NASA CR-
147589

(NASA-CR-147589) SHUTTLE AUTOLAND SUPPORT PROGRAM Final Report, 1 Nov. 1975 - 31 Mar. 1976 (Sperry Flight Systems, Phoenix, Ariz.) 216 p HC $7.75 CSCL 22B

N76-22268

Unclas
G3/18 26804

SPERRY
FLIGHT SYSTEMS

PHOENIX, ARIZONA 85036

SHUTTLE AUTOLAND SUPPORT PROGRAM FINAL REPORT NOVEMBER 1, 1975 THROUGH MARCH 31, 1976

NASA CONTRACT NO. NAS9-14569

SPERRY
FLIGHT SYSTEMS

PHOENIX, ARIZONA

COPY NO.__________

PRINTED IN U.S.A.

APRIL 1976

PUB. NO. 71-0882-00-00

INTRODUCTION

This report summarizes the results of the Space Shuttle Automatic Landing Support program studies by Sperry Flight Systems from November 1, 1975 to March 31, 1976. The sections to follow were previously issued as NASA Shuttle Autoland Memos with numbers corresponding to the following section numbers:

1: Software Definition Review
(TAEM Pre-final and Autoland Only)

2: Software Definition Review
(KPIT Scheduling)

3: Deletion of Air Data

4: June 1975 Aero Data Update and Check Runs

5: Flat Turn Study

6: Guidance Mode Switching Study

7: Flat Turn Study - Additional Data

8: Trajectory Shaping Study

9: Aero Data Tolerances

10: Turbulence Model Study

SECTION 1

SOFTWARE DEFINITION REVIEW
(TAEM PRE-FINAL AND AUTOLAND ONLY)

AUTOLAND MEMO NO: 1

DATE: 12/2/75

COPY: ________

NASA SHUTTLE

AUTOLAND MEMO

TASK NO: 2

TASK NAME: SOFTWARE DEFINITION REVIEW (TAEM PRE-FINAL AND AUTOLAND ONLY)

The enclosed summary chart compares the most recent Guidance, Control and Navigation Software implementation as of the above date in the Sperry Autoland Simulation and the STA Validation Facility with the following FSSR documents and Change Requests (CR's):

GUIDANCE FSSR JULY 75 plus approved or pending CR No.'s 1207, 1208, 1227, 1230, 1250, 1251, 1264 and FINAL FLARE CR dated 27 Aug 1975, 1191.

CONTROL FSSR AUG 75 plus pending CR concerning KPIT gain scheduling.

NAVIGATION FSSR AUG 75

DISTRIBUTION:

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2. E. Capen (R. I.)
3. J. Williams (Sperry)

COMPARISON SUMMARY FOR FSSR, AUTOLAND AND STA
FROM PRE-FINAL TO FINAL FLARE

GUIDANCE - PITCH

	July 75 FSSR + CR	AUTOLAND	STA
Pre-Final	o Frame Time = 961 ms	o Frame Time = 160 ms	o No Pitch Pre-Final Phase
	o N_z Command	o N_z Command	
	o H, H Dot References	o H, H Dot References	
	o A4 _ LL = -320	o A4 _ LL = -320	o Transition to Pitch Capture (Interface) When in Roll Pre-Final
	o K_H = .001 g/ft	o K_H = .001 g/ft	
	o $K_{H\ Dot}$ = .01 g/ft/sec	o $K_{H\ Dot}$ = .01 g/ft/sec	
	o K_{INT} = .02	o K_{INT} = .02	
	o Δ N_zL.L. = -.75g	o Δ N_zL.L. = -.75g	
	o Δ N_zU.L. = 1.50g	o Δ N_zU.L. = 1.50g	
CAPTURE (INTERFACE)	o Frame Time = 160 ms	o Frame Time = 160 ms	o Frame Time = 50 ms
	o N_z Command	o N_z Command	o Pitch Command
	o H and H Dot Reference	o H and H Dot References	o H and γ References
	o Rectangular Integration Throughout A/L	o Rectangular Integration Throughout A/L	o Rectangular Integration Throughout A/L

GUIDANCE - PITCH

	July 75 FSSR + CR	AUTOLAND	STA
CAPTURE (INTERFACE) (cont)	o Z-Transform Lag Throughout A/L $\frac{1 - EXP\ (-AT)}{1 - Z^{-1}\ EXP\ (-AT)}$	o Z-Transorm Lag Throughout A/L $\frac{1 - EXP\ (-AT)}{1 - Z^{-1}\ EXP\ (-AT)}$	o Z-Transform Lag Throughout A/L $\frac{1 - EXP\ (-AT)}{1 - Z^{-1}\ EXP\ (-AT)}$
	o Following phase transitions always on next pass after criteria met	o This and following phase transitions always on next pass after criteria met	o This and following phase transitions always on present pass
	o Engage Criteria:	o Engage Criteria:	o Engage Criteria:
	$\lvert H_{ERROR} \rvert < 1000$ ft	$\lvert H_{ERROR} \rvert < 1000$	Roll Phase = Prefinal
	$\lvert Y \rvert < 1000$ ft	$\lvert Y \rvert < 1000$	
	$\lvert \gamma_{ERROR} \rvert < 4$ deg	$\lvert \gamma_{ERROR} \rvert < 4$	- No Pitch "PREFINAL" Phase
	$\lvert Q_{ERROR} \rvert < 24$ psf	$\lvert Q_{ERROR} \rvert < 24$	
	$H \geqslant 10{,}000$ ft	$h \geqslant 10{,}000$	
	$X > -38{,}700$ ft	$X > -38{,}700$	- Transition directly from Pitch-TAEM to Pitch-Capture based on Roll Phase

GUIDANCE - PITCH

	July 75 FSSR + CR	AUTOLAND	STA
CAPTURE (INTERFACE) (cont)	or	or	
	$\|H_{ERROR}\| < .19H-900$ ft	$\|H_{ERROR}\| < .19h-900$	
	$\|Y\| < (.18h-800)$ ft	$\|Y\| < .18H-800$	
	$\|\gamma_{ERROR}\| < .0007h-3$ deg	$\|\gamma_{ERROR}\| < .0007h-3$	
	$\|Q_{ERROR}\| < 24$ psf	or	
	$10{,}000 > h > 5000$ ft	$h < 5000$	
	$X > -38{,}700$ ft		
	or/and		
	Crew Engage	o Engages Automatically	o Engages Automatically
	o No Open-Loop Predicts	o No Open-Loop Predicts	o Alpha and V DOT Predicts
	o G-Limit = 1g	o G-Limit = 1g	o G-Limit = 7/8g
	o No Pre-Final/Capture Fader	o Pre-Final/Capture Fader	o Alpha Predict G-Limit = 1/2g

GUIDANCE - PITCH

	July 75 FSSR + CR	AUTOLAND	STA
STEEP	o Frame Time = 160 ms	o Frame Time = 160 ms	o Frame Time = 50 ms
	o N_z Command	o N_z Command	o Pitch Command
	o H and H Dot References	o H and H Dot References	o H and γ References
	o Engage Criteria:	o Engage Criteria:	o Engage Criteria:
	$\lvert\gamma_{ERROR}\rvert < 2$ deg	$\lvert\gamma_{ERROR}\rvert < 2$	$\lvert\gamma_{ERROR}\rvert < 2$
	$\lvert H_{ERROR}\rvert < 50$ ft	$\lvert H_{ERROR}\rvert < 50$	$\lvert H_{ERROR}\rvert < 50$
	or	or	or
	$\lvert\gamma_{ERROR}\rvert < 2$ for 4 sec	$\lvert\gamma_{ERROR}\rvert < 2$ for 4 sec	$\lvert\gamma_{ERROR}\rvert < 2$ for 4 sec
	o No Open-Loop Predicts	o No Open-Loop Predicts	o Alpha and V Dot Predicts
	o ALT Gain_{STEEP} = ALT Gain_{CAPT}	o ALT Gain_{STEEP} = ALT Gain_{CAPT}	o ALT $\text{Gain}_{STEEP} \neq$ ALT Gain_{CAPT}
	o H Dot Gain_{STEEP} = H Dot Gain_{CAPT}	o H Dot Gain_{STEEP} = H Dot Gain_{CAPT}	o ATT $\text{Gain}_{STEEP} \neq$ ATT Gain_{CAPT}
	o G-Limit = 1g	o G-Limit = 1g	o G-Limit = 7/8g
			o Alpha Predict G-Limit = 1/2g
	o HGEAR = $1000 + 40\,(VEAS_F - VREF + VG - VTFILT\cos\gamma_{AIR})_{-700}^{900}$	o HGEAR = $1000 + 40\,(VEAS_F - VREF + VG - VTFILT\cos\gamma_{AIR})_{-700}^{900}$	o HGEAR = $1000 + 40\,(520 - VTM)_{-700}^{900}$

GUIDANCE - PITCH

	July 75 FSSR + CR	AUTOLAND	STA
STEEP (cont)	o K_H = .0036 g/ft	o K_H = .0036	
	o K_{HDOT} = .0109 g/ft/sec	o K_{HDOT} = .0109	
	o K_{INT} = .05	o K_{INT} = .05	
PULL-UP/ SHALLOW	o Frame Time = 160 ms	o Frame Time = 160 ms	o Frame Time = 50 ms
	o N_z Command	o N_z Command	o Pitch Command
	o H, H Dot References	o H, H Dot References	o H, γ References
	o ATT. Rate Predict Step	o ATT. Rate Predict Exponential Fade in	o ATT. Rate, Attitude and V Dot Predicts
	o τ_{PRED} = 2 sec	o τ_{PRED} = 2	o τ_{PRED} = 4
	o H Decay = 70' (Constant)	o H Decay = 70' (Constant)	o H Decay = HREFT - HREF (Variable) on Last Pass through circularization
	o G-Limit = 1g	o G-Limit = 1g	o G-Limit = 7/8g
	o Body Flap Retracted to Landing Position	o Body Flap Retracted to -11.7 deg	o No Body Flap Retraction
			o KPRED = $(VTNOM-VTM/13.3)^{+1}_{-2}$ +6

GUIDANCE - PITCH

	July 75 FSSR + CR	AUTOLAND	STA
FINAL FLARE	o Frame Time = 160 ms	o Frame Time = 160 ms	o Frame Time = 50 ms
	o N_z Command	o N_z Command	o Pitch Command
	o $HDOT_{TD1}$ = 4	o $HDOT_{TD1}$ = 4	o $HDOT_{TD}$ = -3
	o $HDOT_{TD2}$ = -4	o $HDOT_{TD2}$ = -4	
	o TAU FLR = 5	o TAU FLR = 5	
	o Open-Loop Predict Using Integrator	o Open-Loop Predict Using Integrator	o "Closed-Loop" Predict using HDOT
			o HDDOT ERROR COMMAND Used
	o K_{INT} = .1	o K_{INT} = .1	o K_{INT} = .4
	o G-Limit = 1g	o G-Limit = 1g	o G-Limit = 7/8g

GUIDANCE - ROLL

	July 75 FSSR + CR	AUTOLAND	STA
PRE-FINAL	o Frame Time = 961 ms	o Frame Time = 160 ms	o Frame Time = 50 ms
	o KY = .02	o KY = .02	o KY = .02
	oKYDOT = .4	o KYDOT = .4	o KYDOT = .4
	o Y Limit = 4000 = (80/.02)	o Y Limit = 4000	o No Phi Limits
	o PHILIM = 60 deg	o PHILIM = 60 deg	
	o Engage Criteria:		o Engage Criteria:
	HAC Phase		$\lvert\psi\rvert \leq 20$ deg
	$\lvert\psi\rvert \leq 20$ deg		RPRED $<$ 50,892'
	<u>or</u>		
	ACQ Phase		
	RNG PRED $<$ RNG_FZ3		
RUNWAY CAPTURE	o Engage Criteria		
	Same as Pitch Capture		
RUNWAY TRACK	o Frame Time = 160 ms	o Frame Time = 160 ms	o Frame Time = 50 ms
	o KY = .1 deg/ft	o KY = .1	o KY = .1
	o KYDOT = 1 deg/ft/sec	o KYDOT = 1	o KYDOT = 1
	o YLIMIT = 500 ft	o YLIMIT = 500	o YLIMIT = 500
	o Y_{INT} = .01	o Y_{INT} = .01	o Y_{INT} = .01

GUIDANCE - ROLL

	July 75 FSSR + CR	AUTOLAND	STA
RUNWAY TRACK (cont)	o $Y_{INT_{LIM}} = 50$ ft	o $Y_{INT_{LIM}} = 50$	o $Y_{INT_{LIM}} = 50$
	o Fader between Pre-Final and A/L	o Fader between Pre-Final and A/L	o No Fader
	o Phi-Max: 30, $h > 5000$	o Phi-Max: 30, $h > 5000$	o Phi-Max: 30, $h > 5000$
	15, $h < 5000$	15, $h < 5000$	15, $h < 5000$
	90, Manual	90, Manual	60, $h > 15{,}000$
		45, $H > 15{,}000$	
	o Engage Criteria:	o Engage Criteria:	
	$\|\psi\| < 2$ deg	$\|\psi\| < 2$ deg	
	$\|Y\| < 50$ ft	$\|Y\| < 50$ ft	
	or	or	
	$\|\psi\| < 2$ deg for 4 sec	$\|\psi\| < 2$ deg for 4 sec	
FLAT TURN	o Roll Command A/L = 0	o No Flat turn Logic	o No Flat turn Logic
	o Yaw Command (Rudder) = Roll Command		

SPEED CONTROL

	July 75 FSSR	AUTOLAND	STA
PRE-FINAL	o Frame Time = 961 ms	o Frame Time = 160 ms	o No Pre-Final Speed Brake Law CAPT/STEEP Speed Law Used
	o QREF = 300 psf	o QREF = 300 psf	$V_{G_{REF}}$ = 460 ft/sec
	o K_Q = 3 deg/psf	o K_Q = 3 deg/psf	
	o DSB_{IC} = 60 deg	o DSB_{IC} = 60 deg	o DSB_{1C} = $DSB_{TAEM\ Last\ Pass}$
			o K_{DSB} = 2 deg/ft/sec
			o $K_{I_{SB}}$ = .1 sec
CAPTURE/ STEEP	o Frame Time = 160 ms	o Frame Time = 160 ms	o Frame Time = 50 ms
	o $V_{EAS_{REF}}$ = 503 ft/sec	o $V_{EAS_{REF}}$ = 503	
	o K_{DSB} = 2 deg/ft/sec	o K_{DSB} = 2	
	o $K_{I_{SB}}$ = .1 sec	o $K_{I_{SB}}$ = .1	
	o DSB_{MIN} = 0	o DSB_{MIN} = 0	o DSB_{MIN} = 0
	o SB_{RATE} = -5 deg/sec	o SB_{RATE} = -5	o SB_{RATE} = -5
	o RHO Sampled Every 960 msec from Air Data	o RHO Look-up Table Every 160 ms	o RHO Look-up Table Every 50 ms
	o Filter = 10 rad/sec	o Filter = 10 rad/sec	o Filter = 10 rad/sec

SPEED CONTROL

	July 75 FSSR	AUTOLAND	STA
PULLUP	o Ramp SB to DSB_{MIN} (Logic in SB Routine)	o Ramp SB to DSB_{MIN} (Logic in SB Routine)	o Ramp SB to DSB_{MIN} (Logic in SB Routine)

CONTROL - PITCH

Aug 75 FSSR + CR	AUTOLAND	STA
o Frame Time = 40 ms, inputs at various rates	o Frame Time = 40 ms	o Frame Time = 50 ms
o N_z Autoland Command	o N_z Autoland Command	o θ Autoland Command
o N_z Auto TAEM Command	o N_z Auto TAEM Command	o θ Auto TAEM Command
o No Faders	o No Faders	o Two Faders
o CSL Limits: (Absolute)	o CSL Limits: (Incremental)	o CSL Limits: (incremental)
$N_z Max_{CSS}$ = 3.75g	$N_z Max_{CSS}$ = 3	$N_z Max_{CSS}$ = 3
$N_z Max_{AUTO}$ = 2.5g	$N_z Max_{AUTO}$ = 1.5	$N_z Max_{AUTO}$ = 1.5
$N_z Min$ = -1g	$N_z Min$ = -2	$N_z Min$ = -2
$Alpha_{Max}$ = 20 deg	$Alpha_{Max}$ = 20	$Alpha_{Max}$ = 20
$Alpha_{Min}$ = -4 deg	$Alpha_{Min}$ = -4	$Alpha_{Min}$ = -4
o Elevon Feedback Lag = 1 rad/sec	o Elevon Feedback Lag = 1 rad/sec	o Elevon Feedback Lag = 1.5 rad/sec
o Lags, Washouts and Higher Order Filters are Bi-linear Transform	o Lags and Washouts are Z-Transform, Higher Order Filters are Bi-Linear Transform	o Lags and Washouts are Z-Transform
o Pre-Separation Logic for ALT	o No Pre-Separation Logic	o No Pre-Separation Logic

CONTROL - PITCH

Aug 75 FSSR + CR	AUTOLAND	STA
o KPIT = 14.73, $M > .45$; 30, $M < .35$	o KPIT = 14.73, $M > .53$; 30, $M < .43$	o KPIT = 220
o Q Feedback *1	o Q Feedback *1	o Q Feedback *3
o N_z Error Sent to Flight Director	o No N_z Error to Flight Director	o No N_z Error to Flight Director

CONTROL

	Aug 75 FSSR + CR	AUTOLAND	STA
ROLL	o Frame Time = 40 ms, inputs at various rates	o Frame Time = 40 ms	o Frame Time = 50 ms
	o Lags and Higher Order Filters are Bi-linear Transform	o Lags are Z-Transform	o Lags are Z-Transform
	o Rectangular Integration	o Rectangular Integration	o Rectangular Integration
	o No Faders	o No Faders	o Two Faders
	o Integral Command Limit = 5 deg	o Integral Command Limit = 10 deg	o Integral Command Limit = 10 deg
	o Pre-Separation Logic	o No Pre-Separation Logic	o No Pre-Separation Logic
	o Phi Error Sent to Flight Director	o No Phi Error Sent to Flight Director	o No Phi Error Sent to Flight Director
	o $K_\phi = 1.5$, $K_p = 1$, $K_I = .2$ $K_{ROLL} = 85$ if Mach < 1.25	o $K_\phi = 1.5$, $K_p = 1$, $K_I = .2$ $K_{ROLL} = 85$	o $K_\phi = 1.5$, $K_p = 1$, $K_I = .2$ $K_{ROLL} = 85$
YAW	o Frame Time = 40 ms, inputs at various rates	o Frame Time = 40 ms	o Frame Time = 50 ms
	o Lags and Higher Order Filters are Bi-Linear Transform	o Lags are Z-Transform	o Lags are Z-Transform
	o No Rudder Fader	o No Rudder Fader	o Rudder Fader Included

CONTROL

	Aug 75 FSSR + CR	AUTOLAND	STA
YAW (cont)	o No Gear Dependent Gains	o No Gear Dependent Gains	o Body Axis Correction Factor is Gear Dependent
	o Rudder Command Lim = 27.1 deg	o Rudder Command Limit = 27.1 deg	o Rudder Command Limit = 25 deg
	o $K_r = 1$, $K_{G\phi} = 1$, $K_{NY} = 20$	o $K_r = 1$, $K_{G\phi} = 1$, $K_{NY} = 20$	o $K_r = 1$, $K_{G\phi} = 1$, $K_{NY} = 20$

NAVIGATION - PRE-LANDING FILTER (IMU, TACAN, BARO)

Aug 75 FSSR	AUTOLAND	STA
o Frame Time = 4 sec	o Frame Time = 160 ms	o Frame Time = 50 ms
o 12-State Vector and covariance matrix update using complete Kalman Filter	o Same as STA	o Six-State Filter with gains multipilied by constant time interval = Frame Time
		o Δ R, Δ ψ, Δ h_{BARO} measurement residuals used in Kalman-form complementary filter transformed to Δ X, Δ Y, Δ Z form using exact partial derivative matrix.
o V_X, V_Y, V_Z and Δ V_X, Δ V_Y, Δ V_Z measurement residuals from IMU used for time update (position and velocity propagation) including misalignments.		o V_X, V_Y, V_Z, $\ddot{h}$ and Δ V_X, Δ V_Y measurement residuals from IMU used for time update.
o Begin Landing Filter when:	o Begin MSBLS processing and Landing Filter when:	o Begin MSBLS processing and Landing Filter when:
High-Rate Autoland Guidance has begun	$h < 12,000$ ft	$h < 15,000$ ft
<u>and</u>		<u>and</u>
$h < 12,000$ ft		MSBLS Range is valid
<u>and</u>		<u>and</u>
MSBLS is valid		MSBLS Azimuth is valid

NAVIGATION - PRE-LANDING FILTER (IMU, TACAN, BARO)

Aug 75 FSSR	AUTOLAND	STA
<u>and</u> Transition to Autoland is not crew inhibited <u>or</u> Nominal Condition for RALT usage has begun		

NAVIGATION - LANDING FILTER (IMU, MSBLS, RALT)

Aug 75 FSSR	AUTOLAND	STA
o Frame Time = 160 ms (Filter) 2 sec (MSBLS) 160 ms (RALT)	o Frame Time = 160 ms	o Frame Time = 50 ms
o R ALT replaces MSBLS elevation data at $X < 4000$ ft ($h \approx 400$ ft). Gain vector for RALT velocity components ($\dot{h}$) is ramped from Ø to nominal value by $X < 3350$ ft ($h \approx 250$ ft)	o RALT begins to replace MSBLS-derived h at h = 400 ft and MSBLS-derived h is blended out by h = 100 ft	o RALT begins to replace MSBLS at h = 400 ft and MSBLS ΔZ is blended out by h = 100 ft
o Six-State Filter with gains scaled by range and time since last MSBLS and IMU update	o Six-State Filter with gains multiplied by constant time = Frame Time	o Six-State Filter with gains multiplied by constant time interval = Frame Time
o Δ_{RNG}, Δ_{AZ}, Δ_{EL}, Δ_{RALT} measurement residuals used in Kalman-form complementary filter transformed to ΔX, ΔY, ΔZ, ΔZ, form respectively using range scaling	o Δ_X, Δ_Y, Δ_Z state residuals used in complementary filter directly, after X, Y, Z computed from combination of AZ, EL, RNG.	o Δ_{AZ}, Δ_{EL}, Δ_{RNG}, Δ_{RALT} measurement residuals used in Kalman-form complementary filter transformed in combination to ΔX, ΔY, ΔZ form using simplified partial derivative matrix.
o V_X, V_Y, V_Z and ΔV_X, ΔV_Y, ΔV_Z measurement residuals from IMU used for time update (position and velocity propagation) including misalignments.	o V_X, V_Y, $\ddot{h}$ from IMU used in complementary filter	o V_X, V_Y, V_Z, $\ddot{h}$ and ΔV_X, ΔV_Y measurement residuals from IMU used for time update.

NAVIGATION - LANDING FILTER (IMU, MSBLS, RALT)

Aug 75 FSSR	AUTOLAND	STA
o Continue to call Pre-Landing Filter but do not process TACAN or BARO Altitude	o No TACAN or BARO Altitude	o No TACAN or BARO Altitude

SECTION 2

SOFTWARE DEFINITION REVIEW
(KPIT SCHEDULING)

AUTOLAND MEMO NO: 2

DATE: 12 Dec 75

COPY: ______

NASA SHUTTLE

AUTOLAND MEMO

TASK NO: 2

TASK NAME: SOFTWARE DEFINITION REVIEW

This memo discusses the impact of a pending CR concerning the Mach scheduling of gain KPIT in the pitch control system.

As shown in Autoland Memo No. 1 on the page labeled CONTROL-PITCH, the FSSR + CR has KPIT ramping from 14.73 to 30 between Mach 0.45 and 0.35. The Sperry Autoland baseline has KPIT ramping from 14.73 to 30 between Mach 0.53 and 0.43. This latter scheduling as recommended by Rockwell was acceptable primarily because the ramping to the higher KPIT gain was completed earlier, prior to exponential capture. The FSSR + CR gain scheduling, however, is not acceptable as the following figures demonstrate:

Figure 1 shows the h vs $\dot{h}$ plot for a constant KPIT value of 22.09 throughout Autoland, which is 1.5 times the present value of 14.73 and was recommended by Sperry during a gain/phase margin study.

Figure 2 shows similar plot for KPIT ramping from 14.73 to 30 between Mach 0.53 and 0.43 as originally recommended by Rockwell. Both are similar and adequately track the shallow G/S as shown by the constant $\dot{h}$ value prior to final flare at about 80 feet. KPIT is constant by the time of exponential capture as desired.

Figure 3 shows a plot for KPIT ramping to 30 between Mach 0.45 and 0.35, as most recently recommended by Rockwell. It can be seen that shallow tracking is poor. This is primarily because the ramping of KPIT is being done later than the previous schedule and is occurring during the critical period of exponential capture. (See Figure 4 for a time history of Mach.)

Thus, it is concluded that the latest KPIT scheduling does not meet the guidance requirements for shallow G/S tracking and stabilization.

Figure 4 is an Autoland time history for KPIT = 22.09, and Figure 5 is also a time history but for KPIT ramped between Mach 0.53 and 0.43. The γ trace of Figure 4 shows adequate tracking of the 3^o shallow glide slope and Figure 5 is better yet. Figure 6 is for KPIT ramped between Mach 0.45 and 0.35. The γ trace shos the flight path angle does not track the shallow glide slope.

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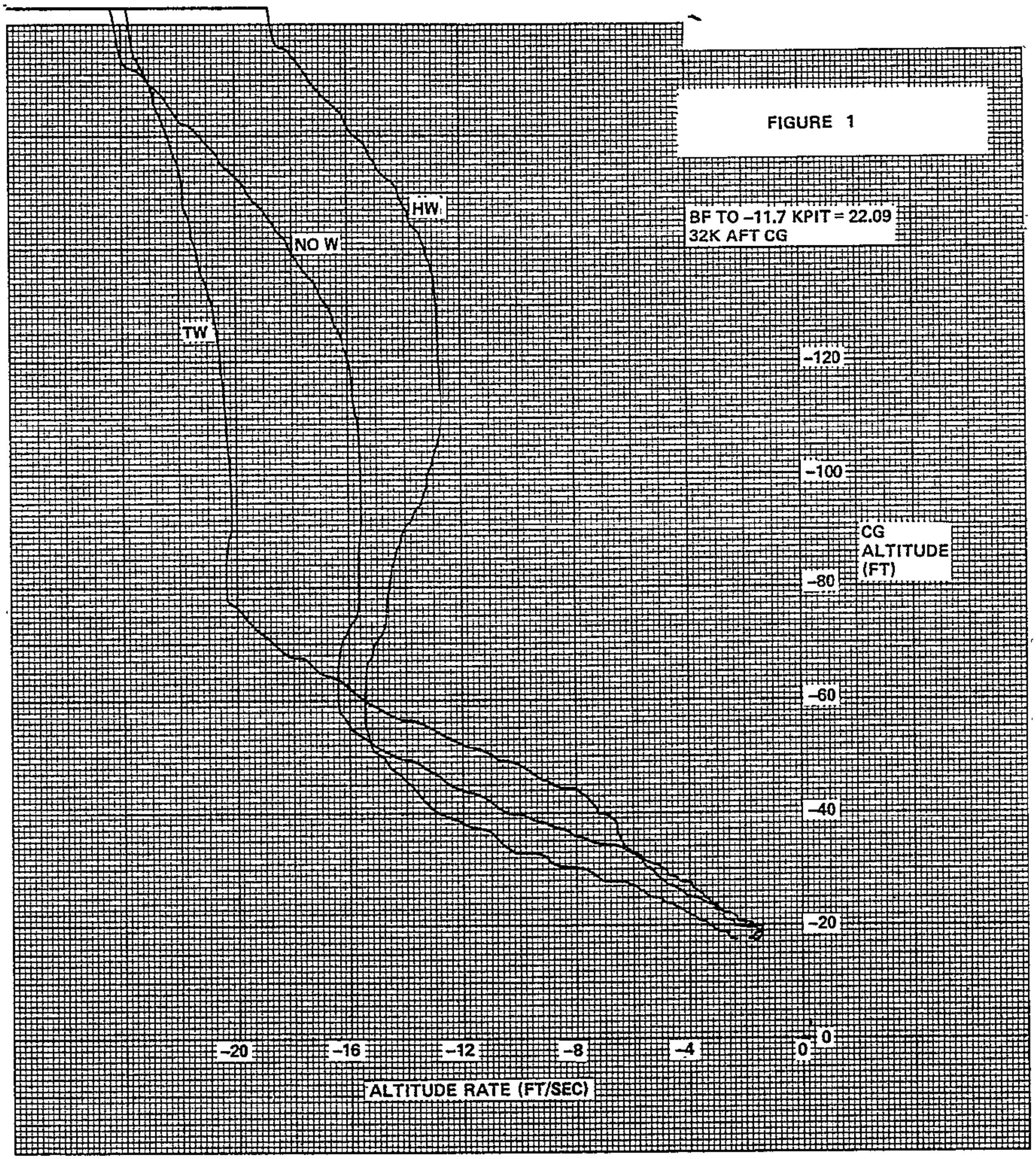
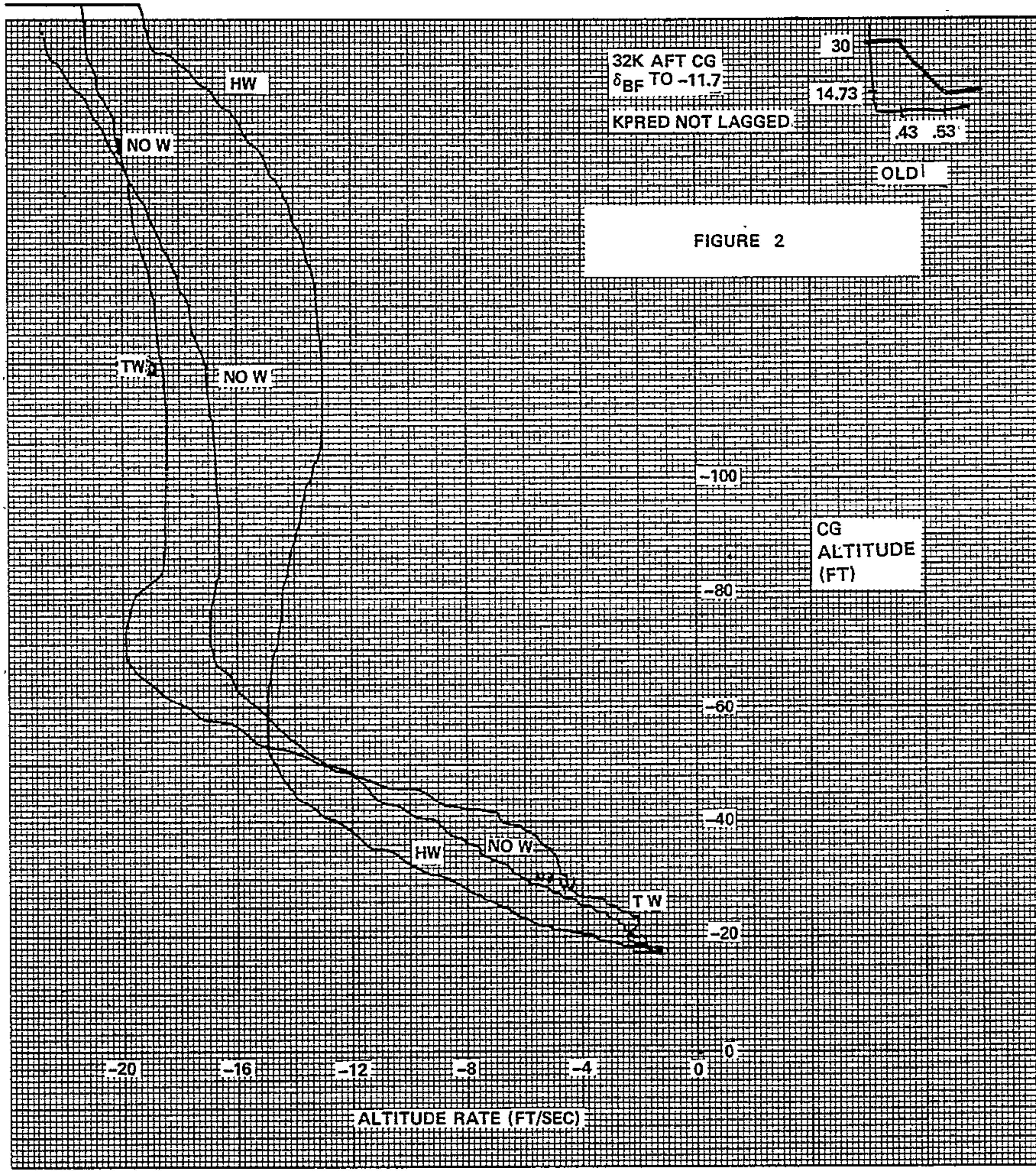
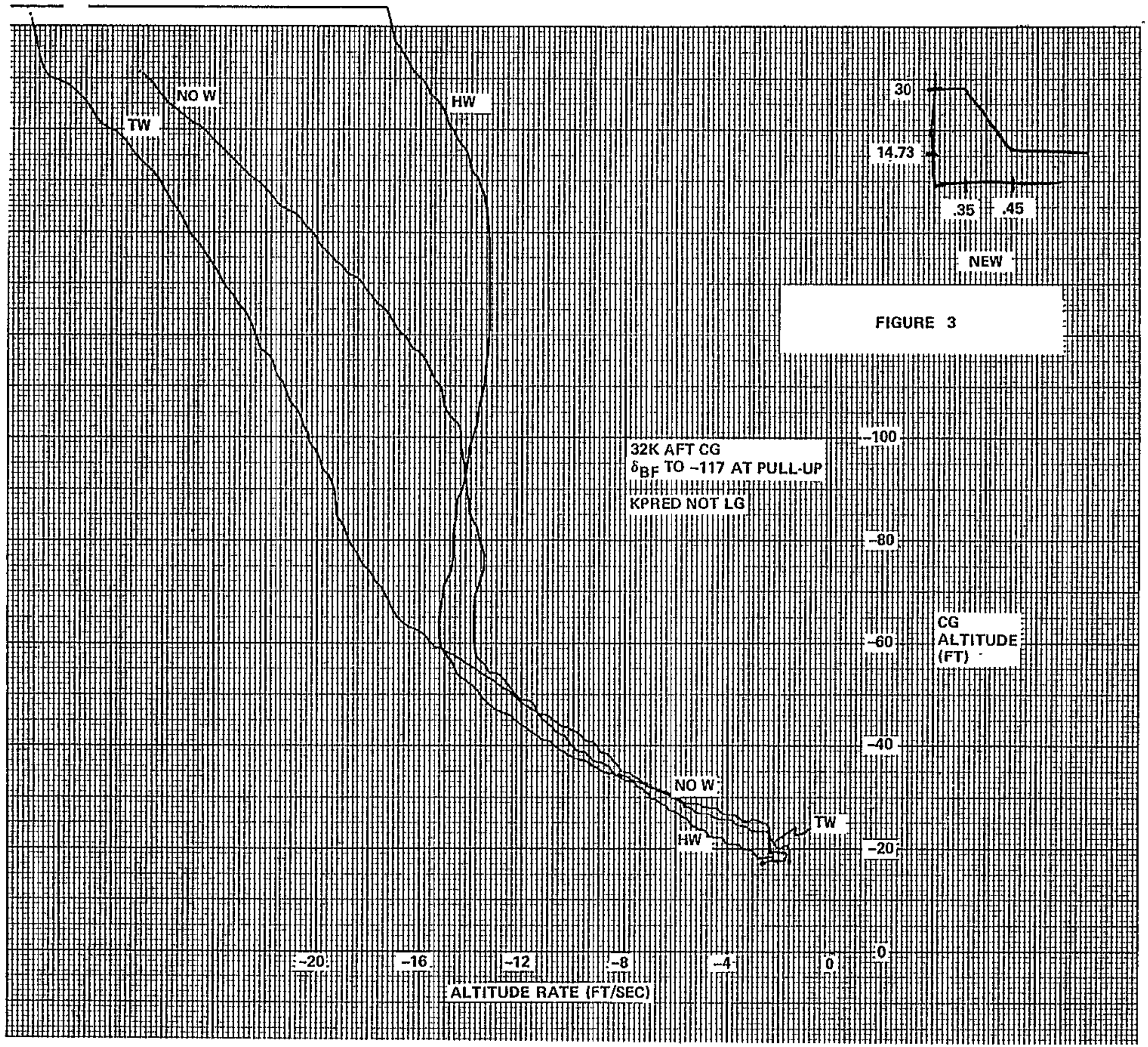


FIGURE 1
BF TO –11.7 KPIT = 22.09
32K AFT CG
HW
NO W
TW
–120
–100
CG
ALTITUDE
(FT)
–80
–60
–40
–20
0
–20
–16
–12
–8
–4
0
ALTITUDE RATE (FT/SEC)



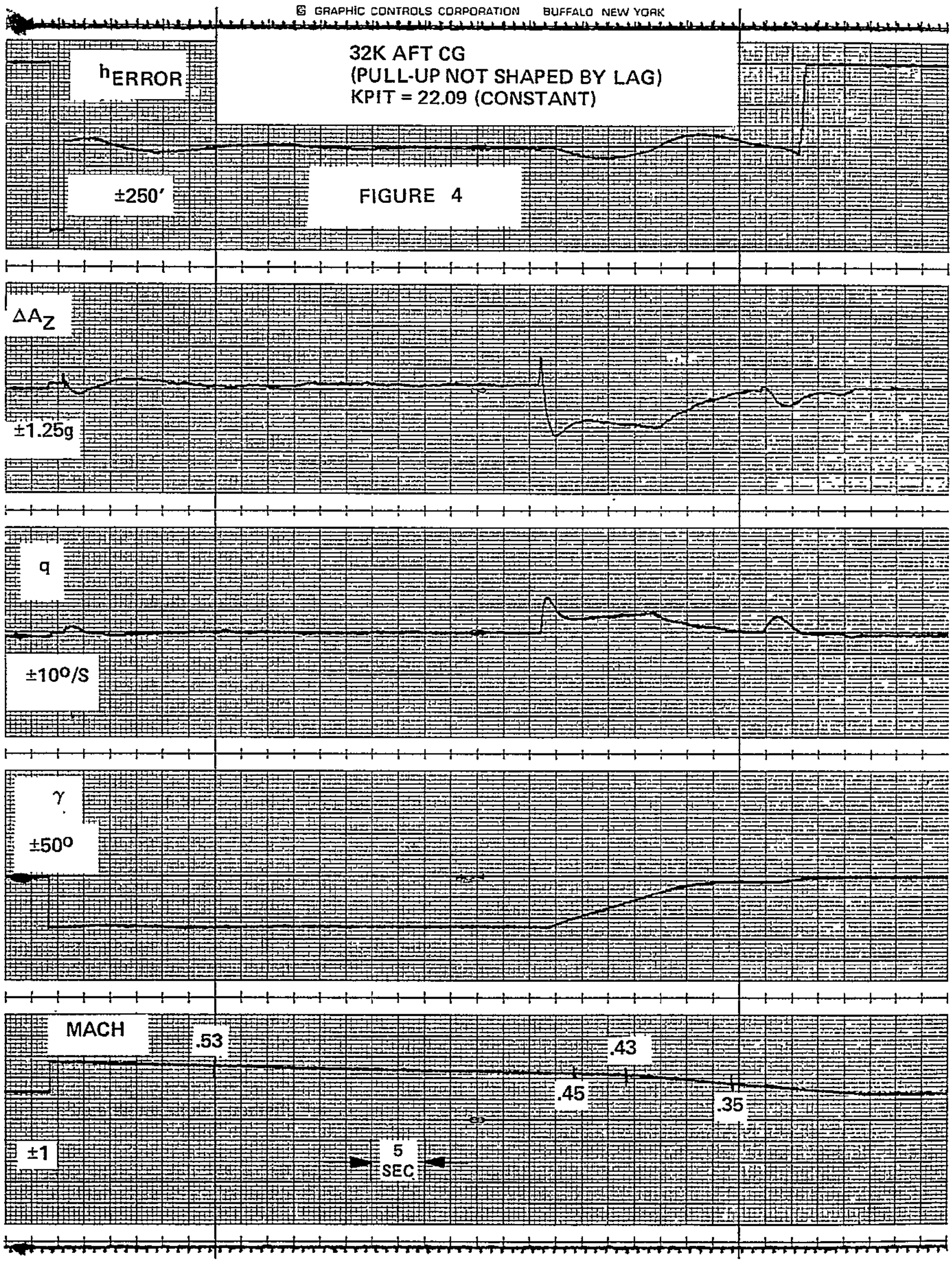
32K AFT CG
δ_{BF} TO −11.7
KPRED NOT LAGGED
30
14.73
.43
.53
OLD
HW
NO W
TW
NO W
HW
NO W
T W
−100
−80
−60
−40
−20
0
CG
ALTITUDE
(FT)
−20
−16
−12
−8
−4
0
ALTITUDE RATE (FT/SEC)

FIGURE 2



NO W
TW
HW
30
14.73
.35
.45
NEW
32K AFT CG
δ_{BF} TO −117 AT PULL-UP
KPRED NOT LG
−100
−80
−60
CG
ALTITUDE
(FT)
−40
NO W
TW
HW
−20
0
−20
−16
−12
−8
−4
0
ALTITUDE RATE (FT/SEC)

FIGURE 3



GRAPHIC CONTROLS CORPORATION BUFFALO NEW YORK
32K AFT CG
(PULL-UP NOT SHAPED BY LAG)
KPIT = 22.09 (CONSTANT)
hERROR
±250′
FIGURE 4
ΔAZ
±1.25g
q
±10°/S
γ
±50°
MACH
.53
.43
.45
.35
±1
5 SEC

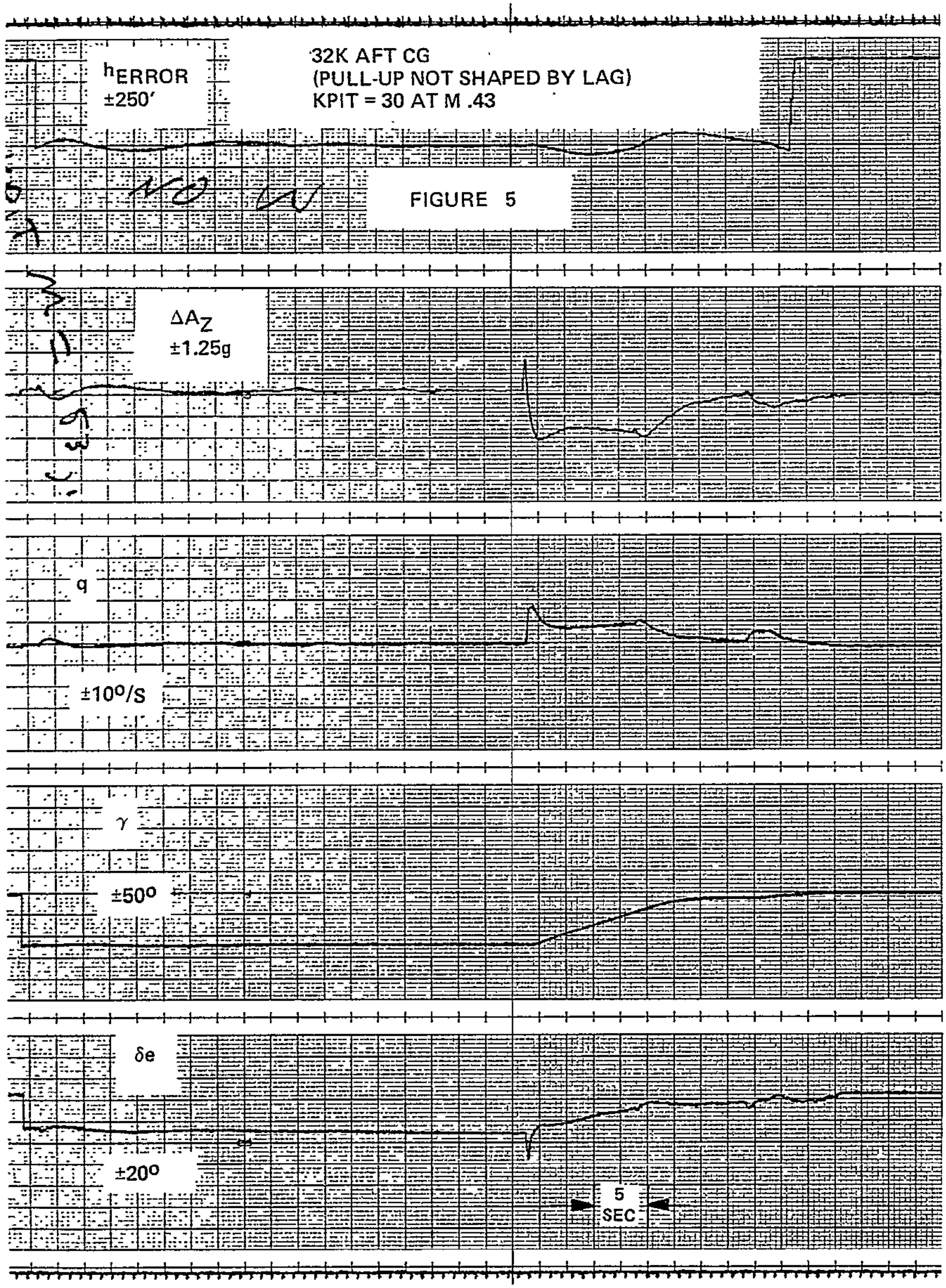
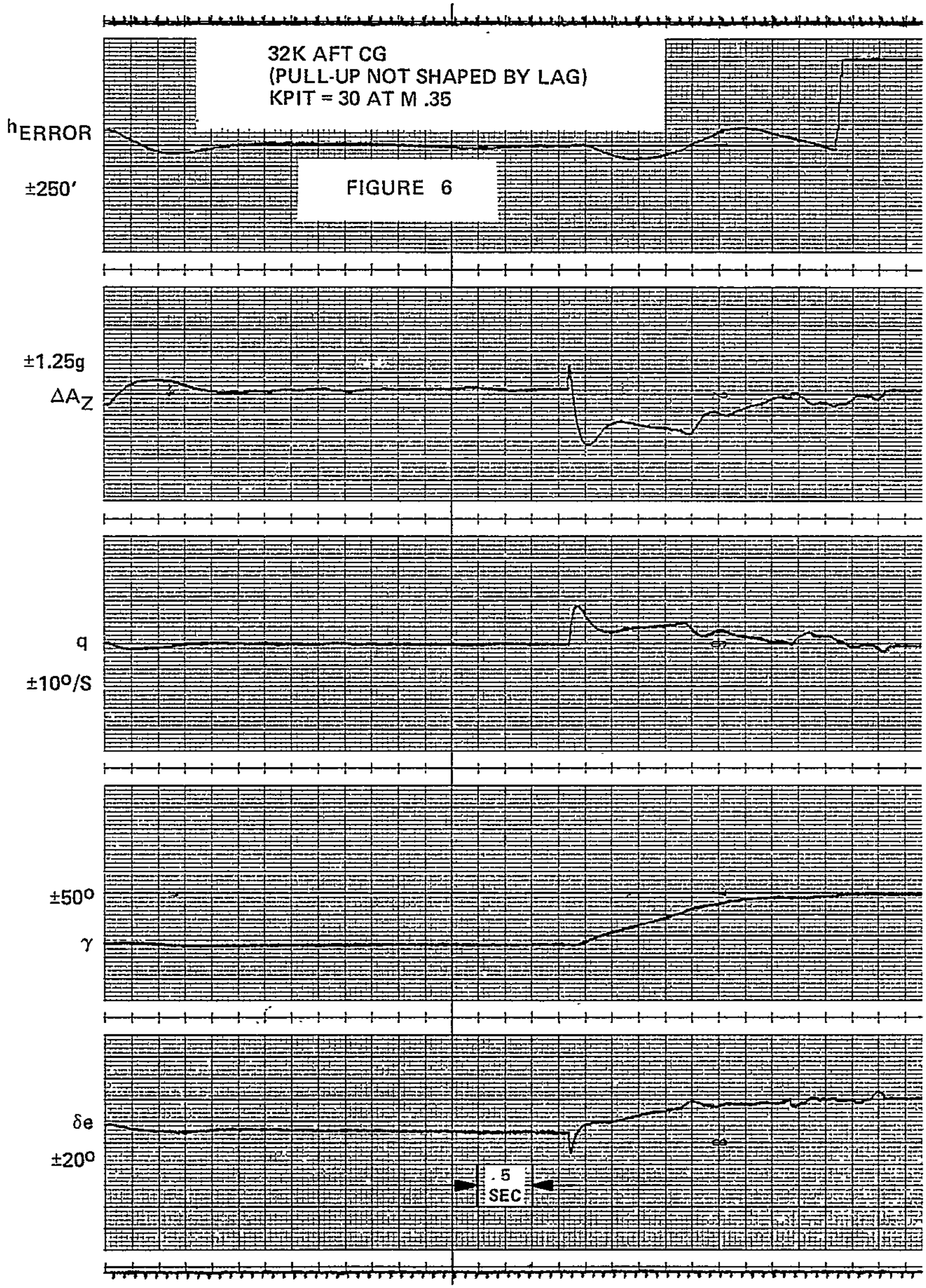

32K AFT CG
(PULL-UP NOT SHAPED BY LAG)
KPIT = 30 AT M .43
hERROR
±250′
ΔAZ
±1.25g
q
±10°/S
γ
±50°
δe
±20°
5
SEC

FIGURE 5



32K AFT CG
(PULL-UP NOT SHAPED BY LAG)
KPIT = 30 AT M .35
FIGURE 6
hERROR
±250′
±1.25g
ΔAZ
q
±10°/S
±50°
γ
δe
±20°
.5 SEC

SECTION 3

DELETION OF AIR DATA

DATE 18 Nov 75

NASA AUTOLAND SUPPORT

(PRELIMINARY INFORMATION)

TASK TITLE: Evaluate Autoland Performance with Air Data Quantities Approximated with INS Quantities.

TASK DESCRIPTION: The enclosure is an initial description of the problem to scope Task No. 3. Only the AUTOLAND automatic mode has been considered. No consideration was given to the manual modes nor to AUTOTAEM.

DISTRIBUTION:

D. Dyer (NASA/JSC)

SPERRY AUTOLAND MEMO NO. 3

DATE: 23 January 1976

COPY: ____________

NASA SHUTTLE

AUTOLAND SUPPORT PROGRAM

TASK NO: 3

TASK NAME: Evaluation of Autoland Performance with Air Data Quantities Approximated with INS Quantities

TASK DESCRIPTION: The enclosure is a detailed report of Task No. 3. Only the Autoland automatic mode was evaluated. No consideration was given to manual modes or to AUTOTAEM.

DISTRIBUTION:

Copy 1. D. Dyer (NASA/JSC)
2. E. Capen (R.I.)
3. G. Carden (R.I.)

I. CONCLUSIONS

An investigation was performed on the Sperry Validation Facility to evaluate Autoland performance for a system employing inertially derived Air Data parameters in place of the nominal Air Data quantities measured by an on-board functioning Air Data system. The baseline Autoland system used for this study constituted the June 1975 140C orbiter with guidance and control defined by the latest FSSRs and pending CRs The primary results and conclusions of this study are as follows:-

(1) Autoland touchdown performance is satisfactory in all respects using inertially derived Air Data quantities. (See, in particular, Tables 2 and 3 and Figures 41 and 42.)

(2) The use of an inertial equivalent speed control law results in maximum $\bar{q}$ values of 360 lb/ft^2 for the 32K payload and approximately 380 lb/ft^2 for a 65K payload. These maximum values occur for the 3σ headwind condition. The maximum $\bar{q}$ for a ØK payload is 300 lb/ft^2.

(3) The use of inertial speed control results in the speed brakes moving to their maximum and minimum authority for 3σ tailwinds and headwinds respectively. The only condition for which this does not occur is in the case of the ØK payload maximum tailwind condition. A maximum speedbrake authority of 67 degrees is required for this case.

(4) Compared to the baseline system with Air Data the inertially derived Air Data system has degraded stability margins. For roll and yaw control this degradation does not exceed 2.5 db for the maximum headwind condition. In the pitch axis this value can approach 4.5 db while flying the steep guidance profile. However, the low frequency and high frequency margins for the combined pitch control and guidance systems do not fall below 10 db and 6 db respectively.

(5) The baseline Autoland system has a pitch

....continued

control gain GDQ that is a function of both Mach number and aerodynamic $\bar{q}$. Considering a pitch gain GDQI = $\frac{22.09}{\sqrt{\bar{q}_I}}$ has the beneficial effect of not having to have a table look-up to compute M_I (inertial Mach number). Using this gain, pitch control/guidance gain margins do not fall below the values specified above and are slightly superior at the high frequency crossover point. Autoland touchdown performance is satisfactory for this system. However, the higher gain at the lower $\bar{q}$ regions of flight that is inherent to the Mach programmable GDQ system does tend to minimize $\dot{h}$ dispersions at touchdown. Mean final flare altitudes are approximately 5 feet higher as well.

(6) The Autoland baseline system defined by the latest FSSRs and pending CRs has not been optimized with respect to the pull-up maneuver and capture and tracking of the -3 degree shallow glide slope. Performance optimization in this area is recommended and will result in higher final flare initiate altitudes and improved and more repeatable final flare performance.

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II INTRODUCTION

The primary task of this study was to evaluate Autoland system performance, assuming the non-availability of an Air Data system. The study was restricted to Autoland only and did not concern itself with Auto Taem, Pre-final, Capture or any Manual Modes.

The logical first step in carrying out this study was to obtain a definition of the Autoland baseline system as it is presently defined by the July 1975 FSSRs in addition to approved or pending Change Requests (CRs). The latter include the following CR numbers: 1207, 1208, 1227, 1230 1250, 1251, 1264, FINAL FLARE CR dated 27 Aug. 1975 and pending CR related to KPIT gain scheduling. This task was accomplished at Sperry and the results summarized in NASA Shuttle Autoland Memo #1 dated 12/2/75. Figures 1 through 9 reflect the Guidance and Control for the baseline system. These block diagrams very closely represent the "July 75 FSSR & CR" system definition and are representative of the system simulated on the Sperry Validation Facility. However, some differences between the two systems were present during the study. They are summarized in the following paragraphs:

1. Autoland control signals in all axes are sampled at a fixed 25/sec rate.
2. Guidance commands to pitch and roll control are updated every 160 msec.
3. Autoland CSL limits remain as defined in the above referenced Memo #1.
4. Lags and washouts are implemented using Z-Transforms and not Bilinear Transforms.
5. The N_Z accelerometer higher order filter and the body bending filter were not simulated on the Sperry Validation Facility.
6. The pitch gain was varied as a function of Mach No. between Mach .53 and .43 rather than between Mach .45 and .35.
7. Fixed gain six state filter used for Navigation. The altitude filter includes a blending function between radar altitude and MSBLS derived altitude between 100 and 400 ft. (See Section III Sperry Autoland Design Definition Document,

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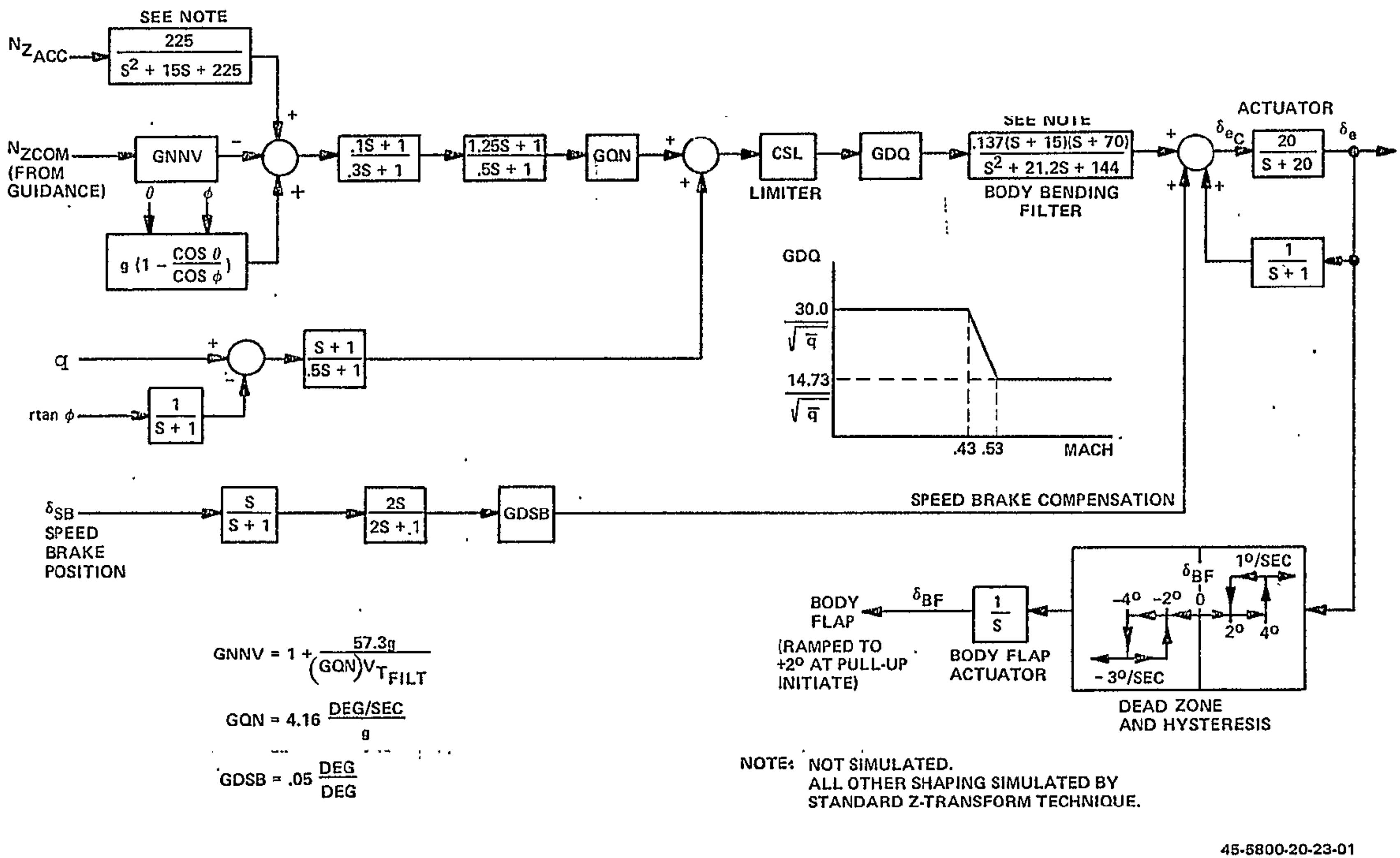
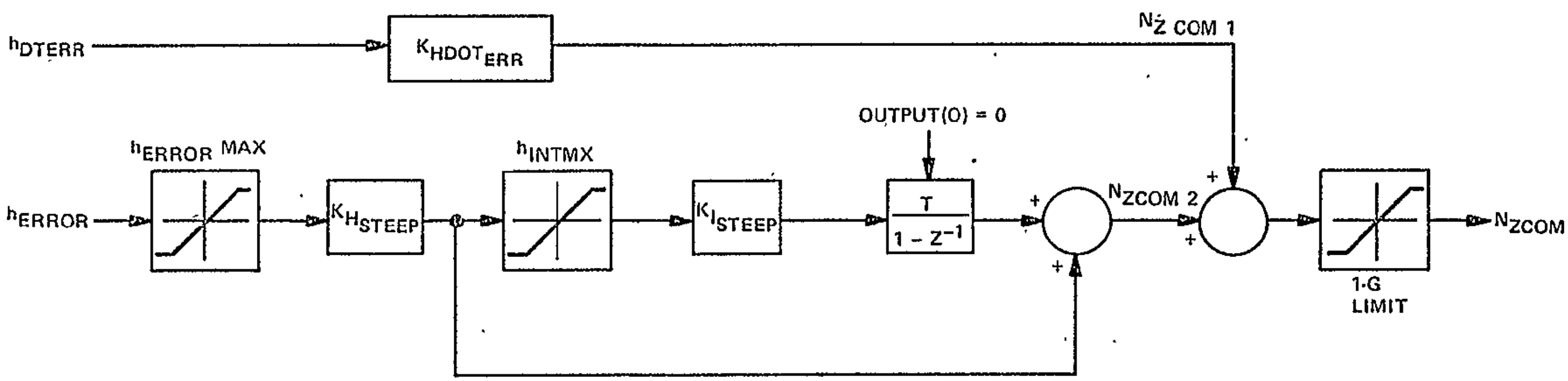


Figure 1
Pitch Control Augmentation Block Diagram



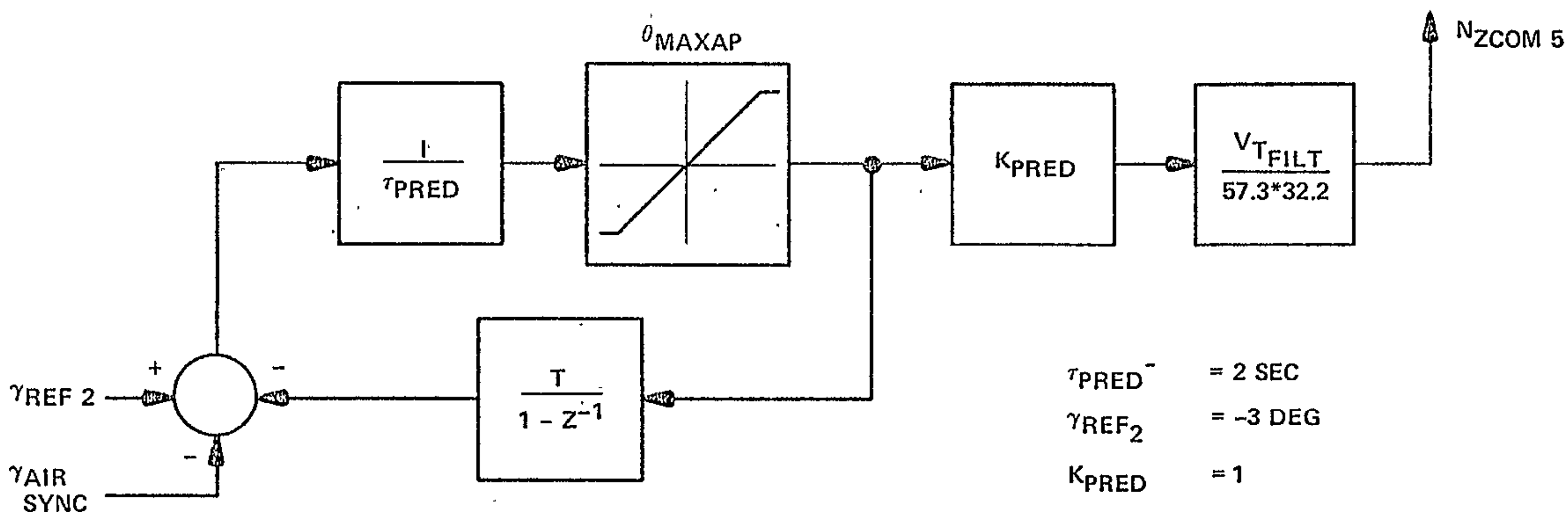
$K_{HDOT_{ERR}} = .0109 \frac{g}{FT/SEC}$, $K_{H_{STEEP}} = .0036 \frac{g}{FT}$ $h_{ERROR\ MAX} = 300$ FT

$K_{I_{STEEP}} = .05$ h_{INTMX} $\doteq 0.18g$

$= 50$ FT

45-5800-20-23-02

Figure 2
Steep Glideslope Block Diagram

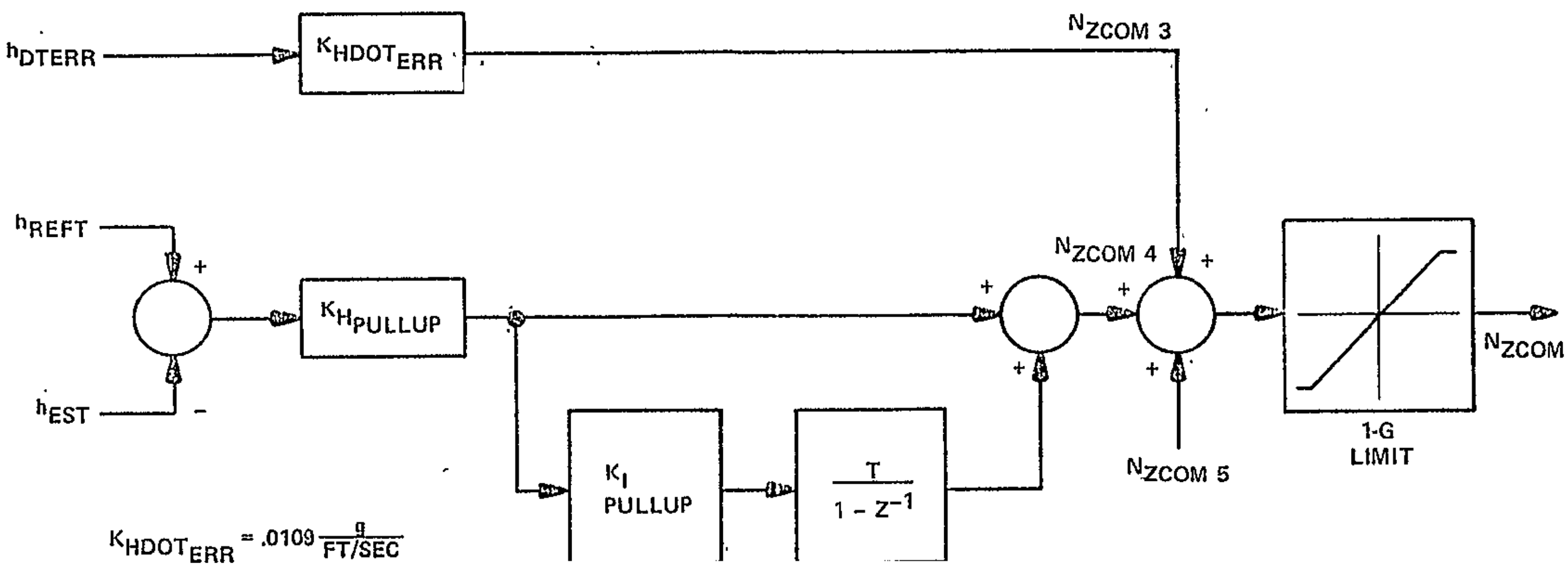


$$\dot{\theta}_{MAX_{AP}} = \frac{N_{Z_{MAX}} \cdot 57.3}{V_{T_{FILT}}}$$

WHERE $N_{Z_{MAX}} = V_I^2 / R$

45-5800-20-23-03

Figure 3
Pullup and Shallow Glideslope
Open Loop Block Diagram



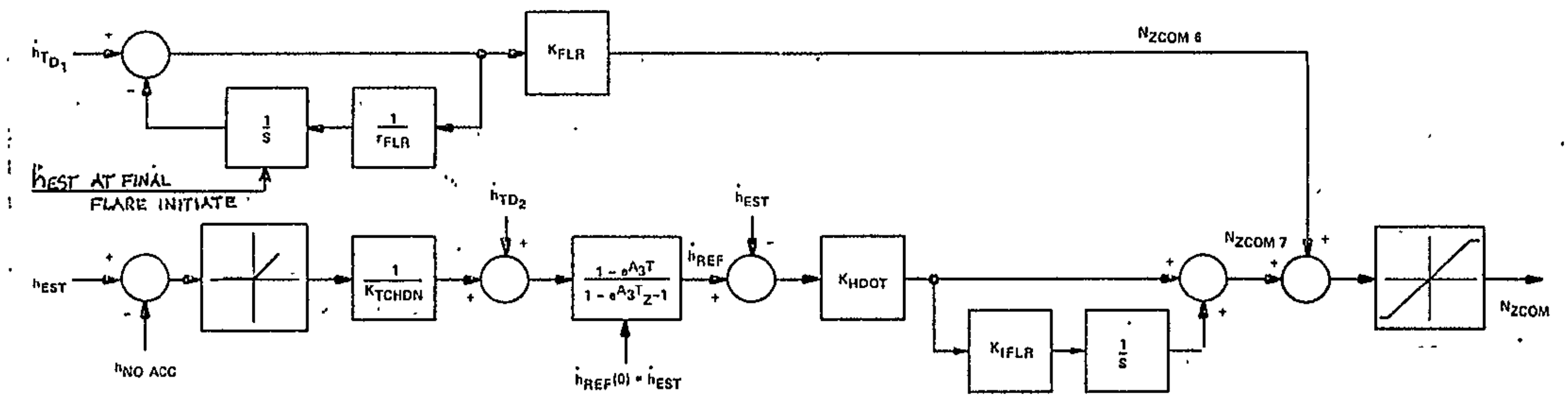
$K_{HDOT_{ERR}} = .0109 \frac{g}{FT/SEC}$

$K_{H_{PULLUP}} = .0046$ g/FT

$K_{I_{PULLUP}} = .05$

45-5800-20-23-04

Figure 4
Pullup and Shallow Glideslope
Closed Loop Block Diagram



$K_{TCHDN} = -4.88 \frac{FT}{FT/SEC}$ $K_{FLR} = 0.004 \frac{g}{FT/SEC}$ $\dot{h}_{TD_1} = +4$ FT/SEC

$K_{HDOT} = 0.02 \frac{g}{FT/SEC}$ $\tau_{FLR} = 5$ SEC $\dot{h}_{TD_2} = -4$ FT/SEC

$K_{IFLR} = 0.1$ $h_{NOACC} = 5$ FT $A_3 = 10$ RAD/SEC

45-5800-20-23-05

Figure 5
Final Flare Block Diagram

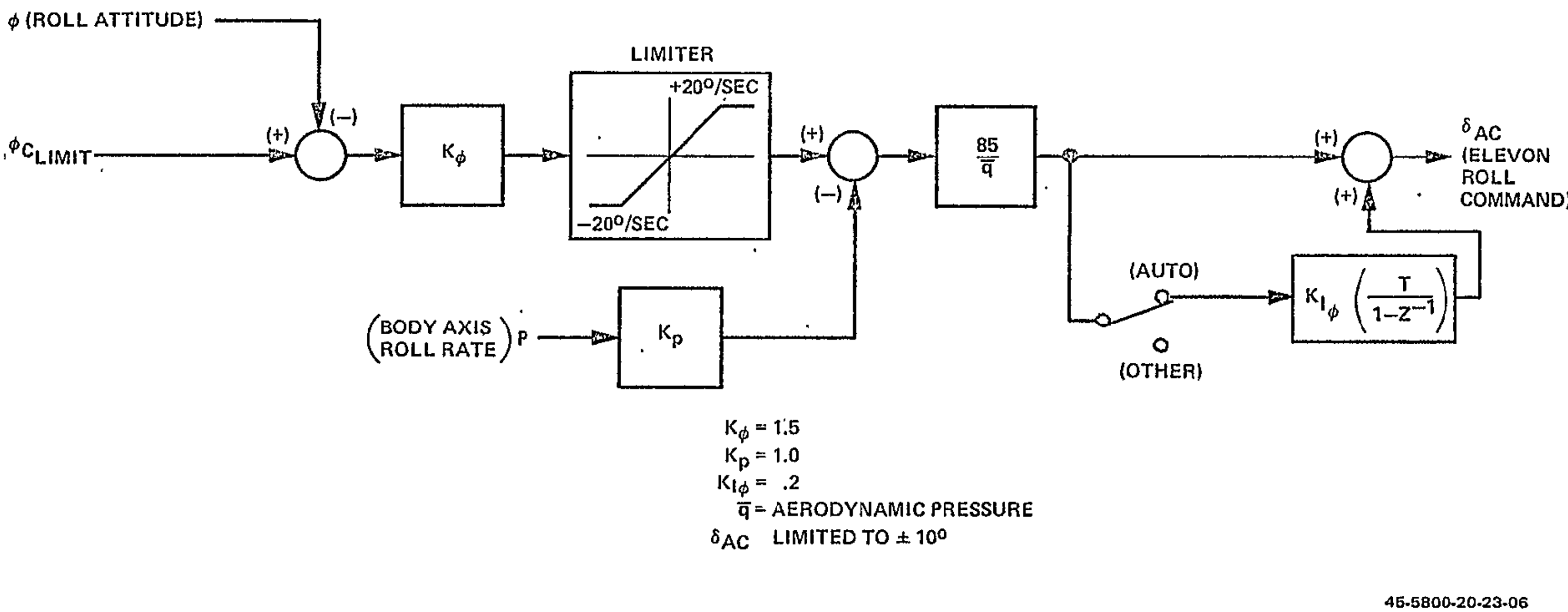
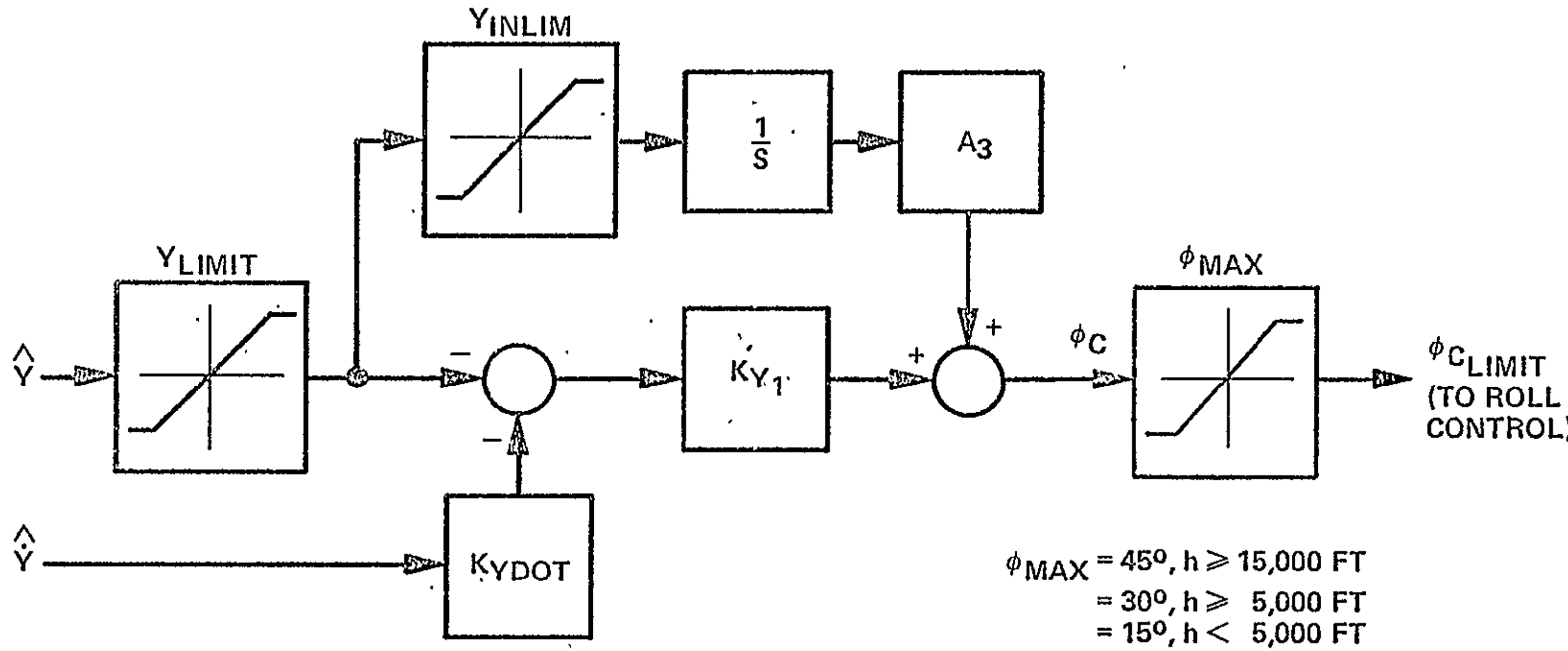


Figure 6
Roll Control Augmentation Block Diagram



$A_3 = -.01$
$Y_{LIMIT} = 500'$
$Y_{INLIM} = 50'$
$K_{Y1} = 0.1$
$K_{YDOT} = 10.0$

45-5800-20-23-07

Figure 7
Lateral Guidance Block Diagram

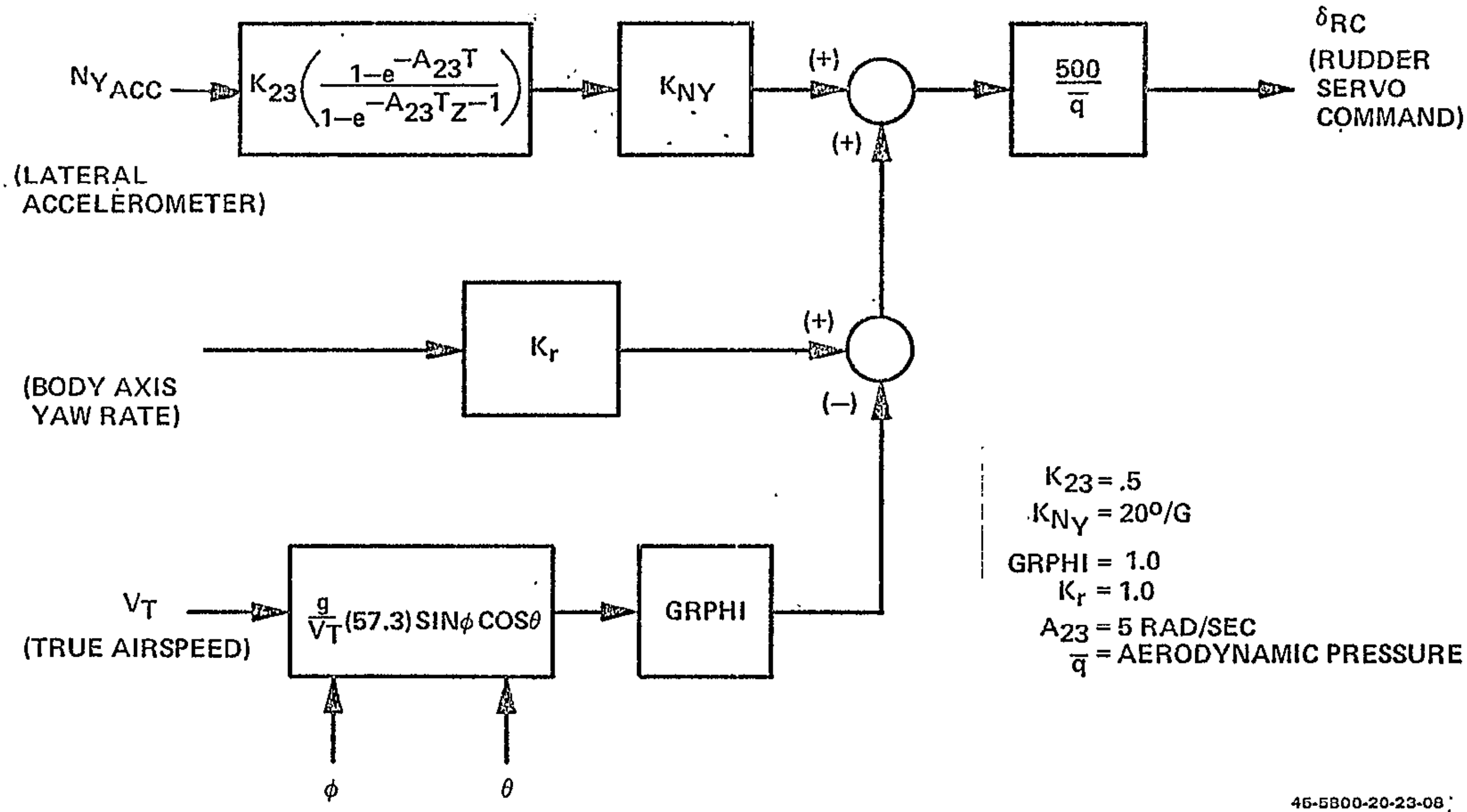


Figure 8
Yaw Control Augmentation Block Diagram

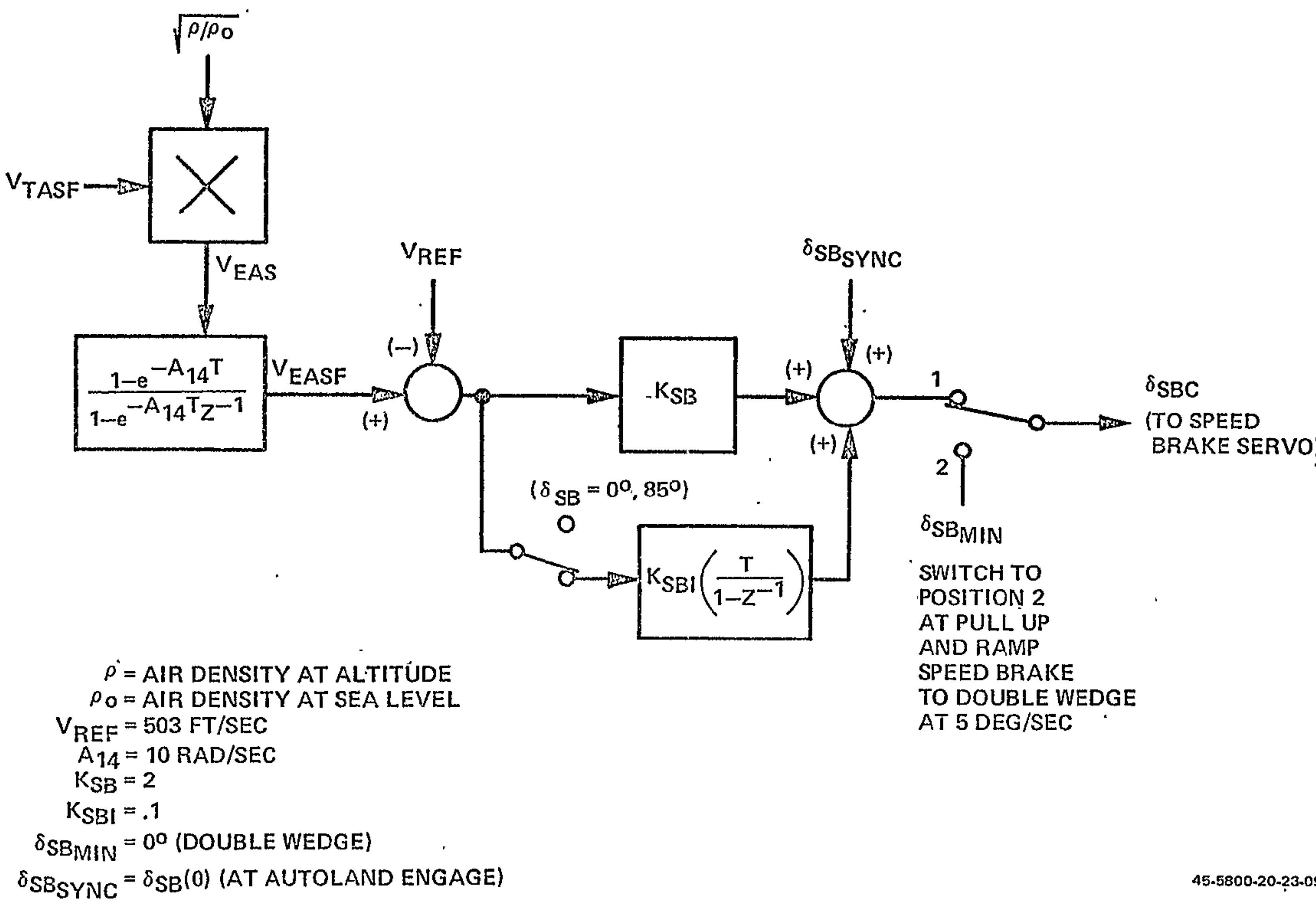


Figure 9
Speed Control System Block Diagram

PDRL #UA01 Sept. 1975, for more detailed information concerning the simulation of the Autoland Navigation System.)

With the exception of item 6 above, none of the other items have any significant impact on the performance of the baseline Autoland system used in this study. Pitch gain programming was retained between the Mach numbers indicated in item 6 in order to minimize dynamic programming changes that would occur during pull-up and shallow glide slope capture. It has been demonstrated that the elevon transients thereby induced have an adverse effect on guidance tracking during this crucial vehicle maneuver and should be eliminated if at all possible. Therefore, Sperry's recommendation is that the pending CR involving the change in pitch gain programming with Mach number be reviewed with all ramifications considered, including gain and phase margin effects, before this modification is officially incorporated into the Autoland system.

Having defined the baseline Autoland $\dot{N}_Z$ system, the system modifications that result from assuming the non-availability of Air Data information were then determined. A summary of these changes is presented in Table 1. This table defines all Air Data dependent parameters used during Autoland as well as the resulting equivalent parameters that would be used using inertially derived Air Data quantities. In the case of the latter, it should be noted that table look-up information using h_{EST} as the independent variable would be used to obtain density altitude and speed of sound information. The assumption of a Standard Atmosphere for the rather reasonable temperature extremes anticipated for Orbiter landings will have no significant impact on system performance during non-standary day conditions. This will permit fixed table look-up data to be used.

The Sperry Autoland Validation Facility was used as the primary tool to evaluate both the individual and relative performance capabilities of the two basic systems defined by Figures 1 through 9 and Table 1. Some of this information such as aerodynamic pressure and Mach number was then used off-line to obtain an indication of the relative impact on system gain margin of the inertially derived Air Data system.

....continued

PARAMETER	BASELINE SYSTEM WITH AIR DATA	BASELINE SYSTEM WITH DERIVED AIR DATA	REFERENCE FIGURE	REMARKS
GNNV	$1 + \frac{57.3\ g}{GQN * V_{T_{FILT}}}$	$1 + \frac{57.3\ g}{GQN * V_I}$	1	$V_I = \sqrt{\dot{h}^2_{EST} + \dot{X}^2_{EST} + \dot{Y}^2_{EST}}$
GDQ	$M = \frac{V_{T_{FILT}}}{\text{SPEED OF SOUND}}$	$M_I = \frac{V_I}{\text{SPEED OF SOUND}}$	1	Speed of sound value obtained from table look-up assuming Standard Day conditions. The independent variable, h_{EST}, is obtained via MSBLS data.
γ_{SYNC}	$\gamma_{AIR\ SYNC} = \gamma_{AIR}$ VALUE DURING LAST PASS THROUGH STEEP	$\gamma_{I_{SYNC}} = \gamma_I$ VALUE DURING LAST PASS THROUGH STEEP	3	$\gamma_{AIR} = \sin^{-1}\left(\frac{\dot{h}_{EST}}{V_{T_{FILT}}}\right)$; $\gamma_I = \sin^{-1}\left(\frac{\dot{h}_{EST}}{V_I}\right)$
$\dot{\theta}_{MAX\ AP}$	$\frac{N_{Z_{MAX}} * 57.3}{V_{T_{FILT}}}$	$\frac{N_{Z_{MAX}} * 57.3}{V_I}$	3	
GAIN	$\frac{V_{T_{FILT}}}{(57.3)(32.2)}$	$\frac{V_I}{(57.3)(32.2)}$	3	
AERODYNAMIC PRESSURE	$\bar{q} = \frac{1}{2}\rho V_T^2$	$\bar{q}_I = \frac{1}{2}\rho_T V_I^2$	1, 6, 8	Air density, ρ_T, obtained from table look-up assuming Standard Day conditions. The independent variable, h_{EST}, is obtained via MSBLS data.
SPEED CONTROL LAW	$\sqrt{\frac{\rho}{\rho_o}}\ V_{TASF} = 503$ FT/SEC	$\sqrt{\frac{\rho_T}{\rho_o}}\ V_I = 503$ FT/SEC	9	
COMMANDED YAW RATE	$r_c = \frac{g}{V_{T_{FILT}}}(57.3)\sin\theta\cos\phi$	$r_{c_I} = \frac{g}{V_I}(57.3)\sin\theta\cos\phi$	8	
GEAR DEPLOYMENT LAW, h_{GEAR}	$h_{GEAR} = h_{NOM} + \text{LIMIT}\left[K_{GRADG}(V_{EASF} - V_{REF}) - K_{GRADG}(V_{T_{FILT}}\cos\gamma_{AIR} - V_G)\right]_{-700}^{+900}$ (FT)	$h_{GEAR} = h_{NOM} + \text{LIMIT}\left[K_{GRADG}(V_{EISF} - V_{REF})\right]_{-700}^{+900}$ (FT)	9	$h_{NOM} = 1000$ FT $K_{GRADG} = 40\ \frac{FT}{FT/SEC}$ $V_G = \sqrt{\dot{X}^2_{EST} + \dot{Y}^2_{EST}}$

SUMMARY OF BASIC AUTOLAND SYSTEM PARAMETER CHANGES RESULTING FROM THE USE OF INERTIALLY DERIVED "AIR DATA" PARAMETERS

TABLE I

Orbiter 140C, June 1975, constituted the basic vehicle model used for the study. Performance evaluation included the following considerations:-

(1) The vehicle was initially trimmed at approximately 12,000 feet, Mach .566, γ_I = -24 deg. and a speed brake setting of 53 deg. These values were kept constant for all weight, CG and wind conditions and account for the initial transients at the start of each run for different payloads and/or wind conditions. (See strip chart traces.)

(2) Four vehicle weight/CG combinations were investigated including

ØK; .675CG
32K; .675CG
32K; .65CG
65K; .65CG

(3) Orbiter trajectory tracking and touchdown performance were determined for no winds and 3σ headwind and tailwind conditions with and without 3σ turbulence.

(4) Monte Carlo runs were obtained for one of the more critical weight/CG combinations.

(5) Vehicle touchdown performance was evaluated for a pitch axis gain, GDQ = $\frac{22.09}{\sqrt{\bar{q}_I}}$. The motivation here was to ascertain if this provided any stability margin benefits for the inertially derived Air Data system while still maintaining satisfactory touchdown performance. This would also eliminate the need for a speed of sound table look-up that would be required if GDQ for the inertially derived Air Data system is to be programmed as a function of Mach number (M_I).

(6) In the process of flying the guidance trajectory to touchdown, basic Autoland event occurrences included:-

....continued

(a) the retraction of speed brakes to Ø deg., i.e. the double wedge position.
(b) retrimming the body flap from +22.5 deg. to +2 deg.

Both of these events are initiated at the pull-up altitude of approximately 1910 ft. Retraction rates for each system are -5 deg/sec. and -3 deg/sec. respectively.

(7) Also basic to the Autoland system was the fact that final flare initiate altitudes were constrained to occur between minimum and maximum landing gear reference altitudes of 60 ft and 200 ft respectively.

.....continued

III DISCUSSION OF RESULTS

1.0 BASELINE SYSTEM PERFORMANCE - NOMINAL AIR DATA

Autoland performance for the Baseline System with nominal Air Data is illustrated and summarized by the following:-

(a) Strip chart traces - Figures 10,11,12,17,18,19,24, 25,26,31,32,33.
(b) h vs h final flare phase plane plots - Figures 15,22, 29,36.
(c) Monte Carlo touchdown data - Figures 41,42.
(d) Autoland Trajectory and Touchdown Performance - Table 2.

An evaluation of this data indicates that system performance is generally satisfactory in all respects. Touchdown parameters fall well within the desirable limits. However, one primary area where further guidance improvement might be desirable is associated with capture of the -3 deg. shallow glide slope. This is illustrated by both the strip chart traces and the phase plane plots Overshoot of the shallow glide slope by approximately 0.8 deg. to -2.2 deg. occurs with the orbiter falling 30 to 40 feet below the reference altitude. Before recovery to the -3 deg. reference path can be effected, final flare occurs at a mean CG altitude of approximately 80-85 feet. No attempt was made to optimize this particular aspect of the Autoland guidance during the study. It was recognized that this constituted a slight deficiency in system trajectory guidance, but would not detract from the primary objective of the study which was to make a relative comparison of system performance with actual Air Data and using inertially derived Air Data quantities. It should also be recognized that this was the first time that this exact Autoland configuration was being evaluated on the Sperry Validation Facility. Hence, the impact of such aspects as ramping the body flap to +2 deg. during pull-up and removing the lag on K_{PRED} were expected to necessitate at least some re-optimization of the pull-up guidance. Direct compensation for the moment producing effects of speed brake and body flap ramping might also have to be considered to improve shallow glide slope capture and tracking.

....continued

Besides the performance data for the four weight/CG combinations summarized in Table 2, Figures 41 and 42 illustrate the results of 400 runs for the 32K payload aftCG condition with random winds and turbulence. This condition was deemed to be the most adverse as suggested by observations made while considering worst case 3σ winds and turbulence conditions. In Figures 41 and 42, the triangles represent maximum and minimum values. The ends of the horizontal bar represent plus and minus standard deviations. The vertical bar centered on the horizontal one represents the mean of each parameter. In the case of the X touchdown data, the shallow glide slope of $-3°$ was assumed to intercept the runway at a point 1500 feet from threshold. As the figures indicate, overall system touchdown performance is satisfactory in all respects.

As suggested previously above, one further test was made with each weight/CG combination. Touchdown performance was observed and recorded for 3σ headwind and tailwind conditions with corresponding 3σ turbulence. Ten to twelve runs were made for each configuration in order to ascertain any potential trends toward radically unacceptable performance. No such trends were observed. The worst touchdown rate encountered for the tailwind condition was -7.4 ft/sec. A maximum attitude of 11.1 deg. for the 65K payload condition was also noted. For the maximum headwind with turbulence condition, the largest touchdown rate encountered for the OK and 32K payloads was 9.1 ft/sec. The largest touchdown attitude was 15.0 deg. The results were somewhat worse for the 65K payload. Besides three touchdown rates of -9.2 ft/sec., a maximum value of -12.9 ft/sec was also measured for one of the twelve runs. The corresponding attitudes for the above were approximately 16.1 deg. and 17.3 deg. respectively. Although it is true the statistical sample was small, these rather high values for $\dot{h}$ and θ for the 65K payload condition tend to reinforce the previous recommendation that further investigation and optimization of the presently defined baseline Autoland system would appear to be in order.

2.0 BASELINE SYSTEM PERFORMANCE - INERTIALLY DERIVED AIR DATA SYSTEM

Autoland performance for the Baseline System using inertially derived Air data quantities is illustrated and summarized by the following:-

....continued

(a) Strip chart traces - Figures 13,14,20,21,27, 28,34,35.
(b) h vs $\dot{h}$ final flare phase plane plots - Figures 16,23,30,37.
(c) Monte Carlo Touchdown data - Figures 41,42.
(d) Autoland Trajectory and Touchdown Performance - Table 3.

The strip charts for the no wind condition are identical to those for the Baseline System with nominal Air Data and correspond to Figures 10,17,24,31. Some pertinent comments regarding this system are as follows:-

(1) Touchdown performance as summarized by Table 3 and the Monte Carlo runs of Figures 41 and 42 is satisfactory in all respects. Performance is generally as good as or superior to the Baseline System with Air Data. In particular, the maximum pitch attitudes at touchdown are reduced.
(2) The above conclusion appears to extend to the extreme 3σ headwind/tailwind with turbulence environment as well. The total of 40-45 runs taken for the four payload/CG combinations for each wind direction resulted in no vertical touchdown rate exceeding -5.0 ft/sec. The maximum pitch attitude of 12.1 degrees was encountered for the 65K payload condition.
(3) The speed control law for this system must of necessity revert to an equivalent inertial airspeed one. As the data of Table 3 indicates, this results in a maximum aerodynamic $\bar{q}$ of 360-380 lb/ft^2 for the 32K, 65K payload 3σ headwind condition.

(4) For a γ_{STEEP} of -24 deg. and the inertial speed control law, the speed brakes assume their extreme position of Ø deg. and 85 deg. respectively for 3σ headwind and tailwind conditions. This occurs for both the 32K and 65K payload conditions. For ØK the mean speed brake position is Ø deg. for a 3σ headwind and 67 deg. for a 3σ tailwind. These values contrast with mean speed brake positions for the Baseline System with Air Data. In the case of the latter, the only condition where the speedbrake is driven to its

....continued

stop is for the 65K 3σ tailwind case.

(5) The inability to monitor for winds in the inertially referenced Air Data system necessitates a change in the landing gear deployment law as indicated in Table I. Gear deployment now becomes a function of speed error only. The results, as indicated in Table 3, is that the gear in most cases is no longer deployed at the extremes of 300 ft and 1900 ft for 3σ headwinds and tailwinds respectively. In fact, for the 65K payload 3σ headwind condition, there tends to be a "reversal" in gear deployment altitude in that the landing gear deploys at 1275 ft (nominal deployment altitude is 1000 ft) rather than in the direction of minimum altitude deployment. The impact of this overall effect for all payloads is not very significant. Maximum touchdown velocities are within desirable limits and well below the specified safe tire speed limit of 210 Kts, even for the 65K payload condition.

(6) The capture and tracking of the -3 deg. shallow glideslope is slightly worse for this system compared to the nominal system with Air Data. The system performance is such that overshoot to -2 deg. occurs during the pull-up and capture maneuver, particularly in the case of the ØK and 32K payloads with aftCG. This results in lower flare initiate altitudes since the orbiter falls 40-45 ft below the reference altitude for certain cases. Touchdown performance is not adversely affected; however, as the results of Table 3 and Figures 41 and 42 clearly indicate. The constraint of a minimum final flare initiate altitude of 60 ft, landing gear reference, comes prominently into play in many of these cases. It was precisely because of this concern that an initial series of Monte Carlo runs was taken for the ØK payload condition. The results, though not included in this report, confirm that for this case, touchdown performance is satisfactory in all respects.

(7) The use of inertially derived Air Data results in the scheduling of the yaw and roll control gains as an inverse function of $\bar{q}_I = \frac{1}{2}\rho V_I^2$.

....continued

Relative to the baseline Autoland system using nominal Air Data parameters, this results in a maximum control system gain increase of 2-2.5 db for 3σ headwinds and a 1-1.5 db gain decrease for 3σ tailwinds. The impact of this effect is not significant with respect to either roll or yaw performance or to their respective gain and phase margins.

(8) The measured value of the pitch control gain, GDQ, for the 3σ headwind condition shows a maximum gain increase of approximately 4.5 db relative to the nominal baseline system using Air Data. This maximum increase in gain occurs during the initial portion of the Autoland guidance steep profile. Its effect translates into an almost equivalent gain margin increase for the low frequency cross-over point for the conditionally stable ØK and 32K payload aftCG orbiter. However, an equivalent adverse effect results with respect to the gain margin for the high frequency cross-over for all orbiter payload conditions. Based upon steep glide slope analysis, this would reduce the gain margin to approximately 7 db. (See handout, Shuttle Automatic Landing Support Program Review Meeting, August 19, 1975. This meeting was held at Rockwell.) For the lower $\bar{q}$ regions between pull-up and touchdown, the corresponding maximum gain increases measured were approximately 1.5 db.

For the 3σ tailwind condition, a maximum reduction in GDQ gain during steep guidance of 1.4 db was meausred for the 32K payload conditions. Application of this data to the gain and phase margin information supplied in the above reference would still result in a low frequency gain margin of approximately 11.7 db. At the lower $\bar{q}$ region between shallow capture and touchdown, this maximum reduction in GDQ reduces to a rather insignificant 0.6 db.

To evaluate the impact of going to a non-programmable GDQ gain with Mach number, a GDQI value of $\frac{22.09}{\sqrt{\bar{q}_I}}$ was selected. This resulted in system gain margins at least

....continued

equivalent to or 1 db superior to those quoted above for steep. For the lower $\bar{q}$ regions, an approximate 2.5 db gain margin reduction occurs at the low frequency crossover for a worst case 3σ tailwind and a conditionally stable system (ØK and 32K AFT CG). In addition, the choice of this gain eliminates the need for a table look-up for computing $M_I = \dfrac{V_I}{\text{speed of sound}}$. The results corresponding to this modification are summarized in Table 4. Typical illustrative runs are presented for the ØK payload condition in Figures 38,39,40. Touchdown performance is satisfactory in all respects. However, the slightly larger altitude errors induced during shallow capture (lower mean final flare altitudes) suggest that the tighter control loop gain, exemplified by $\dfrac{30}{\sqrt{\bar{q}_I}}$ for Mach programmable GDQ, is preferable in this flight region. This conclusion is also supported by the touchdown performance observed for the 3σ headwind and tailwind runs made with maximum turbulence for the GDQI gain of $\dfrac{22.09}{\sqrt{\bar{q}_I}}$. For this limited number of runs higher mean altitude rates at touchdown are observed while the rms $\dot{h}$ dispersions tended to be greater as well.

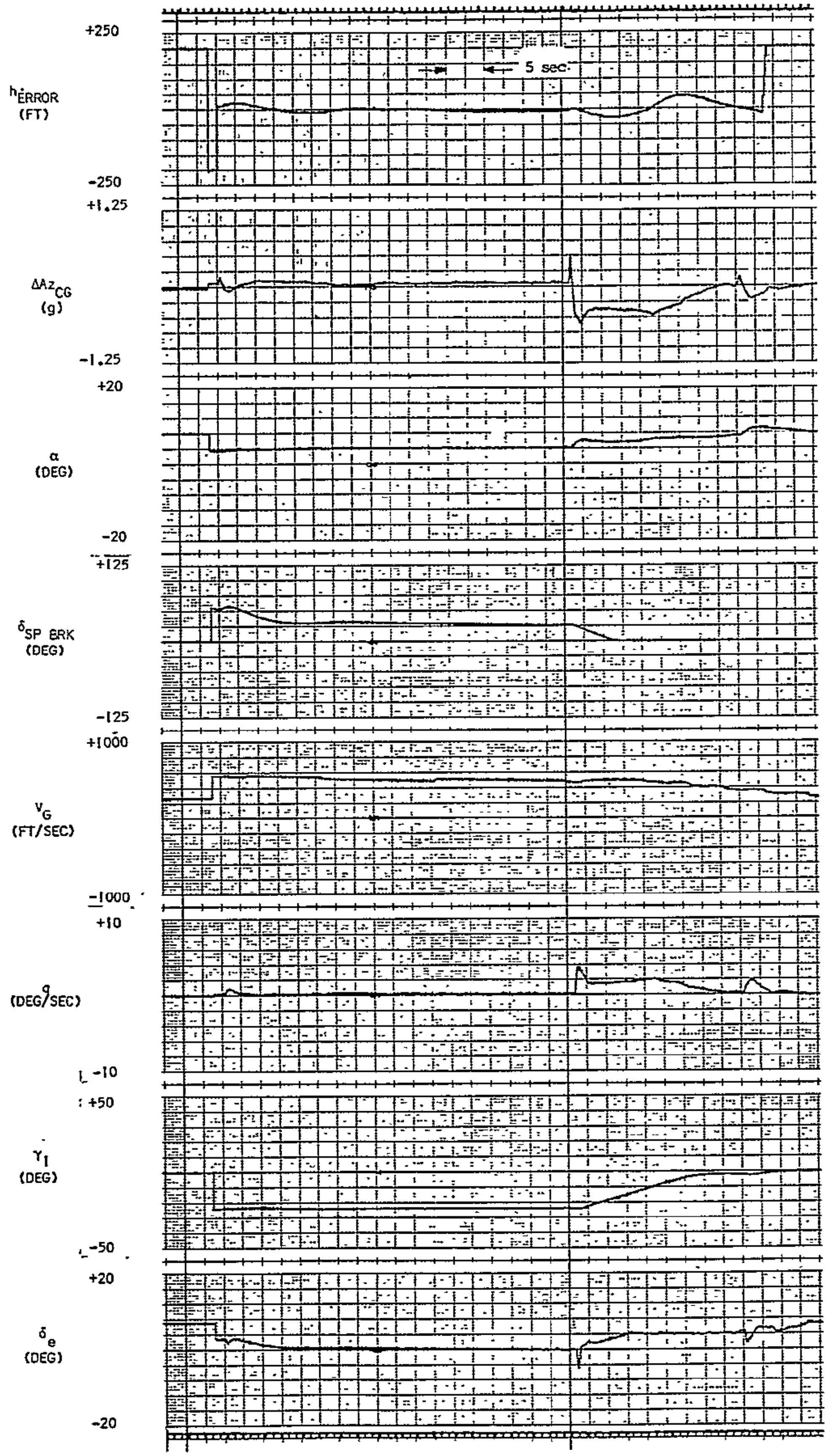


Figure 10

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING NOMINAL AIR DATA PARAMETERS, 140C ORBITER JUNE 1975. ØK PAYLOAD; .675 CG; NO WIND

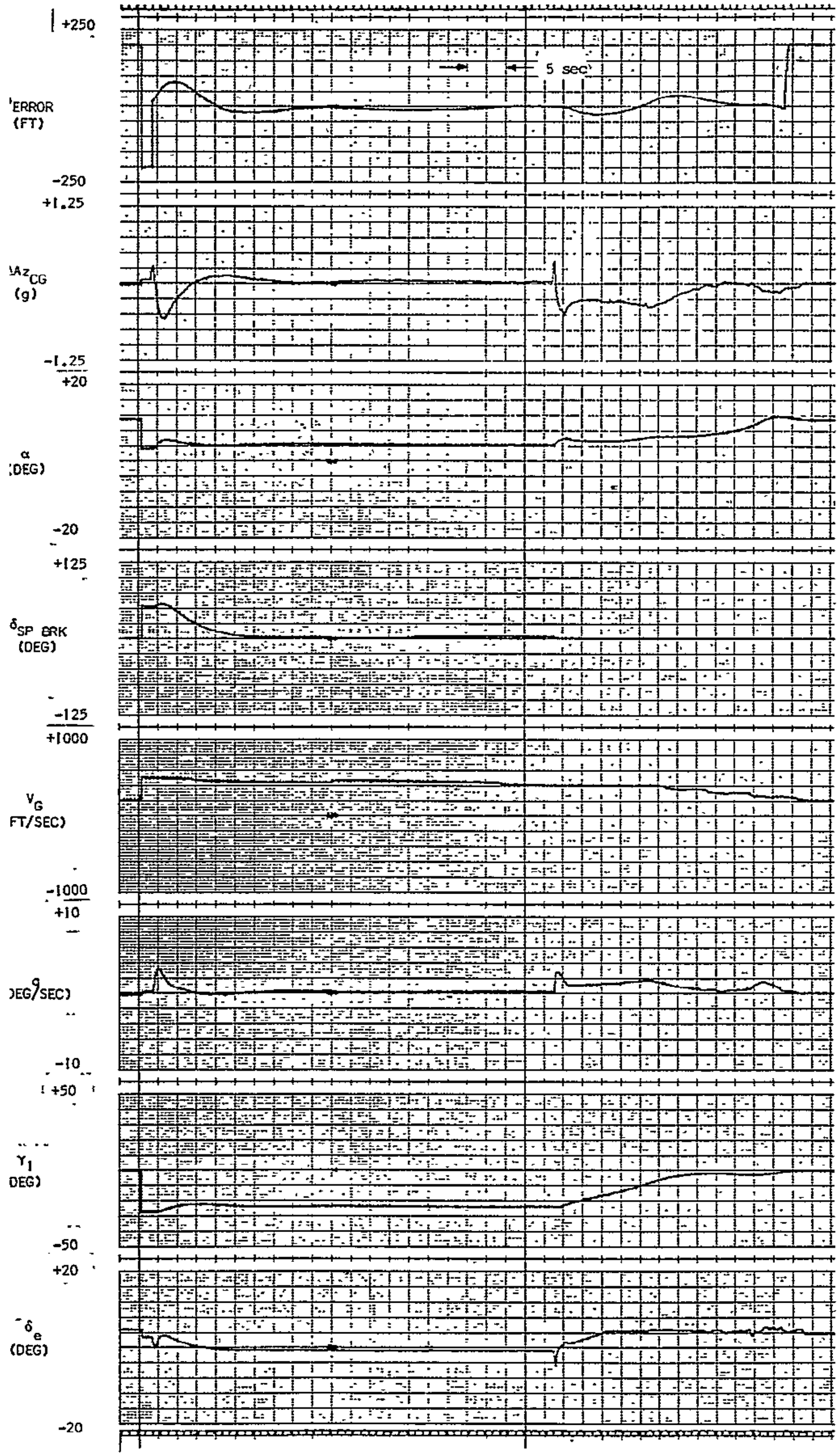


Figure 11

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING
NOMINAL AIR DATA PARAMETERS, 140C ORBITER
JUNE 1975. ØK PAYLOAD; .675 CG; MAX HEADWIND

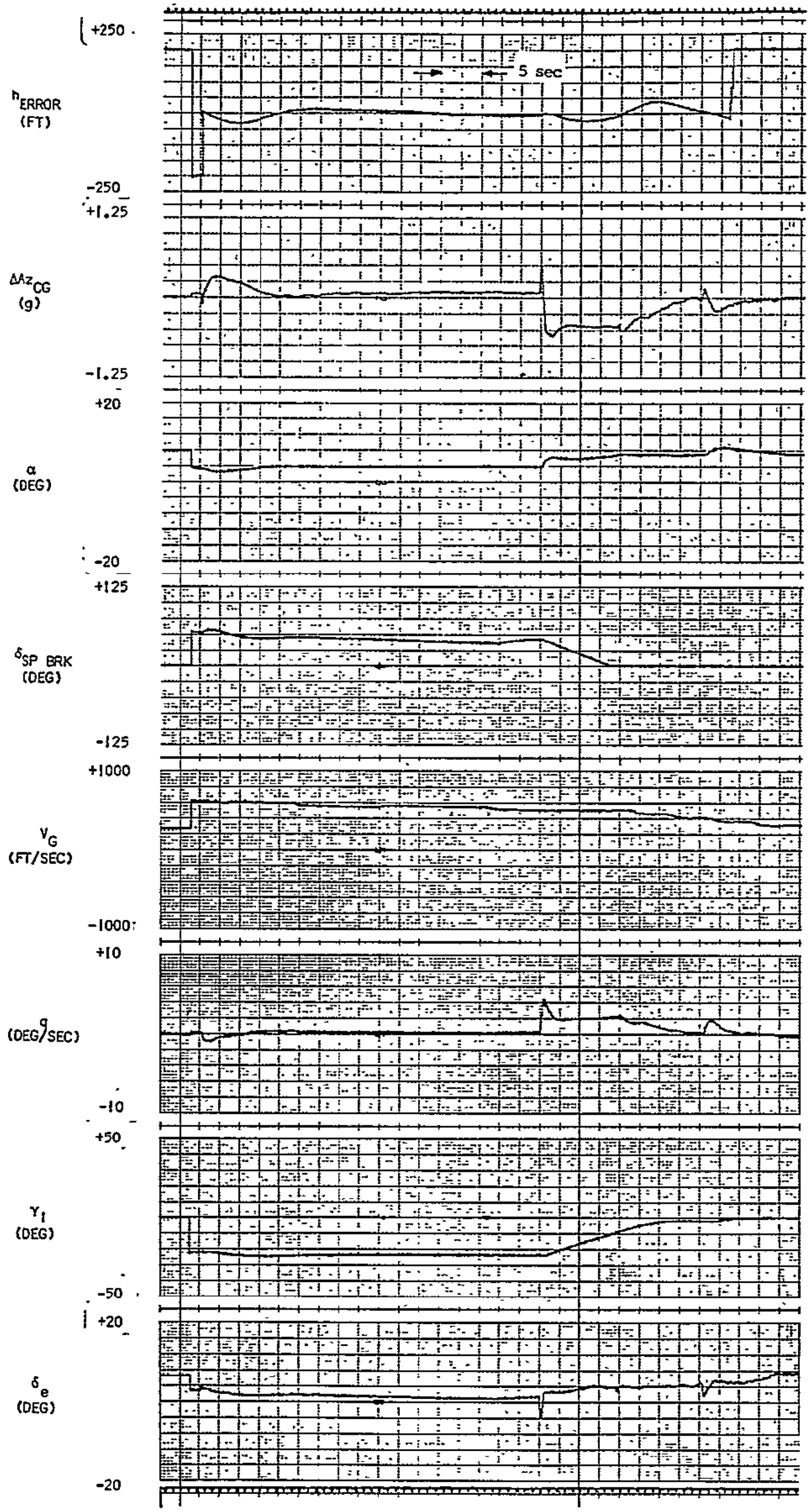


Figure 12

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING NOMINAL AIR DATA PARAMETERS, 140C ORBITER JUNE 1975. ØK PAYLOAD; .675 CG; MAX TAILWIND

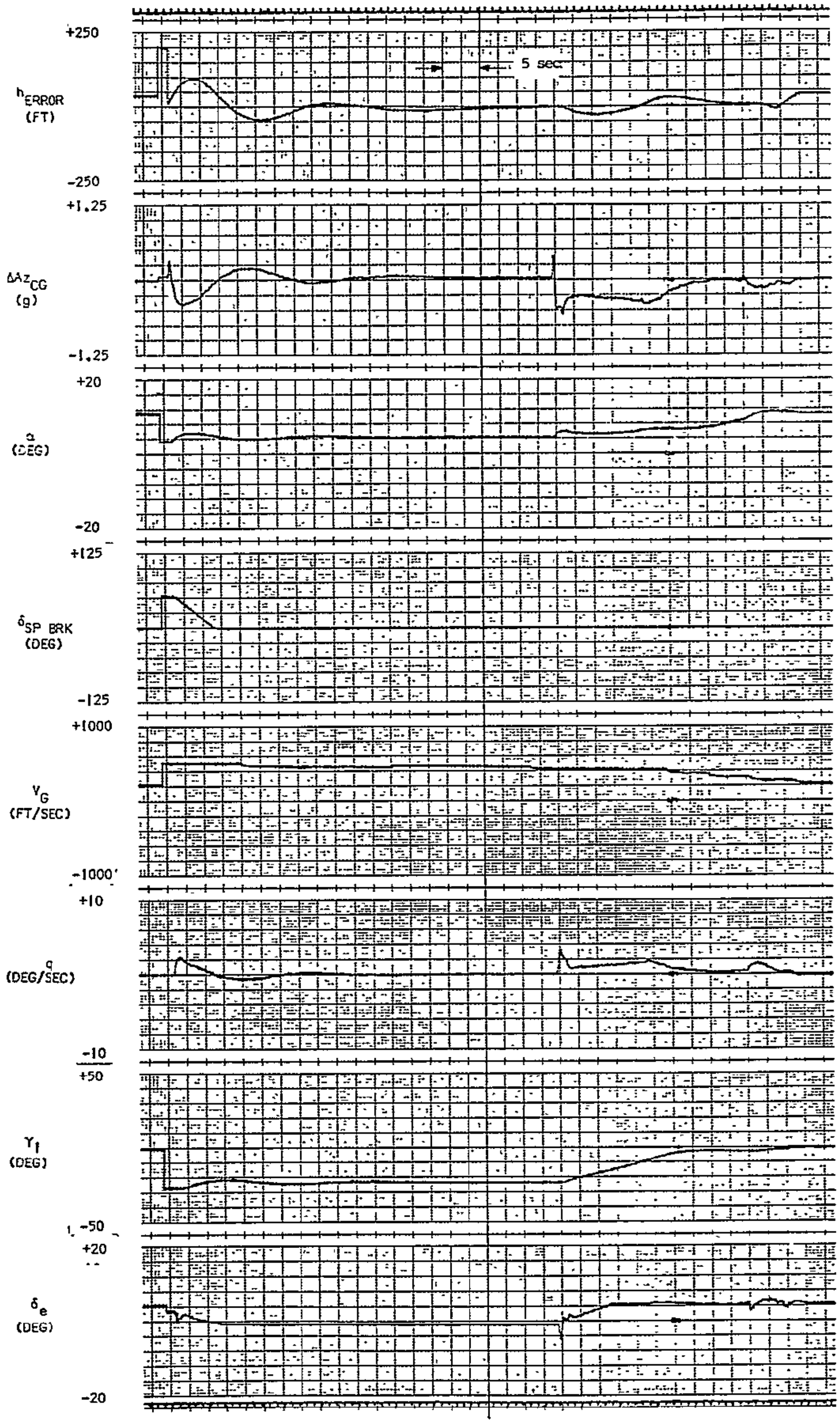


Figure 13

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING
INERTIALLY DERIVED "AIR DATA" PARAMETERS,
140C ORBITER JUNE 1975.
ØK PAYLOAD; .675 CG; MAX HEADWIND

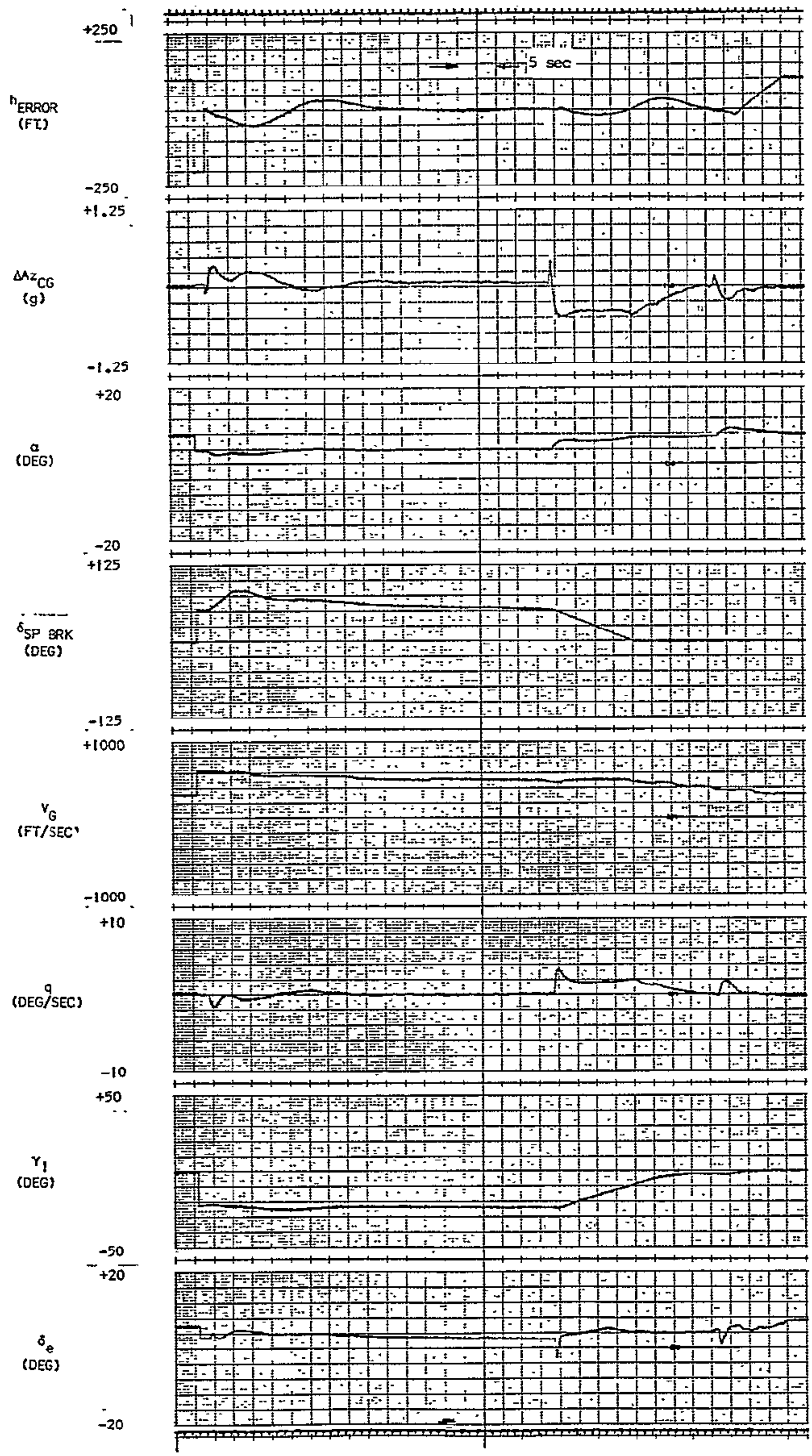


Figure 14

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING INERTIALLY DERIVED "AIR DATA" PARAMETERS, 140C ORBITER JUNE 1975.
ØK PAYLOAD; .675 CG; MAX TAILWIND

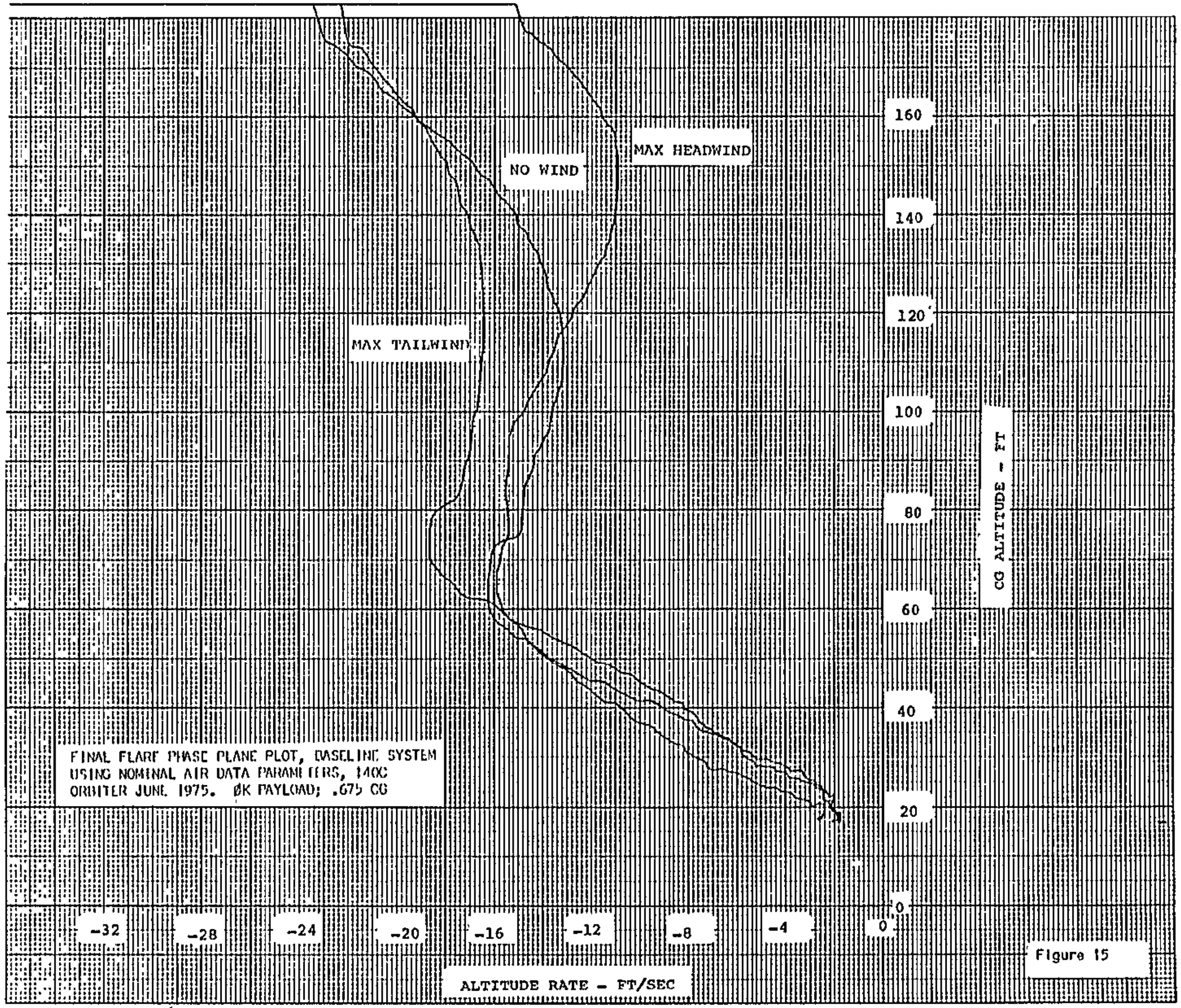
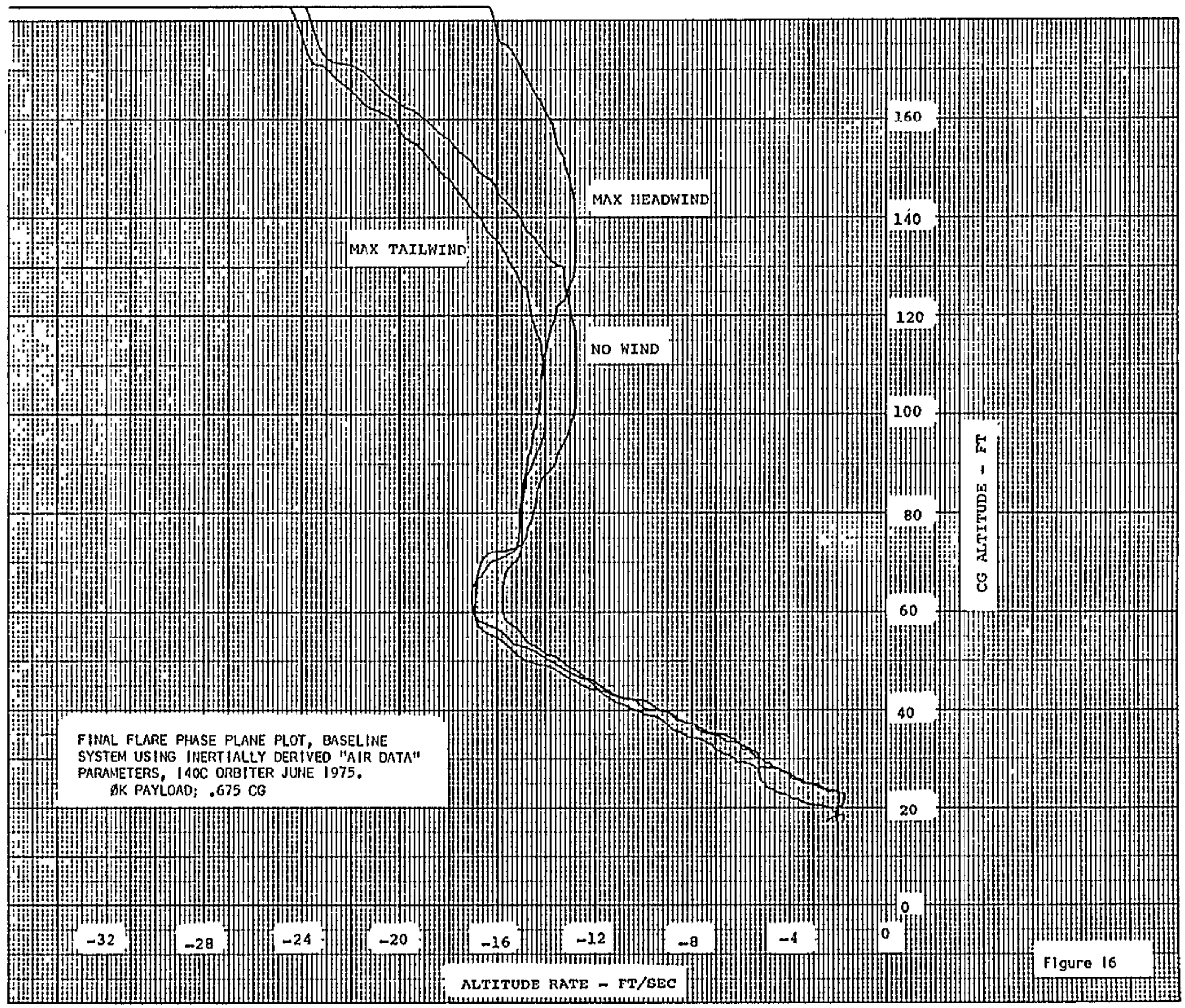

MAX HEADWIND
NO WIND
MAX TAILWIND
160
140
120
100
80
60
40
20
0
CG ALTITUDE - FT
-32
-28
-24
-20
-16
-12
-8
-4
0
ALTITUDE RATE - FT/SEC
FINAL FLARE PHASE PLANE PLOT, BASELINE SYSTEM USING NOMINAL AIR DATA PARAMETERS, 140C ORBITER JUNE 1975. ØK PAYLOAD; .675 CG

Figure 15


MAX HEADWIND
MAX TAILWIND
NO WIND
160
140
120
100
80
60
40
20
0
CG ALTITUDE - FT
FINAL FLARE PHASE PLANE PLOT, BASELINE SYSTEM USING INERTIALLY DERIVED "AIR DATA" PARAMETERS, 140C ORBITER JUNE 1975.
ØK PAYLOAD; .675 CG
-32
-28
-24
-20
-16
-12
-8
-4
0
ALTITUDE RATE - FT/SEC
Figure 16

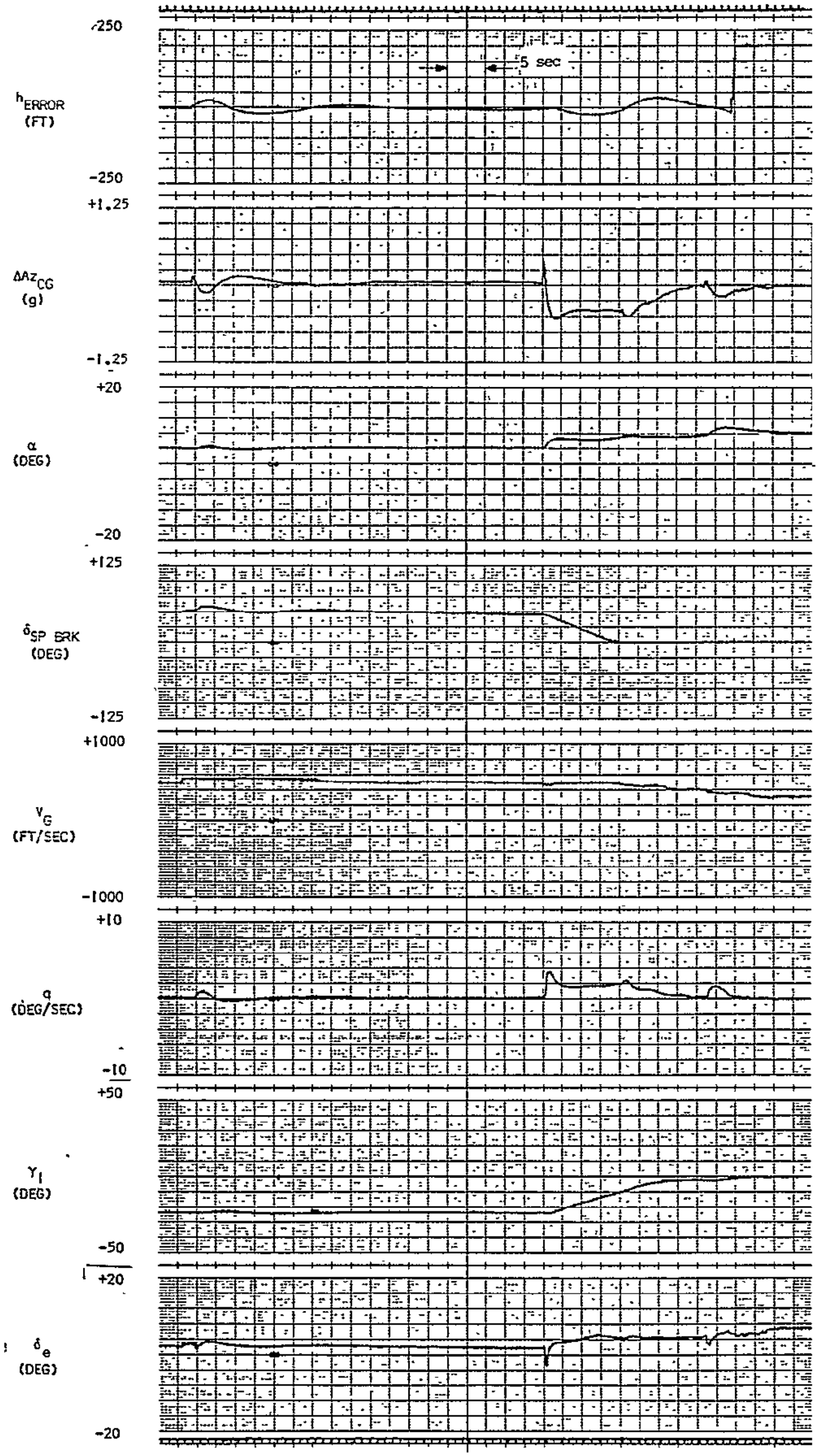


Figure 17

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING NOMINAL AIR DATA PARAMETERS, 140C ORBITER JUNE 1975. 32K PAYLOAD; .675 CG; NO WIND

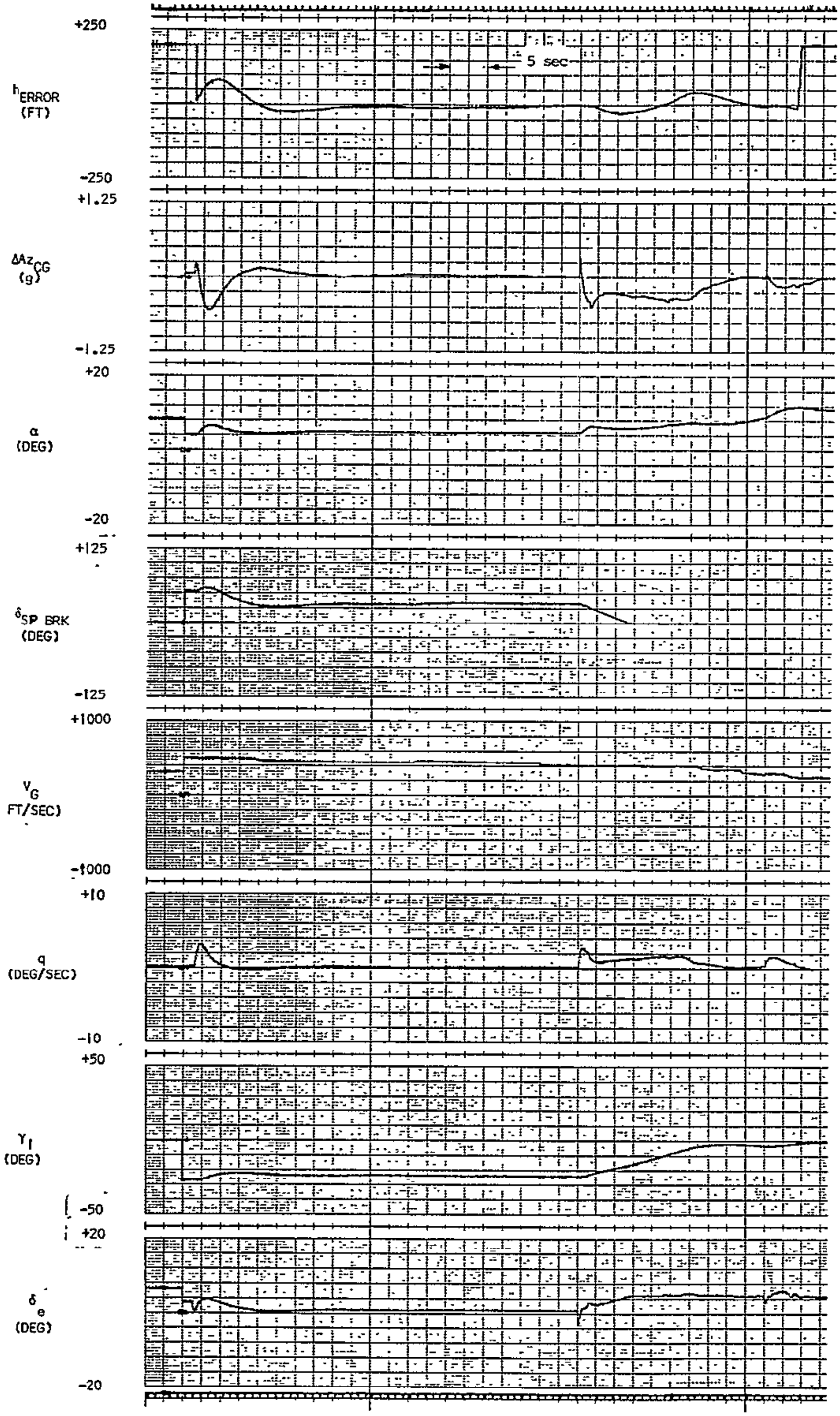


Figure 18

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING
NOMINAL AIR DATA PARAMETERS, 140C ORBITER
JUNE 1975. 32K PAYLOAD; .675CG; MAX HEADWIND

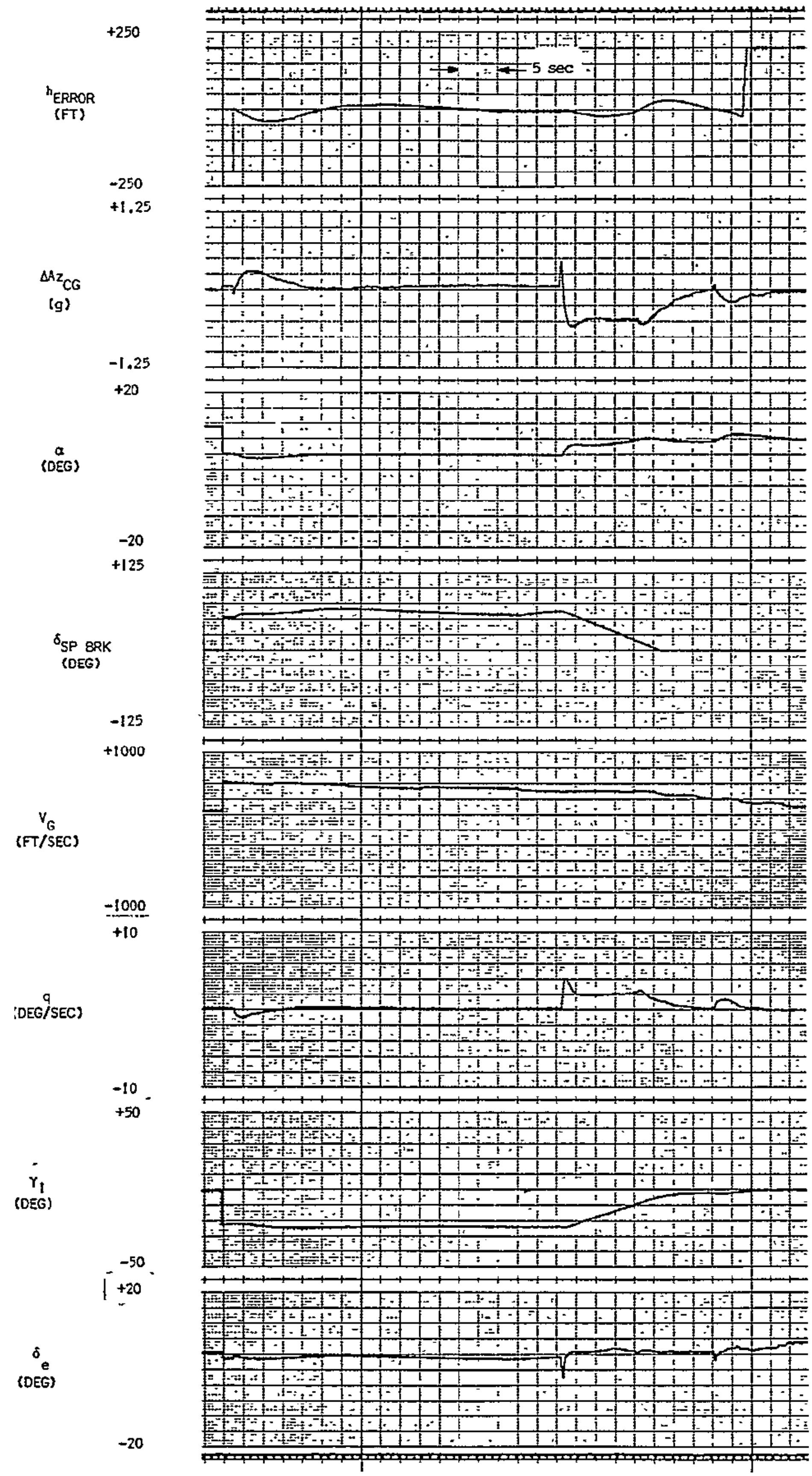


Figure 19

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING
NOMINAL AIR DATA PARAMETERS, 140C ORBITER
JUNE 1975. 32K PAYLOAD; .675 CG; MAX TAILWIND

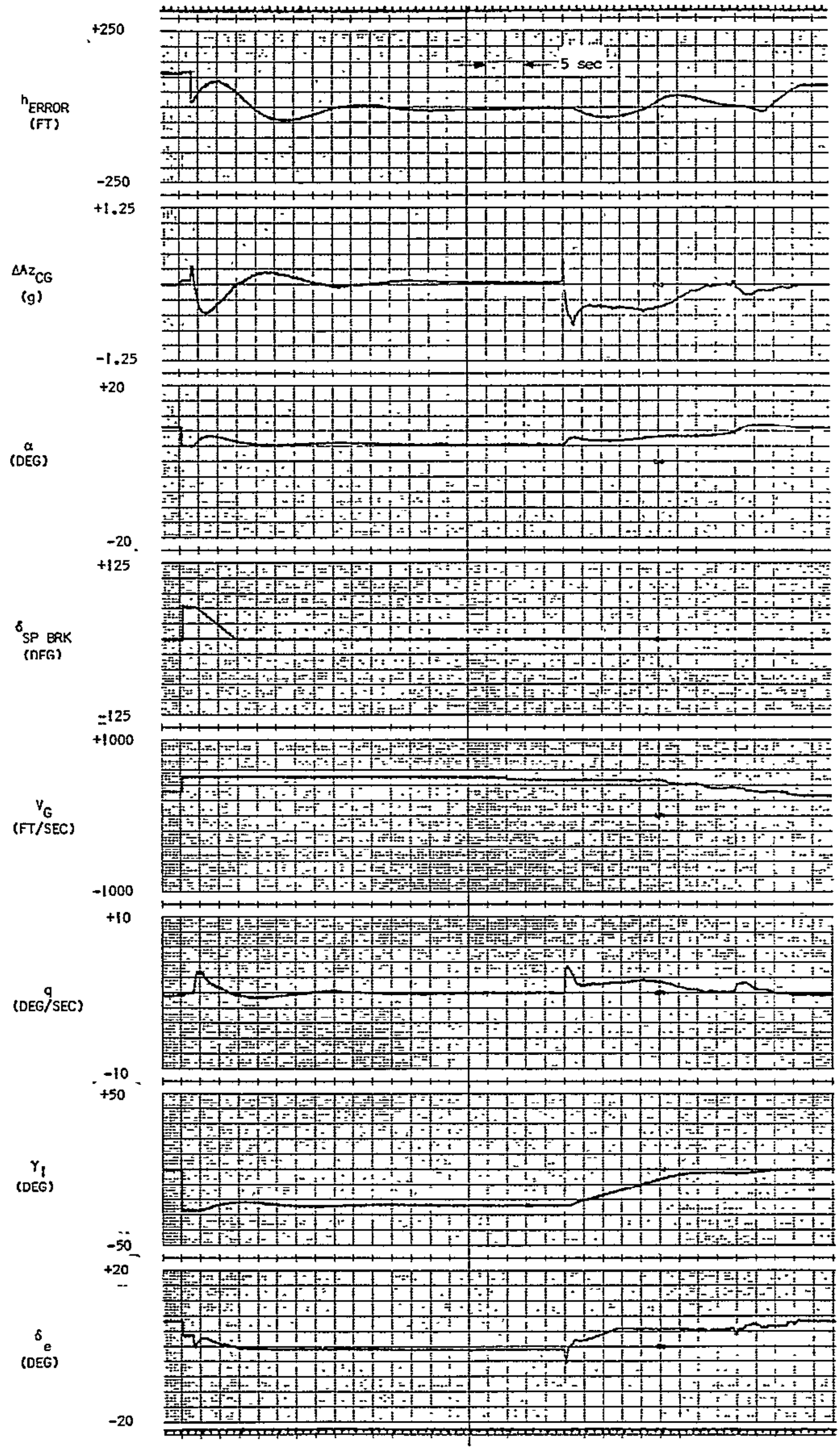


Figure 20

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING INERTIALLY DERIVED "AIR DATA" PARAMETERS, 140C ORBITER JUNE 1975.
32K PAYLOAD; .675 CG: MAX HEADWIND

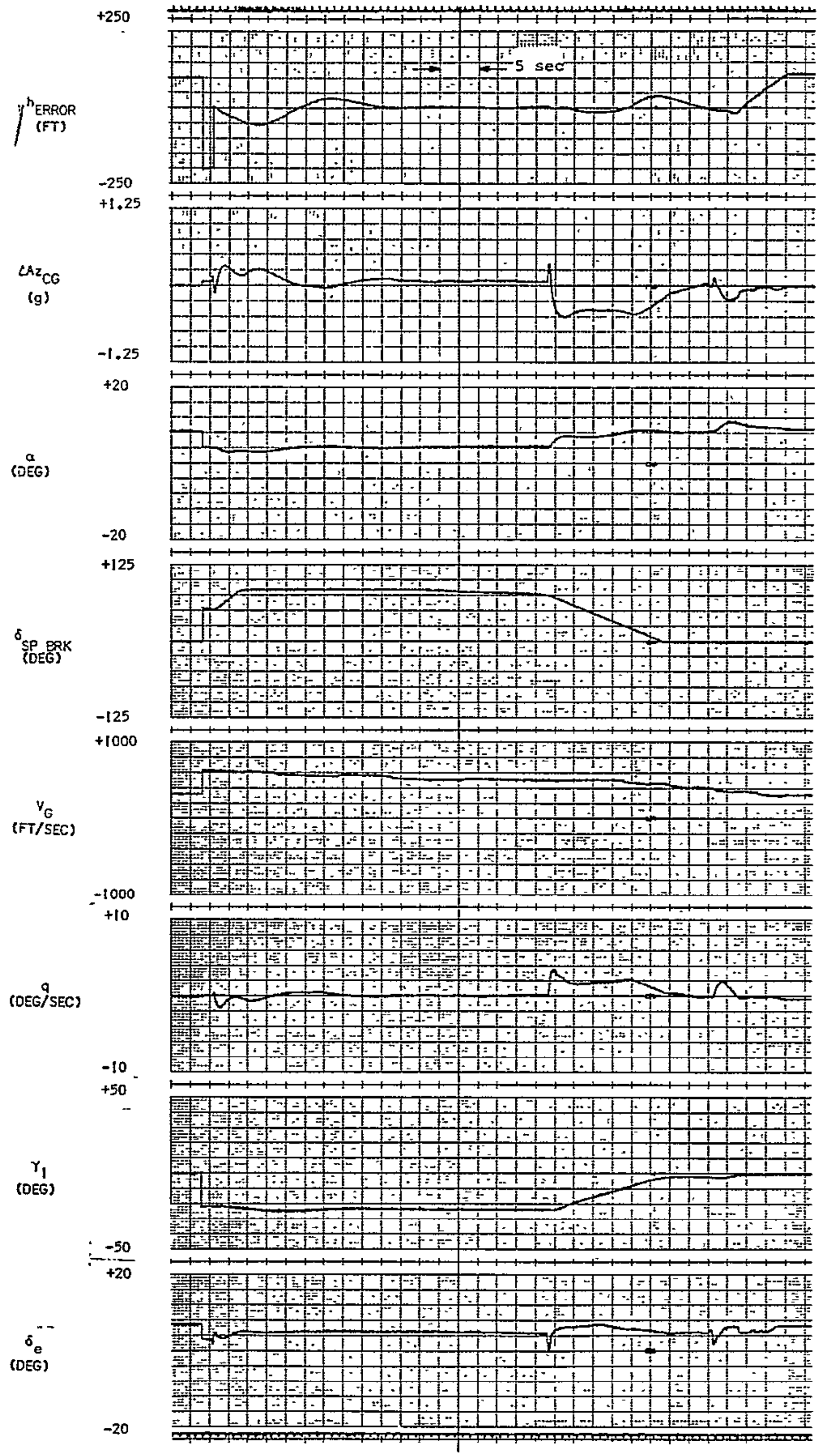
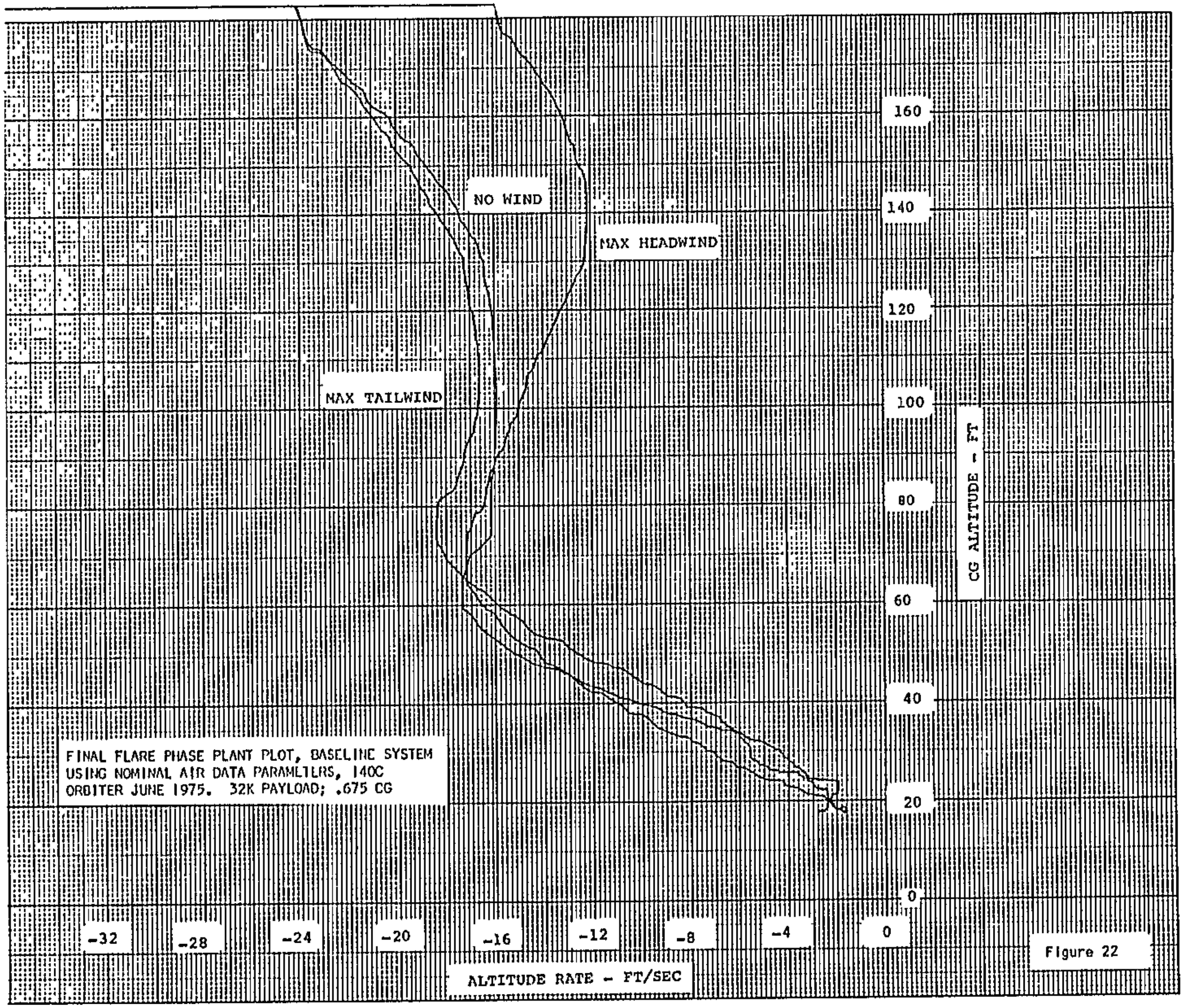


Figure 21

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING INERTIALLY DERIVED "AIR DATA" PARAMETERS, 140C ORBITER JUNE 1975.
32K PAYLOAD; .675 CG; MAX TAILWIND



NO WIND
MAX HEADWIND
MAX TAILWIND
FINAL FLARE PHASE PLANT PLOT, BASELINE SYSTEM
USING NOMINAL AIR DATA PARAMETERS, 140C
ORBITER JUNE 1975. 32K PAYLOAD; .675 CG
160
140
120
100
80
60
40
20
0
CG ALTITUDE - FT
-32
-28
-24
-20
-16
-12
-8
-4
0
ALTITUDE RATE - FT/SEC

Figure 22

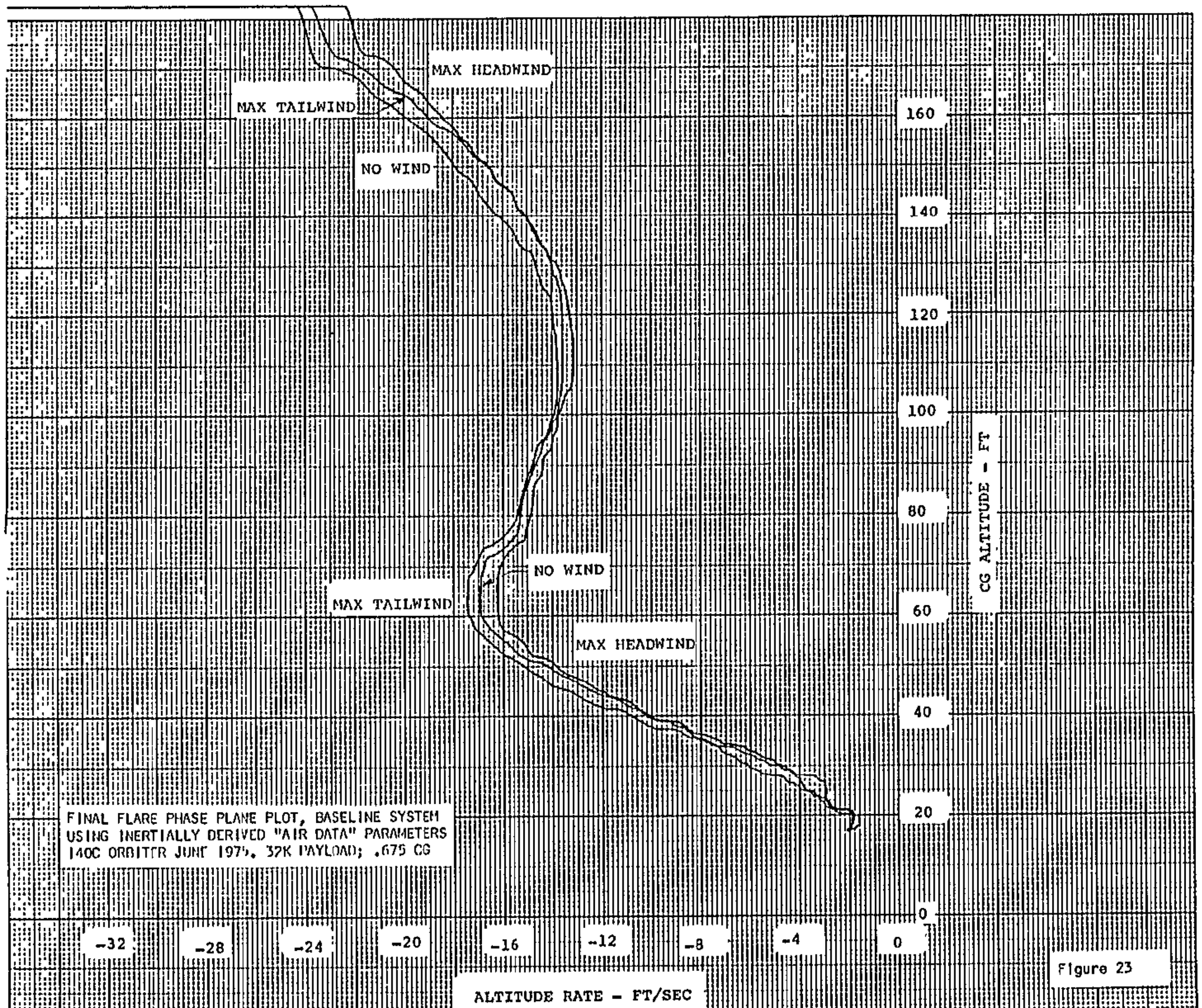

MAX HEADWIND
MAX TAILWIND
NO WIND
NO WIND
MAX TAILWIND
MAX HEADWIND
160
140
120
100
80
60
40
20
0
CG ALTITUDE - FT
FINAL FLARE PHASE PLANE PLOT, BASELINE SYSTEM
USING INERTIALLY DERIVED "AIR DATA" PARAMETERS
140C ORBITER JUNE 1975, 32K PAYLOAD; .675 CG
-32
-28
-24
-20
-16
-12
-8
-4
0
ALTITUDE RATE - FT/SEC

Figure 23

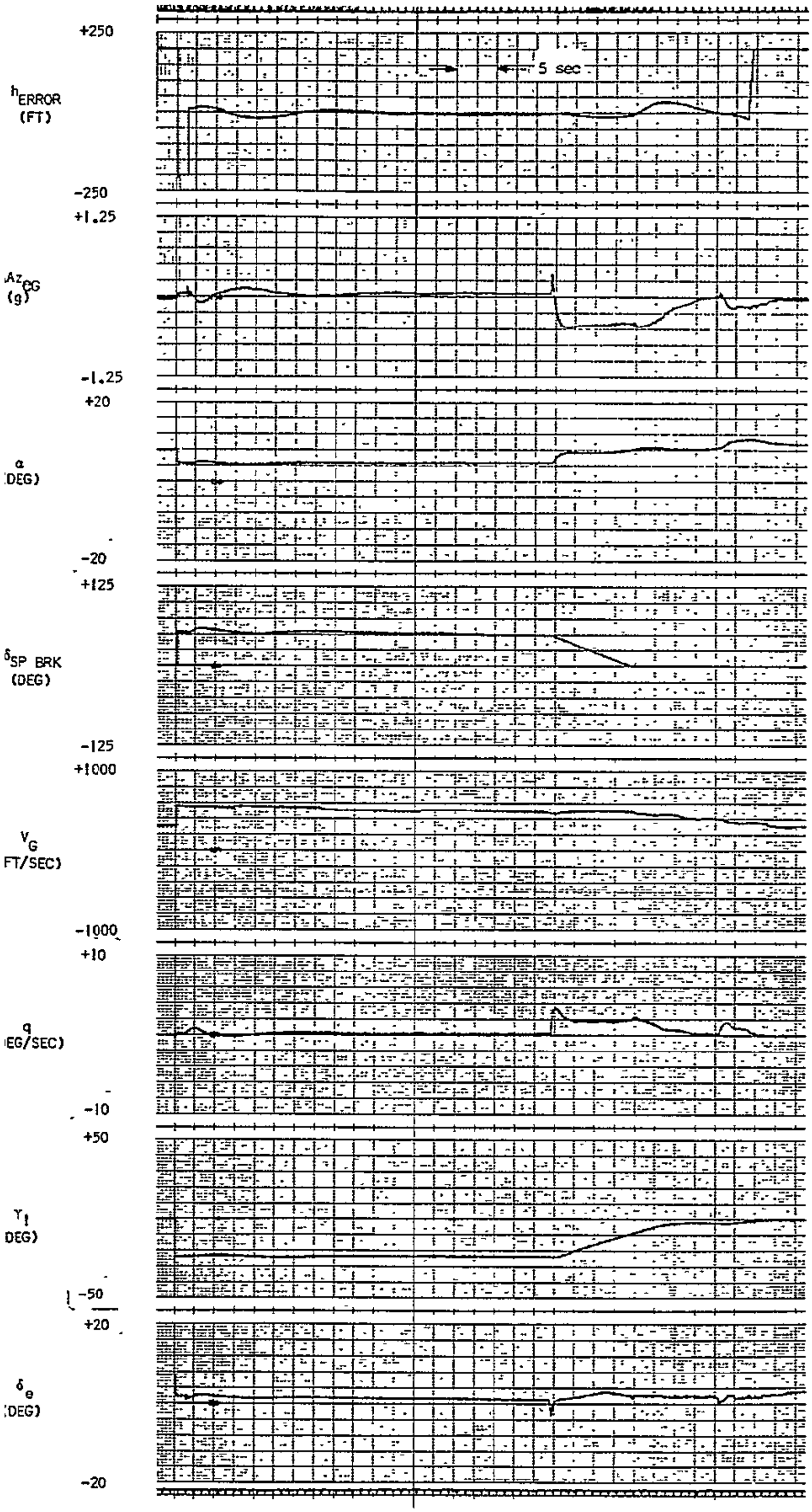


Figure 24

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING
NOMINAL AIR DATA PARAMETERS; 140C ORBITER
JUNE 1975. 32K PAYLOAD; .65 CG; NO WIND

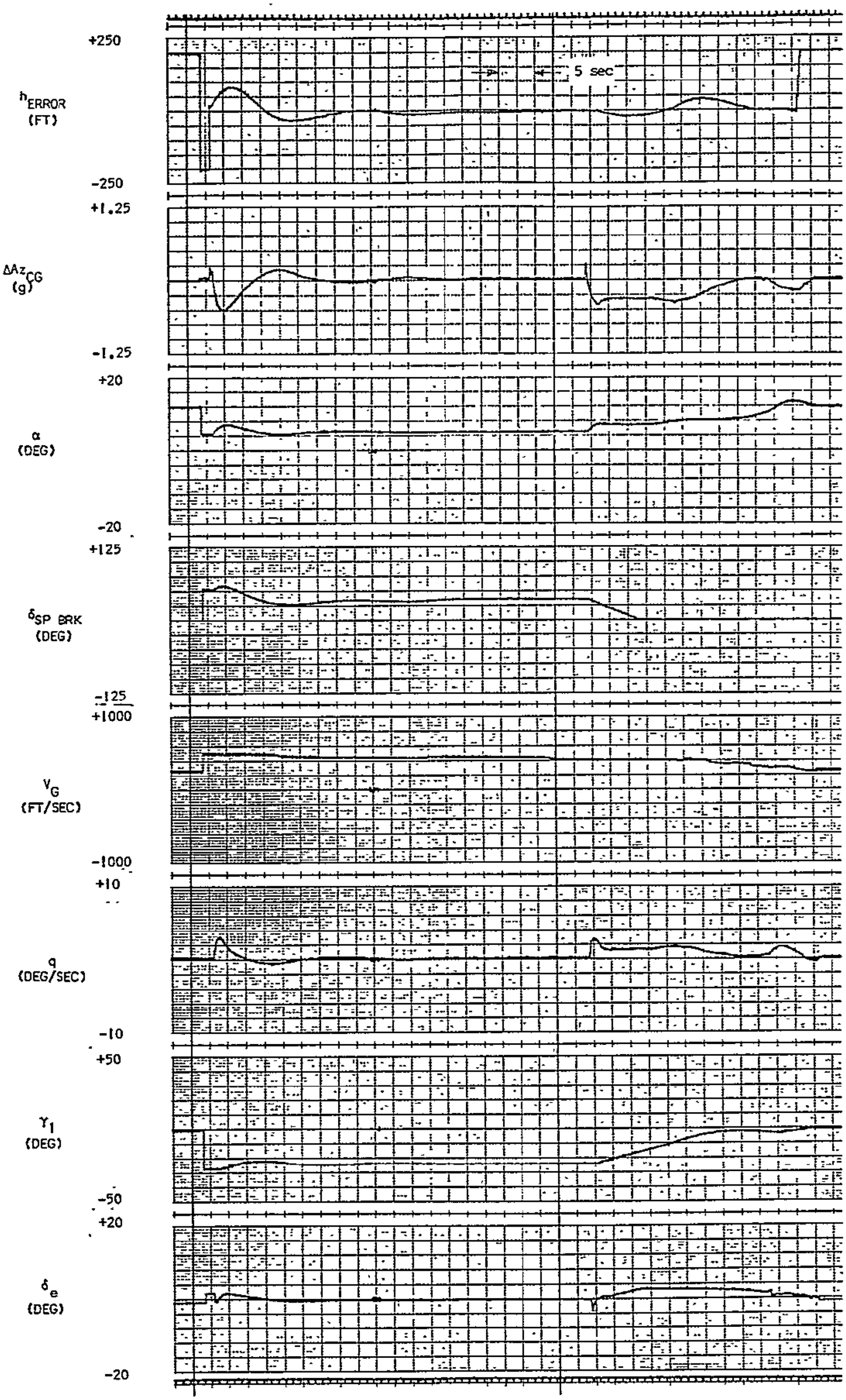


Figure 25

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING
NOMINAL AIR DATA PARAMETERS: 140C ORBITER
JUNE 1975. 32K PAYLOAD; .65 CG; MAX HEADWIND

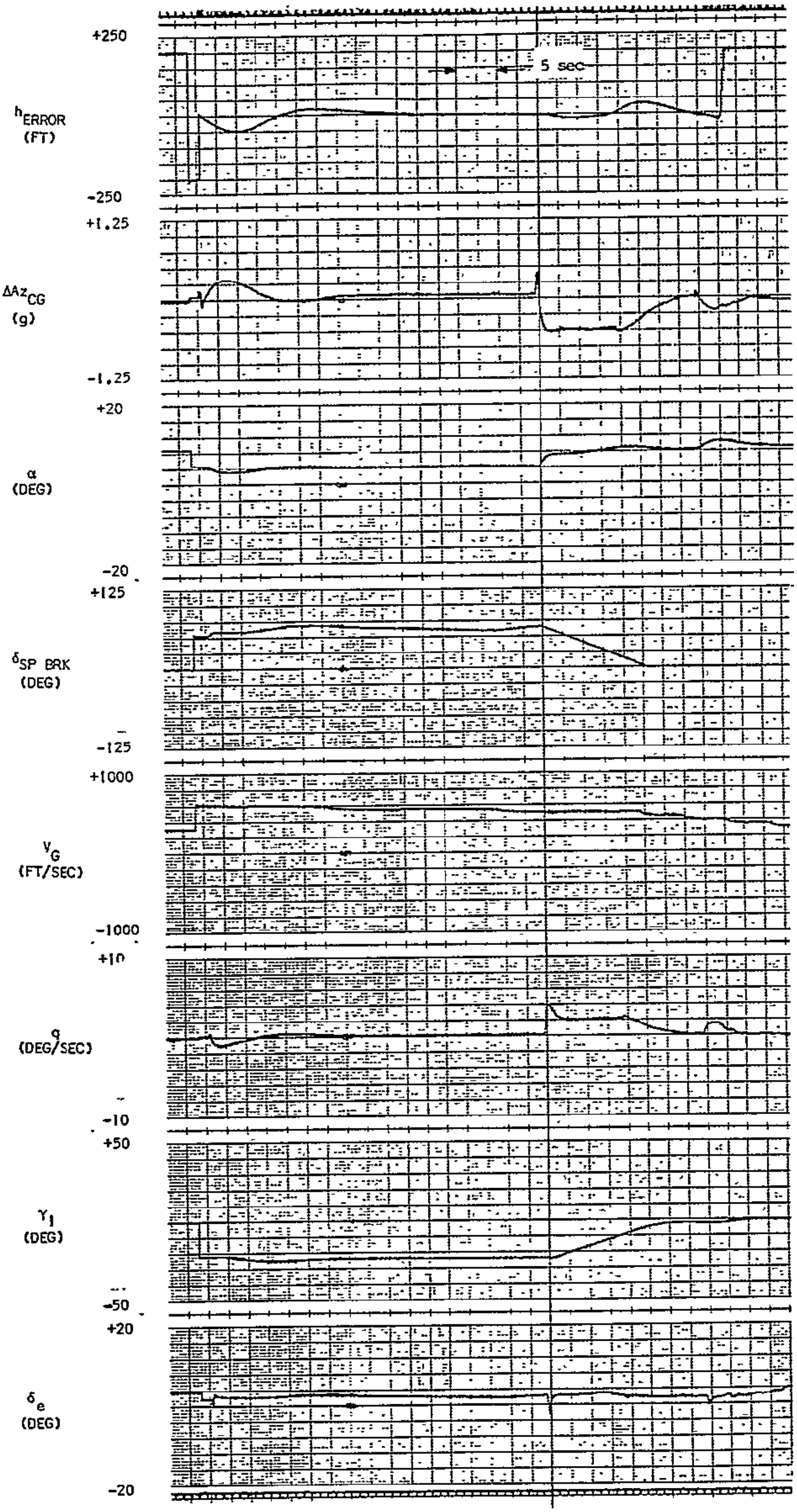


Figure 26

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING NOMINAL AIR DATA PARAMETERS; 140C ORBITER JUNE 1975. 32K PAYLOAD; .65 CG; MAX TAILWIND

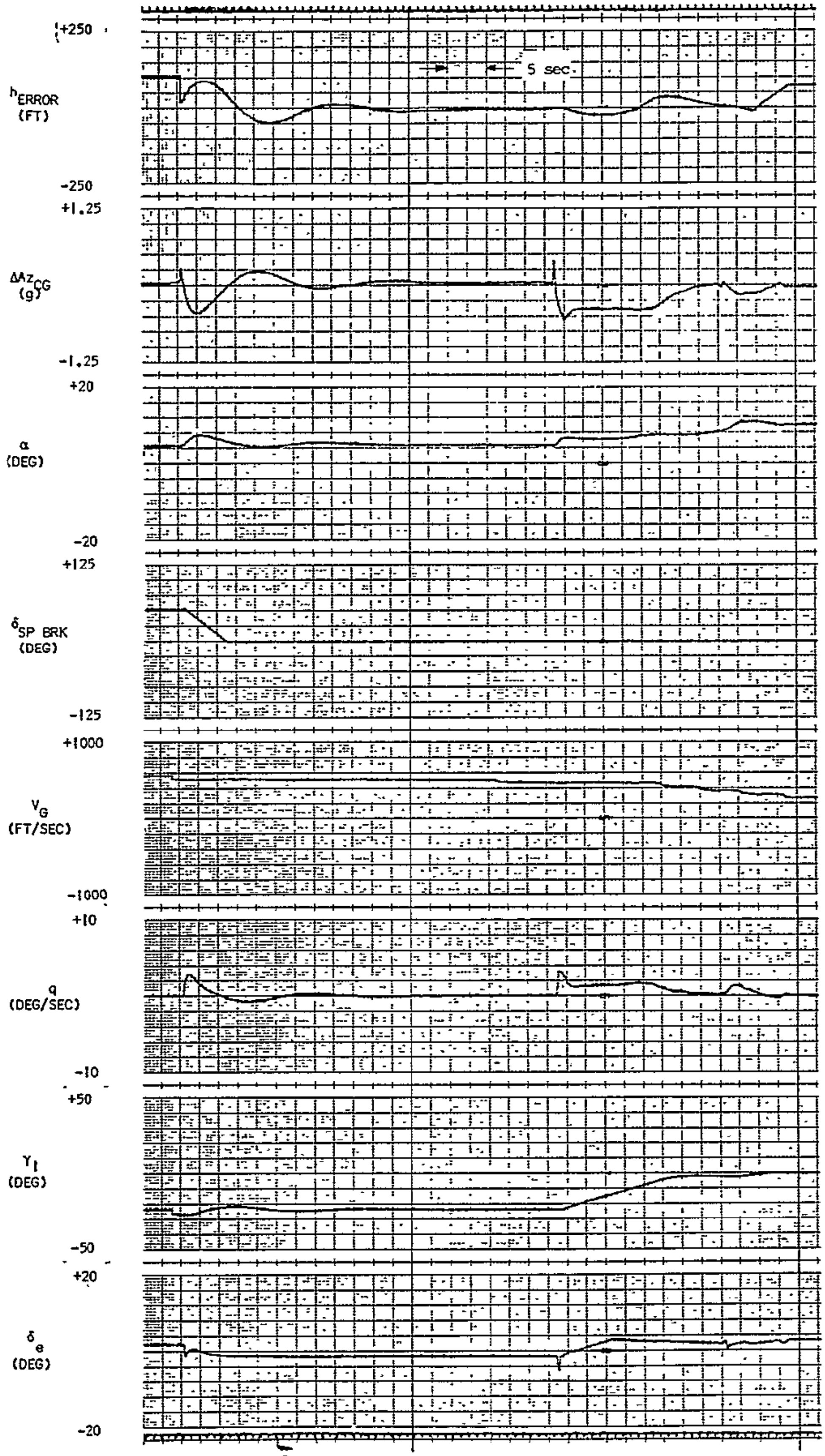


Figure 27

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING
INERTIALLY DERIVED "AIR DATA" PARAMETERS,
140C ORBITER JUNE 1975.
32K PAYLOAD; .65 CG; MAX HEADWIND

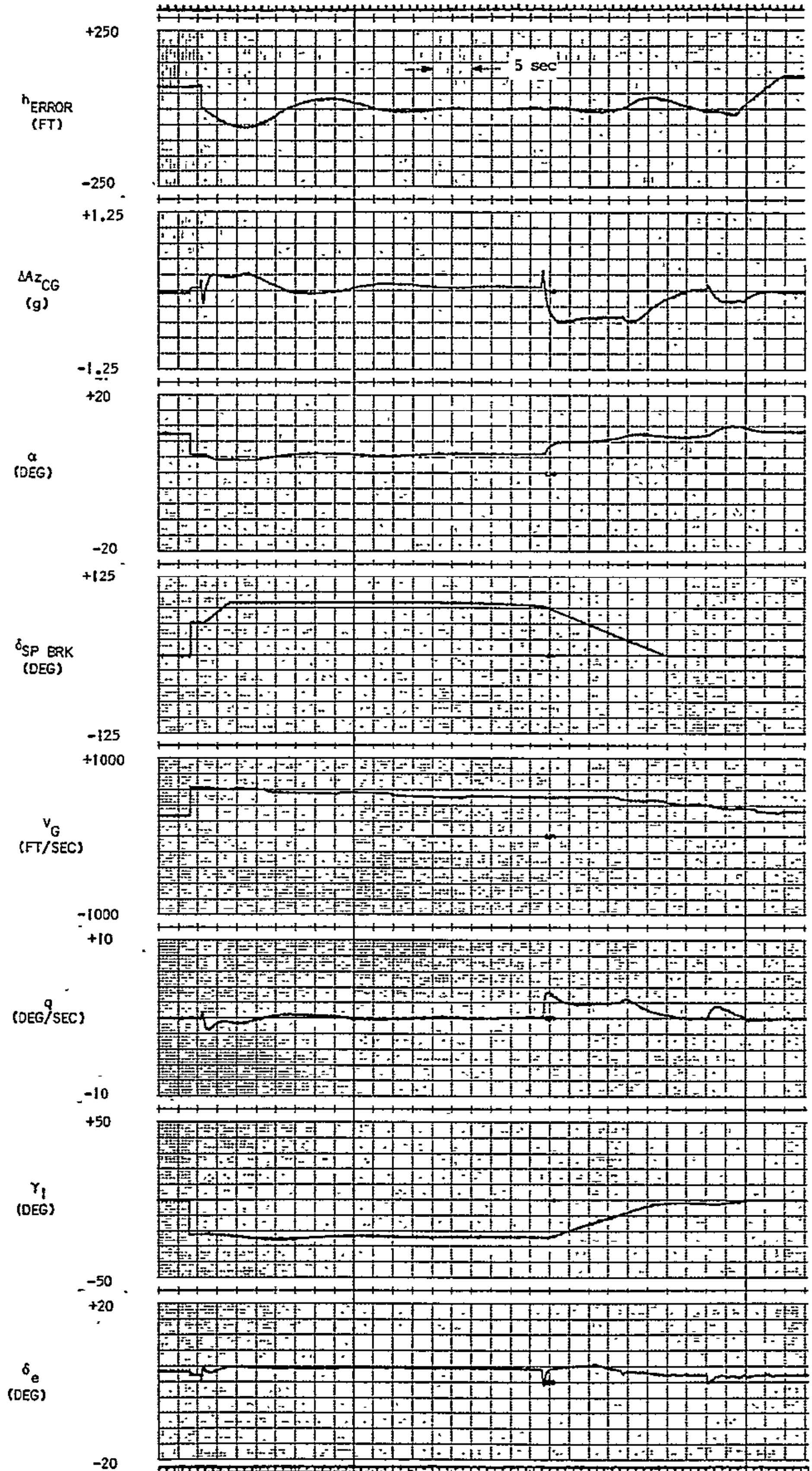


Figure 28

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING
INERTIALLY DERIVED "AIR DATA" PARAMETERS,
140C ORBITER JUNE 1975.
32K PAYLOAD; .65 CG; MAX TAILWIND

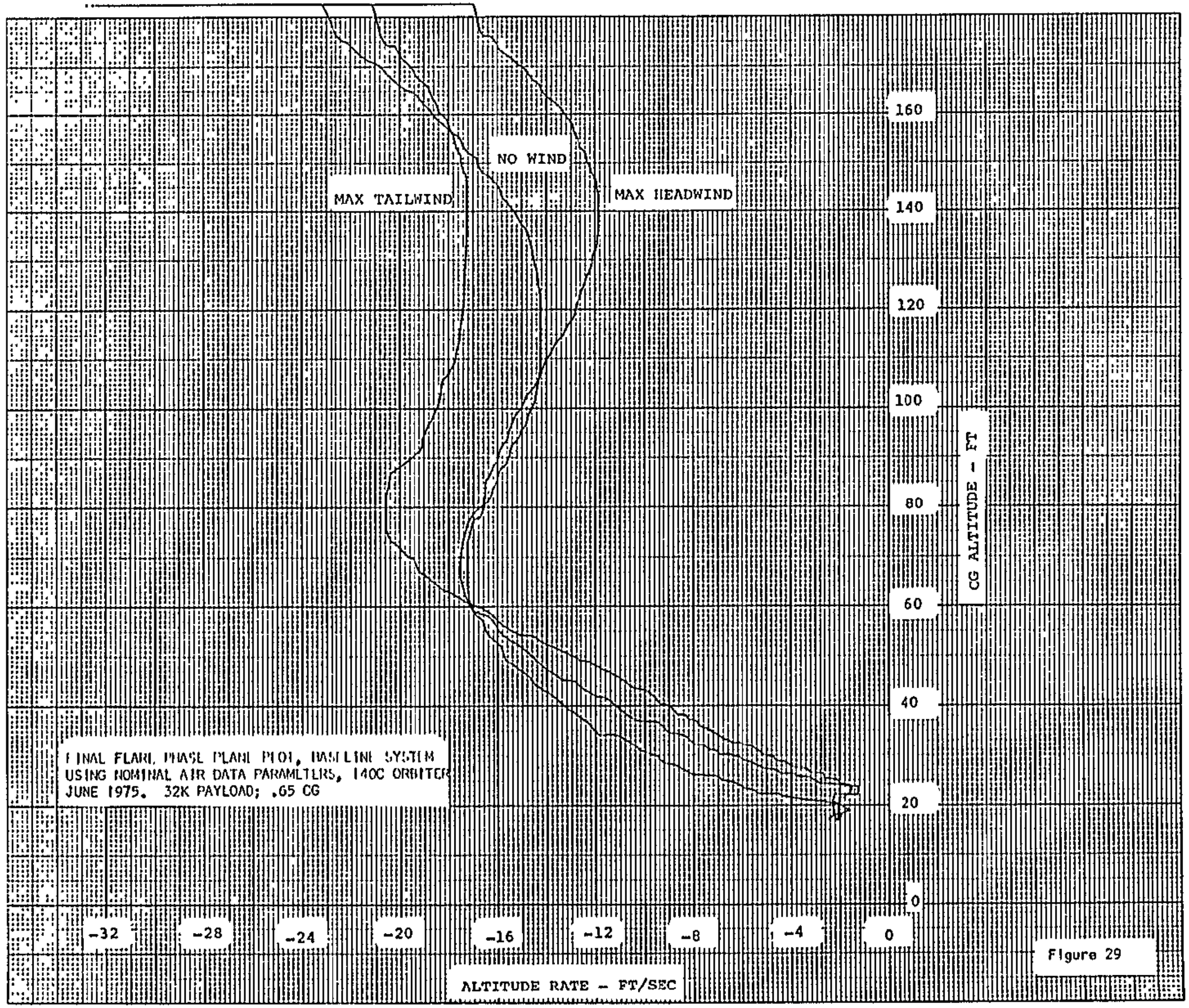
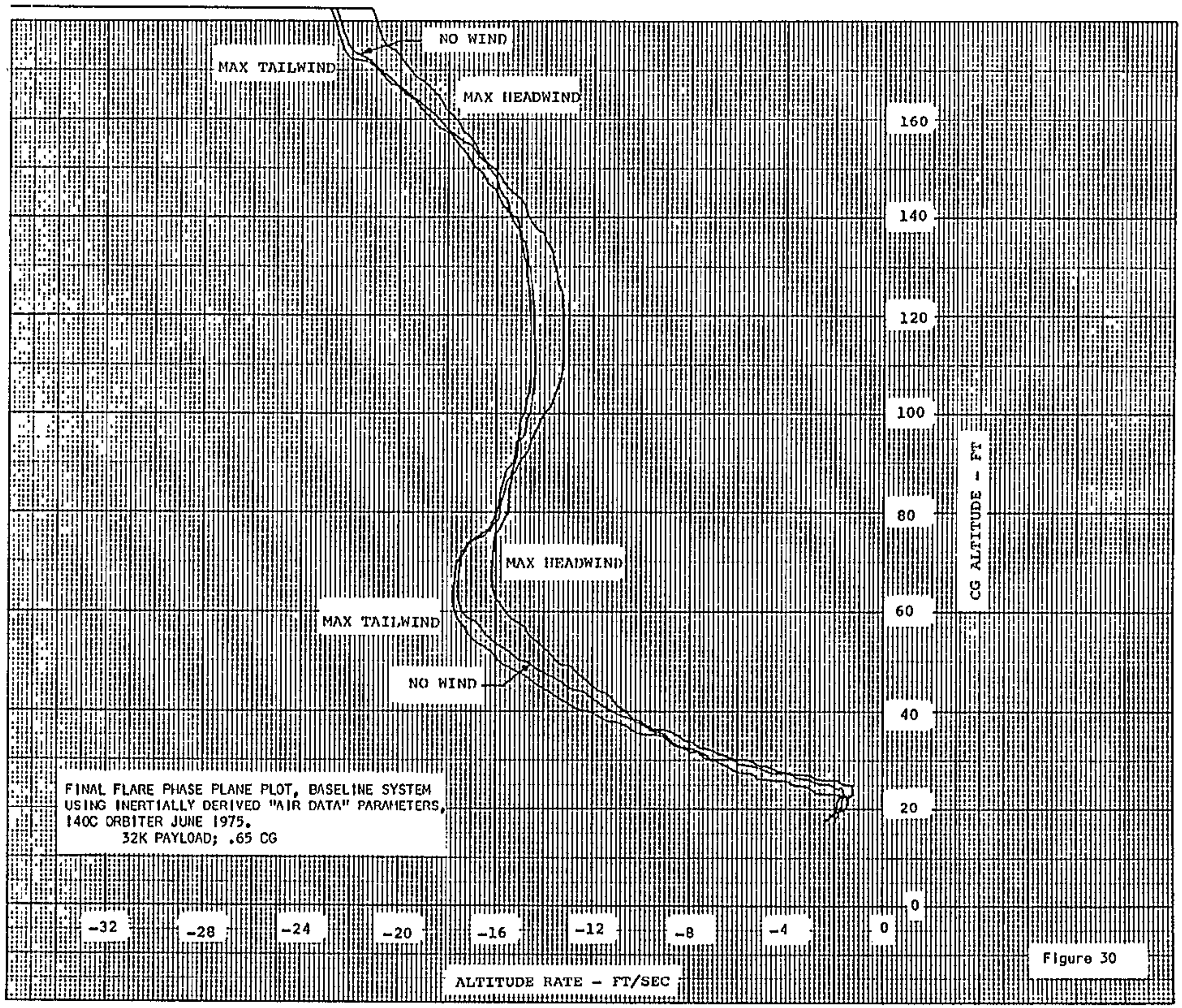

NO WIND
MAX TAILWIND
MAX HEADWIND
160
140
120
100
80
60
40
20
0
CG ALTITUDE - FT
FINAL FLARE PHASE PLANE PLOT, BASELINE SYSTEM
USING NOMINAL AIR DATA PARAMETERS, 140C ORBITER
JUNE 1975. 32K PAYLOAD; .65 CG
-32
-28
-24
-20
-16
-12
-8
-4
0
ALTITUDE RATE - FT/SEC

Figure 29



NO WIND
MAX TAILWIND
MAX HEADWIND
160
140
120
100
80
60
40
20
0
CG ALTITUDE - FT
MAX HEADWIND
MAX TAILWIND
NO WIND
FINAL FLARE PHASE PLANE PLOT, BASELINE SYSTEM USING INERTIALLY DERIVED "AIR DATA" PARAMETERS, 140C ORBITER JUNE 1975.
32K PAYLOAD; .65 CG
-32
-28
-24
-20
-16
-12
-8
-4
0
ALTITUDE RATE - FT/SEC

Figure 30

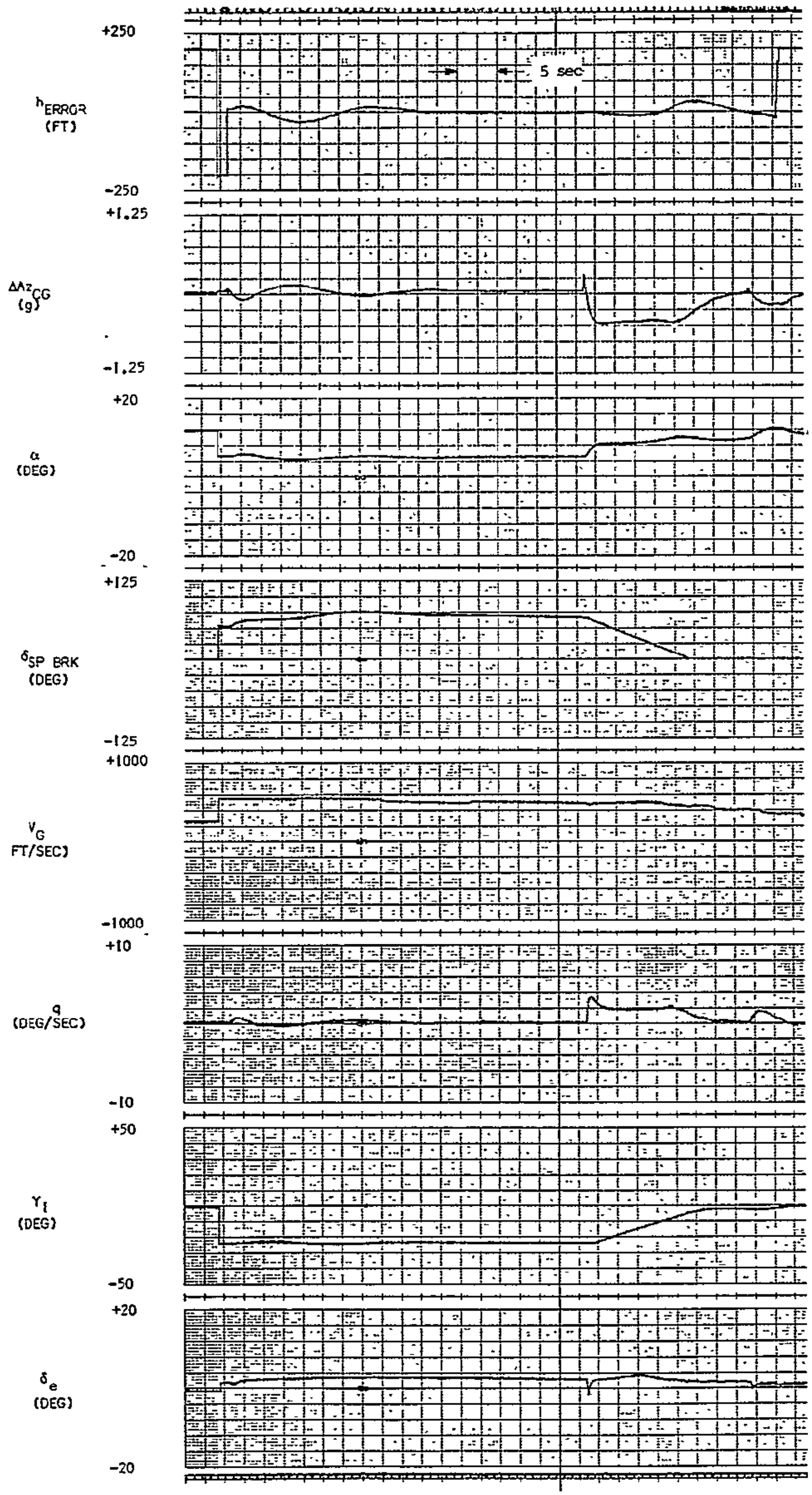


Figure 31

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING NOMINAL AIR DATA PARAMETERS, 140C ORBITER JUNE 1975. 65K PAYLOAD; .65 CG; NO WIND

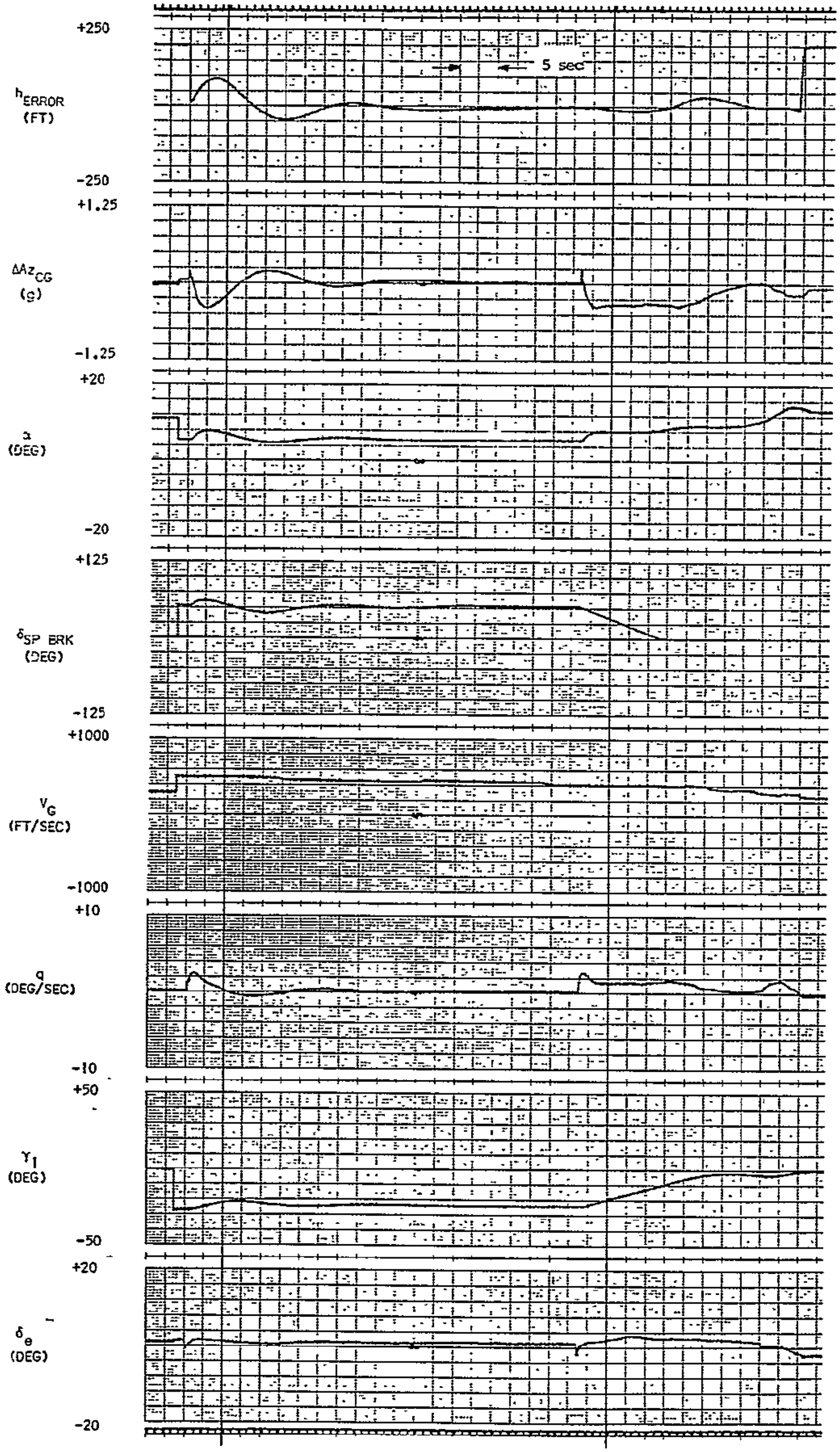


Figure 32

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING NOMINAL AIR DATA PARAMETERS, 140C ORBITER JUNE 1975. 65K PAYLOAD; .65 CG; MAX HEADWIND

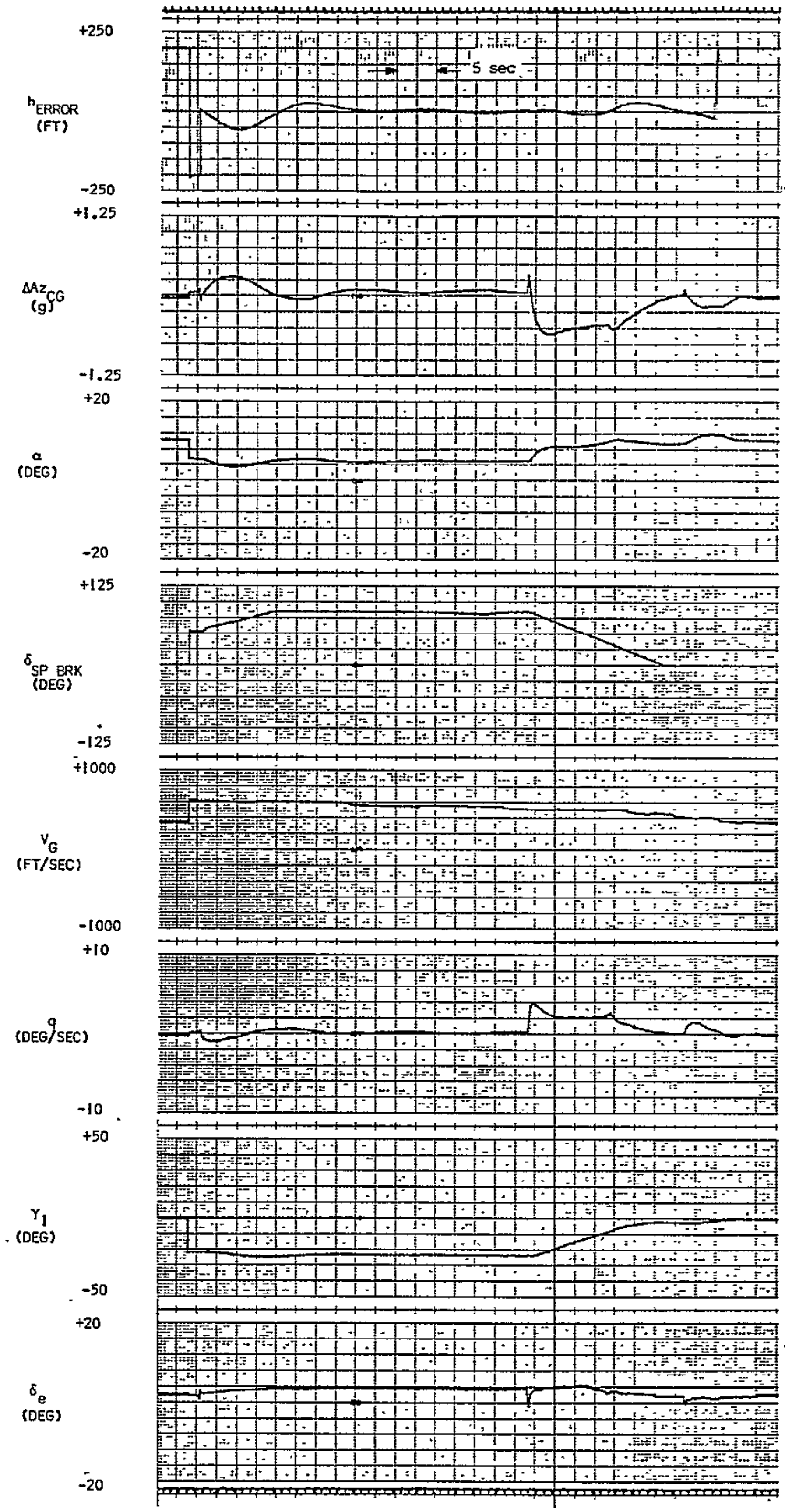


Figure 33

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING
NOMINAL AIR DATA PARAMETERS, 140C ORBITER
JUNE 1975. 65K PAYLOAD; .65 CG; MAX TAILWIND

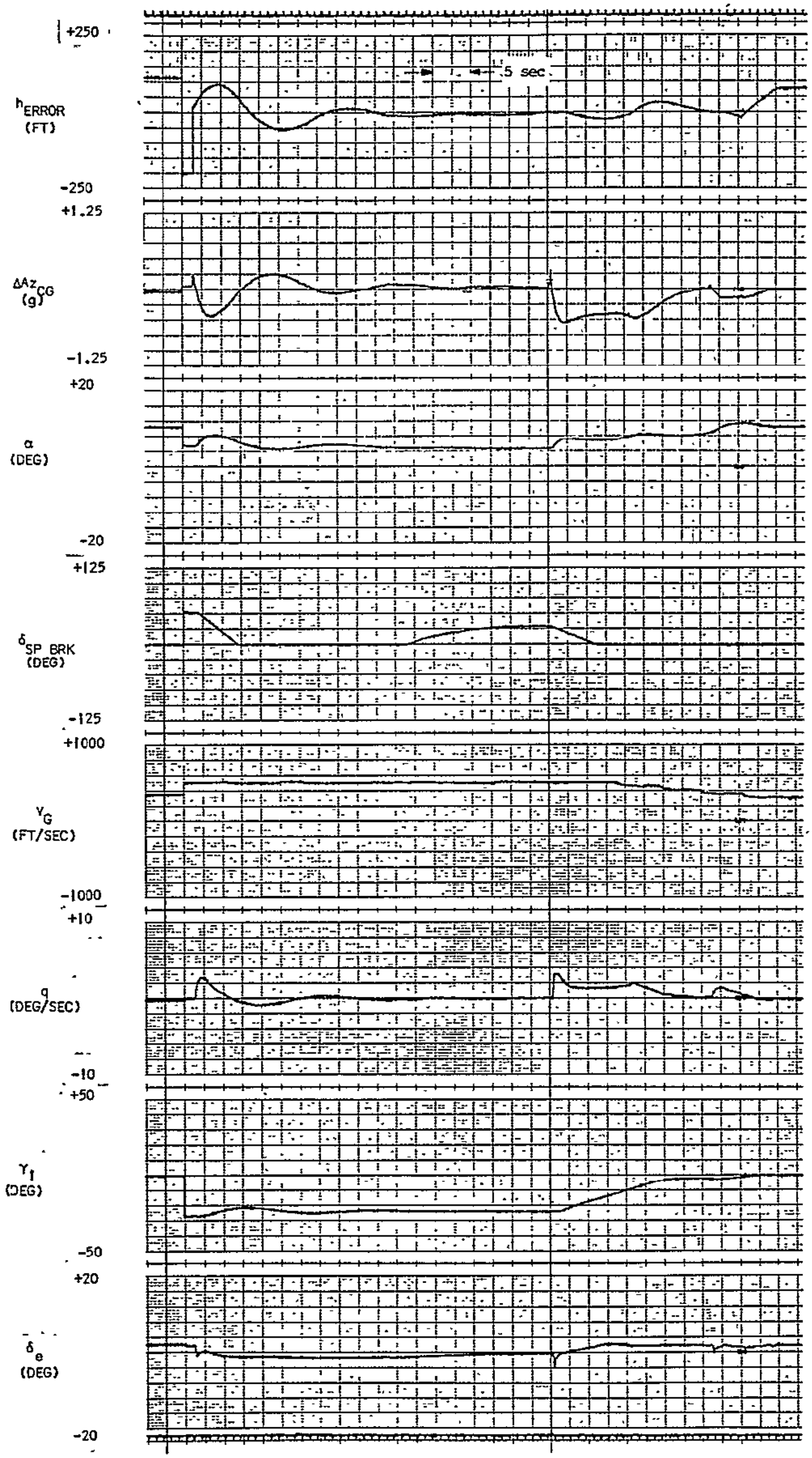


Figure 34

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING INERTIALLY DERIVED "AIR DATA" PARAMETERS, 140 ORBITER JUNE 1975.
65K PAYLOAD; .65 CG; MAX HEADWIND

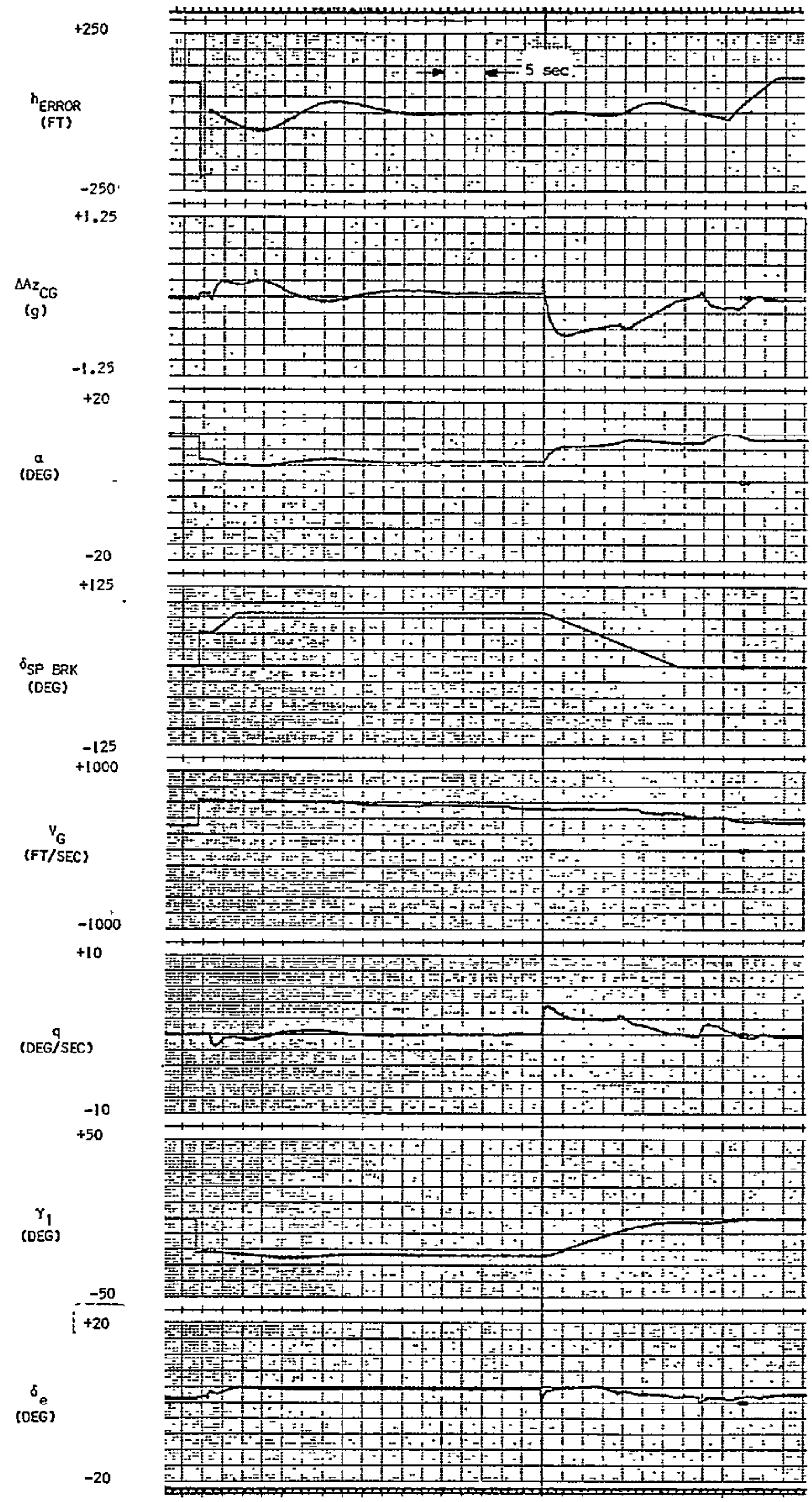


Figure 35

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING INERTIALLY DERIVED "AIR DATA" PARAMETERS, 140C ORBITER JUNE 1975.
65K PAYLOAD; .65 CG; MAX TAILWIND

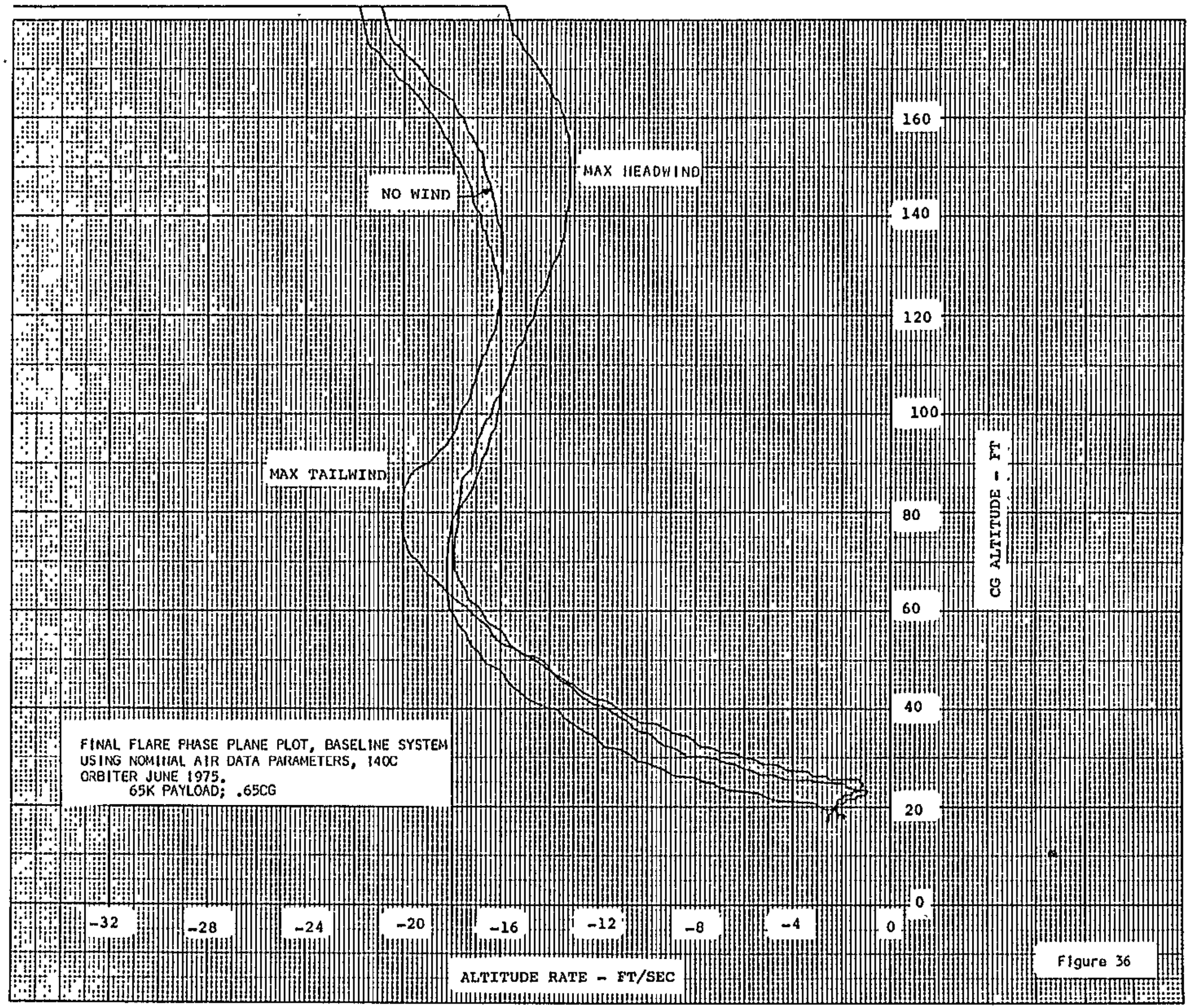

MAX HEADWIND
NO WIND
MAX TAILWIND
160
140
120
100
80
60
40
20
0
CG ALTITUDE - FT
-32
-28
-24
-20
-16
-12
-8
-4
0
ALTITUDE RATE - FT/SEC
FINAL FLARE PHASE PLANE PLOT, BASELINE SYSTEM
USING NOMINAL AIR DATA PARAMETERS, 140C
ORBITER JUNE 1975.
65K PAYLOAD; .65CG

Figure 36

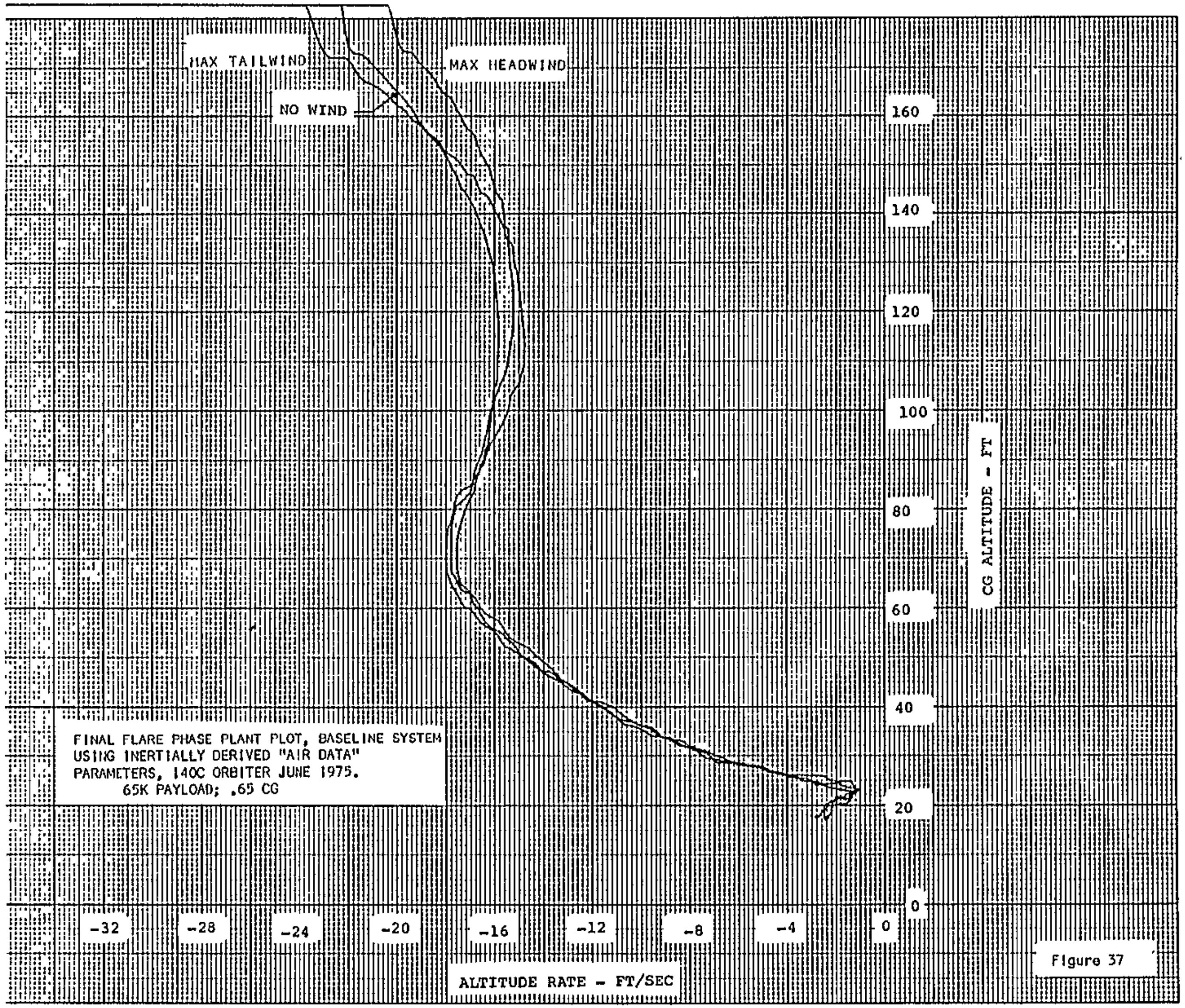

MAX TAILWIND
MAX HEADWIND
NO WIND
FINAL FLARE PHASE PLANT PLOT, BASELINE SYSTEM USING INERTIALLY DERIVED "AIR DATA" PARAMETERS, 140C ORBITER JUNE 1975. 65K PAYLOAD; .65 CG
160
140
120
100
80
60
40
20
0
CG ALTITUDE - FT
-32
-28
-24
-20
-16
-12
-8
-4
0
ALTITUDE RATE - FT/SEC

Figure 37

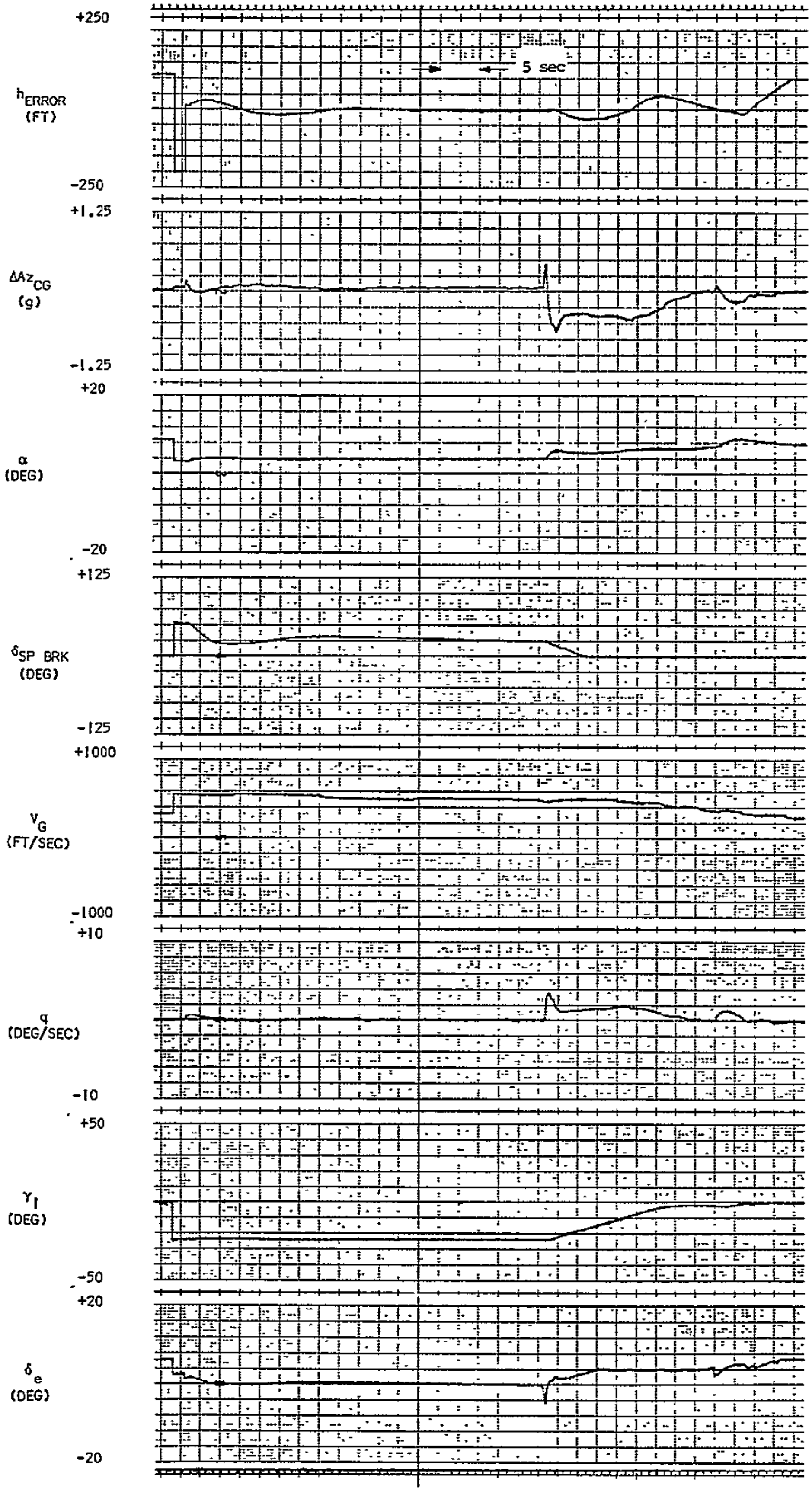


Figure 38

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING INERTIALLY DERIVED "AIR DATA" PARAMETERS AND GDQ = $\frac{22.09}{\sqrt{\bar{q}_I}}$

140C ORBITER JUNE 1975
ØK PAYLOAD; .675 CG; NO WIND

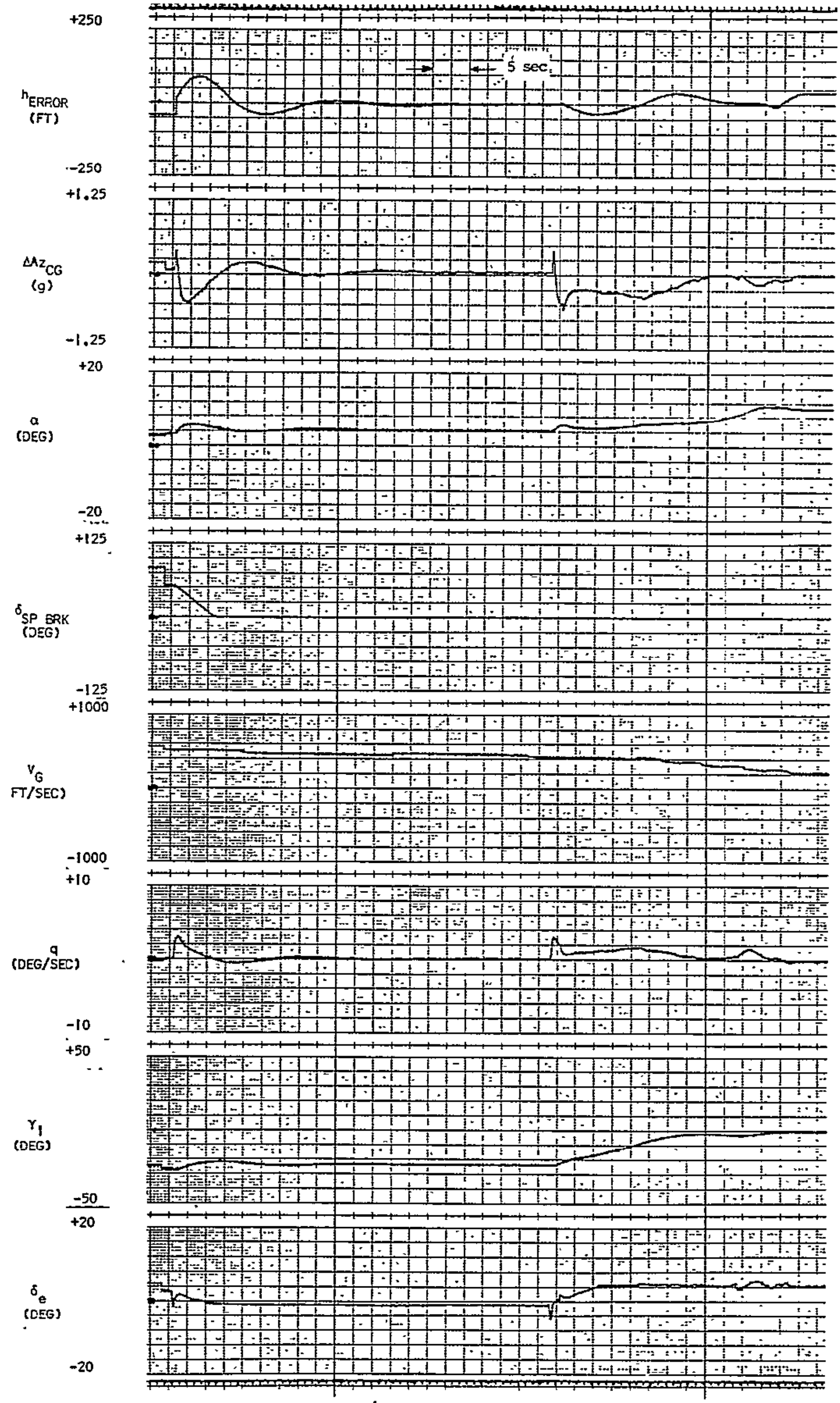


Figure 39

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING INERTIALLY DERIVED "AIR DATA" PARAMETERS AND GDQ = $\frac{22.09}{\sqrt{\bar{q}_I}}$

140C ORBITER JUNE 1975
ØK PAYLOAD; .675 CG; MAX HEADWIND

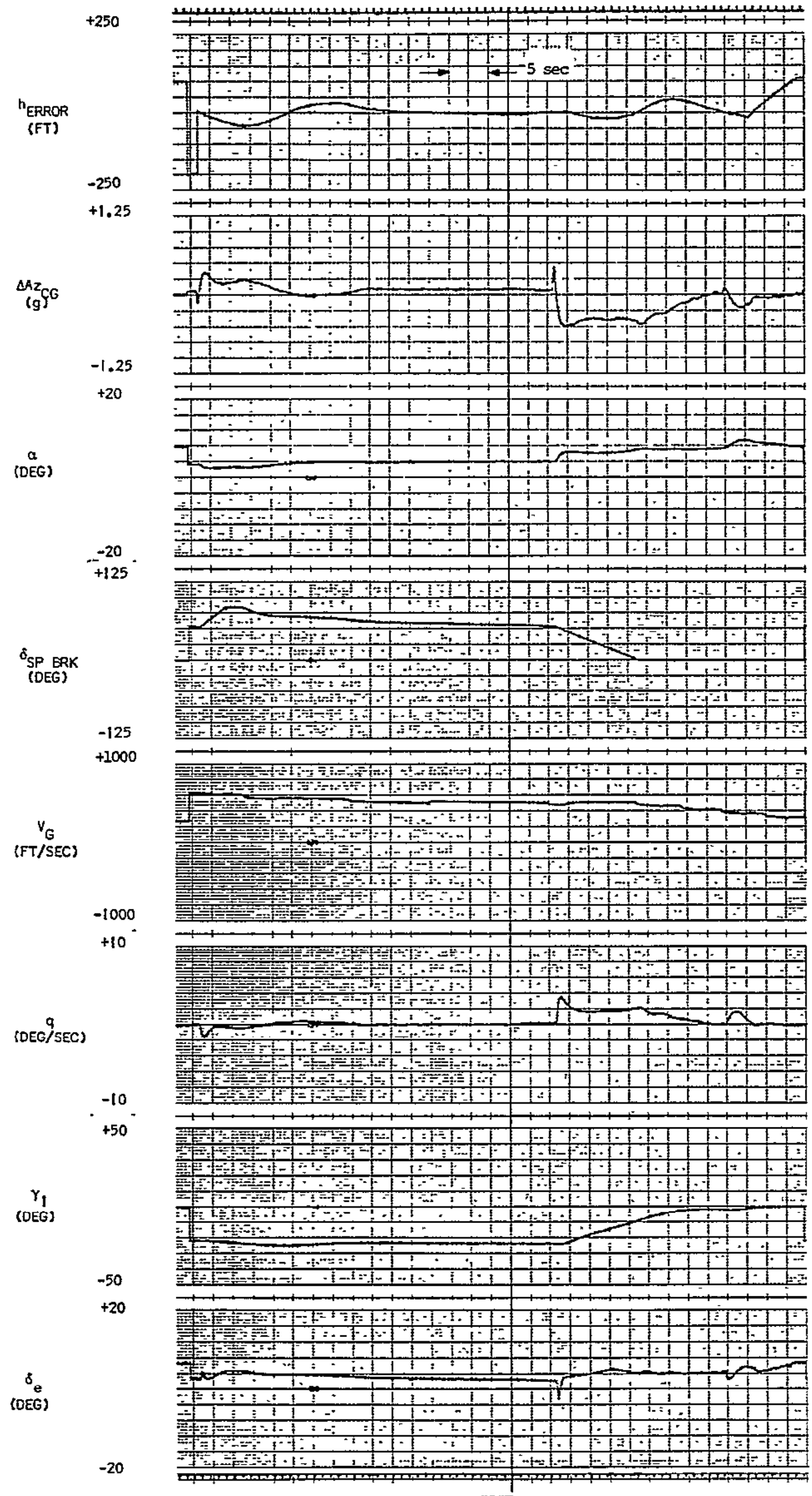


Figure 40

AUTOLAND TRAJECTORY, BASELINE SYSTEM USING INERTIALLY DERIVED "AIR DATA" PARAMETERS AND $GDQ = \frac{22.09}{\sqrt{\bar{q}_I}}$

140C ORBITER JUNE 1975

ØK PAYLOAD; .675 CG; MAX TAILWIND

AUTOLAND TRAJECTORY AND TOUCHDOWN PERFORMANCE FOR NO WIND, MAX HEADWIND, AND MAX TAILWIND CONDITIONS (NO TURBULENCE)

BASELINE SYSTEM USING NOMINAL AIR DATA PARAMETERS

PAYLOAD	CG	WIND	$\bar{q}_{max}$ lb/Ft^2	MEAN SP BRK POSITION FOR $\gamma_{STEEP} = -24°$ $\delta_{SP\ BRK}$ DEG	GEAR DEPLOYMENT ALTITUDE h_{GEAR} FT	CG FINAL FLARE INITIATE ALTITUDE h_{FCG} FT	TOUCHDOWN PERFORMANCE $\dot{h}$ FT/SEC	X_{THRESH} FT	V_T FT/SEC	V_G FT/SEC	θ DEG
ØK	.675	None	300	30	833	75	-2.1	3345	279.8	279.5	7.5
ØK	.675	Max HW	300	2	300	75	-2.3	2179	249.0	227.0	10.5
ØK	.675	Max TW	300	45	1900	76	-1.5	3687	291.6	302.5	6.8
32K	.675	None	300	50	900	79	-1.9	3483	300.0	299.6	8.0
32K	.675	Max HW	300	30	300	82	-2.3	2424	267.2	245.0	10.9
32K	.675	Max TW	308	60	1900	89	-2.0	3879	307.2	318.0	7.6
32K	.65	None	300	52	900	86	-2.1	3456	303.1	302.4	9.2
32K	.65	Max HW	300	35	300	81	-2.6	2279	275.8	253.7	11.5
32K	.65	Max TW	308	65	1900	87	-1.8	3855	310.8	321.3	8.7
65K	.65	None	305	70	920	88	-2.6	3388	311.3	311.1	10.5
65K	.65	Max HW	305	50	300	85	-2.1	2005	295.6	272.7	13.1
65K	.65	Max TW	315	85	1900	91	-2.1	3812	318.8	329.3	10.0

TABLE 2

AUTOLAND TRAJECTORY AND TOUCHDOWN PERFORMANCE FOR NO WIND, MAX HEADWIND, AND MAX TAILWIND CONDITIONS (NO TURBULENCE)

BASELINE SYSTEM USING INERTIALLY DERIVED AIR DATA PARAMETERS

PAYLOAD	CG	WIND	$\bar{q}$ MAX lb/FT^2	MEAN SP BRK POSITION FOR $\gamma_{STEEP} = -24°$ $\delta_{SP\ BRK}$ DEG	GEAR DEPLOYMENT ALTITUDE h_{GEAR} FT	CG FINAL FLARE INITIATE ALTITUDE h_{FCG} FT	TOUCHDOWN PERFORMANCE $\dot{h}$ FT/SEC	X_{THRESH} FT	V_T FT/SEC	V_G FT/SEC	θ DEG
ØK	.675	NONE	300	30	900	77	-1.9	3316	280.7	280.4	7.6
ØK	.675	MAX HW	300	0	300	76	-2.2	2250	251.5	229.0	10.3
ØK	.675	MAX TW	300	67	875	75	-1.7	3550	282.9	293.7	7.4
32K	.675	NONE	308	50	920	79	-1.9	3488	299.8	299.3	8.0
32K	.675	MAX HW	365	0	575	80	-1.8	2965	291.6	269.2	8.6
32K	.675	MAX TW	300	85	1020	79	-1.6	3698	289.8	300.1	8.7
32K	.65	NONE	308	52	933	80	-1.7	3444	303.2	302.9	9.1
32K	.65	MAX HW	360	0	575	81	-2.5	2886	299.1	276.8	9.5
32K	.65	MAX TW	300	85	1080	89	-2.5	3513	293.1	303.6	10.1
65K	.65	NONE	315	70	1000	92	-2.9	3365	310.4	310.1	10.7
65K	.65	MAX HW	380	0	1275	89	-2.4	3034	313.7	291.7	10.1
65K	.65	MAX TW	315	85	1900	87	-2.0	3710	319.3	329.8	10.1

TABLE 3

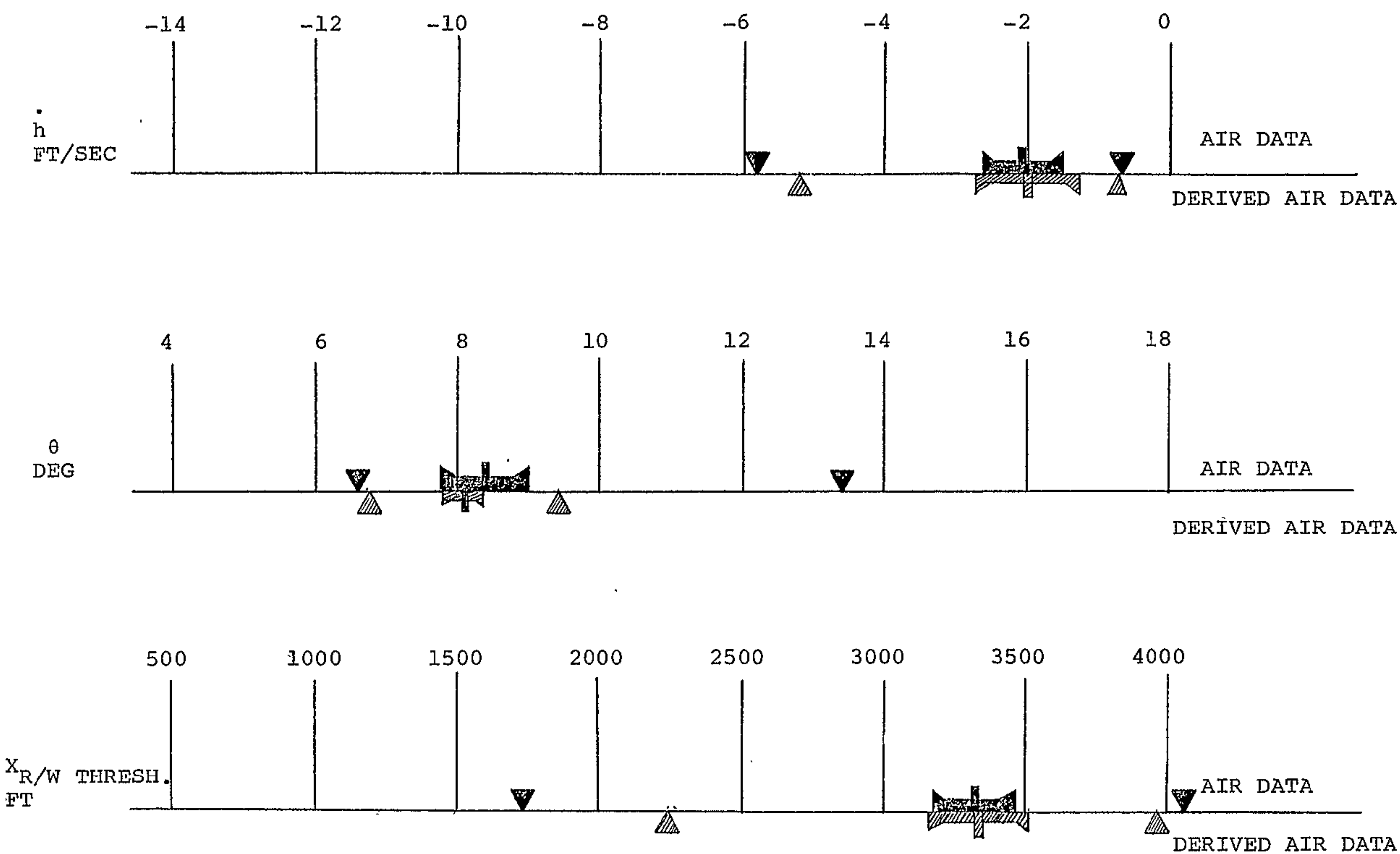


FIGURE 41

TOUCHDOWN PERFORMANCE FOR RANDOM WINDS WITH TURBULENCE (400 RUNS)
32K PAYLOAD; .675 CG; 140C ORBITER, JUNE 1975

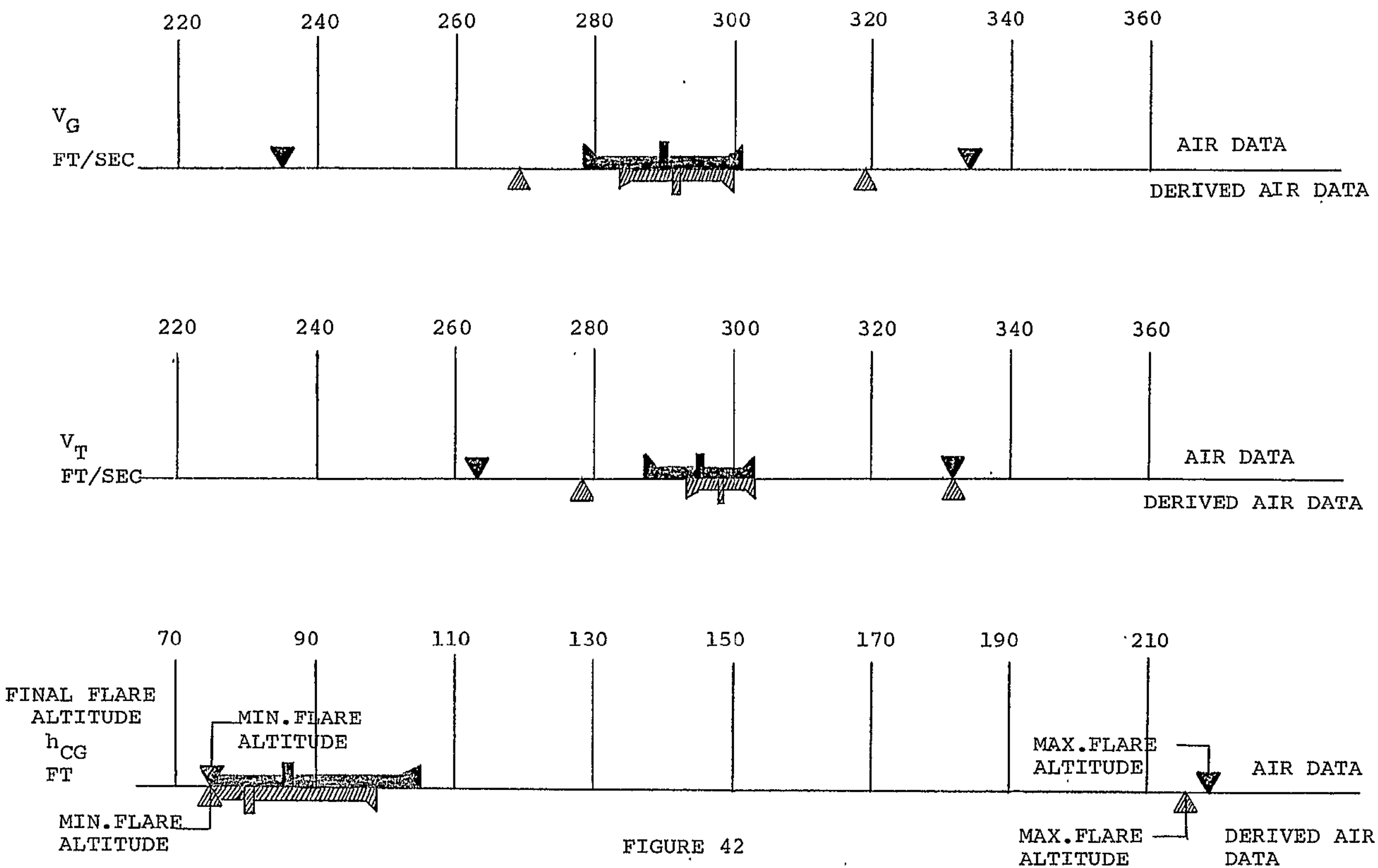


FIGURE 42
TOUCHDOWN PERFORMANCE FOR RANDOM WINDS WITH TURBULENCE (400 RUNS)

32K PAYLOAD; .675 CG; 140C ORBITER, JUNE 1975

AUTOLAND TRAJECTORY AND TOUCHDOWN PERFORMANCE FOR NO WIND, MAX HEADWIND AND MAX TAILWIND CONDITIONS (NO TURBULENCE)

BASELINE SYSTEM USING INERTIALLY DERIVED AIR DATA PARAMETERS AND $GDQ = \frac{22.09}{\sqrt{\bar{q}_I}}$

PAYLOAD	CG	WIND	MEAN SP BRK POSITION FOR $\gamma_{STEEP} = -24°$ $\delta_{SP\ BRK}$ DEG	$\bar{q}_{MAX}$ lb/FT^2	GEAR DEPLOYMENT ALTITUDE h_{GEAR} FT	CG FINAL FLARE INITIATE ALTITUDE h_{FCG} FT	TOUCHDOWN PERFORMANCE $\dot{h}$ FT/SEC	X_{THRESH} FT	V_T FT/SEC	V_G FT/SEC	θ DEG
ØK	.675	NONE	30	300	900	78	-2.0	3368	280.3	279.9	7.5
ØK	.675	MAX HW	0	300	300	76	-2.4	2275	250.7	228.6	10.1
ØK	.675	MAX TW	67	300	875	76	-1.7	3615	282.9	293.5	7.4
32K	.675	NONE	50	308	920	76	-2.0	3551	298.5	298.2	7.9
32K	.675	MAX HW	0	365	575	76	-2.2	2961	291.6	269.6	8.2
32K	.675	MAX TW	85	300	1020	77	-1.9	3658	290.2	300.9	8.5
32K	.65	NONE	52	308	933	77	-2.0	3428	303.4	303.0	8.9
32K	.65	MAX HW	0	360	575	77	-2.4	2868	300.9	278.5	9.2
32K	.65	MAX TW	85	300	1080	82	-1.9	3485	294.0	304.6	10.0
65K	.65	NONE	70	315	1000	85	-2.9	3291	314.9	314.4	10.4
65K	.65	MAX HW	0	380	1275	82	-2.8	3078	313.8	291.3	10.2
65K	.65	MAX TW	85	315	1900	84	-1.7	3805	316.4	327.1	10.2

TABLE 4

SECTION 4

JUNE 1975 AERO DATA UPDATE AND CHECK RUNS

AUTOLAND MEMO NO. 4

DATE: 14 January 1976

COPY: ______________

NASA SHUTTLE

AUTOLAND MEMO

TASK NO: 1

TASK NAME: June 1975 Aero Data Update and Check Runs

The enclosed time histories (Figures 1 through 24) show the response of the June 1975 150,000 lb forward CG free vehicle to 1° elevon, aileron and rudder steps using the Sperry 6 DOF simulation. In order to compare these to similar runs made by Rockwell which used linearized perturbation equations, the initial values of acceleration ($\dot{V}$, $\ddot{h}$) and vertical velocity ($\dot{h}$) were forced to zero in the Sperry simulation. The responses agreed with the Rockwell results within 10 to 15%, the best agreement occurring during the initial seconds of the response. The errors are due to the different implementations of the vehicle dynamics and strip chart inaccuracies.

Table I defines the symbology used by Sperry for the time histories. Table II shows the trim conditions as determined and used by the Sperry simulation, and Table III shows the Rockwell initial conditions. (Note that in actuality the Rockwell $\dot{V}_{ic} = 0$). The last two tables closely agree.

Figures 25, 26 and 27 are free vehicle step responses using the Sperry simulation, showing total values of the various parameters. These runs include the initial values of velocity and acceleration and are a more complete verification of the entire vehicle. They are examples of a portion of the standard set of check runs Sperry will issue periodically. Figures 28 through 39 compare results.

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2. E. Capen (R.I.)

3. G. Carden (R.I.)

TABLE I

SYMBOLOGY USED FOR

SPERRY TIME HISTORIES

Symbol	Definition	Computation
ΔV	Change in Total Velocity	$V-V_{initial}$
$\Delta\alpha$	Change in Angle of Attack	$\alpha-\alpha_{initial}$
Δq	Change in Pitch Rate	$q-q_{initial}$
Δh	Change in CG Altitude	$h-h_{initial}$
$\Delta\beta$	Change in Sideslip	$\beta-\beta_{initial}$
Δp	Change in Roll Rate	$p-p_{initial}$
Δr	Change in Yaw Rate	$r-r_{initial}$
ΔAy	Change in Lateral Acceleration	$Ay-Ay_{initial}$

TABLE II

SPERRY INITIAL TRIM CONDITIONS

Altitude h_{CG} (Ft)	Dynamic Pressure $\bar{q}$ (psf)	α (deg)	δ_e (deg)	θ (deg)	$\dot{V}$ (fps^2)	V (fps)	Mach
10K	95	11.78	-1.68	-12.28	4.9	333.8	.31
	375	3.17	.03	-20.87	-7.2	657	.61
5K	95	12.15	-1.77	-11.91	4.94	304.8	.28
	375	3.17	.02	-20.89	-6.75	605.4	.55
1K	95	12.58	-0.47	- 2.42	-0.39	286.9	.26
	375	3.54	0.66	-11.45	-12,3	569.6	.51
25	95	10.11	4.05	10.14	- 8.23	282.3	.25
	375	2.17	6.35	2.18	-26.09	561.4	.5

Alt	δ_{BF}°	δ_{SB}°	GEAR
10K	22.5	30	UP
5K	22.5	30	UP
1K	10	0	DOWN
25	-11.7	0	DOWN

TABLE III

ROCKWELL INITIAL TRIM CONDITIONS

Altitude h (Ft)	Dynamic Pressure $\bar{q}$ (psf)	α (deg)	δ_e (deg)	θ (deg)	$\dot{V}$ (fps^2)	V (fps)	Mach
10K	95	12.0	-1.78	-12.0	5.34	329.0	.31
	375	3.25	0.15	-20.8	-7.08	653.6	.61
5K	95	12.0	-1.81	-12.0	5.37	304.6	.28
	375	3.25	.04	-20.75	-6.82	605.1	.55
1K	95	12.9	-0.58	- 2.1	-0.09	286.9	.26
	365	3.52	0.82	-11.5	12.8	570.0	.51
25	95	10.8	4.05	10.8	-8.19	282.8	.25
	375	2.38	6.10	2.38	-24.1	561.9	.5

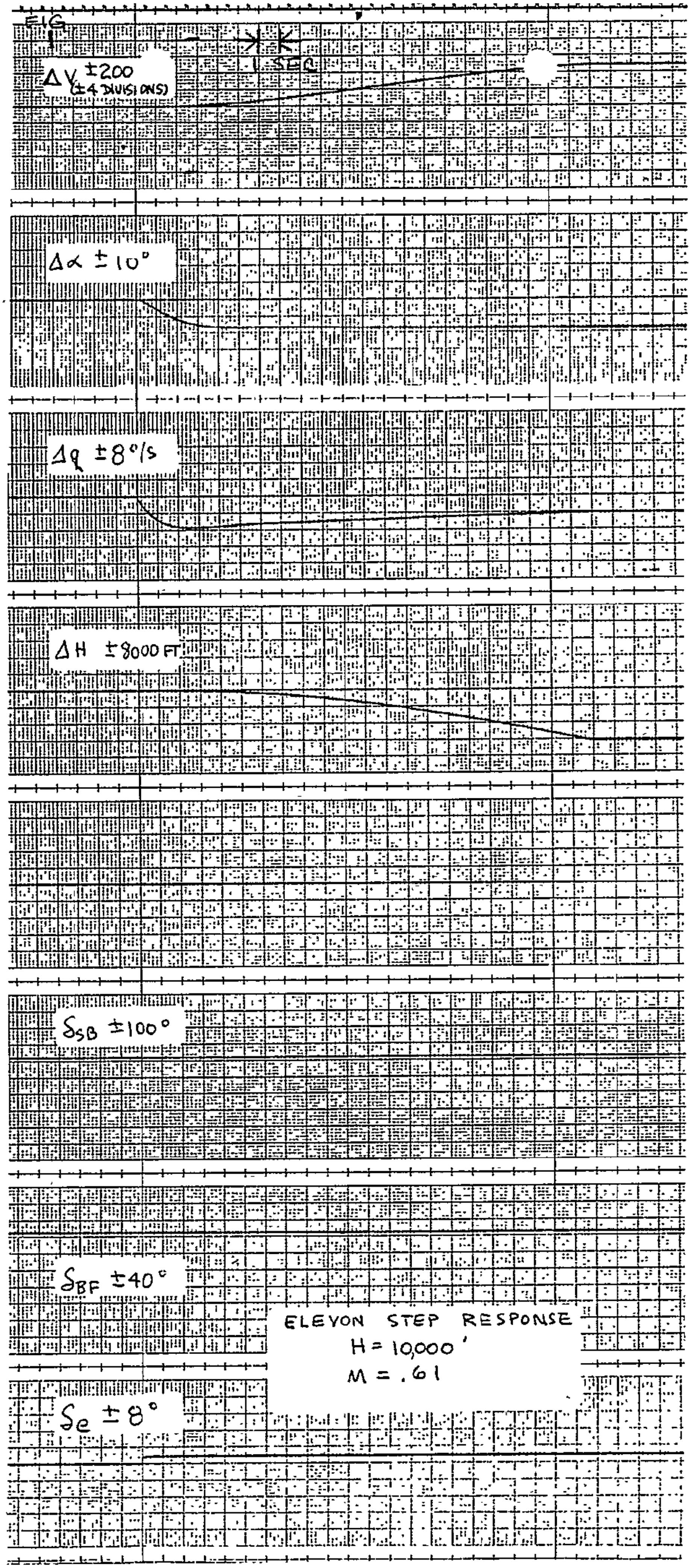
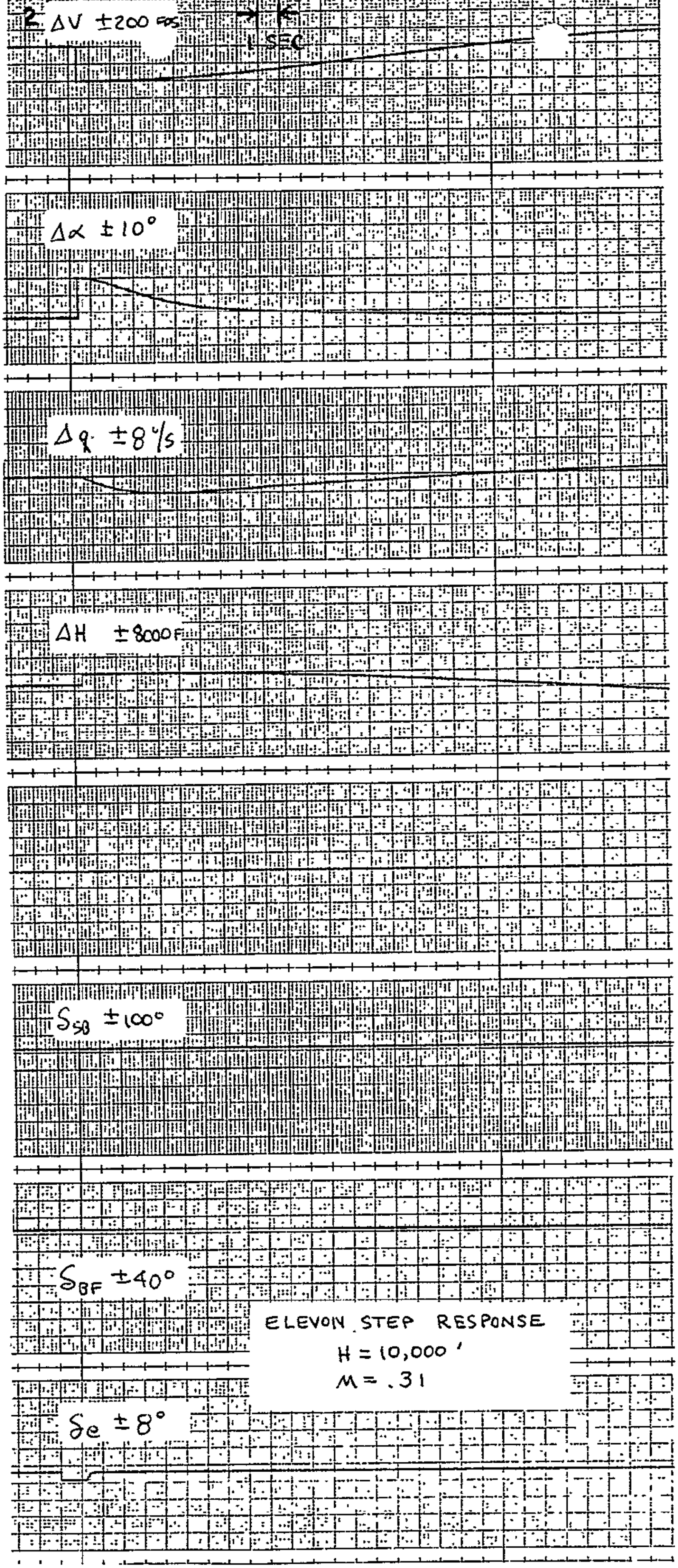

1 SEC
ΔV ±200 (±4 DIVISIONS)
Δα ±10°
Δq ±8°/s
ΔH ±8000 FT
δSB ±100°
δBF ±40°
δe ±8°
ELEVON STEP RESPONSE
H = 10,000'
M = .61

FIG. 2



ELEVON STEP RESPONSE
H = 10,000 '
M = .31

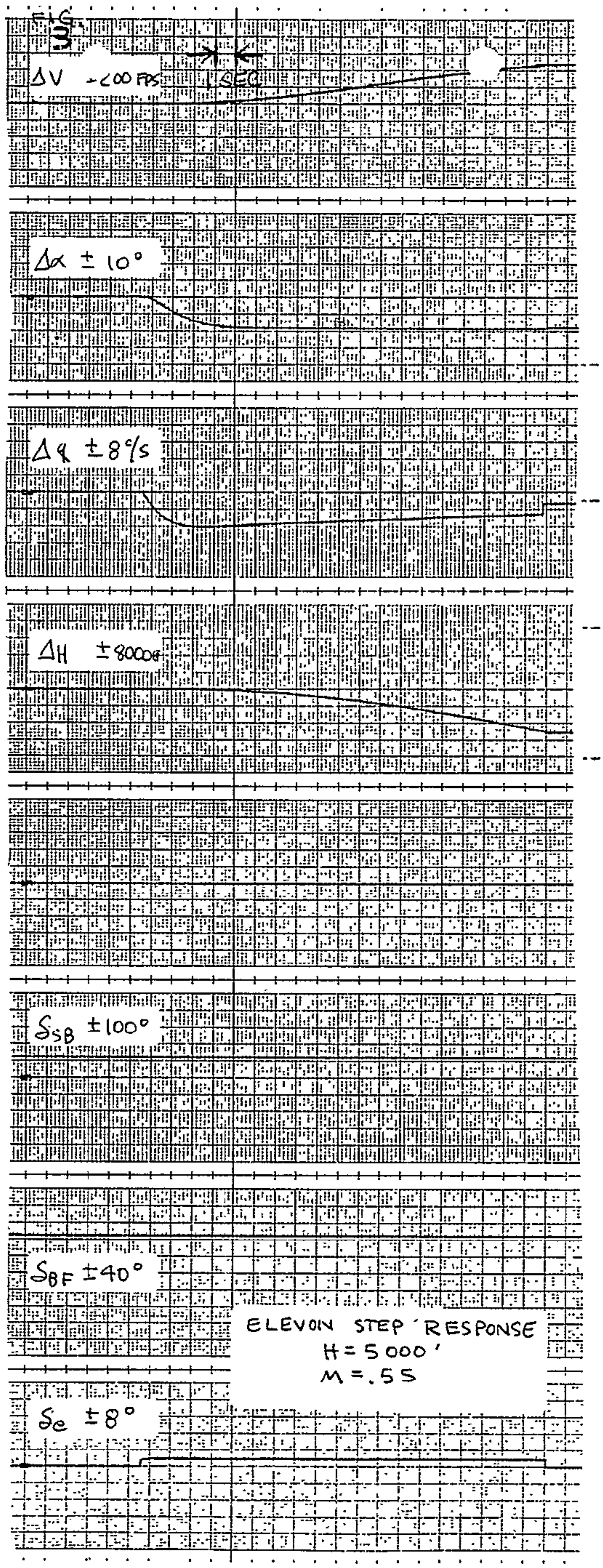


FIG. 3
ΔV ±200 FPS
1 SEC
Δα ±10°
Δq ±8°/s
ΔH ±8000 ft
δSB ±100°
δBF ±40°
ELEVON STEP RESPONSE
H = 5000'
M = .55
δe ±8°

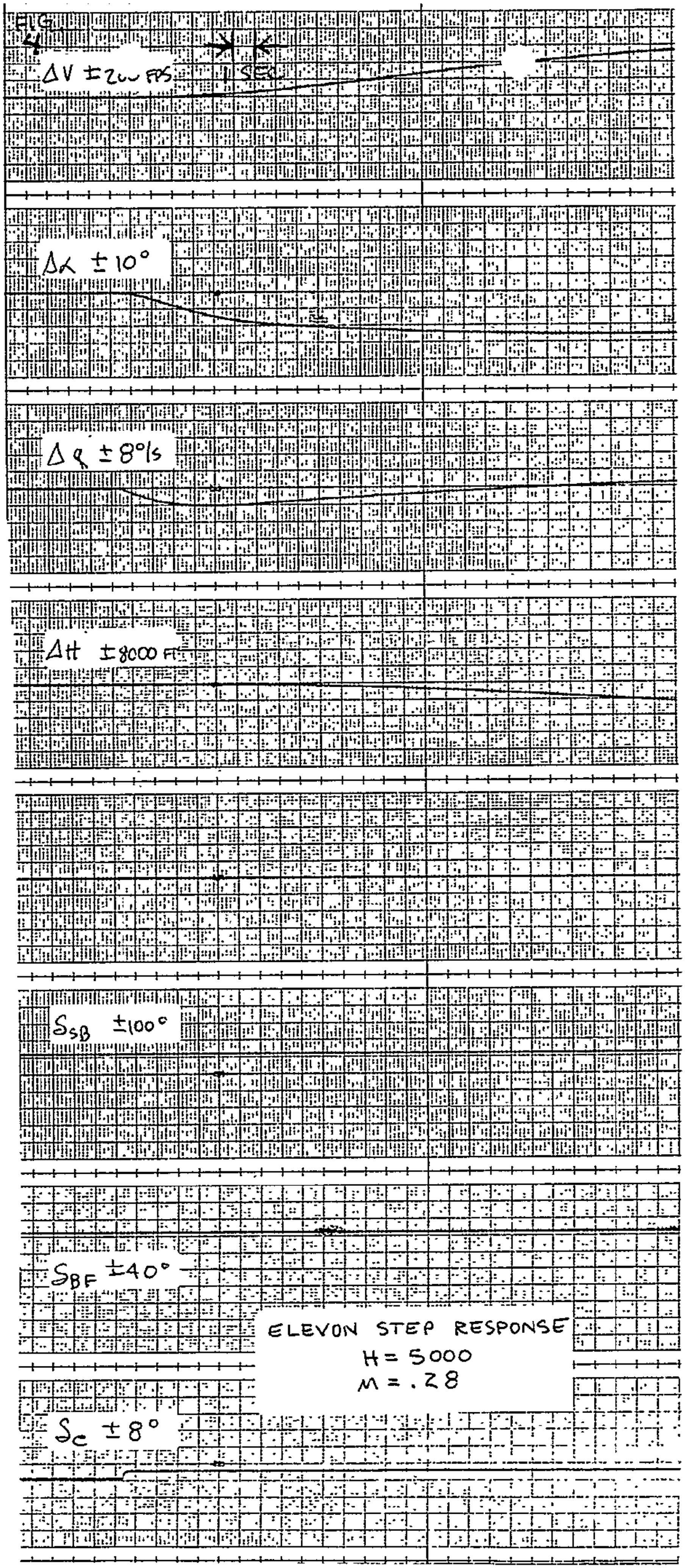


FIG. 4
ΔV ±200 FPS
1 SEC
Δα ±10°
Δq ±8°/s
ΔH ±8000 FT
δ_{SB} ±100°
δ_{BF} ±40°
δ_C ±8°
ELEVON STEP RESPONSE
H = 5000
M = .28

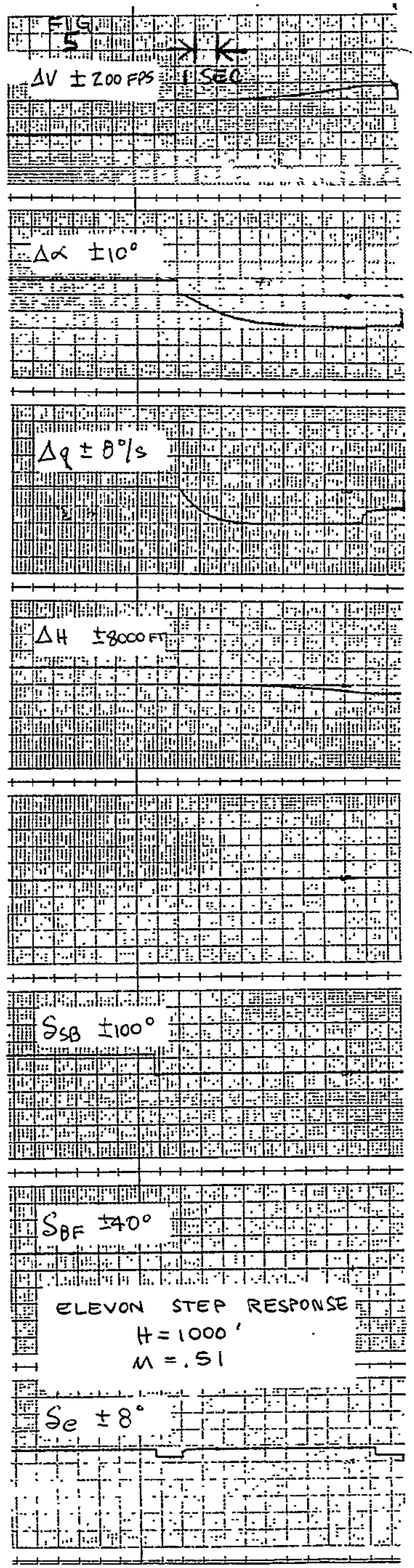


FIG 5
1 SEC
ΔV ±200 FPS
Δα ±10°
Δq ±8°/s
ΔH ±8000 FT
δSB ±100°
δBF ±40°
ELEVON STEP RESPONSE
H = 1000'
M = .51
δe ±8°

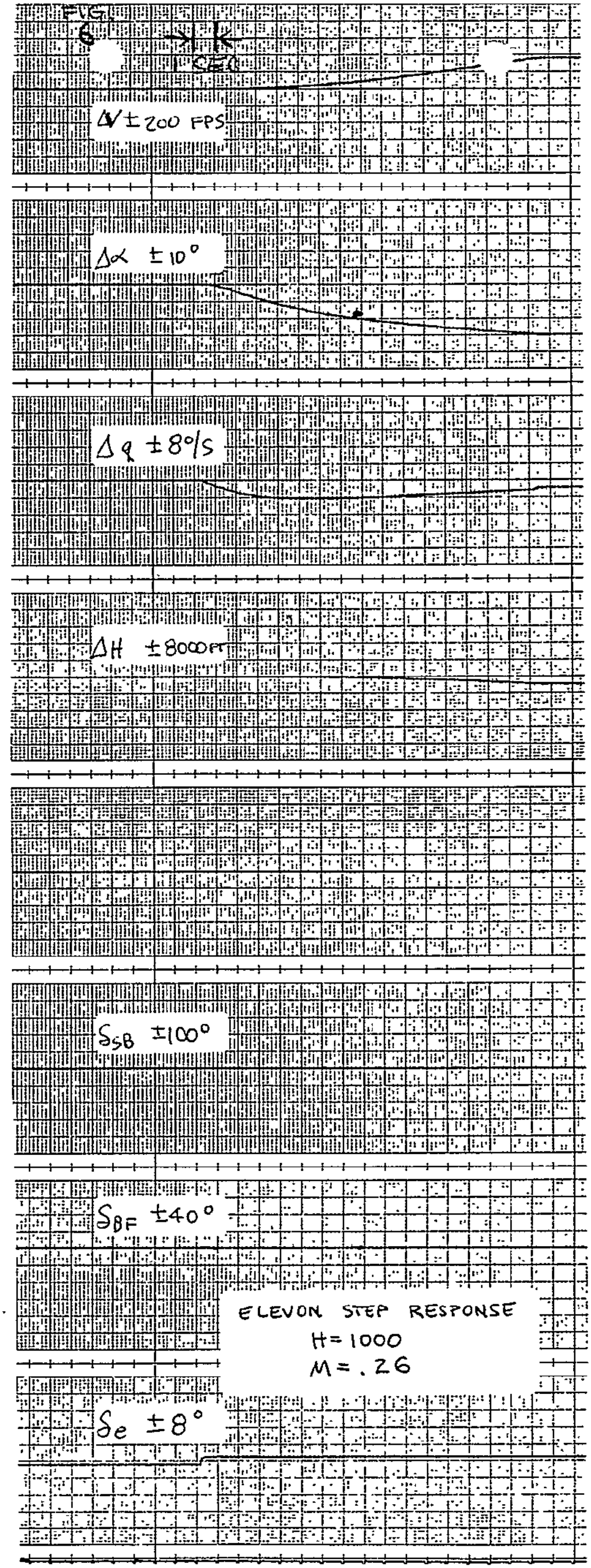
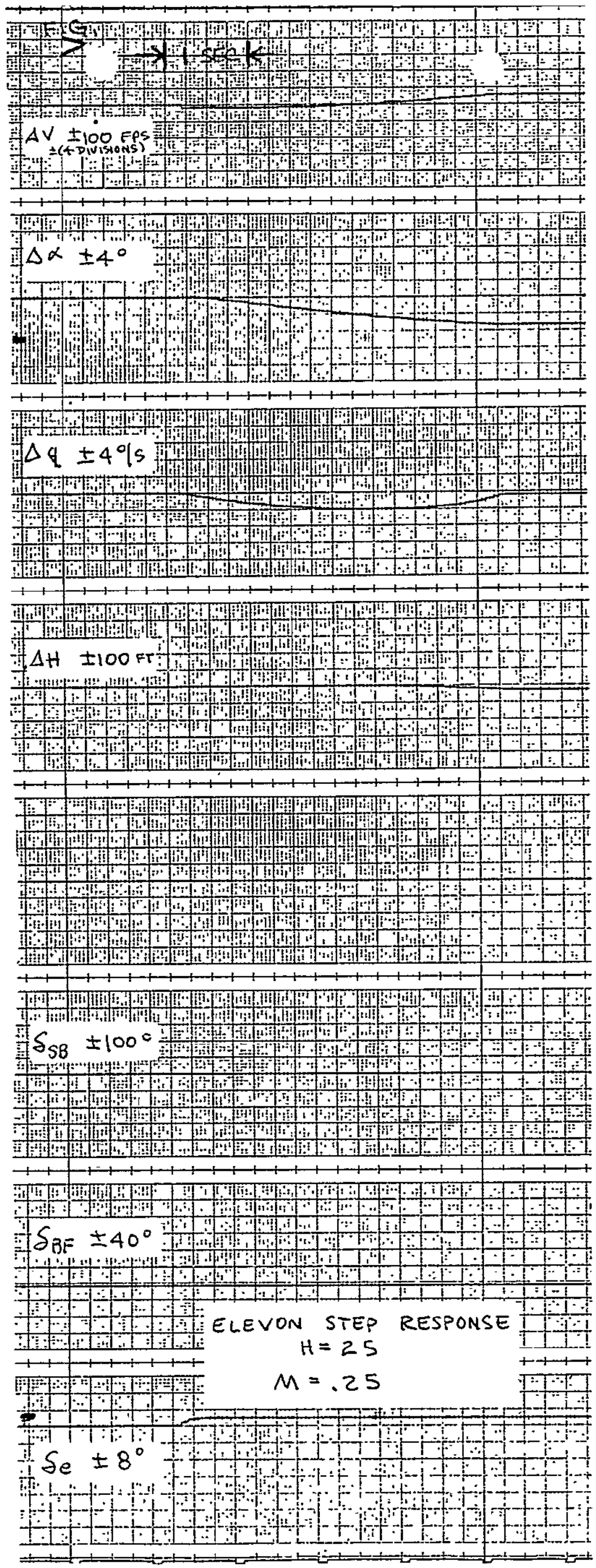


FIG 6
1 SEC
ΔV ±200 FPS
Δα ±10°
Δq ±8°/s
ΔH ±8000FT
δSB ±100°
δBF ±40°
ELEVON STEP RESPONSE
H=1000
M=.26
δe ±8°



ΔV ±100 FPS
±(4 DIVISIONS)
Δα ±4°
Δq ±4°/s
ΔH ±100 FT
δSB ±100°
δBF ±40°
ELEVON STEP RESPONSE
H = 25
M = .25
δe ±8°

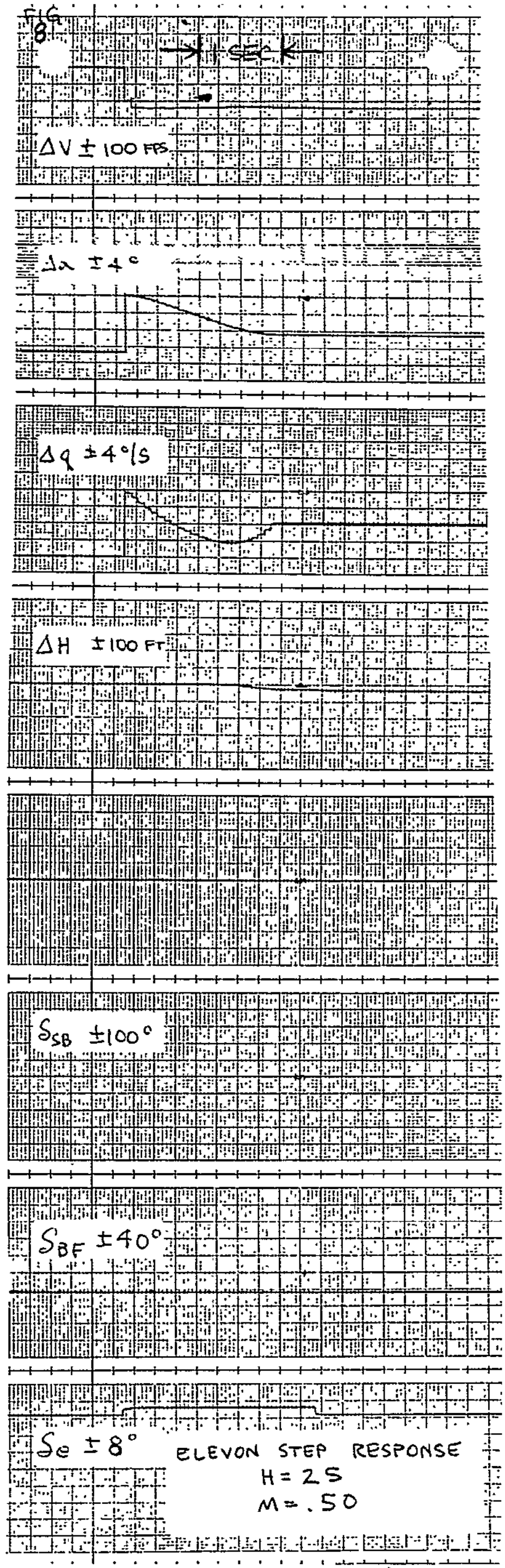


FIG 8
→| 1 SEC |←
ΔV ± 100 FPS
Δα ± 4°
Δq ± 4°/S
ΔH ± 100 FT
δSB ± 100°
δBF ± 40°
δe ± 8°
ELEVON STEP RESPONSE
H = 25
M = .50

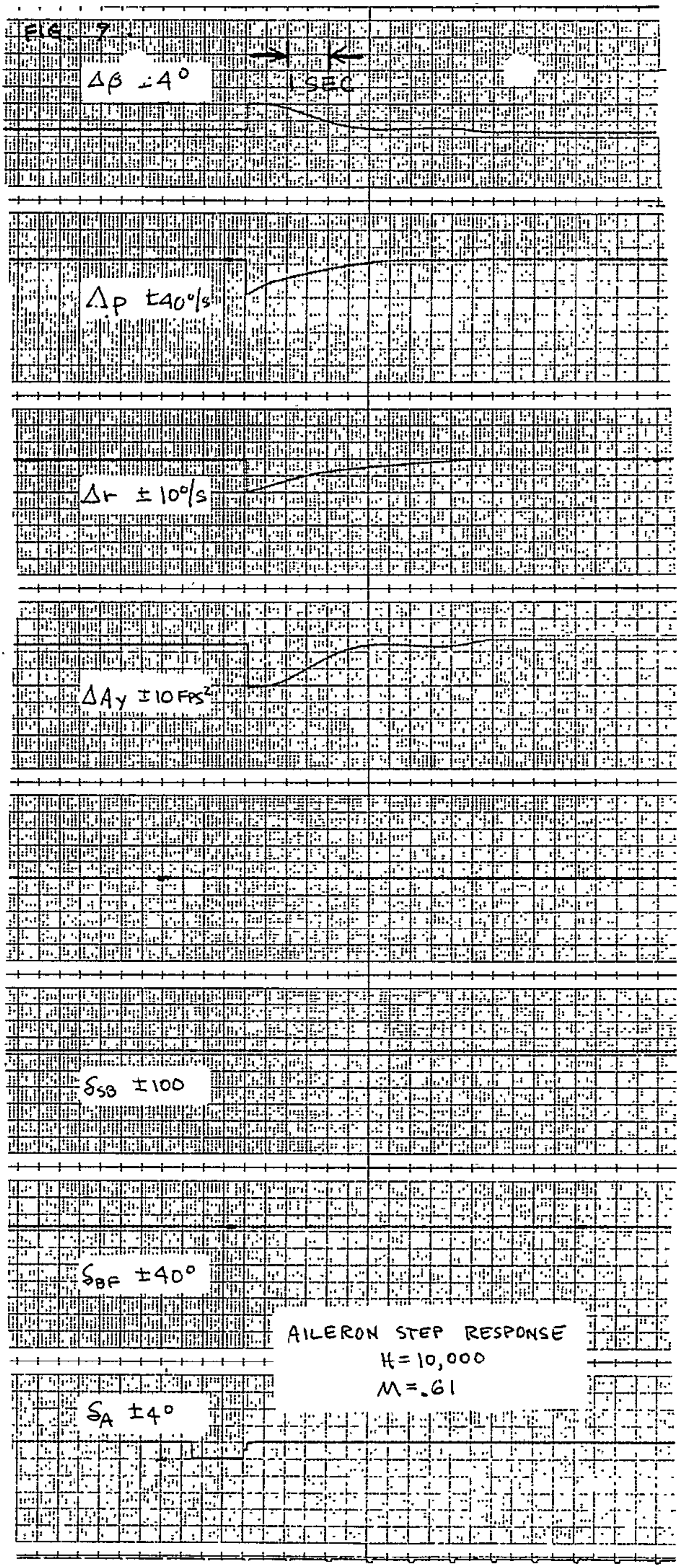


FIG. 7
1 SEC
Δβ ±4°
Δp ±40°/s
Δr ±10°/s
ΔAy ±10 FPS²
δSB ±100
δBF ±40°
δA ±4°
AILERON STEP RESPONSE
H = 10,000
M = .61

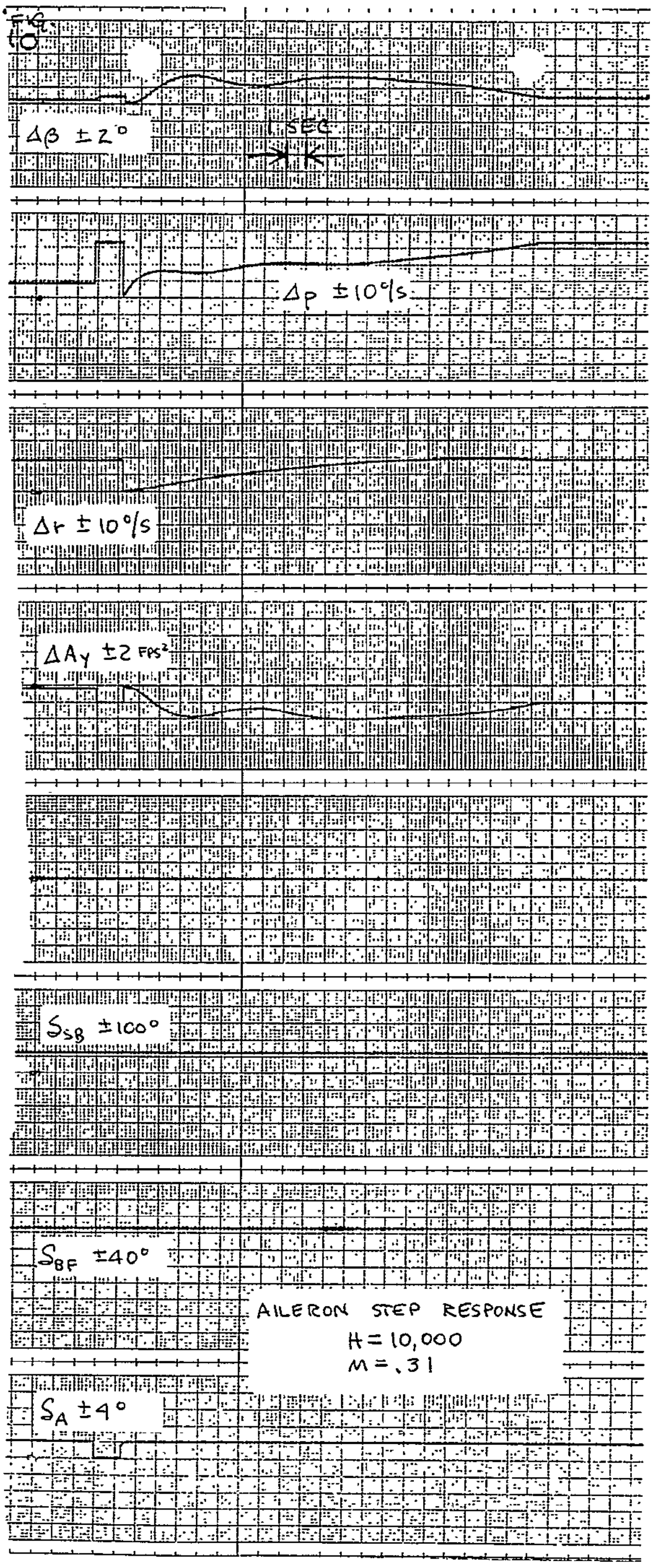


FIG. 10
Δβ ±2°
1 SEC
Δp ±10°/s
Δr ±10°/s
ΔAy ±2 FPS²
δSB ±100°
δBF ±40°
δA ±4°
AILERON STEP RESPONSE
H = 10,000
M = .31

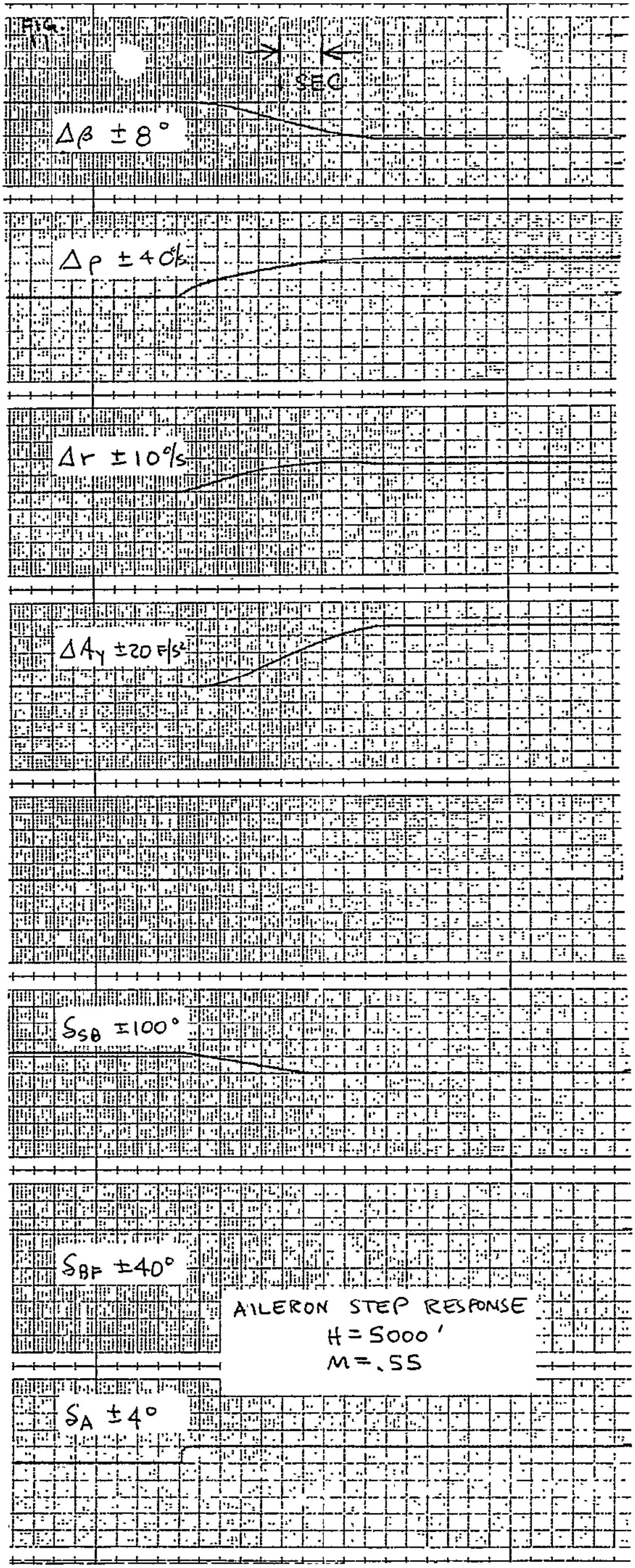


FIG.
1 SEC
Δβ ±8°
Δp ±40°/s
Δr ±10°/s
ΔAy ±20 F/s²
δSB ±100°
δBF ±40°
AILERON STEP RESPONSE
H = 5000'
M = .55
δA ±4°

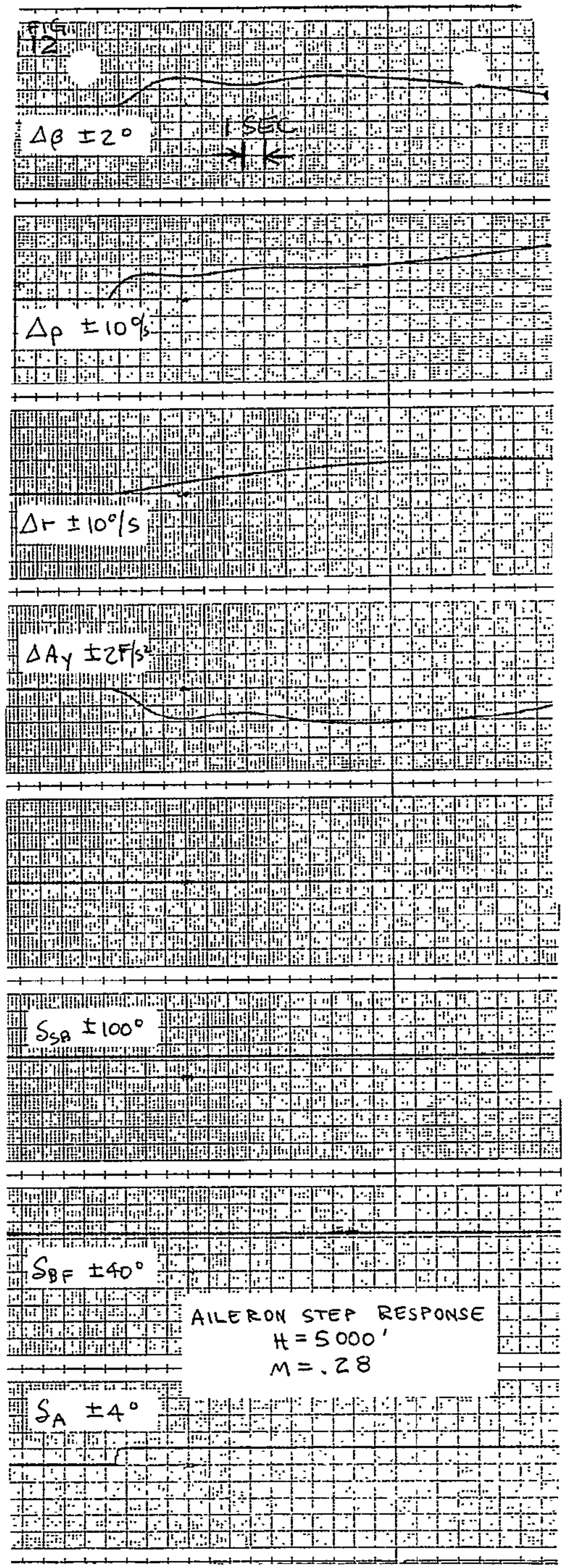


FIG 12
Δβ ±2°
1 SEC
Δp ±10°/s
Δr ±10°/s
ΔAy ±2F/s²
δSB ±100°
δBF ±40°
AILERON STEP RESPONSE
H = 5000'
M = .28
δA ±4°

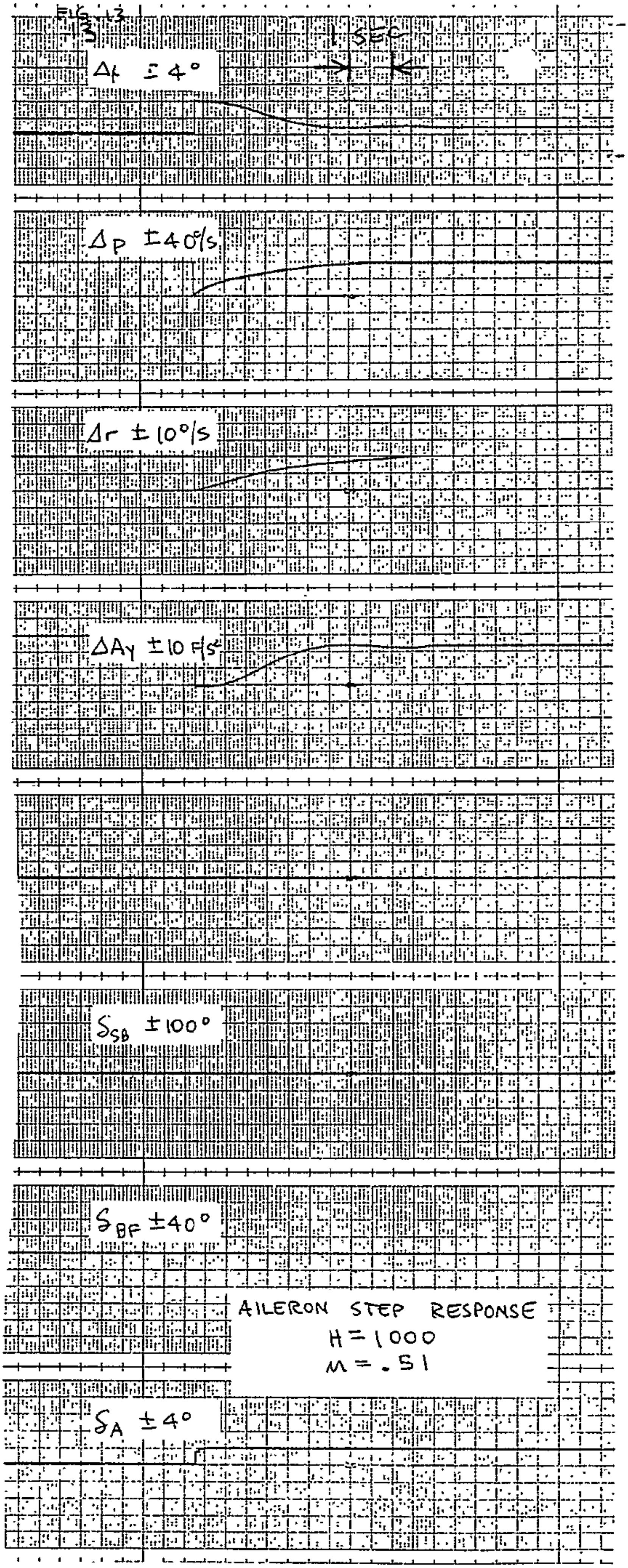


FIG. 13
1 SEC
Δφ ±4°
Δp ±40°/s
Δr ±10°/s
ΔAy ±10 F/s²
δSB ±100°
δBF ±40°
AILERON STEP RESPONSE
H = 1000
M = .51
δA ±4°

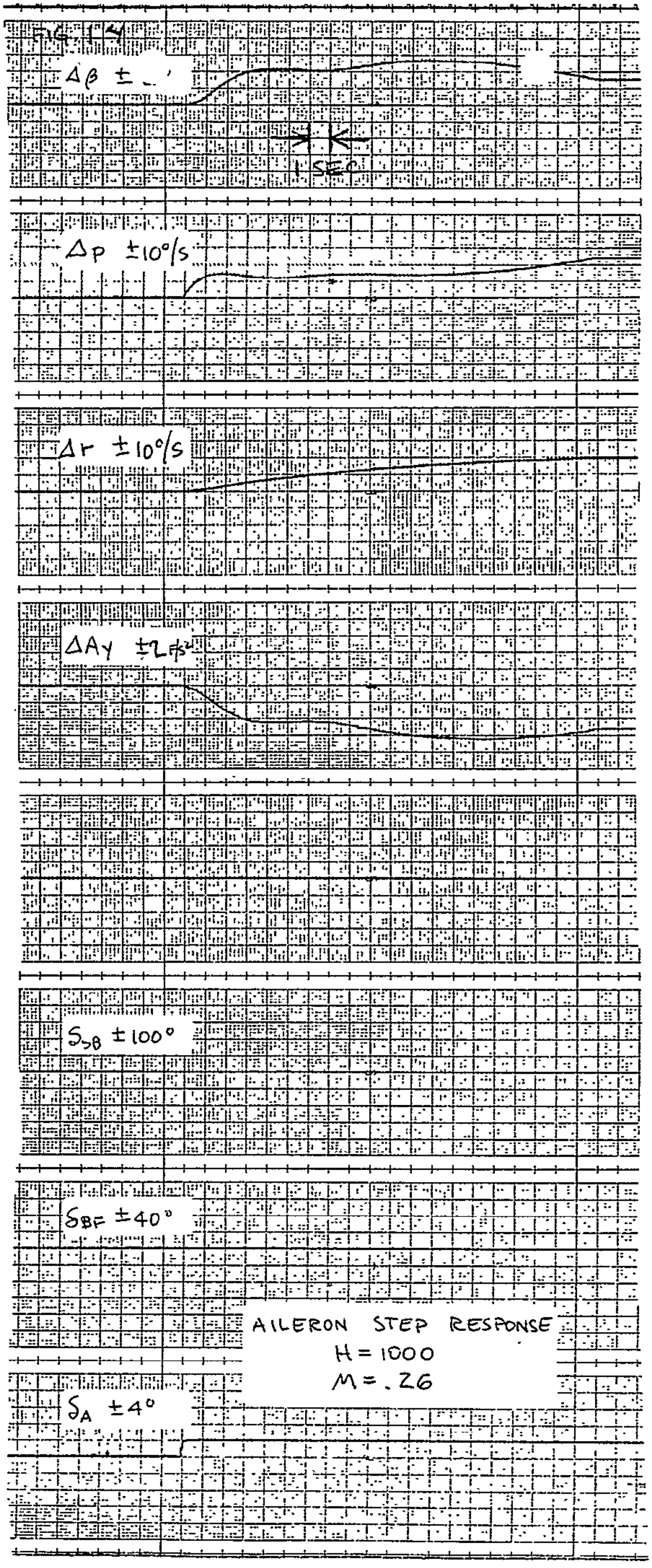


FIG. 14
Δβ ± _ '
1 SEC
Δp ±10°/s
Δr ±10°/s
ΔAy ±2 ft/s²
δSB ±100°
δBF ±40°
AILERON STEP RESPONSE
H = 1000
M = .26
δA ±4°

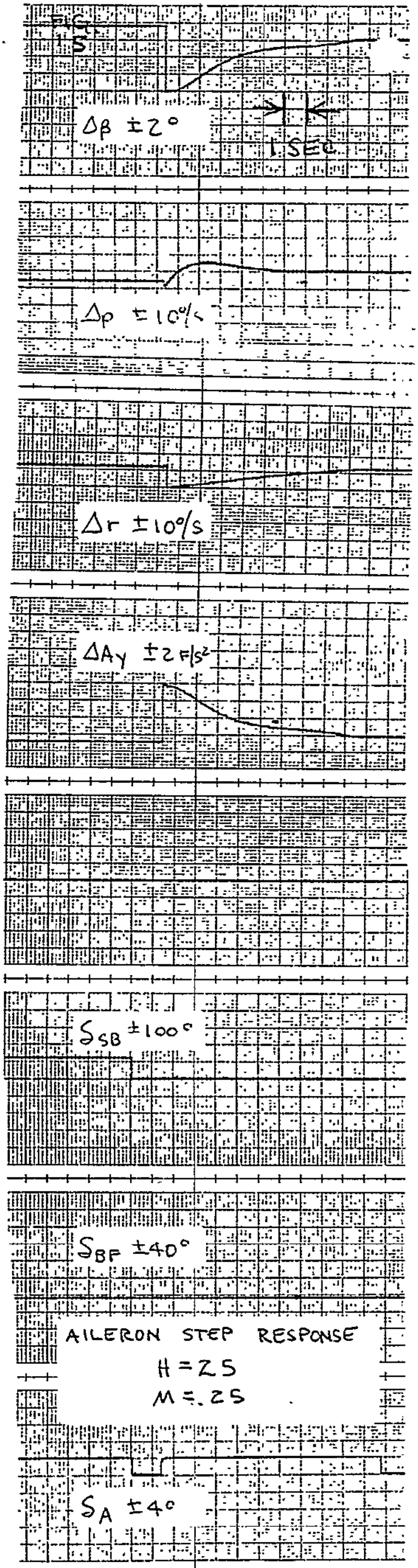


FIG 5
$\Delta\beta$ ±2°
1 SEC
Δp ±10°/s
Δr ±10°/s
ΔA_y ±2 F/s²
δ_{SB} ±100°
δ_{BF} ±40°
AILERON STEP RESPONSE
H = 25
M = .25
δ_A ±4°

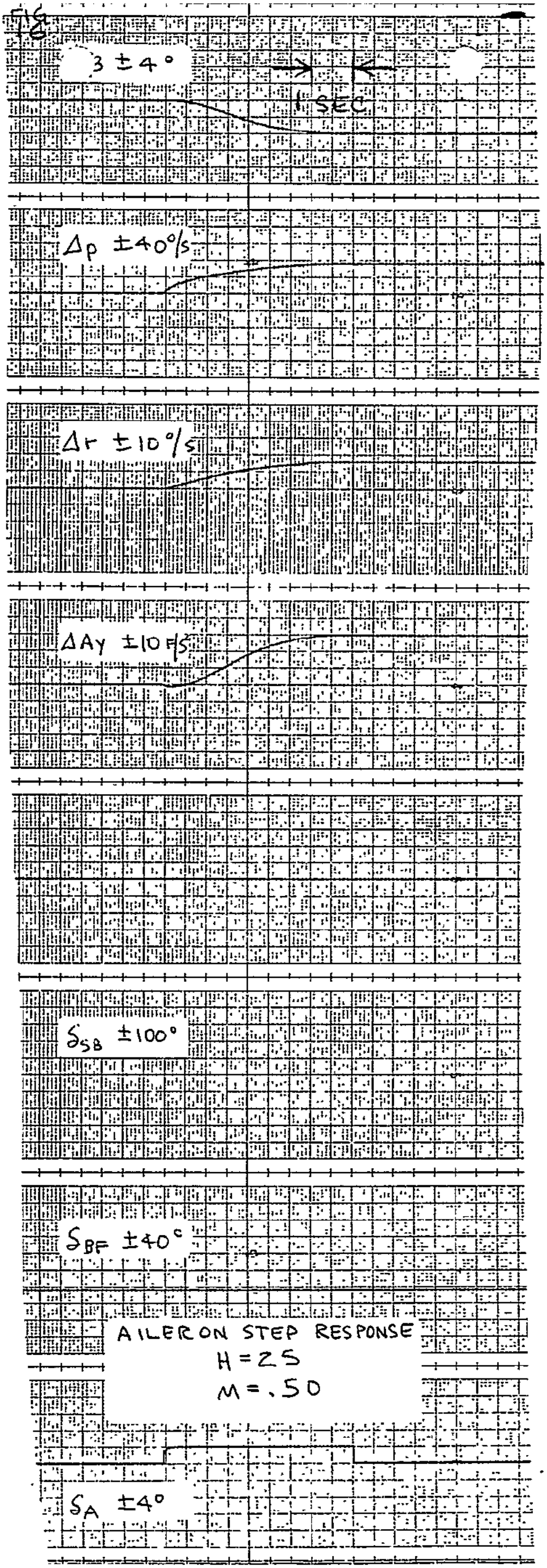


FIG 6
3 ±4°
1 SEC
Δp ±40°/s
Δr ±10°/s
ΔAy ±10 F/S
δSB ±100°
δBF ±40°
AILERON STEP RESPONSE
H = 25
M = .50
δA ±4°

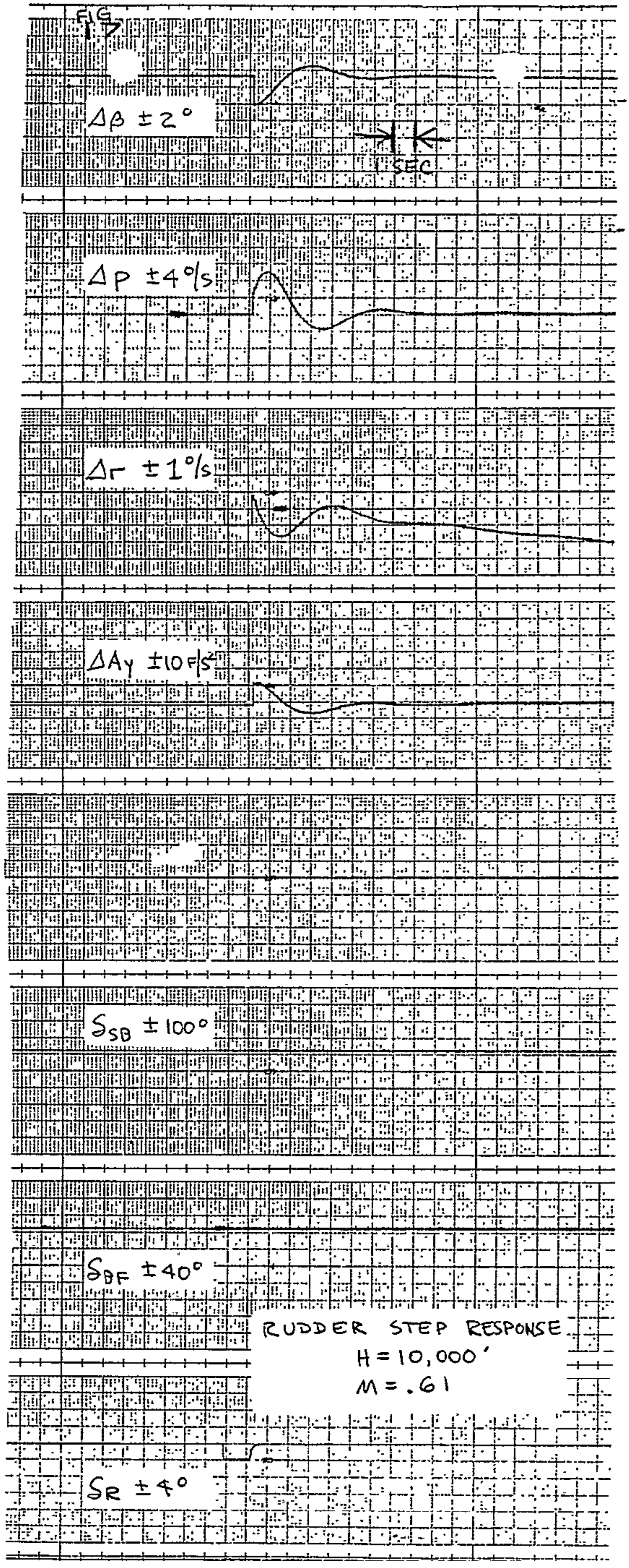
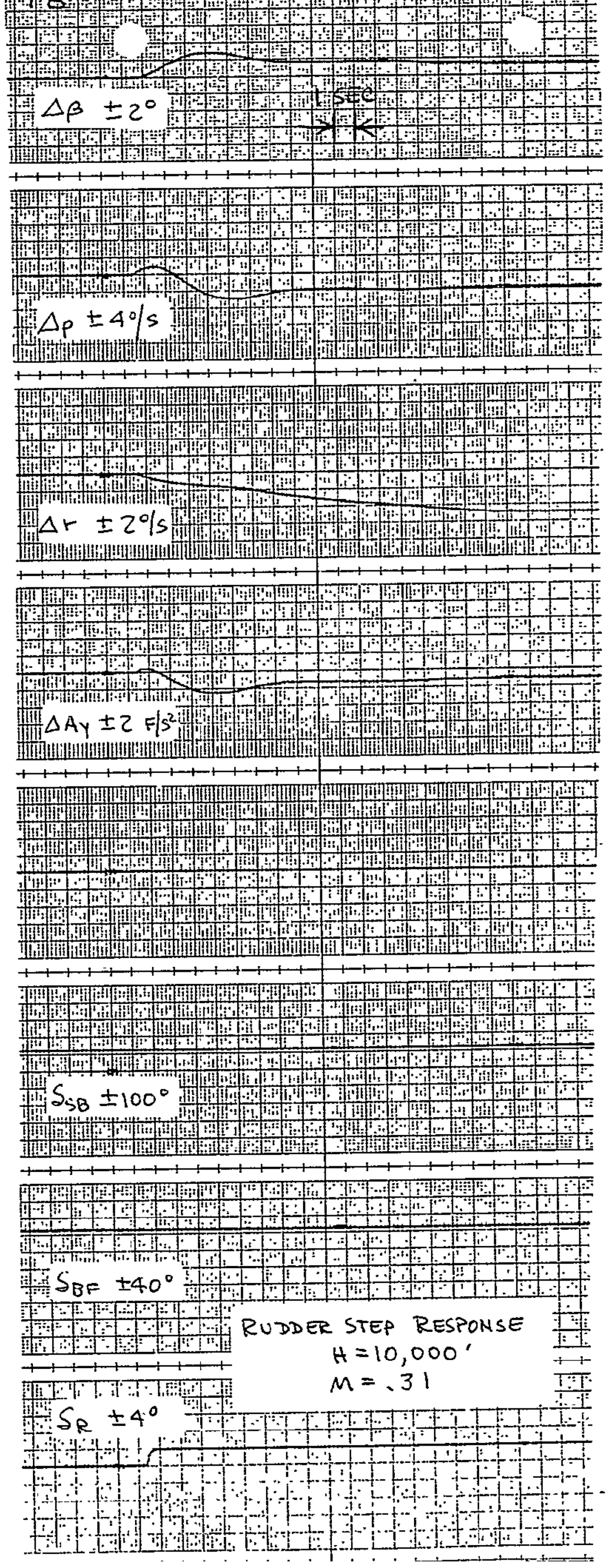


FIG 17
Δβ ±2°
1 SEC
Δp ±4°/s
Δr ±1°/s
ΔAy ±10 F/S²
δSB ±100°
δBF ±40°
RUDDER STEP RESPONSE
H = 10,000′
M = .61
δR ±4°

FIG. 18



Δβ ±2°
1 SEC
Δp ±4°/s
Δr ±2°/s
ΔAY ±2 F/S²
δSB ±100°
δBF ±40°
RUDDER STEP RESPONSE
H=10,000'
M=.31
δR ±4°

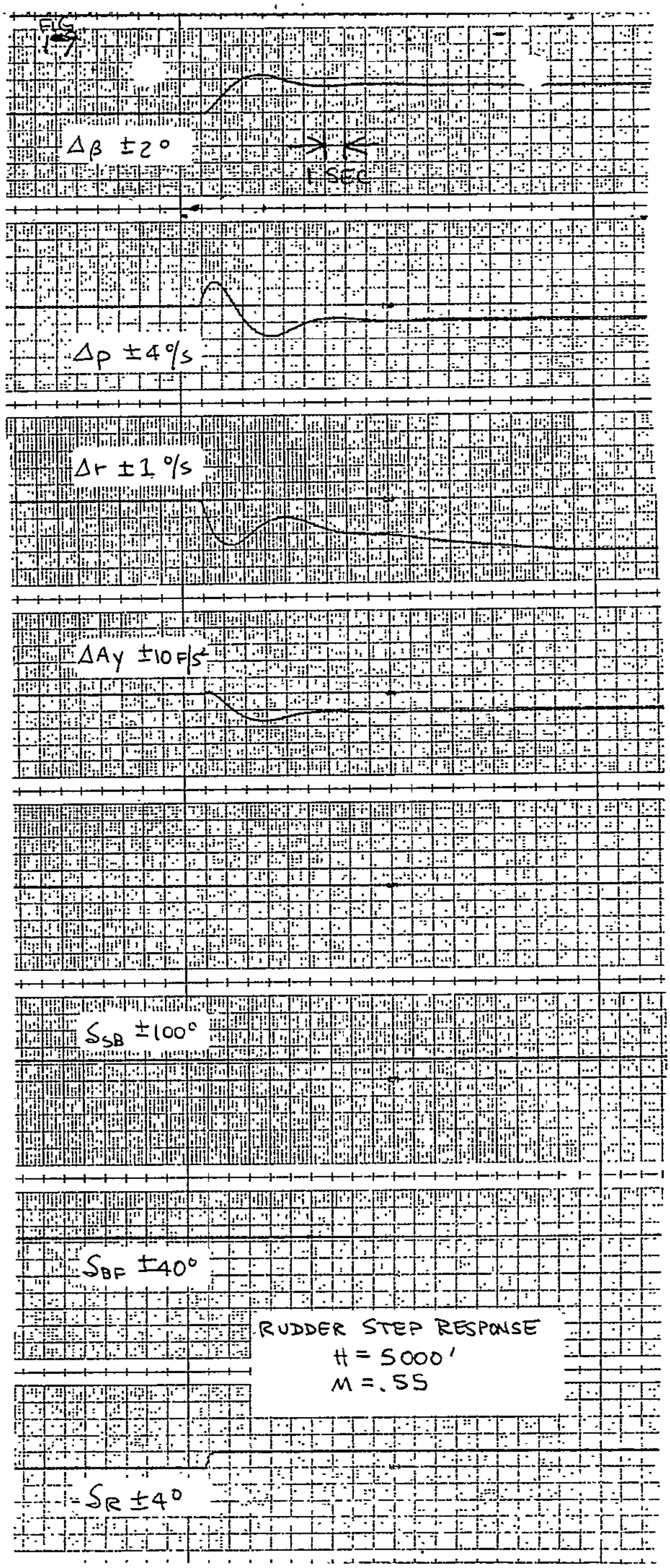


FIG
Δβ ±2°
1 SEC
Δp ±4°/s
Δr ±1 °/s
ΔAy ±10 F/s²
δSB ±100°
δBF ±40°
RUDDER STEP RESPONSE
H = 5000'
M = .55
δR ±4°

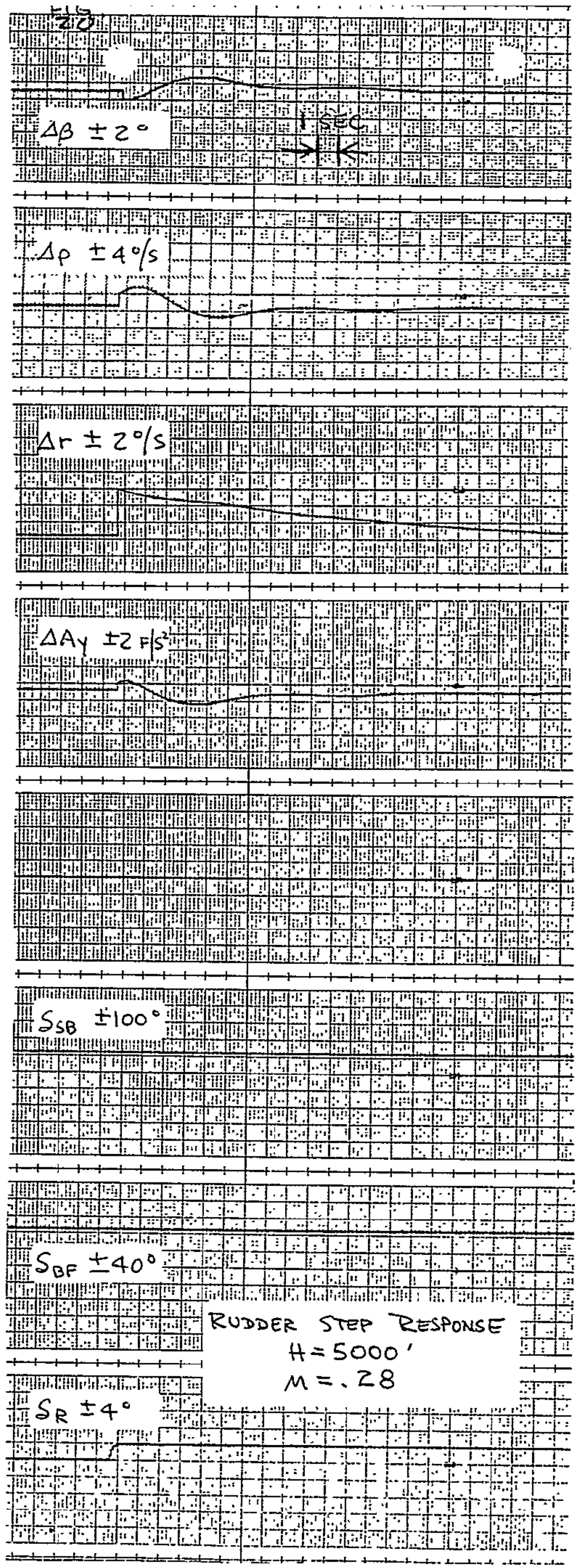


FIG 2.17
Δβ ±2°
1 SEC
Δp ±4°/s
Δr ±2°/s
ΔAy ±2 F/S²
δSB ±100°
δBF ±40°
RUDDER STEP RESPONSE
H = 5000'
M = .28
δR ±4°

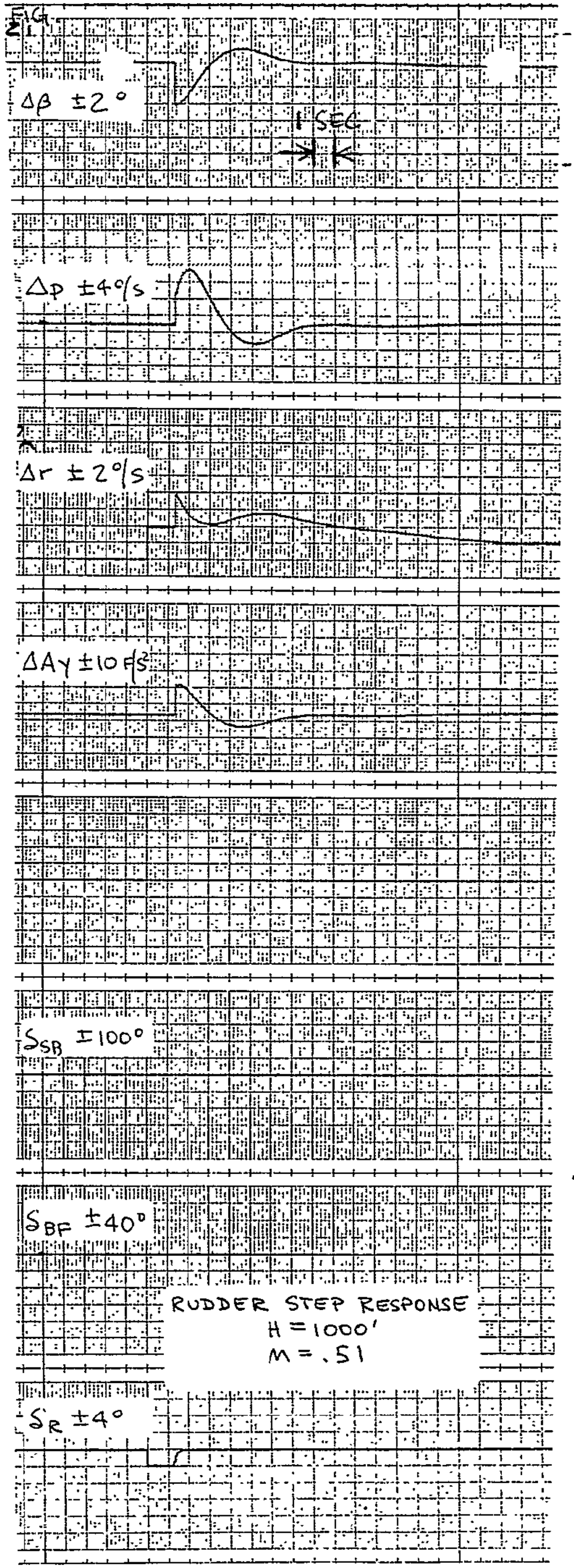


FIG.
Δβ ±2°
1 SEC
Δp ±4°/s
Δr ±2°/s
ΔAy ±10 F/S²
δSB ±100°
δBF ±40°
RUDDER STEP RESPONSE
H = 1000'
M = .51
δR ±4°

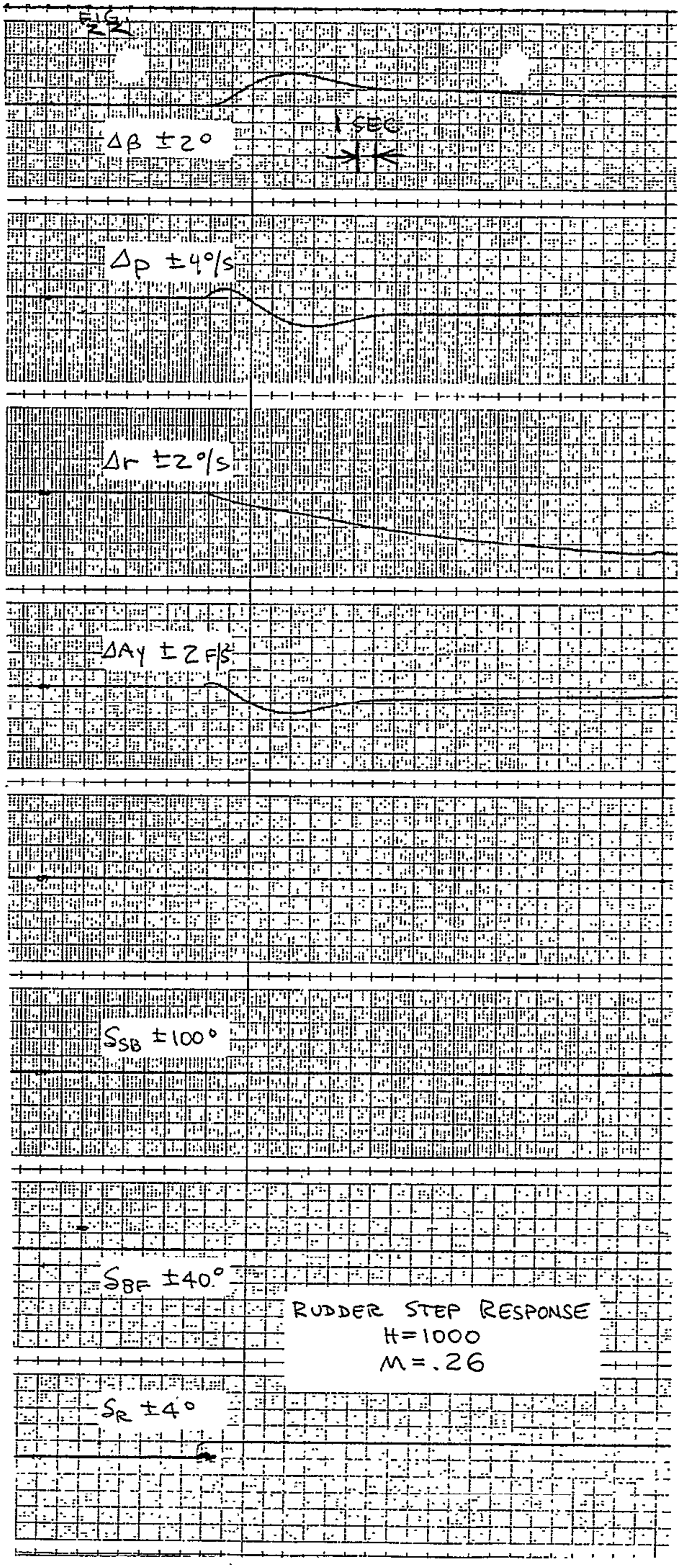


FIG. 22
Δβ ±2°
1 SEC
Δp ±4°/s
Δr ±2°/s
ΔAy ±2 F/S²
δSB ±100°
δBF ±40°
RUDDER STEP RESPONSE
H=1000
M=.26
δR ±4°

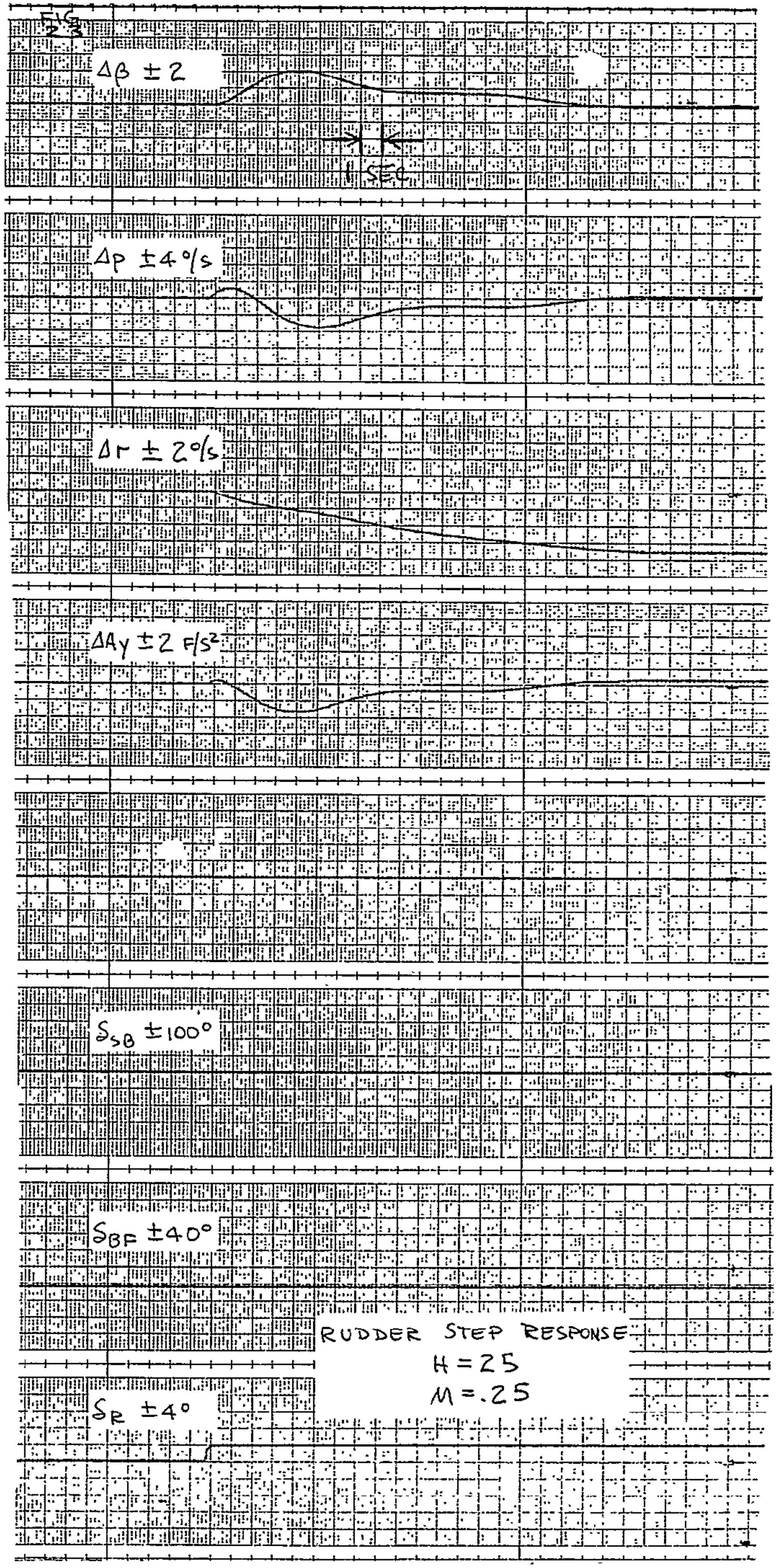


FIG 23
Δβ ±2
1 SEC
Δp ±4°/s
Δr ±2°/s
ΔAy ±2 F/S²
δSB ±100°
δBF ±40°
δR ±4°
RUDDER STEP RESPONSE
H = 25
M = .25

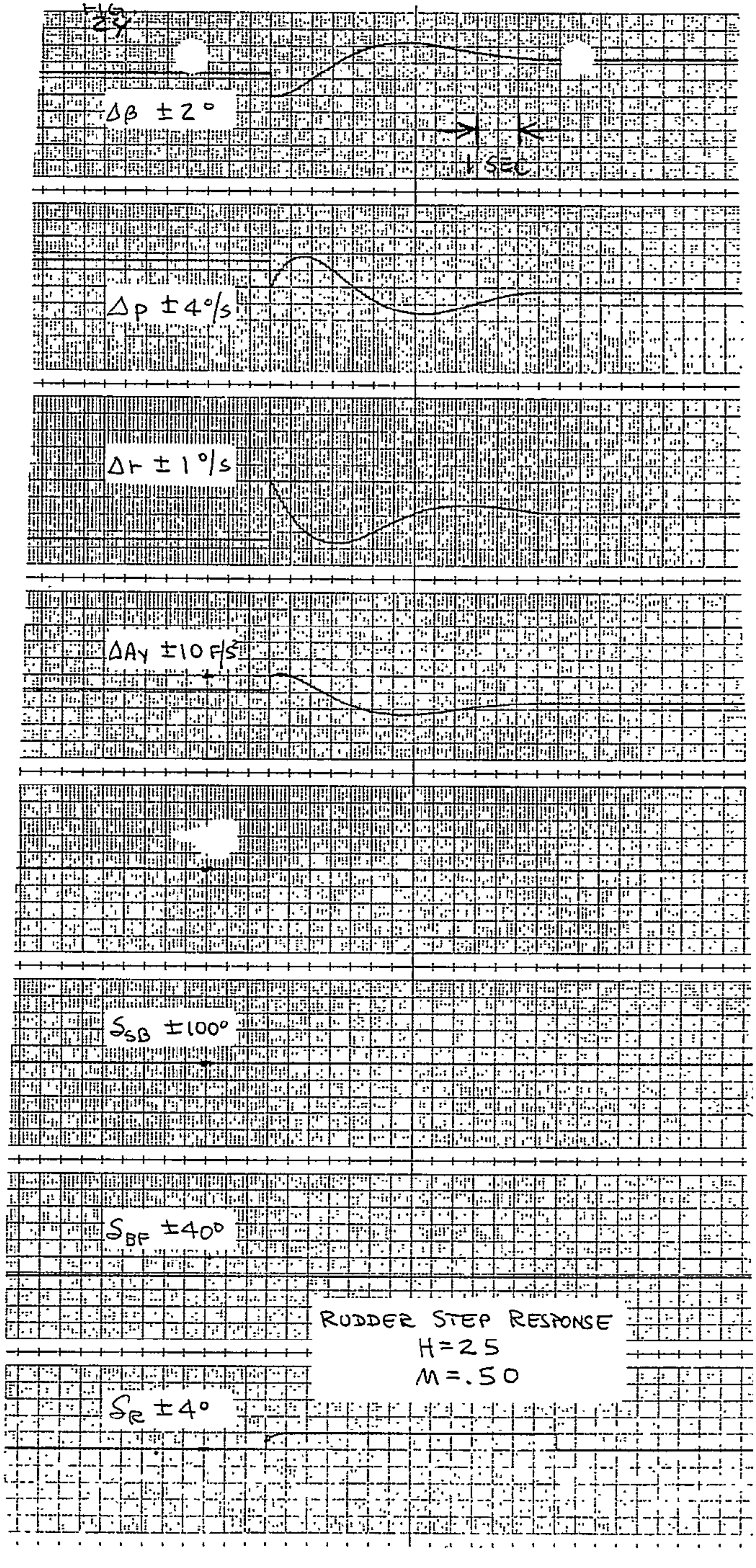
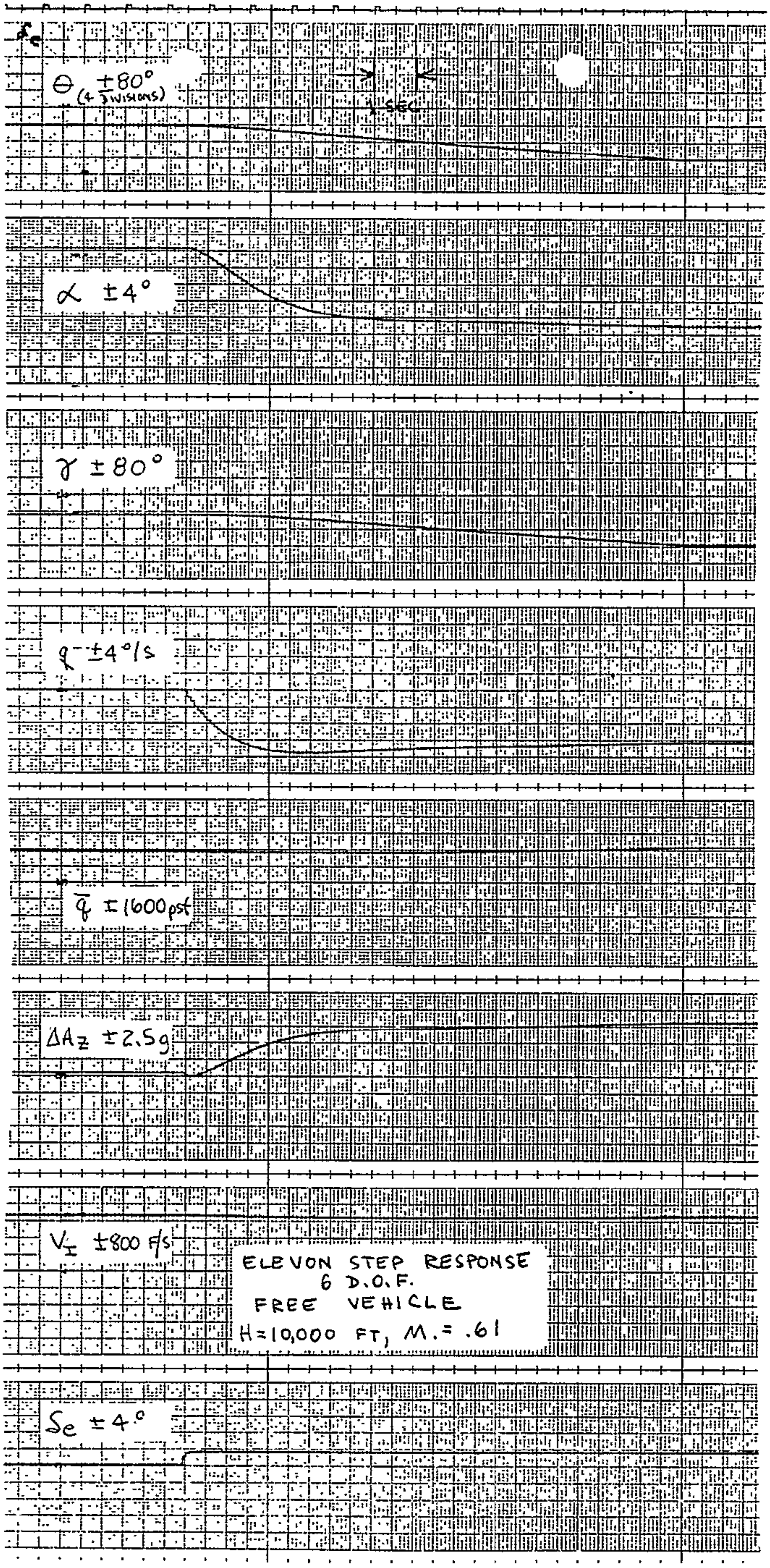
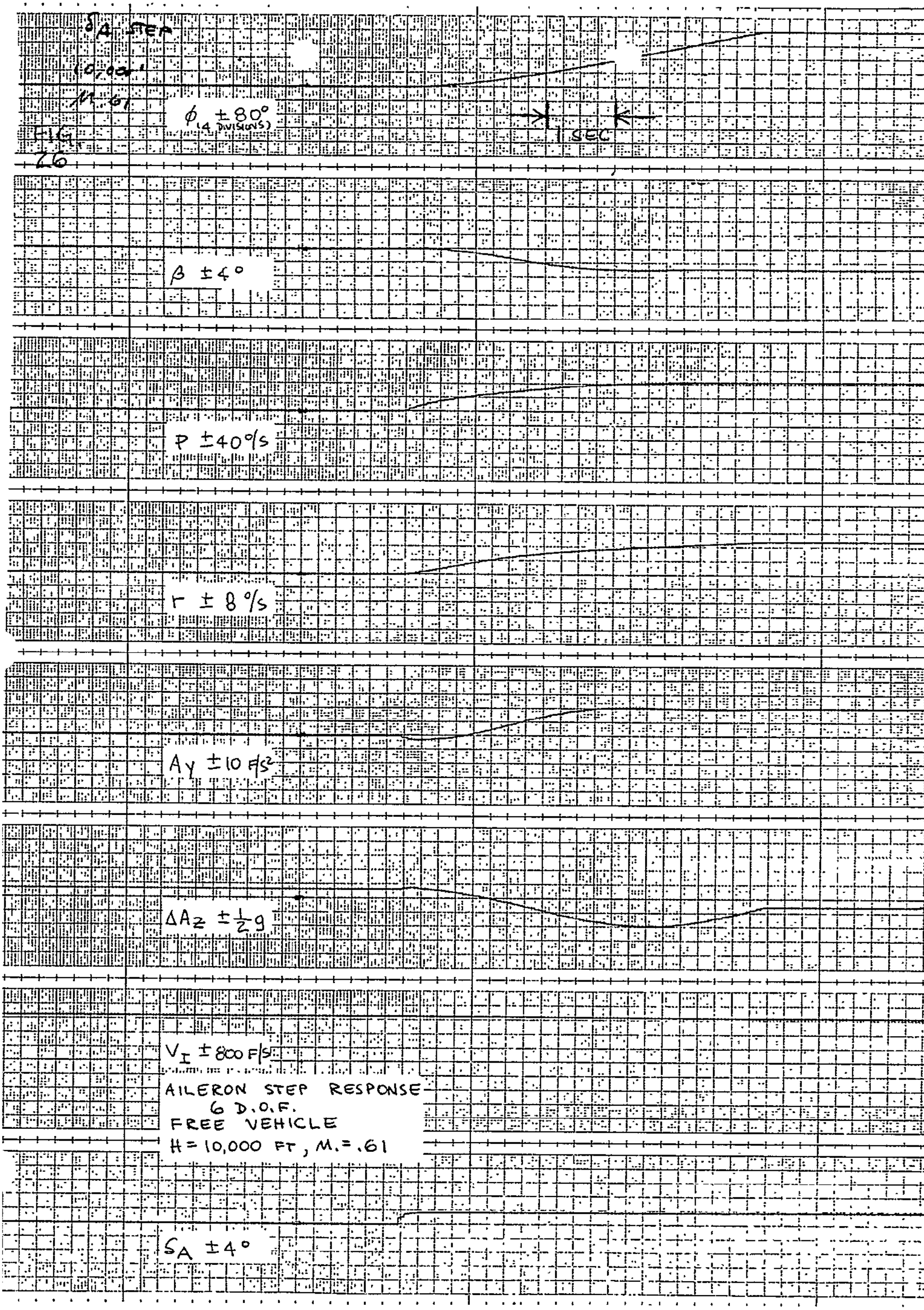


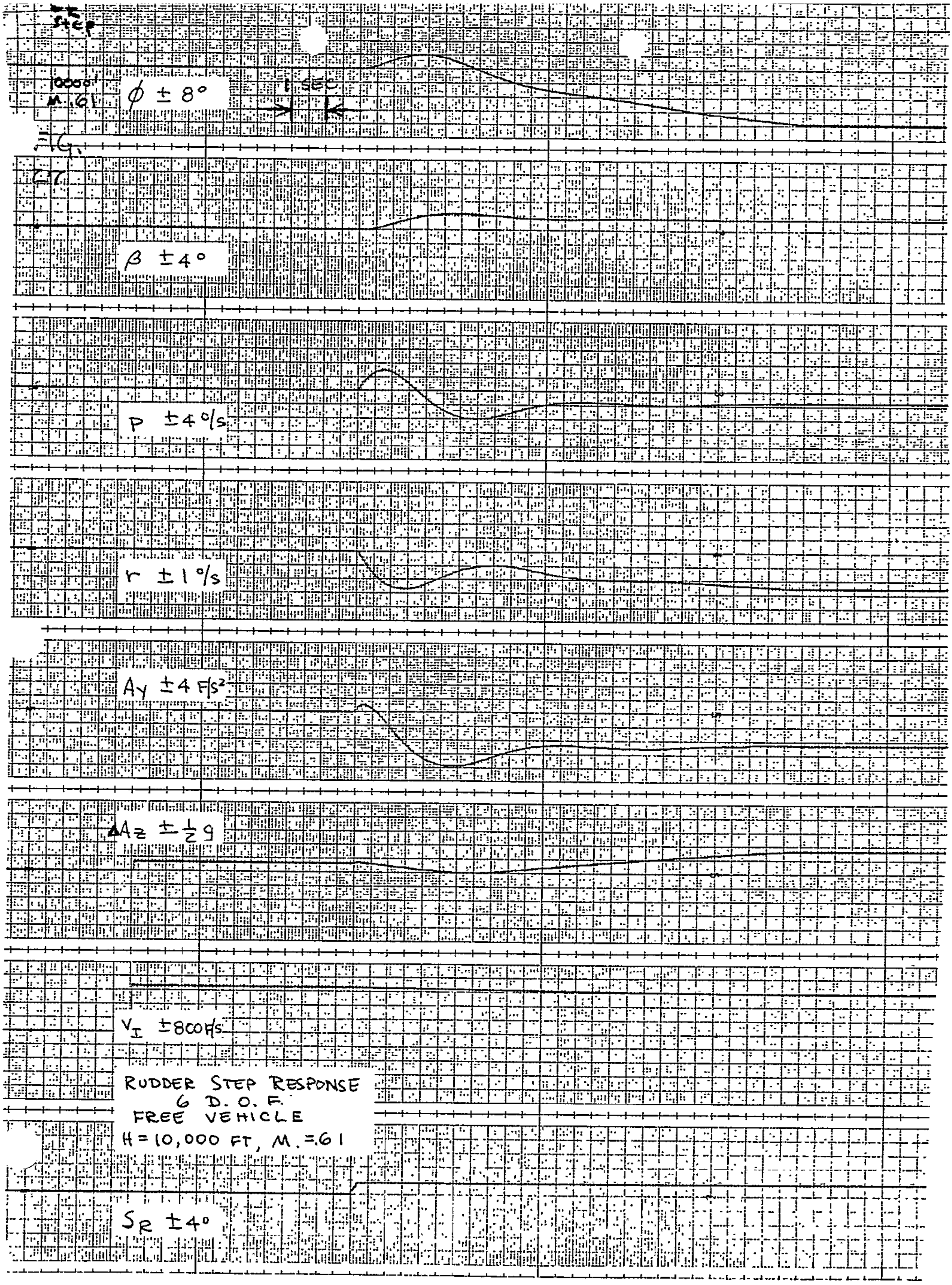
FIG. 24
Δβ ±2°
1 SEC
Δp ±4°/s
Δr ±1°/s
ΔAy ±10 F/S²
δSB ±100°
δBF ±40°
δR ±4°
RUDDER STEP RESPONSE
H=25
M=.50

FIG. 25



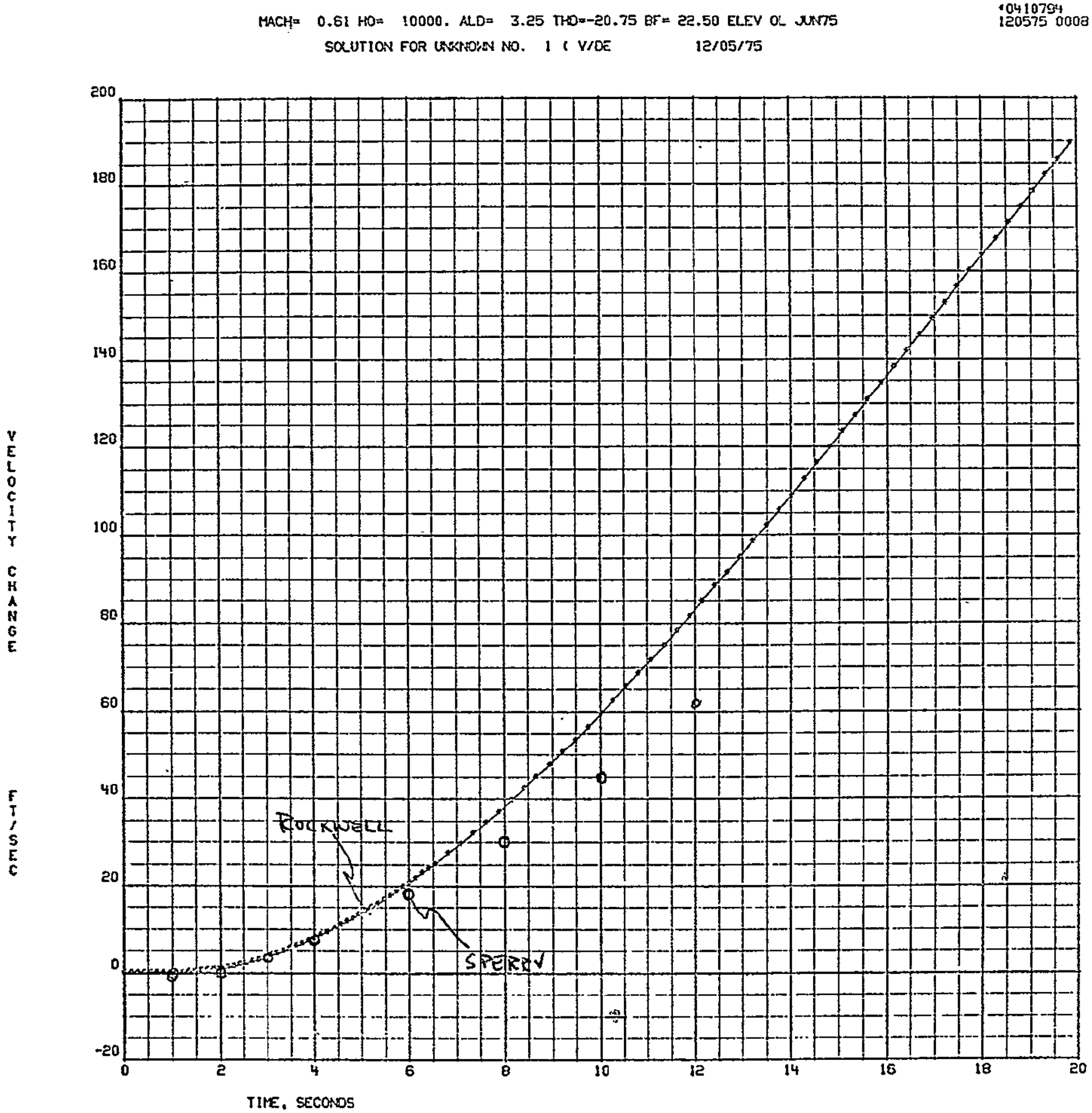
δ_e
Θ ±80° (4 DIVISIONS)
1 SEC
α ±4°
γ ±80°
q ±4°/s
$\bar{q}$ ±1600 psf
ΔA_z ±2.5g
V_I ±800 F/S
ELEVON STEP RESPONSE
6 D.O.F.
FREE VEHICLE
H=10,000 FT, M. = .61
δ_e ±4°


δA STEP
10,000'
M .61
FIG
26
φ ±80°
(4 DIVISIONS)
1 SEC
β ±4°
P ±40°/s
r ±8°/s
AY ±10 F/S²
ΔAZ ±½g
VI ±800 F/S
AILERON STEP RESPONSE
6 D.O.F.
FREE VEHICLE
H = 10,000 FT, M. = .61
δA ±4°



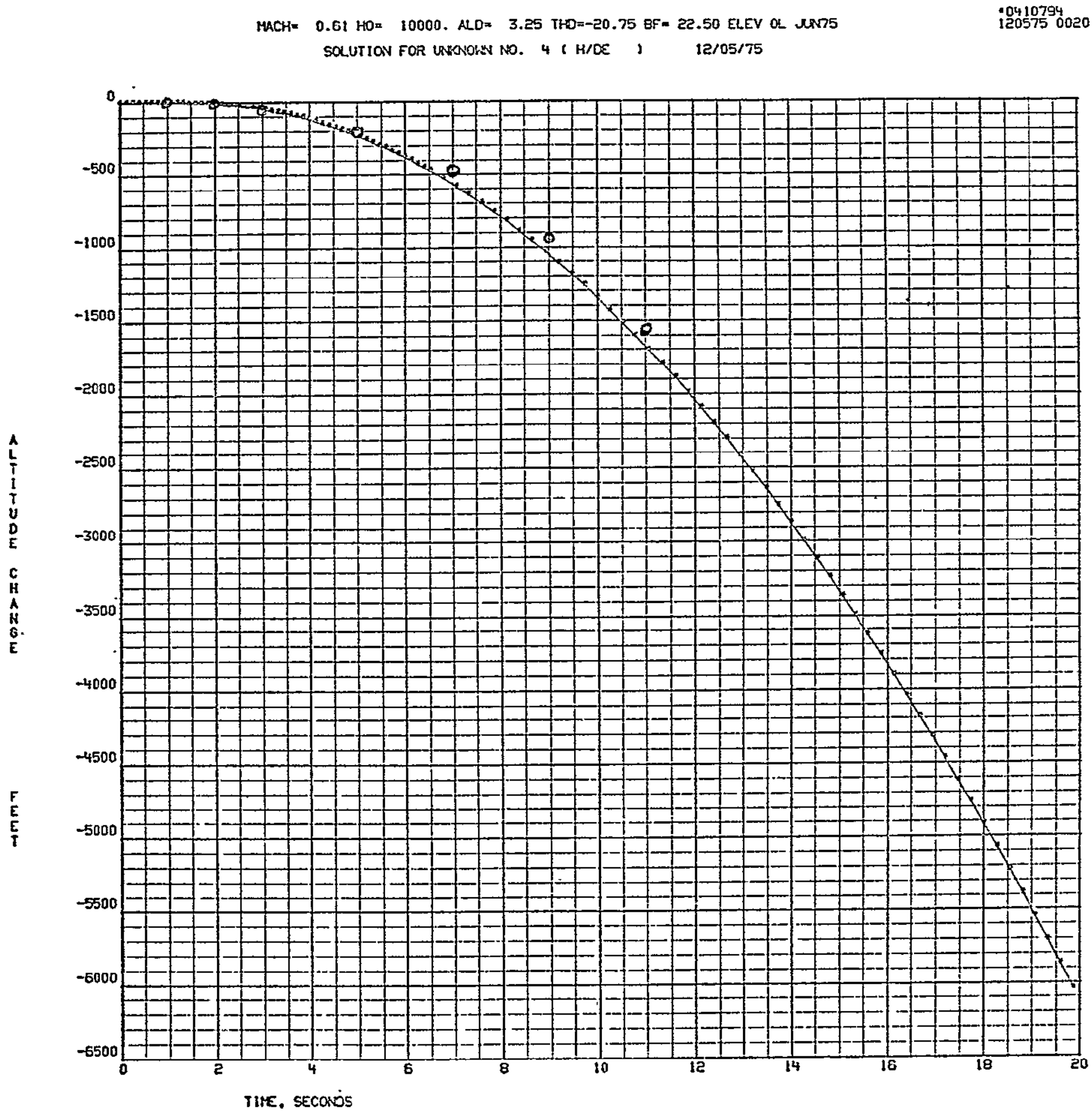
Step
10000'
M .61
φ ± 8°
1 SEC
FIG.
27
β ± 4°
p ± 4 °/s
r ± 1 °/s
Ay ± 4 F/s²
ΔAz ± ½ g
VI ± 800 F/s
RUDDER STEP RESPONSE
6 D. O. F.
FREE VEHICLE
H = 10,000 FT, M. = .61
δR ± 4°

FIG. 28



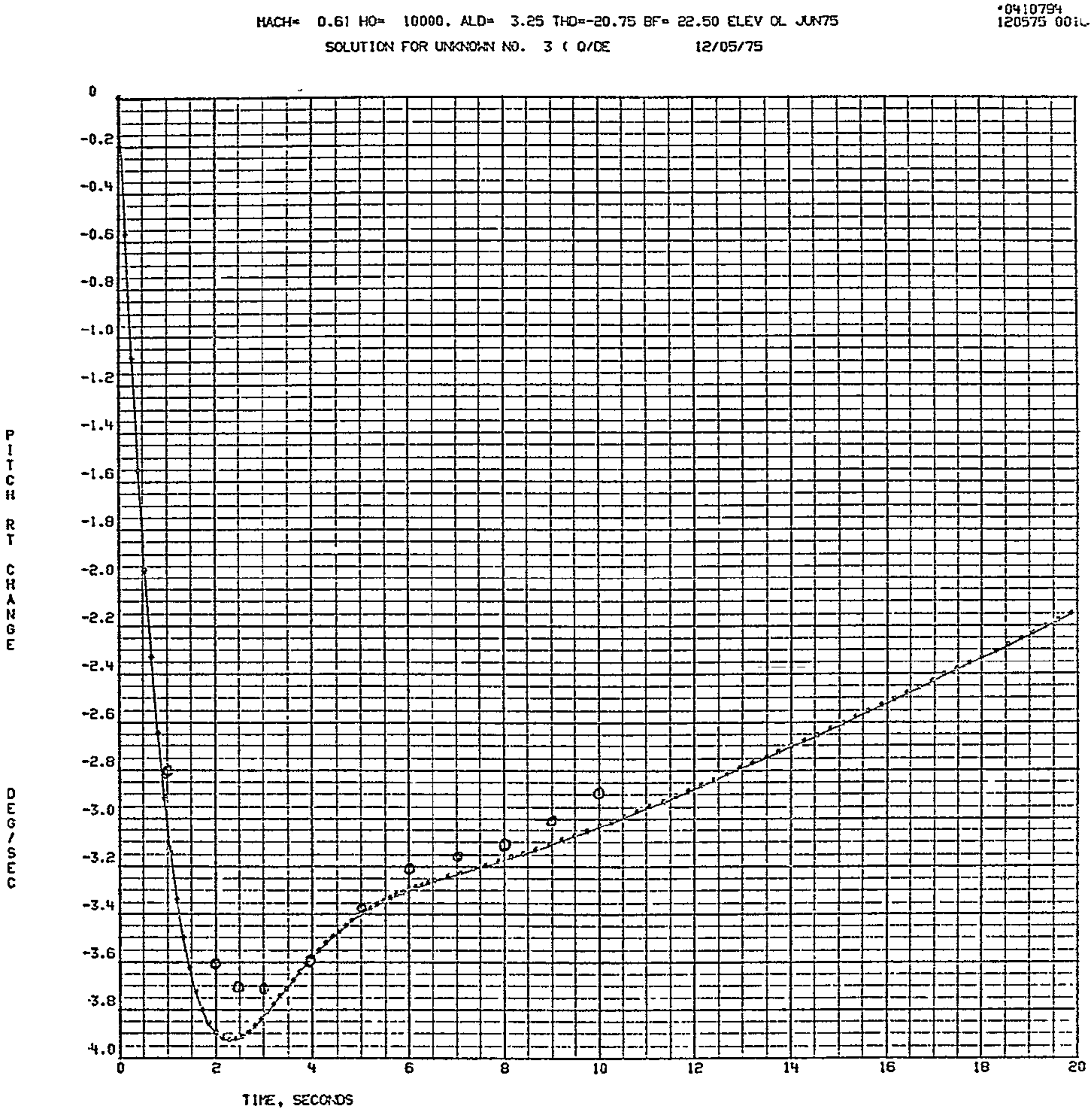
MACH= 0.61 HO= 10000. ALD= 3.25 THD=-20.75 BF= 22.50 ELEV OL JUN75
SOLUTION FOR UNKNOWN NO. 1 (V/DE 12/05/75
*0410794
120575 0008
VELOCITY CHANGE FT/SEC
TIME, SECONDS
200
180
160
140
120
100
80
60
40
20
0
-20
0
2
4
6
8
10
12
14
16
18
20
ROCKWELL
SPERRY

FIG. 29



*0410794
120575 0020
MACH= 0.61 HO= 10000. ALD= 3.25 THD=-20.75 BF= 22.50 ELEV OL JUN75
SOLUTION FOR UNKNOWN NO. 4 (H/DE)   12/05/75
ALTITUDE CHANGE FEET
0
-500
-1000
-1500
-2000
-2500
-3000
-3500
-4000
-4500
-5000
-5500
-6000
-6500
0
2
4
6
8
10
12
14
16
18
20
TIME, SECONDS

FIG. 30



MACH= 0.61 HO= 10000. ALD= 3.25 THO=-20.75 BF= 22.50 ELEV OL JUN75
SOLUTION FOR UNKNOWN NO. 3 (Q/DE 12/05/75
*0410794
120575 001
0
-0.2
-0.4
-0.6
-0.8
-1.0
-1.2
-1.4
-1.6
-1.8
-2.0
-2.2
-2.4
-2.6
-2.8
-3.0
-3.2
-3.4
-3.6
-3.8
-4.0
PITCH RT CHANGE DEG/SEC
0
2
4
6
8
10
12
14
16
18
20
TIME, SECONDS

FIG. 31

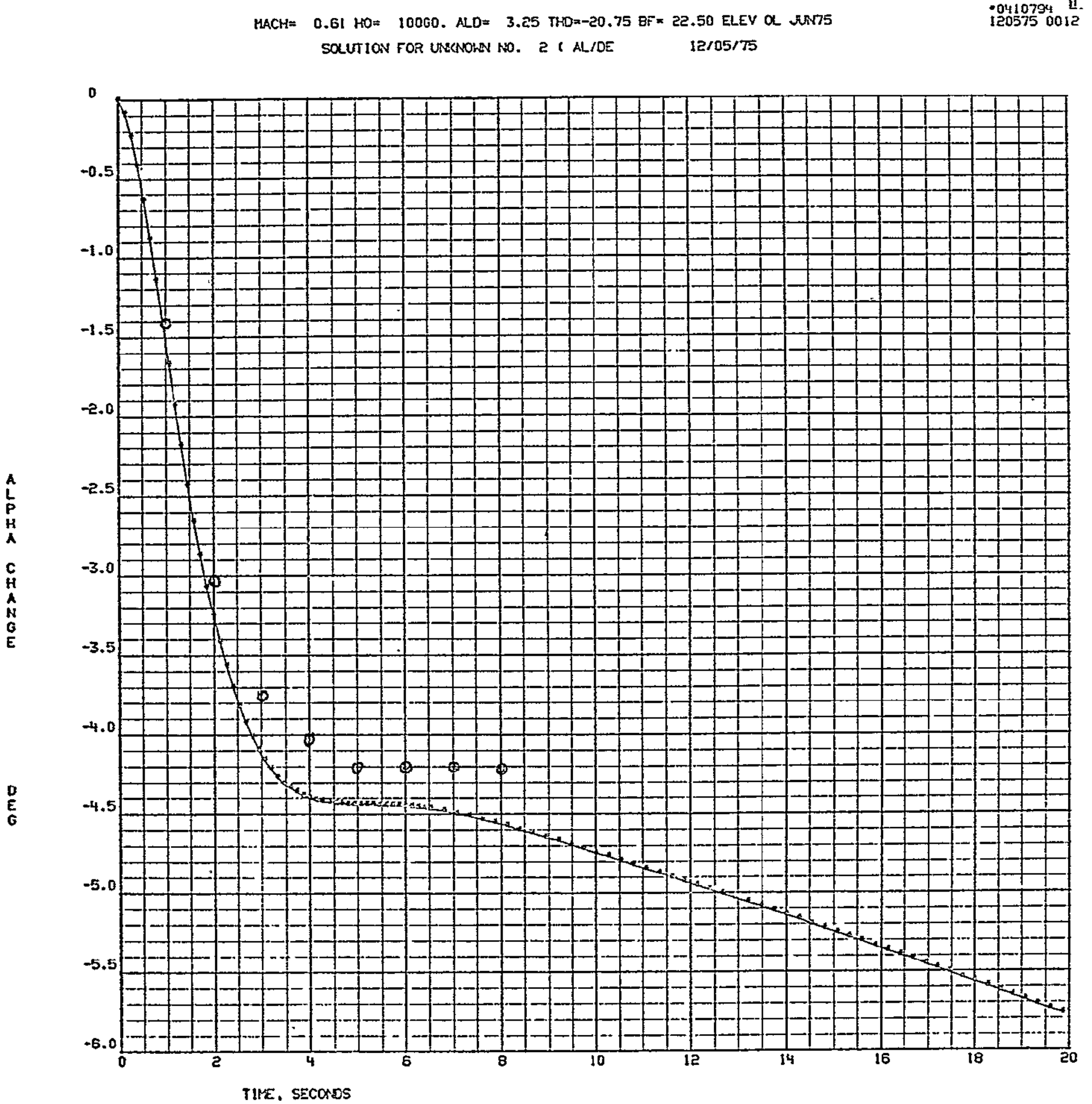
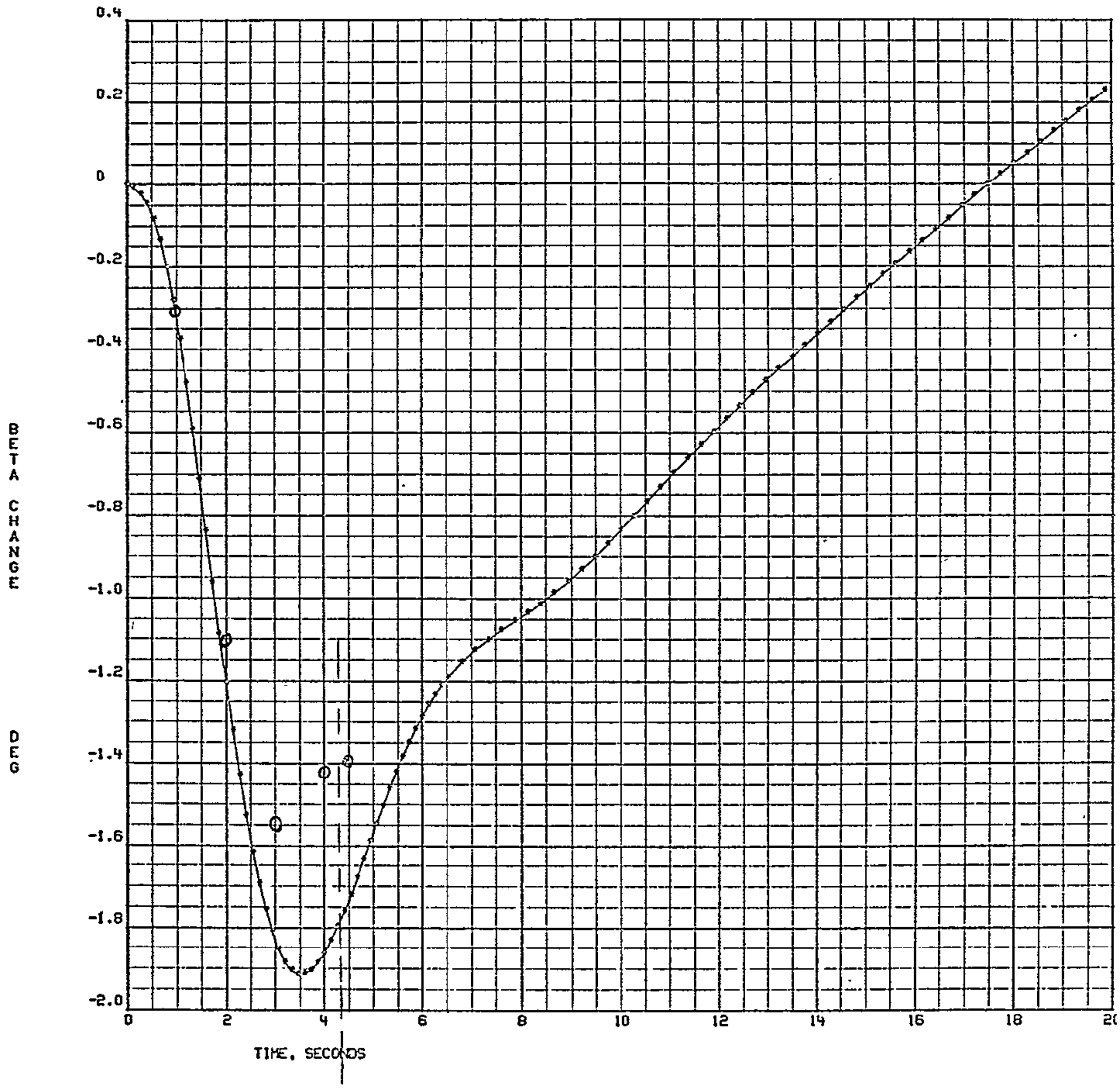

MACH= 0.61 HO= 10000. ALD= 3.25 THD=-20.75 BF= 22.50 ELEV OL JUN75
SOLUTION FOR UNKNOWN NO. 2 (AL/DE 12/05/75
*0410794
120575 0012
0
-0.5
-1.0
-1.5
-2.0
-2.5
-3.0
-3.5
-4.0
-4.5
-5.0
-5.5
-6.0
ALPHA CHANGE DEG
0
2
4
6
8
10
12
14
16
18
20
TIME, SECONDS

FIG. 32

MACH= 0.61 HO= 10000. ALD= 3.25 THD=-20.75 BF= 22.50 AIL OL JUN75

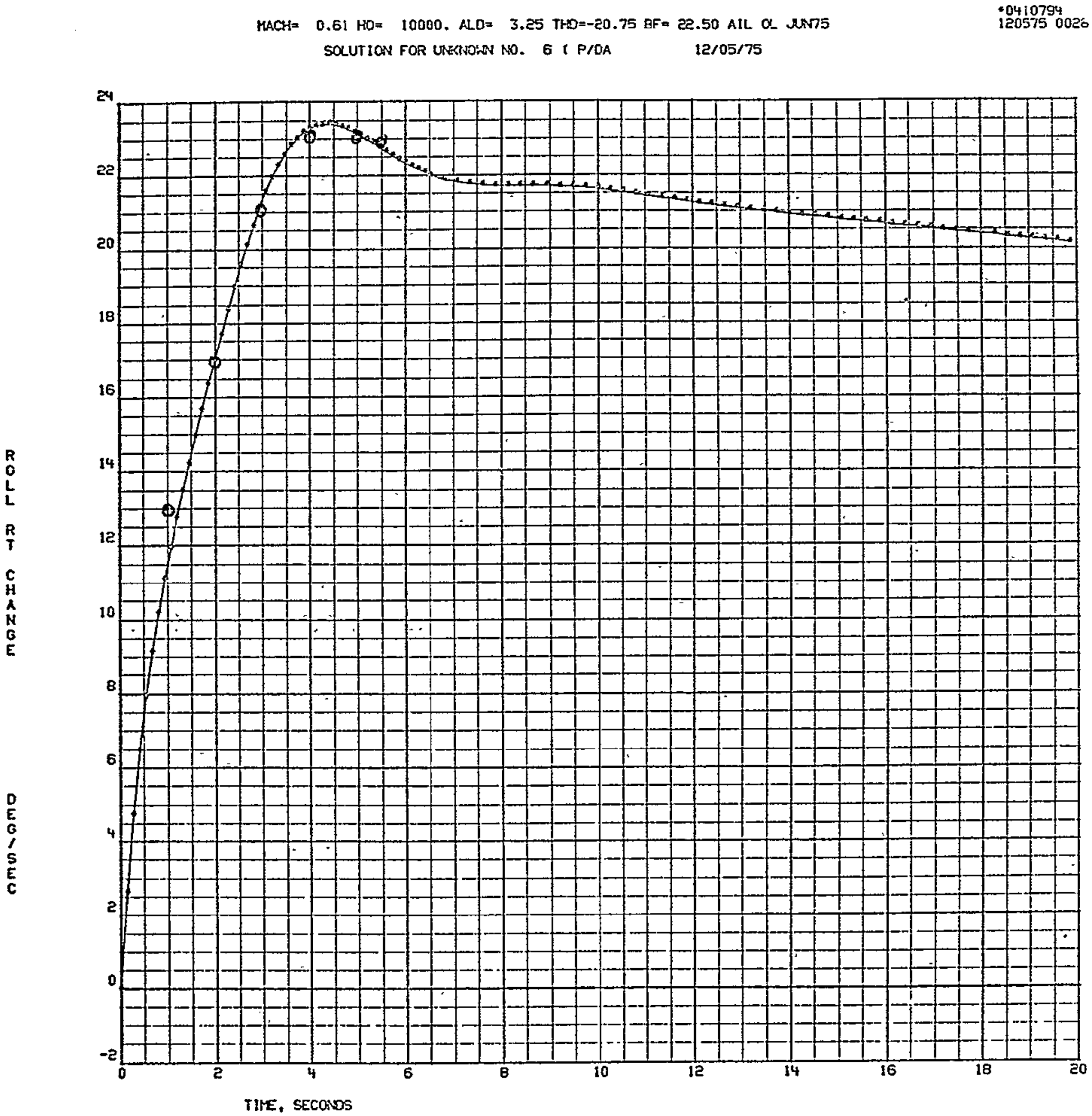
SOLUTION FOR UNKNOWN NO. 5 (B/DA 12/05/75

*0410794
120575 0024



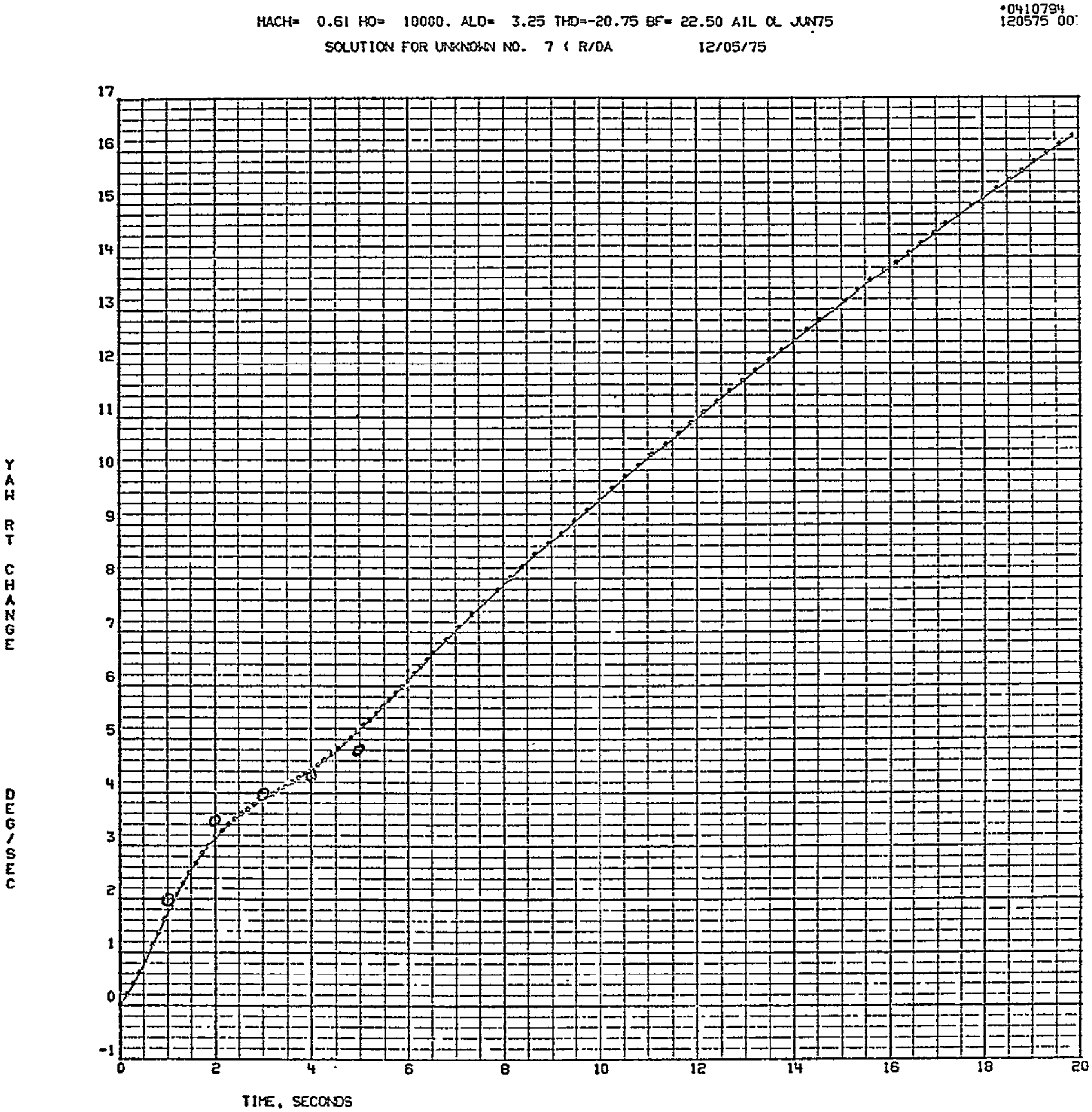
NOTE THAT FOR
$\phi \geq 90°$ THE SPERRY SIM
TRIGNOMETRIC
SUBROUTINES ARE
INVALID

FIG. 33



*0410794
120575 0026
MACH= 0.61 HO= 10000. ALD= 3.25 THD=-20.75 BF= 22.50 AIL OL JUN75
SOLUTION FOR UNKNOWN NO. 6 (P/DA 12/05/75
ROLL RT CHANGE DEG/SEC
TIME, SECONDS
24
22
20
18
16
14
12
10
8
6
4
2
0
-2
0
2
4
6
8
10
12
14
16
18
20

FIG. 34



MACH= 0.61 HO= 10000. ALD= 3.25 THD=-20.75 BF= 22.50 AIL OL JUN75
*0410794
120575 00
SOLUTION FOR UNKNOWN NO. 7 (R/DA 12/05/75
YAW RT CHANGE DEG/SEC
TIME, SECONDS
17
16
15
14
13
12
11
10
9
8
7
6
5
4
3
2
1
0
-1
0
2
4
6
8
10
12
14
16
18
20

FIG. 35

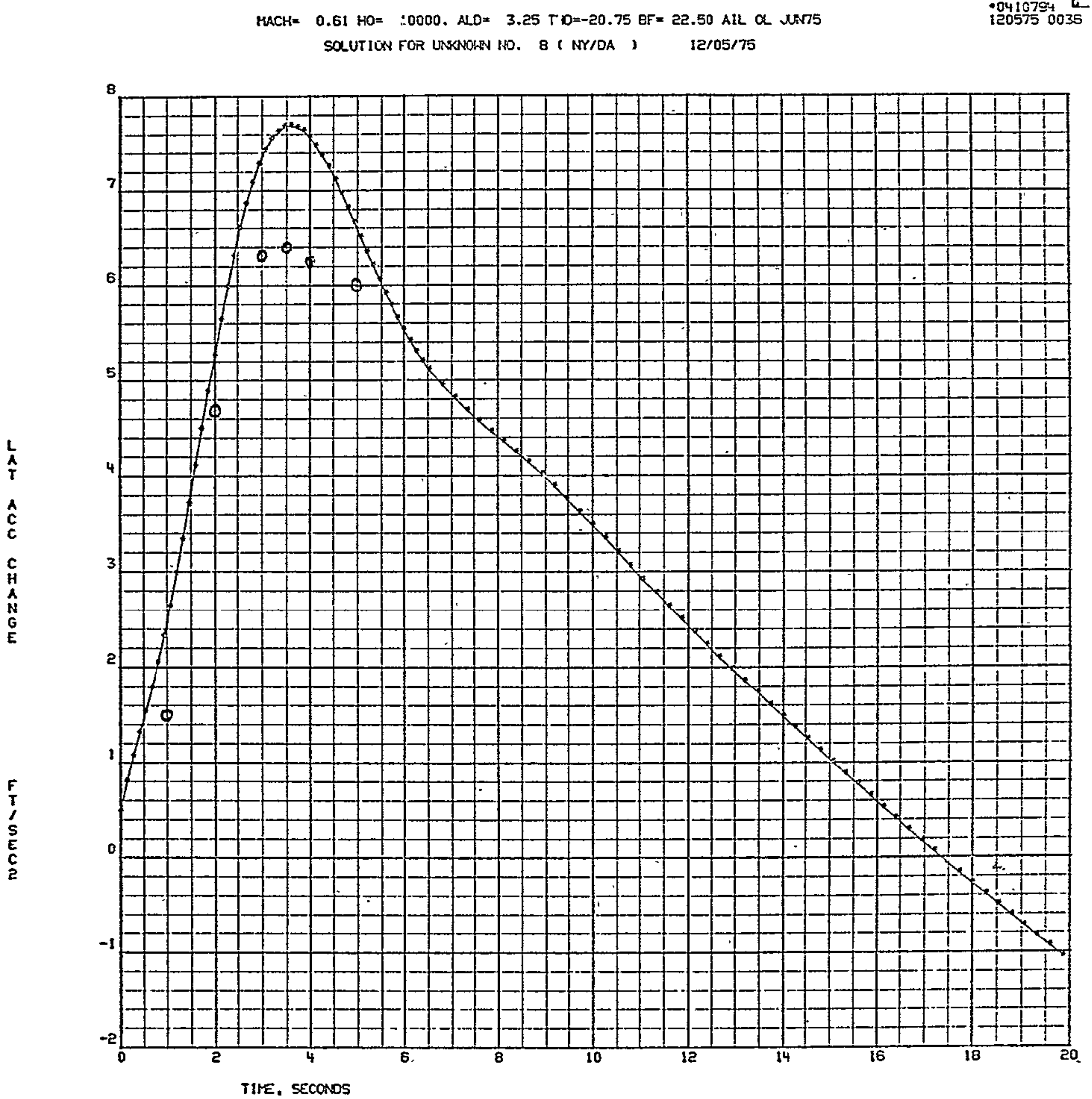
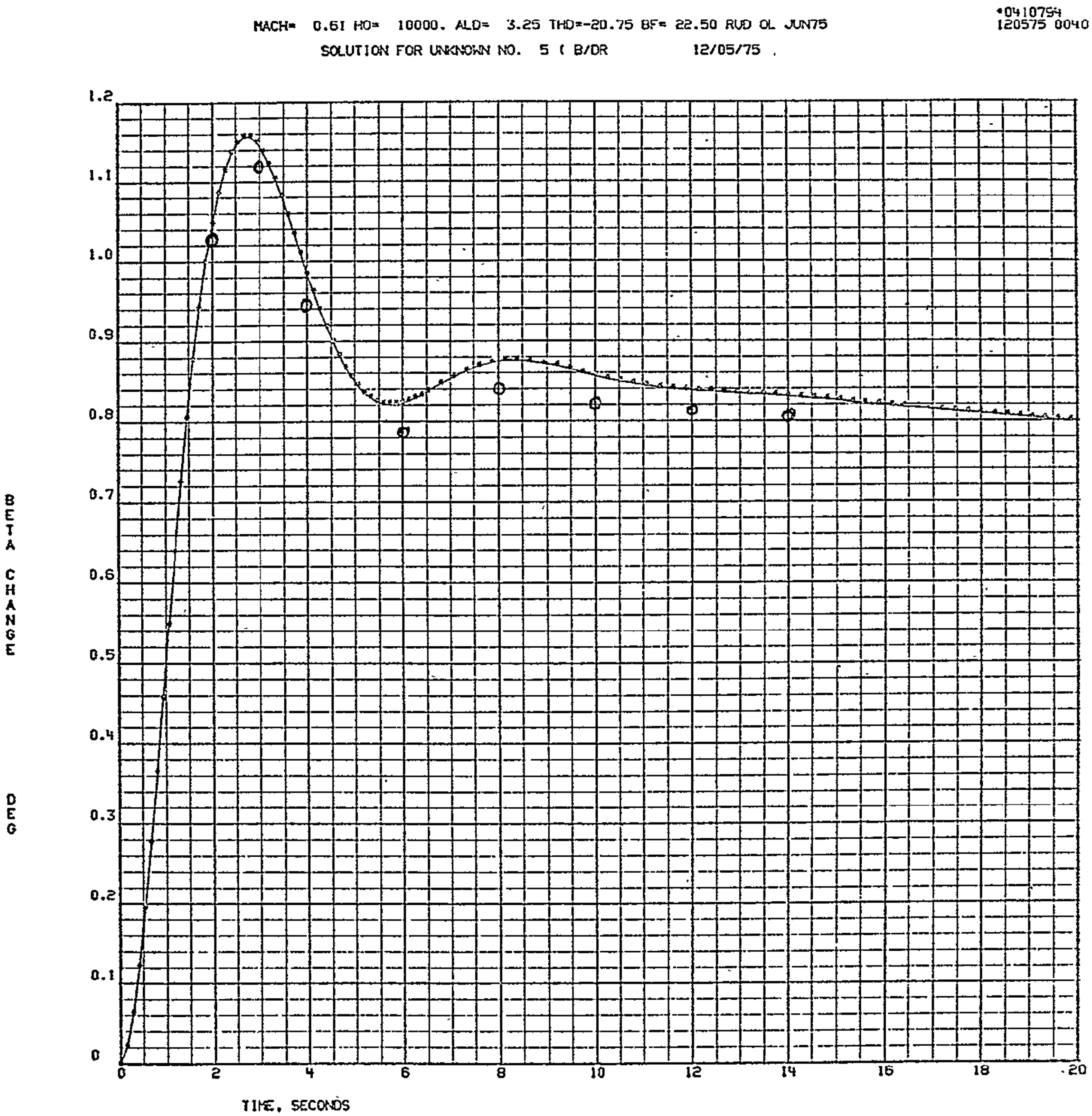
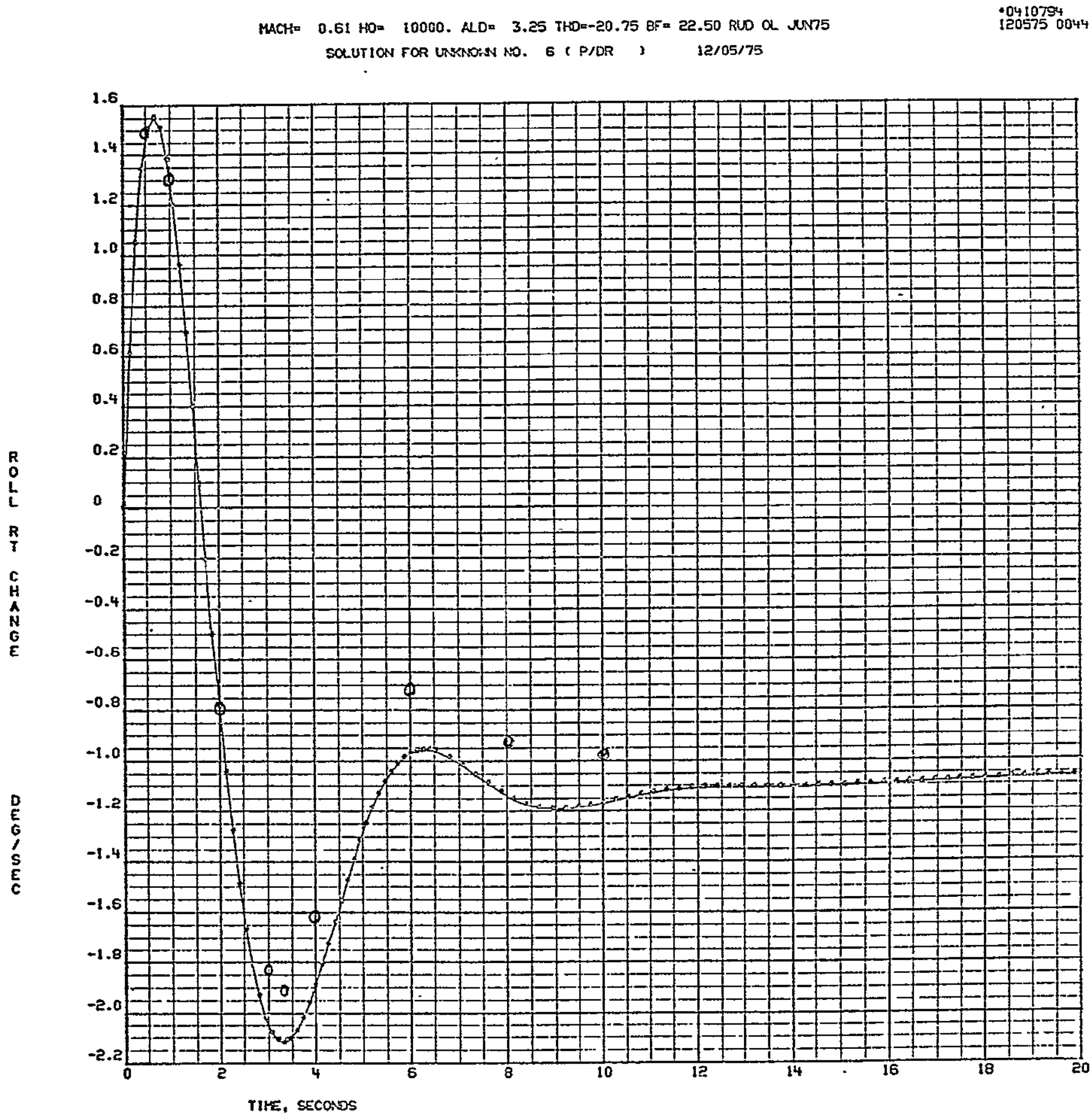

MACH= 0.61 HO= 10000. ALD= 3.25 TID=-20.75 BF= 22.50 AIL OL JUN75
SOLUTION FOR UNKNOWN NO. 8 (NY/DA) 12/05/75
*0410794
120575 0036
LAT ACC CHANGE FT/SEC2
TIME, SECONDS
8
7
6
5
4
3
2
1
0
-1
-2
0
2
4
6
8
10
12
14
16
18
20

FIG. 36



*0410754
120575 0040
MACH= 0.61 HO= 10000. ALD= 3.25 THD=-20.75 BF= 22.50 RUD OL JUN75
SOLUTION FOR UNKNOWN NO. 5 (B/DR 12/05/75 .
BETA CHANGE DEG
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1.0
1.1
1.2
0
2
4
6
8
10
12
14
16
18
20
TIME, SECONDS

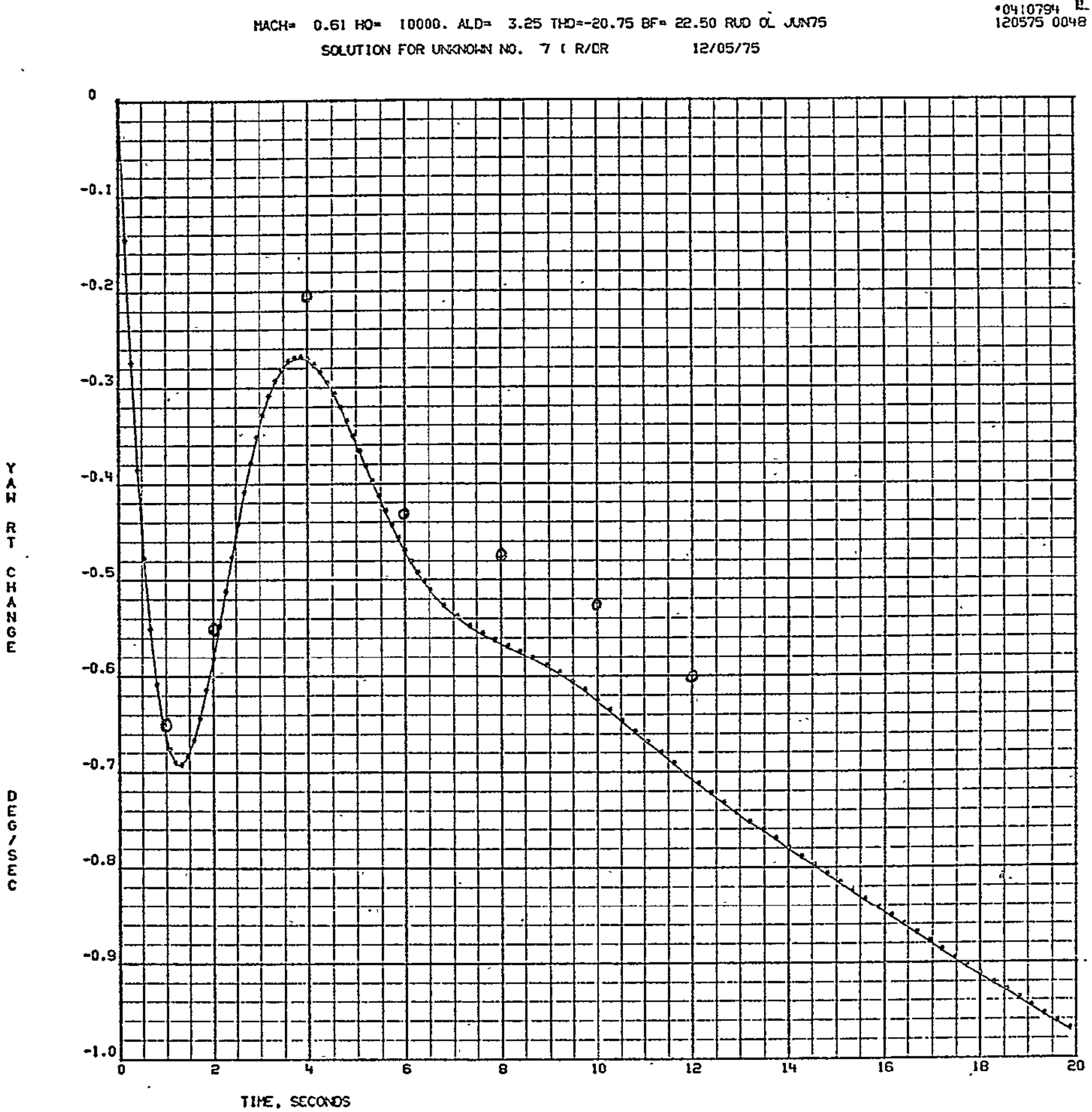
FIG. 37



*0410794
120575 0044
MACH= 0.61 HO= 10000. ALD= 3.25 THD=-20.75 BF= 22.50 RUD OL JUN75
SOLUTION FOR UNKNOWN NO. 6 (P/DR)
12/05/75
ROLL RT CHANGE DEG/SEC
TIME, SECONDS
1.6
1.4
1.2
1.0
0.8
0.6
0.4
0.2
0
-0.2
-0.4
-0.6
-0.8
-1.0
-1.2
-1.4
-1.6
-1.8
-2.0
-2.2
0
2
4
6
8
10
12
14
16
18
20

17

FIG. 38



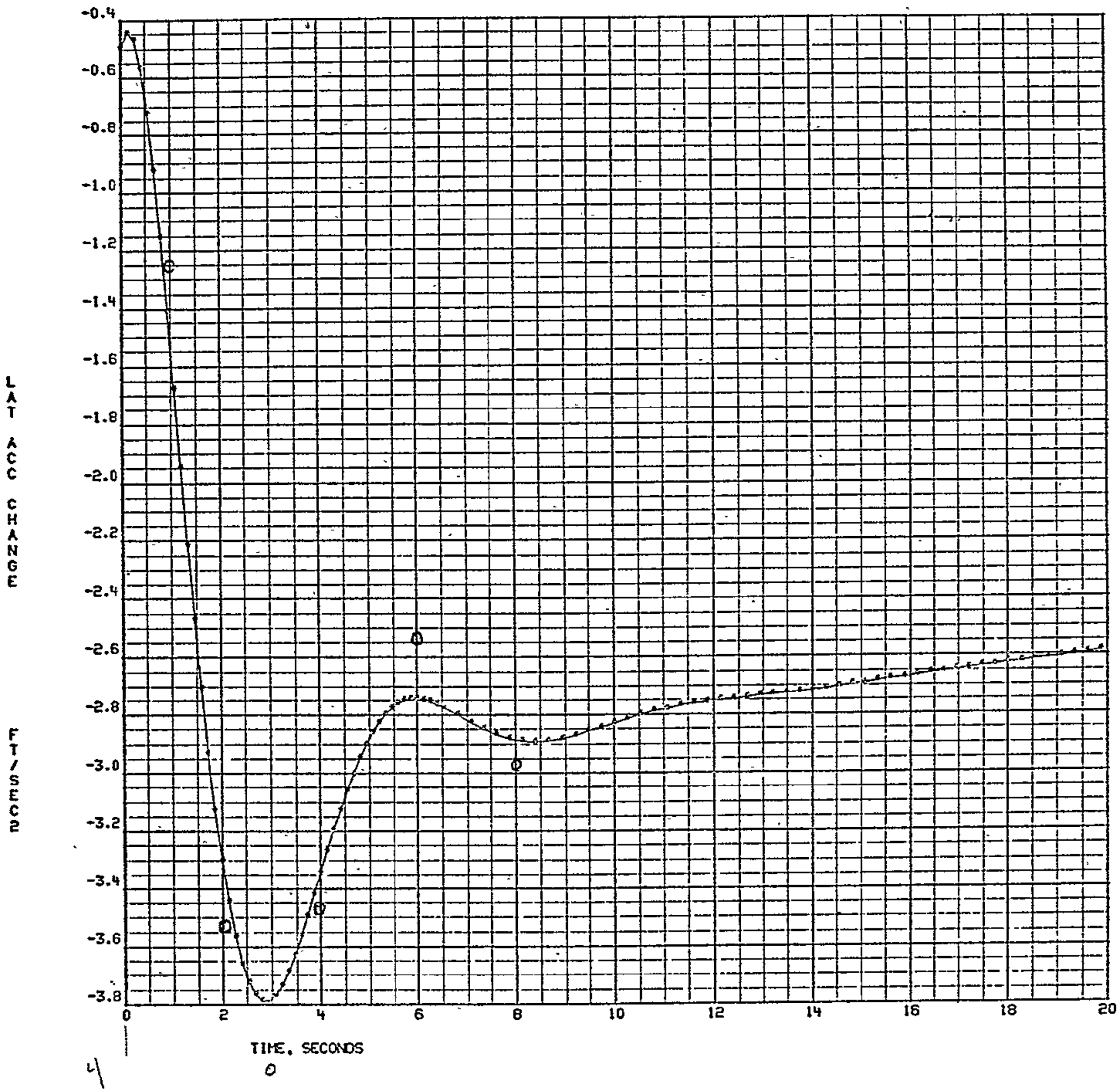
*0410794
120575 0048
MACH= 0.61 HO= 10000. ALD= 3.25 THD=-20.75 BF= 22.50 RUD OL JUN75
SOLUTION FOR UNKNOWN NO. 7 (R/DR 12/05/75
YAW RT CHANGE DEG/SEC
0
-0.1
-0.2
-0.3
-0.4
-0.5
-0.6
-0.7
-0.8
-0.9
-1.0
0 2 4 6 8 10 12 14 16 18 20
TIME, SECONDS

FIG. 39

MACH= 0.61 HO= 10000. ALD= 3.25 THD=-20.75 BF= 22.50 RUD OL JUN75

SOLUTION FOR UNKNOWN NO. 8 (NY/DR) 12/05/75

*0410794
120575 0052



SECTION 5

FLAT TURN STUDY

SPERRY AUTOLAND MEMO NO: 5

DATE: 26 January 1976

COPY:

NASA SHUTTLE

AUTOLAND SUPPORT PROGRAM

TASK NO: 4

TASK NAME: Flat Turn Study

TASK DESCRIPTION: The Flat Turn ("wings level maneuvering near touchdown") subphase of Autoland as presently documented in the FSSR has never been fully evaluated with regard to its effect upon vehicle stability and touchdown performance. In addition, the real reason for its inclusion has always been unclear.

It will be shown in this study that nothing is gained by using Flat Turn during autoland, and in fact that it has an adverse effect upon touchdown performance.

Figure 1 shows the present 15 July 1975 FSSR implementation of the roll and touchdown guidance laws. Using the defined gains of KY1 = 0.1, KYDOT = 10, A3LOC = -0.01 and KR1 = 1, the time history of an automatic flight is shown in Figure 2 for the ØK payload, forward CG vehicle. In this case a Flat Turn altitude of 5 feet was used, (as defined in RM/Moding Sequencing FSSR, 15 July 1975) and at this point a 20 foot Y error step was introduced which shifts the Y reference 20 feet to the right of the centerline. Little time before touchdown is available for lateral maneuvering to correct this error, although the rudder initially deflects in a proper negative direction to zero the error. For a Flat Turn altitude of 20 feet, the response is as shown in Figure 3, in which the rudder rapidly oscillates between -6° and $+12^{\circ}$ for one cycle before touchdown. Roll attitude reaches $+2.5^{\circ}$ and -4°. A Flat Turn altitude of 150 feet is used in Figure 4, in which a lateral instability is very evident. The Y error trace on channel 1 is seen to initially move in the correct direction but it rapidly diverges. Note that at the start of Flat Turn the vehicle roll attitude shows a divergent oscillation even though the roll attitude command is zeroed. It is concluded that the FSSR gain KR1 is unsatisfactory.

It has been suggested that the Y and Y dot gains be reduced by one-half to decrease roll activity in turbulence (C.S. Draper Shuttle Memo No. 28, dated 14 August 1975). Using these gains (KY1 = 0.05, KYDOT = 5 and KR1 = 1), results in the response shown in Figure 5 using a Flat Turn altitude of 150 feet to allow enough time to gage the stability. The oscillation is evident here, also.

Reducing KR1 to 0.7 results in Figure 6 in which a damped response zeros the Y error, but results in a Y overshoot. This was unavoidable due to the Y dot value having to increase sufficiently to result in the required yaw rate command. Sideslip angle and heading angle changes are also much smoother and are not as disconcerting to the pilot as the oscillatory motion shown in sideslip and heading in Figure 5. This latter criteria is actually more important than the zeroing of Y error, since the rate of Y correction is dependent upon values of roll attitude and lateral velocity at the start of Flat Turn. It should be noted that even in this more stable response, roll angle increases during Flat Turn and "wings level" is attained through a convergent oscillation. Figures 7 and 8 show the results for a 20 and 5 foot Flat Turn respectively. A minimum of oscillation in sideslip and heading angle is seen. Roll attitude, however, begins to increase at Flat Turn in both cases.

If it is not desired to halve the lateral guidance gains KY1 and KYDOT, a corresponding set of responses as described above can be obtained by halving gain KR1, from 0.7 to 0.35. However, as is shown in Figure 9, the Y response to a Flat Turn Y step is slightly different because the initial conditions at Flat Turn are different due to the higher lateral gains, causing Y dot to be smalled prior to Flat Turn. A 20 foot Flat Turn is shown in Figure 10 using these gains. Sideslip and heading angle response is seen to be very similar to the corresponding Figure 7. Roll motion is still evident throughout Flat Turn, showing the coupling of the Yaw into the roll axis despite a zero roll guidance command.

Flat Turn is initiated at 150, 20 and 5 feet in the next set of runs without a Y step and with a maximum 3 σ headwind and turbulence (Figures 11, 12, 13) where KY1 = .05, KYDOT = 5, KY1 =.7. Roll motion during Flat Turn is seen to be only slightly less than without Flat Turn, shown by comparing Figures 11 and 13 and observing the Ch. 3 trace (roll attitude). Heading angle is larger during Flat Turn than without. These observations are borne out by a series of Monte Carlo runs in similar wind conditions, as tabulated in Table I. Heading angle standard deviation for 25 runs is higher for a 20 foot Flat Turn than no Flat Turn (0 feet), whereas roll angle is slightly less.

A typical time history for 3 σ crosswinds and turbulence is shown in Figure 14 for a 20 foot altitude Flat Turn, with KY1 = .05, $K\dot{Y}DOT$ = 5, KR1 = .7. Touchdown performance for this case is tabulated in Table II. Heading angle at touchdown is seen to be larger with Flat Turn in than out. (9° vs. 6.7°). In fact, it gets within 1° of the maximum angle of 10 degrees. The increase in roll angle of 1° between a 20 foot Flat Turn and no Flat Turn is negligible. No difference in performance is shown between the 5 and 20 foot Flat Turn altitudes, the latter value being recently suggested for coping with navigation errors due to loss of scan beam near touchdown.

It is concluded that heading at touchdown is increased when using Flat Turn in turbulence and that it does little to improve performance. It is recommended that Flat Turn be removed from Autoland.

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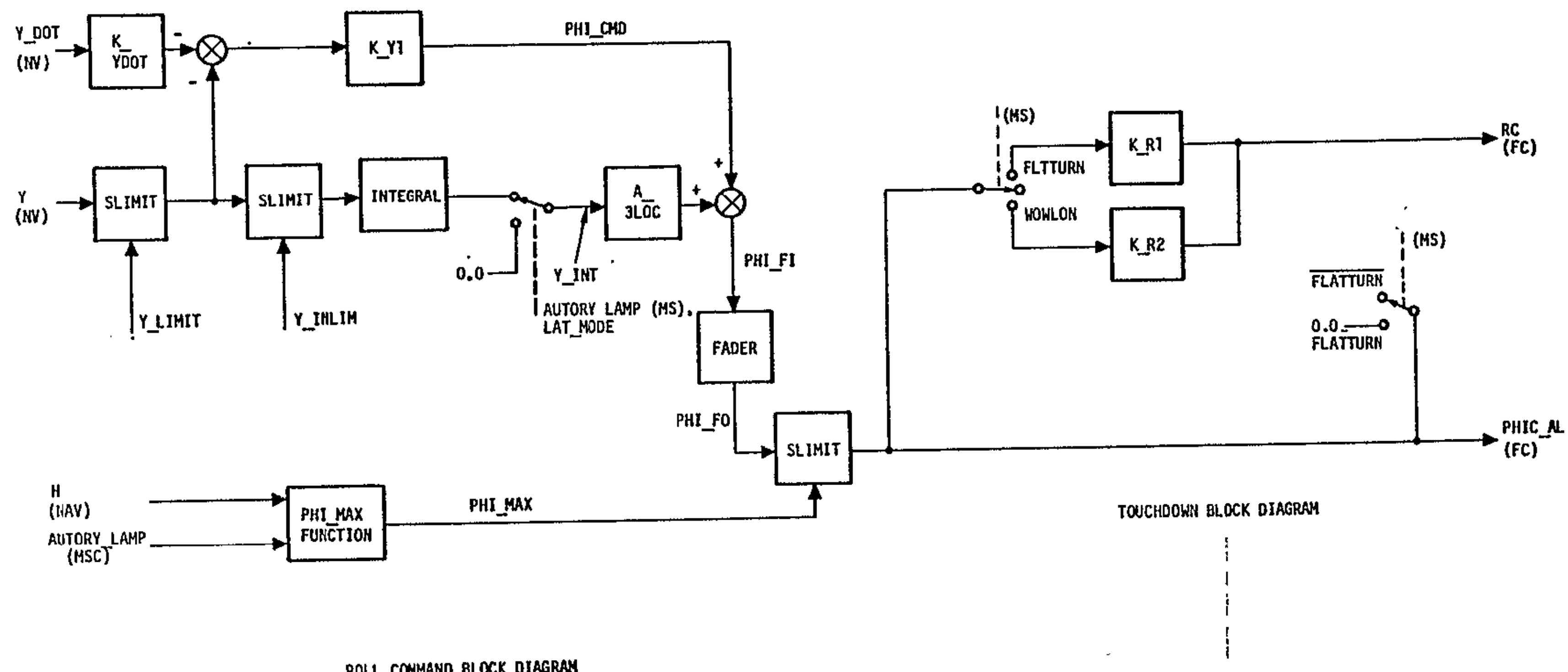


FIG. 1

Lateral Guidance Block Diagram

4-221, 4-222

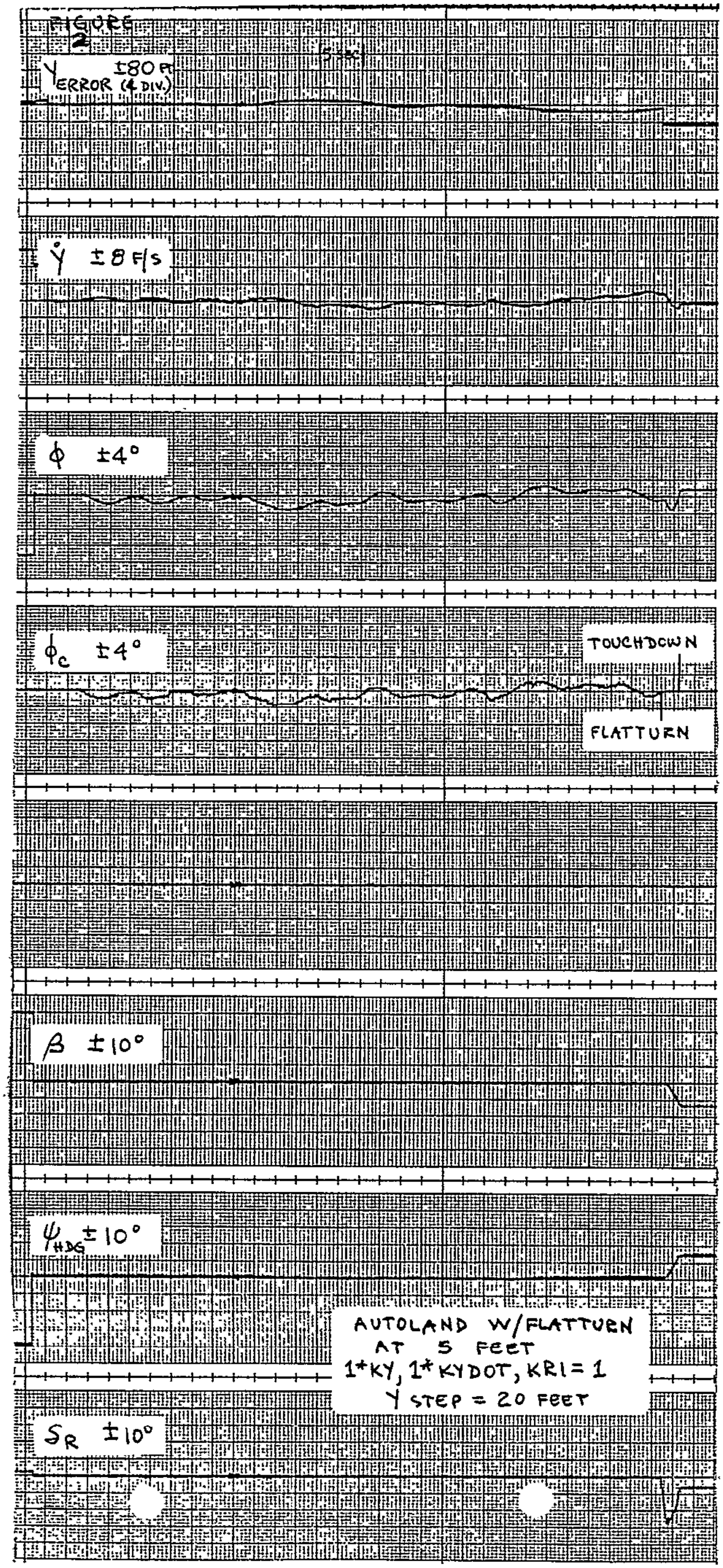


FIGURE 2
Y ERROR ±80 FT (4 DIV.)
Ẏ ±8 F/S
φ ±4°
φc ±4°
TOUCHDOWN
FLATTURN
β ±10°
ψHDG ±10°
AUTOLAND W/FLATTURN
AT 5 FEET
1*KY, 1*KYDOT, KR1 = 1
Y STEP = 20 FEET
δR ±10°

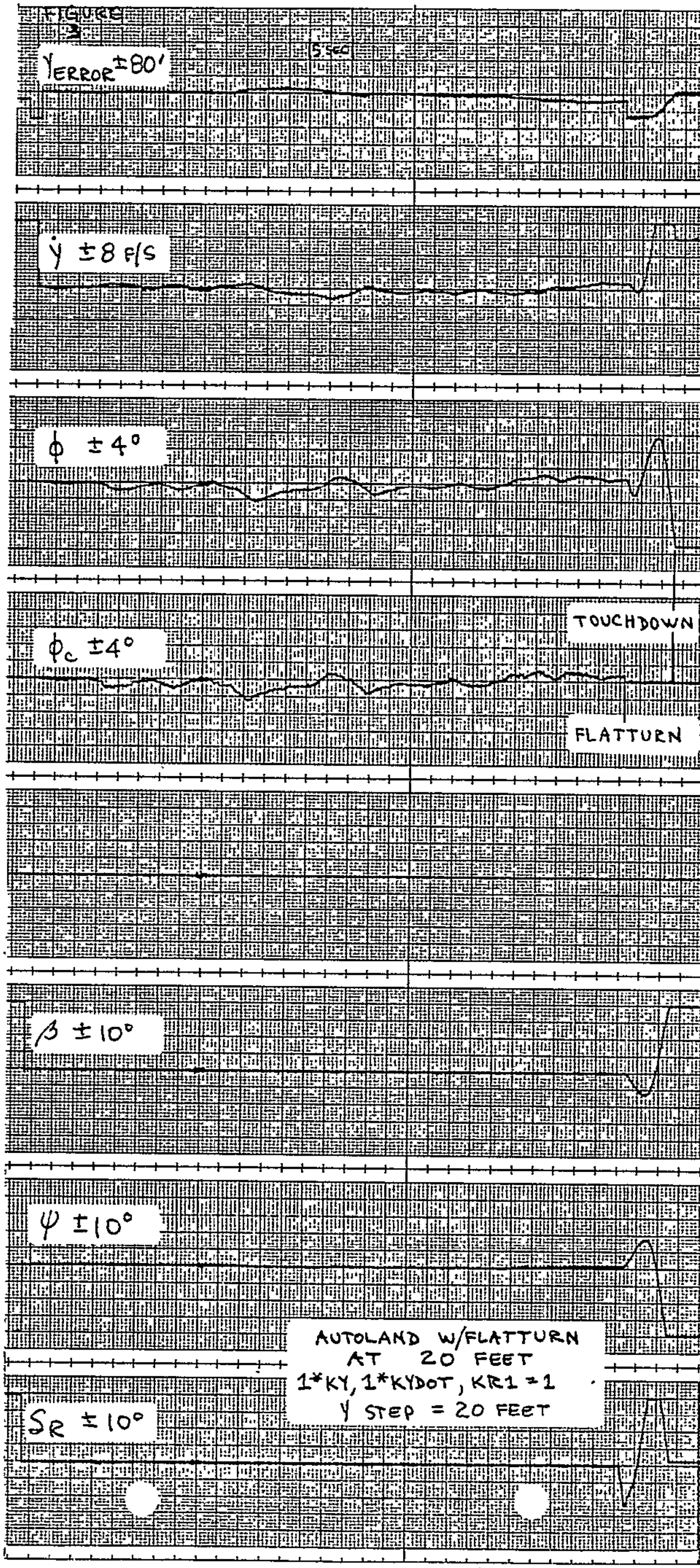
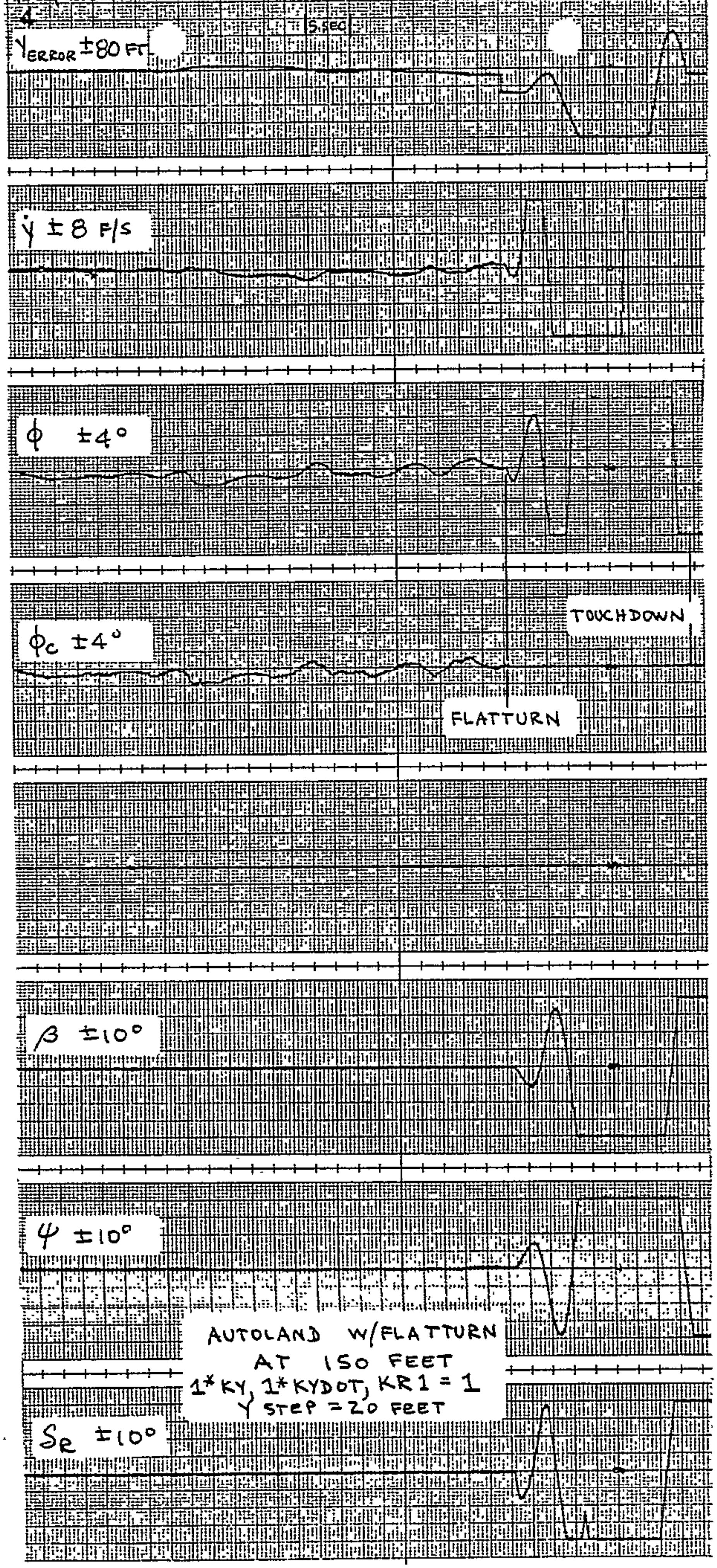


FIGURE 2
5 SEC
Y_{ERROR} ±80'
$\dot{y}$ ±8 f/s
ϕ ±4°
ϕ_c ±4°
TOUCHDOWN
FLATTURN
β ±10°
ψ ±10°
AUTOLAND W/FLATTURN
AT 20 FEET
1*KY, 1*KYDOT, KR1 = 1
Y STEP = 20 FEET
δ_R ±10°

FIGURE 4



5 SEC
Y_{ERROR} ±80 FT
$\dot{y}$ ±8 F/S
φ ±4°
$φ_c$ ±4°
TOUCHDOWN
FLATTURN
β ±10°
ψ ±10°
AUTOLAND W/FLATTURN
AT 150 FEET
1* KY, 1* KYDOT, KR1 = 1
Y STEP = 20 FEET
$δ_R$ ±10°

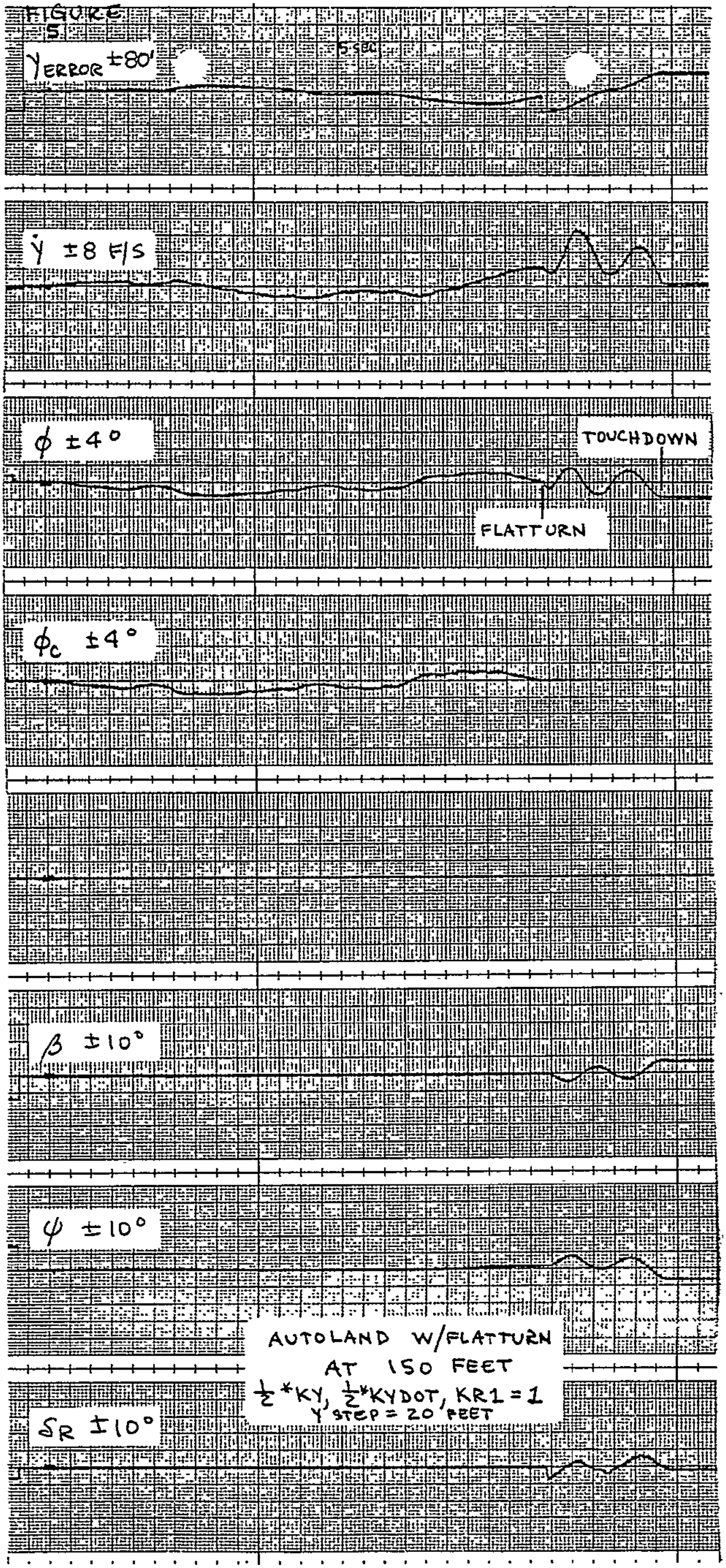


FIGURE 5
Y_{ERROR} ±80'
5 SEC
$\dot{Y}$ ±8 F/S
ϕ ±4°
TOUCHDOWN
FLATTURN
ϕ_c ±4°
β ±10°
ψ ±10°
AUTOLAND W/FLATTURN
AT 150 FEET
$\frac{1}{2}$*KY, $\frac{1}{2}$*KYDOT, KR1 = 1
Y STEP = 20 FEET
δ_R ±10°

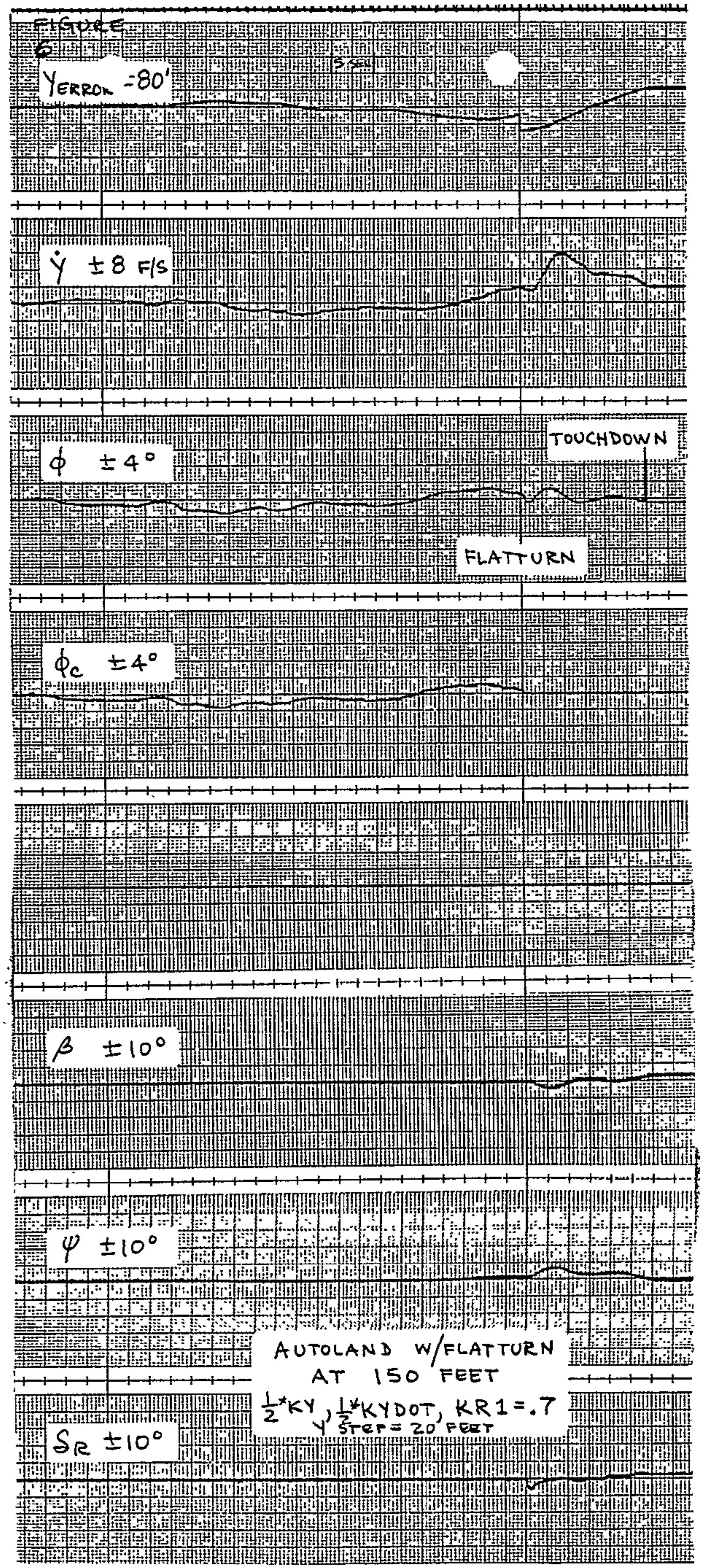


FIGURE 6
Y ERROR ±80'
5 SEC
Ẏ ±8 F/S
φ ±4°
TOUCHDOWN
FLATTURN
φc ±4°
β ±10°
ψ ±10°
AUTOLAND W/FLATTURN
AT 150 FEET
1/2*KY, 1/2*KYDOT, KR1=.7
Y STEP = 20 FEET
δR ±10°

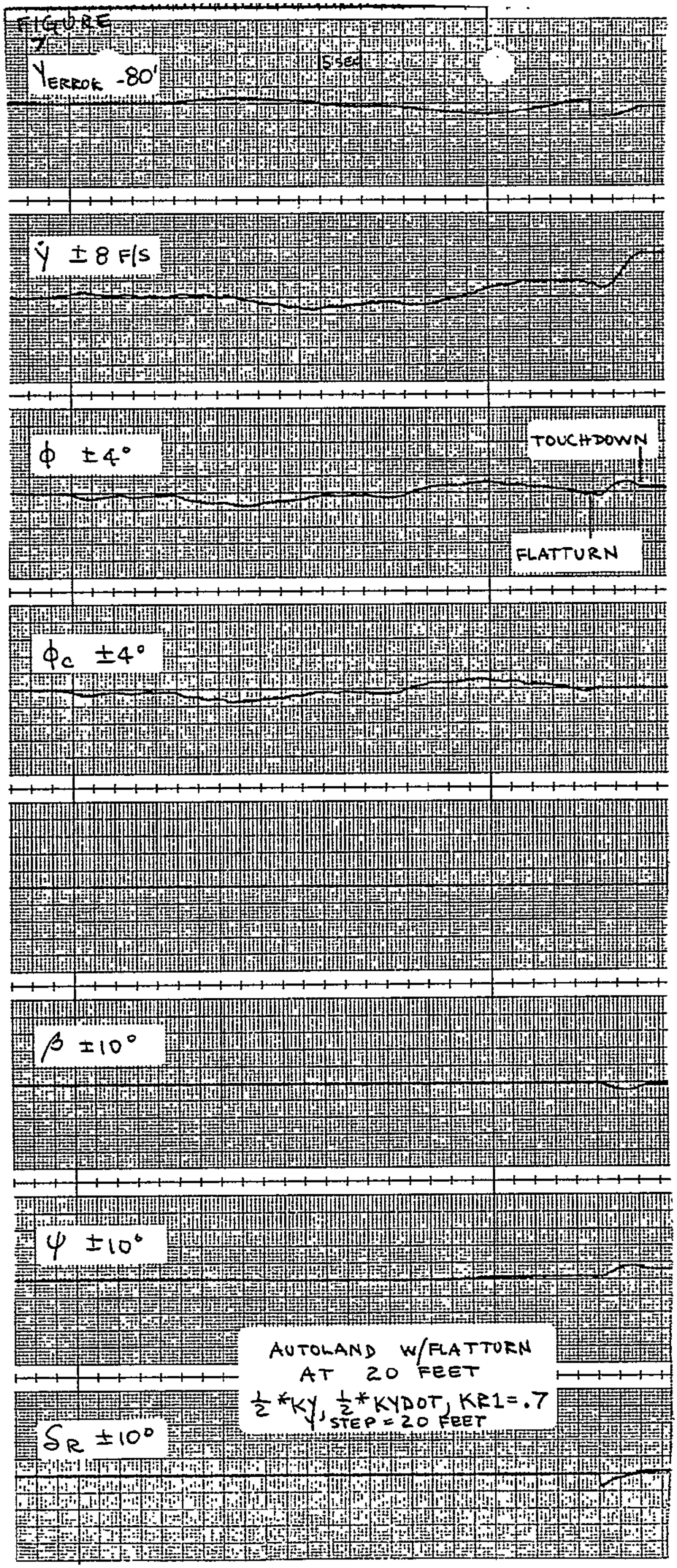


FIGURE 7
Y_{ERROR} -80'
5 SEC
$\dot{Y}$ ±8 F/S
ϕ ±4°
TOUCHDOWN
FLATTURN
ϕ_C ±4°
β ±10°
ψ ±10°
AUTOLAND W/FLATTURN
AT 20 FEET
$\frac{1}{2}$*KY, $\frac{1}{2}$*KYDOT, KR1=.7
Y STEP = 20 FEET
δ_R ±10°

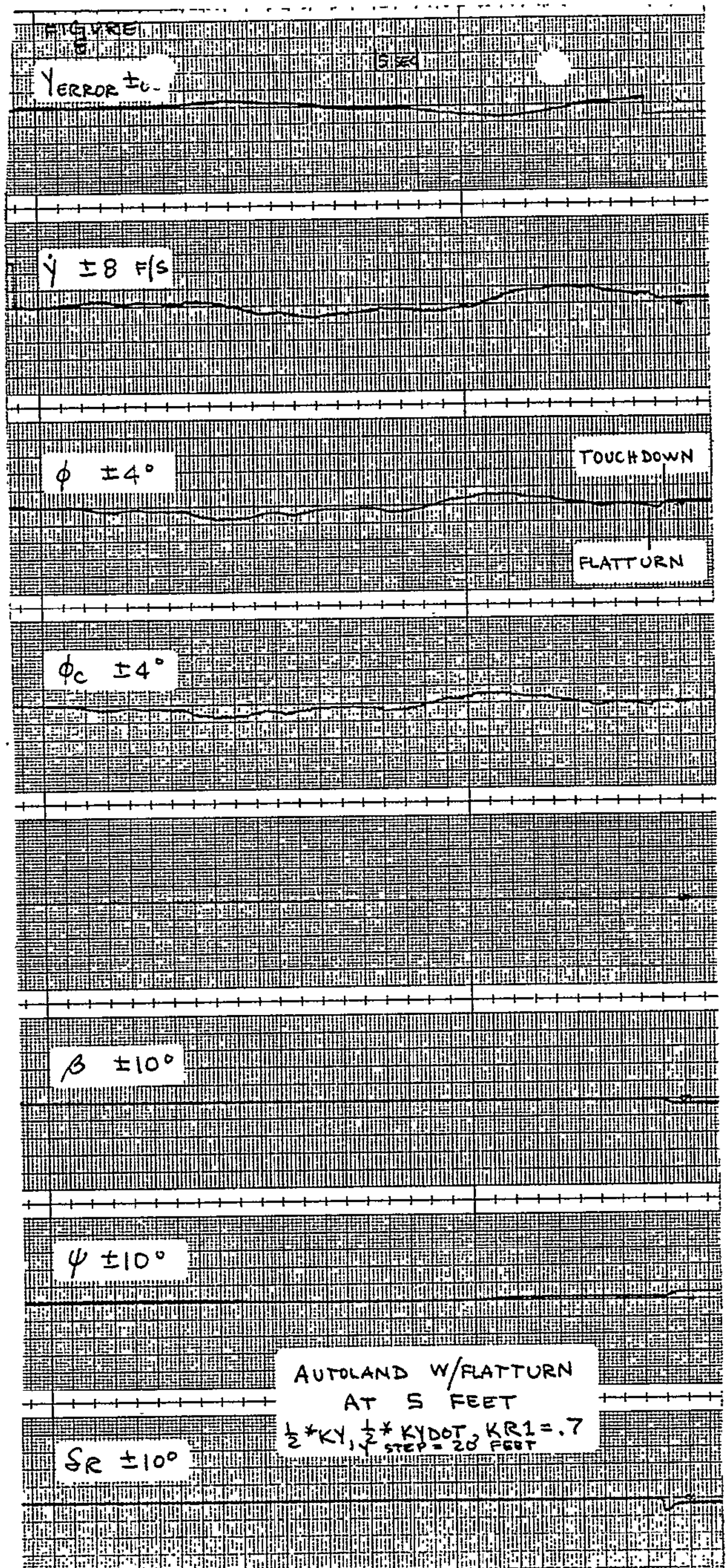


FIGURE 6
5 SEC
Y ERROR ±6
ẏ ±8 F/S
φ ±4°
TOUCHDOWN
FLATTURN
φc ±4°
β ±10°
ψ ±10°
AUTOLAND W/FLATTURN
AT 5 FEET
½*KY, ½*KYDOT, KR1 = .7
Y STEP = 20 FEET
δR ±10°

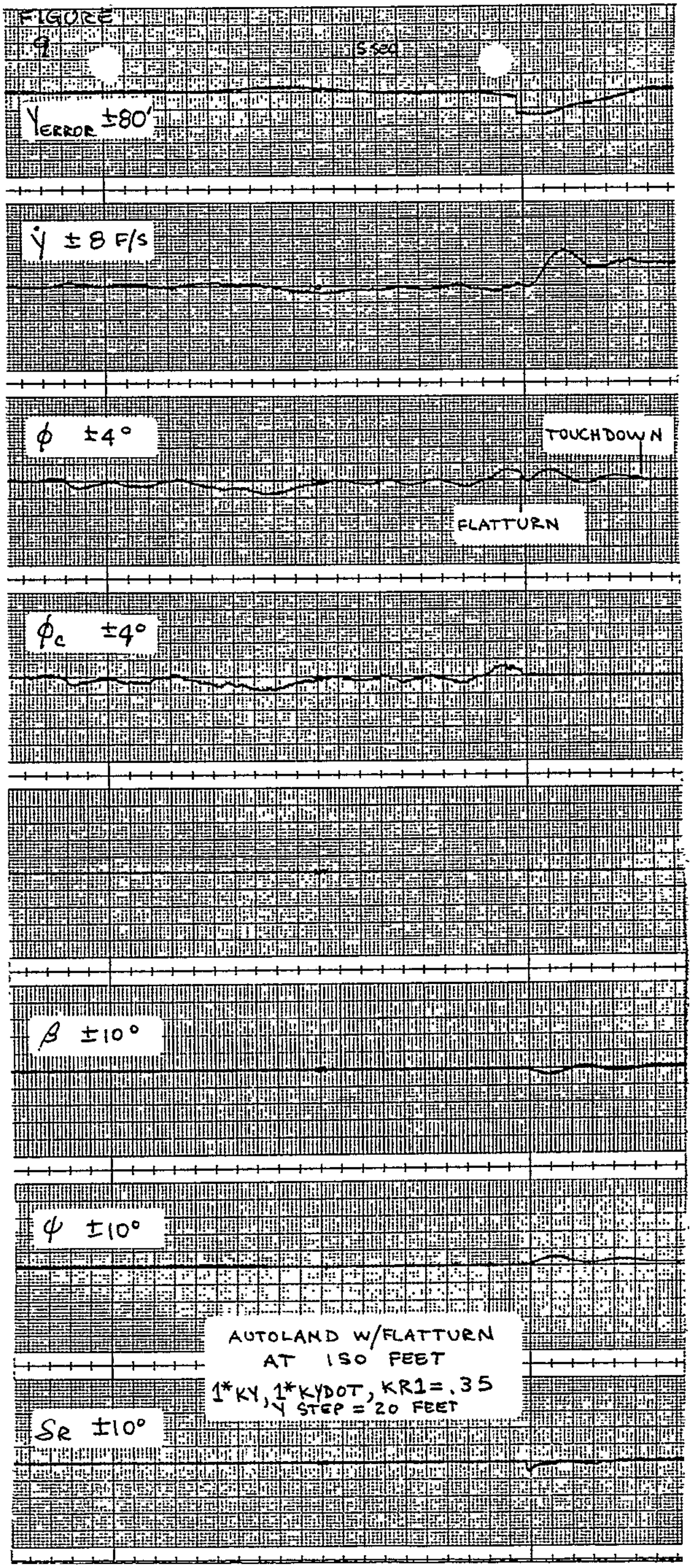
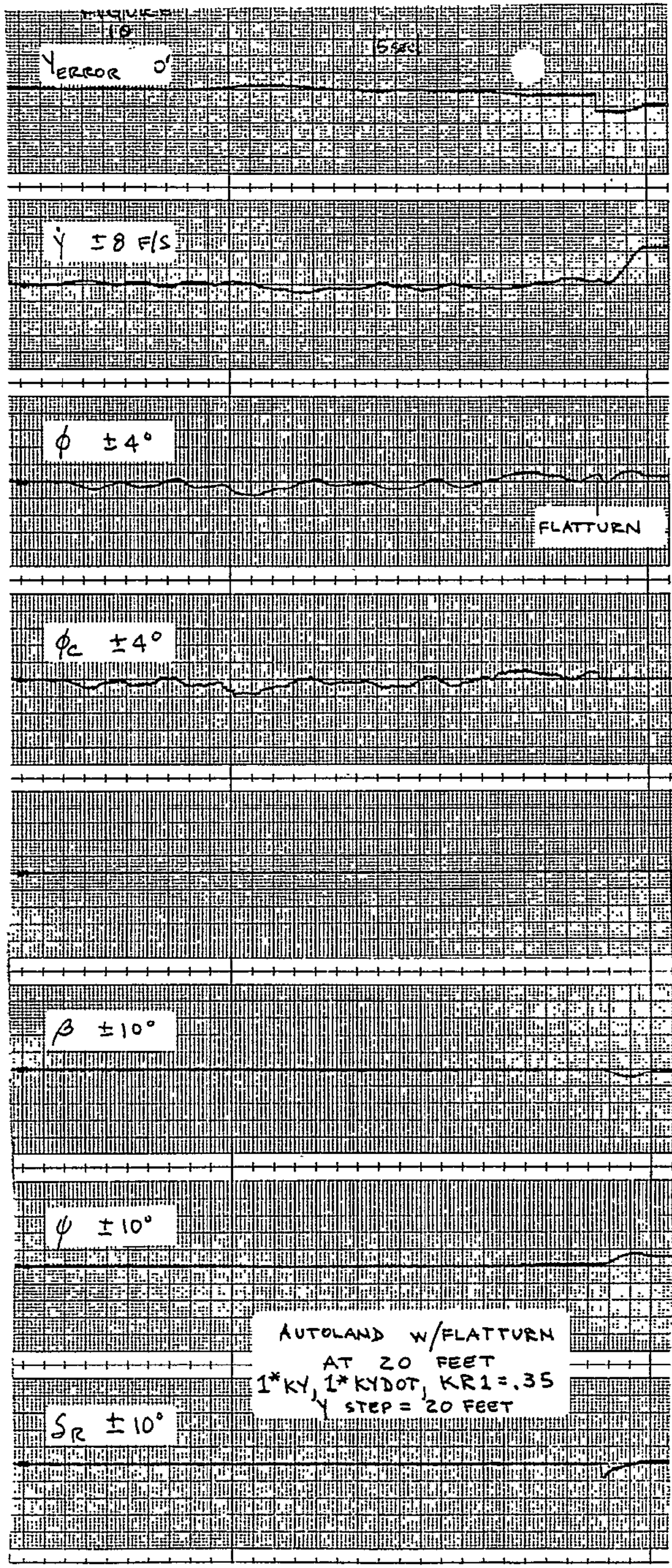

5 sec
Y_{ERROR} ±80'
$\dot{Y}$ ±8 F/S
ϕ ±4°
TOUCHDOWN
FLATTURN
ϕ_c ±4°
β ±10°
ψ ±10°
AUTOLAND W/FLATTURN
AT 150 FEET
1*KY, 1*KYDOT, KR1=.35
Y STEP = 20 FEET
δ_R ±10°

FIGURE 9



10
Y ERROR 0'
15 sec
Ẏ ±8 F/S
φ ±4°
FLATTURN
φc ±4°
β ±10°
ψ ±10°
δR ±10°
AUTOLAND W/FLATTURN
AT 20 FEET
1*KY, 1*KYDOT, KR1=.35
Y STEP = 20 FEET

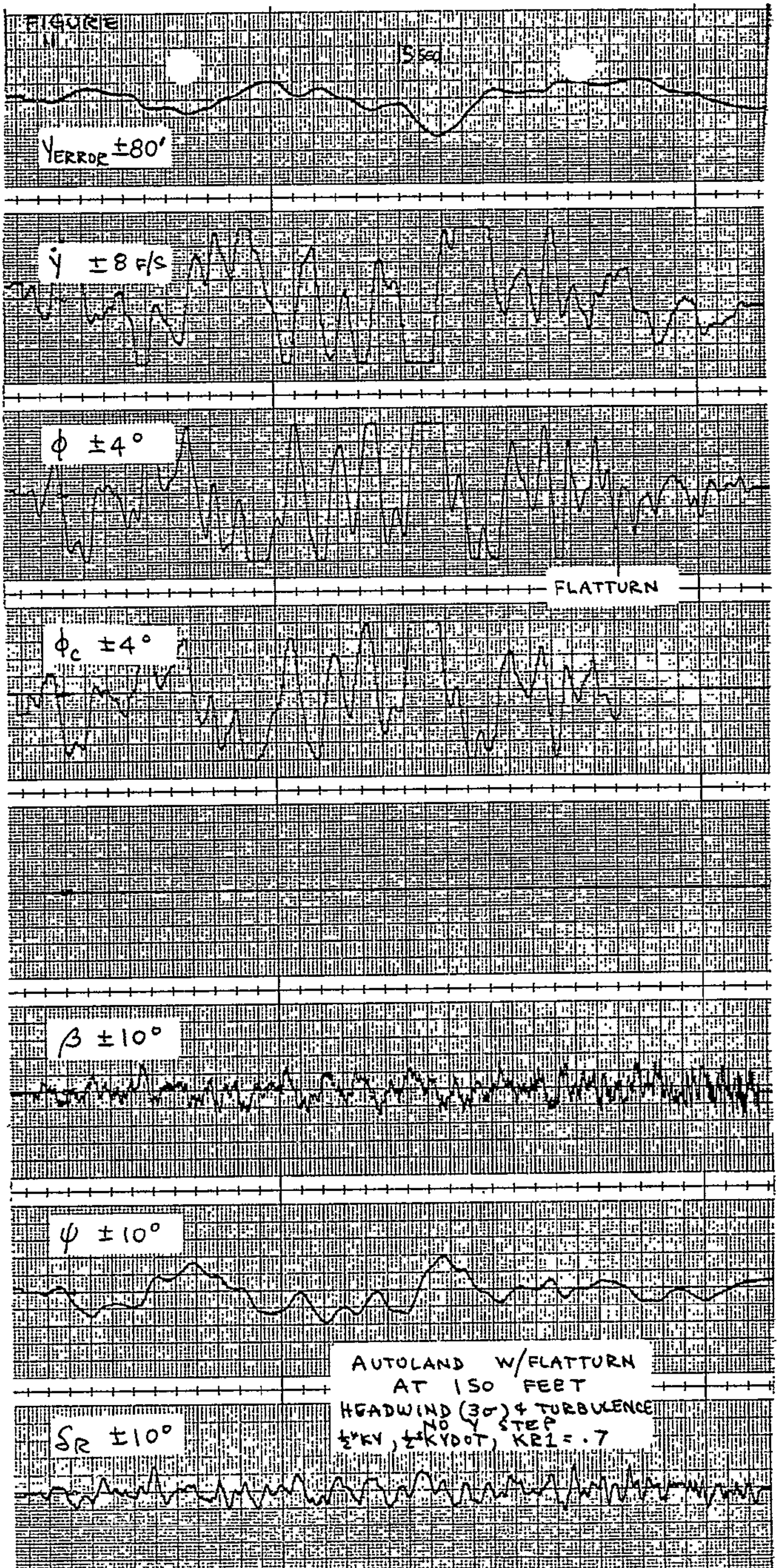


FIGURE
11
15 sec
Y_{ERROR} ±80'
$\dot{y}$ ±8 F/S
ϕ ±4°
FLATTURN
ϕ_c ±4°
β ±10°
ψ ±10°
AUTOLAND W/FLATTURN
AT 150 FEET
HEADWIND (3σ) & TURBULENCE
NO Y STEP
½ KY, ½ KYDOT, KR1 = .7
δ_R ±10°

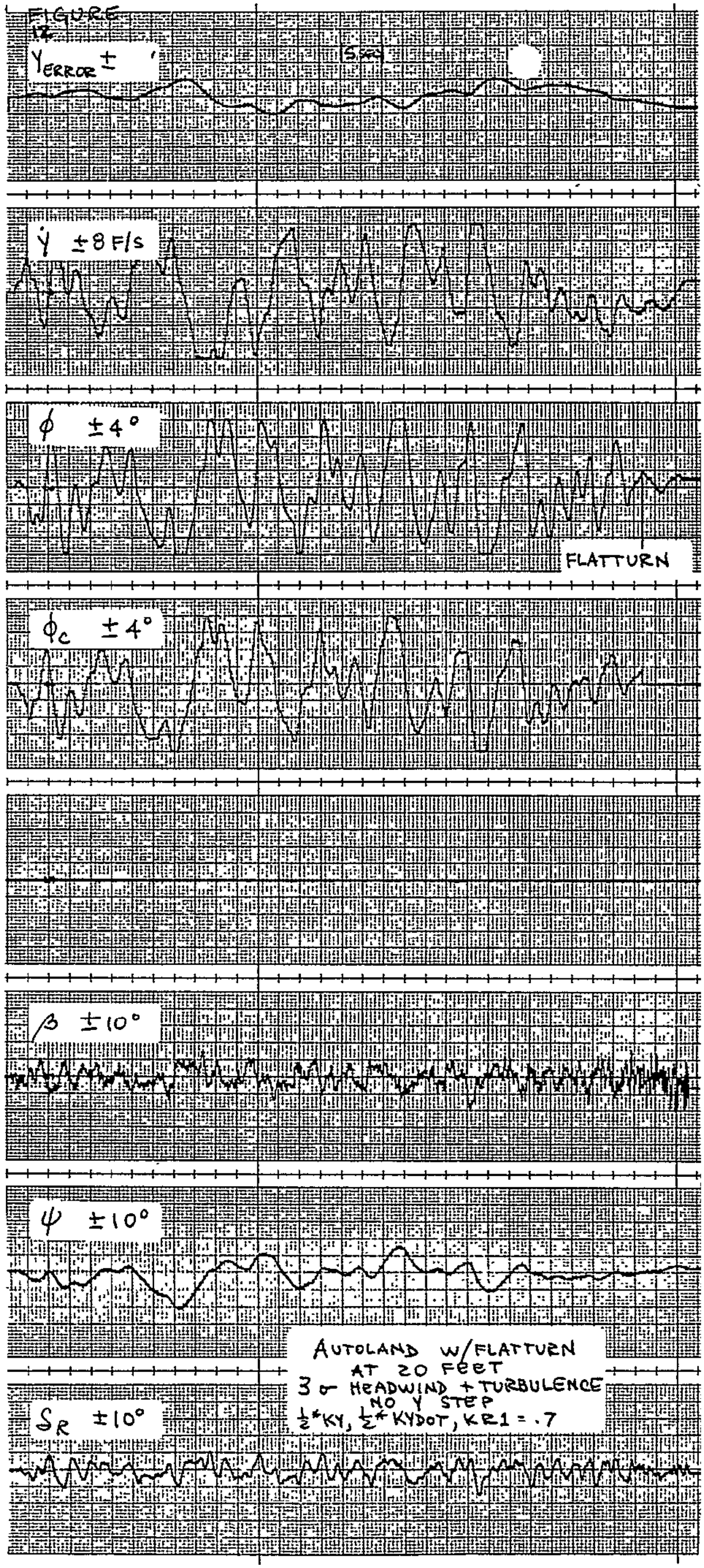


FIGURE 12
Y ERROR ±
15 sec
ẏ ±8 F/S
φ ±4°
FLATTURN
φc ±4°
β ±10°
ψ ±10°
AUTOLAND W/FLATTURN
AT 20 FEET
3σ HEADWIND + TURBULENCE
NO Y STEP
½*KY, ½*KYDOT, KR1 = .7
δR ±10°

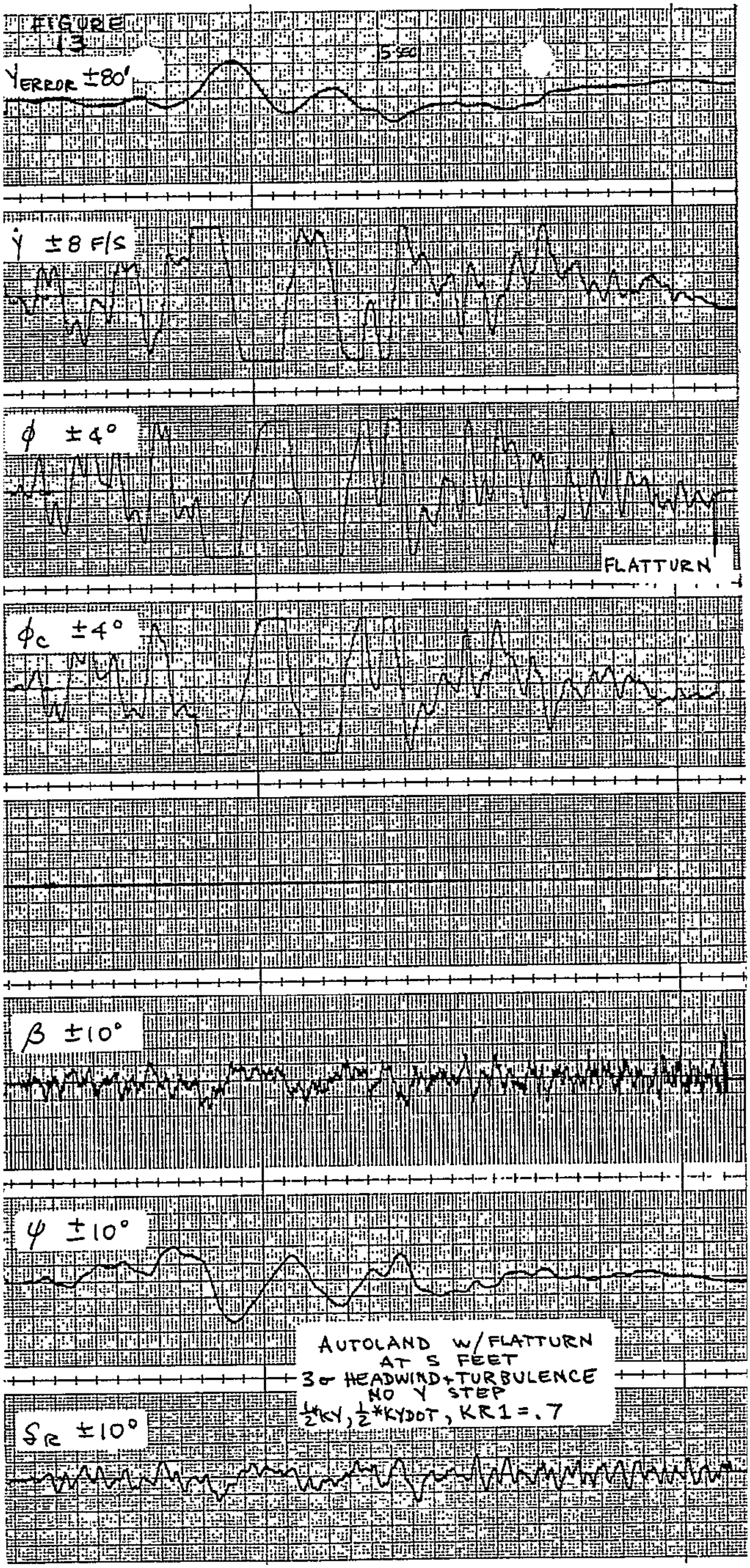

5 sec
Y ERROR ±80'
Ẏ ±8 F/S
φ ±4°
FLATTURN
φc ±4°
β ±10°
ψ ±10°
AUTOLAND W/FLATTURN
AT 5 FEET
3σ HEADWIND + TURBULENCE
NO Y STEP
½*KY, ½*KYDOT, KR1 = .7
δR ±10°

FIGURE 13

TABLE I

TOUCHDOWN PERFORMANCE IN 3σ HEADWINDS

AND TURBULENCE WITH FLAT TURN

INITIATED AT VARIOUS ALTITUDES

KY1 = .05, KYDOT = 5, KR1 = 0.7

ØK PAYLOAD, FORWARD CG 25 RUNS

Flat Turn Gear Altitude (Ft)		Y (Ft)	ψ (Vehicle Heading WRT Runway Centerline) (Deg)	φ (Deg)
0	Mean	- 1	.23	0
	Std. Dev.	12	.61	.64
	Max	23	1.34	1.23
	Min	-21	-1.03	-1
5	Mean	- 7	.33	.04
	Std. Dev.	12	.68	.22
	Max	13	1.4	.54
	Min	-27	- .83	- .34
20	Mean	- 3	.29	- .03
	Std. Dev.	9	.89	.42
	Max	16	1.97	.62
	Min	-31	-1.6	- .93

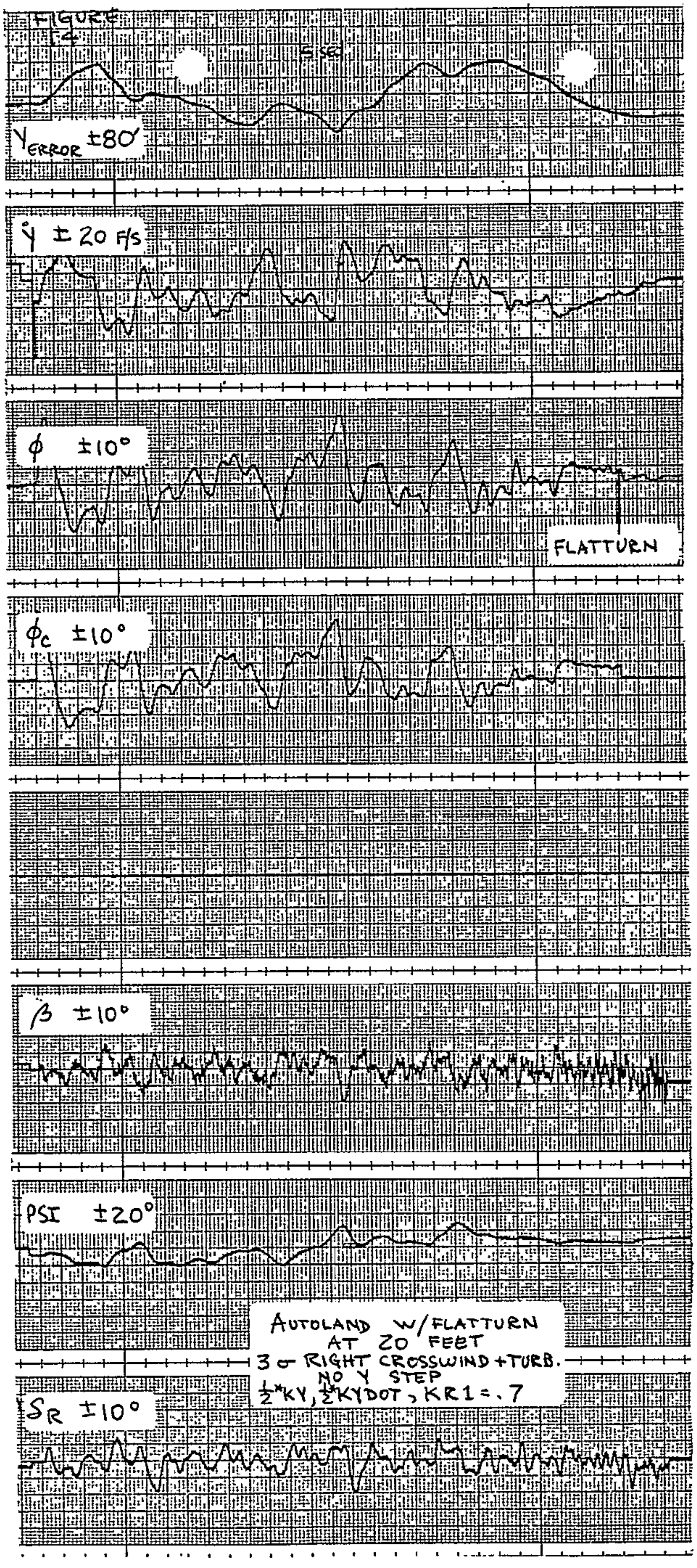


FIGURE 14
5 SEC
Y_{ERROR} ±80'
$\dot{y}$ ± 20 F/S
ϕ ±10°
FLATTURN
ϕ_c ±10°
β ±10°
PSI ±20°
AUTOLAND W/FLATTURN AT 20 FEET
3 σ RIGHT CROSSWIND + TURB.
NO Y STEP
½*KY, ½*KYDOT, KR1 = .7
δ_R ±10°

TABLE II

TOUCHDOWN PERFORMANCE IN
3σ RIGHT CROSSWINDS AND TURBULENCE
WITH FLAT TURN INITIATED AT
VARIOUS ALTITUDES
KY1 = .05, KYDOT = 5, KR1 = 0.7
ØK PAYLOAD, FORWARD CG - 25 RUNS

Flat Turn Gear Altitude (Ft)		Y (Ft)	ψ (Vehicle Heading WRT Runway Centerline) (Deg)	ϕ (Deg)
0	Mean	-11	5.37	.91
	Std. Dev.	10	.6	.7
	Max	11	6.75	2.21
	Min	-28	4.53	0
5	Mean	-13	6.16	.26
	Std. Dev	14	1.07	.46
	Max	18	8.31	1.15
	Min	-46	3.7	- .86
20	Mean	-12	5.87	.15
	Std. Dev.	11	1.07	.4
	Max	11	8.98	1.01
	Min	-48	4.28	- .65

SECTION 6

GUIDANCE MODE SWITCHING STUDY

SPERRY AUTOLAND MEMO No: 6

DATE: 17 Mar 76

COPY: ________

NASA SHUTTLE

AUTOLAND SUPPORT PROGRAM

TASK NO: 9

TASK NAME: Guidance Mode Switching Study

TASK DESCRIPTION: This task reviews the Approach and Landing Moding and Sequencing and recommends the necessary software modifications to guarantee that the guidance provide the crucial pullup command at H_{EST} = H_PULLUP. In addition other facets of Approach and Landing control and guidance moding, phasing, sequencing, and switching have been reviewed and recommendations for system modifications delineated along with the justification for same.

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DISCUSSION: A study was made of the Approach and Landing mode sequencing and selection to evaluate the interdependence and interrelationship among the various guidance phases of the Approach and Landing major mode. In addition, the study was extended to consider certain flight control system mode/phase combinations in order to evaluate their logical implementation with respect to the Approach and Landing Guidance Mode. The primary objectives of the study were twofold in nature.

1. To make certain, in the case of the guidance system itself, that failure to properly sequence while in either the automatic (Autoland) or manual mode would not result in the orbiter and crew being exposed to a potentially unsafe environment.

2. To recommend, based upon a basic understanding of the system and previous Sperry applicable experience, possible alternatives to some of the presently implemented flight control/guidance mode sequencing and phasing logic including override considerations.

This study is concerned with the Orbiter Flight Tests (OFT) and the Approach and Landing Tests (ALT). Reference was made to the following FSSRs.

a) OFT Guidance FSSR SD 76-SH-0001 dated 21 Nov 75

b) OFT Flight Control FSSR SD 76-SH-0007 dated 21 Nov 75

c) ALT Guidance FSSR SD 74-SH-0273B Rev. B dated 15 July 75

d) ALT Flight Control FSSR SD 74-SH-02714A

However, the general considerations and recommendations delineated in the memo are intended to apply to the operational shuttle vehicle as well.

For convenience, Table 1 summarizes the TAEM and Approach and Landing Guidance phases for OFT and ALT. Each area where some specific system modification is recommended is discussed separately below and justification for the recommended change provided.

ITEM 1

Modification: Pullup Command at Pullup Altitude

Include the following statements as part of the Approach and Landing Major Mode phase sequencing logic.

1) at H_{EST} = H-PULLUP set P-MODE = 3.
2) Call subroutine H-GEAR (computation of gear deployment altitude). In addition, allow the LAT-MODE phasing to occur per its present sequencing, independent of the P-MODE phasing.

Justification:

The incorporation of this logic will guarantee that the flare maneuver will occur whether or not the normal sequencing from P-MODE = 1 to 2 has or has not occurred. The present implementation necessitates that a prior P-MODE = 2 state (steep glide slope) must have existed when passing through H-PULLUP before the flare or pullup maneuver will occur and its command displayed on the pitch flight director. This can impose a potentially unsafe condition to the orbiter and crew when it is recognized that at the normal pullup altitude of approximately 1910 ft. the orbiter is on a -24 deg glide slope with a descent rate of about -210 ft/sec. It is readily possible to construct a set of conditions, particularly in manual flight, where the necessary criteria may not have been satisfied for the system to have engaged steep glide slope (P-MODE = 2) prior to reaching the pullup altitude. Hence automatic pullup will not occur if in Autoland nor would the appropriate flight director command be displayed if in manual.

The computation of the gear deployment altitude is normally called during the last pass through steep, P-MODE = 2. Hence if the P-MODE = 2 state is never achieved, then computation of H-GEAR should be made during the first pass through flare after P-MODE is set equal to 3.

Except for the incorporation of integral control of Y_{error} for steep glide slope, the lateral control law for LAT-MODE = 1, Trajectory Capture, and LAT-MODE = 2, Steep Glide Slope, are identical. Forcing the system to assume a LAT-MODE = 2 state simultaneous with forcing the P-MODE to a value of 3 with a large Y offset existing can result in a large Y overshoot of the runway centerline. Hence it is recommended that a natural progression should be permitted between LAT-MODE - 1 and LAT-MODE = 2 per the criteria specified in the Guidance FSSR.

ITEM 2

<u>Modification</u>: <u>Automatic Transition to Approach and Landing</u>

Make the exiting from the TAEM Major Mode result in an <u>Automatic Transition</u> to the Approach and Landing Major Mode so long as the necessary criteria for transferring between TAEM Pre-final and Trajectory Capture are met.

<u>Justification</u>:

The OFT Guidance FSSR (pgs 3-1 and 3-5) indicates that it is intended to carry out such a procedure from the Entry Major Mode through Approach and Landing. The specific statement to this effect of pg. 3-1 is however contradicted by the conclusion to be drawn from an examination of Figure 3.1-3, pg. 3-4. The implication here is that Automatic Approach and Landing (Autoland) can only be engaged by operation of the Keyboard "PRO" button. This in turn is only possible providing the CRT presents the crew with the appropriate message implying that the transition criteria between TAEM Pre-final and Approach and Landing Trajectory Capture have been satisfied. Otherwise the crew must type in the appropriate message using the keyboard. If the latter does constitute the normal procedure for initiating the Approach and Landing Major Mode, then this approach is not the optimum one to employ. The tracking of the steep glide slope while managing orbiter energy, plus the subsequent pull-up and final flare maneuvers are not simple tasks for the crew to carry out. This is especially true when they have been exposed to prolonged periods of weightlessness in excess of 7 days, for example, and their operating efficiency has thereby been degraded. Under extreme hazards that result in crew incapacitation, for whatever reason, it is obviously desirable to have the automatic mode as the primary mode as well. Hence the recommendation is to make the Entry, TAEM, and Approach and Landing Guidance fully automatic along with the transitioning associated between these Major Modes. The crew would have the capability to override the automatic system and revert to manual control under any circumstances they deemed appropriate.

ITEM 2A

Modification: Eliminate I-PHASE = 3, TAEM PREFINAL

Transition automatically to Approach and Landing, P-MODE = 1, TRAJECTORY CAPTURE, from TAEM I-PHASE = 2, Heading Alignment. Transition criteria would include the present requirements for phasing between TAEM Heading Alignment and TAEM Pre-final in addition to imposing some limits on Y_{error}, range and altitude, e.g. $H \leq 15{,}000$ ft, $X_{THRESH} \geq -40{,}000$ ft. The latter considerations, including Y_{error} constraints would naturally recognize the limits of MSBLS acquisition capabilities. A further modification would include automatic switch-over to the CRT FINAL APPROACH DISPLAY (FAD) upon selection of Approach and Landing P-MODE = 1.

Justification:

The present system which defines a TAEM prefinal phase and an Approach and Landing Trajectory Capture phase is essentially an artificial separation between two modes which have the same objective, namely to guide the vehicle to the Approach and Landing Steep Glideslope at a $\bar{q}$ of approximately 300 psf. While differences presently do exist in some of the control loop gains and limits between PREFINAL and TRAJECTORY CAPTURE, a single set of control laws and fixed system gains should be adequate in this flight regime. Appropriate limits on H and Y error would also be selected to guarantee smooth and satisfactory rates of zeroing out H, Y and $\bar{q}$ errors in the process of capturing the steep glide slope. Mode changing to steep tracking $\begin{pmatrix} \text{P-MODE} = 2 \\ \text{LAT-MODE} = 2 \end{pmatrix}$ where the only control law change would be associated with switching in the path integrators would occur according to the presently defined criteria. The net result would be a marked simplification in the shuttle guidance, a reduction in software requirements and a somewhat longer time period associated with capturing and tracking the steep glide slope. The latter, besides being a desirable objective in itself, would also extend the probability of the orbiter always sequencing from P-MODE = 1 to P-MODE = 2 before arriving at the pullup altitude. Hence for most situations a sequencing to P-MODE = 3 would occur naturally at H-PULLUP, resulting in the proper pullup maneuver being initiated and the command appropriately displayed on the flight director.

The FAD display during Approach and Landing provides both a horizontal and vertical situation display. The display for $X_{THRESH} \geq -40{,}000$ ft. should have adequate sensitivity to display the largest anticipated H and Y errors. The FAD will thereby provide a desirable positive indication in the automatic mode that the orbiter has progressed into the final Approach and Landing phase of flight. The crew will retain the ability to manually select any display at any time.

ITEM 3

<u>Modification</u>: <u>Steep Capture at 5000 Feet</u>

Having made the transitioning between Major Modes automatic per ITEM 2, at H_{EST} = 5000 ft. force the Approach and Landing Major Mode into the Trajectory Capture phase if the P-MODE = 1 phasing has not occurred automatically by the time this altitude has been reached.

<u>Justification</u>:

This is a logical extension of ITEM 2 whereby automatic transitioning between Entry, TAEM and then to Approach and Landing will tend to guarantee that if Trajectory Capture has not occurred by the time the orbiter has descended to 5000 ft, the vehicle should not fall very far outside the required capture window. At this altitude, the Autoland system should be capable of bringing the orbiter to an adequate landing. The incorporation of the modification associated with ITEM 1 will guarantee that pull-up will occur at 1910 feet even if the sequencing from P-MODE = 1 to P-MODE = 2 has not occurred by the time the orbiter has descended to this altitude.

ITEM 4

Modification: Lock-out Autoland if Below 5000 Feet and Manual Mode

Preclude transition from manual to Automatic mode below 5,000 ft.

Justification:

This recommendation is predicated on the assumption that having reverted to the manual mode during steep, it is uncertain that the altitude and γ errors will be small enough by the time H-PULLUP is attained (some 13.5 seconds later between 5000 ft. and 1910 ft.) such that adequate touchdown performance will result. This consideration naturally incorporates some margin of safety, and in the final analysis can only be firmly verified by simulator studies. Obviously some basic "window" requirements can be determined at each 1000 ft. altitude decrement, for example, which if satisfied could enable re-engagement at any time prior to H-PULLUP. This would complicate the system logic, thereby increasing software requirements. In any event, because it is crucial that the spatial pullup trajectory be followed to as high a degree of accuracy as possible in order to ensure satisfactory final flare performance, the lower limit associated with reverting to the automatic mode should under no circumstances be less than 2000 ft.

ITEM 5

Modification: Deletion of "Hot Stick" Concept

Abolish the "HOT STICK" concept of overriding the automatic mode and replace this by controlling disengagement via the Rotation Hand Controller (RHC) AUTO DISENGAGE pushbutton. Actuation of this pushbutton should result in reversion of both the pitch and roll/yaw axes to the CSS mode while retaining speed control in the automatic mode. Selection of AUTO VS. MANUAL can be controlled via normal operation of the Glare Shield Pushbutton for each axis of control.

Justification:

The present implementation of the "HOT STICK" results in Automatic disengagement of either the pitch or/and the roll axis for RHC control deflections greater than 14.4 degrees. The values correspond to roll and pitch authorities of approximately 62.5%. With no fading between CSS and AUTO provided in the control system, use of the "HOT STICK" concept will result in inordinately high transients in either axis. While it is true that faders can be incorporated to transfer control more smoothly between AUTO and MANUAL under these circumstances, this can very well defeat the prime purpose of the override which is to provide the pilot with control as quickly as possible in order to correct for some abnormal deviation from the desired flight path. Hence a more logical strategy is to use the RHC AUTO DISENGAGE pushbutton to revert to CSS while applying smooth corrective action proportional to the RHC deflection. This would also eliminate the possibility of any accidental deflection of the RHC that would result in disengagement of the AUTO mode. This type of Autopilot override is in fairly standard use in most transport and fighter airplanes today.

ITEM 6

Modification: Automatic Speed Control

Provide for the selection of automatic speed control independent of manual mode selection in pitch and roll/ yaw.

Justification:

In order to minimize pilot workload and provide more efficient energy management, the capability to retain automatic speed control independent of pitch and roll/ yaw AUTO engagement is recommended. While it may be true that manual speed control is no more difficult than the similar task using throttle in a transport airplane, the control of energy during Autoland is of shorter duration and more critical with respect to subsequent pullup, final flare and touchdown performance. Hence the ability to retain automatic speed control independent of other pitch mode selection should be implemented in the shuttle system.

ITEM 7

Modification: Automatic Gear Deployment

Modify gear deployment logic to make the gear deploy automatically at H_{EST} = H-GEAR.

Justification:

The gear is presently armed automatically at an altitude of 3000 ft. However, computation of H-GEAR results in an appropriate message to the crew to manually deploy the landing gear at the computed H-GEAR altitude. Even for a worst case automatic deployment situation corresponding to a ØK payload condition, 3σ headwind and inadvertent gear deployment at 3000 ft., the touchdown limits of $\dot{h} = -9$ ft/sec and $\theta = 16$ deg. are barely exceeded. This should not constitute an unsafe environment and its probability of occurring should be extremely low in any event. Therefore to avoid potentially hazardous landings for slow crew reaction time when the deployment message is flashed at an H-GEAR altitude minimum of 300 ft. and for the case where the crew may have become incapacitated, automatic landing gear deployment is recommended.

GUIDANCE ORBITAL FLIGHT TEST (OFT) AND APPROACH AND LANDING TEST (ALT) MAJOR MODE AND PHASING DEFINITIONS

	OFT (FSSR SD 76-SH-0007-21 Nov 75)		ALT (FSSR SD 74-SH-0273B-15 July 75)	
TAEM GUIDANCE	Major Mode 306		Major Mode 203	
	Phase		Phase	
	0	S-Turn	0	Not Considered for ALT
	1	Acquisition	1	Not Considered for ALT
	2	Heading Alignment	2	Not Considered for ALT
	3	Prefinal	3	Prefinal
APPROACH AND LANDING GUIDANCE	Major Mode 307		Major Mode 204	
	Phase (P-Mode)		Phase (P-Mode)	
	1	Trajectory Capture (Lat. Mode = 1)	1	Trajectory Capture (Lat-Mode = 1)
	2	Steep Glide Slope (Lat-Mode = 2)	2	Steep Glide Slope (Lat-Mode = 2)
	3	Flare and Shallow Glide Slope	3	Flare and Shallow Glide Slope
	4	Final Flare	4	Final Flare
	5	Touchdown	5	Touchdown
	6	Rollout		
			Major Mode 205	
			Rollout	

NOTES:

1. Other major modes for OFT not delineated.
2. The matrix of mode transitions (p 3-5 of OFT FSSR) indicates that entering the Approach and Landing Major Mode can only be accomplished via exiting from the TAEM Guidance Major Mode.

Table 1

SECTION 7

FLAT TURN STUDY - ADDITIONAL DATA

SPERRY AUTOLAND MEMO NO: 7

DATE: 16 Feb 76

COPY: ________

NASA SHUTTLE
AUTOLAND SUPPORT PROGRAM

TASK NO: 4

TASK NAME: FLATTURN STUDY-ADDITIONAL DATA

TASK DESCRIPTION: This memo provides additional information concerning FLATTURN Memo No. 5. Data is shown here for a 40-foot FLATTURN in both Tables I and II, which also reproduce data for the 0 and 20-foot FLATTURN altitudes. The parameter β (angle in the body axis X-Y plane from the body longitudinal axis to the air mass velocity vector component in the body X-Y plane) is shown in Table II for all altitudes. For Table I, β is about equal but of opposite sign to ψ (angle from the body longitudinal axis to the runway centerline) and is therefore not shown explicitly.

Regarding a question of what the FLATTURN response would be for one-half the lateral guidance gains including ½ A3_LOC, attention is directed to Figure 1 where FLATTURN is initiated at 150 feet at which point a 20-foot Y error is introduced (KY - .05, KYDOT = 10, A3_LOC =-.005, KR1 = 1.0, $KYDOT_{TOTAL}/KY = 10$). Comparing this run to Figure 5 of Autoland Memo No. 5, it can be seen that the response is slightly more oscillatory, as the ψ trace best indicates. With KR1 = 0.7, the response of Figure 2 results and can be compared to Figure 6 of Memo No. 5, which indicates the latter to be slightly better damped. Assuming that Figure 1 of this memo is marginally stable, the KR1 = 0.7 indicates a -3 db gain margin. For a -6 db margin, then KR1 = 0.5, with the resulting response in Figure 3. Better damping is seen especially in roll.

As Tables I and II indicate, however, little is gained using FLATTURN. Better justification for the inclusion of FLATTURN in Autoland must be provided before further analysis is carried out.

NOTE: Tables I and II in Autoland Memo No. 5 are incorrectly titled. With the lateral guidance implemented as shown in the July 1975 FSSR, then for one-half the lateral gains, the values should be: KY = .05, KYDOT = 10. In this way, the ratio of KYDOT to KY remains 10 (KYDOT . KY/KY = 10).

Tables I and II are correctly titled in this memo.

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TABLE I

(ADDITIONS TO MEMO NO. 5)

TOUCHDOWN PERFORMANCE IN 3 σ HEADWINDS
AND TURBULENCE WITH FLAT TURN
INITIATED AT VARIOUS ALTITUDES
KY1 = .05, KYDOT = 10, KR1 = 0.7
ØK PAYLOAD, FORWARD CG-25 RUNS

Flat Turn Gear Altitude (Ft)		Y (Ft)	$\bar{\psi}$ (Vehicle Heading WRT Runway Centerline) (Deg)	ϕ (Deg)
0	Mean	- 1	.23	0
	Std. Dev.	12	.61	.64
	Max	23	1.34	1.23
	Min	-21	-1.03	-1
20	Mean	- 3	.29	- .03
	Std. Dev.	9	.89	.42
	Max	16	1.97	.62
	Min	-31	-1.6	- .93
40	Mean	-.2	0.4	.13
	Std. Dev.	14	1.28	.47
	Max	41	2.64	.81
	Min	-31	-3.04	-.97

TABLE II

(ADDITIONS TO MEMO NO. 5)

TOUCHDOWN PERFORMANCE IN

3σ RIGHT CROSSWINDS AND TURBULENCE

WITH FLAT TURN INITIATED AT

VARIOUS ALTITUDES

KY1 = .05, KYDOT = 10, KR1 = 0.7

ØK PAYLOAD, FORWARD CG - 25 RUNS

Flat Turn Gear Altitude (Ft)		Y (Ft)	ψ (Vehicle Heading WRT Runway Centerline) (Deg)	ϕ (Deg)	β (Deg)
0	Mean	-11	5.37	.91	- .48
	Std. Dev.	10	.6	.7	2.04
	Max	11	6.75	2.21	2.75
	Min	-28	4.53	0	-4.26
20	Mean	-12	5.87	.15	- .36
	Std. Dev.	11	1.07	.4	2.33
	Max	11	8.98	1.01	4.25
	Min	-48	4.28	- .65	-3.83
40	Mean	- 5	4.98	- .17	.17
	Std. Dev.	10	.59	.41	2.07
	Max	14	6.20	.62	4.37
	Min	-30	4.0	-.91	-4.19

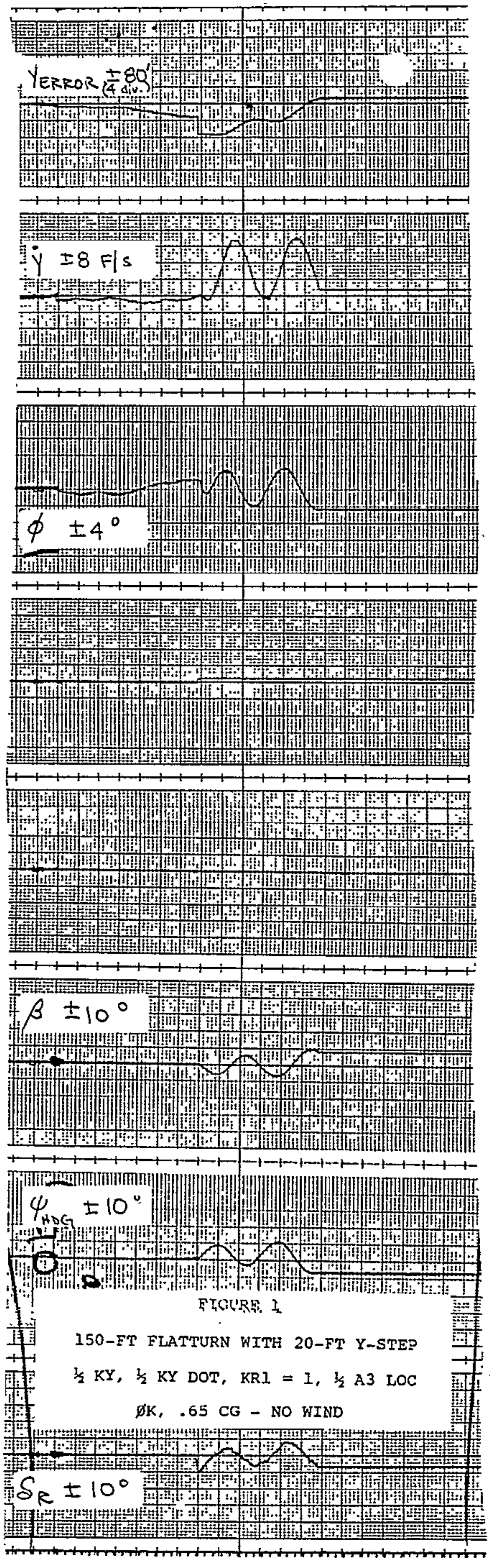

FIGURE 1

150-FT FLATTURN WITH 20-FT Y-STEP

½ KY, ½ KY DOT, KR1 = 1, ½ A3 LOC

ØK, .65 CG - NO WIND

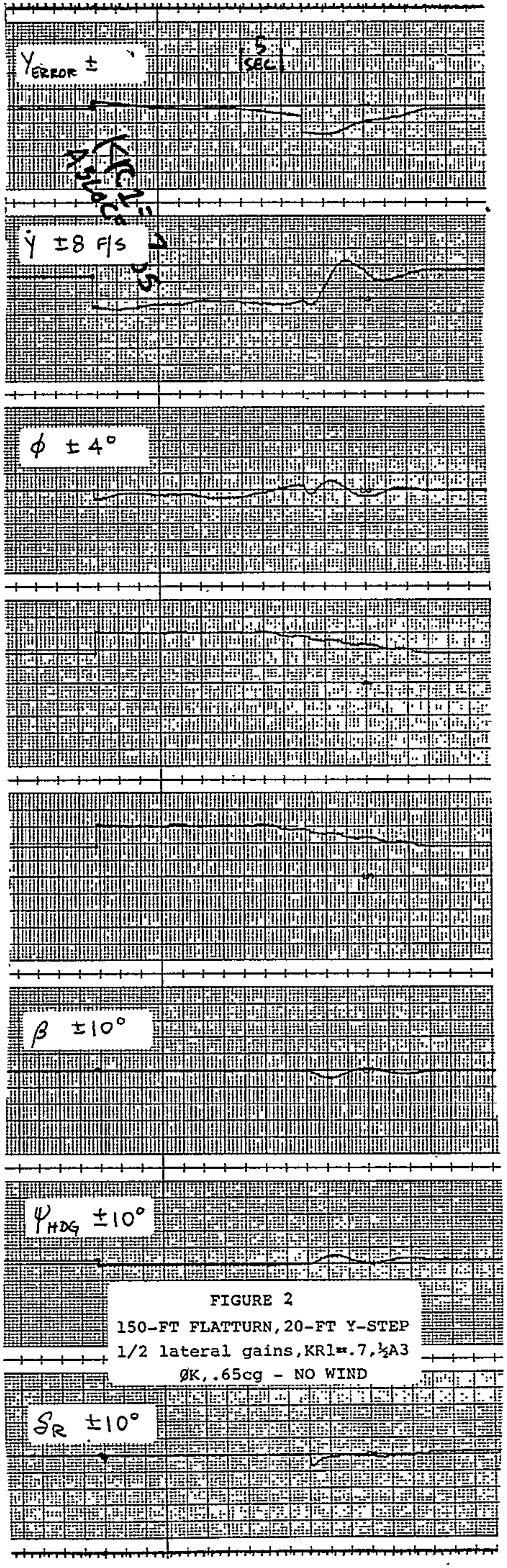

FIGURE 2
150-FT FLATTURN, 20-FT Y-STEP
1/2 lateral gains, KR1=.7, ½A3
ØK, .65cg - NO WIND

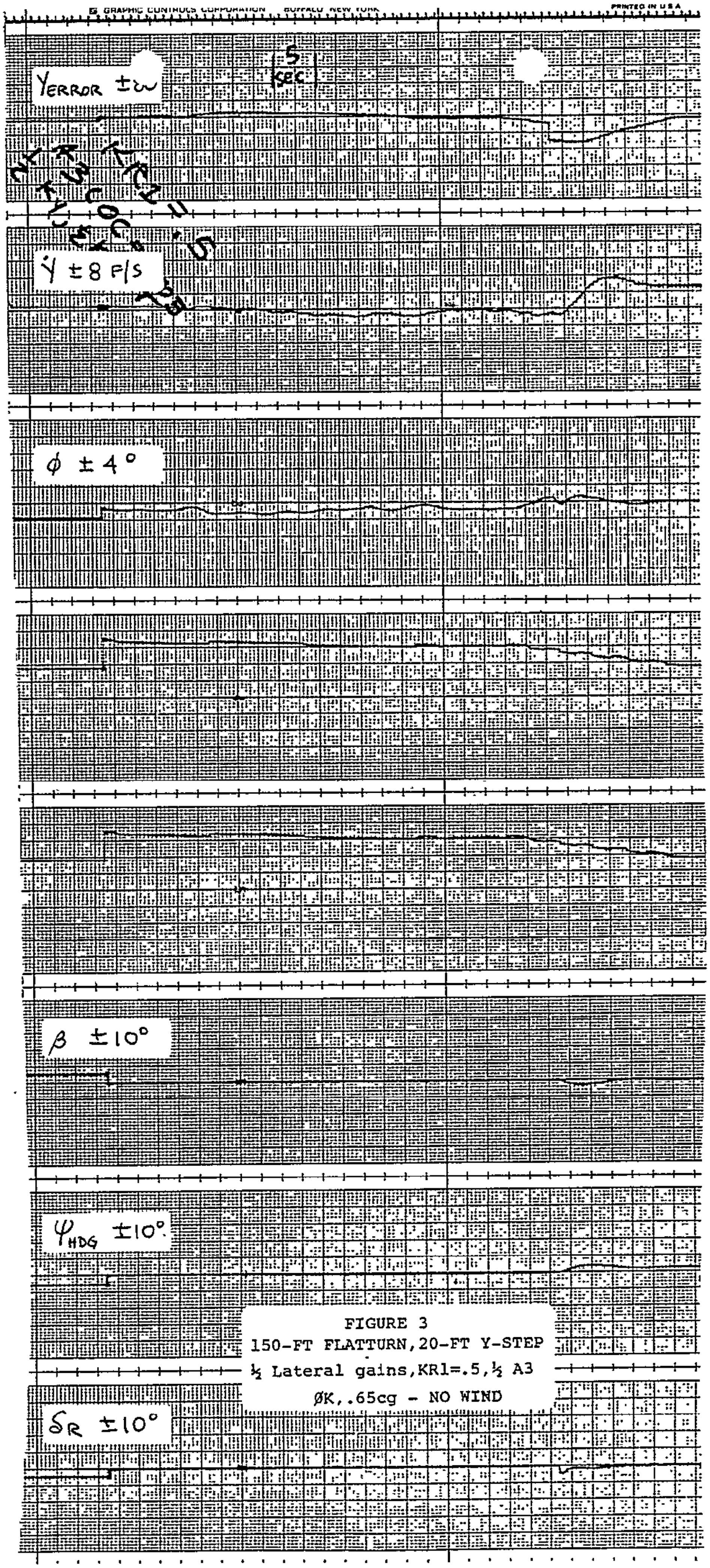

FIGURE 3
150-FT FLATTURN, 20-FT Y-STEP
½ Lateral gains, KR1=.5, ½ A3
ØK, .65cg - NO WIND

SECTION 8

TRAJECTORY SHAPING STUDY

SPERRY AUTOLAND MEMO NO: 8

DATE: 23 Feb 76

COPY: ______

NASA SHUTTLE

AUTOLAND SUPPORT PROGRAM

TASK NO: 6

TASK NAME: Trajectory Shaping

TASK DESCRIPTION: This memo describes some recent data taken for three candidate Autoland trajectories, as suggested by H. W. Hartsfield. Table I lists the three trajectories ($3/4^{\circ}$, $1\frac{1}{2}^{\circ}$ and 3° shallow glideslopes) and associated constants and gains. The gains were developed given the general requirements for a period of stabilized flight on a shallow glideslope after pullup; good pullup and shallow tracking and acceptable touchdown performance in 3 σ headwinds and tailwinds; and a pullup of 8/10-g for the $3/4^{\circ}$ and $1\frac{1}{2}^{\circ}$ glideslopes, and 1/2-g for the 3° glideslope. A 25° steep glideslope was used throughout with a speed reference of 503 fps . It has been concluded from the data to follow that the 3° shallow glideslope demonstrates the best and safest performance, especially during the flare and shallow glideslope subphase. The $3/4^{\circ}$ and $1\frac{1}{2}^{\circ}$ cases were developed and optimized to sufficient extent to allow comparison with the 3° case on an equal basis. Major gain and constant changes had to be made in pullup in the $1\frac{1}{2}^{\circ}$ and $3/4^{\circ}$ glideslopes in order to improve shallow tracking and performance in turbulence. These included increasing K_HDOT to .015 and .02 respectively, and NZ_MAX to 1.5 g. It will be shown that this higher pullup $\dot{h}$ gain of .015 should be used in the present 3° baseline system also to improve shallow glideslope tracking.

Figure 1 shows the three trajectories from 1800 feet to touchdown. It can be seen that the final flare maneuver is almost unneeded for the $3/4^{\circ}$ shallow. However, since

Continued..............

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TASK
DESCRIPTION: Cont'd

the trajectory is so near the ground at the end of pullup (within 50 feet) and the pullup itself is done at a low altitude (1260 feet) thereby increasing its criticality, navigation errors could cause the maneuver to be totally unacceptable. This will be shown in later Monte Carlo runs. It would appear that the criticality of final flare has been reduced slightly using the lowest glide-slope, while the pullup has become more critical.

Figure 2 shows altitude vs. attitude rate for the three glideslopes. For the lower glideslopes, it can be seen that the $\dot{h}$ values remain high for a longer period of time and are reduced to safe levels very late in the trajectory. For instance, at 500 feet the 3° shallow $\dot{h}$ has reduced from -225 fps to -90 fps, whereas the $1\frac{1}{2}^{\circ}$ and $3/4^{\circ}$ glideslopes are still descending at rates greater than -100 fps (-130 and -140 respectively). When the vehicle is near the ground at 200 feet, the 3° shallow trajectory has begun to stabilize on its glideslope with an $\dot{h}$ of -25 fps. However, the $1\frac{1}{2}^{\circ}$ case is still in pull-up with an $\dot{h}$ of -50 fps, while the $3/4^{\circ}$ is even faster with a -75 fps descent rate at this low altitude. It appears that a minor final flare maneuver to reduce the vehicle's $\dot{h}$ from -25 to -2 fps, as in the 3° case, has been traded for a "less critical" final flare, but with an extremely critical pullup maneuver that starts at a low altitude and must reduce the $\dot{h}$ from -225 to -10 fps. A pullup from a higher altitude to a 3° shallow with a final flare near touchdown would be less critical overall.

It should be noted that good performance was obtained by increasing the $\dot{h}$ gain (K_HDOT) in pullup from .0109 to .015 for the 3° and $1\frac{1}{2}^{\circ}$ glideslopes, and to .02 for the $3/4^{\circ}$ case. This performance is evidenced in Figures 3, 4, and 5 which plot h vs. $\dot{h}$ from 200 feet to touchdown. The period of shallow glideslope stabilization is seen in each case when a constant $\dot{h}$ is maintained prior to final flare. It is seen that this capture period is lower in altitude as the shallow glide slope is lowered.

Touchdown parameters for these cases of no wind, 3 σ headwind and 3 σ tailwind (no turbulence) are noted in Table II.

TASK
DESCRIPTION: Cont'd.

Basic performance from 12,000 feet to touchdown is shown in the time histories of Figures 6 through 14. The pullup profile for the 3° shallow case (Figure 6-8) is slightly different than previous time histories (for example, Memo No. 3 and the ØK payload "nominal air data" runs, Figures 10-12) because in this case the steep glideslope was 25°, not 24°, requiring that the pullup geometry be changed (see Table I). In fact, the 3° runs shown here perform much better on the shallow glideslope due primarily to the increased $\dot{h}$ gain in pullup/shallow glideslope. It is recommended that this gain be used for subsequent design studies. (Simulator responses to an α-gust and a step elevon were analyzed and showed improved stability. A closer look at the ratio between the h and $\dot{h}$ gains in the steep glideslope subphase (.0109/.0036 = 3) also says the pullup $\dot{h}$ gain should be increased to maintain the same ratio (.015/.0046 = 3.2). An off-line analysis confirming this increased gain should be carried out. The goal of this memo was to primarily develop 3 types of trajectories that could be compared on an equal basis.)

The recorded value of γ_T (Channel 7) on all the time histories illustrates the accurate capturing of the shallow glideslope.

Figures 15, 16 and 17 show the results of a set of 80 Monte Carlo runs, made up of forty 3σ headwind plus turbulence runs and forty 3σ tailwind plus turbulence runs with navigation errors. Touchdown results are plotted in Figure 18 for X distance, and are tabulated in Table III for other parameters. It is seen that the $3/4^{\circ}$ shallow glideslope has the largest X dispersion in the headwind case. The 3° shallow shows the largest X value of all the runs but has the smallest standard deviation. The $\dot{h}$ performance was the best for the $1\frac{1}{2}^{\circ}$ case, with the $3/4^{\circ}$ case exceeding -9 ft/sec in one run. The $\dot{h}$ means were about the same for all three glideslopes. Pitch attitude at touchdown never exceeded 14°. The maximum tire speed of 350 ft/sec was exceeded in two runs (353 and 356 ft/sec) for the lowest glideslope in tailwinds. The 3° glideslope, when compared to the others, had the smallest ground speed standard deviation overall.

TASK DESCRIPTION: Cont'd.

Figure 17 for the $3/4^{o}$ case illustrates at least three times in which the vehicle got within 5 feet of the ground 1500 feet from threshold. This never occurred with the $1\frac{1}{2}^{o}$ or 3^{o} glideslopes, although the $1\frac{1}{2}^{o}$ case skims the runway very near threshold in many cases. The 3^{o} shallow has the best record of touchdown performance.

It is noted that the NZ-MAX limit on the pullup command had to be increased to 1.5 g in order to obtain good performance for the $3/4^{o}$ and $1\frac{1}{2}^{o}$ cases. Time histories of A_Z did show intermittent peaking at or slightly above a 1.5 g incremental value during pullup in 75% of the $3/4^{o}$&$1\frac{1}{2}^{o}$ runs in turbulence. It is recommended that this limit be increased to 1.5 for the baseline (3^{o}) system also, since N_Z commands at or near this limit can occur in 3σ turbulence (but less often than with the 8/10-g pullup).

TABLE I

CONSTANTS AND GAINS USED
FOR $3/4^{\circ}$, $1\frac{1}{2}^{\circ}$ AND 3° SHALLOW GLIDESLOPES

CONSTANT OR GAIN	SHALLOW GLIDESLOPE		
	$3/4^{\circ}$	$1\frac{1}{2}^{\circ}$	3°
XZERO	5000	5000	5000 feet
GAMMA_REF_1	- 25	- 25	- 25 deg
GAMMA_REF_2	- 3/4	- $1\frac{1}{2}$	- 3 deg
X_EXP	- 4153	- 4421	-4087 feet from threshold
H_K	10553	10616	16986 feet
R	10497	10497	16795 feet
X_K	- 2796	- 2930	-1684 feet from threshold
H_FF	50	100	200 feet
H_MIN	10	30	60 feet
TAU_TD2	4.88	4.88	4.88 sec
TAU_TD1	5	5	5 sec
H_TD2_DOT	- 3	- 2	- 4 feet/sec
H_TD1-DOT	3	3	4 feet/sec
H_NO_ACC	5	5	5 feet
TAU_EXP	1.67	1.67	1.67 sec
H_DECAY	73	73	73 feet
H_FLARE	1260	1320	1980
H_CLOOP	1040	1103	1764 feet
TAU_GAMMA	2	2	2 sec
K_H	.0046	.0046	.0046
K_HDOT	.02	.015	.015
NZ_MAX	1.5	1.5	1.5 g

TABLE I

CONSTANTS AND GAINS USED
FOR 3/4°, 1½° AND 3° SHALLOW GLIDESLOPES

CONSTANT OR GAIN	SHALLOW GLIDESLOPE		
	3/4°	1½°	3°
K_FLR	.004	.004	.004
K_IFLR	0.1	0.1	0.1
K_HDOT	.0175	.0175	.0175
PULLUP KPRED SHAPING	1-sec lag	1-sec lag	1-sec lag
BODY FLAP IC	+22.5	+22.5	+22.5 deg
BODY FLAP	Ramp to +2	Ramp to +2	Ramp to +2 deg
KPIT Mach Schedule from 14.73 to 30	.53, .43	.53, .43	.53, .43
V_REF	503	503	503 feet/sec
FLATTURN	None	None	None
Landing Gear Down Altitude	FSSR Algorithm	FSSR Algorithm	FSSR Algorithm

Steep glideslope subphase gains are the same as July 15, 1975 + CR FSSR.

TABLE II

TOUCHDOWN PARAMETERS

ØK PAYLOAD, .65 CG, JUNE 75 AERODATA

Shallow G/S	Wind	$\dot{h}$ (Ft/Sec)	θ (Deg)	X (Threshold, Ft)	V_T (Ft/Sec)	V_G (Ft/Sec)	Flare Alt (Gear/Ft)
$3/4^\circ$	No Wind	-1.7	7	1927	304	304	21
	H.W.	-2.6	9	1071	278	256	10
	T.W.	-1.1	6.6	2352	311	322	18
$1\frac{1}{2}^\circ$	No Wind	-1.4	9	3006	276	276	41
	H.W.	-2	11	1598	261	238	29
	T.W.	-1.1	8.6	3629	283	294	47
3°	No Wind	-1.6	8	3306	290	290	80
	H.W.	-2.3	10	2098	265	243	58
	T.W.	-1.1	7.5	3986	290	301	94

TABLE III

TOUCHDOWN PARAMETER	SHALLOW GLIDESLOPE		HEADWIND + TURBULENCE	TAILWIND + TURBULENCE
$\dot{h}$ (Ft/Sec)	$3/4^{\circ}$	Mean	- 3.4	- 2.7
		Std. Dev.	1.6	1.5
		Max.	- 0.2	- 0.2
		Min	- 9.4	- 6.9
	$1\frac{1}{2}^{\circ}$	Mean	- 3.3	- 3
		Std. Dev.	1	1
		Max.	- 0.9	- 0.8
		Min.	- 5.4	- 5.7
	3°	Mean	- 3.3	- 3.3
		Std. Dev.	1.5	1.6
		Max.	- 0.7	- 0.4
		Min	- 7.7	- 7.2
$X_{Threshold}$ (Ft)	$3/4^{\circ}$	Mean	510	1910
		Std. Dev	460	680
		Max	1550	3020
		Min	-580	- 20
	$1\frac{1}{2}^{\circ}$	Mean	1440	3270
		Std. Dev.	320	620
		Max	2020	4350
		Min	490	1650
	3°	Mean	1930	3640
		Std. Dev.	290	360
		Max.	2400	4650
		Min	1380	2910
θ (Deg)	$3/4^{\circ}$	Mean	8.1	7
		Std. Dev.	1.1	1.1
		Max	11.6	9.9
		Min.	6.4	5.2
	$1\frac{1}{2}^{\circ}$	Mean	10.8	9
		Std. Dev.	1.2	0.9
		Max	14.1	10.6
		Min	8.3	7.1

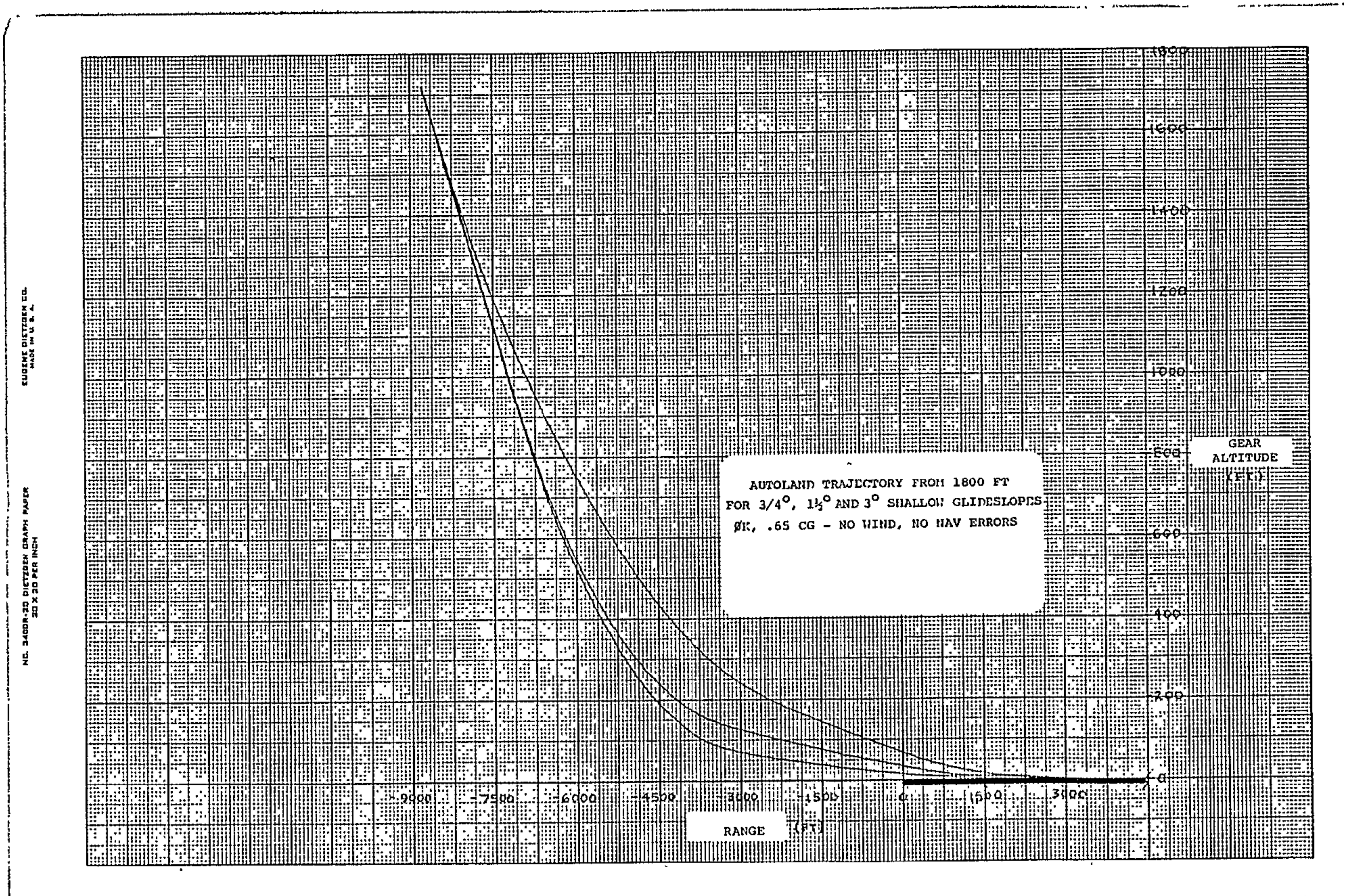
TABLE III

TOUCHDOWN PARAMETER	SHALLOW GLIDESLOPE		HEADWIND + TURBULENCE	TAILWIND + TURBULENCE
	3°	Mean	10.9	7.7
		Std. Dev.	1	0.9
		Max	13.2	10.5
		Min	9.2	6.1
V_G (Ft/Sec)	3/4°	Mean	270	321
		Std. Dev.	15	17
		Max.	302	356
		Min	235	286
	1½°	Mean	240	292
		Std. Dev.	12.5	10
		Max.	270	318
		Min	218.	274
	3°	Mean	239.6	305.7
		Std. Dev.	7	11.4
		Max	258	334
		Min	228	276

*NAV Errors (1σ):

	Elevation	±.03 deg
	Azimuth	±.05 deg
	Range	±100 feet
	Radar Altimeter	±2 feet

C-3



AUTOLAND TRAJECTORY FROM 1800 FT
FOR 3/4°, 1½° AND 3° SHALLOW GLIDESLOPES
ØK, .65 CG - NO WIND, NO NAV ERRORS
GEAR
ALTITUDE
(FT)
RANGE
(FT)

Figure 1

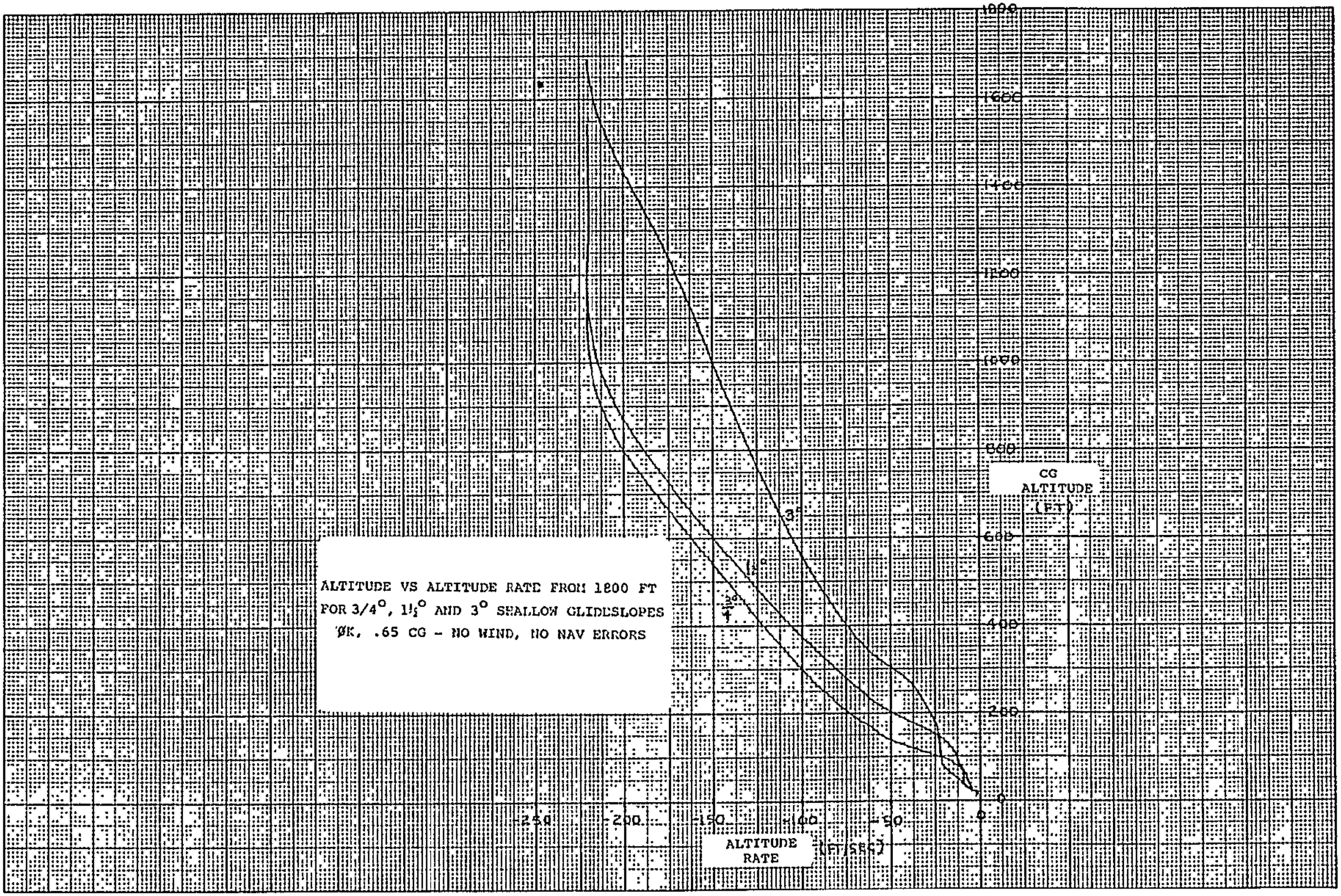

ALTITUDE VS ALTITUDE RATE FROM 1800 FT
FOR 3/4°, 1½° AND 3° SHALLOW GLIDESLOPES
ØK, .65 CG - NO WIND, NO NAV ERRORS
CG
ALTITUDE
(FT)
1800
1600
1400
1200
1000
800
600
400
200
0
ALTITUDE
RATE
(FT/SEC)
-250
-200
-150
-100
-50
0
3°
1½°
3/4°

Figure 2

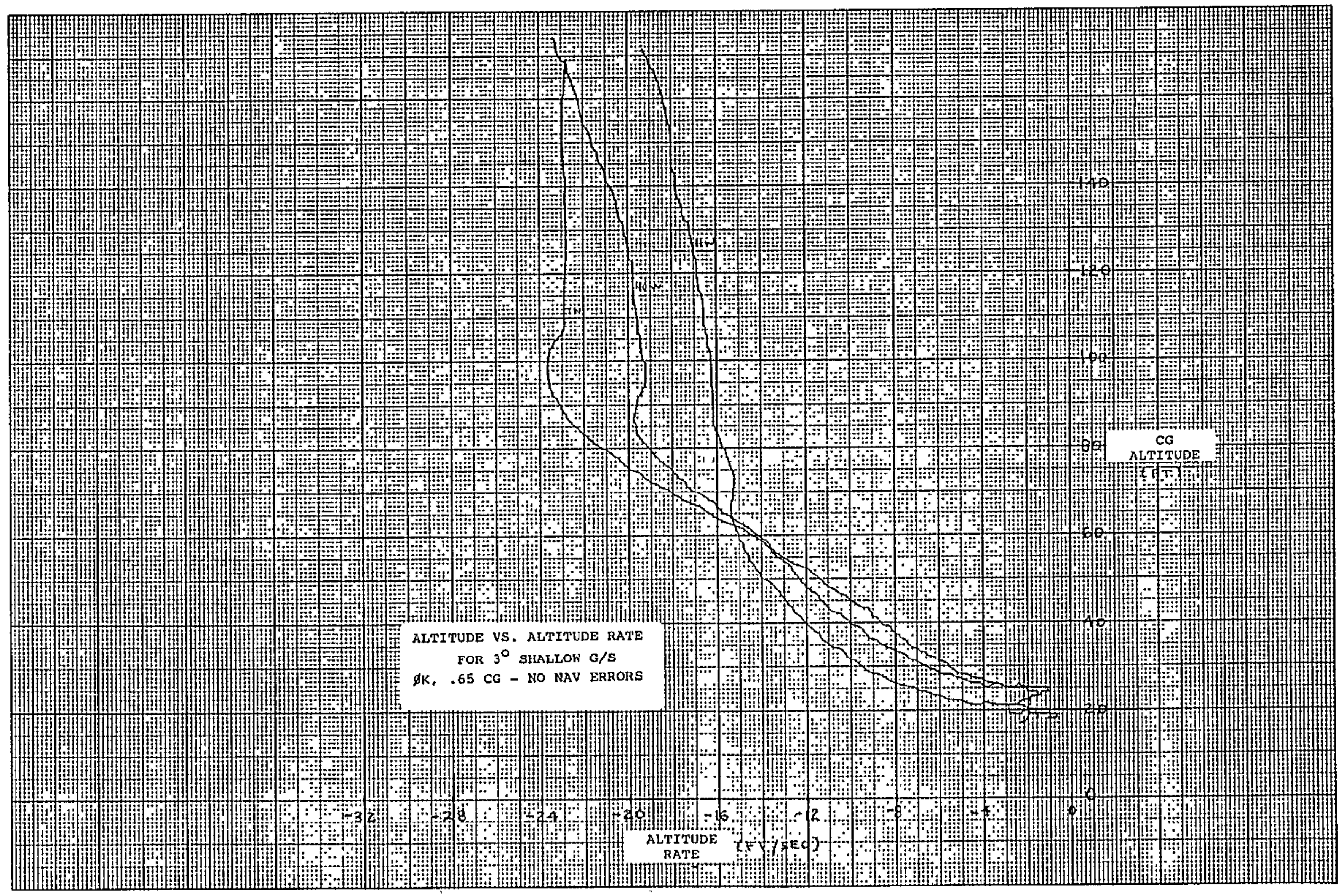
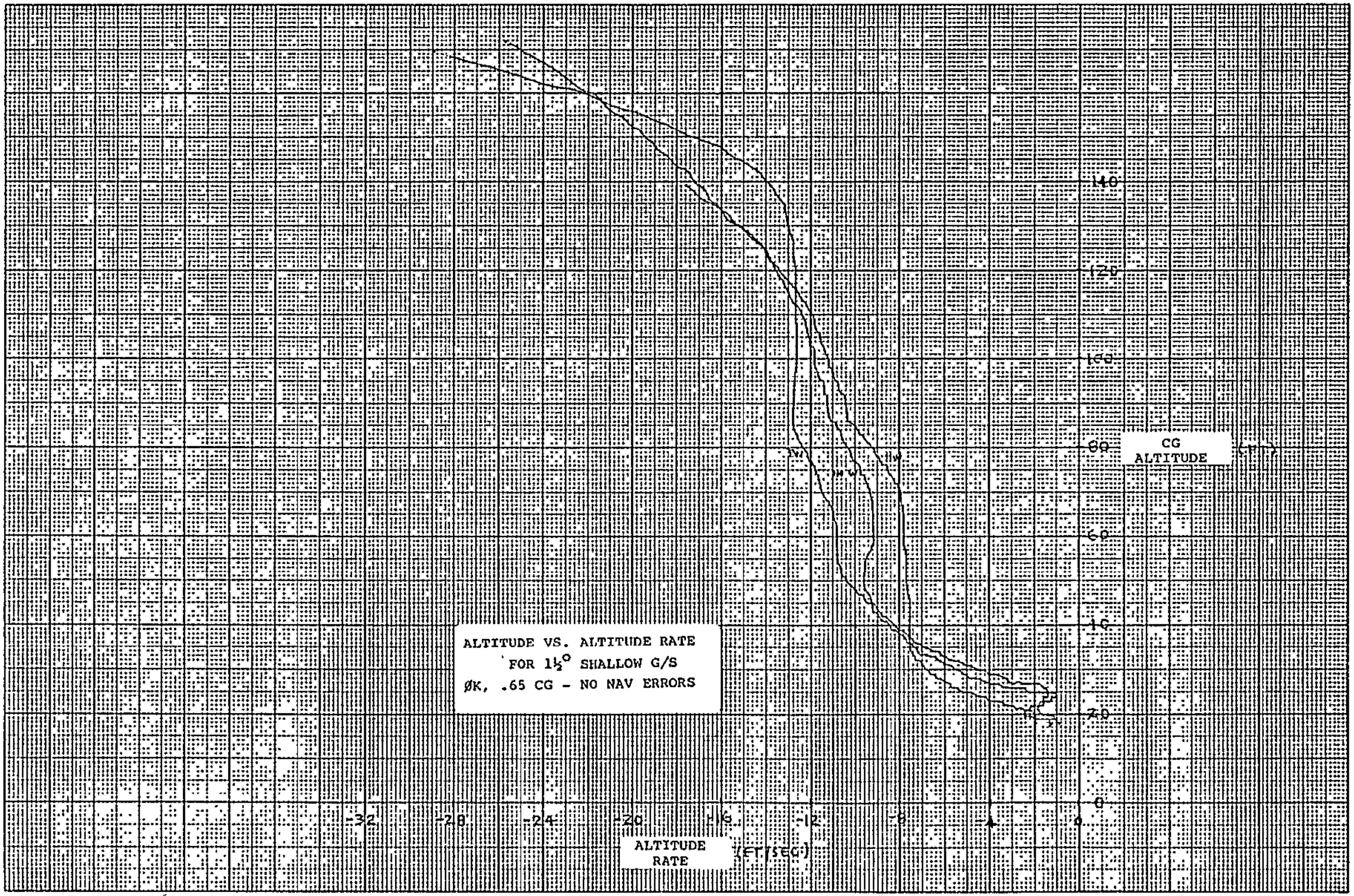
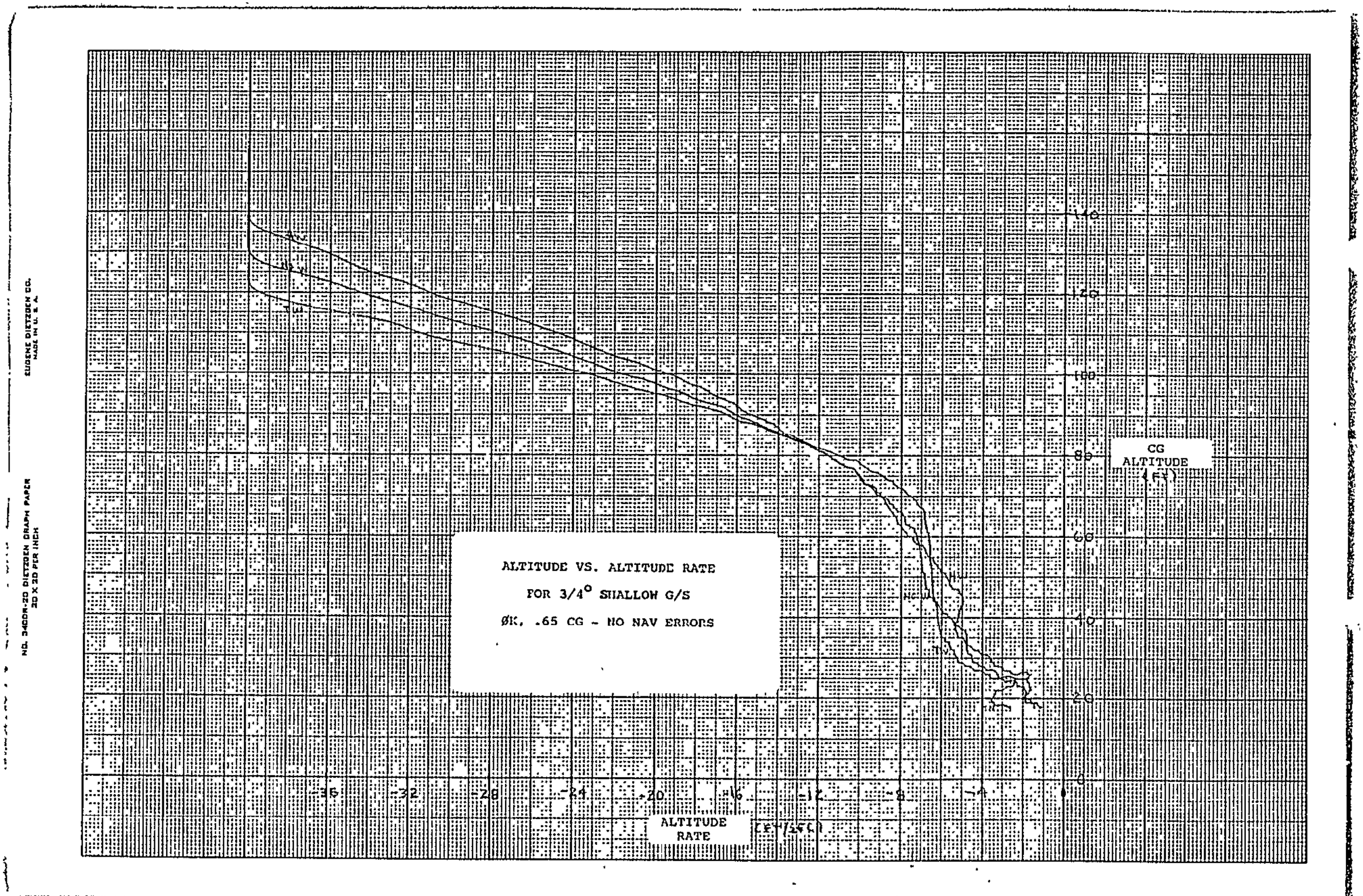

ALTITUDE VS. ALTITUDE RATE
FOR 3° SHALLOW G/S
ØK, .65 CG - NO NAV ERRORS
CG
ALTITUDE
(FT)
140
120
100
80
60
40
20
0
-32
-28
-24
-20
-16
-12
-8
-4
0
ALTITUDE
RATE
(FT/SEC)

Figure 3



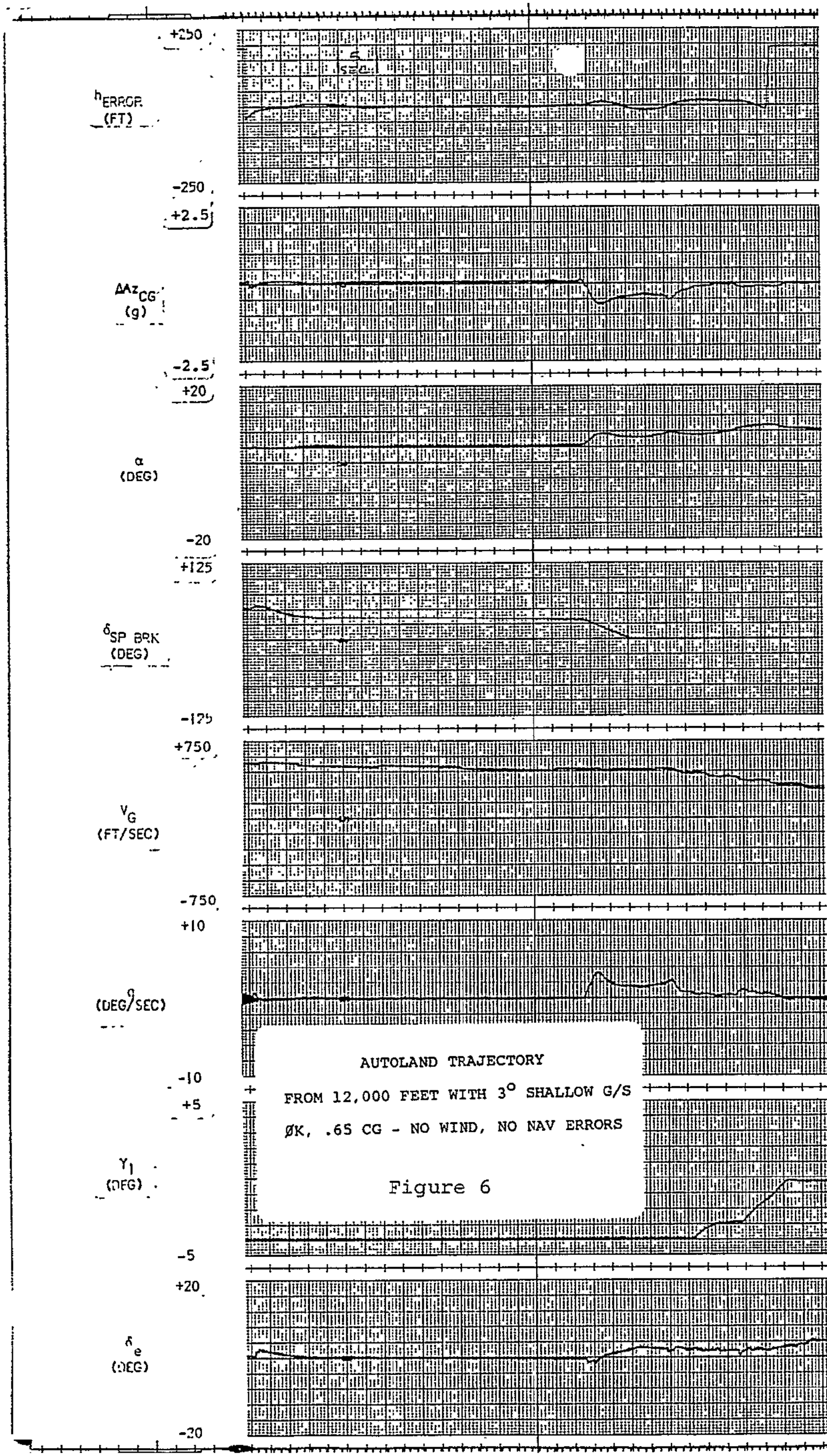
ALTITUDE VS. ALTITUDE RATE
FOR 1½° SHALLOW G/S
ØK, .65 CG - NO NAV ERRORS
CG
ALTITUDE
(FT)
140
120
100
80
60
40
20
0
-32
-28
-24
-20
-16
-12
-8
-4
0
ALTITUDE
RATE
(FT/SEC)

Figure 4



ALTITUDE VS. ALTITUDE RATE
FOR 3/4° SHALLOW G/S
ØK, .65 CG - NO NAV ERRORS
CG
ALTITUDE
(FT)
ALTITUDE
RATE
(FT/SEC)
-36
-32
-28
-24
-20
-16
-12
-8
-4
0
140
120
100
80
60
40
20
0

Figure 5

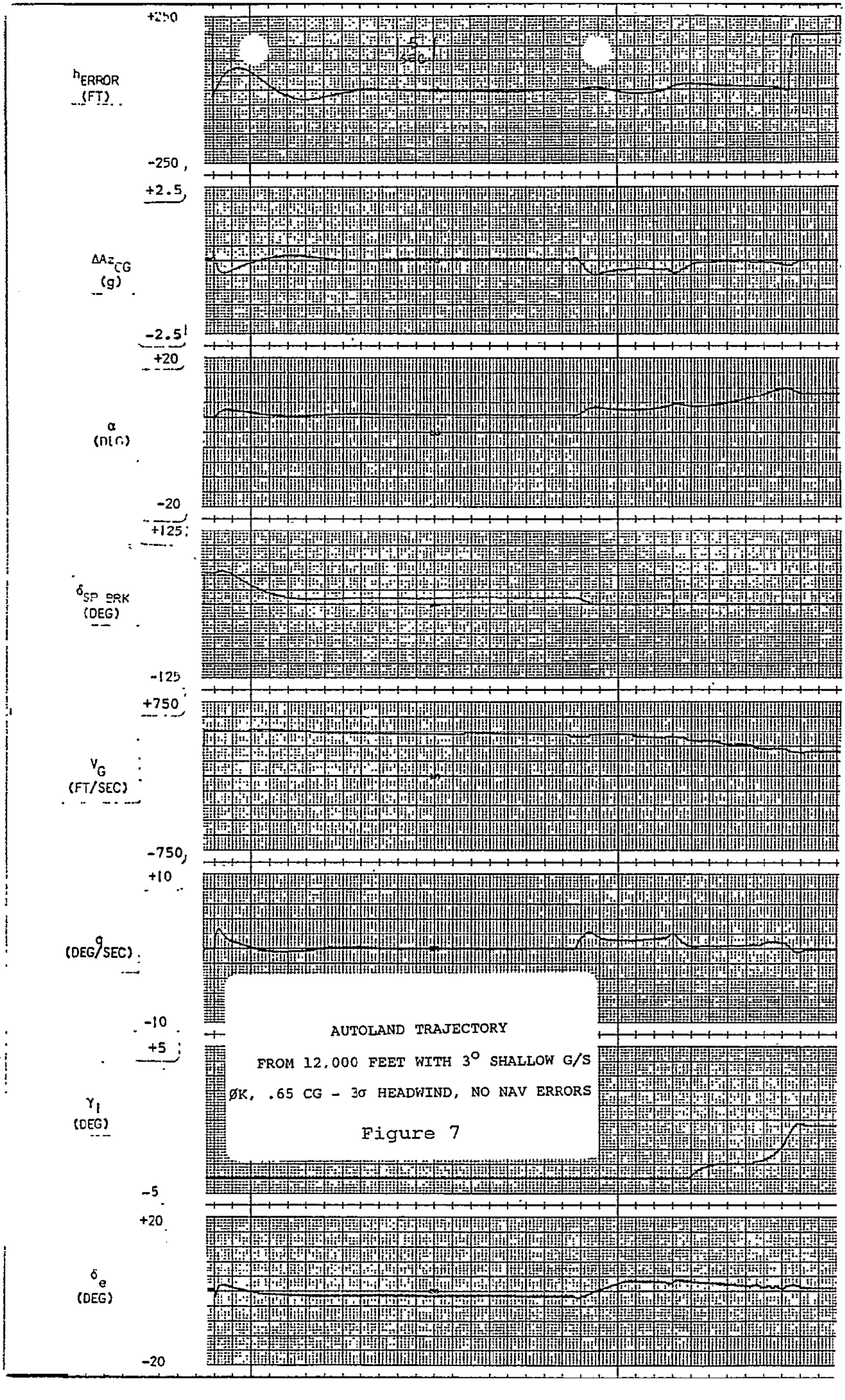


AUTOLAND TRAJECTORY

FROM 12,000 FEET WITH 3° SHALLOW G/S

ØK, .65 CG - NO WIND, NO NAV ERRORS

Figure 6

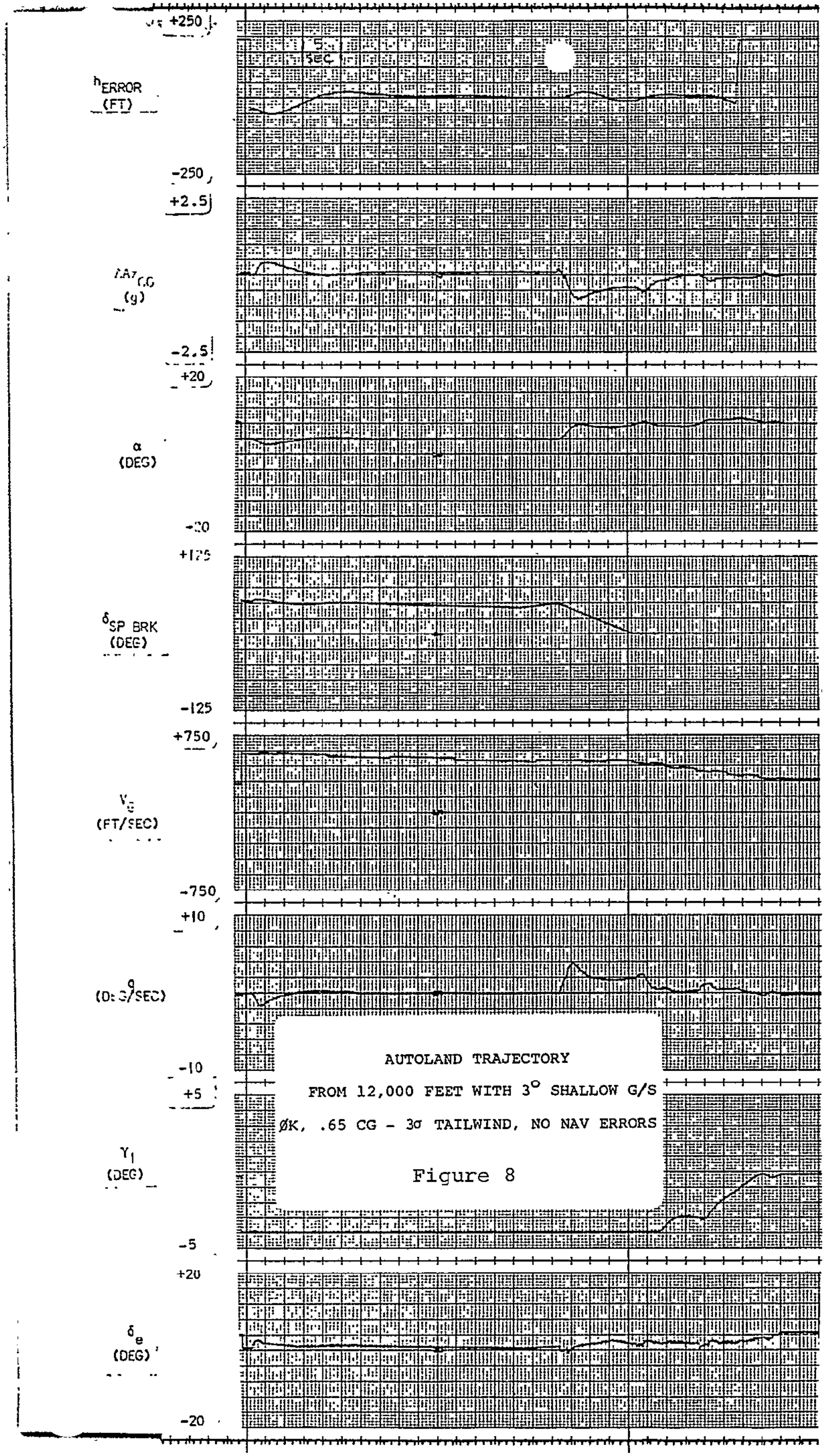


AUTOLAND TRAJECTORY

FROM 12,000 FEET WITH 3° SHALLOW G/S

ØK, .65 CG - 3σ HEADWIND, NO NAV ERRORS

Figure 7

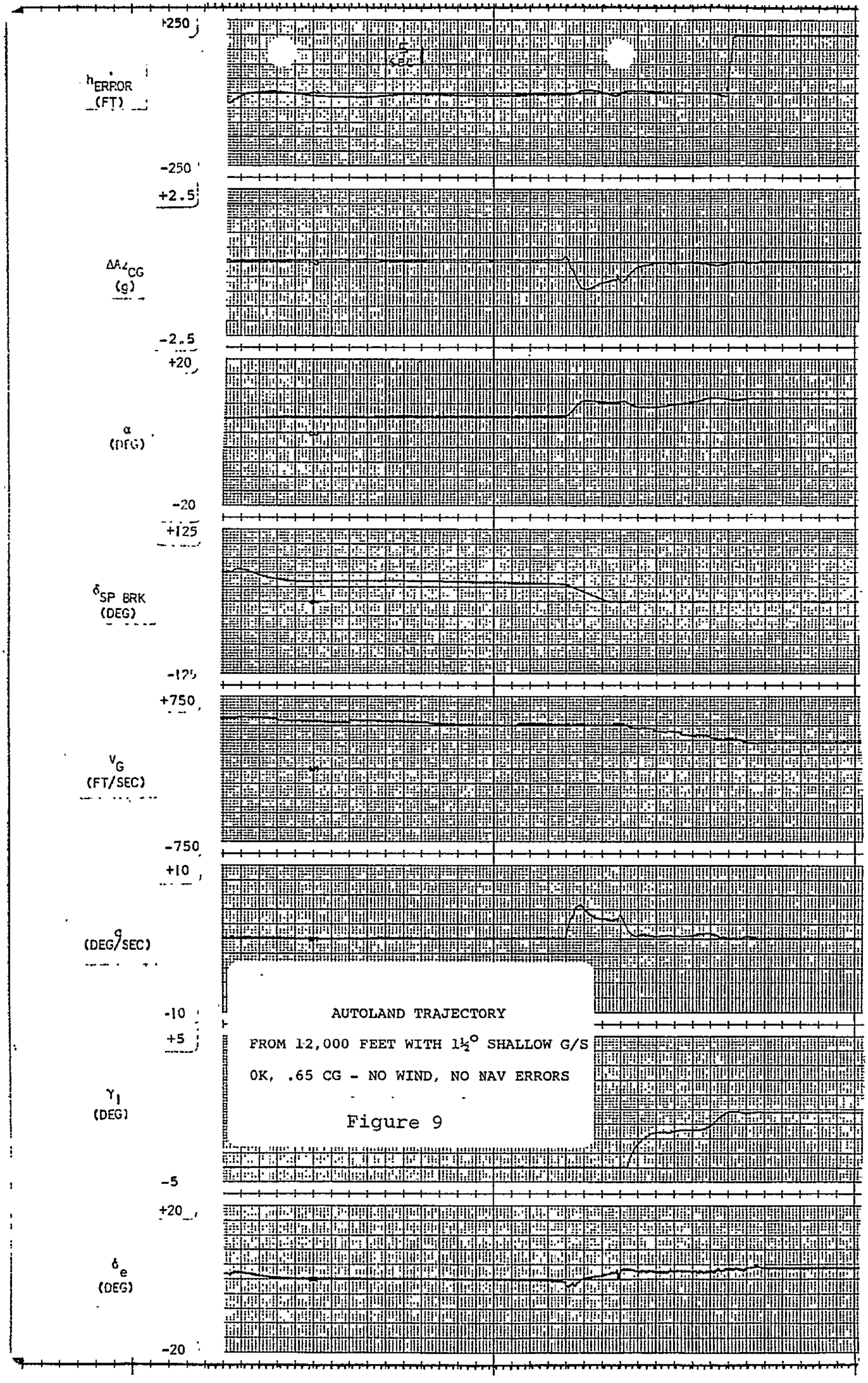


AUTOLAND TRAJECTORY

FROM 12,000 FEET WITH 3° SHALLOW G/S

ØK, .65 CG - 3σ TAILWIND, NO NAV ERRORS

Figure 8

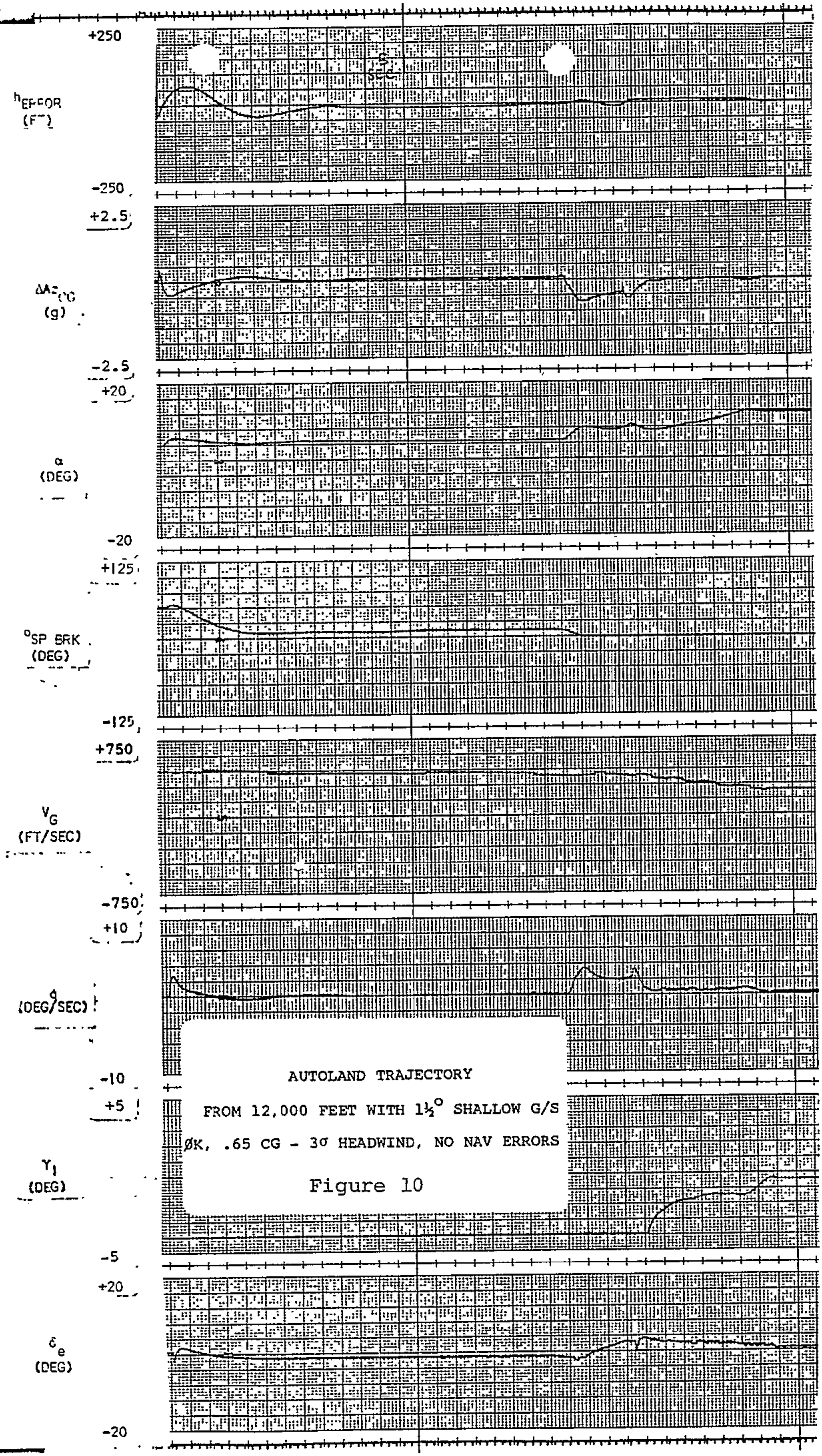


AUTOLAND TRAJECTORY

FROM 12,000 FEET WITH 1½° SHALLOW G/S

OK, .65 CG - NO WIND, NO NAV ERRORS

Figure 9

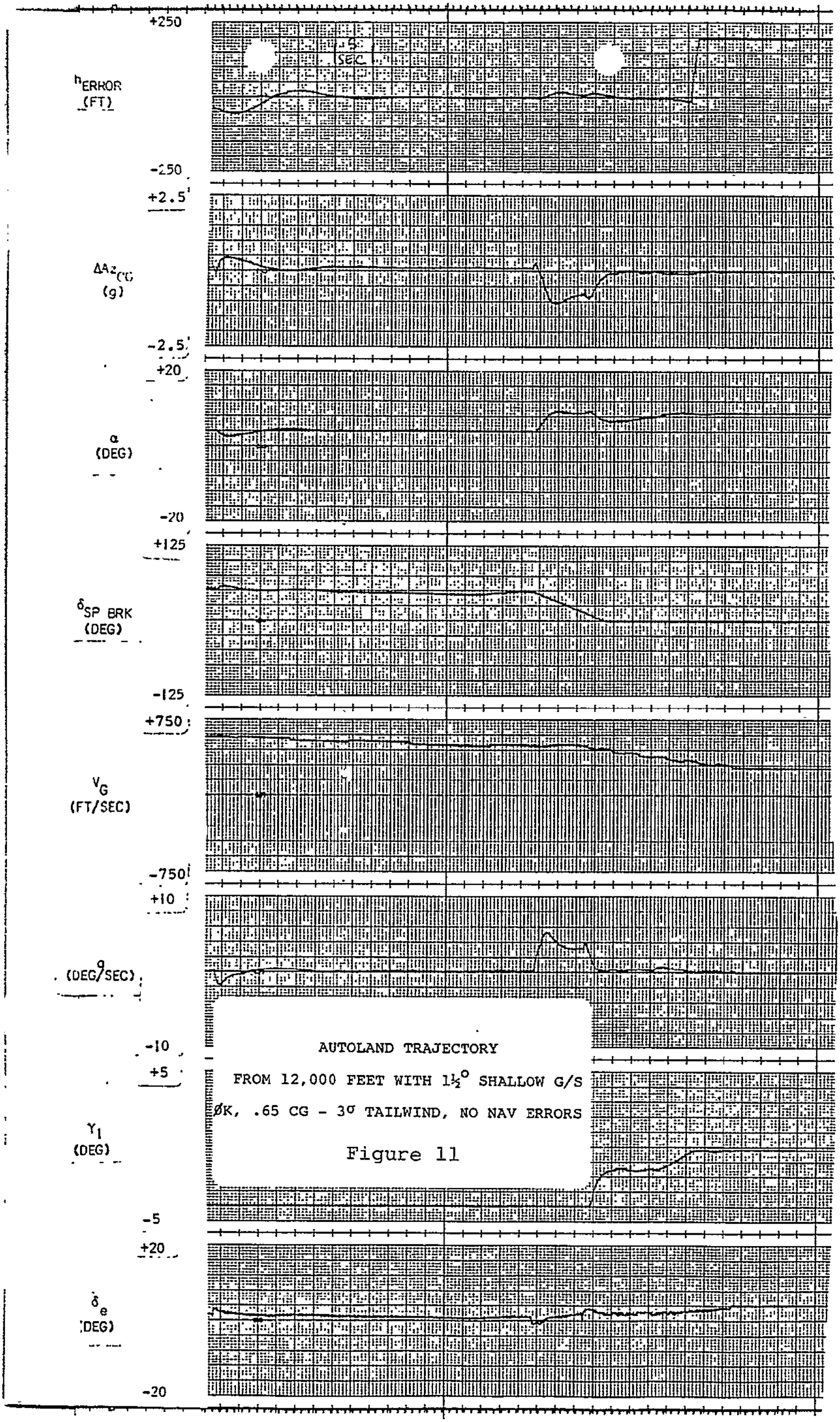


AUTOLAND TRAJECTORY

FROM 12,000 FEET WITH 1½° SHALLOW G/S

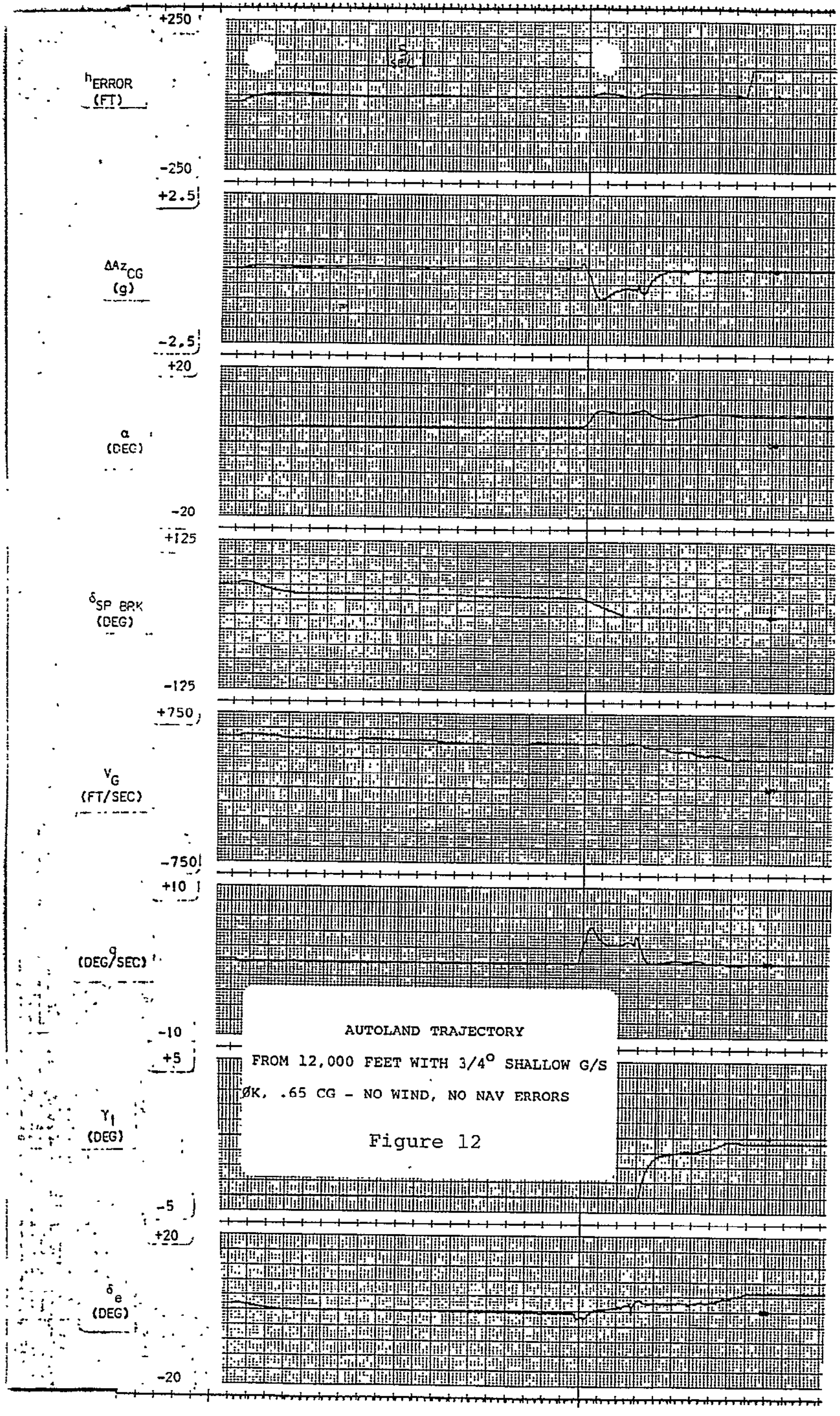
ØK, .65 CG - 3σ HEADWIND, NO NAV ERRORS

Figure 10



AUTOLAND TRAJECTORY
FROM 12,000 FEET WITH 1½° SHALLOW G/S
ØK, .65 CG - 3σ TAILWIND, NO NAV ERRORS

Figure 11

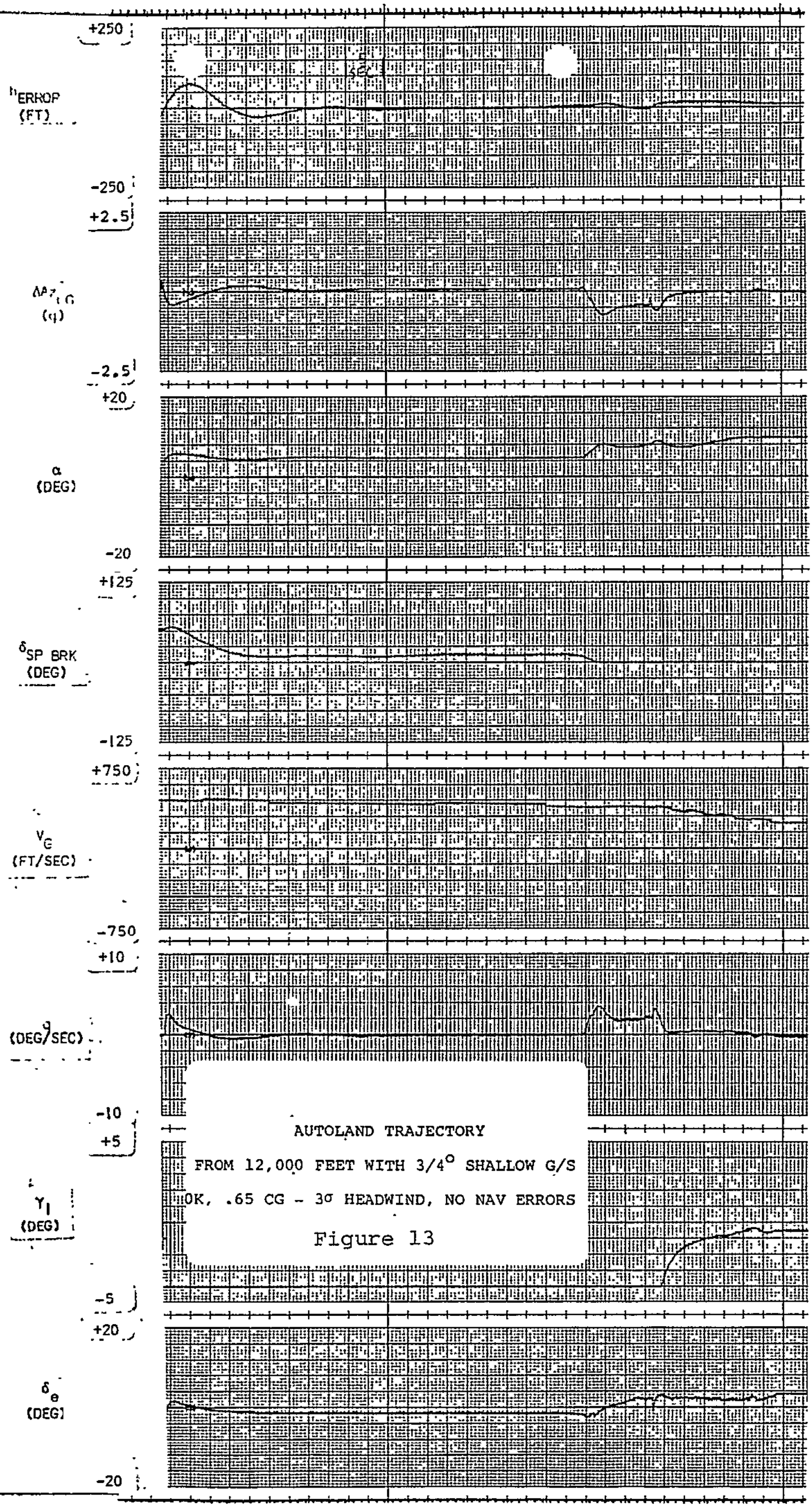


AUTOLAND TRAJECTORY

FROM 12,000 FEET WITH 3/4° SHALLOW G/S

ØK, .65 CG - NO WIND, NO NAV ERRORS

Figure 12

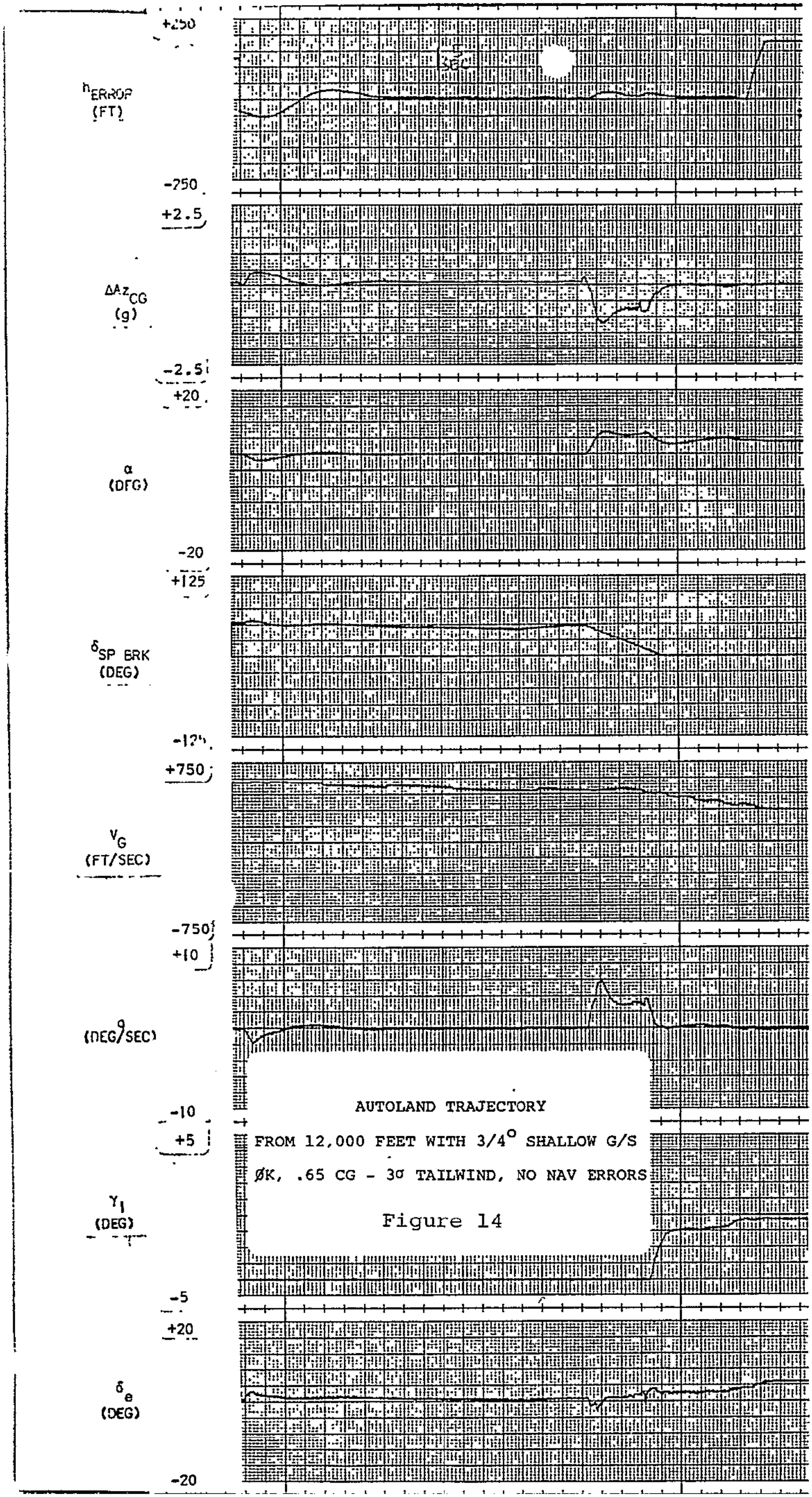


AUTOLAND TRAJECTORY

FROM 12,000 FEET WITH 3/4° SHALLOW G/S

0K, .65 CG - 3σ HEADWIND, NO NAV ERRORS

Figure 13



AUTOLAND TRAJECTORY

FROM 12,000 FEET WITH 3/4° SHALLOW G/S

ØK, .65 CG - 3σ TAILWIND, NO NAV ERRORS

Figure 14

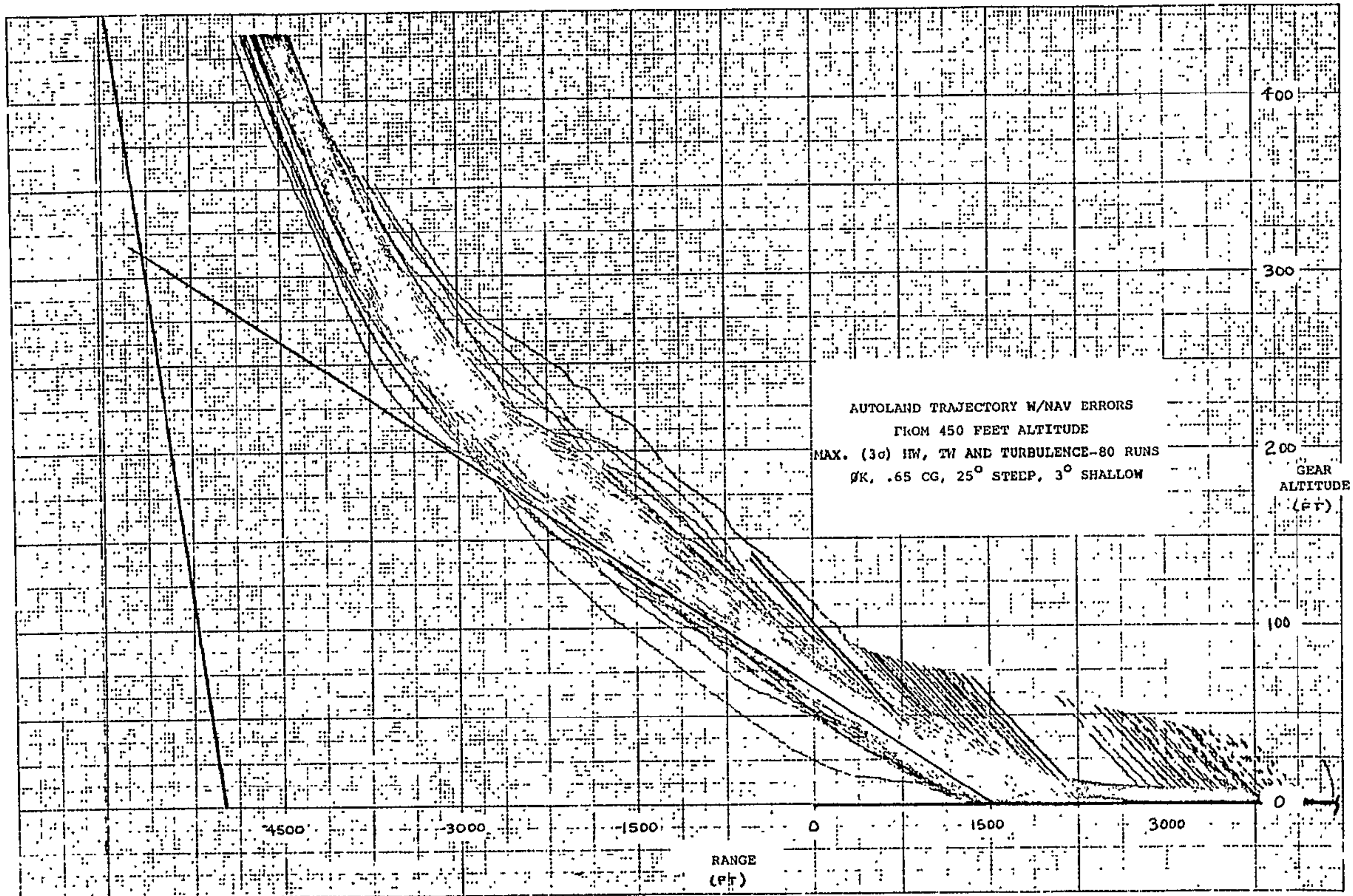

AUTOLAND TRAJECTORY W/NAV ERRORS
FROM 450 FEET ALTITUDE
MAX. (3σ) HW, TW AND TURBULENCE-80 RUNS
ØK, .65 CG, 25° STEEP, 3° SHALLOW
400
300
200
100
0
GEAR ALTITUDE (FT)
-4500
-3000
-1500
0
1500
3000
RANGE (FT)

Figure 15

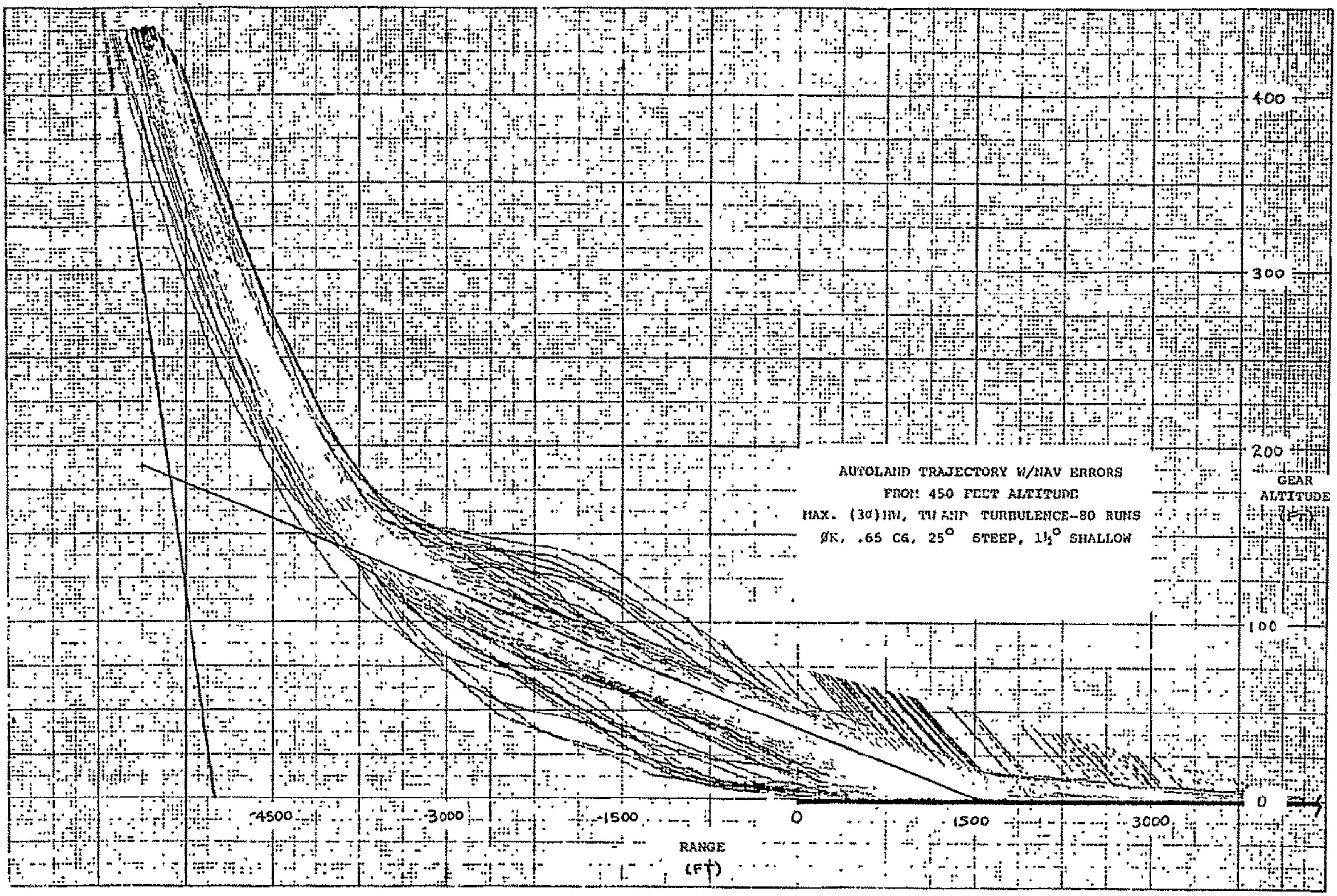

AUTOLAND TRAJECTORY W/NAV ERRORS
FROM 450 FEET ALTITUDE
MAX. (3σ)HW, TW AND TURBULENCE-80 RUNS
ØK, .65 CG, 25° STEEP, 1½° SHALLOW
400
300
200
100
0
GEAR
ALTITUDE
(FT)
-4500
-3000
-1500
0
1500
3000
RANGE
(FT)

Figure 16

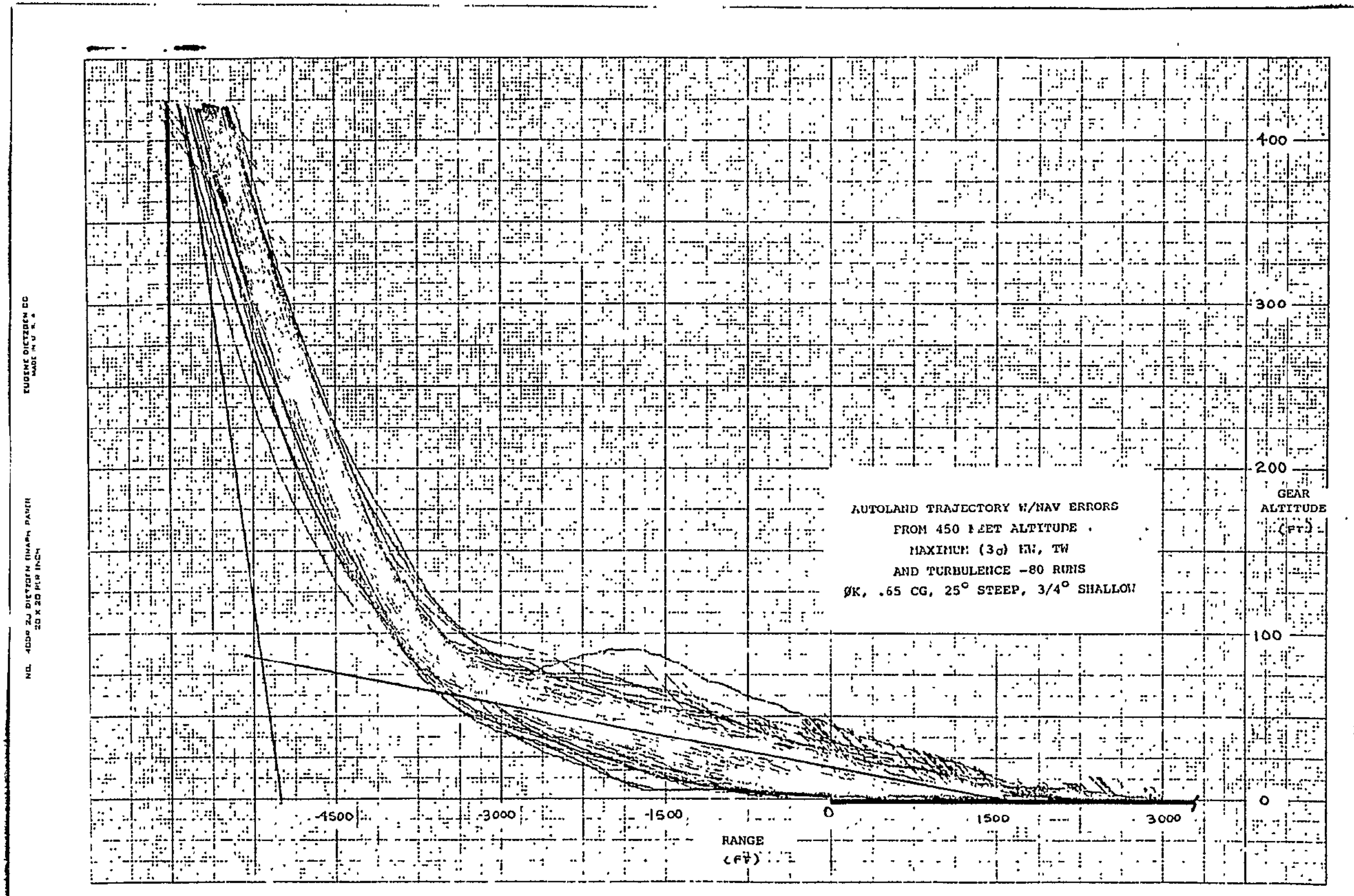

AUTOLAND TRAJECTORY W/NAV ERRORS
FROM 450 FEET ALTITUDE
MAXIMUM (3σ) HW, TW
AND TURBULENCE -80 RUNS
ØK, .65 CG, 25° STEEP, 3/4° SHALLOW
400
300
200
100
0
GEAR ALTITUDE (FT)
-4500
-3000
-1500
0
1500
3000
RANGE (FT)

Figure 17

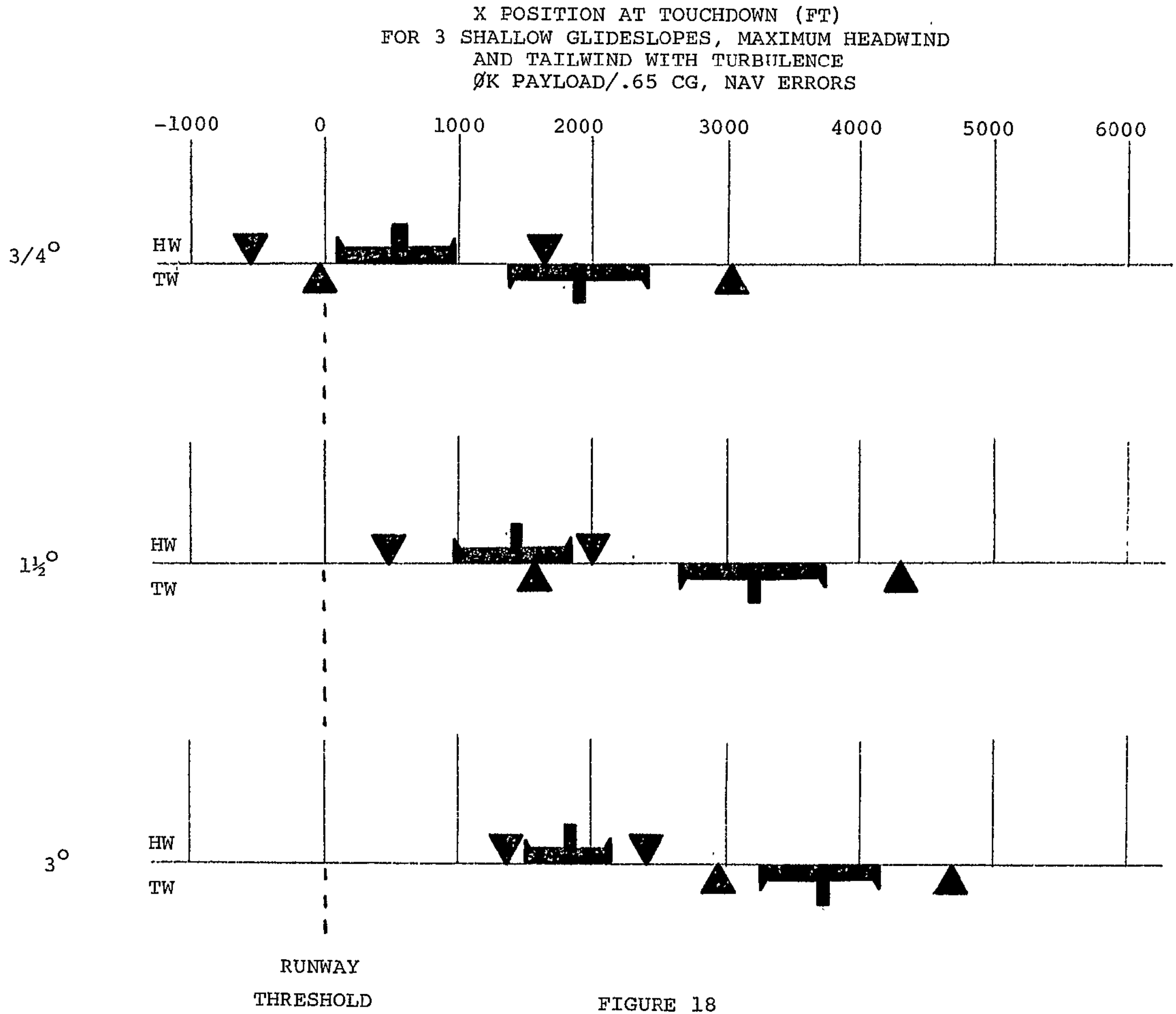

X POSITION AT TOUCHDOWN (FT)
FOR 3 SHALLOW GLIDESLOPES, MAXIMUM HEADWIND
AND TAILWIND WITH TURBULENCE
ØK PAYLOAD/.65 CG, NAV ERRORS
-1000
0
1000
2000
3000
4000
5000
6000
3/4°
HW
TW
1½°
HW
TW
3°
HW
TW
RUNWAY
THRESHOLD

FIGURE 18

SECTION 9

AERO DATA TOLERANCES

AUTOLAND MEMO NO. 9

TO BE PROVIDED

SECTION 10

TURBULENCE MODEL STUDY

AUTOLAND MEMO NO. 10

TO BE PROVIDED