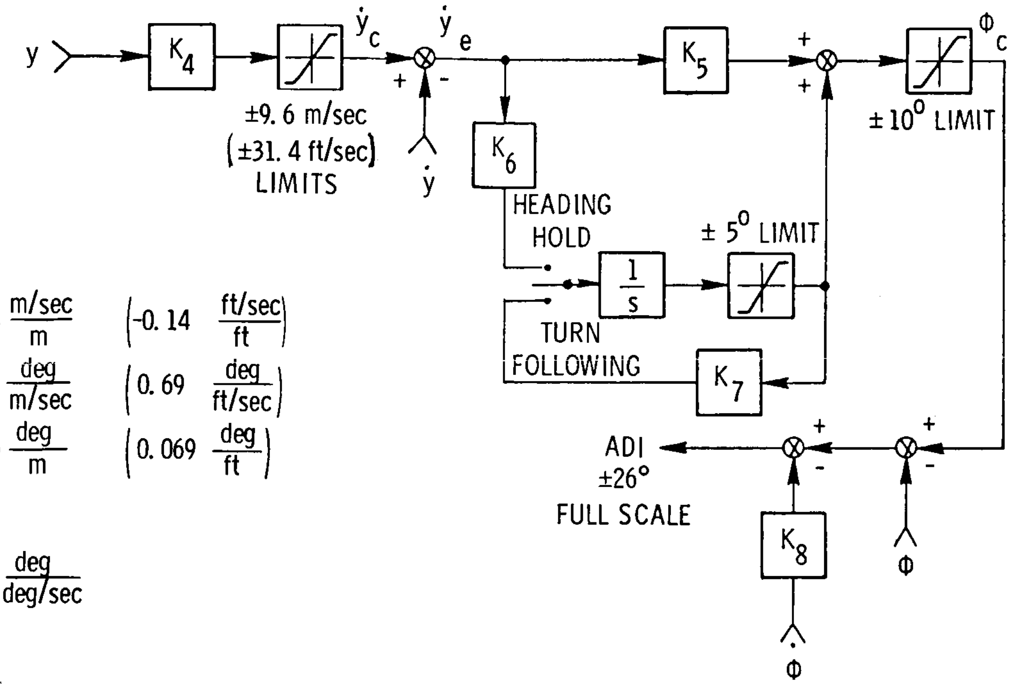


Figure 8.- Flight-director pitch command.



$K_4 = -0.14\ \frac{m/sec}{m}$ $\left(-0.14\ \frac{ft/sec}{ft}\right)$

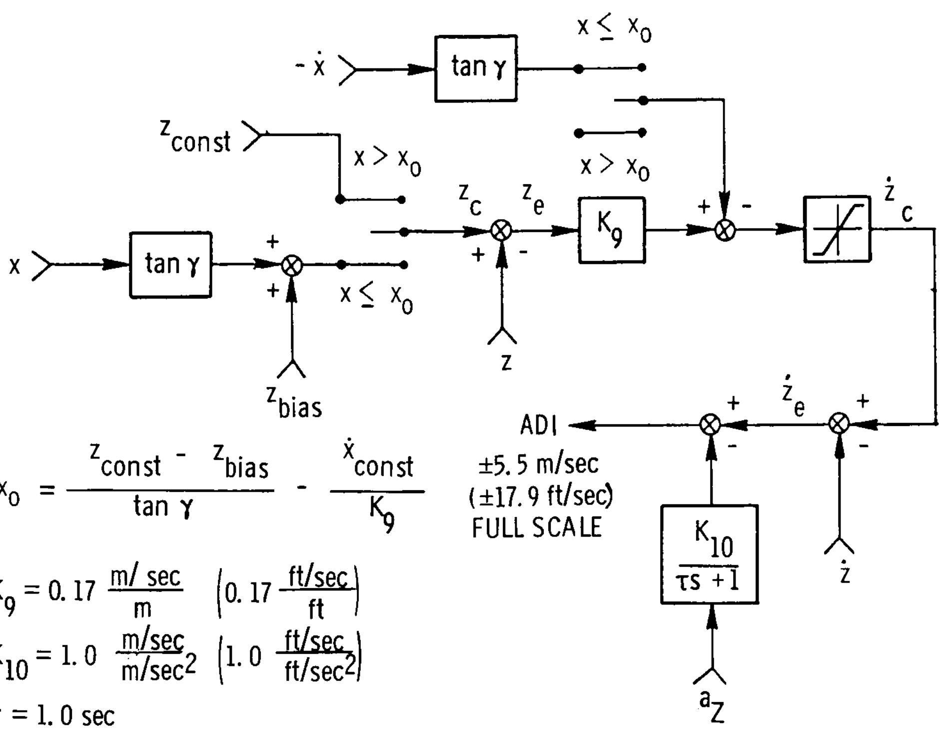
$K_5 = 2.26\ \frac{deg}{m/sec}$ $\left(0.69\ \frac{deg}{ft/sec}\right)$

$K_6 = 0.226\ \frac{deg}{m}$ $\left(0.069\ \frac{deg}{ft}\right)$

$K_7 = -0.2$

$K_8 = 0.35\ \frac{deg}{deg/sec}$

Figure 9.- Flight-director roll command.



$$x_0 = \frac{z_{const} - z_{bias}}{\tan\gamma} - \frac{\dot{x}_{const}}{K_9}$$

$K_9 = 0.17\ \frac{m/sec}{m}$ $\left(0.17\ \frac{ft/sec}{ft}\right)$

$K_{10} = 1.0\ \frac{m/sec}{m/sec^2}$ $\left(1.0\ \frac{ft/sec}{ft/sec^2}\right)$

$\tau = 1.0$ sec

Figure 10.- Flight-director power command.

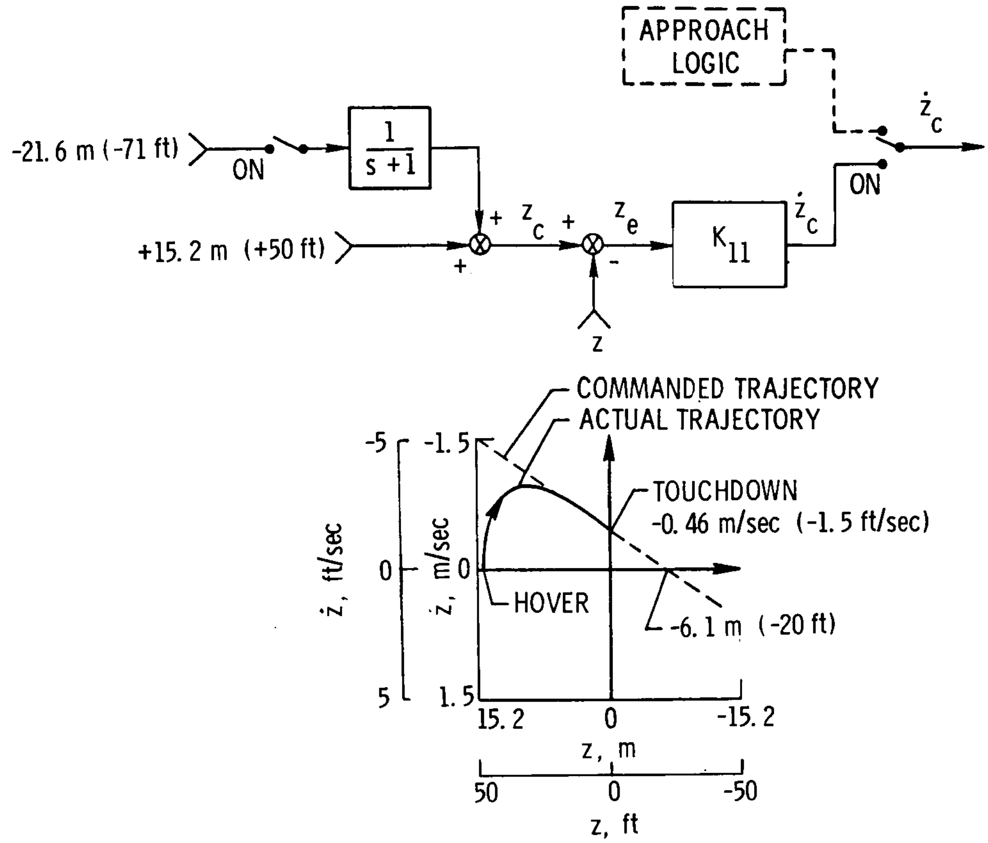


Figure 11.- Vertical letdown logic. $K_{11} = 0.07 \frac{m/sec}{m} \left(0.07 \frac{ft/sec}{ft}\right)$.

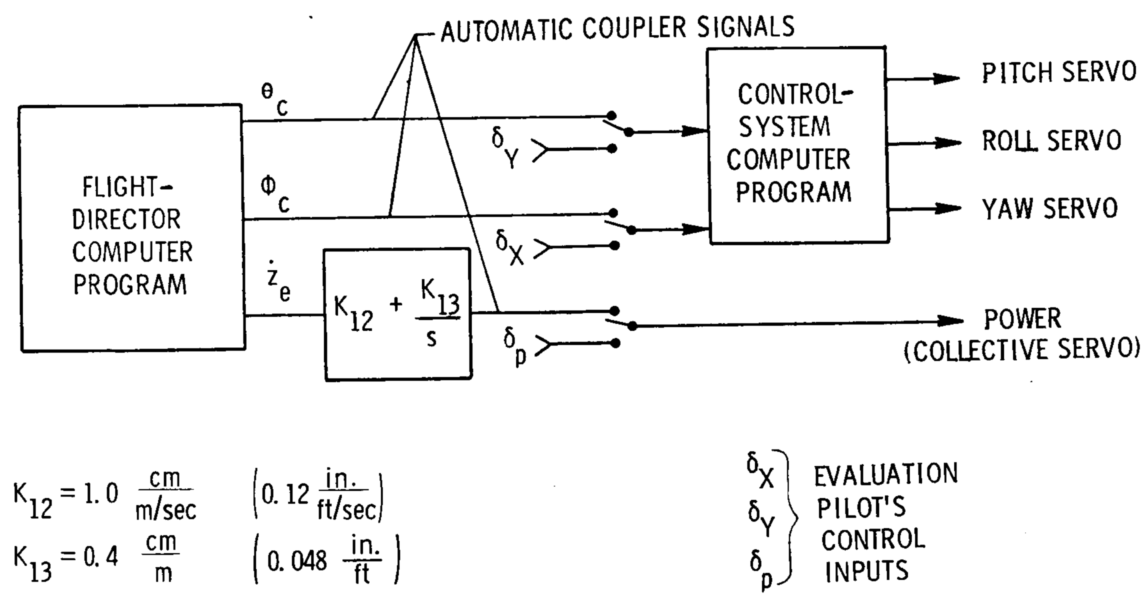


Figure 12.- Automatic approach and landing coupler.

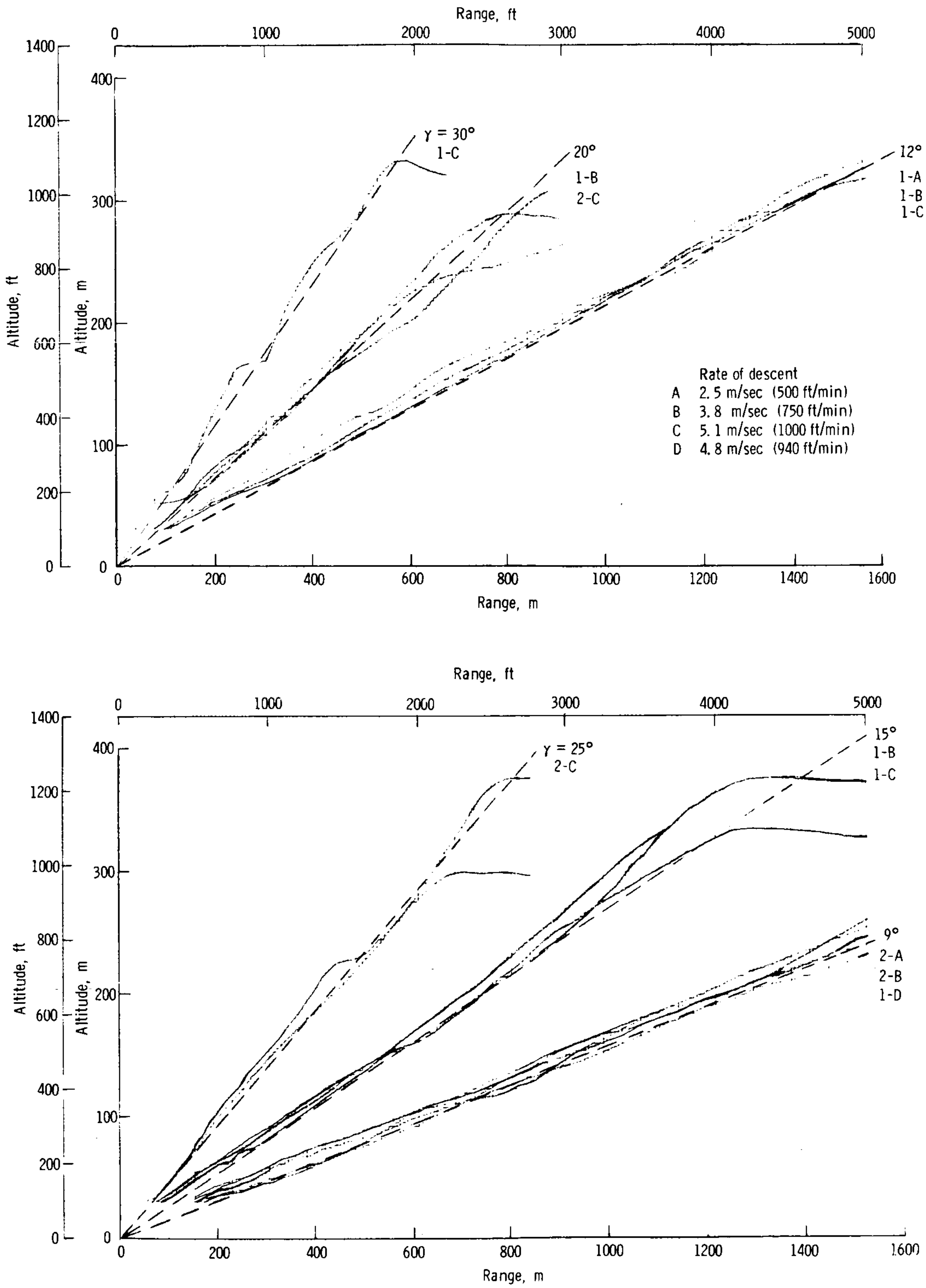


Figure 13.- Constant-speed approach performance. Dashed lines indicate desired track.

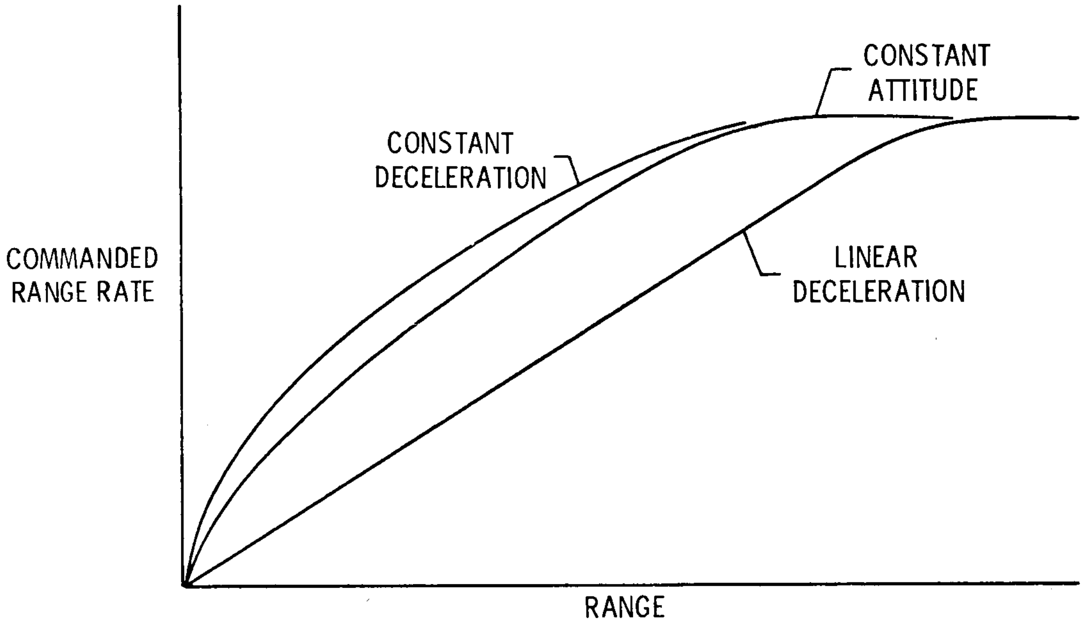
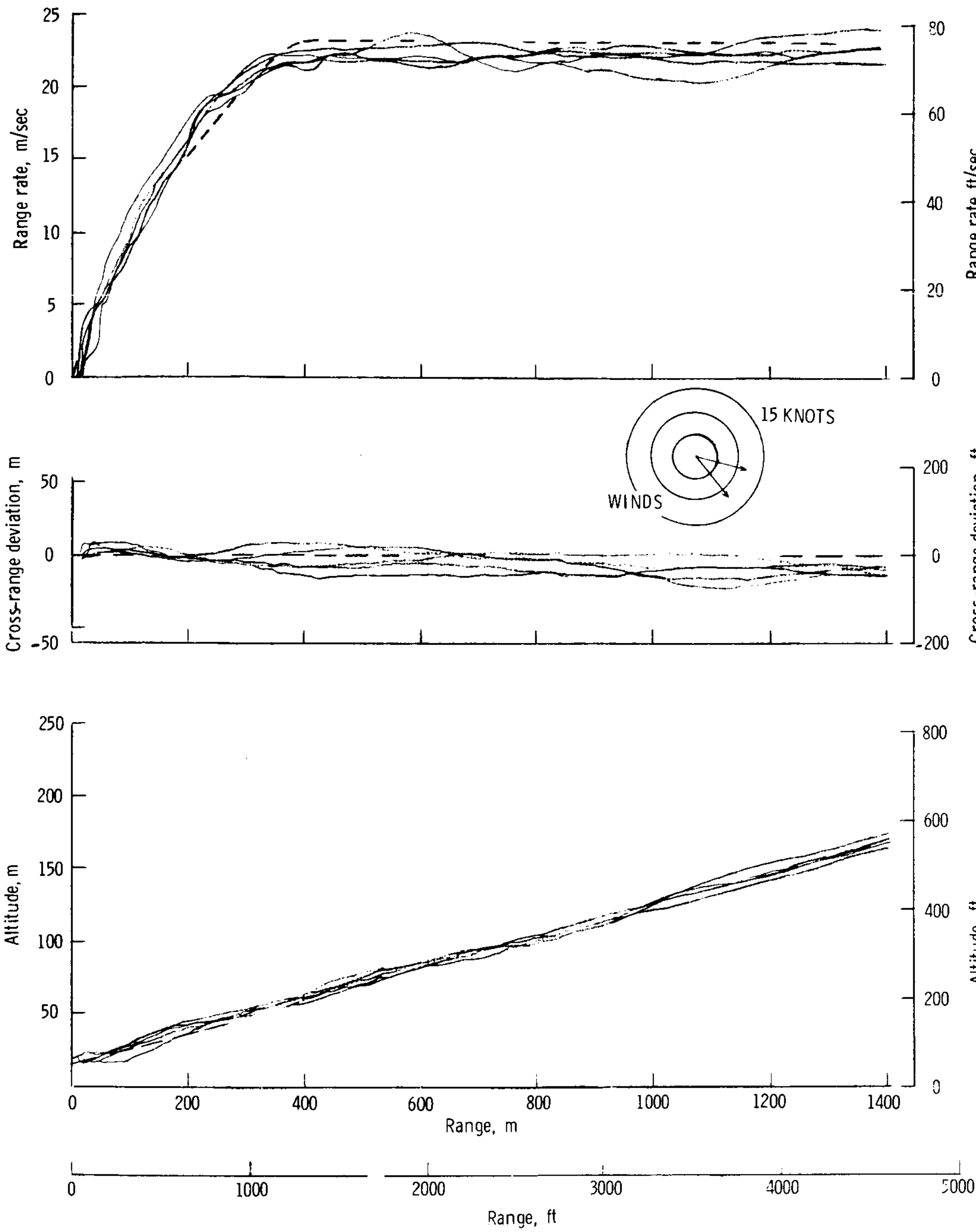
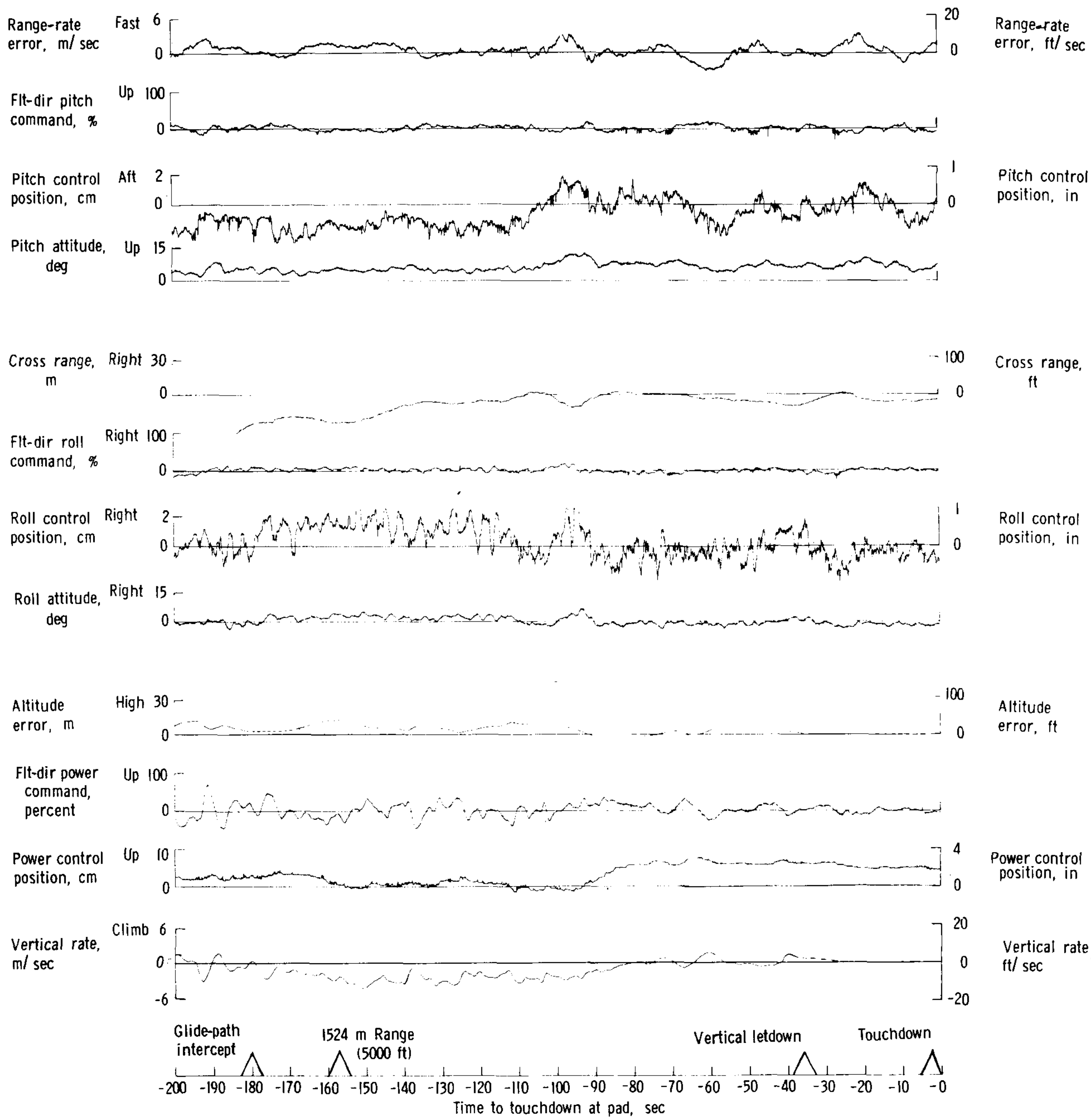


Figure 14.- Characteristic shapes of the deceleration profiles.



(a) Tracking performance.

Figure 15.- Manual decelerating approaches at 6°.
Dashed lines indicate desired track.



(b) Time history.

Figure 15.- Concluded.

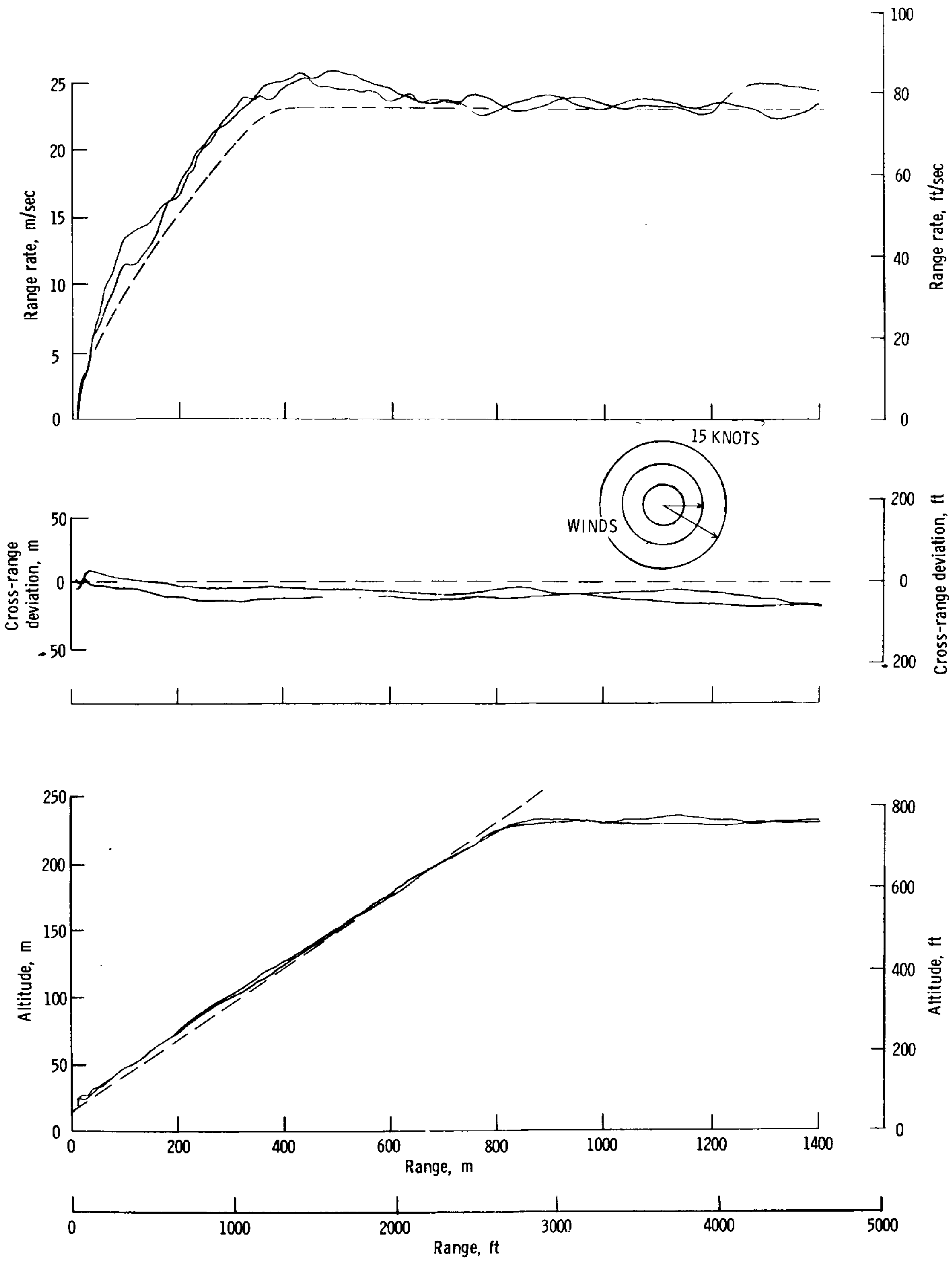


Figure 16.- Manual decelerating approaches at 15°. Dashed lines indicate desired track.

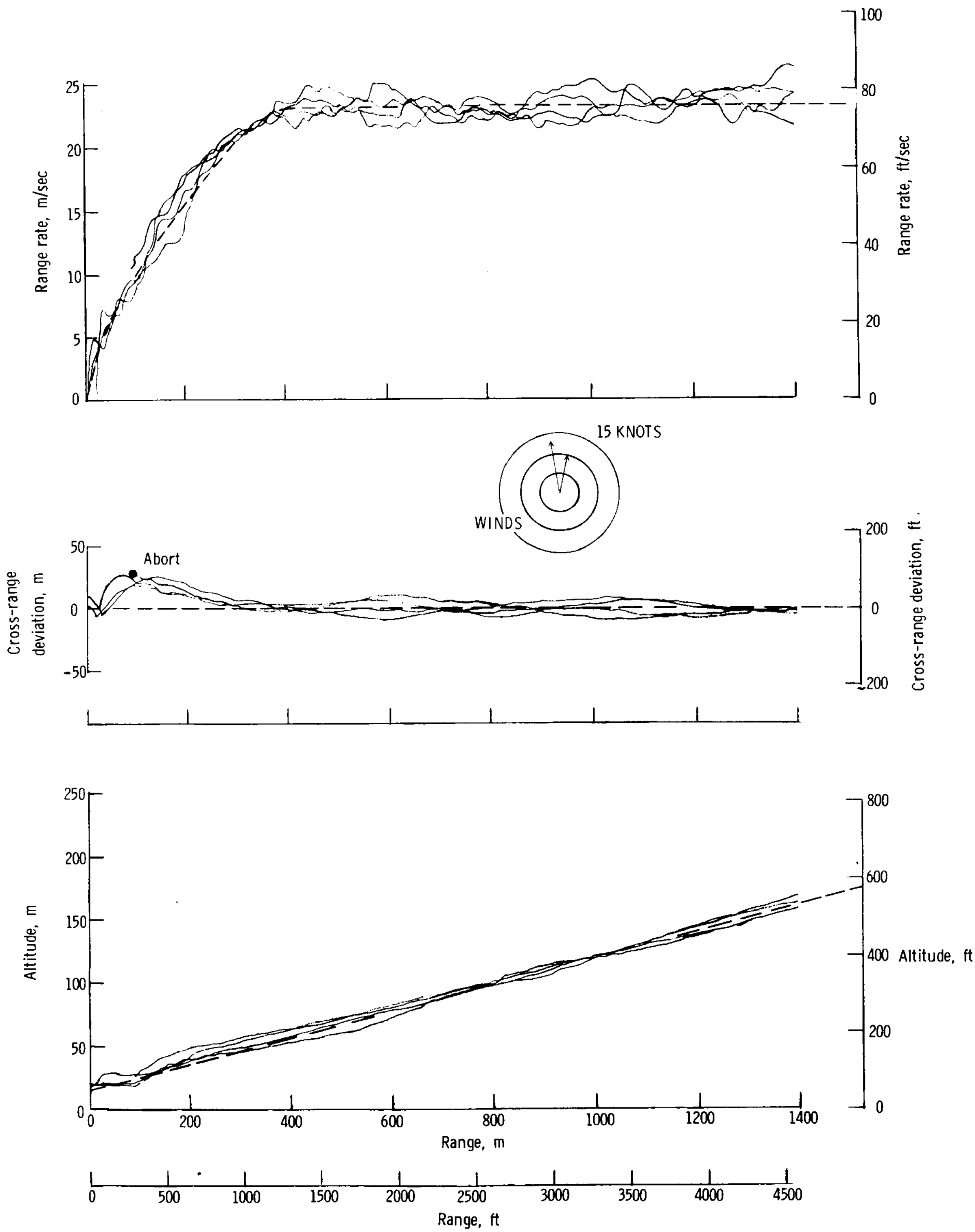
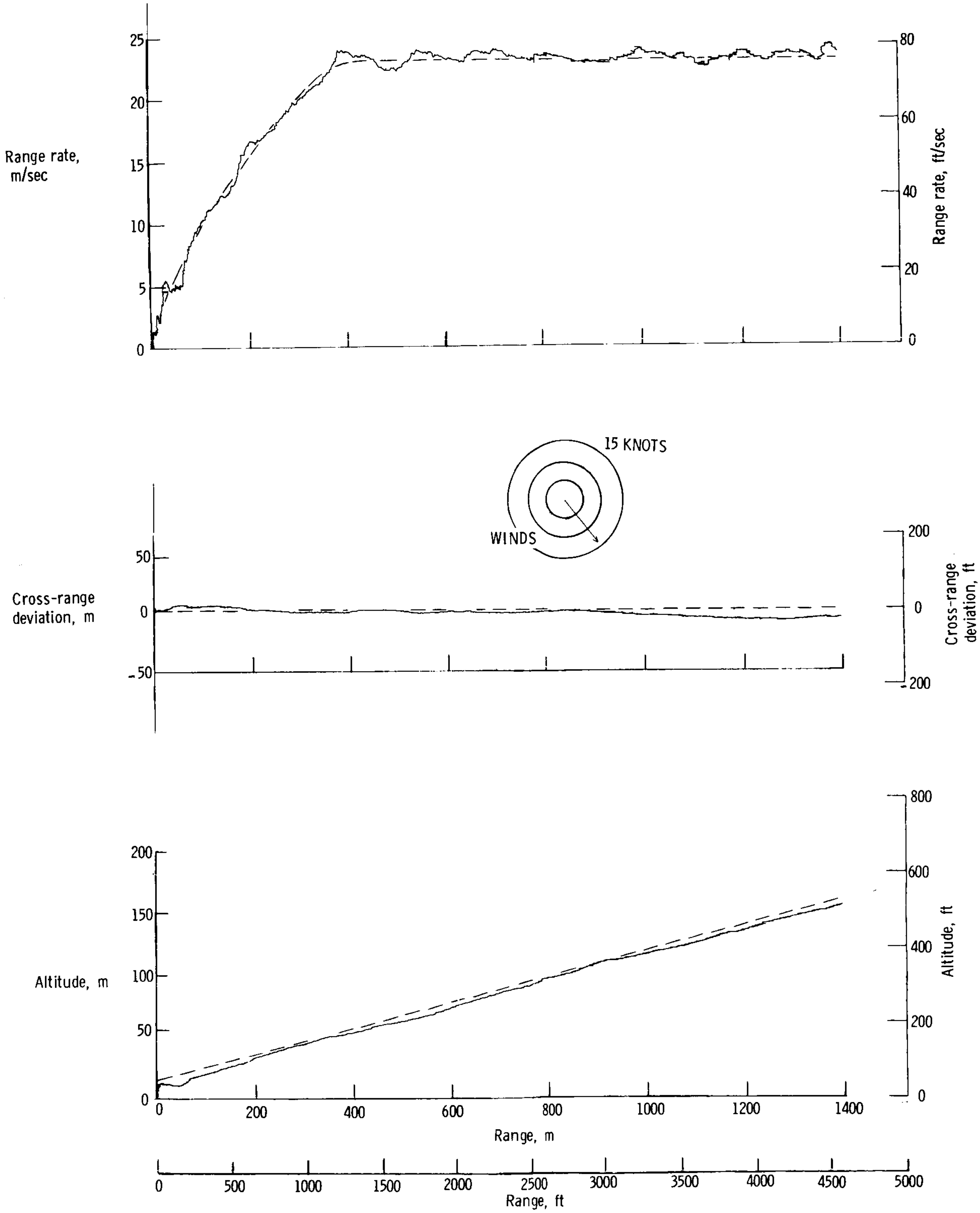
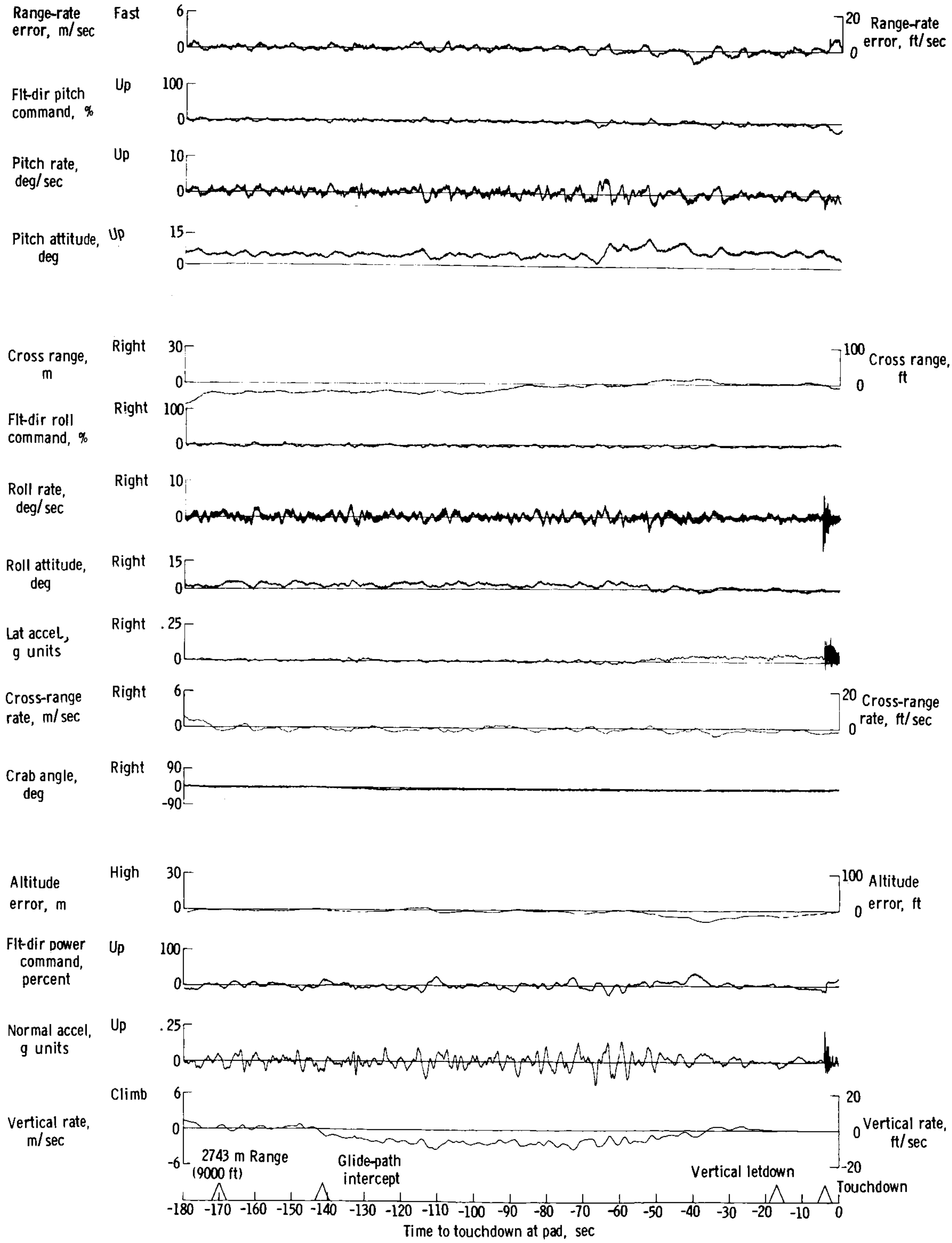


Figure 17.- Manual decelerating approaches of 6° in strong crosswinds of 10 to 13 knots. Dashed lines indicate desired track.



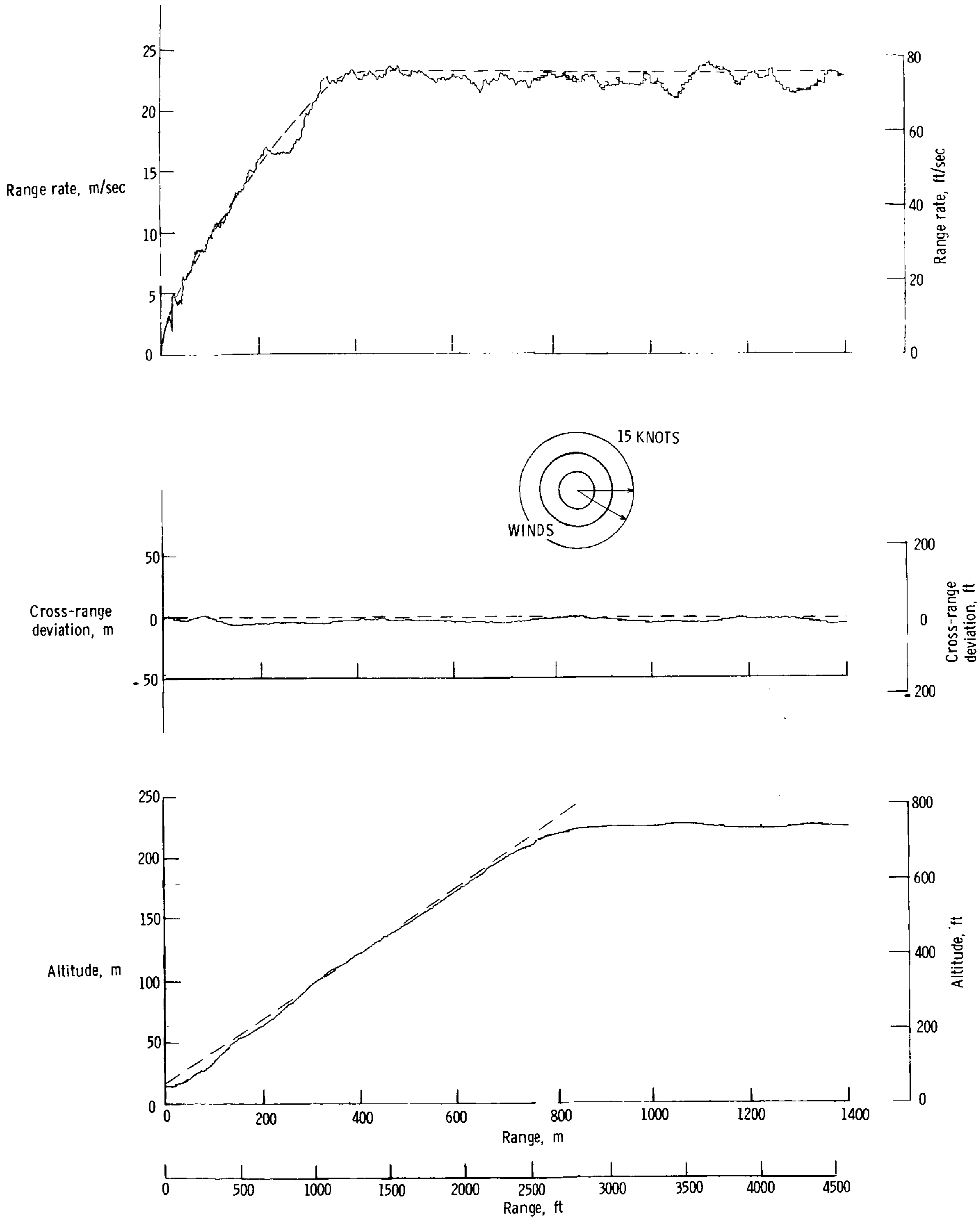
(a) Tracking performance.

Figure 18.- Performance of automatic system for $\gamma = 6^{\circ}$. Dashed lines indicate desired track.



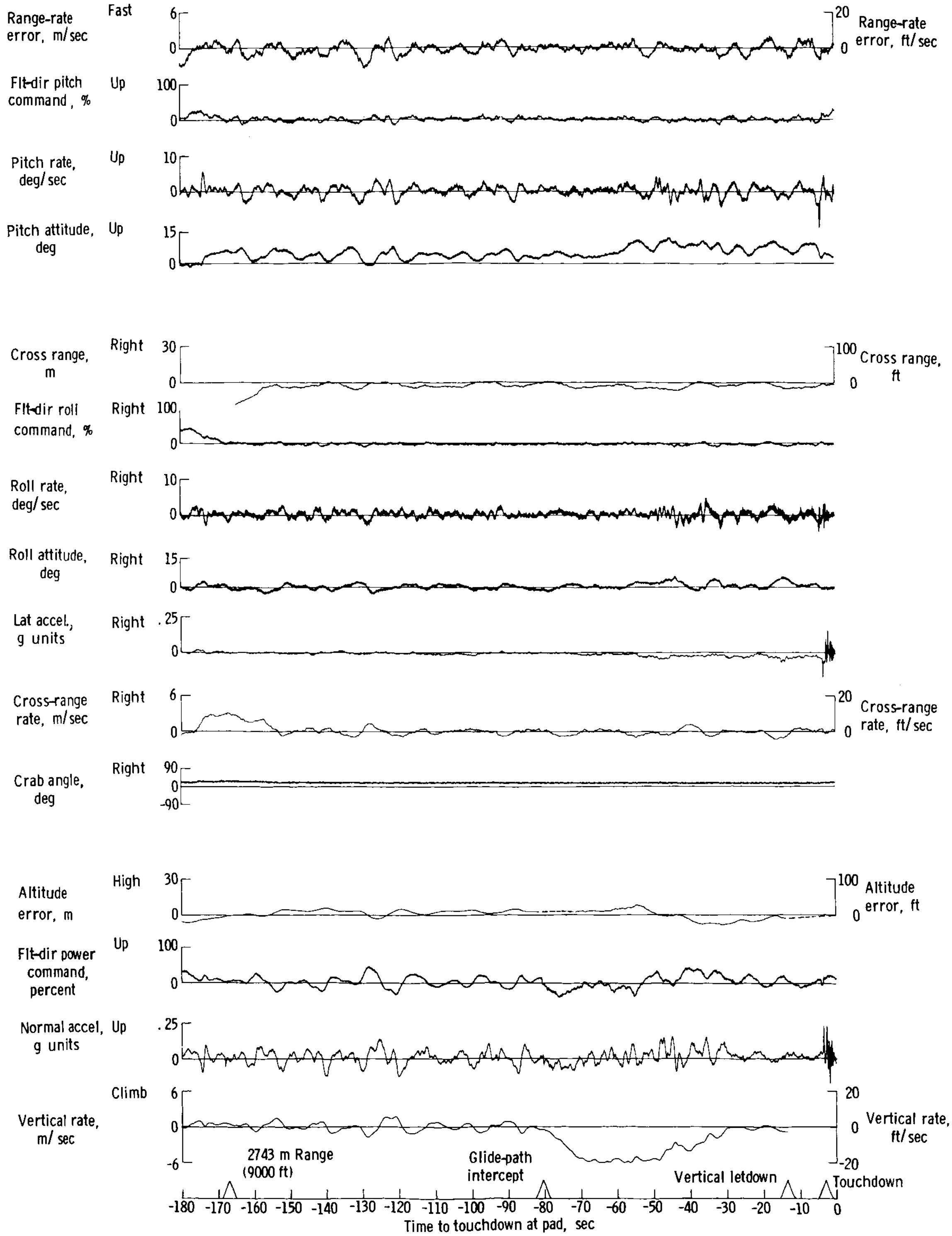
(b) Time history.

Figure 18.- Concluded.



(a) Tracking performance.

Figure 19.- Performance of automatic system for $\gamma = 15^{\circ}$. Dashed lines indicate desired track.



(b) Time history.

Figure 19.- Concluded.

NASA-Langley, 1974 L-9311