(304)

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PROJECT APOLLO

A PRELIMINARY SIMULATION STUDY OF THE REQUIRED VELOCITY METER SENSITIVITY AND TOUCHDOWN CONTROL PERFORMANCE DURING LUNAR LANDING

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SUMMARY

The landing and touchdown portion of the powered descent phase of the LEM mission was studied utilizing facilities assigned to the Guidance and Control Division. A fixed-base simulator containing an attitude hand controller, descent engine throttle, and pilot displays was used to represent the LEM cockpit. The six-degrees-of-freedom equations of motion were solved utilizing analog and analog-digital computing equipment. The main areas under study during the simulation were: (1) the effect of the forward and lateral velocity meter scale sensitivity on pilot control of horizontal touchdown velocities, (2) the effect of radar noise on pilot controlled touchdown velocities, (3) the effect of landing gear probe length on touchdown velocities and the ability of the pilot to perform a manual shutdown of the descent engine, and (4) the effect of automatic and manual scale change of the horizontal velocity indicator on touchdown conditions.

The results of this study indicate that a meter sensitivity of 0.075 inches per ft/sec provides the pilot with a velocity indication sufficiently accurate to land the spacecraft within the horizontal landing gear design limits. Also, the study results showed that pilot reaction time to the engine shutdown signal averaged approximately 0.32 second and could be expected to be less than 0.65 second 99.9% of the time. Using test data obtained during the study, calculations revealed that a probe length of the order of 4 feet would be necessary to assure that the pilots would have the engine off at touchdown.

INTRODUCTION

Previous simulations of manually controlled lunar landings have provided preliminary answers to some questions and identified other areas where further effort is required to provide more quantitative or more statistically valid data on which to base design requirements and operating procedures. This simulation study was intended to take advantage of the results of previous studies and by means of carefully controlled experiments determine the effect of the following on the probable limits of pilot controlled touchdown conditions:

1. Forward and lateral velocity meter sensitivity
2. Radar noise
3. Landing gear probe length
4. Pilot reaction time
5. Automatic and manual scale change of the forward-lateral velocity meter

Improvements were made in the probe light and engine off switch location and in the accurate representation of engine shutdown characteristics relative to earlier simulations. The touchdown conditions of interest were the components of vehicle velocity, attitude and attitude rate, and the engine on or off status.

DESCRIPTION OF SIMULATION

The simulation was implemented by coupling a partial simulation of the LEM cockpit containing the necessary pilot controls and flight displays to an analog solution of the equations of motion. An out-the-window display of the lunar surface was generated by a virtual image optical system and a special purpose digital computer. A complete description of the simulation is contained in the appendix of this note.

TEST PROCEDURE

Because of the large number of parameters associated with the study, the test procedure was divided into three parts to reduce the number of production runs required to obtain meaningful statistical data. All pilots were trained to a consistent level of performance prior to making data runs.

Part 1. - The objective was to obtain preliminary data which could be used to form the basis for determining the probe length, meter sensitivity, and whether manual or automatic velocity scale changing should be used. To do this, a test matrix consisting of 3 probe lengths (3, 4, and 5 feet), the four velocity meter faces (±2, ±4, ±10, and ±20) shown in figure 1, and manual and automatic velocity scale changes was set up. Sensitivities of the four meter faces were 0.375, 0.1875, 0.075, and 0.0375 inches/ft/sec for the ±2, ±4, ±10, and ±20 meter faces, respectively.

Part 2. - The primary objective of part 2 was to further evaluate the scale sensitivity for the forward and lateral velocity indicator in the presence of radar noise. Meter scales used were the same as in part 1. A three-foot probe length (arbitrarily selected) was used for the engine cutoff signal. Three pilots each made 15 runs for each of the meter faces, both with and without radar noise. Radar noise was simulated by filtering three Gaussian noise generators to limit the frequency to approximately 1 cps and adding the resulting noise to the signals driving the velocity displays in each of the three channels. The RMS value of forward velocity noise was 0.4 ft/sec while the RMS of lateral velocity varied linearly from 1.25 to 1.00 ft/sec as a function of altitude from 200 feet to touchdown. Vertical velocity noise also varied linearly from 1.0 to .75 ft/sec through the same altitude range.

Part 3. - Part 3 was conducted to obtain statistical data on touchdown velocities using the meter sensitivity determined from parts 1 and 2. The ±10 feet/second velocity scale (.075 in/ft/sec sensitivity) and a three-foot probe length were used. Five pilots flew 60 runs each for a total of 300 data runs.

Additional Tests

Several additional tests of a limited nature were made to (1) determine the effect of manual throttle control on touchdown parameters, (2) examine the effect of pressure suit operation on touchdown parameters, (3) investigate the effect of bias errors in the altimeter on the engine shutdown, and (4) determine the effect of a translational maneuver on pilot control of touchdown parameters.

TEST MANEUVER

The test maneuver for each part consisted of the latter portion of the landing approach phase and the final vertical letdown to the lunar surface. Initial conditions were:

Forward velocity	15 ft/sec
Descent rate	-8 ft/sec
Altitude	200 feet
Range to landing site	300 feet

Pilot Briefing

Before the test subjects began practice runs, each one was given an exensive briefing concerning the simulation mechanization, the simulation objectives, and landing procedures. The subjects were: (1) shown the location of all meters, indicators, and control devices and the function, reading, and scaling of each explained; (2) given a review of the simulation objectives; (3) given an explanation of the effects of using radar velocities with corresponding noise on landing conditions and its effect on the meter face and engine cutoff, and (4) instructed that it was mandatory to have the descent engine cut off as quickly as possible after the engine cutoff light actuated and that the forward and lateral velocities were to be as close to zero as possible at touchdown, not just less than the design limits.

Landing Instructions - After initiation of the landing maneuver (starting from the initial conditions previously noted), the pilots were instructed to pitch back and null the forward and lateral velocities when the landing area was reached. The spacecraft was then to be pitched forward to zero attitude and the descent to the surface completed. During this maneuver, the pilots were told to reduce the rate of descent to -6 ft/sec at an altitude of 100 feet, to -4 feet/second at an indicated altitude of 50 feet, and maintain the -4 feet/second descent rate until the probe light activated at which time the descent engine was to be manually shut off.

EVALUATION OF TEST RESULTS

The evaluation of the results of parts 1 and 2 were by pilot comments and simplified statistical analysis. This method of evaluation was used because the purpose of this portion of the simulation was to reduce the test matrix to a manageable number of variables.

A detailed analysis was performed to obtain the statistical characteristics of the data of part 3. Tests were made on the data to determine (1) statistical dependence of test variables and (2) normalcy of test data. Depending on whether the data were normally or non-normally distributed, the results were analyzed using standard techniques or order statistics as explained in reference 1. For normally distributed variables, the results are expressed in terms of means and standard deviations. The variables having indeterminant distributions are expressed as the probability that they would be within an upper and lower bound with a given confidence level; i.e., $\Pr(X_1 < \xi < X_2) = X$, with a confidence level of g. For this simulation study, sufficient data were obtained to establish the limits on the variables with a 95% (X) probability of occurrence and a confidence level of 93.5% (g).

DISCUSSION OF TEST RESULTS

The discussion of test results is presented in four sections. The first three sections contain the results from the pilot comments and statistical analysis whereas the results discussed in the last section are merely observations of what occurred during the tests.

Results of Part 1

Variation of the probe length influenced only the vertical touchdown velocity. Table I shows that the vertical velocity increased roughly 1 ft/sec for each additional foot of probe length. Data relative to engine-on landings were scattered. Results of this portion of the study showed 2 engine-on landings with the 3-foot probe, none for the 4-foot probe and one engine-on case for the 5-foot probe. The test data showed no abnormal vertical velocities at touchdown for any of the probe lengths studied. The average pilot reaction time to the probe light signal was calculated to be 0.37 seconds. A review of the test data indicated the pilot had a reaction time of 0.97 second for the 5-foot probe length engine-on landing and it was concluded in this isolated case he became too preoccupied with other control tasks to react normally to the probe light signal. In the cases of the engine-on landings with the 3-foot probe, the test data showed that both pilots had reaction times of about 0.75 second which was only .15 second over the maximum allowable time of 0.6 second for the descent rate being used. For these landings, it was obvious that the difference between the allowable 0.6 second reaction time for the 3-foot probe and 4 ft/sec descent rate and the average pilot reaction time of 0.37 second did not provide sufficient margin to assure that the engine would always be off at touchdown. Back calculations using the test data for these two runs show that the two engine-on landings would have been eliminated using a 4-foot probe length. Based on these results, it was decided to obtain data relative to pilot reaction time during the statistical runs and use these data to provide a design basis for selection of the probe length necessary for manual shutdown of the descent engine.

Table I also shows that the forward and lateral velocities were not a function of manual or automatic switching of the velocity display. However, the pilots expressed a strong preference for manual switching and the remainder of the study was conducted using manual switching. The final selection of velocity scale switching procedures should be based on operational considerations and the results of a simulation of the entire powered descent maneuver.

Results of Part 2

Table II lists the average of the absolute values of horizontal touchdown velocity for the different meter faces both with and without radar noise. From this table, it is seen that without noise the touchdown velocities are lowest for the most sensitive meter face and highest for the least sensitive. However, with radar noise added, the lowest velocities occur for the ±10 fps meter face as seen in figure 2. This indicates that when noise is added to the more sensitive meter scales it became distracting to the pilots and they found it difficult to use the meter for a nulling task. On this basis, the meter sensitivity chosen for detailed study was 0.075 in/ft/sec (±10 ft/sec scale). The primary advantages of this meter scale are: (1) sufficient resolution for accurate landings while still being sufficiently insensitive to smooth the radar noise, (2) the ±10 scale is large enough so that the scale change is made at a high enough altitude to not interfere with critical control requirements during touchdown, and (3) the range is large enough to make a final translation without multiple scale changes.

Results of Statistical Data Runs

The results of the statistical analysis indicate the non-normally distributed variables without radar errors are within the present design limits as shown in the following table which gives the bounds such that the

$$\Pr \left\{ X_1 < \xi < X_2 \right\} = .95$$

		Forward Velocity ft/sec	Lateral Velocity ft/sec	Vertical Velocity ft/sec	Yaw Attitude deg	Pitch Rate deg/sec	Yaw Rate deg/sec	Roll Rate deg/sec
LOWER LIMITS	X_1	-1.38	-1.49	4.89	-2.86	-0.78	-0.19	-2.44
UPPER LIMITS	X_2	1.41	1.54	6.88	1.57	0.66	0.25	0.82

0.935 Confidence Level

Means of the forward and lateral velocities are very near zero and the average vertical touchdown velocity is of the order of 6 feet/second. The attitude rates, with the exception of the roll rate, are low which is consistent with the observed pilot control techniques.

Pitch and roll attitude, which were normally distributed, have means and standard deviations as indicated in the table below:

	Pitch Attitude	Roll Attitude
	deg	deg
Mean	-0.15	-0.93
Standard Deviation	1.02	1.09

Summing the expected 3σ excursions of pitch and roll attitude about their respective means shows that the lower bounds exceed the present landing gear design limit of 3 degrees. The upper 3σ bounds, however, remain within the design limit. The most severe case is that of roll attitude where the expected 3σ angle will be of the order of 4 degrees. There is very little chance that this can be lowered because the pilot must compensate for the attitude caused the c.g. offset to maintain a nulled lateral velocity.

Statistical addition of the expected one sigma radar errors (no IMU errors considered) of 1.0, 0.9, and 0.5 feet/second with zero means in the forward, lateral, and vertical axes, respectively, caused the translational velocities to become normally distributed and, as would be expected, to increase. The resulting means and standard deviations of these variables are contained in the table below:

	Forward Velocity ft/sec	Lateral Velocity ft/sec	Vertical Velocity ft/sec
Mean	-0.08	0.15	5.64
Standard Deviation	1.13	1.17	0.67

The expected 3σ excursion of all the variables about their means are still within the present design limits, but the forward and lateral velocities are beginning to approach a value which might become critical. However, the absolute value of the vector sum of the forward and lateral velocities; i.e., $(V_x^2+V_y^2)^{\frac{1}{2}}$, is of more interest than the individual velocities. Statistical theory shows that a sum of squares based on two squares from data whose distributions are normal with means equal to zero and equal standard deviations are distributed as "chi-square" with two degrees of freedom. Because the simulation data essentially meet these conditions, the probability of being less than a specified value can be directly and accurately determined. For these simulation data, it can be shown that the probability of the horizontal velocity being less than 4 ft/sec is 0.9976. To state this as has been done previously

$$\Pr \left\{ (V_z^2+V_y^2)^{\frac{1}{2}} > 4.00 \right\} = 0.0024$$

which means that 0.24% of the landings will have horizontal velocities exceeding the translational velocity design limit.

<u>Probability Contours of Landing Velocities</u> - To define the existed landing velocity envelope, probability contours for combined vertical and horizontal velocities were calculated. For ease of computation, it was assumed that forward, lateral, and vertical velocities were normally distributed (as they were with radar errors present) and that forward and lateral velocities had zero means and equal standard deviations (very nearly true). The contours and the probability of being on or inside a given contour is presented in figure 3. For example, the 0.99 contour line contains 99% of the possible combinations of vertical and horizontal velocities at landing. It should also be noted the data are based on results obtained using a three-foot probe. However, the effect of increasing the probe length merely shifts the contours upward vertically at about 1 ft/sec/ft of probe length. Conversely, shortening the probe lowers the contour at the same rate per foot decrease of probe length for small changes. A detailed explanation of the mechanics of calculating these probability contours is given in reference 2.

Descent Engine Cutoff Control

The 300 statistical data runs produced a total of 4 engine-on landings. A review of the test data revealed that the longest pilot reaction time for an engine-on landing was 0.67 second. Back calculating using the test data for this particular run, shows that this case would have been reduced to an engine-off landing had the probe length been of the order of 4 feet.

A $\log_e$ plot of reaction time as a function of the probability of occurrence indicates the log of the reaction time to be normally distributed (figure 4). Inspection of this figure shows that 99.9% of the reaction times will be less than 0.654 second and that the average reaction time is 0.312 second.

These data, together with the data on vertical velocity at touchdown, can be used to analytically derive a probe length that provides a specific probability of having the engine off at touchdown. This result must be balanced against the probability of high vertical touchdown velocity caused by the possible combination of a 3σ low pilot reaction time with the longer probe length required to insure engine off touchdown in the 3σ high reaction time case. These results will be reported separately, but it can be seen qualitatively that the necessity to find a best compromise on probe length stems directly from the necessity to accommodate the $\pm 3\sigma$ tolerance on pilot reaction time. This could be avoided and a lower maximum vertical velocity assured if the engine shutoff prior to touchdown were not critical or could be accomplished automatically.

Results of Additional Test Cases

The results of the several additional test conditions examined are based on a very limited number of production runs. Therefore, the results as presented are mere statements of what occurred rather than conclusions and should be treated as such.

Manual Control of Descent Engine Throttle _ In all cases run, the spread of touchdown conditions increased and engine-on landings were frequent. Specifically, in rate command without rate of descent control, there were 7 engine-on landings and one run where the vertical velocity exceeded 10 ft/sec in 30 runs. There is no doubt that the distance between the descent engine throttle and engine off switch attributed significantly to the large number of engine-on landings. However, it can also be hypothesized that additional operational training in this control mode would significantly reduce, if not altogether eliminate, the engine-on landings.

Pressure Suit Operation - The pressure suit operation indicated that some spread increase in touchdown conditions might be experienced in a pressurized suit, but that unpressurized operation did not appreciably change the test results from those obtained in the 300 statistical runs. There were, however, some unexpected problems in pressure suit operations: (1) It was difficult with the glove on, regardless of pressurization, to feel whether the thumb was on the FIRE or STOP engine switch (figure 5a) and a visual check was required before depressing the switch which directed the pilot's attention from the other control tasks; (2) The incremental ROD switch spring force was too light for the pilots to sense actuation with the glove on; (3) In pressurized operation, it was difficult to make a pitch-up maneuver because of the pivot point of the attitude rotational controller and the wrist rest position; and (4) It was difficult and clumsy to make a manual scale change on the cross pointer velocity display in a pressurized suit (figure 5b). These problems require additional investigations at both MSC and at GAEC using the III-B (Landing) and Full Mission Engineering Simulator (FMES) simulations.

Translational Maneuver and Altitude Bias - With the final translation maneuver, the problem of engine-on landing occurred again. The test results showed 7 engine-on landings in 28 separate runs. Here again, additional training would probably reduce the problem.

The altimeter bias errors (0 to 20 feet) did not affect pilot response. No engine-on landings occurred, and there was no apparent effect on touchdown conditions. The most probable reason for the bias not contributing adverse effects is that the pilots implicitly believed the probe light signal and therefore were relatively unconcerned about the altimeter reading below an altitude of the order of 25 feet.

CONCLUSIONS

1. The meter sensitivity for forward and lateral velocity display should be 0.075 inches per ft/sec, with at least $\pm$10 ft/sec full scale deflection.

2. The effect of radar noise on velocity meter sensitivity is to force a compromise between the maximum sensitivity desired for accurate control of touchdown conditions and the minimum sensitivity desired to eliminate the meter response to the noise. The first conclusion is a result of that compromise.

3. With the $\pm$10 ft/sec velocity meter, 3 ft. probe, and the procedure used, the results indicate a 0.9976 probability of the landing being within the landing gear velocity design limits. This conclusion is based on three sigma radar errors of 1.5, 2.7, and 3.0 ft/sec in the vertical, lateral, and forward axes, respectively.

4. Manual or automatic dwitching of velocity meter scales did n ot affect touchdown conditions, but the pilots expressed a strong preference for manual switching.

5. The average pilot reaction time to the probe light signal was 0.32 sec.

6. A four-foot probe, approximately, would have been required to eliminate all the reasonably valid engine-on landings.

REFERENCES

1. An Introduction to Mathematical Statistics. H. D. Brunk, Ginn & Co. 1960

2. Procedures outlined by A. H. Feiverson, Theory and Analysis Office, Computation and Analysis Division, MSC

3. Project Apollo Internal Note, MSC-IN-EG-65-25 entitled "Simulation Study of the Powered Descent Phase of the LEM Mission from Transition to Touchdown", by Richard Reid and Herbert G. Patterson

TABLE I - AVERAGE TOUCHDOWN VELOCITIES (PART I OF SIMULATION)

Meter Sensitivity (in/ft/sec)	Meter Scale	Probe Length (ft)	Scale Switching	AVERAGE TOUCHDOWN VELOCITIES (ft/sec)		
				FORWARD	LATERAL	VERTICAL
0.375	±2	3	M	.20	.14	5.15
			A	.14	.19	5.20
		4	M	.12	.15	6.31
			A	.17	.10	6.05
		5	M	.22	.21	6.89
			A	.12	.12	6.93
0.1875	±4	3	M	.34	.39	5.13
			A	.37	.47	5.19
		4	M	.28	.23	6.21
			A	.27	.23	6.13
		5	M	.34	.26	7.22
			A	.24	.23	7.15
0.075	±10	3	M	.33	.59	5.09
			A	.31	.32	5.25
		4	M	.24	.36	6.34
			A	.41	.36	6.24
		5	M	.30	.36	7.02
			A	.31	.42	7.30
0.0375	±20	3	M	.50	.48	5.22
		4	M	.67	.28	6.22
		5	M	.53	.52	6.90

TABLE II - AVERAGE ABSOLUTE VALUES OF HORIZONTAL TOUCHDOWN VELOCITIES FOR DIFFERENT METER FACES

(With and Without Radar Noise)

Meter Sensitivity in/ft/sec	Meter Face ft/sec	Forward Velocity ft/sec	Forward Velocity* ft/sec	Lateral Velocity ft/sec	Lateral Velocity* ft/sec
0.375	±2	.17	.36	.24	.46
0.1875	±4	.19	.33	.27	.37
0.075	±10	.30	.28	.24	.35
0.0375	±20	.41	.49	.57	.42
0.01875	±40	.66	.76	.53	.66

*With radar noise

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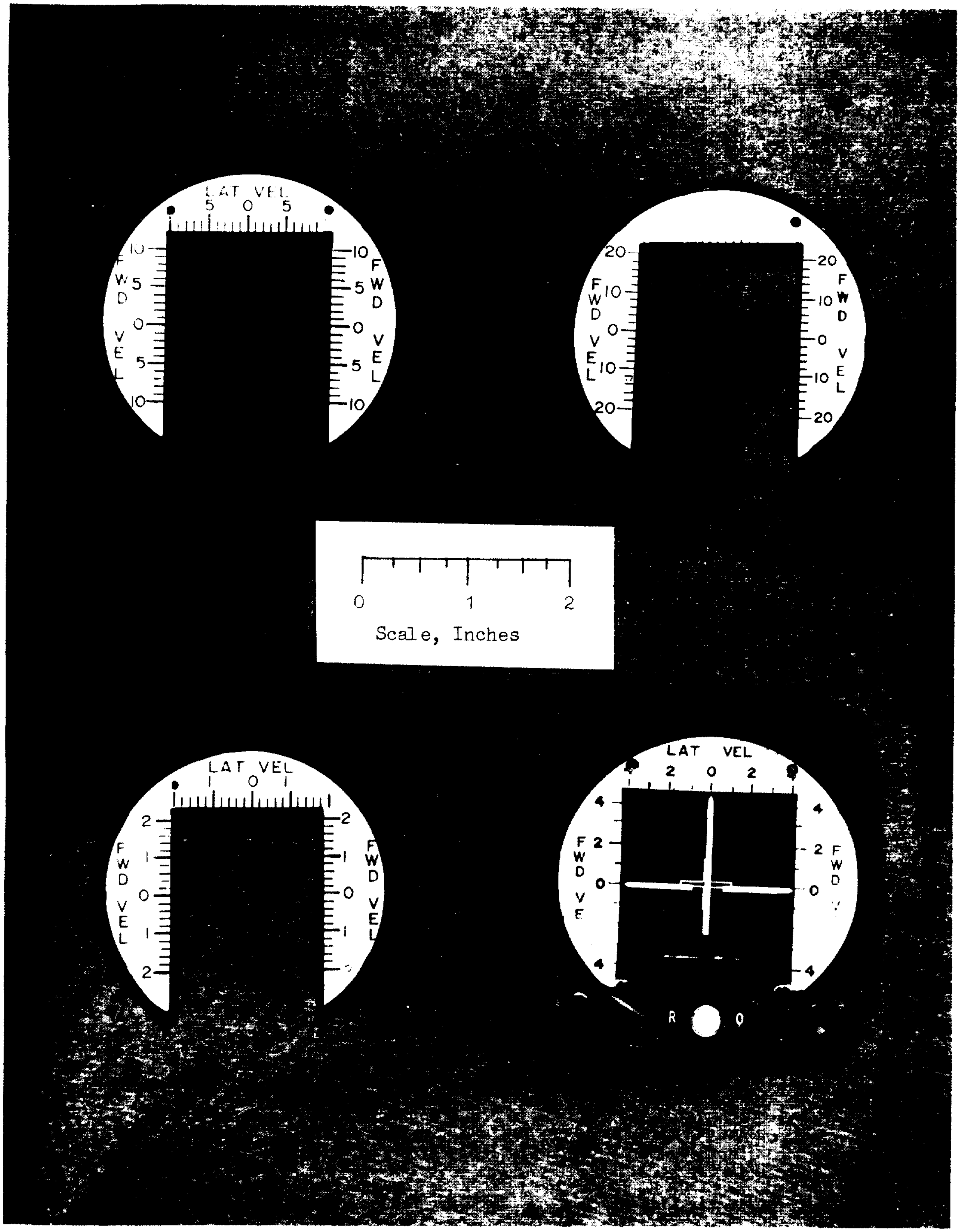
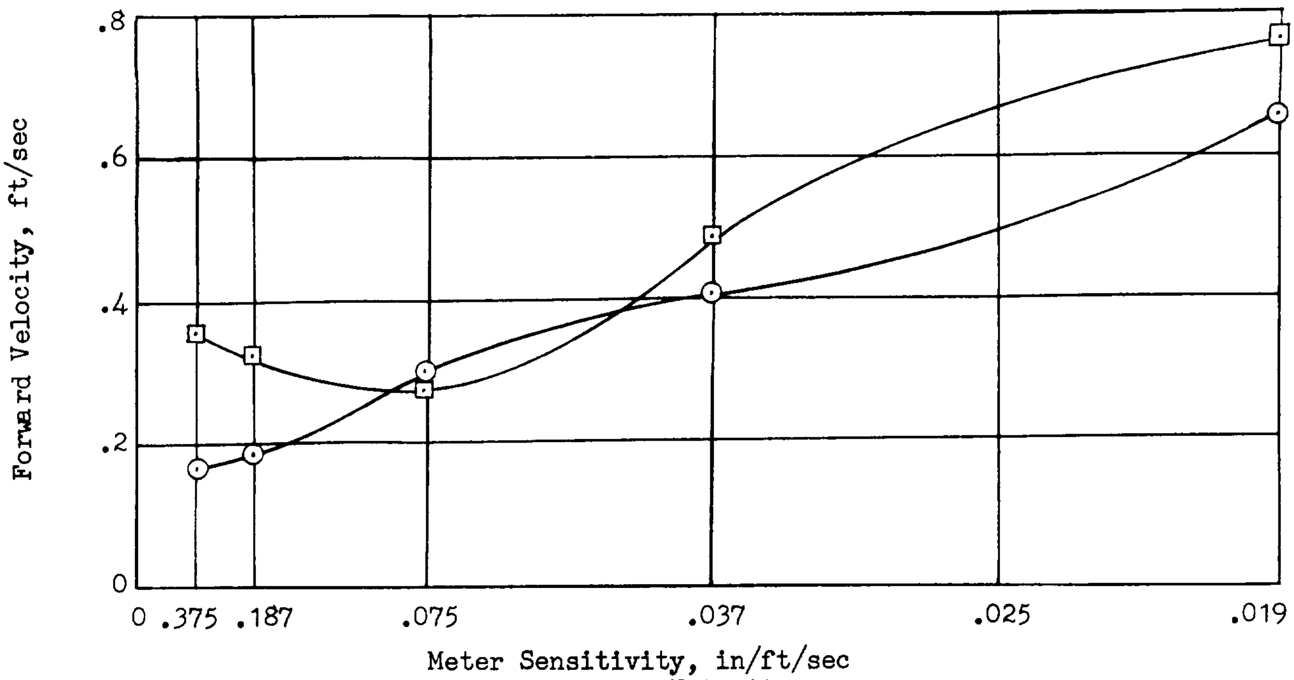


Figure 1.-Forward-Lateral velocity meter faces

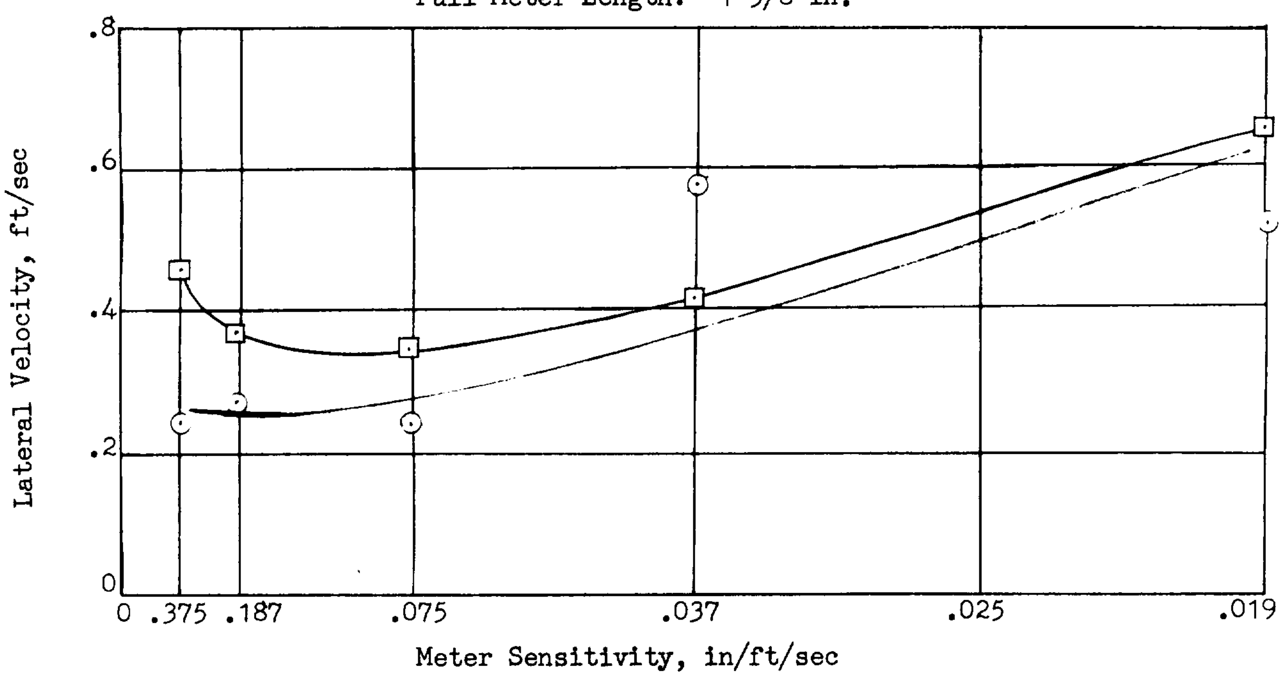


A. Forward Velocity

With radar noise

Without radar noise

Full Meter Length: 1 5/8 in.



B. Lateral Velocity

Touchdown Velocity Error as a Function
of Meter Sensitivity

Figure 2

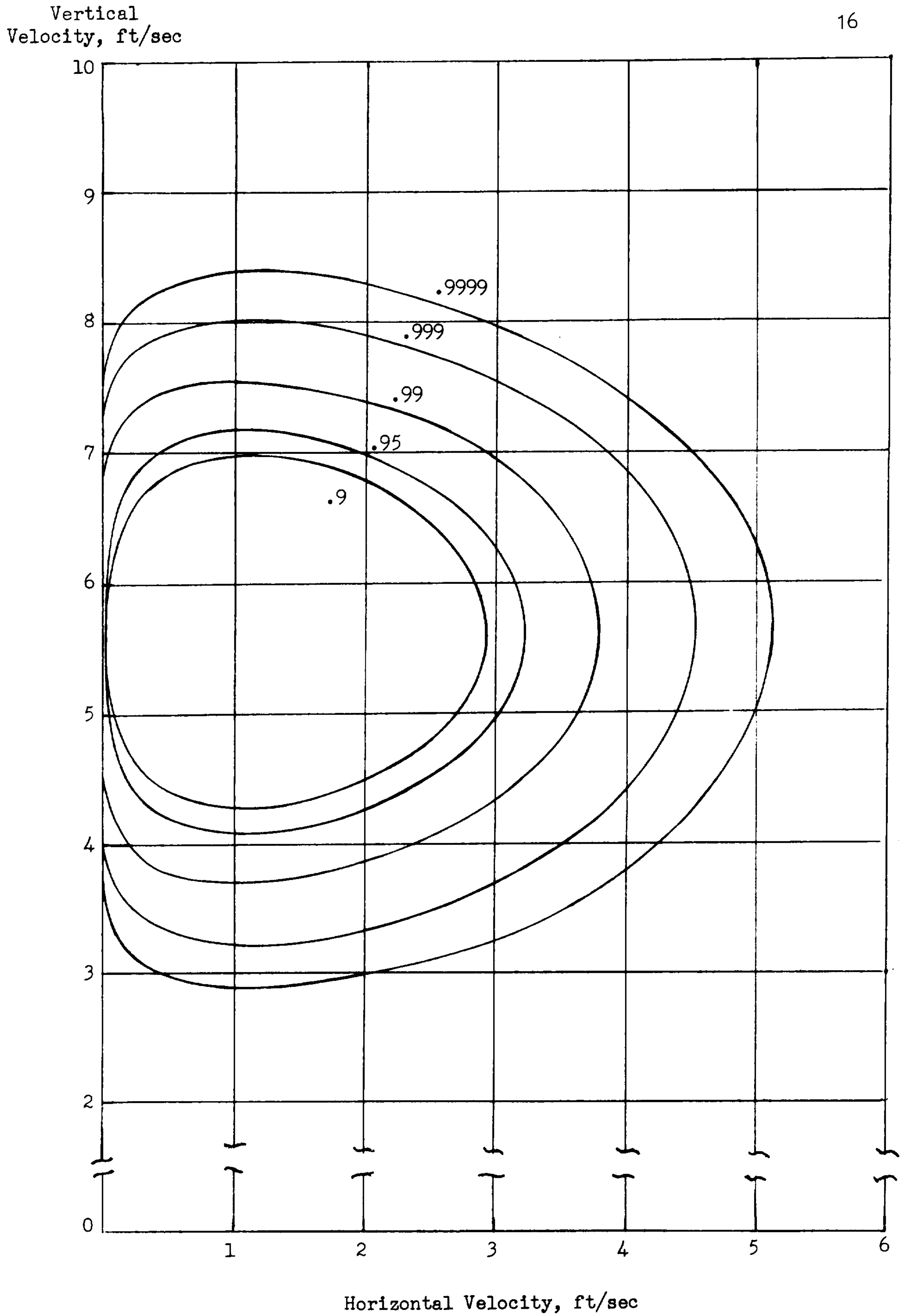


Figure 3.-Touchdown velocity probability contours

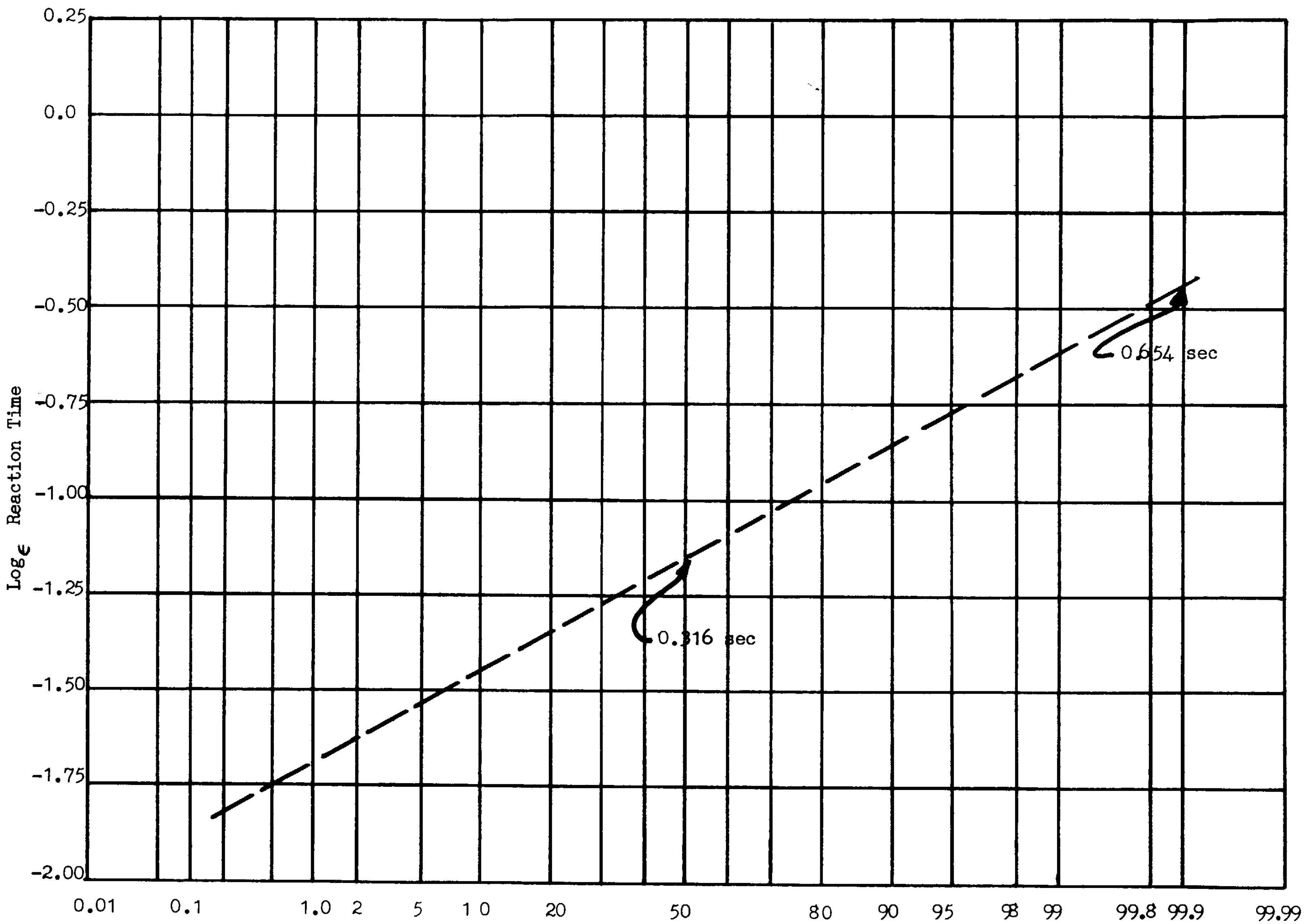


Figure 4 - Probability of Occurrence (%)

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(a) Position of hands

Figure 5.-Pressure suit operation

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(b) Scale change operation

Figure 5.-Concluded

APPENDIX

DESCRIPTION OF SIMULATION

Characteristics of Simulated Vehicle

The initial conditions assumed for the physical parameters of the LEM spacecraft were:

Quantity	Symbol Value
Mass, slugs	456.2
Roll inertia, slug-ft^2	13500
Pitch inertia, slug-ft	11250
Yaw inertia, slug-ft	10850
Product of inertia in the X_b - Z_b plane, slug - ft^2	-312
Product of inertia in the X_b - Y_b plane, slug - ft^2	-41
Product of inertia in the Y_b Z_b plane, slug - ft^2	-208
Distance from origin to c.g. along X_b, ft	.008
Distance from origin to c.g. along Y_b, ft	.117
Distance from c.g. to main engine gimbal in the Z_b direction, ft	4.920

The general configuration of the simulated vehicle is shown in figure A.

Equations of Motion

The equations of motion are essentially those of reference 3. Because the test runs began at low altitude and the characteristics of the vehicle changed very little by touchdown, considerable simplifications in the equations were made. The velocities and positions were computed in the inertial axis with gravity components removed from the forward and lateral velocity equations because the moon was considered flat and non-rotating for this problem. The only other simplification was the X-Y product of the inertia was dropped from the body angular rate equations.

Control System

The attitude control was provided for three modes of operation:

1. Rate command-attitude hold (RCAH)
2. Rate command (RC)
3. Direct thruster operation

The linear pulse ratio modulation used in the RCAH and RC modes was generated from a jet select modulator and logic box. The pitch attitude control system used in the simulation is shown in figure B. The circuitry from the roll and yaw channels was identical. The detent switches switched the mode of the attitude follower circuit so that the output of the follower either followed the input signal or held the last value of the input signal. This system also included an inhibition circuit which prevented the follower circuit from holding the last value of the input signal until the sum of the absolute value of the vehicle attitude rates were below a preselected magnitude; i.e., $|p| + |q| + |r| \leq 2$ deg/sec. The simulation was simplified in that transformation of attitude error signal to the proper body axis rate command was neglected. Large roll and yaw angles were avoided during the simulation so the effect was considered negligible.

The c.g. offset from the main engine thrust vector produces moments about the pitch and roll body axes of the spacecraft which in turn produce steady state error signals in the pitch and roll control axes. For the short time of the test runs in this study, a constant c.g. offset was assumed. Initial values of engine gimbal position were used in the equation to prevent initial moments being created by the c.g. offset. In addition, an initial roll angle of approximately -1 deg was used in the program to prevent an initial lateral acceleration caused by the engine gimbal angle.

Rate of Descent (ROD) Command - The mechanization of the rate of descent mode used during the study is shown in figure C. The ROD was commanded through the descent engine throttle and/or through a two position center-off switch located on the throttle housing. The ROD command mode was activated only when the attitude control switch was in RCAH position and the thrust control switch in the AUTO position, (figure D), but it automatically switched to direct engine thrust control whenever the throttle deflection exceeded 51° (throttle soft stop location).

Simulator Cockpit

The simulator cockpit used in the study consisted of the pilot chairs, attitude controller, throttle, and spacecraft display panel enclosed in a partial modkup of the LEM spacecraft cabin.

Astronaut chairs - The actual spacecraft has a harness-type arrangement to restrain the pilots, but chairs were used in this simulation for pilot comfort. However, the chairs were positioned so that the window view angle was the same as the harness arrangement.

Attitude controller - The attitude controller used to actuate the attitude jets was a prototype of the Block I command module three-axis hand controller.

Throttle - The throttle used in the simulation was a duplicate of the proposed LEM throttle as of August 15, 1963 (figure E). A single linear potentiometer controlled the output voltage for the throttle and full throttle angular range was from 0° to 66°. There was a soft stop at 51° which indicated to the pilots the throttle deflection point where the ROD command mode was disengaged.

Spacecraft display panel - A photograph of the spacecraft display panel used in the simulation is presented in figure D. The instruments shown on the panel are forward and lateral velocity, ΔV, % fuel remaining, % thrust, FDAI, altitude, altitude rate, T/W ratio, and a clock for indicating elapsed time of flight.

The sensitivity of the forward and lateral velocity meter was varied as part of the study. The scale on the altitude and altitude rate meters were to the same sensitivity as the type meters planned for the LEM. The FDAI in the actual LEM displays the output of the Gimbal Attitude Servo Transformation Assembly (GASTA) which in turn is driven by the gimbal angles of the inertial platform. In the simulation, the FDAI was driven by the spacecraft angular position angles θ, ψ , and ϕ. The platform was alined with the target point so that zero reading of the FDAI at the target point indicated that spacecraft X_b axis was alined with the local horizontal and pointed along the lunar equator in the negative direction of the moon rotation. Also, for a zero FDAI reading at the target point, Y_b was in the direction of the positive moon rotation axis and Z_b was directed toward the center of the moon. A zero FDAI reading (ψ , θ, ϕ=0) at the target meant that the body axis was alined with the inertial axes (figure A).

Probe simulation - The simulated vehicle had a probe on each landing gear pad and a logic circuit was developed which required 2 probe contacts before the probe light (figure D) lighted.

Out-The-Window Display

The out-the-window display was generated by the Visual Space Flight Simulator which is an electronic system designed to simulate the visual environment of the space vehicle. The display consists of special purpose digital units and specially designed television display units. The digital units accept numerical data describing the position and attitude rates of the vehicle and performed the computations necessary to produce the appropriate perspective pictures of the environment of the spacecraft with respect to the generated plane (lunar surface). The display units use the results of the computations to produce color pictures on shadow mask television cathode ray tubes which were presented to the pilot through an optical system in the form of a virtual image. The landing site, as established in this study, was 512 feet square and consisted of green and yellow checkerboard squares having an area of 16 feet on a side. The target area was centered in a blue and red area of 2,048 feet square having squares of various sizes. The rest of the plane consisted of a repetitive green and yellow pattern.

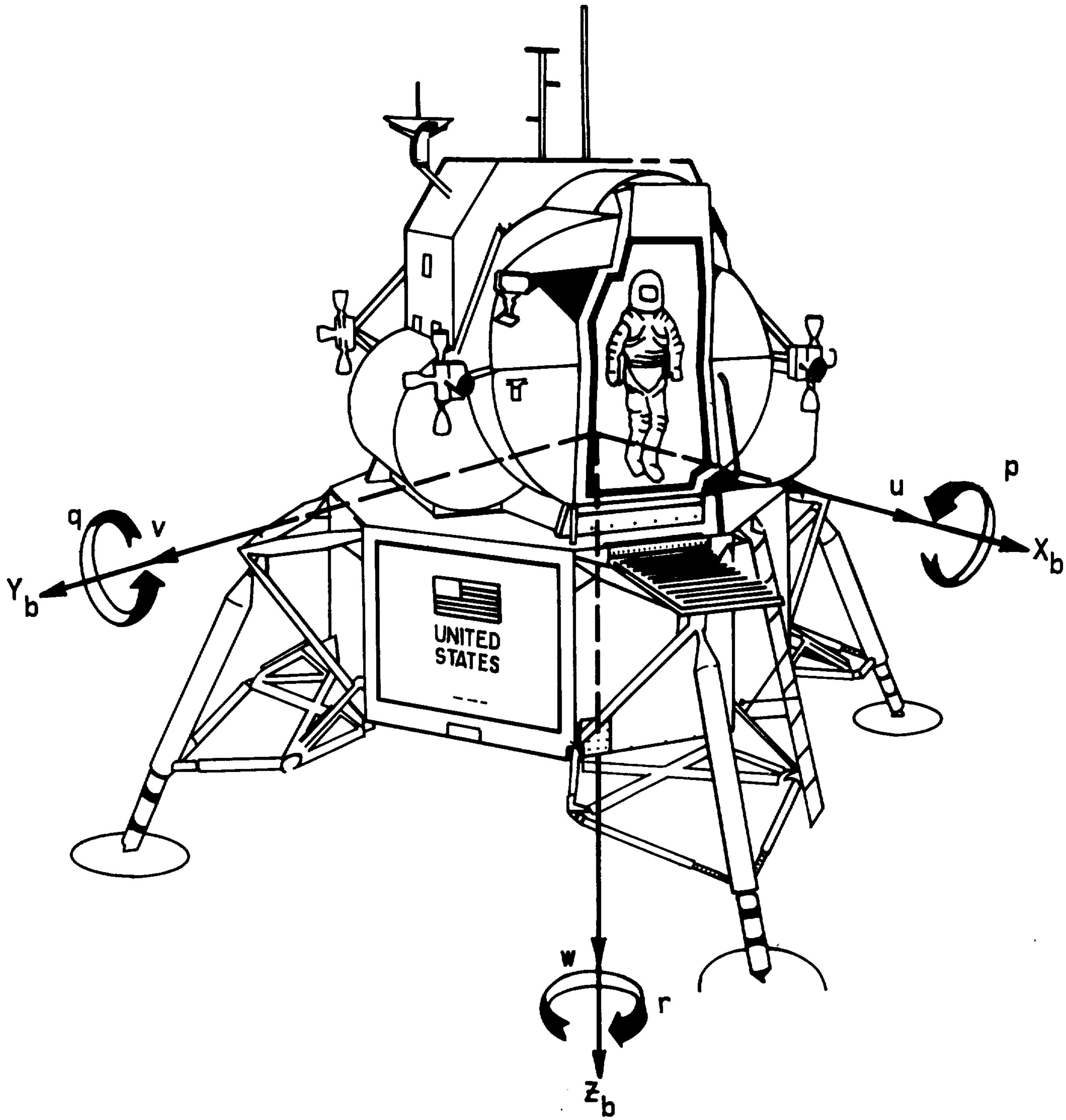

Figure 1.- General configuration of simulated vehicle.

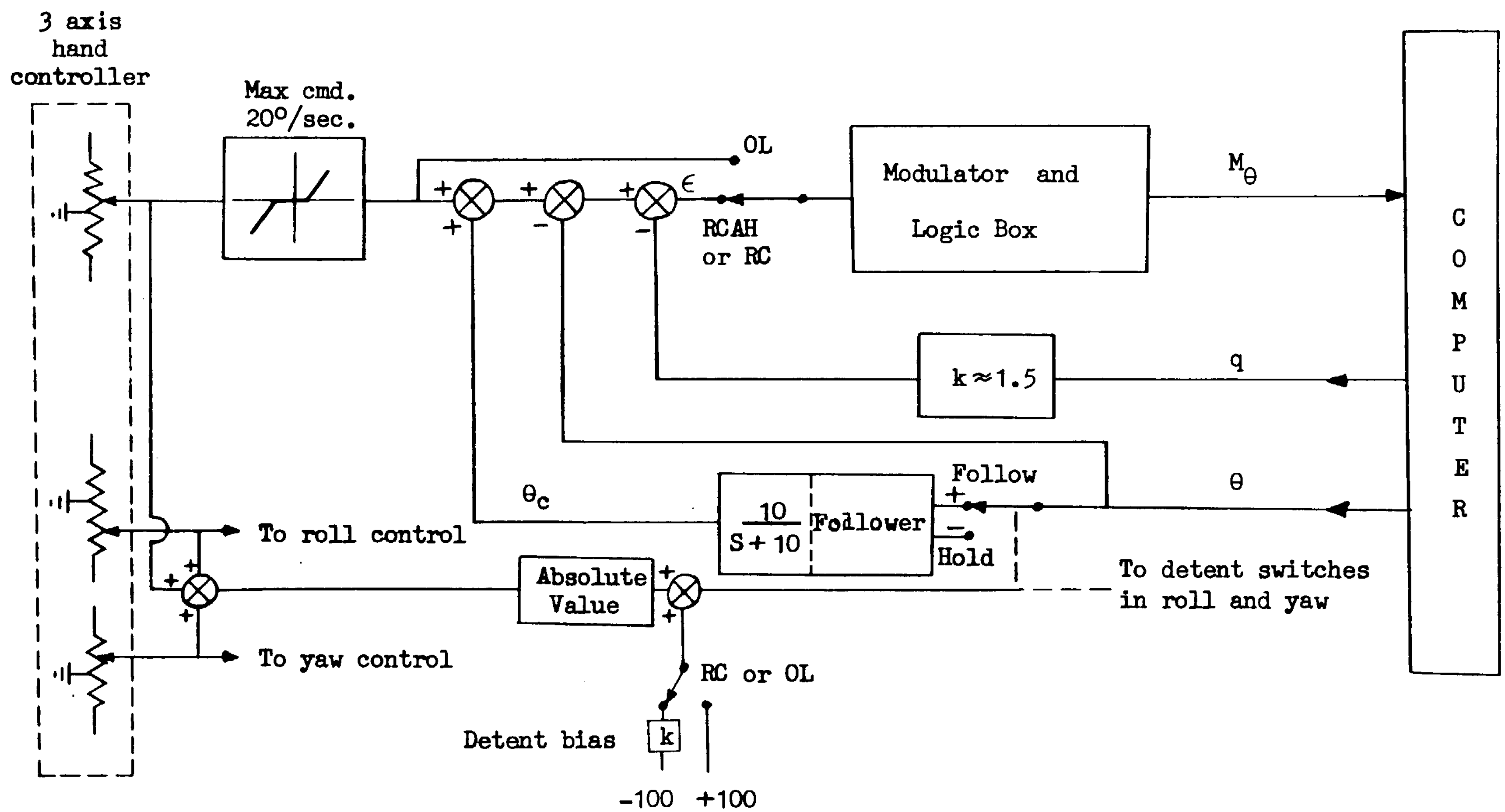

Figure B. - Attitude control system for pitch axis.

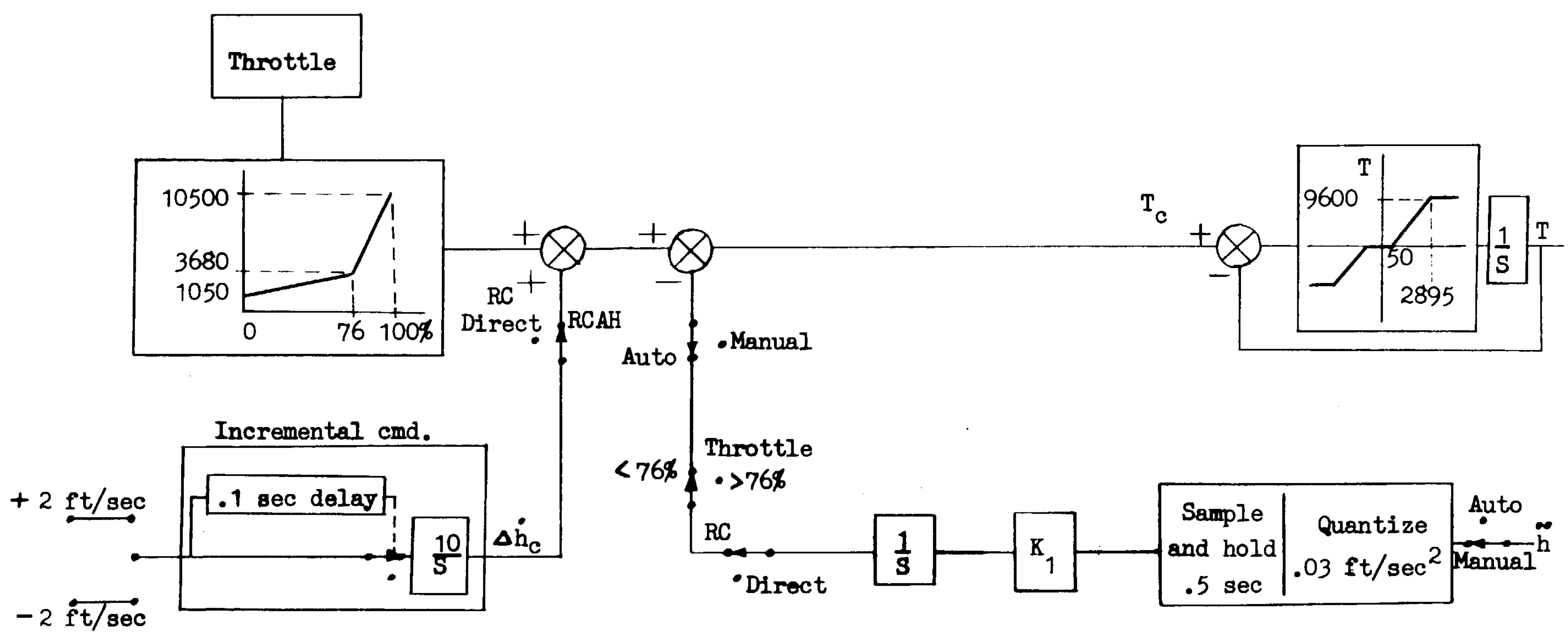

Figure C. - Rate of descent control systems.

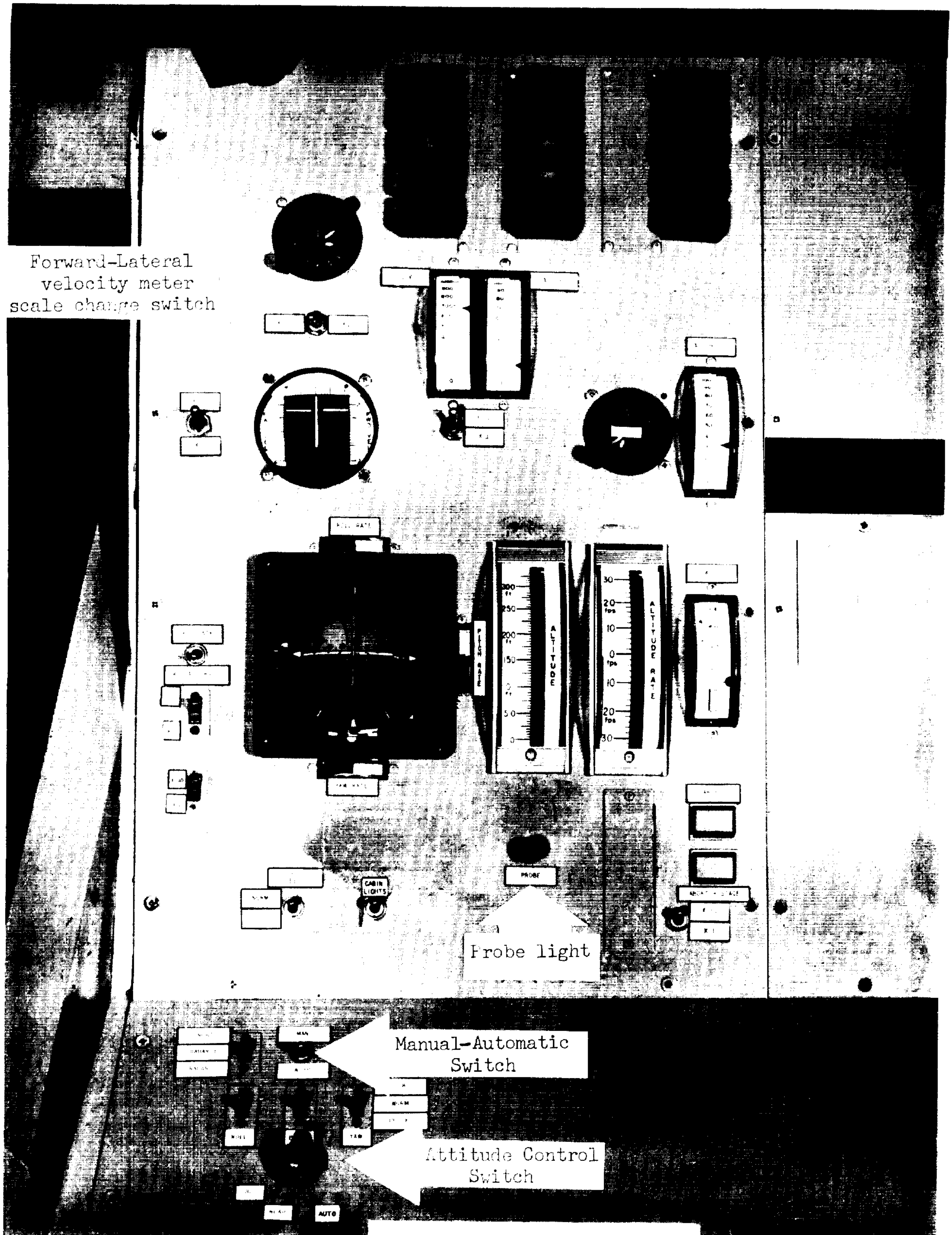

Figure D.-Instrument Panel

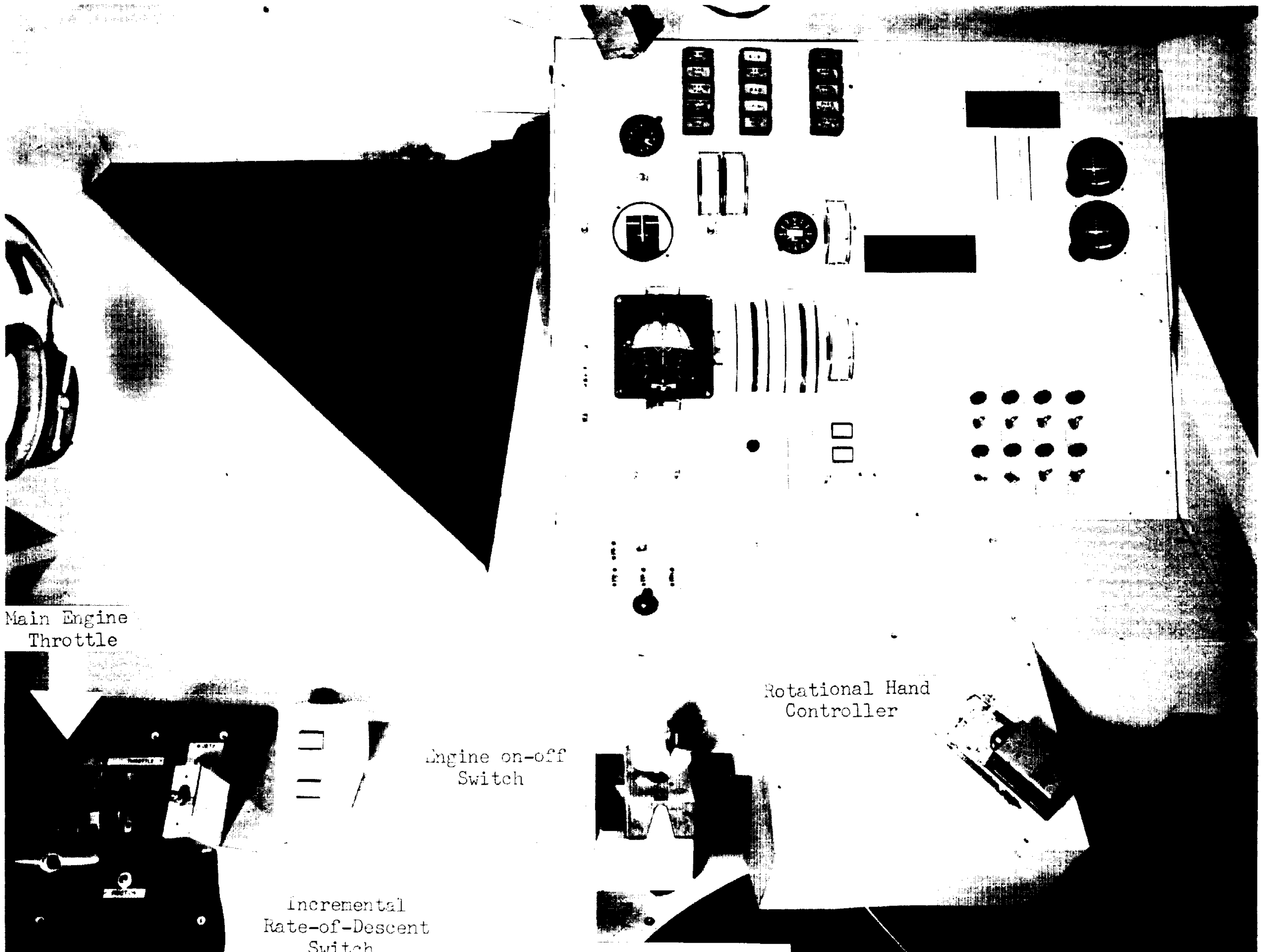

Figure E.-Simulator cockpit