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SIMULATION STUDY OF PROPOSED YAW DAMPER SYSTEMS FOR THE B-58 AIRCRAFT

R. L. HAAS

TECHNICAL REPORT AFFDL-TR-66-136

SEPTEMBER 1966

AIR FORCE FLIGHT DYNAMICS LABORATORY
RESEARCH AND TECHNOLOGY DIVISION
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SIMULATION STUDY OF PROPOSED YAW DAMPER SYSTEMS FOR THE B-58 AIRCRAFT

R. L. HAAS

FOREWORD

This report was initiated as the result of an analog simulation of the lateral-directional equations of motion of the B-58 aircraft and associated inner-loop controllers.

The study was undertaken at the request of the Systems Engineering Group (SEG) which was desirous of additional information for increased confidence in their program to provide a redundant lateral-directional stability augmentation system for the B-58.

The simulation program was accomplished through the joint efforts of SEG, Mr. Andes and Mr. Taylor and AFFDL, Mr. Haas. The program was conducted in the Control Techniques Simulation Facility (FDCL) during March and April 1965.

Conclusions and recommendations were agreed upon by both SEG and AFFDL. Results have been utilized by SEG in the Redundant Yaw Damper Program.

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This technical report has been reviewed and is approved.

H. W. BASHAM
Chief, Control Elements Branch
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AF Flight Dynamics Laboratory

ABSTRACT

Operational problems encountered with the B-58 aircraft led the Air Force to direct the prime contractor, General Dynamics, to redesign the lateral-directional augmentation system incorporating a limit cycle adaptive system in the yaw axis.

This work is part of a combined AFFDL/SEG in-house study to review the contractor's recommendations and conclusions concerning the B-58 Redundant Yaw Damper Program. Two stability augmentation systems, one fixed gain, the other limit cycle adaptive, were evaluated on a three degree-of-freedom lateral-directional simulation of the B-58 aircraft. The equations of motion were based on small perturbation assumptions. A "cockpit" from which pilots could "fly" the aircraft was included. Simulation results generally were in agreement with information provided by the contractor. Presentation of data is essentially limited to coverage of topics not discussed in other reporting.

Low speed controllability with either augmentation system is not considered satisfactory.

Structural modes are not adequately defined and interaction with controller modes could not be evaluated.

There exists enough doubt about the adequacy of both the adaptive and the pure fixed gain approaches to question the worth of either type as designed.

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SYMBOLS AND NOTATION

SYMBOL	DEFINITION	UNITS
α	Incremental Angle of Attack	Deg
α_t	Trim Angle of Attack	Deg
β	Side Slip Angle	Deg
$\dot{\beta}$	Rate of Change of Side Slip Angle	Deg/Sec
δ_a	Aileron Deflection from Trim; Positive Left Aileron Trailing Edge Up	Rad, Deg
δ_{a_D}	Aileron Deflection Due to Damper	Deg
δ_r	Rubber Deflection from Trim; Positive Trailing Edge Left	Rad, Deg
δ_{r_D}	Rudder Deflection Due to Damper	Deg
δ_s	Control Stick Deflection	Deg
$\delta_a/\dot{\phi}$	Aileron Deflection Per Roll Rate Gain	Deg/Deg/Sec
$\delta_a/\dot{\psi}$	Aileron Deflection Per Yaw Rate Gain	Deg/Deg/Sec
δ_r/a_y	Rudder Deflection Per Lateral Acceleration Gain	Deg/Ft/Sec^2
δ_r/δ_a	Rudder Deflection Per Aileron Deflection Gain	Deg/Deg
$\delta_r/\dot{\psi}$	Rudder Deflection Per Yaw Rate Gain	Deg/Deg/Sec
Δ	Incremental Change	-
ζ	Damping Ratio	-
T_a	Servo Time Constant	Seconds
$\dot{\phi}$	Incremental Roll Angle; Positive Right Wing Down	Rad, Deg
$\ddot{\phi}$	Roll Rate	Rad/Sec, Deg/Sec
ϕ	Roll Acceleration	Rad/Sec^2, Deg/Sec^2
ψ	Yaw Angle	Deg

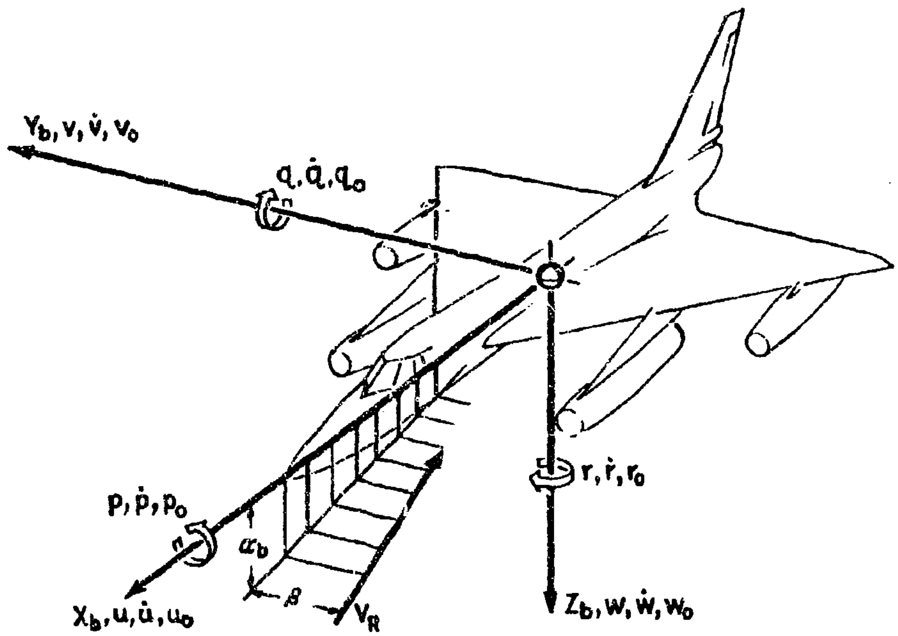
SYMBOL	DEFINITION	UNITS
$\dot{\psi}$	Yaw Rate	Rad/Sec, Deg/Sec
$(\psi+\beta)$	Heading Error	Deg
ω_a	First Order Servo Break Frequency	Rad/Sec
ω_o	Undamped Natural Frequency	Rad/Sec
a	Speed of Sound	Ft/Sec
A_y	Lateral Acceleration	Ft/Sec^2,"g's"
ARI	Aileron-Rudder Interconnect	-
b	Wing Span	Feet (56.82)
$\bar{c}$	Mean Aerodynamic Chord	Feet (36.17)
C_l	Rolling Moment Coefficient, $\frac{\text{(Rolling Moment)}}{qSb}$	-
C_{l_β}	$dC_l/d\beta$	/deg
$C_{l_{\dot\beta}}$	$dC_l/d(\frac{\dot\beta b}{2V})$	/rad
$C_{l_{\delta_a}}$	$dC_l/d\delta_a$	/deg
$C_{l_{\delta_r}}$	$dC_l/d\delta_r$	/deg
C_{l_p}	$dC_l/d(\frac{\dot\phi b}{2V})$	/rad
C_{l_r}	$dC_l/d(\frac{\dot\psi b}{2V})$	/rad
C_n	Yawing Moment Coefficient (N/q Sb)	-
C_{n_β}	$dC_n/d\beta$	/deg
$C_{n_{\dot\beta}}$	$dC_n/d(\frac{\dot\beta b}{2V})$	/rad
$C_{n_{\delta_a}}$	$dC_n/d\delta_a$	/deg
$C_{n_{\delta_r}}$	$dC_n/d\delta_r$	/deg
C_{n_p}	$dC_n/d(\frac{\dot\phi b}{2V})$	/rad
C_{n_r}	$dC_n/d(\frac{\dot\psi b}{2V})$	/rad
C_t	Thrust Coefficient (T/q S)	-
C_y	Lateral Force Coefficient (Y/q S)	-

SYMBOL	DEFINITION	UNITS
Cy_{β}	$dCy/d\beta$	/deg
$Cy_{\delta a}$	$dCy/d\delta_a$	/deg
$Cy_{\delta r}$	$dCy/d\delta_r$	/deg
Cy_p	$dCy/d(\frac{\dot{\phi} b}{2V})$	/rad
Cy_r	$dCy/d(\frac{\dot{\psi} b}{2V})$	/rad
c.g.	Center of Gravity	% MAC
f( )	Function of Variable Inclosed in Parenthesis	-
g	Acceleration Due to Gravity	g's, ft/sec^2
G.W.	Gross Weight	Pounds
h	Altitude	Feet
$\dot{h}$	Rate of Change of Altitude	Ft/Sec
I_x, I_y, I_z	Inertia About x, y, z axes, Respectively	Slug ft^2
J_{xz}	Product of Inertia About x and z Axis	Slug ft^2
K	Kilo or Thousand	-
K_I	Integrator Gain	1/Sec
K_x	I_x/q Sb	Sec^2
K_{xz}	J_{xz}/q Sb	Sec^2
K_z	I_z/q Sb	Sec^2
M	Mach or Mach Mode	-
N	Yawing Moment	Ft/Lbs
P	Rolling Angular Velocity	Rad/Sec
q Q	Dynamic Pressure	Lbs/Ft^2
S	LaPlacian Operator	-
S	Wing Area	Ft^2(1542)
t	Time	Seconds

SYMBOL	DEFINITION	UNITS
T	Thrust Along Thrust Line	Pounds
tr	Subscript Indicating Trim Condition	-
w/o	Washout	-
X	x Axis; Along Projection of Relative Wind	-
Y	y Axis; Perpendicular to x and z Axes (Lateral Axis)	-
Y	Lateral Force	Pounds
Z	z Axis; Perpendicular to x and y Axes (Vertical Axis)	-
C_{HA}	Aileron Hinge Moment Coefficient	
C_{HA_α}	$dC_{HA}/d\alpha$	/deg
C_{HA_δ}	$dC_{HA}/d\delta$	/deg
C_{HA_R}	$dC_{HA}/d\dot{\Phi}(\frac{b}{2V})$	/rad
C_{HO}	Elevon Hinge Moment Coefficient at Zero Elevon Deflection	/deg
C_{HR}	Rudder Hinge Moment Coefficient	/deg
$C_{HR_{\delta_r}}$	$dC_{HR}/d\delta_r$	/deg
$C_{HR_{\delta_a}}$	$dC_{HR}/d\delta_a$	/deg
C_{HR_β}	$dC_{HR}/d\beta$	/deg
HM_A	Aileron Hinge Moment	ft-lbs
HM_R	Rudder Hinge Moment	ft-lbs
MA_A	Moment Area of Aileron	ft^2
MA_R	Moment Area of Rudder	ft^2
$Ry_{\delta_r}, Rl_{\delta_r}, Rn_{\delta_r}$	Flexibility Ratios for Rudder	-
Xcg	Location of cg on x-axis	% MAC
$\overline{X}cg$	Reference Location of cg on x-axis	25% MAC
Zcg	Location of cg on z-axis	ft

SYMBOL	DEFINITION	UNITS
$\bar{Z}cg$	Reference Location of cg on z-axis	8.67 ft
L	Rolling Moment Due to Transverse Fuel Shift	ft-lbs
K_1	Constants Defining Transverse Fuel Shift	ft-lb/g-sec
K_2	Constants Defining Transverse Fuel Shift	ft-lb/g
K_{2_0}	Constants Defining Transverse Fuel Shift	ft-lb/g
$f(A_Y)$	Constants Defining Transverse Fuel Shift	-

AXES AND RELATED ANGLES



INTRODUCTION

This report is a summary of a lateral-directional simulation study of the B-58 aircraft. The effort was undertaken in order to provide SEG with technical data for evaluation of proposed yaw augmentation system changes in the B-58.

Numerous incidents/accidents incurred in operational usage of the B-58 were considered the fault of poor reliability in the lateral-directional stability augmentation system (Reference 1). As a result of this, the Air Force directed General Dynamics to design and incorporate a new system, the basic philosophy being to significantly increase reliability while maintaining or improving existing handling qualities. The pertinent details of the system specification are listed below.

YAW STABILITY AUGMENTATION SPECIFICATION

Performance Requirements (Flying Qualities)

The modified yaw stability augmentation (S/A) system shall meet the requirements of MIL-F-8785 as amended by this specification. In case of conflict, the requirements of this specification shall apply.

Ranges of Operation

MIL-F-8785 as amended by this specification shall apply throughout the design operational envelope (Mach Number and Altitude) including all operational configurations, center of gravity positions, and aircraft weights within structural and maneuverability limits.

Amended Requirements

General: All requirements shall be met with the pilot out of the loop and with the pilot in the loop controlling in his normal manner, (e.g., during and subsequent to an engine failure, the pilot normally will attempt to hold a wings-level attitude by use of aileron stick inputs only).

Lateral Damping

Acceptable damping of the lateral-directional oscillations and acceptable values of the ϕ/Ve rolling parameters are indicated in Figure 1 of MIL-F-8785 except as modified below. Residual undamped oscillations may be tolerated only if the amplitude is no greater than 0.2 degree peak-to-peak sideslip. Dampers on configuration shall meet bombing and firing requirements.

Dutch Roll Natural Frequency

Acceptable values of the dutch roll natural frequency (ω_D) are given by $1 < \omega_D < 6$ rad/sec.

Rolling Capability

(Exception to MIL-F-8785) It shall be possible to roll to and stabilize at 60 degrees of bank angle in three seconds or less.

Lateral Frequency Ratio

The ratio of the second order numerator frequency (ω_ϕ) to the dutch roll frequency (ω_D) of the roll-rate-to-aileron transfer function shall be given by $0.9 < \omega_\phi/\omega_D < 1.0$ for $\zeta_D < 0.3$; and $0.9 < \omega_\phi/\omega_D < 1.0$ for $\zeta_D > .3$. The ratio ϕ/β shall not be > 6.0 nor < 1.0. The ω_ϕ/ω_D requirement shall be considered an objective for $\zeta_D > 0.35$. However, the dutch roll damping ratio, ζ_D should not be degraded in attempting to comply with the ω_ϕ/ω_D objective.

Turn Coordination

Automatic turn coordination shall mean the automatic reduction of sideslip during banking maneuvers. The automatic turn coordination should not allow a maximum transient sideslip larger than 8.5 degrees to develop during rolls to 60 degrees bank angle with abrupt aileron input at 1.4 V_{spa}. V_{spa} is to be interpreted as the speed at which 1 "g" flight is maintained at 17 degrees angle of attack at the particular flight gross weight, center of gravity, and altitude. At all higher speed flight conditions, the maximum transient lateral acceleration should, as an objective, not exceed 0.1 "g" and shall not exceed a maximum allowable of 0.3 "g" during maximum abrupt aileron input rolls to 60 degrees bank angle. Steady state lateral acceleration shall not exceed 0.03 "g".

Sudden Engine Failure

After an outboard engine failure, at any permissible flight condition, with the other engines developing maximum A/B power for that flight condition, the resulting maximum sideslip must not exceed that specified in Table 1.

TABLE 1

ALLOWABLE SIDESLIP FOLLOWING SUDDEN ENGINE FAILURE

Mach No.	Maximum Sideslip (Degrees)
*0.3	8.5
0.6	6.5
1.24	3.5
1.6	3.5
2.0	3.5

* Or the speed associated with 1 "g" flight at 17 degrees angle of attack at the particular loading condition.

Saturation Characteristics

The system shall exhibit stable and unmagnified response to disturbances which cause control saturation; i.e., system shall be designed so that it will not be amplitude sensitive.

Spiral Mode Time Constant

The spiral mode time constant (T_S) may be divergent, but the rate of divergence shall not be so great that, following a small disturbance in bank, the bank angle is doubled in less than 20 seconds.

Wing Heaviness

Correction for the wing heaviness caused by transverse fuel shift due to lateral acceleration shall be provided. The performance shall be considered acceptable if the lateral acceleration is maintained at approximately zero in the steady-state condition. In any event, wing heaviness shall be controlled, at the worst loading conditions from a wing heaviness consideration of half-full aft fuel tank, such that not more than one degree of aileron control will be required to maintain a wings-level trimmed attitude. This is to be accomplished assuming symmetrical thrust from the engines and no rudder command by the pilot. As an objective, this should be accomplished without an integral of rudder per lateral acceleration gain.

Cross Wind Requirements

The pilot shall be able to exercise directional control in order to sideslip the aircraft for landing in a crosswind per paragraph 3.4.11.1 of MIL-F-8785, without having steady state yaw damper opposition to the pilot's rudder command. Thus, it will be necessary to make provisions for deactivating the rudder-per-side-acceleration gain to accomplish intentional sideslips for crosswind operation and to provide yaw damping during these intentional sideslips.

DESIGN REQUIREMENTS

Yaw Augmentation

General: The yaw S/A design will be based on the following specific modifications to the existing S/A and flight control provisions.

a. A fixed mechanical aileron-rudder interconnect of 1:1 with no electrical interconnect.

b. The aileron-per-yaw-rate, $\delta_a/\dot{\psi}$, shall be switched in automatically upon loss of yaw S/A. A positive interlock shall be provided to insure that the $\delta_a/\dot{\psi}$ signal is not fed into the roll damper when the yaw damper is functioning.

c. A fixed aileron per roll rate gain in the roll S/A function.

Self-Adaptive Gain Changing

A self-adaptive type gain changer of flight proven capability shall be used. The acceptable technique uses a high gain limit cycle. The adaptive logic shall be so designed that response to gust disturbances, structural bending effects and electrical noise inputs will not result in gain changes that degrade system performance below minimum requirements specified herein.

The gain changer time response characteristics shall be stable and adequate during normal rapid changes in aircraft flight conditions and characteristics.

The SEG/AFFDL in-house simulation effort was a comparative evaluation of the Bendix adaptive system and the General Dynamics fixed gain system (General Dynamics favors a fixed gain system even though the Air Force spec requires a limit cycle adaptive system). A detailed description of the Bendix system is presented in Reference 4.

General Dynamics performed a comprehensive simulation study as part of the Redundant Yaw Damper Program. This in-house evaluation spot checked the results of the General Dynamics evaluation and their assessment of problems and then carried the effort to additional areas, covering items considered unresolved by AFFDL and SEG.

This report only documents those problem areas not reported elsewhere by General Dynamics but considered by the Air Force to be worthy of note. Significant problems previously reviewed by the contractor include the sensitivity of the adaptive system to random (gust or pilot) inputs and its tendency to reduce gain under this condition (Bendix study) and the amplitude sensitivity of both systems (General Dynamics study).

No attempt was made to evaluate the present stability augmentation system. This has been the subject of numerous contractor and Air Force efforts in the past (Reference 1). The present aircraft has been considered as having acceptable flying qualities for augmented mode of operation (poor to unacceptable unaugmented).

It should be emphasized that results obtained through this effort are only as good as the available data and the assumptions made. According to General Dynamics, the derivative $C_{n\delta_r}$ has somewhat different and more variant values than those used in this Air Force study. This would represent greater variation in surface effectiveness resulting in a greater airframe gain variation than that encountered in this study. Previous computer analyses of the B-58 aircraft have been based on small perturbation assumptions including purely linear aerodynamic characteristics. This study also attempted to determine system performance with nonlinear C_{n_β} and C_{l_β} included when side-slip (β) is forced to large excursions. Additionally, a limited parametric study was accomplished to evaluate the performance of the two proposed systems with reasonable variation of the predicted linear aerodynamic characteristics.

Structural mode data supplied by the contractor is not acceptable for analysis to determine structural mode interaction with other system modes.

The analysis recently performed by General Dynamics in evaluating the augmentation systems did not include a man in the loop, with the exception that a simulated pilot inputs were used for the rolling pullout maneuver (RPO) and for single engine failure (SEF) corrective action. It is considered that these inputs were optimized to obtain desired performance for these two problems. In order to better evaluate the aircraft's performance and whether or not it meets spec requirements, a cockpit was utilized as part of this Air Force simulation, and a number of runs were made with a pilot "flying" the simulation.

DISCUSSION

The approach taken in this study was to evaluate the proposed yaw augmentation systems through a three-degree-of-freedom lateral-directional analog computer simulation of the B-58 aircraft. An existing General Dynamics analog computer program was used as the basis for the simulation which was set up at the Control Techniques Simulation Facility. This was done in order to insure direct correlation between contractor and Air Force results and to utilize data provided by General Dynamics in Reference 2. In general, the General Dynamics and the Air Force simulations can be considered identical. The simulation of aerodynamic characteristics is based on small perturbation assumptions and is considered inadequate for any situation where relatively large amplitude disturbances are encountered (the equations are no longer valid and also, the derivatives are nonlinear). Simulations of the control systems did include all nonlinearities and were representative of the actual systems. Whenever possible, results obtained in this study were compared with previous analog and/or digital simulation results in order to verify the simulation. A problem frequently encountered with the limit-cycle type of adaptive system is the interaction of structural modes and controller modes. Data supplied by General Dynamics and their analog representation of this data are considered unrealistic and unuseable for analysis of the structural mode problem. Analog circuit diagrams of the GD representation of structural modes are presented, but were not utilized. The question of mode interaction remains unanswered.

The equations simulated are listed in Figure 1. Analog circuit diagrams are presented in Appendix A. In order to study response to outboard engine failure (SEF), a portion of the function generation, the thrust decay curves (Reference 2), was mechanized on a Litton digital computer rather than with the analog representation utilized by General Dynamics.

Pot settings (see Appendix B) were determined by a digital computer program (Reference 2) which additionally presents free aircraft transfer functions for the selected flight condition.

Review of data obtained from a previous in-house study (not covered by this report) to compare the performance of the Bendix adaptive system and a Honeywell adaptive system (equivalent performance for the conditions investigated) showed very large excursions in sideslip for the rolling pullout maneuver (RPO) and sudden engine failure (SEF) at the low speed flight conditions. This led to generation of the plot shown in Figure 2 for use in this study. The β for zero slope was estimated through review of B-58 and other high performance aircraft data for particular flight conditions.

The analog representation of this curve is shown in Appendix A. Admittedly, this is a rough approximation, but the available aerodynamic data does not warrant a more sophisticated representation of the nonlinearity.

The sensitivity of the control system's performance for small amplitude disturbances to errors in predicted aerodynamics was evaluated by varying the value (up to $\pm 20\%$) of various derivatives.

1. $mA_y = mv(\dot{\Psi} + \dot{\beta}) - W \sin\Phi = Qs[C_{y_{\delta_r}}\delta_r + C_{y_\beta}\beta + \frac{b}{2V}(C_{y_{\dot\beta}}\dot{\beta} + C_{y_p}\dot{\Phi} + C_{y_r}\dot{\Psi}) + (\frac{\partial^2 C_y}{\partial\delta_a \partial\delta_{a}})\delta_a\beta|\delta_a| + C_{y_{\delta_a}}\delta_a]$

2. $\ddot{\Phi} = \frac{Qsb}{I_x}[C_{l_{\delta_r}}\delta_r + C_{l_{\delta_a}}\delta_a + C_{l_\beta}\beta + \frac{b}{2V}(C_{l_{\dot\beta}}\dot{\beta} + C_{l_p}\dot{\Phi} + C_{l_r}\dot{\Psi}) + (\frac{\partial^2 C_l}{\partial\delta_a \partial\delta_a})\delta_a\beta|\delta_a|] + \frac{I_{xz}}{I_x}\ddot{\Psi} + \frac{I_{xz}}{I_x}\sin(\alpha - \epsilon) + \frac{\Psi_{\omega}}{I_x}$

3. $\ddot{\Psi} = \frac{Qsb}{I_z}[C_{n_{\delta_r}}\delta_r + C_{n_{\delta_a}}\delta_a + C_{n_\beta}\beta + \frac{b}{2V}(C_{n_{\dot\beta}}\dot{\beta} + C_{n_p}\dot{\Phi} + C_{n_r}\dot{\Psi}) + (\frac{\partial^2 C_n}{\partial\delta_a \partial\delta_a})\delta_a\beta|\delta_a|] + \frac{I_{xz}}{I_z}\ddot{\Phi} + \frac{\Psi}{I_z}\cos(\alpha - \epsilon)$

4. TAIL LOAD $= Qs[C_{Y_{\delta_r V.T.}}\delta_r + C_{Y_{\delta_a V.T.}}\delta_a + C_{Y_{\beta V.T.}}\beta + \frac{b}{2V}(C_{Y_{\dot\beta V.T.}}\dot{\beta} + C_{Y_{p V.T.}}\dot{\Phi} + C_{Y_{r V.T.}}\dot{\Psi}) + (\frac{\partial^2 C_Y}{\partial\delta_a \partial\delta_a})_{V.T.}\delta_a\beta|\delta_a|]$

5. RUDDER HINGE MOMENT $= 2MA_r Q(C_{h_{\delta_r}}\delta_r + C_{h_{\delta_a}}\delta_a + C_{h_\beta}\beta)$

6. AILERON HINGE MOMENT $= 2MA_a Q(C_{h_{\delta_a}}\delta_a + \frac{b}{2V}C_{h_p}\dot{\Phi} + C_{h_0})$

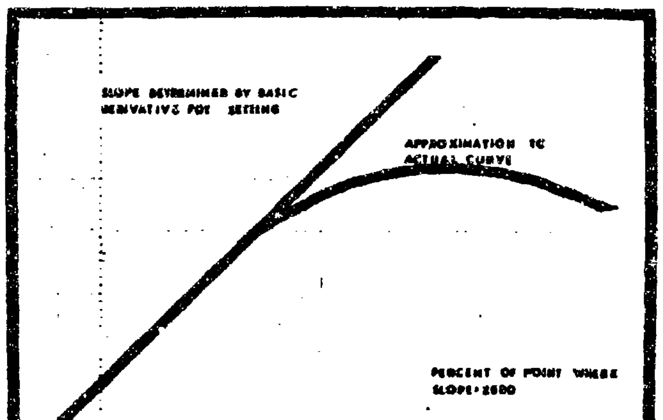
7. $L_{fuel} = \frac{1}{S+1}[K_1(a_n)\int_0^t A_Y dt + K_{2_n}(K_2 \omega_r)]$ note: $\int_0^t A_Y dt$ LIMIT IS 210.5 ft/sec

8. $\dot{\Phi}_{SENSOR} = \frac{31.5}{S+31.5}[\dot{\Phi}_{STABILITY}\cos(\alpha - 5^\circ) - \dot{\Psi}_{STABILITY}\sin(\alpha - 5^\circ)]$

9. $\dot{\Psi}_{SENSOR} = \frac{31.5}{S+31.5}[\dot{\Phi}_{STABILITY}\sin(\alpha - 5^\circ) + \dot{\Psi}_{STABILITY}\cos(\alpha - 5^\circ)]$

10. $A_{Y_{SENSOR}} = \frac{14}{S+14}(A_Y)$

EQUATIONS OF MOTION
FIGURE 1



APPROXIMATION OF NON-LINEARITIES
FIGURE 2

Figure 3 is a listing of the flight conditions checked, and Figure 4 is a listing of the tests run at each flight condition with the coding utilized on the time histories.

The cockpit was used to determine trends in overall controllability. Detailed pilot evaluation/interpretation was not attempted.

A block diagram of the adaptive system is presented in Figure 5, and Figure 6 is the block diagram for the fixed gain system.

Equipment Used

The simulation equipment used consisted of two EAI 231R analog computers (Figure 7), a Litton CG-820 digital computer (Figure 8) with D to A interface, and a cockpit (Figure 9) with rudder pedals and a "formation type" control stick. Flight information was presented to the pilot through an all-attitude indicator and a sideslip indicator (no motion or external visual cues were provided).

FLIGHT CONDITIONS CHECKED

Condition	Mach	Altitude	Gross Weight	Center of Gravity
002	.33	0 ft	150,000#	28%
004	.239	0 ft	80,000#	28%
006	.91	0 ft	150,000#	28%
008	.91	40,000 ft	120,000#	30%
010	1.36	25,000 ft	90,000#	33%
011	1.2	40,000 ft	120,000#	33%
013	2.0	44,200 ft	120,000#	33%

FIGURE 3

BASIC PERFORMANCE DATA

Run 1 – Free A/F check

Run 2A – All dampers in

2B – Roll damper out

Run 3A – RPO (0-60°)

3B – RPO with nonlinear C_{n_β}

3C – RPO with pilot (WR indicates rudder used)

3D – RPO with pilot and nonlinear C_{n_β}

3E – RPO with nonlinear C_{n_β} and C_{l_β}

3F – RPO with pilot and nonlinear C_{n_β} and C_{l_β}

Run 4A – (1-cosωt) β gust (2°)

Run 5A – Outboard engine out (SEF)

5A' – Engine out with nonlinear C_{n_β}

5A" – Engine out with nonlinear C_{n_β} and C_{l_β}

Run 5B – Engine out (wings level)

5B' – Engine out (wings level) with nonlinear C_{n_β}

5B" – Engine out (wings level) with nonlinear C_{n_β} and C_{l_β}

Run 5C – Engine out with pilot (primes as above; WR indicates rudder used)

Runs 6 – 9 are sensitivity analysis

(6 C_{n_r}, 7 C_{l_p}, 8 C_{l_β}, 9 C_{n_β} with A+ and B-)

RUN CODE

XXX	F OR M	XX
Designates Flight Case	Designates Fixed Gain or Adaptive System	Denotes Run

Example: 013 F 2A is Flight Case 013, Fixed Gain System, Run 2A (all dampers in)

FIGURE 4

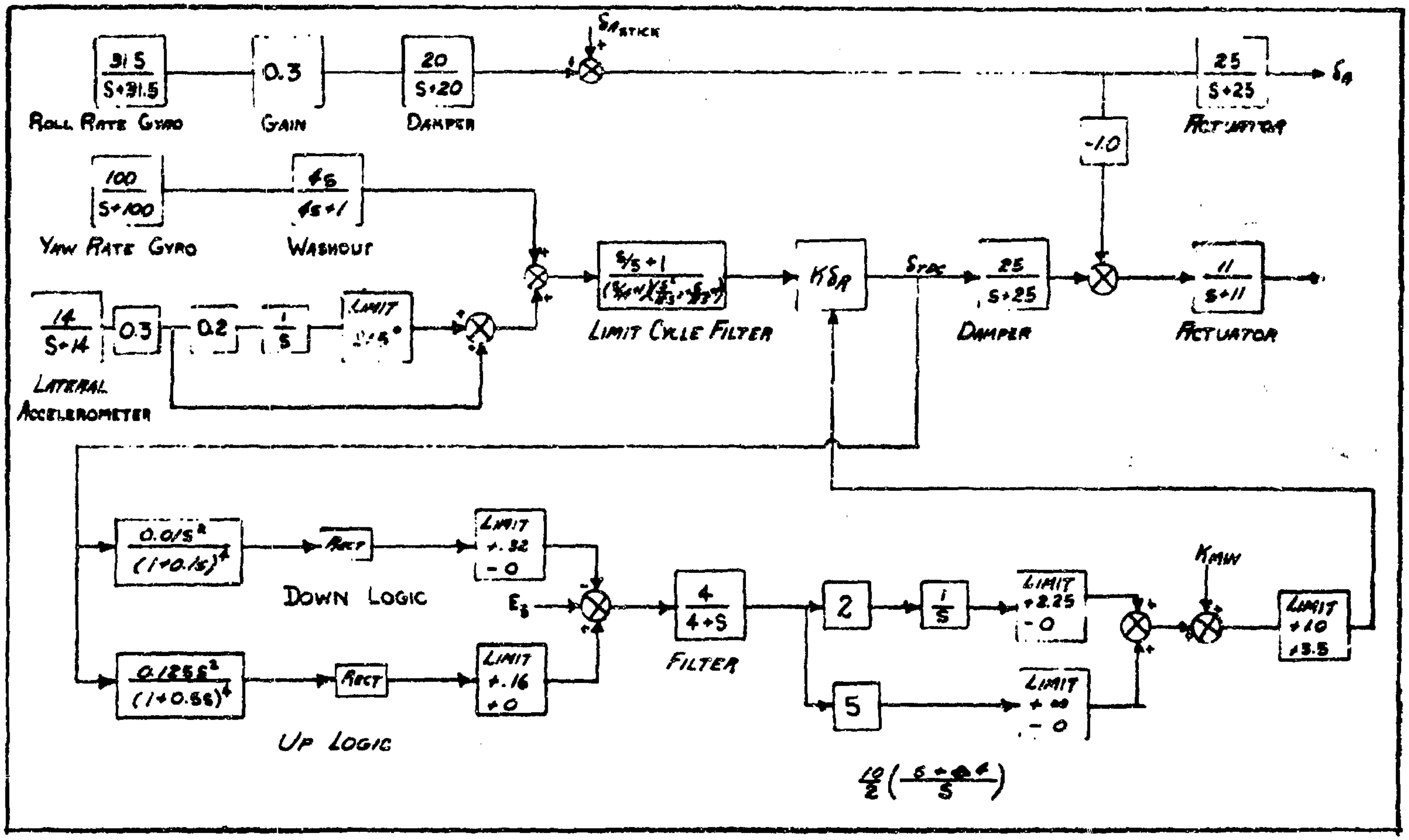


FIGURE 5 BLOCK DIAGRAM - ADAPTIVE SYSTEM

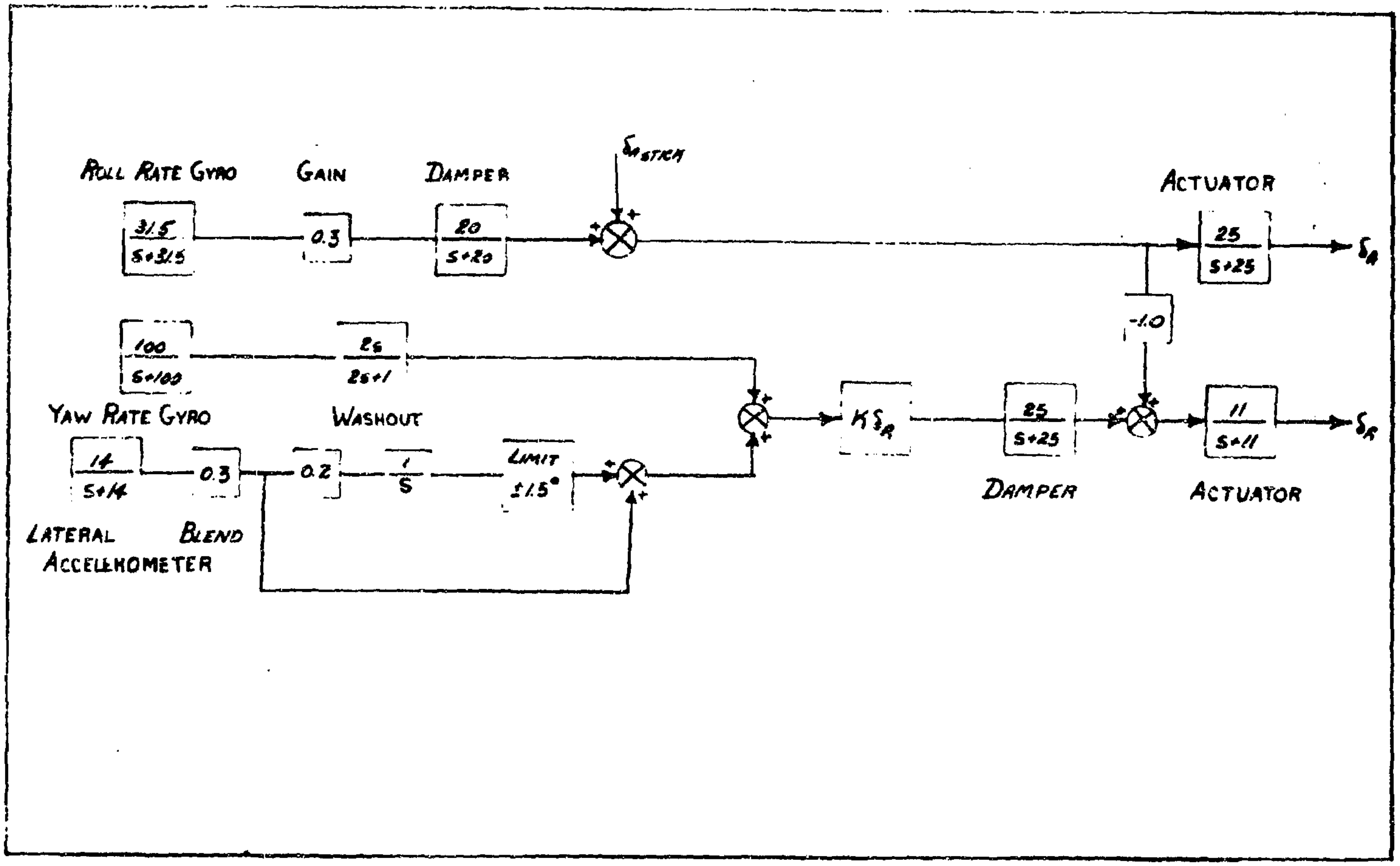


FIGURE 6 BLOCK DIAGRAM - FIXED GAIN SYSTEM

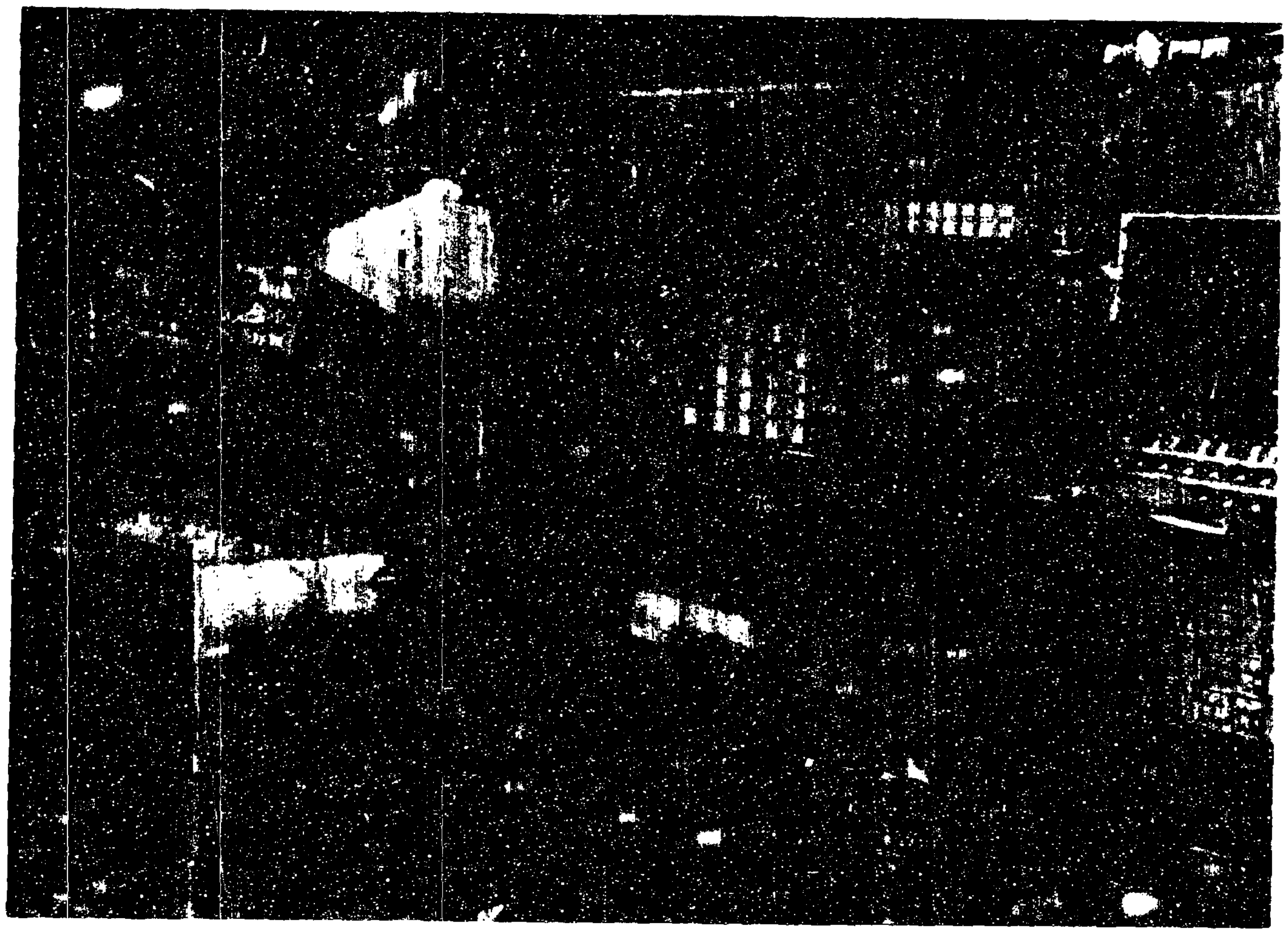


FIGURE 7 ANALOG COMPUTER

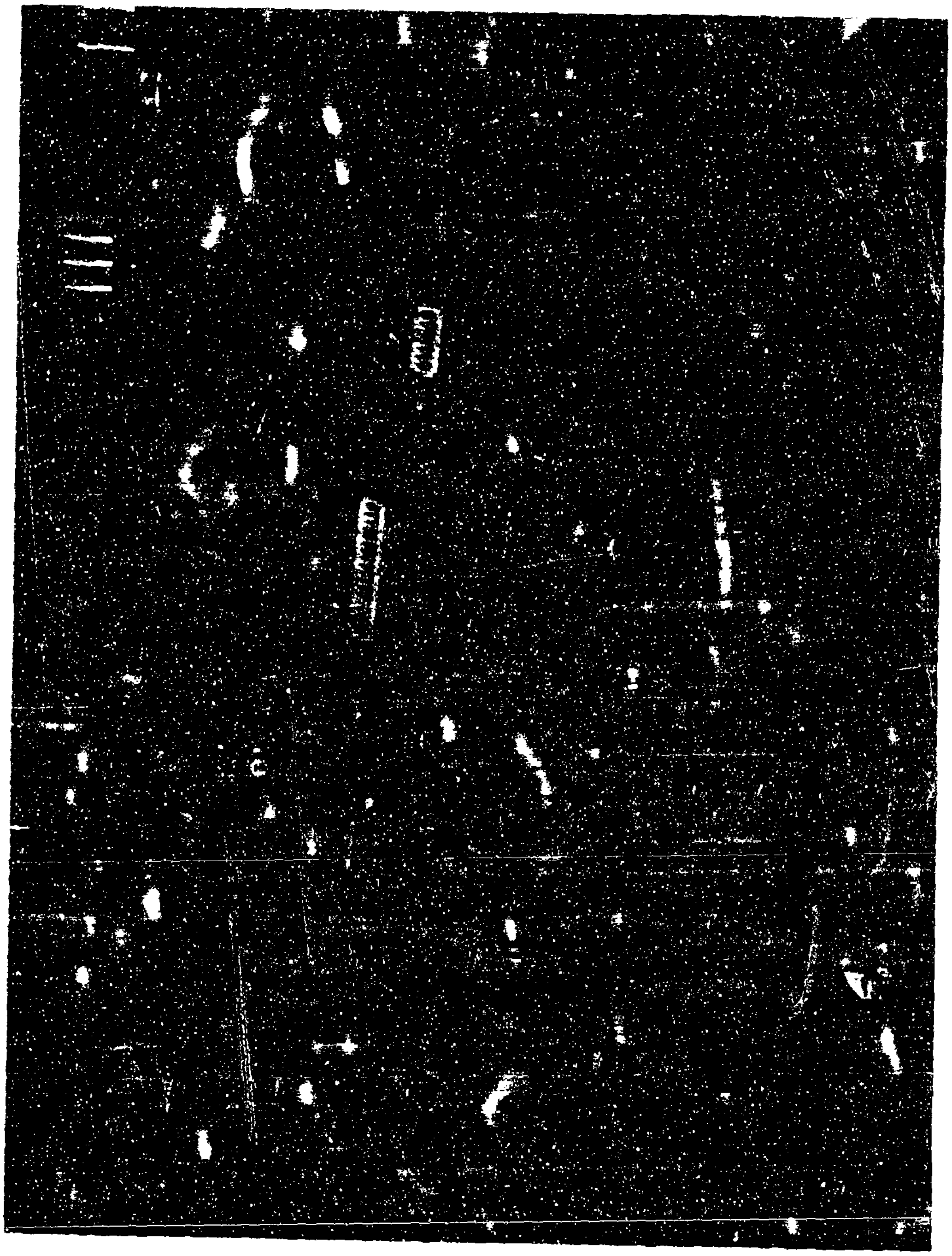


FIGURE 8 DIGITAL COMPUTER

FIGURE 9 COCKPIT

RESULTS

A summary of the results obtained in this study is presented in Figure 10. A brief review of each flight condition is then presented including noteworthy time histories with appropriate discussion. For each flight condition evaluated, free airframe responses were checked for agreement with tabulated dynamic characteristics provided by General Dynamics in Reference 2. In all cases, very close agreement exists. For the low speed flight conditions where sideslip was large for RPO and SEF (augmented vehicle), the results are questionable because of small perturbation assumptions. When the curve in Figure 2 was utilized to represent nonlinear C_{n_β} and C_{l_β} for these low speed conditions (note that equations were not changed), controllability is a problem, and the system can be driven to divergence. General Dynamics did not concur with our representation of nonlinear C_{n_β} and C_{l_β} (Figure 2) and referenced FZE-4-020 (Reference 3). Review of this document provided no justification for great faith in any curve because of insufficient data points. However, it did indicate that for the low speed $17^\circ\,\alpha$ conditions C_{n_β} is zero at $20^\circ\,\beta$ and C_{l_β} is zero at $25^\circ\,\beta$. Consequently, those results including nonlinearities letting the slope of C_{n_β} and/or C_{l_β} be zero at 15° (for low speed, $\alpha = 17^\circ$ conditions only) are invalid. Some of these results are included to graphically show trends caused by the nonlinearities. In those situations where only C_{n_β} slope is zero at $20^\circ\,\beta$, results are considered valid; but for both C_{n_β} and C_{l_β}, results are conservative.

FLIGHT CONDITION	FREE AIRFRAME			FIXED GAIN SYSTEM				ADAPTIVE GAIN SYSTEM			
				ROLL DAMPER IN		ROLL DAMPER OUT		ROLL DAMPER IN		ROLL DAMPER OUT	
	ξ	ω		ξ	ω	ξ	ω	ξ	ω	ξ	ω
002	-.017	1.395		.275	1.340	.025	1.395	.345	1.309	.018	1.395
004	-.021	1.208		.285	1.158	.133	1.197	.265	1.164	.128	1.198
006	.063	2.180		.370	2.025	.370	2.025	.430	1.968	.400	1.997
008	.016	1.487		.350	1.392	.225	1.443	.350	1.392	.290	1.492
010	.095	2.09		.214	2.041	.312	1.872	.230	2.033	.312	1.872
011	.061	1.96		.350	1.863	.312	1.756	.305	1.866	.330	1.850
013	.074	1.281		.243	1.174	.250	1.240	.347	1.201	.350	1.200

FIGURE 10 SUMMARY OF RESULTS

Flight Condition 002

Flight condition 002 is one of the two low speed flight conditions. Results for this condition, along with those for flight condition 004, were significantly affected when nonlinear C_{l_β} and C_{n_β} were incorporated. For this flight condition, the free airframe is unstable. Response of the aircraft with the fixed gain system (roll damper on) was well damped; however, low amplitude residual oscillation exists. The ARI variation produced no significant results as the response to the 2° initial sideslip input was roughly the same for the various ARI positions selected. For the fixed gain system, the response of the aircraft to the 2° initial beta input with the roll damper out is poor. The airplane is lightly damped (close to neutrally stable). For the adaptive system, the response to a 2° beta initial input (all dampers on) is satisfactory; however, again the small residual oscillations were noted. For the case of the roll damper out, the aircraft is very lightly damped. For the rolling-pullout maneuver with the fixed gain system and with the simulated pilot, the responses were not satisfactory. Including the nonlinearities with the slopes going to zero at $20^\circ\beta$ produced an unflyable situation as the aircraft was not controllable with the simulated pilot. Utilizing the cockpit with an actual pilot, the aircraft was only controllable with large application of rudder. Rolling velocity reversal did occur, and it was impossible to precisely control the aircraft rolling from $+30^\circ$ to -30°. Pilots generally were not able to precisely roll to a given roll attitude or to roll out on a desired heading. Varying the ARI for the rolling-pullout maneuver, again produced no significant changes. The 2° β gusts were also applied to both the fixed and the adaptive gain systems without any significant results being recorded. For this flight condition under the engine out test, the aircraft rolls (with either system) over rapidly with no wings level control. For the fixed gain system with the wings level, circuit results were acceptable. With the fixed gain, nonlinear C_{n_β} , and the wings level circuit in, the aircraft rolls over. Lateral acceleration, tailload, and sideslip are excessive.

With the wings level, fixed gain, nonlinear C_{n_β} and C_{l_β} , the aircraft again rolls over. The ARI was changed (.3, .5, .7), but it produced little differences in the traces. For the adaptive system, engine out with wings level circuit in and no nonlinearities, the aircraft recovers quickly. However, with the wings level circuit in, engine out and nonlinear C_{n_β} , the aircraft slowly rolls over. Sideslip was held to 16°. With the nonlinear C_{n_β} and C_{l_β} , the wings level circuit in the aircraft rolls out of control. With the fixed gain for engine out flown from the cockpit, the aircraft was quite controllable with no nonlinearities in the circuit; however, rudder was used. With the nonlinear C_{n_β} , the pilot was again able to control the aircraft utilizing rudder. The same performance held true for nonlinear C_{n_β} and C_{l_β} . Generally, the aircraft was controllable for this condition. The adaptive system appeared easier to fly than the fixed gain system. The simulated pilot (wings level) significantly degrades performance. Variation in derivatives for this flight case did not produce significant results.

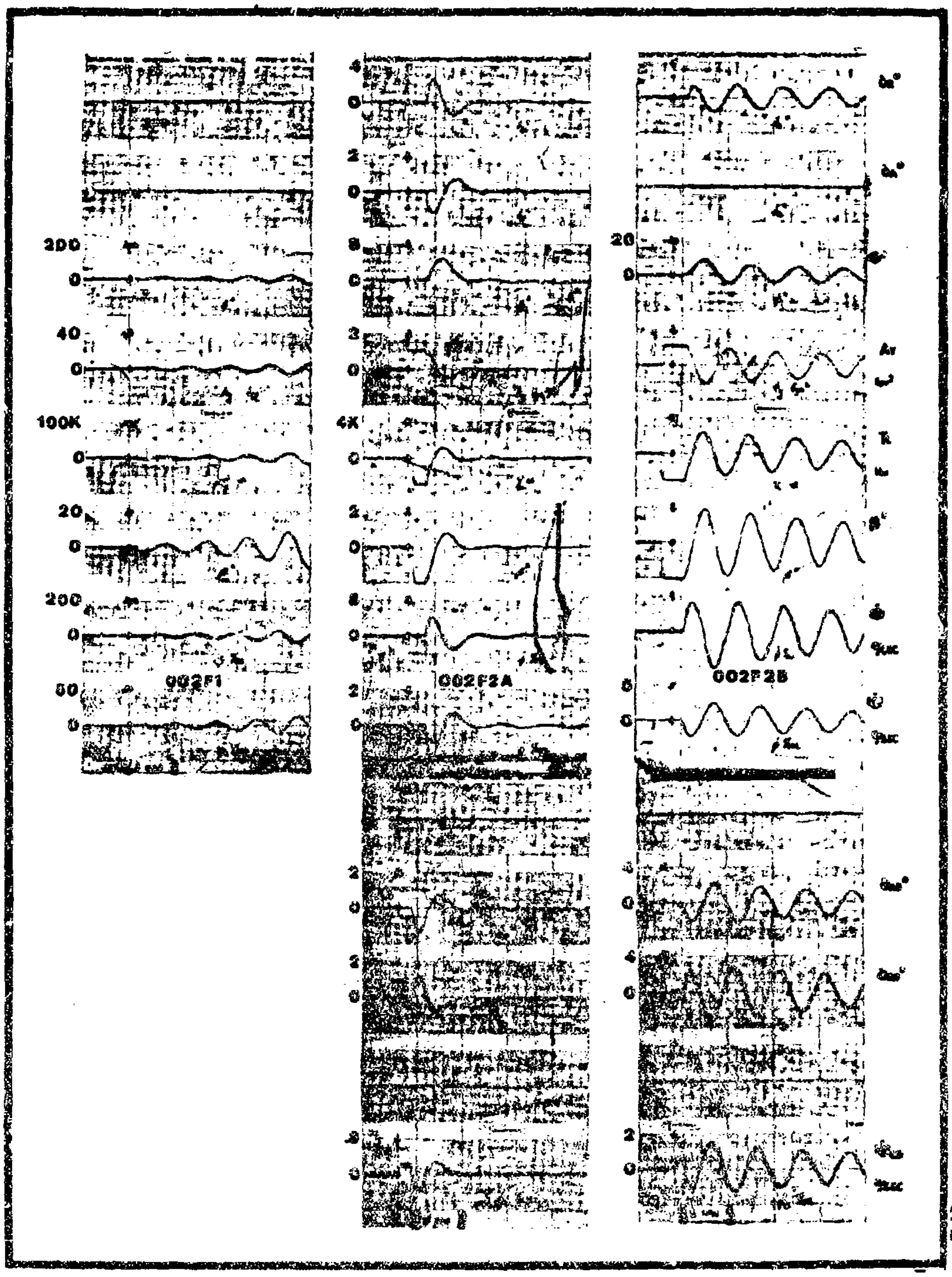


FIGURE 11 INITIAL SIDESLIP

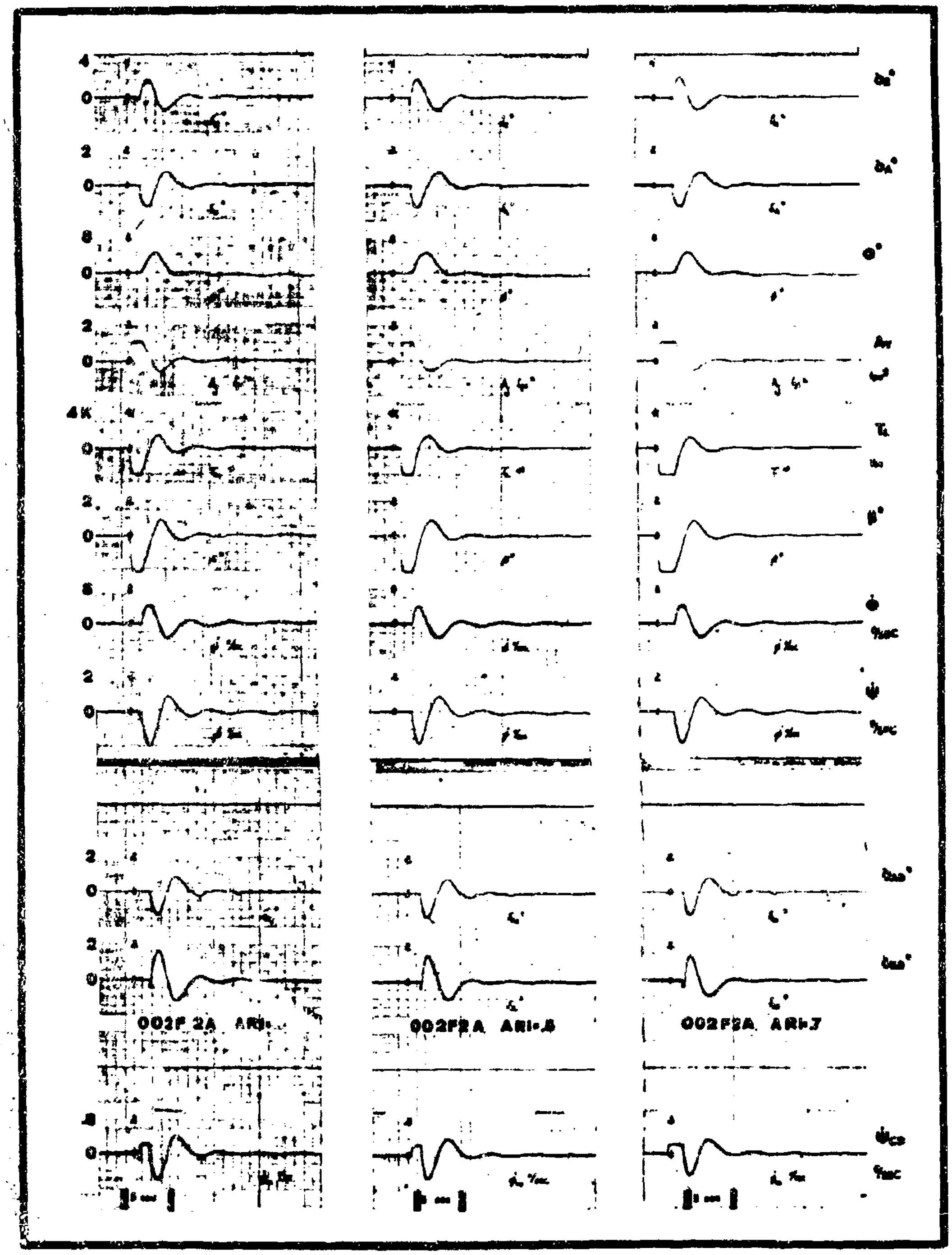


FIGURE 12 ARI VARIATION, INITIAL SIDESLIP

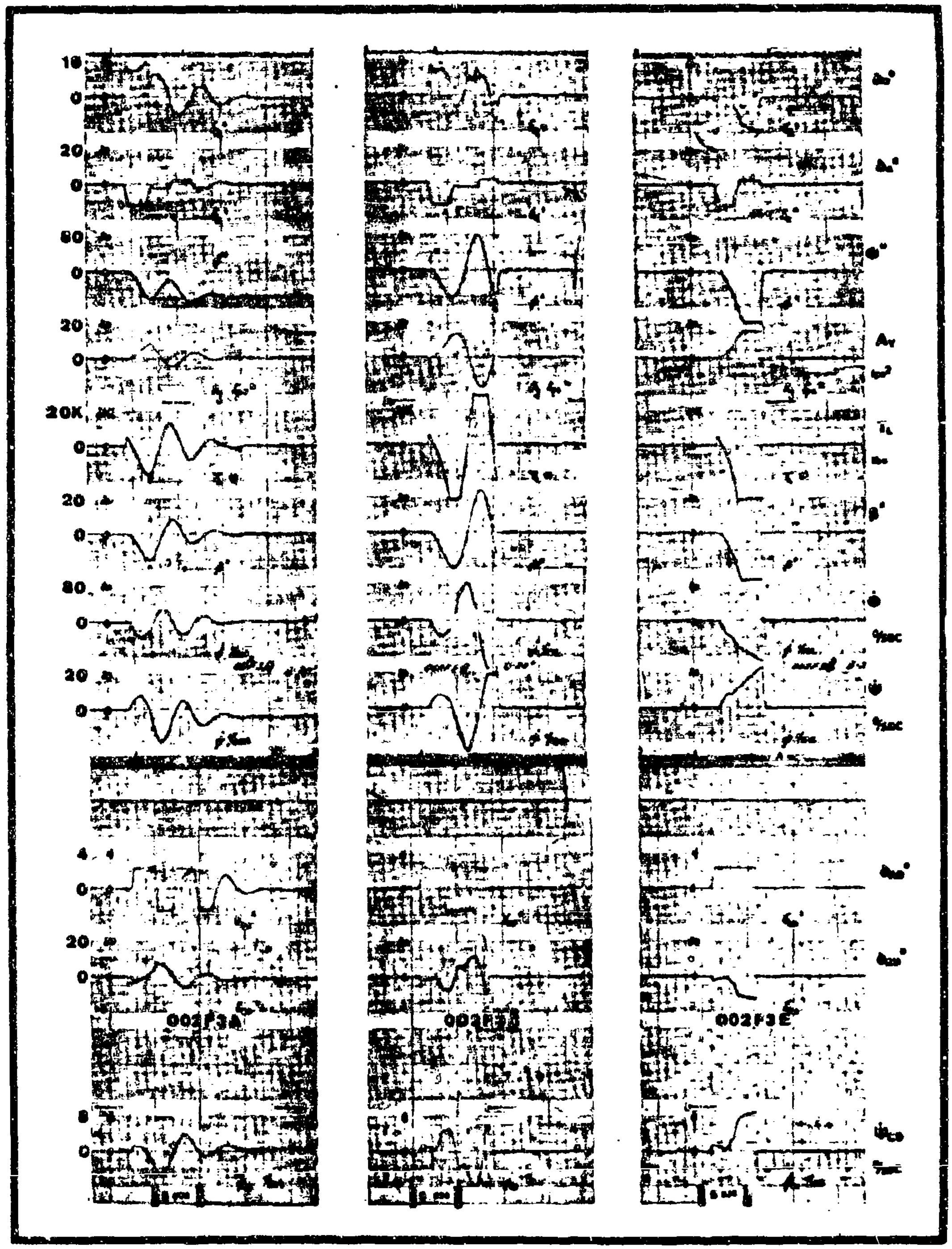


FIGURE 13 RPO MANEUVER (SLOPES = 0 at $\beta = 20^{\circ}$)

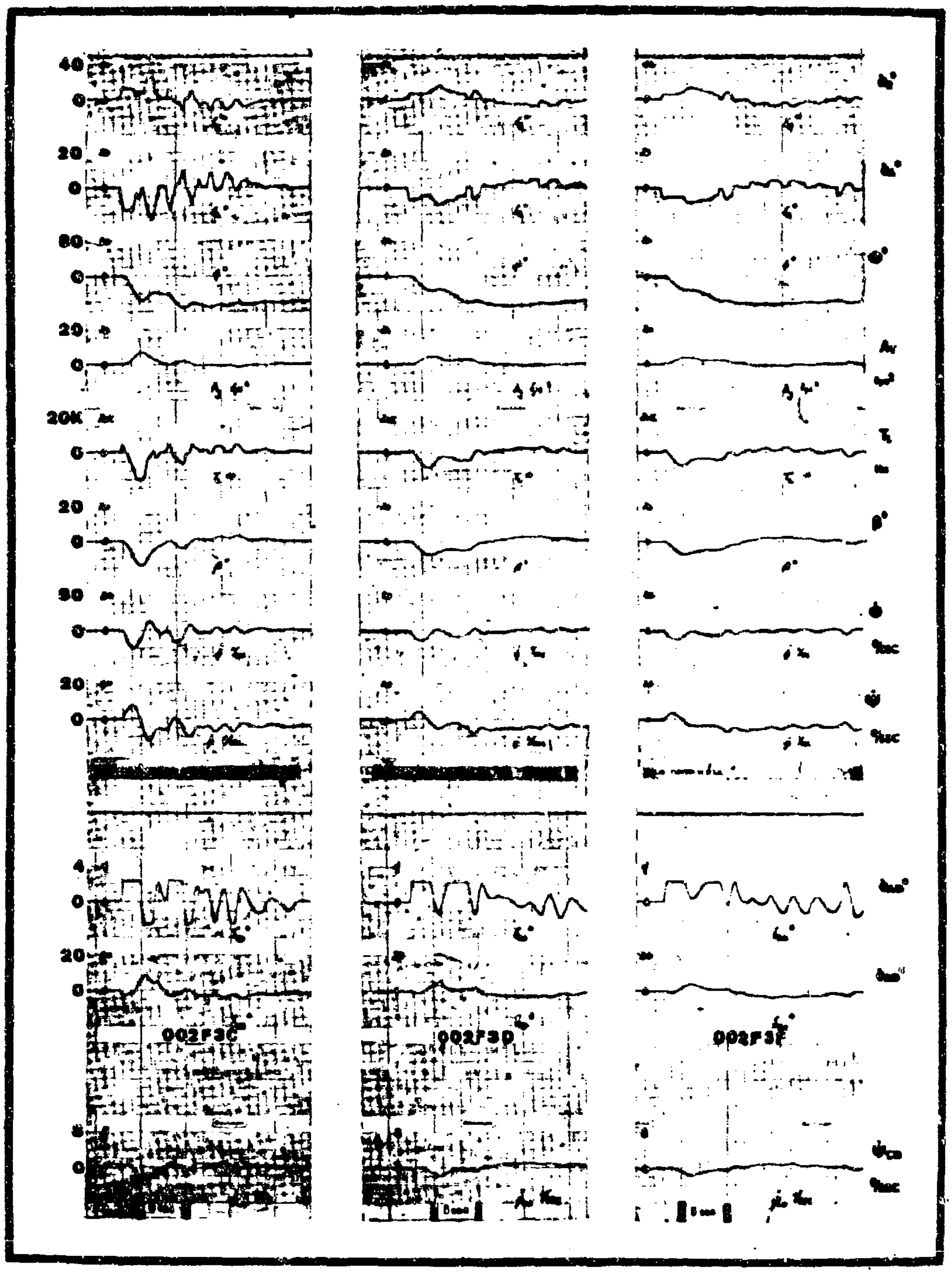


FIGURE 14 HPO WITH PILOT (SLOPES = 0 at β = 20°)

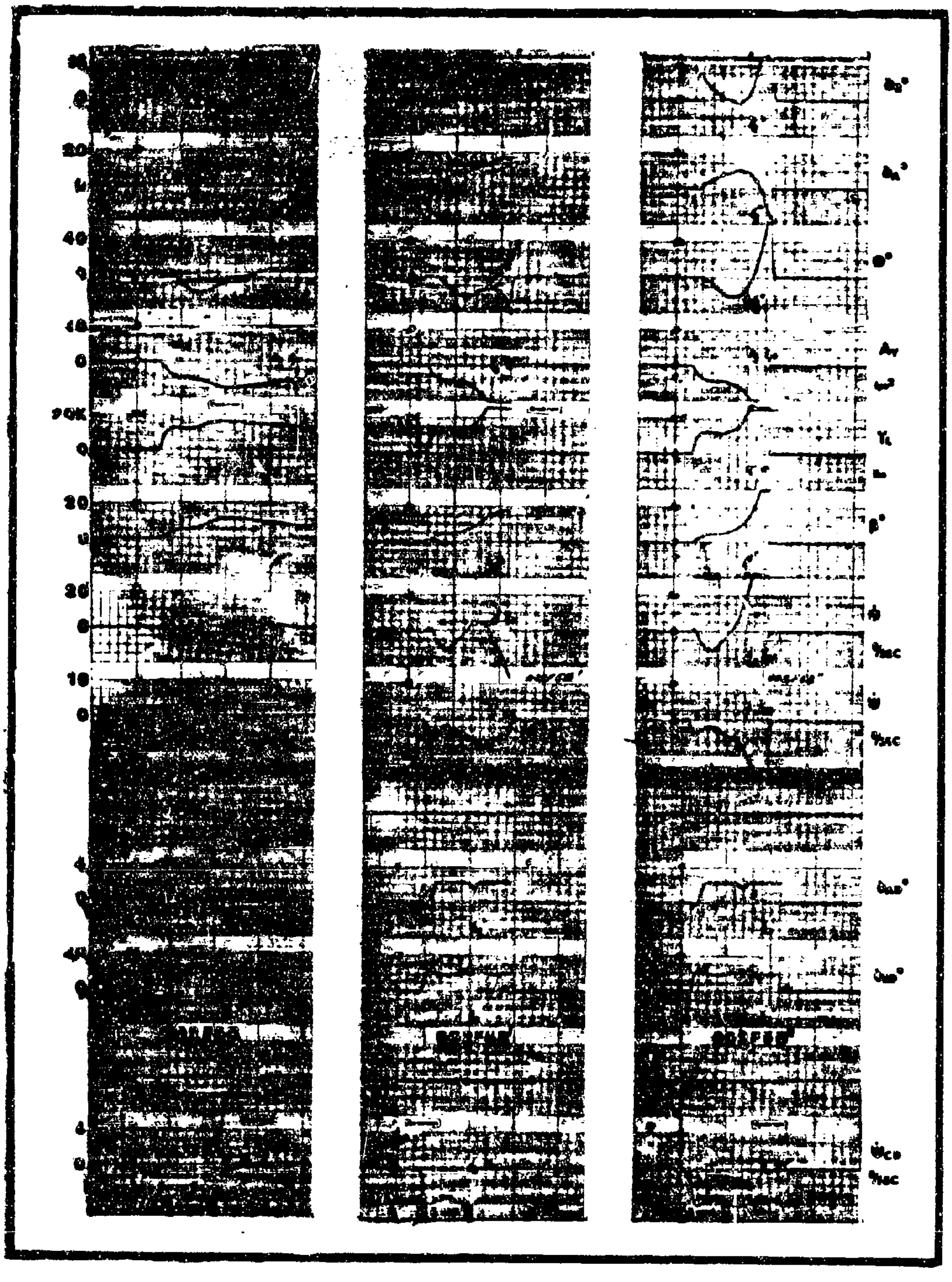


FIGURE 15 ENGINE OUT WITH WINGS LEVEL CIRCUIT (SLOPES = 0 at $\beta = 20°$)

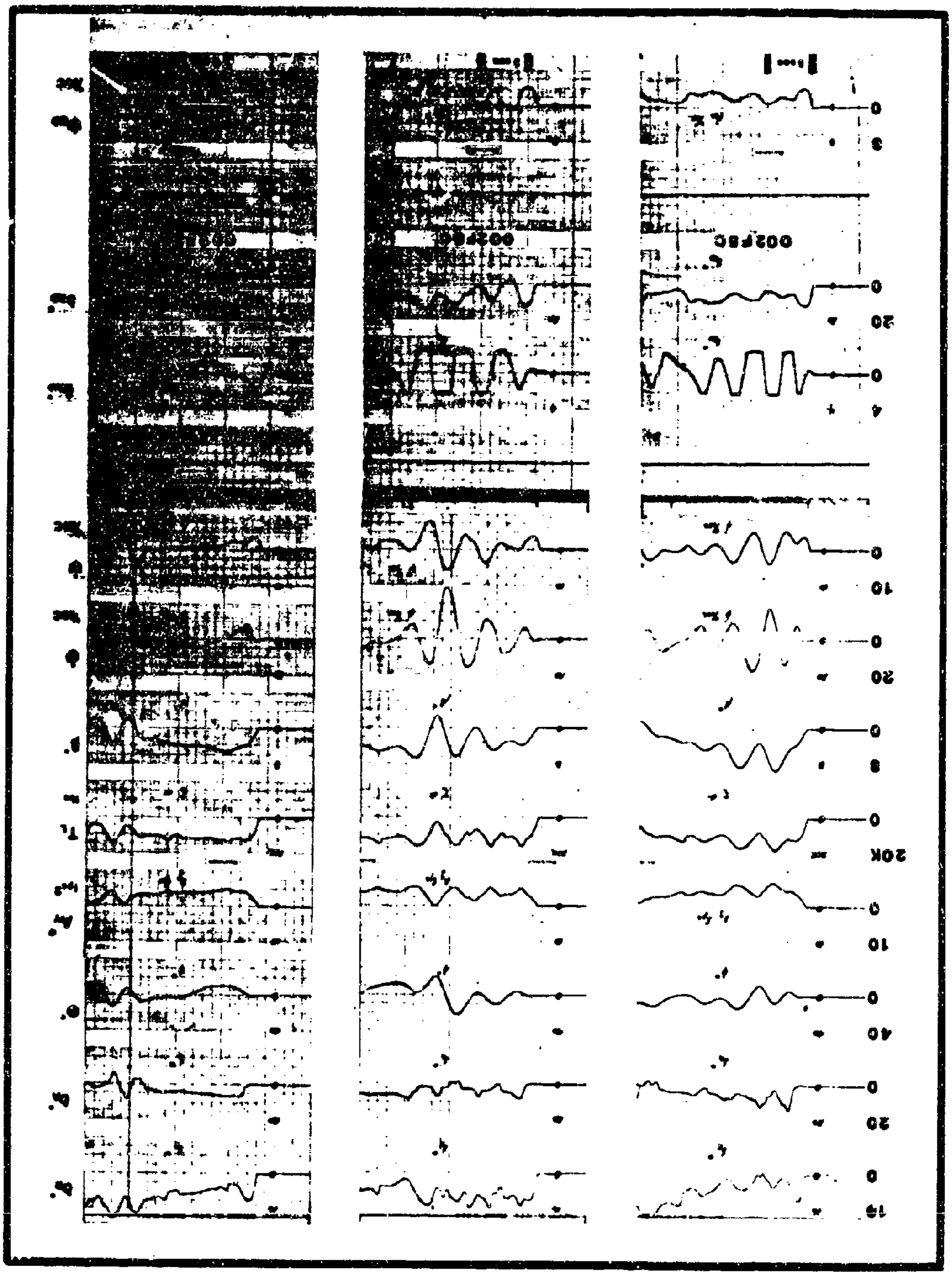


FIGURE 16 ENGINE OUT WITH PILOT (SLOPES = 0 at $\beta = 20°$)

Flight Condition 004

This flight condition is one of the two for which General Dynamics' time histories were available. Generally, the responses obtained in this study matched those provided by the contractor.

At this flight condition, the free airframe is unstable. With both roll and yaw augmentation operating, the fixed and the adaptive gain systems exhibited acceptable response to a two-degree initial sideslip input. With roll damper out, both configurations were lightly damped.

The rolling-pullout (RPO) maneuver, as performed by the simulated pilot, matched the contractor's time histories for both the fixed gain and adaptive gain systems. The sideslip obtained during the RPO was roughly 15° (without aerodynamic nonlinearities). With nonlinearities introduced, the results were considerably different. For the fixed gain system with C_{n_β} going to 0 at 15° β and also at 20° β, the system is stable; however, excessive rolling velocity reversal occurs and considerable difficulty exists in rolling from 0 to 60°. In general, performance of the RPO maneuver is poor. In attempting to fly from the cockpit with the same nonlinearities, it was not possible to precisely control the aircraft in a roll from 20° to -20°. The same comments are also true for the adaptive system. When both C_{n_β} and C_{l_β} were set up to be nonlinear, the results were worse. The aircraft was very difficult to control for all attempts to fly from the cockpit. Preliminary runs were made with roll control only; and after the first sequence of runs, the pilot also utilized the rudder pedals. Well coordinated rudder inputs provided significant improvement in controllability. Sideslip was held to lower amplitudes; and consequently, the aircraft did not go out of control as quickly. A $2^\circ \beta$ gust input was introduced to the system, and the results were insignificant; but for a $15^\circ \beta$ gust (nonlinear C_{n_β}), the system diverged. The fixed gain and the adaptive gain systems exhibited equivalent responses to the gust. For the fixed gain system, simulated pilot in the loop, (wings level circuit) rolling and sideslip were not severe for engine failure. Engine failure was also checked with nonlinear C_{n_β} and C_{l_β} . The aircraft is unflyable with nonlinearities included. For the engine out test, there was little difference in performance between the fixed gain and the adaptive gain systems. Without either nonlinearity and a pilot "flying" from the cockpit, rudder pedal input was required to prevent divergence. With nonlinear C_{n_β} and C_{l_β} , the aircraft was unflyable, the controlling factor being the excessive sideslip which exceeded 20° in $3\frac{1}{2}$ seconds. With the fixed gain system and nonlinear C_{n_β} , the aircraft was flyable if rudder pedal was applied. The rudder damper saturated at 20° for pilot controlling wings level only and the aircraft diverged. Results for the adaptive system were comparable; again sideslip and pilot rudder input being the dominating characteristics. A sensitivity analysis was conducted by varying C_{n_r}, C_{l_p} , C_{l_β} , and C_{n_β} . (Accomplished by changing the potentiometer settings.) There did not appear to be any appreciable difference in the system response to small initial β inputs.

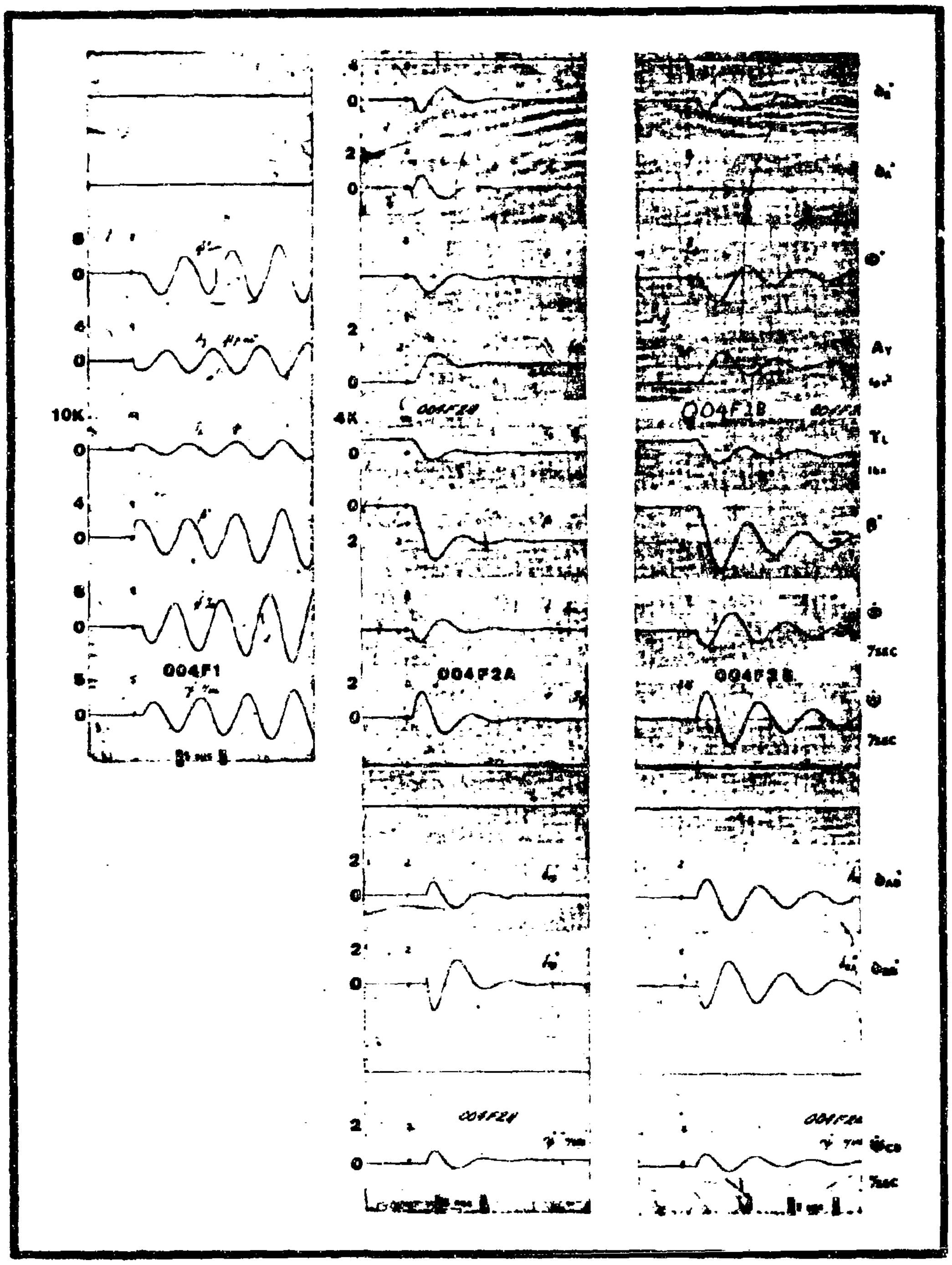


FIGURE 17 INITIAL SIDESLIP

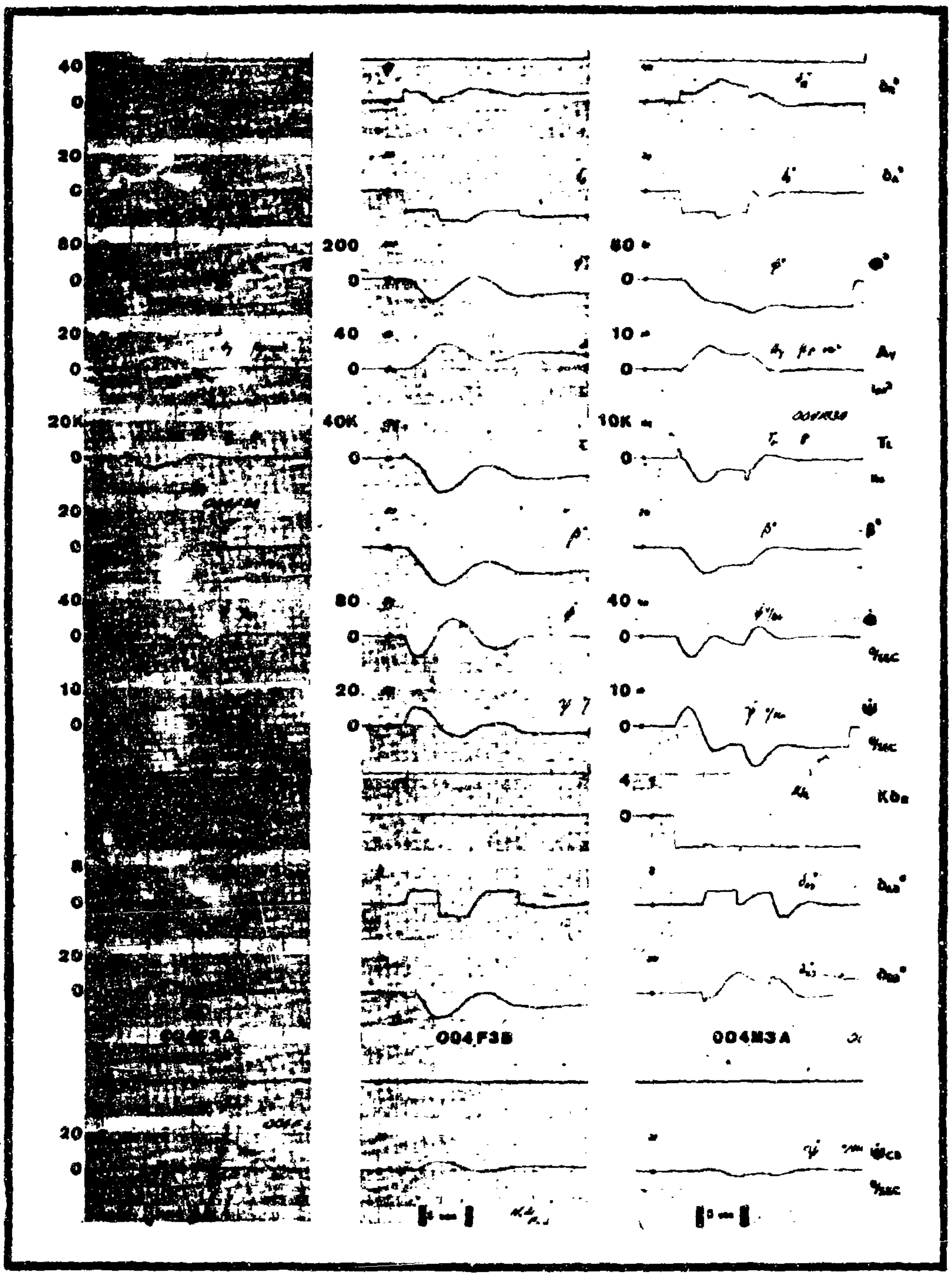


FIGURE 18 RFC MANEUVER (SLOPE = 0 at $\beta = 15^{\circ}$)

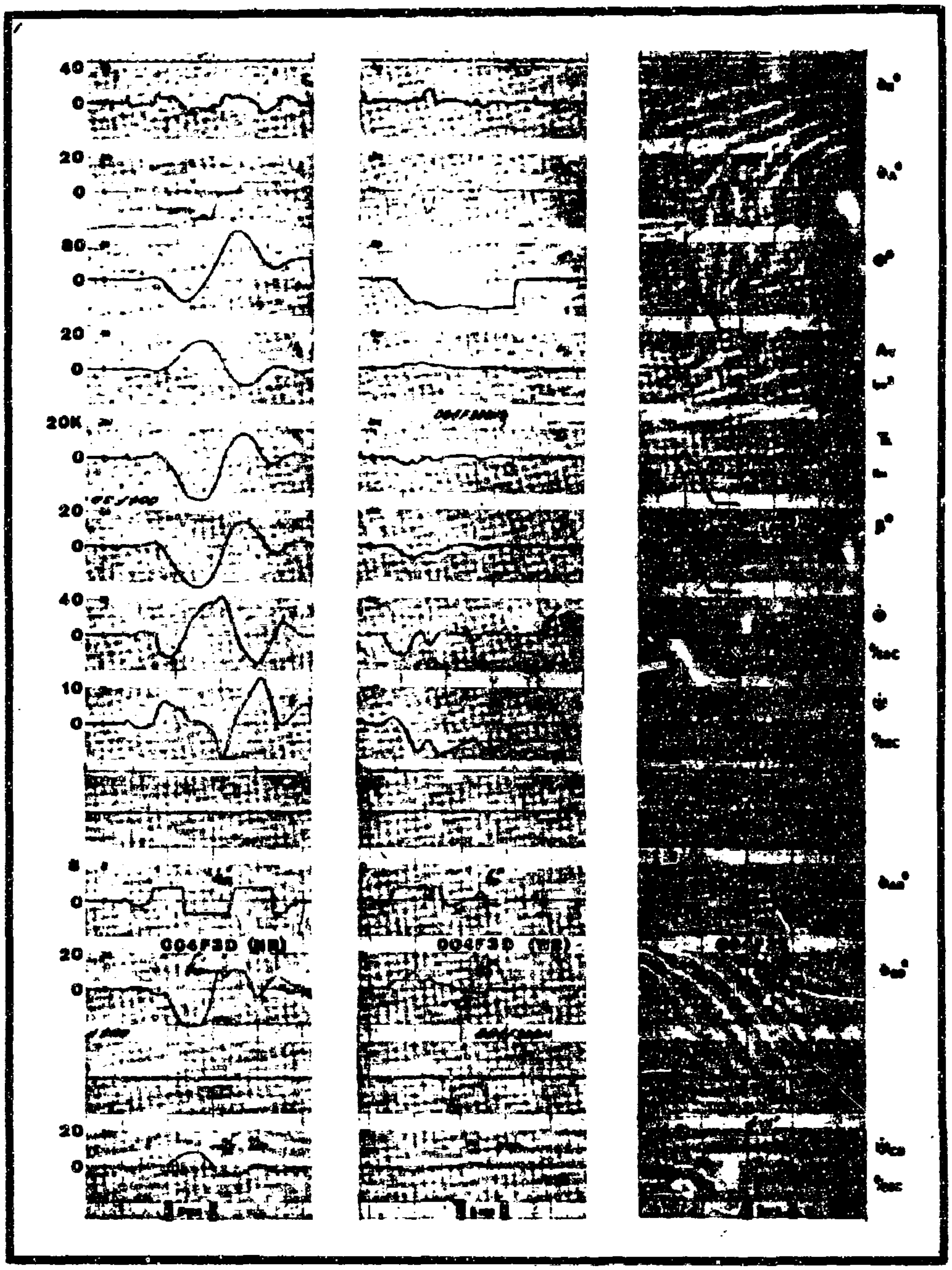


FIGURE 19 RFO WITH PILOT (SLOPES = 0 at $\beta = 15^{o}$)

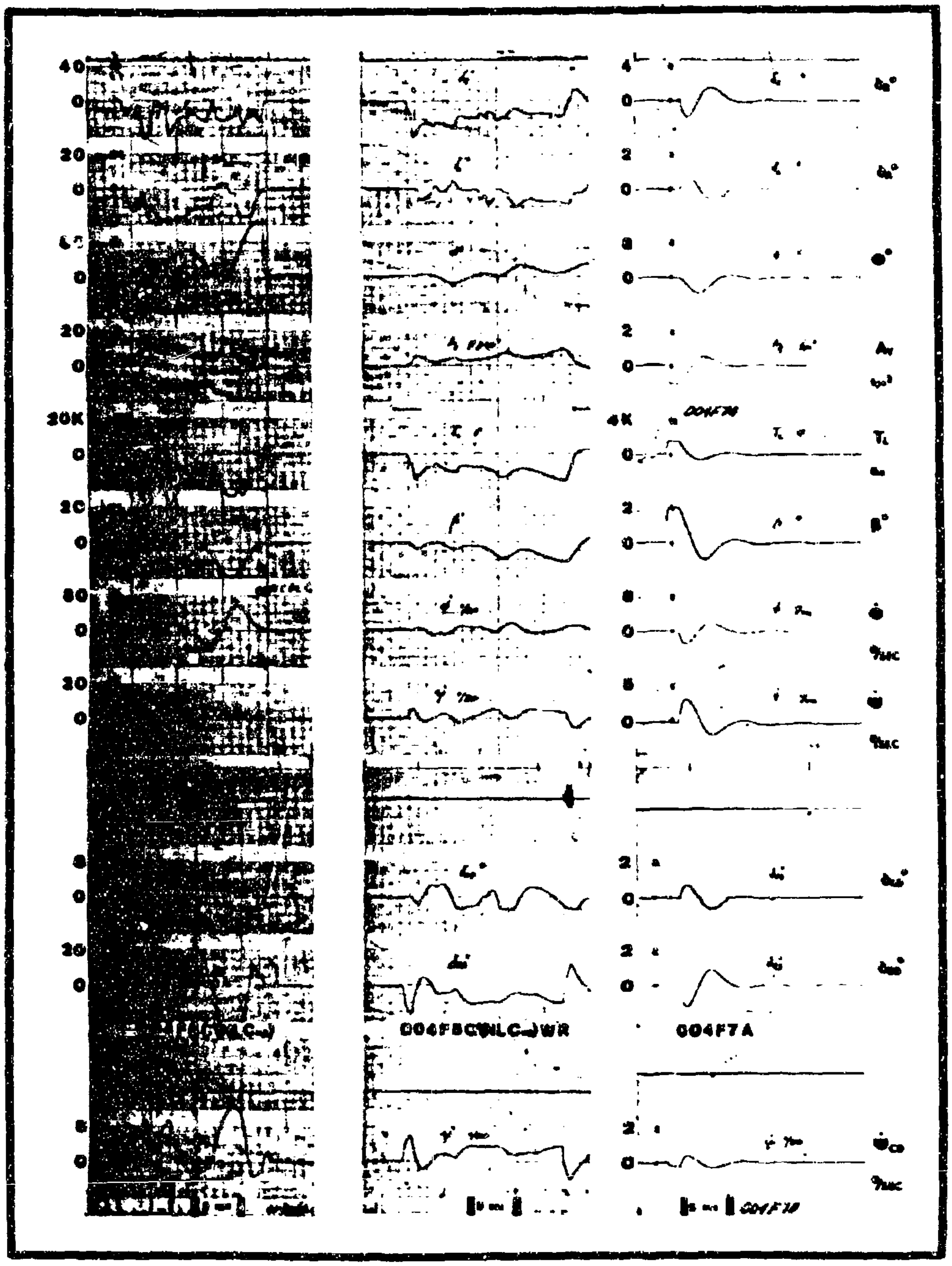


FIGURE 20 ENGINE OUT WITH PILOT (SLOPE = 0 at $\beta = 20^{\circ}$)

Flight Condition 006

The free airframe responses agreed with the data found in the General Dynamics data requirements document (Reference 1) used as reference for verification of system performances throughout this simulation. For the damped airframe, fixed gain system, all dampers on, responses are satisfactory. Variation of the Aileron Rudder Interconnect (ARI) did not appear to change the damped airframe traces. On this flight case, the roll damper out trace appears to be satisfactory . . the fixed gain system, and the ARI was again changed to .3, .5, and .7 with no appreciable difference indicated in the traces. The RPO maneuver was successfully accomplished with the fixed gain system. For the RPO maneuver with the actual pilot in the loop, there was no significant sideslip. For the 0 to 60° RPO maneuver with the adaptive system, the response looked quite good. With the pilot "flying" from the cockpit, no problems were encountered.

For this flight case with the fixed gain, the engine out responses were run for the aircraft, ARI set at .3, .5, and .7. These changes were made without significantly altering the basic trace for the ARI = 1.0, which was satisfactory.

With the engine failure and the adaptive system, the aircraft rolls over quite rapidly without the wings level circuit.

With the wings level circuit in, the system appears to be lightly damped with residual oscillations.

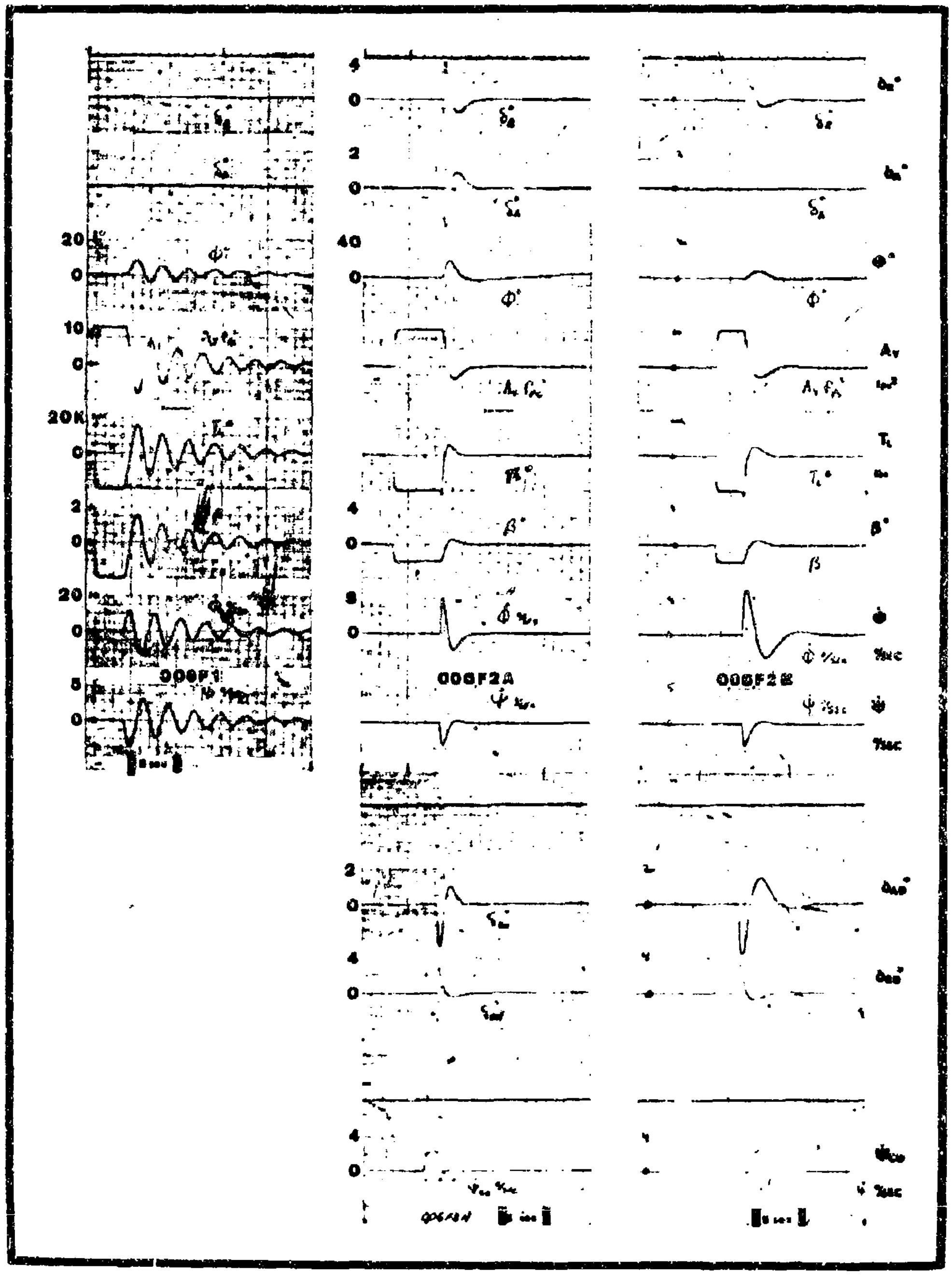


FIGURE 21 INITIAL SIDESLIP

Flight Condition 008

The free airframe is very lightly damped. The damped airframe with the fixed gain system was satisfactory. Variation of the ARI apparently had no effect with respect to the aircraft response for a 2^{o} initial β input. The aircraft response with the roll damper out was only moderately damped although satisfactory. The response to a 2^{o} initial sideslip input for the adaptive system, roll damper on, looks slightly better than the fixed gain response. Again the ARI variation had no effect. The rolling-pullout maneuver for the fixed gain system looks good; responses were satisfactory with regards to spec requirements. Again in this condition, the roll damper authority was inadvertently limited to slightly less than 3^{o}; however, the data obtained with the dampers on is still considered valid. "Flying" the rolling-pullout maneuver from the cockpit was relatively easy in this flight condition, although as in the other cases, it was somewhat touchy and dependent upon the pilot applied rudder. For both the fixed gain and the adaptive gain system, the response of the aircraft to low amplitude gust inputs was acceptable. In the engine out tests, both systems performed satisfactorily. The airplane was quite flyable from the cockpit. There was no saturation in the yaw channel. The engine out cases were run with ARI variations, and again no noticeable effect was found. The parameter variation study produced no noticeable effects (the derivatives were varied $\pm 20\%$ in this case).

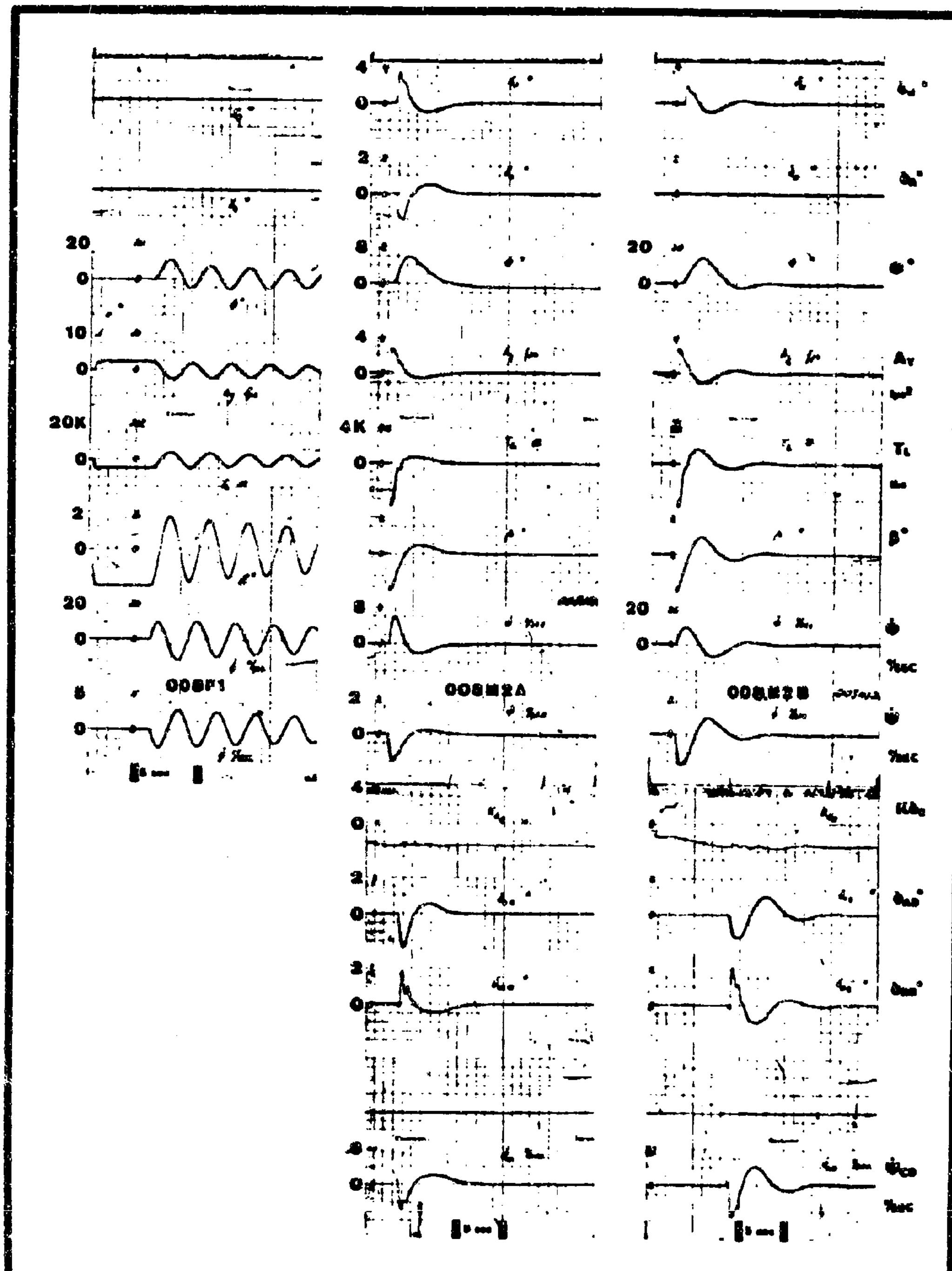


FIGURE 22 INITIAL SIDESLIP

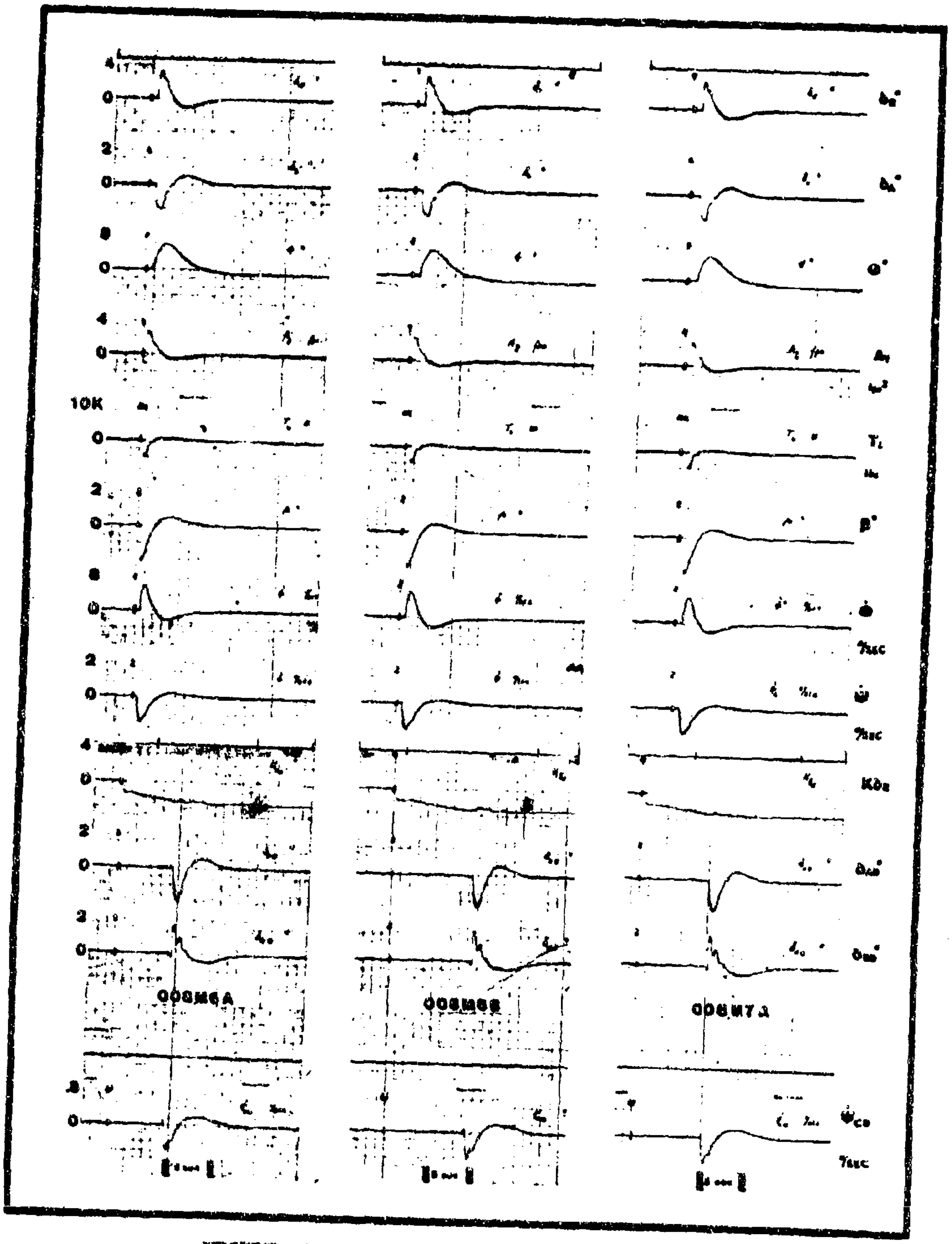


FIGURE 23 VARIATION OF DERIVATIVES

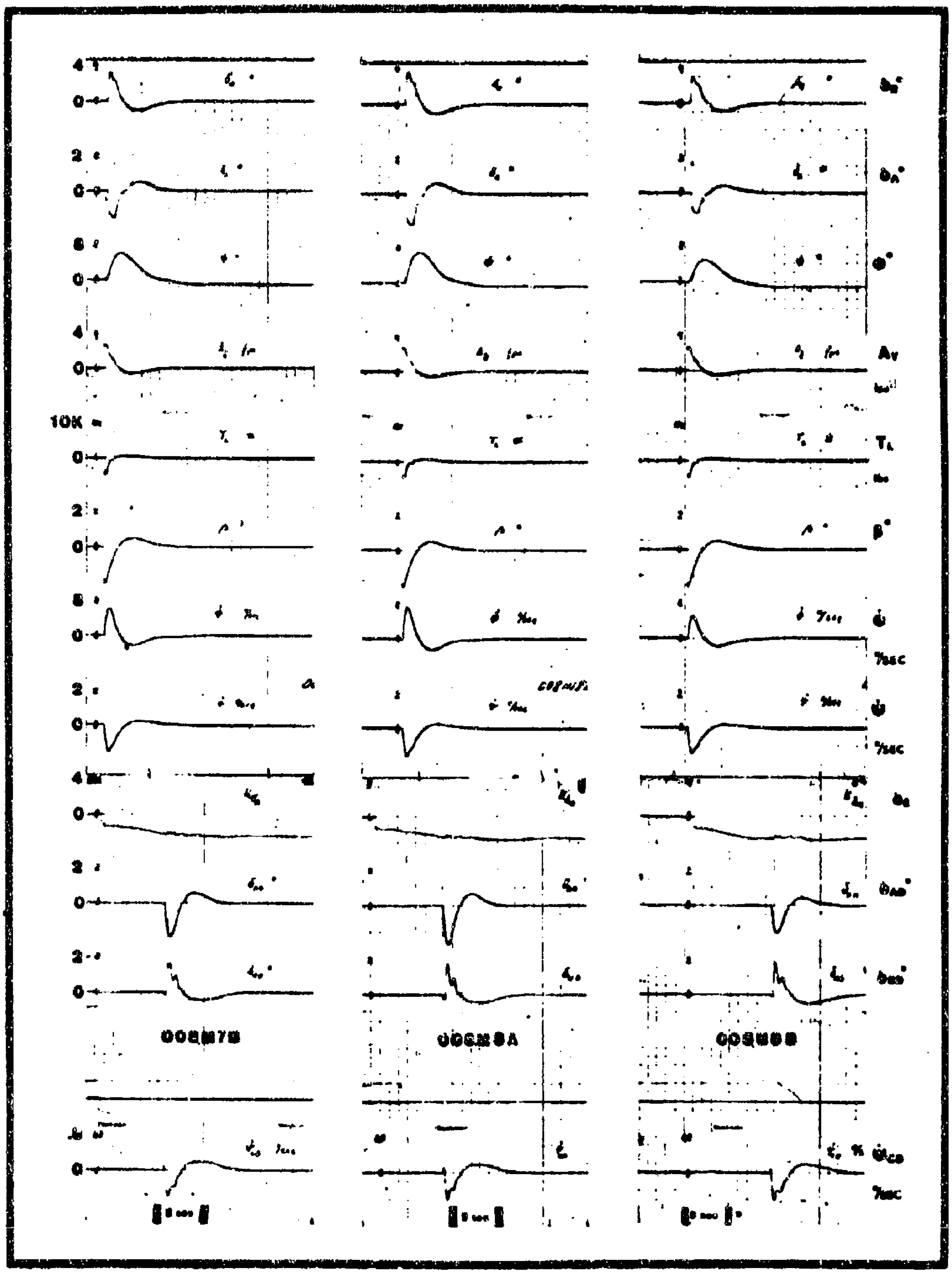


FIGURE 24 VARIATION OF DERIVATIVES

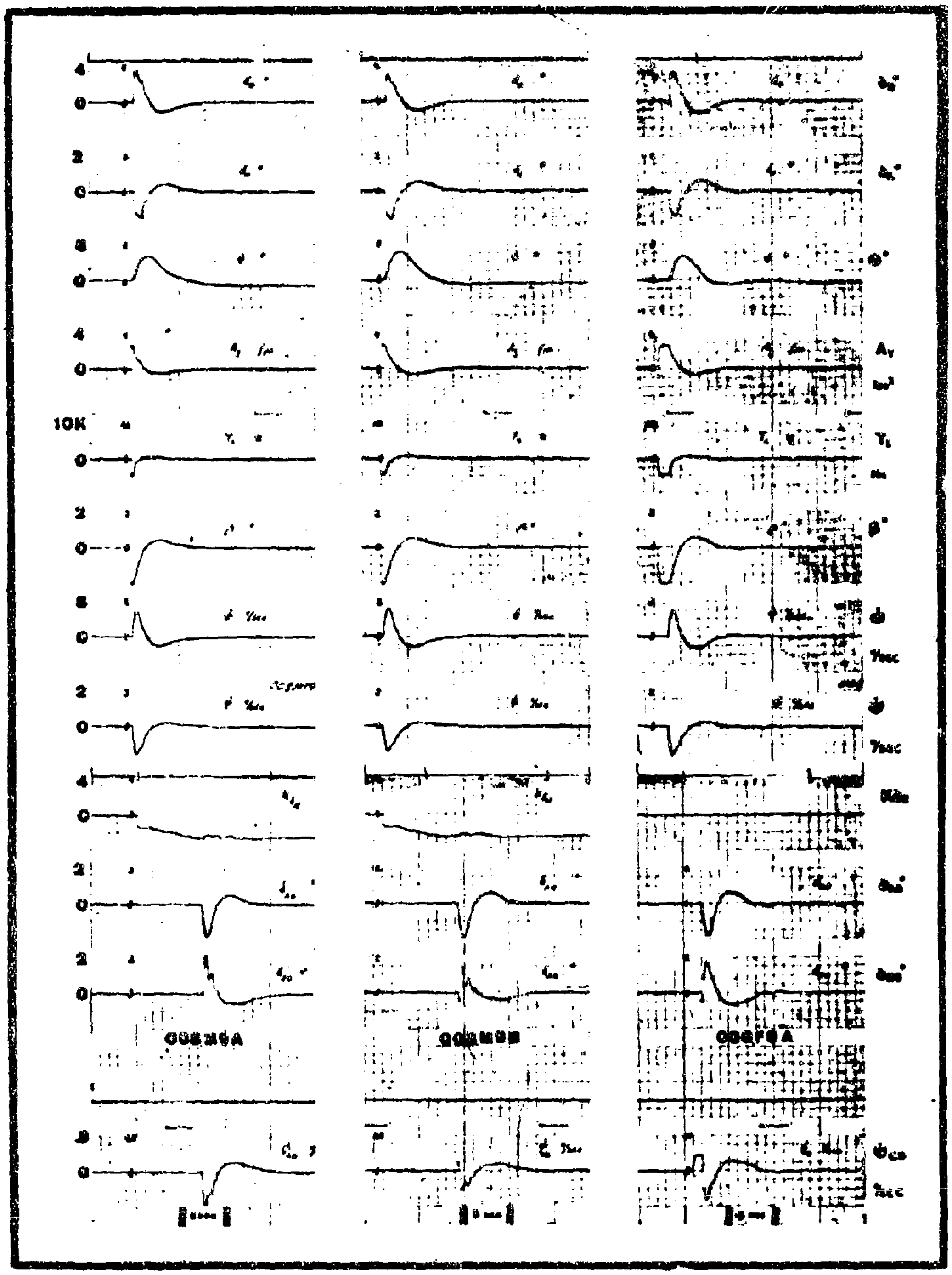


FIGURE 25 VARIATION OF DERIVATIVES

Flight Condition 010

The free airframe is heavily damped, and the period and the transient peak ratio agrees with the printed digital data received from General Dynamics. For fixed gain, (damped airframe) the β initial condition responses are heavily damped. Changing the mechanical aileron rudder interconnect to .7, .5, and .3 did not appear to have any major effect on the damped airframe as the traces remained about the same. For the fixed gain yaw damper with the roll damper out, traces are satisfactory. Changing the rudder-aileron interconnect again did not cause any significant change in response. The adaptive system with roll damper in and also with roll damper out performed satisfactorily, giving well damped responses. No ARI changes were recorded for the adaptive system. For the RPO maneuvers with the simulated pilot in the loop, the traces are satisfactory. The roll attitude holds at 60° indicating no spiral divergence. Sideslip, tailload, and lateral acceleration are small for this flight condition. For the fixed gain system, pilot in the cockpit, and the nonlinear C_{n_β} and nonlinear C_{l_β}, the system exhibited good flying characteristics. For the RPO maneuver with the adaptive system, the traces are satisfactory. The nonlinearities did not have any effect. Pilot in the loop characteristics were satisfactory. Sideslip, tailload, and lateral acceleration were small (β did not attain sufficient amplitude to enter the nonlinear approximation).

No single engine failures were run for this flight condition. In general, a pronounced tendency for ringing was noted at this flight condition.

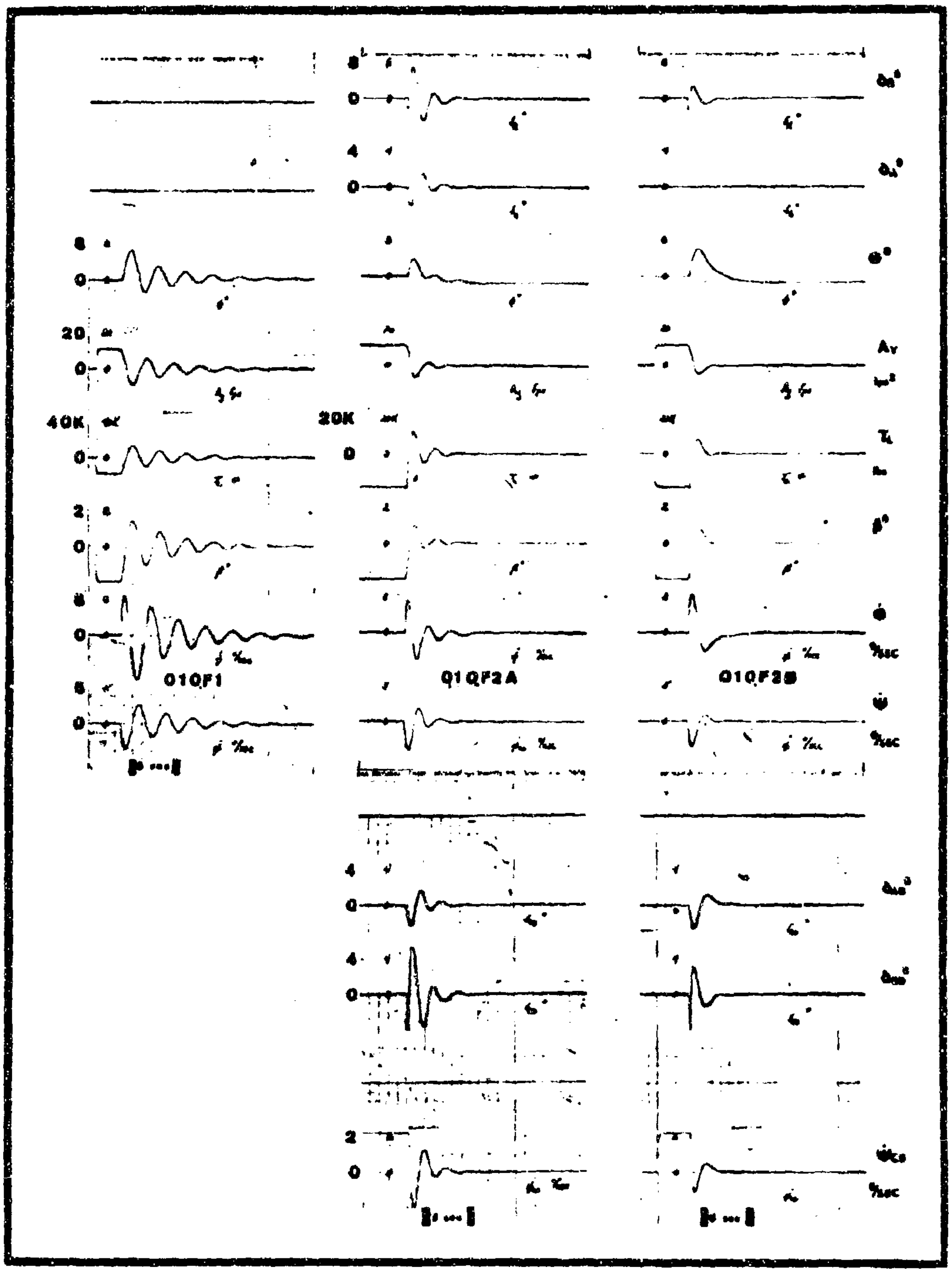


FIGURE 26 INITIAL SIDESLIP

Flight Condition 011

The free airframe is lightly damped. Response of the airframe, with either the fixed gain or the adaptive gain system, roll and yaw augmentation on, looks quite good. The airplane has a tendency to roll off, indicating spiral divergence. The cases for the roll damper out appear to be well damped. Variation of the mechanical aileron-rudder interconnect had negligible effect for the 2° β initial condition. For both the fixed gain and the adaptive systems, the rolling and pullout maneuvers were satisfactory (sideslip amplitude was small). For this flight condition, the roll damper limits drifted to approximately 2.5° which is less than the actual 3° damper authority. While performing the rolling-pullout maneuver from the cockpit, it was noted that the airplane was sensitive to control inputs, tending to cause over-control. The system was stable in the maneuver, and the sideslip amplitudes were reasonably low. Utilization of the rudder pedals significantly improved the performance. The low amplitude gust input to the simulation was handled satisfactorily by both systems. The aircraft responses to engine failure for both the fixed gain and the adaptive systems were satisfactory. Tailloads, lateral acceleration, and sideslip were acceptable. There was no problem in controlling the aircraft when "flying" the simulation through the cockpit. Generally, no significant effect was noted for the variation of stability derivatives with small amplitude disturbance inputs.

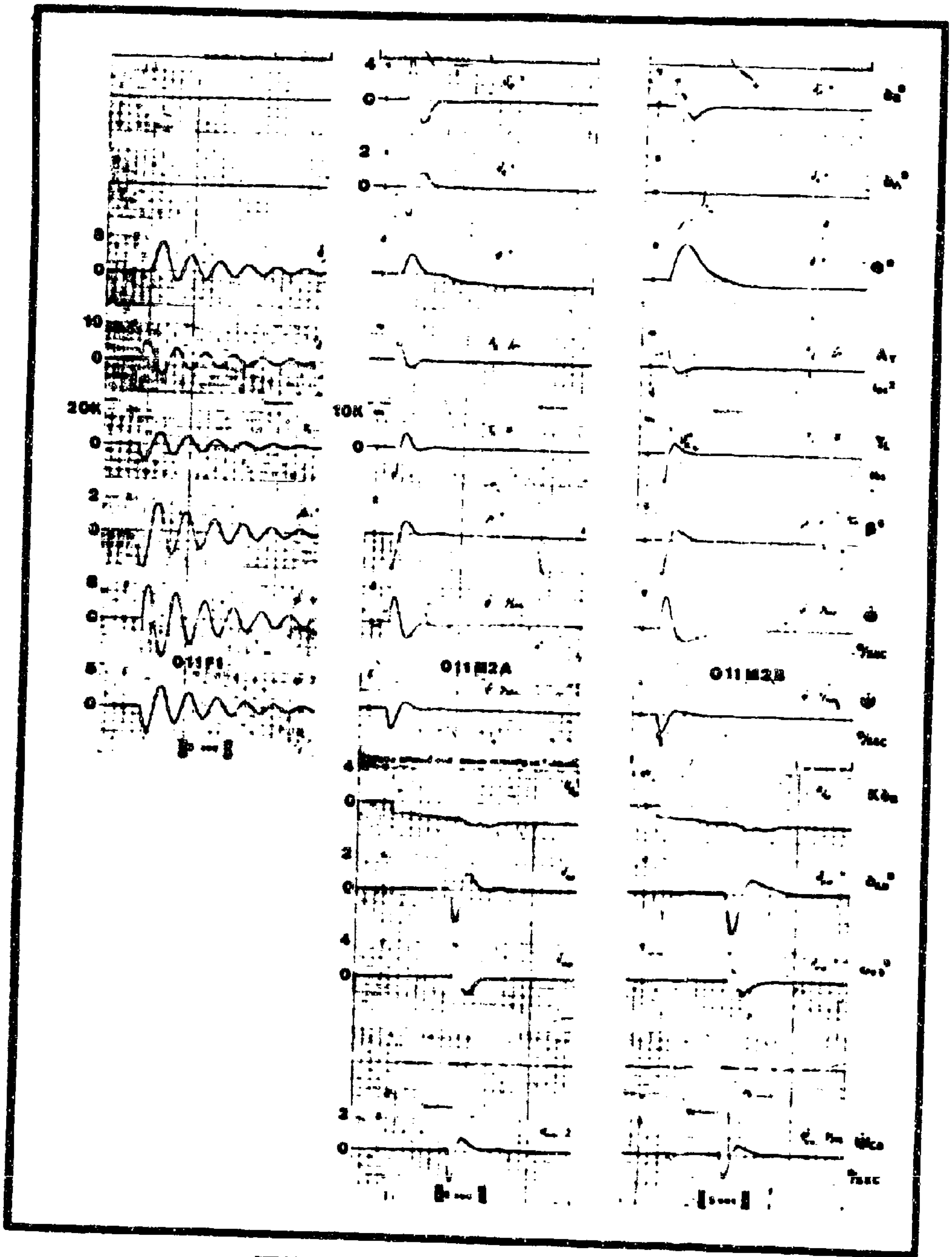


FIGURE 27 INITIAL SIDESLIP

Flight Condition 013

This is the second case for which we had General Dynamics time histories to compare with ours. Complete agreement was obtained. For this flight condition, the free airframe is moderately damped. With the fixed gain yaw augmentation, roll damper in, the response to a 2° initial β is well damped. This is also true with the roll damper out.

For the fixed gain RPO maneuvers, the system performed quite well; the pilots (human and computer) had no trouble accomplishing the $0^\circ - 60^\circ$ roll meneuver and holding. Tailload, sideslip, and lateral acceleration were small. The 2° β gust was used to disturb the system, but the results were insignificant. With the fixed gain system for the engine out run with no pilot, the aircraft rolled over quite rapidly. With the simulated pilot in the loop (wings level circuit), response to the engine failure is acceptable with slight ϕ, β, T_L, and A_Y. It is significant to note that large (over 15°) yaw damper inputs were required to compensate for engine failure.

In general, the adaptive system demonstrated similar performance.

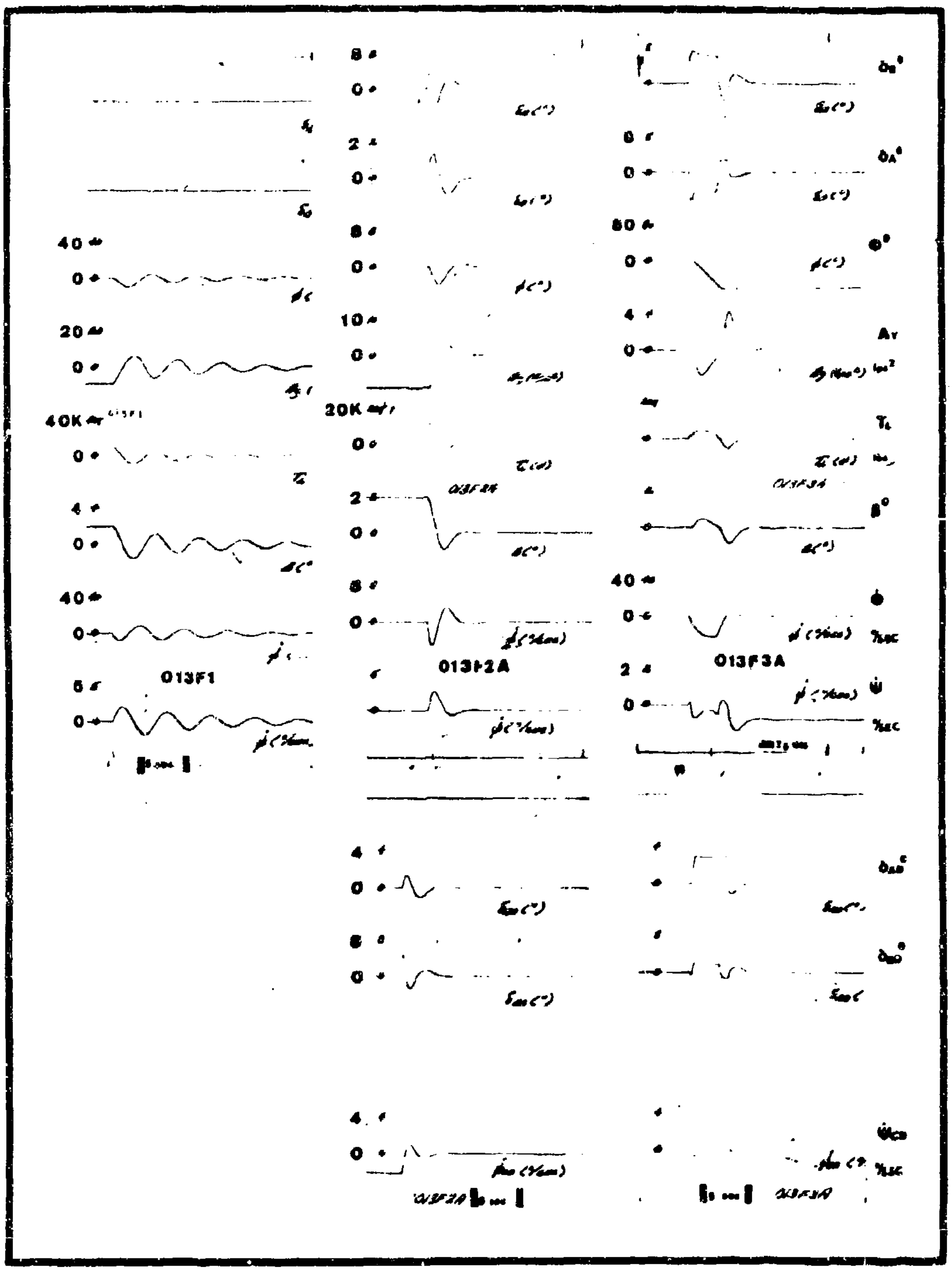


FIGURE 28 INITIAL SIDESLIP

Flight Condition 013

This is the second case for which we had General Dynamics time histories to compare with ours. Complete agreement was obtained. For this flight condition, the free airframe is moderately damped. With the fixed gain yaw augmentation, roll damper in, the response to a 2° initial β is well damped. This is also true with the roll damper out.

For the fixed gain RPO maneuvers, the system performed quite well; the pilots (human and computer) had no trouble accomplishing the 0° - 60° roll maneuver and holding. Tailload, sideslip, and lateral acceleration were small. The 2° β gust was used to disturb the system, but the results were insignificant. With the fixed gain system for the engine out run with no pilot, the aircraft rolled over quite rapidly. With the simulated pilot in the loop (wings level circuit), response to the engine failure is acceptable with slight ϕ, β, T_L, and A_Y. It is significant to note that large (over 15°) yaw damper inputs were required to compensate for engine failure.

In general, the adaptive system demonstrated similar performance.

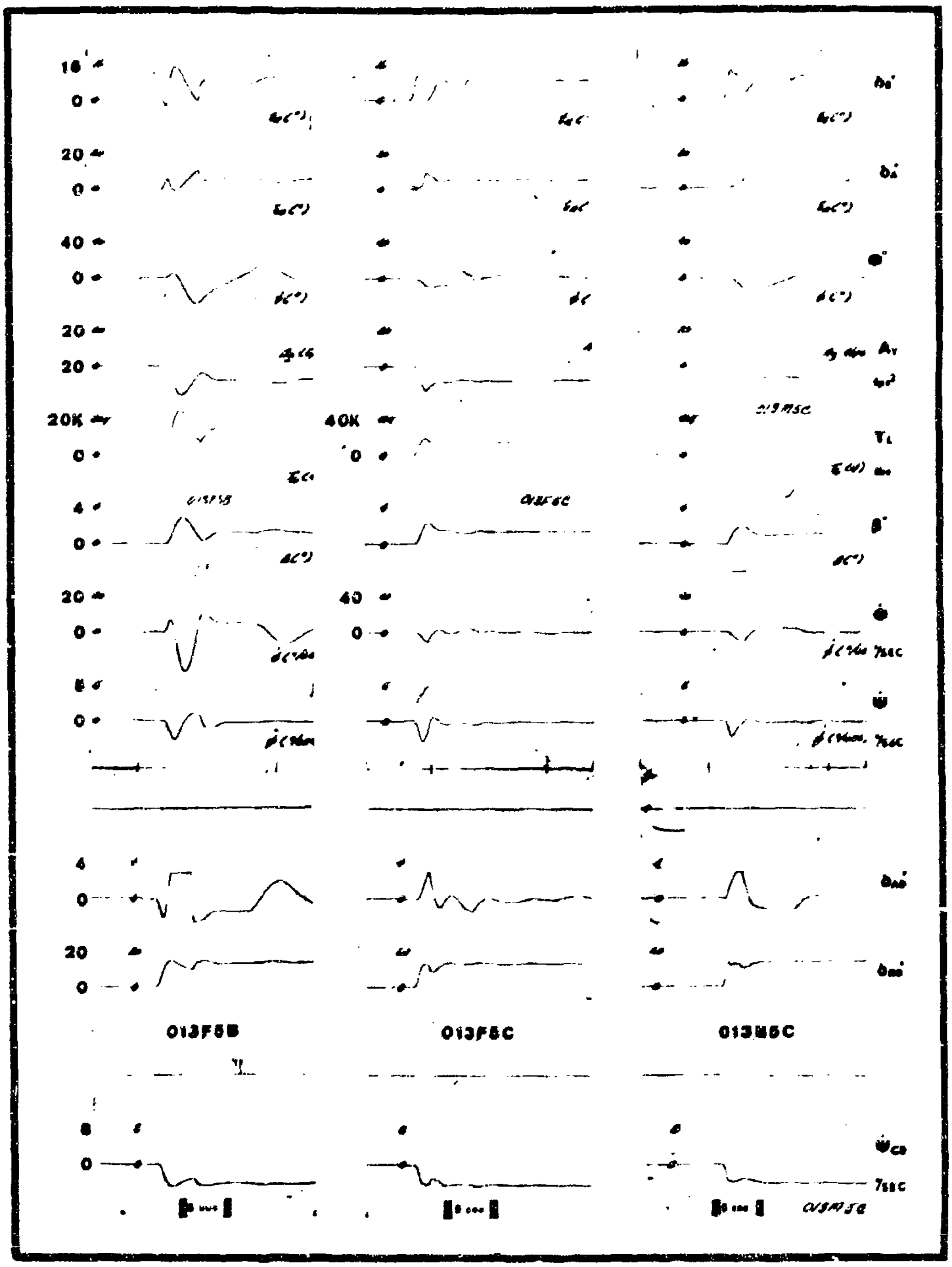


FIGURE 29 ENGINE OUT

CONCLUSIONS

Air Force and contractor simulation studies to date (including this study) have not been adequate for accurate flight safety determination and handling qualities evaluations of the B-58 aircraft.

Use of GD specified transfer functions for simulating the pilot in specific maneuvers and emergency recovery tasks in varying flight conditions is invalid. Assumptions of small disturbances and linear aerodynamics are not appropriate for the B-58.

For the problems encountered with the B-58, a sophisticated motion simulator with appropriate external visual cues is considered an essential piece of equipment for obtaining a very high degree of confidence in ground based simulation results.

Neither of the evaluated control systems is acceptable as mechanized. Both systems exhibit amplitude sensitivity (General Dynamics study). The adaptive system gain drives down for random input signals (Bendix study). Predicted aerodynamic characteristics are still undergoing change which leaves the adequacy of the fixed gain system in doubt. Both systems exhibit undesirable characteristics for low speed, high angle of attack flight when subjected to large disturbance inputs (large aileron deflection, sudden engine failure and large amplitude gust disturbances). Unsatisfactory definition of structural modes leaves the question of the adaptive system interaction with these modes unanswered.

The low dutch roll damping demonstrated by both systems at flight condition 002 when the roll damper was out is not considered [illegible] fail operational performance for single failure.

It is suggested that the poor low-speed characteristics are also exhibited by the present aircraft configuration and are the underlying cause of the sometimes encountered "stick lock problem".

RECOMMENDATIONS

It is recommended that the following actions be taken by the organization referred to.

SEG should undertake a comprehensive review of the Redundant Yaw Damper Program. This review should include (1) a study of the requirement specifying a limit cycle system, (2) a comprehensive simulation effort by General Dynamics incorporating aerodynamic nonlinearities including a cockpit with three degree of freedom motion and visual cues, and (3) a program to correct known deficiencies in the control systems.

SEG should proceed with the flight test program for the redundant damper system (with desired system modifications). As part of this program, the fixed gain system should be evaluated in parallel with the adaptive system (this can be done by driving the adaptive gain to the required fixed level). Results of the flight test program should be reviewed with caution, keeping in mind the test pilot is in an idealized environment with reference to the operational pilot. Flight test results should also be correlated with results of the above mentioned simulation program. Any discrepancies should be completely resolved before the system is approved for retrofit.

SEG should look into a controller incorporating multiple (two or three) fixed gains/compensators with simple switching as a back-up system in the event neither the adaptive or the fixed gain system is adequate.

AFFDL should increase research efforts in the area of analysis and synthesis of multiple input systems. This should include application of advanced control techniques to lateral-directional control. The existing heavy emphasis on longitudinal control should be removed.

In addition to seeking the universal controller, AFFDL should spend equal effort on application of advanced techniques to specific control problems and should emphasize the limitations and nonuniversality of proposed control techniques. This should include exposure of any lack of knowledge regarding a system and/or any intended application.

AFFDL should provide updated stability and control and flight control system specifications or drop the standardized approach to these specs.

Contractors in general should cease the present practice of extreme optimism concerning aerodynamic stability and control characteristics. A practically realizable aerodynamics/flight control package should be sought from proposal stage through operational use.

REFERENCES

1. Report of the B-58 Flight Control System Review Board, 5 December 1962.

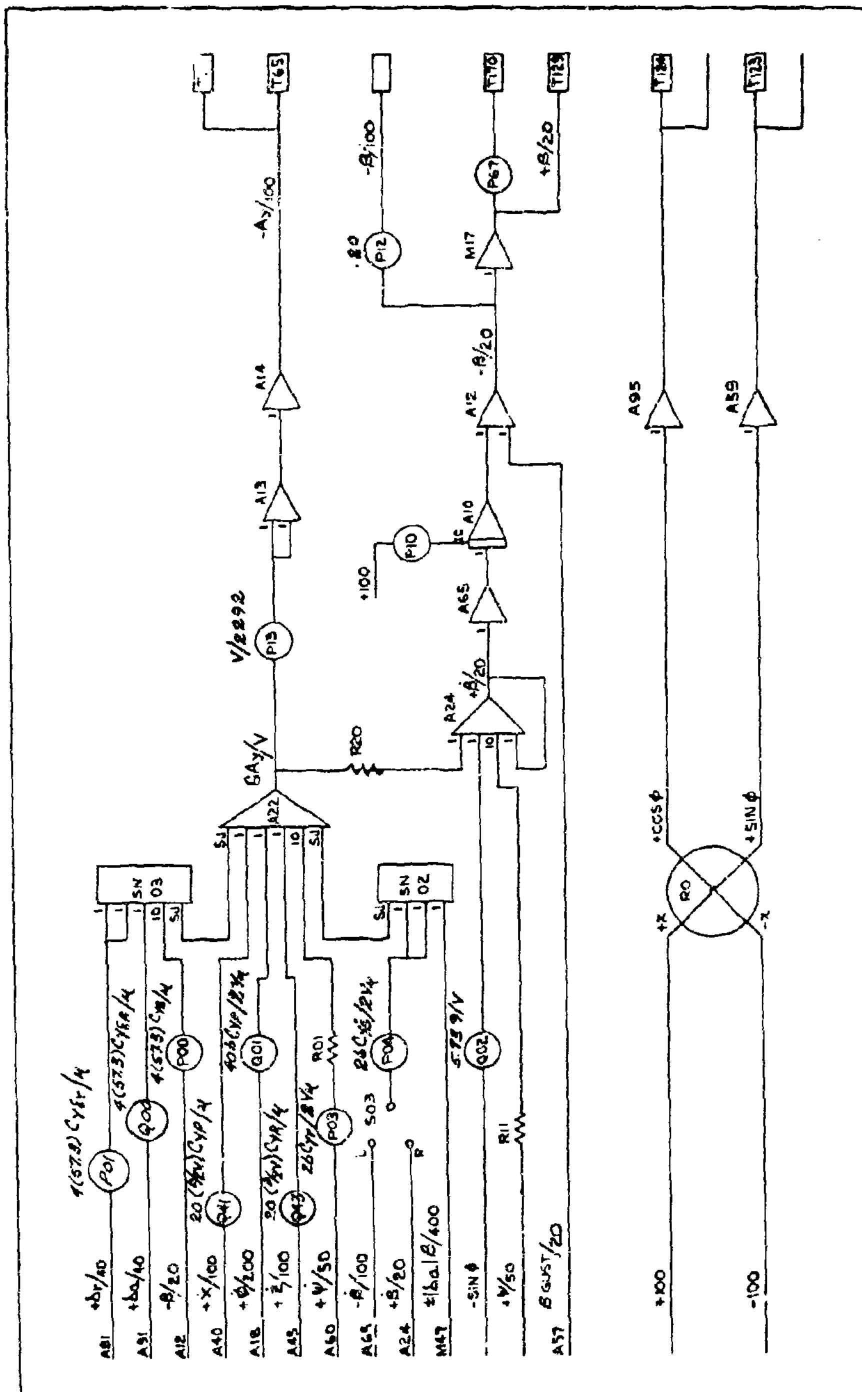
2. Data Requirements for Determination of System Performance for Vendor Proposal, 22 June 1964.

3. Bahnman, M. W., Analytical Prediction of the B-58A Spin Characteristics - Phase I, FZE-4-020, 6 February 1961.

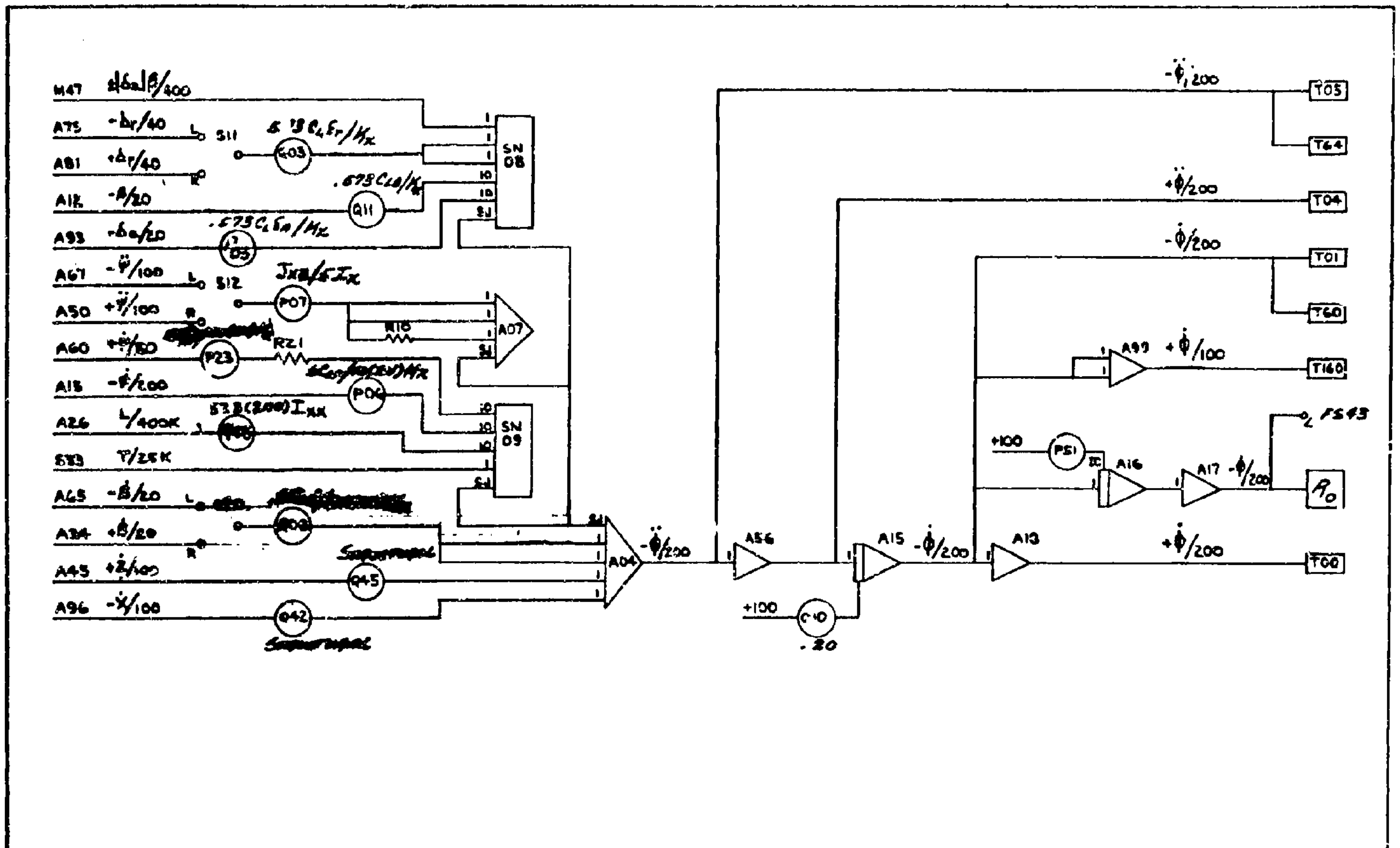
4. Triple Redundant Yaw Damper System for the B-58 Weapon System, 10 December 1964.

APPENDIX A

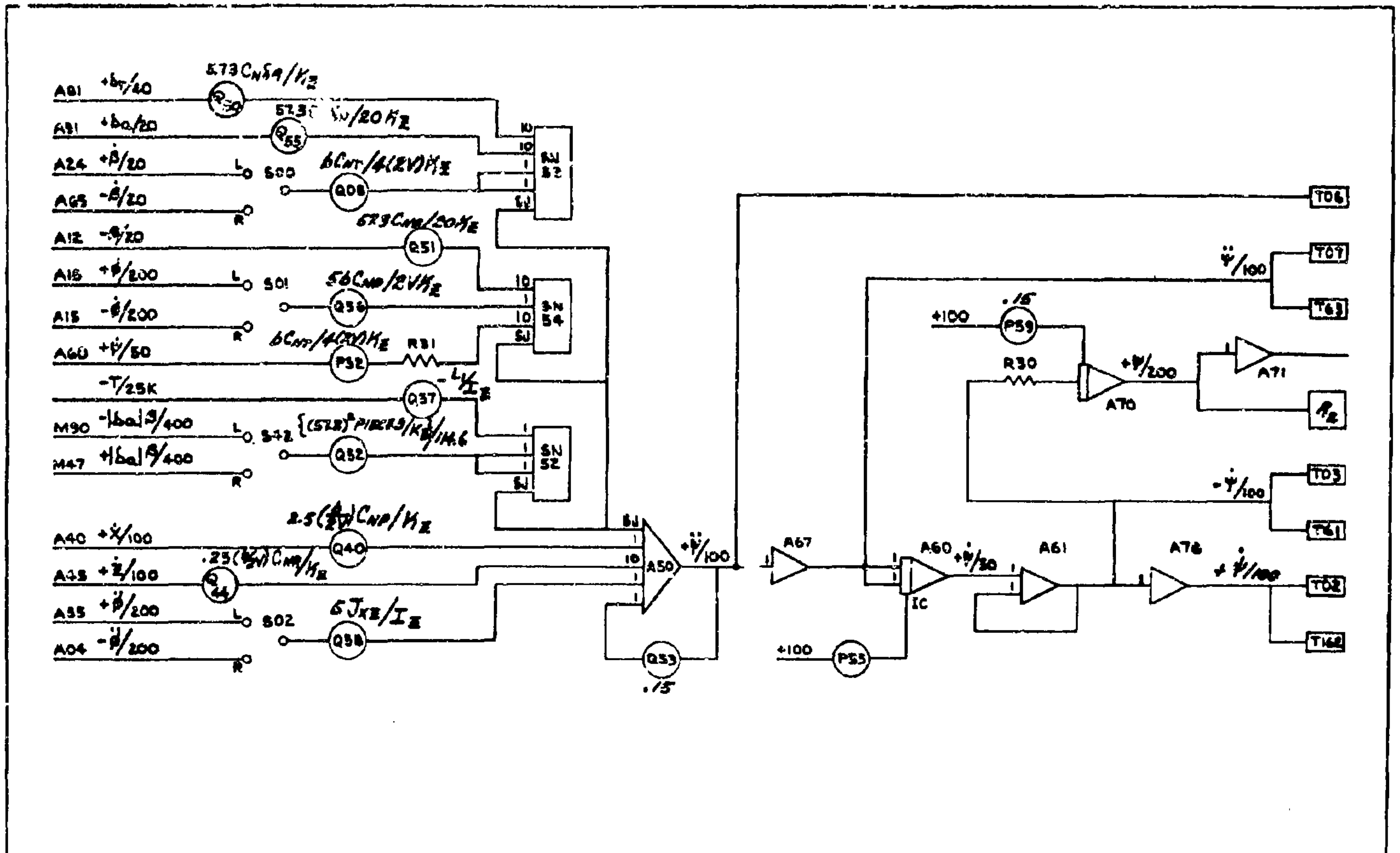
COMPUTER CIRCUIT DIAGRAMS



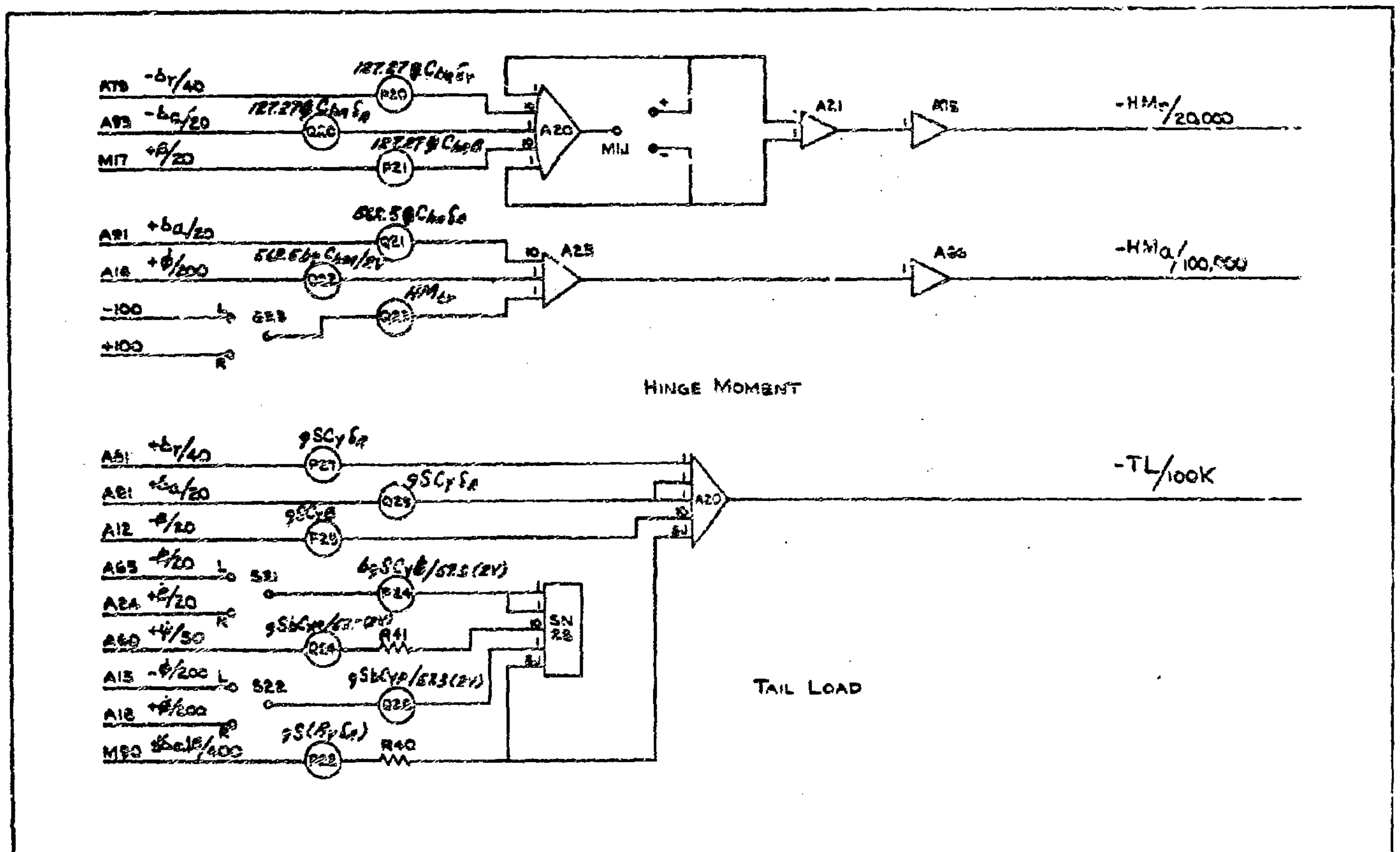
LATERAL ACCELERATION EQUATION



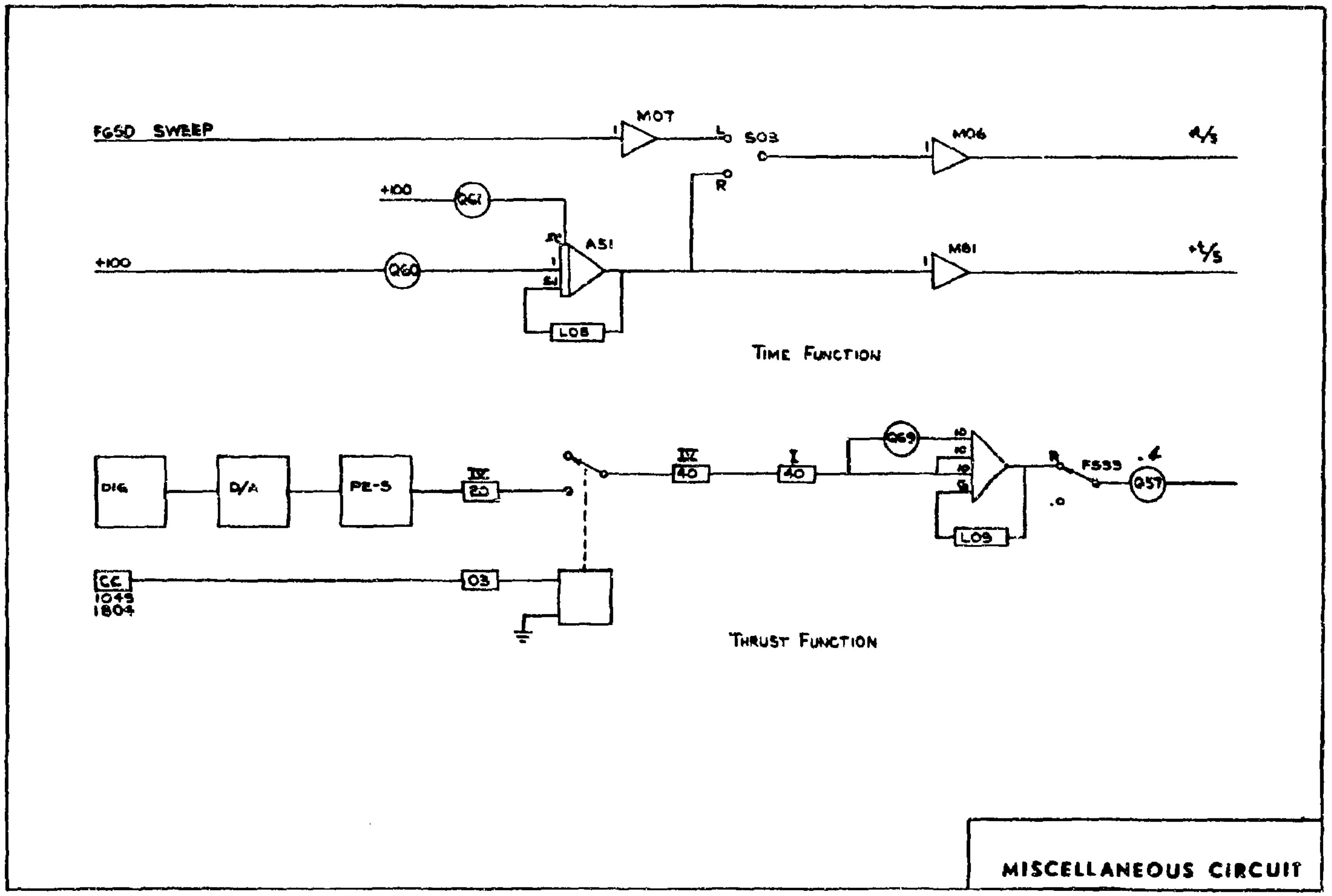
ROLLING MOMENT EQUATION

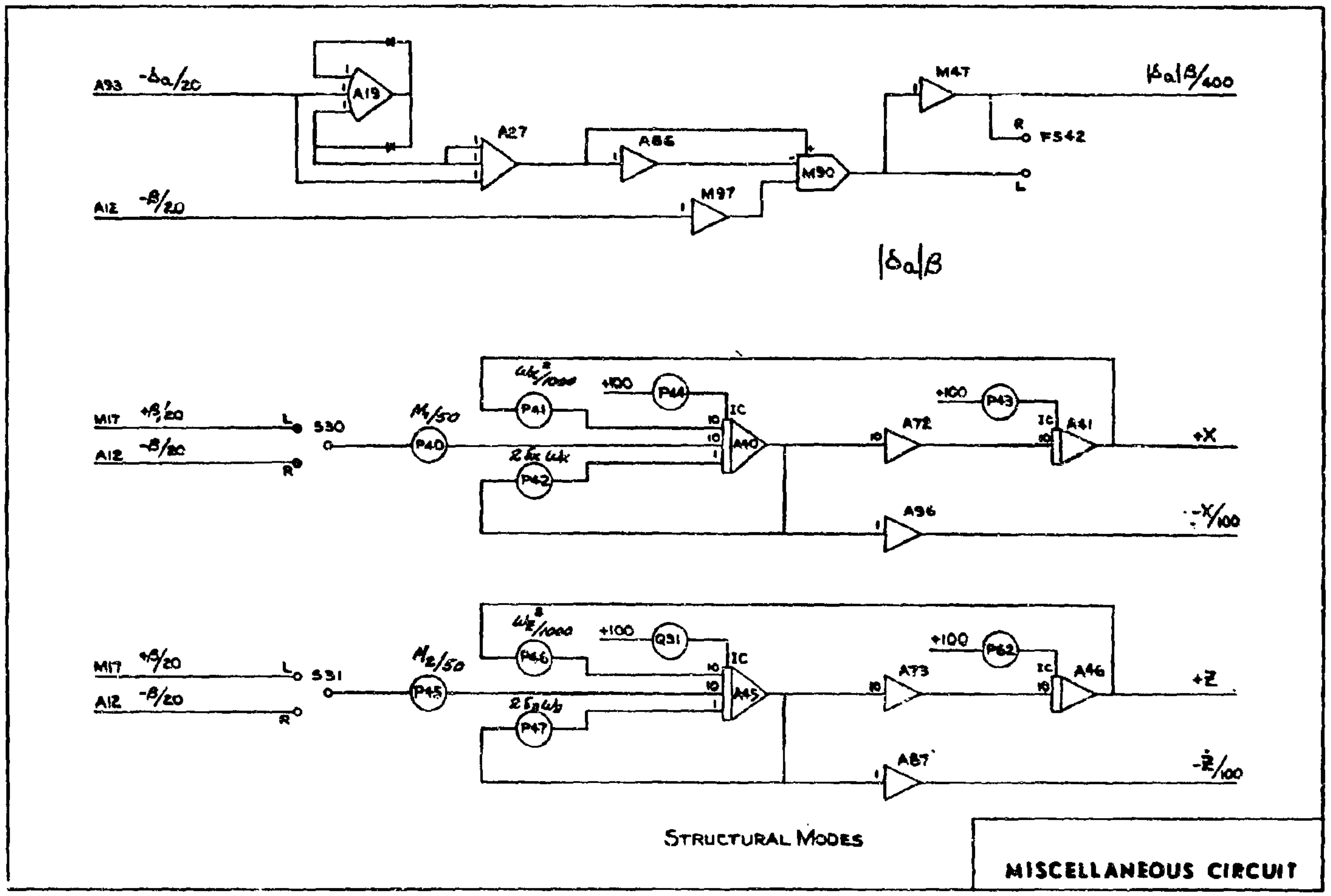


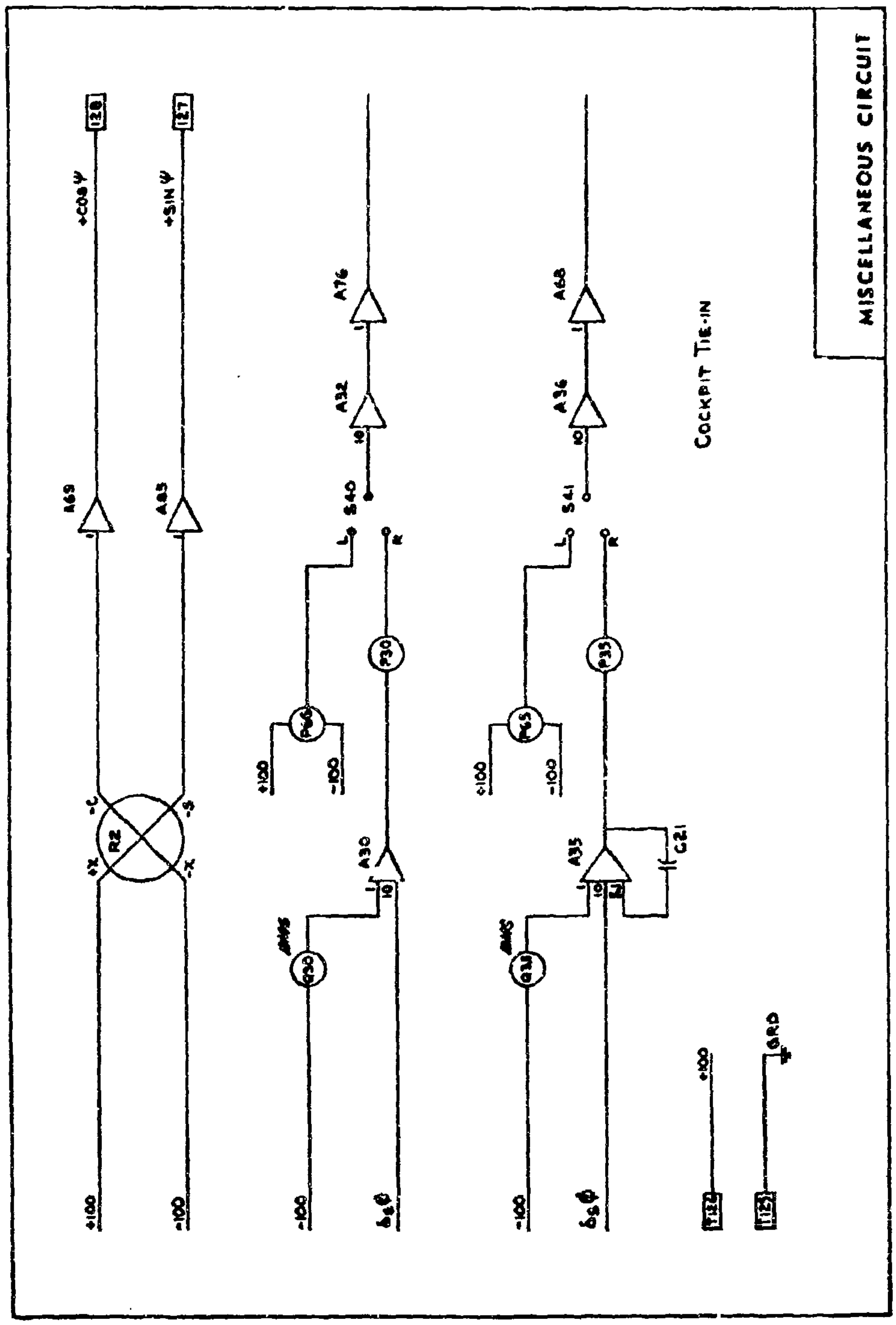
YAWING MOMENT EQUATION

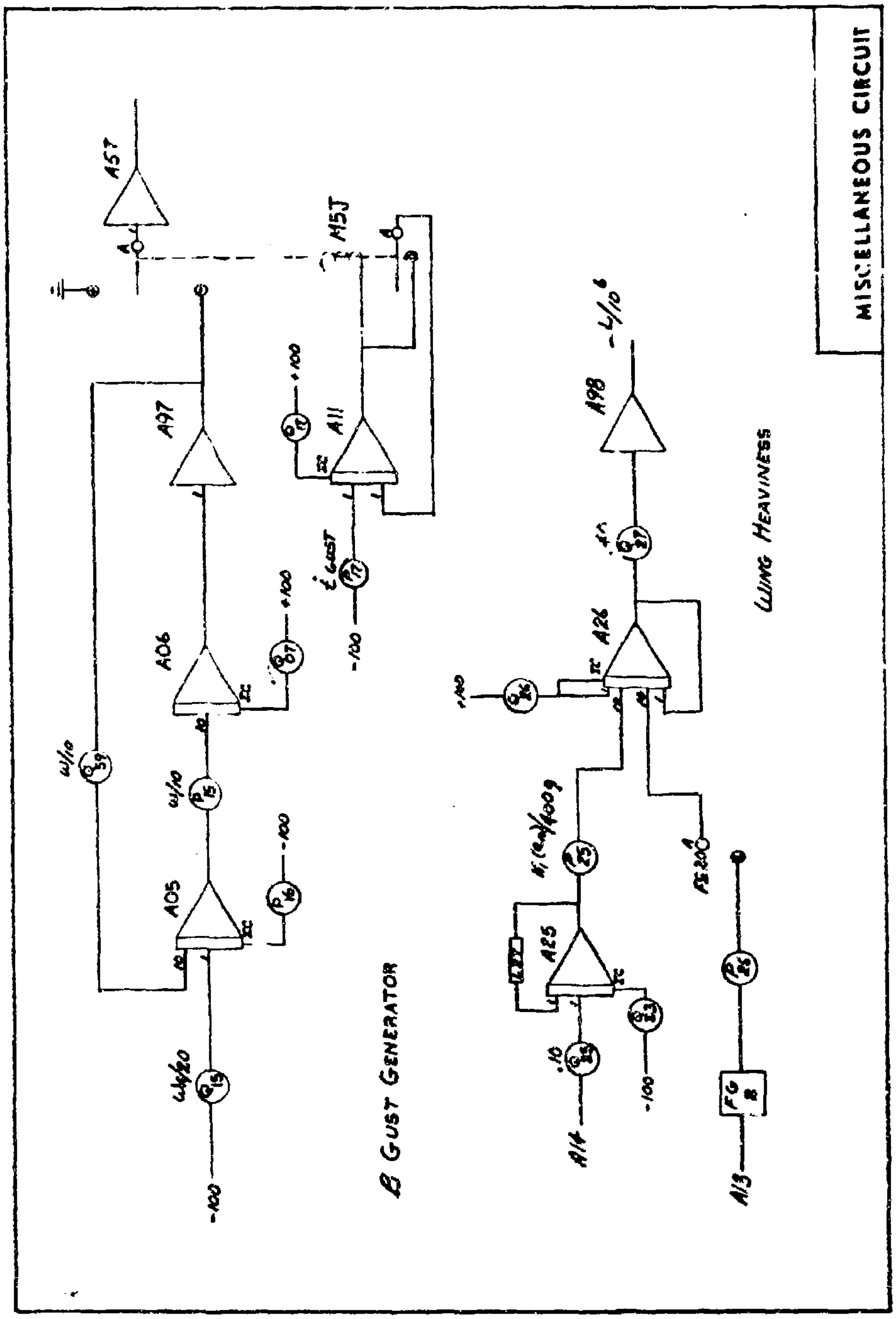


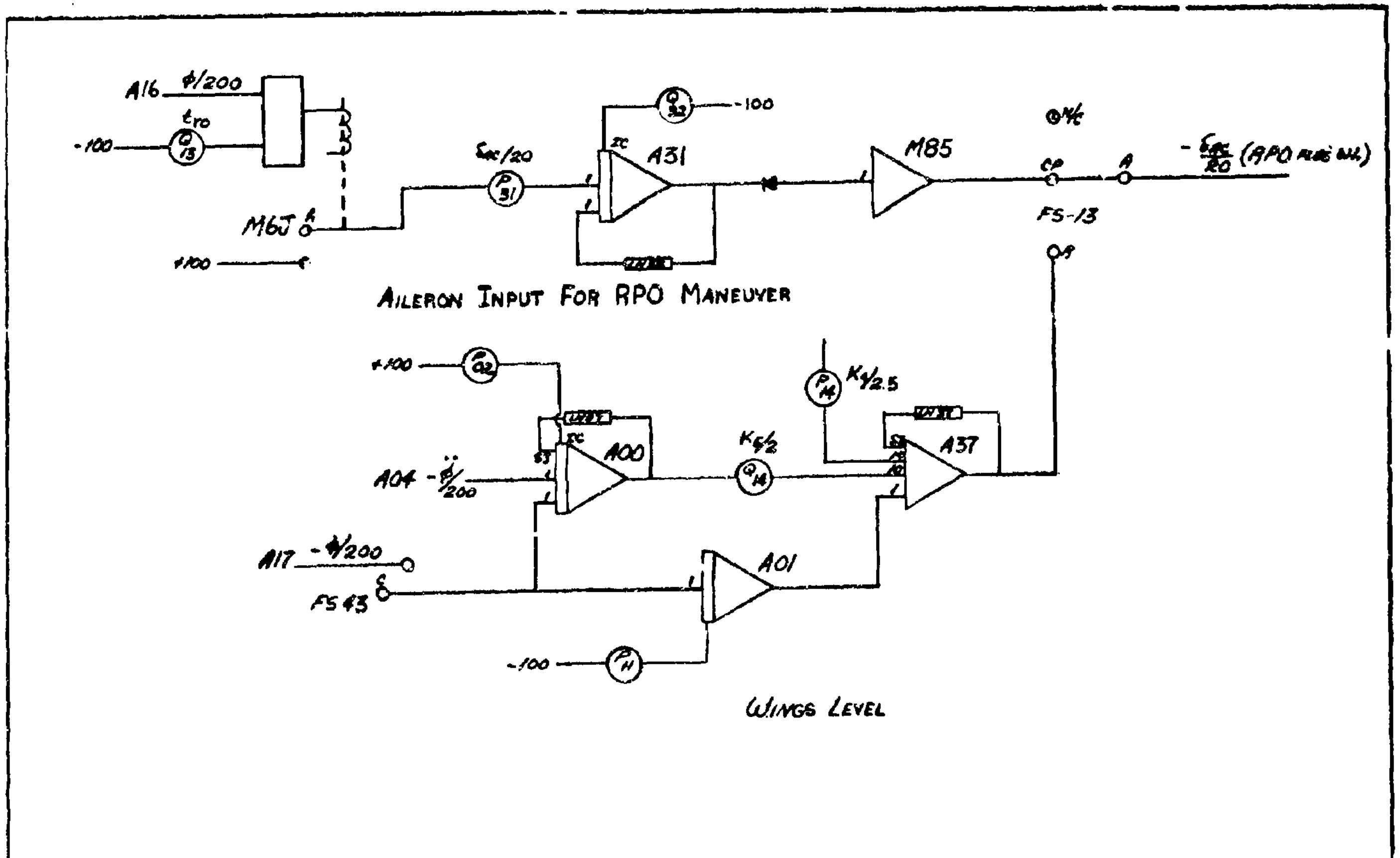
MISCELLANEOUS CIRCUIT


FG5D SWEEP
M07
L
S03
R
M06
-t/s
+100
Q61
Q60
AS1
L08
M81
+t/s
TIME FUNCTION
DIG
D/A
PE-5
IV
20
IV
40
I
40
Q69
L09
R
FS33
.4
Q57
CC
1045
1804
03
THRUST FUNCTION
MISCELLANEOUS CIRCUIT

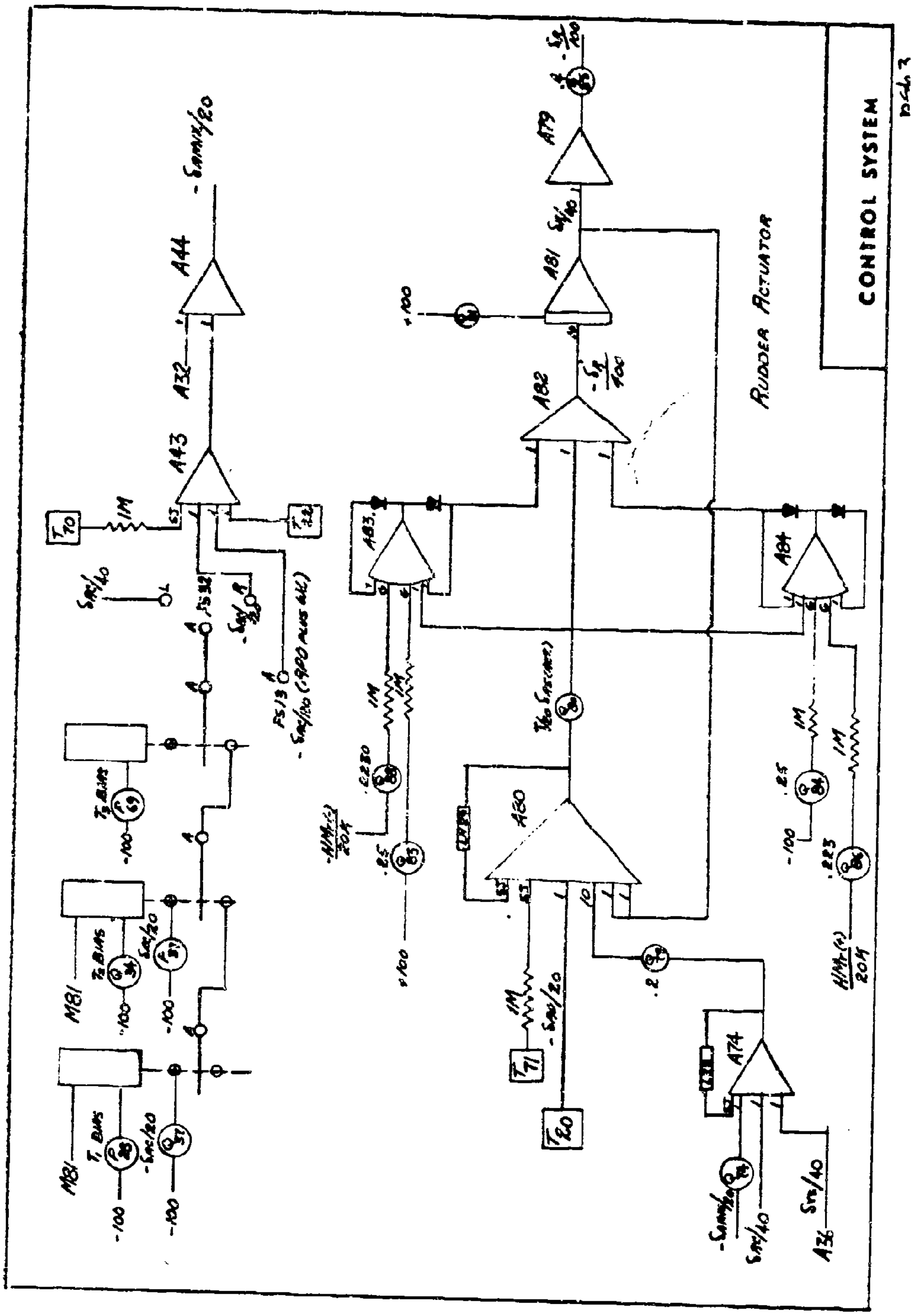

A93 -δa/20
A19
A27
A86
M90
M97
M47
|δa|β/400
R F542
L
A12 -β/20
|δa|β
M17 +β/20
A12 -β/20
S30
M1/50
P40
P41
P42
+100
P44
IC
A40
A72
P43
A41
+X
A96
-X/100
M17 +β/20
A12 -β/20
S31
M2/50
P45
P46
P47
Q31
A45
A73
P62
A46
+Z
A87
-Ż/100
STRUCTURAL MODES
MISCELLANEOUS CIRCUIT

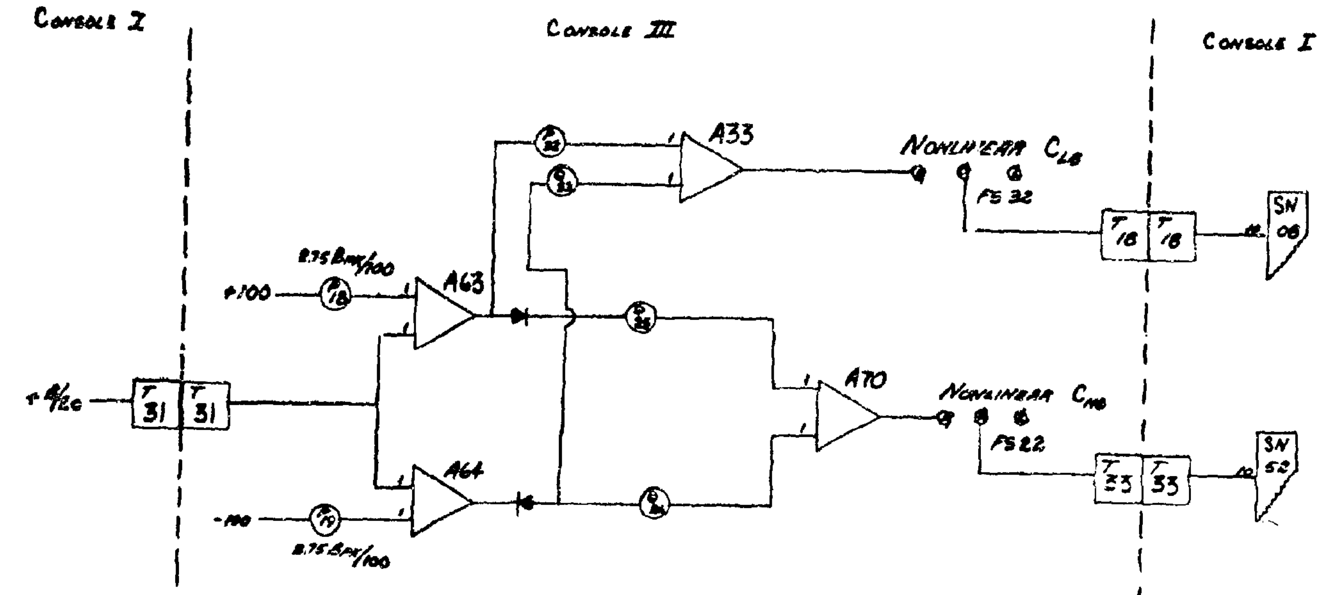

MISCELLANEOUS CIRCUIT
COCKPIT TIE-IN
+100
-100
+COS Ψ
+SIN Ψ
128
127
A69
A85
RZ
-100
-100
+100
-100
+100
-100
P30
P35
A30
A35
C21
S40
S41
A32
A36
A76
A68
GRD
1.124
1.125


MISCELLANEOUS CIRCUIT
β GUST GENERATOR
WING HEAVINESS
A05
A06
A97
A57
A11
M5J
A25
A26
A98
A14
A13
-100
+100



MISCELLANEOUS CIRCUIT

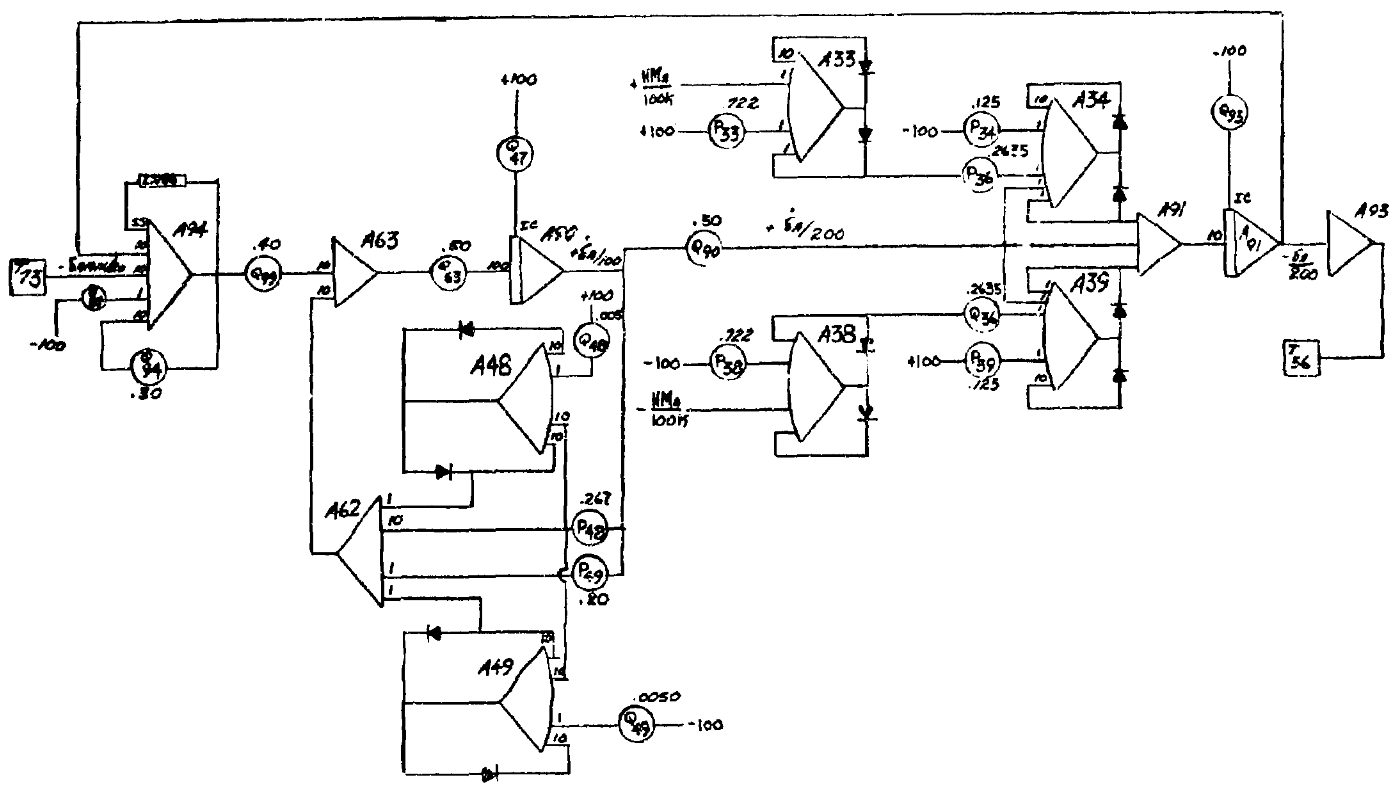

CONTROL SYSTEM
RUDDER ACTUATOR
A43
A44
A80
A81
A82
A83
A84
A74
A79
M81



B_{PX} = 3.11 (MACH NO.) + 10.74 WHEN M < 1.0

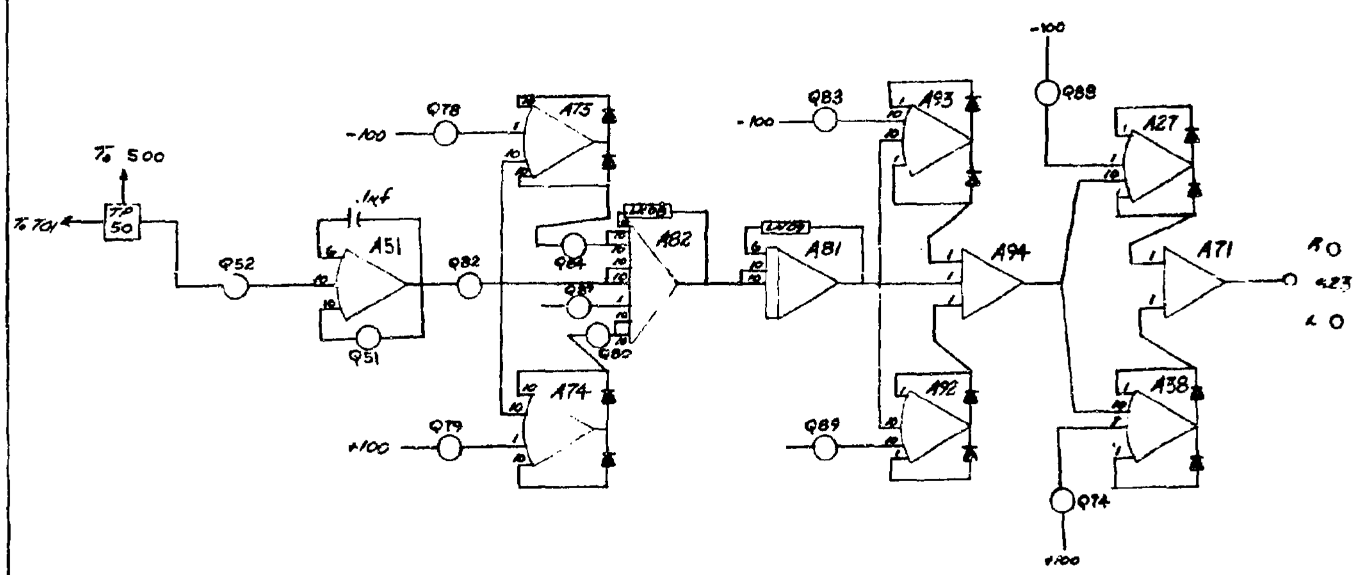
NONLINEAR C_{HB} : WITH B_{PX} INITIAL CONDITION ON P10 ADJUST Q25 SO A70 OUTPUT = OUTPUT Q51 THEN REVERSE SIGN AND ADJUST Q26

MISCELLANEOUS CIRCUIT



Elevon Actuator

CONTROL SYSTEM

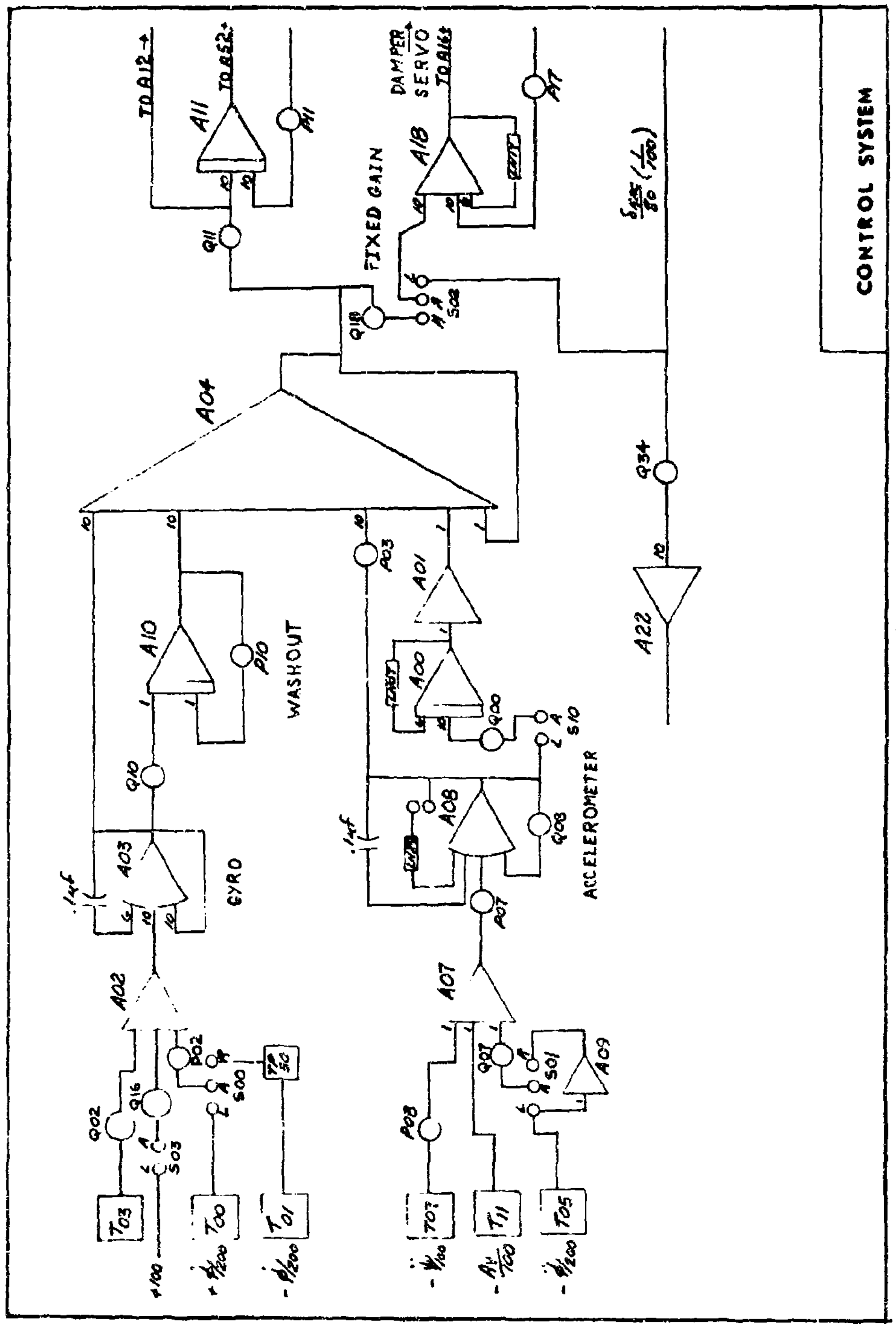


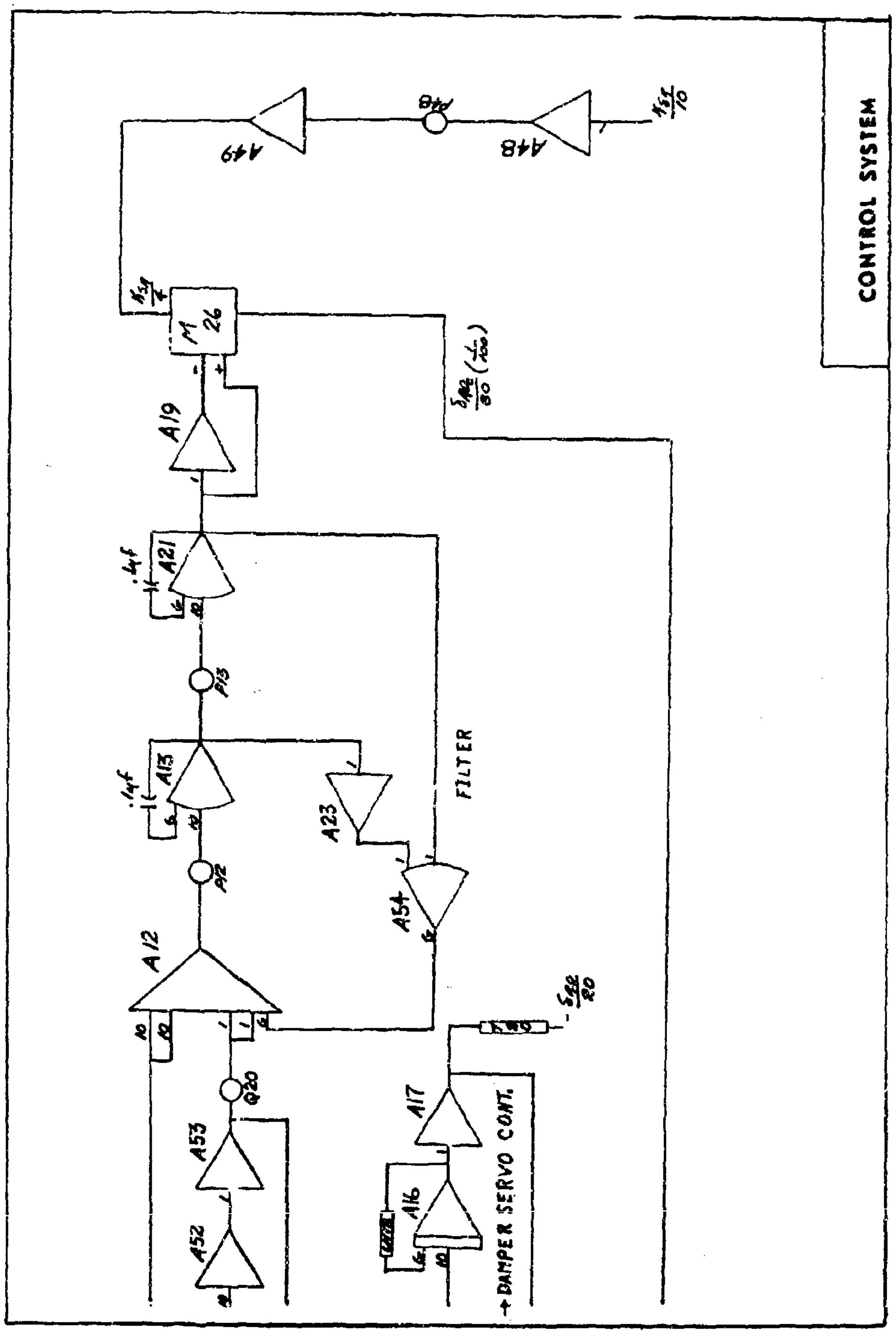
SYSTEM	S00	S01	S02	S03	S10	S20	S23	S30
SELF ADAPTIVE	*	*	R	C	L	R	R	R
FIXED GAIN	*	*	L	C	L	R	R	R

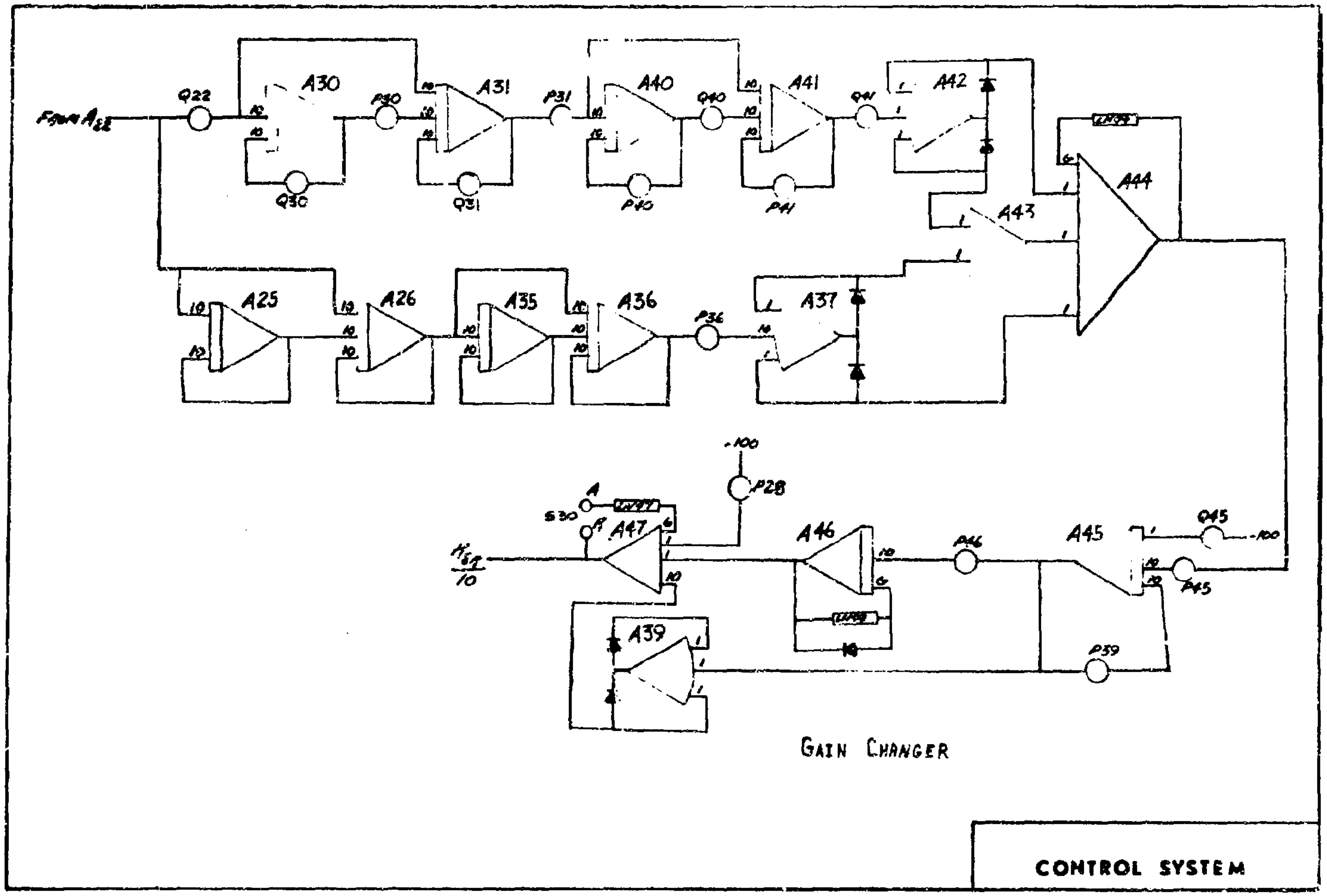
* FUNCTION OF FLIGHT CONDITION

ROLL DAMPER SERVO

CONTROL SYSTEM


CONTROL SYSTEM
GYRO
WASHOUT
ACCELEROMETER
FIXED GAIN
DAMPER SERVO
A04
A10
A11
A18
A22
A02
A03
A07
A08
A00
A01
A09


CONTROL SYSTEM
A49
A48
P48
M
26
A19
A21
A13
P15
P12
A12
A23
A54
FILTER
Q20
A53
A52
A17
A16
DAMPER SERVO CONT.


A30
A31
A40
A41
A42
A43
A44
Q22
P30
P31
Q40
Q30
Q31
A25
A26
A35
A36
A37
P36
-100
P28
A
S30
A47
A46
A45
Q45
P45
P39
A39
GAIN CHANGER
CONTROL SYSTEM

APPENDIX B

POTENTIOMETER SETTINGS

FLT NUMBER	FACTOR	SOLVING for	AMPL.	INPUT	POT SETTINGS FOR LISTED FLIGHT CONDITIONS 0002	0004	0006	0008	0010	0011	0013
P01	4(57.3) Cyδr/u	Ay/v	A-22	+δr/40	.0685	.0960	.1103	.0583	.0461	.0444	.0431
Q00	4(57.3) Cyδa/u	Ay/v	A-22	+δa/20	.0805	.1123	.1366	.0723	.1029	.0855	.0196
P00	4(57.3) Cyβ/u	Ay/v	A-22	-β/20	.0360	.0461	.1128	.0343	.0966	.0512	.0550
P04	2bCyβ̇/2Vu	Ay/v	A-22	±β̇/20	.0031	.0060	.0012	.0006	.0018	.0008	.0011
P03	2b Cyr/2Vu	Ay/v	A-22	+ψ̇/50	.0073	.0131	.0071	.0028	.0039	.0026	.0012
Q01	40b Cyp/2Vu	Ay/v	A-22	+φ̇/200	.2410	.4721	.0454	.0367	.0027	.0129	.0014
Q03	5.73 Clδr/Kx	φ̈/100	A-04	±δr/40	.0079	.0075	.2047	.0536	.0843	.0785	.0597
P05	.573 Clδa/Kx	φ̈/200	A-04	-δa/20	.0341	.0232	.0562	.0613	.0237	.0242	.0368
Q11	.573 Clβ/Kx	φ̈/200	A-04	-β/20	.0412	.0244	.2016	.0839	.1586	.1005	.0982
P08	bClβ̇/20(2V)Kx	φ̈/200	A-04	±β̇/20	.0016	.0011	.00 2	.002	.0068	.0034	.0071
P23	bClr/20(2V)Kx	φ̈/200	A-04	+ψ̇/50	.0067	.0045	.0330	.0096	.0198	.0127	.0115
P06	bClp/10(2V)Kx	φ̈/200	A-04	-φ̇/200	.0036	.030	.1013	.0316	.0965	.0460	.0944
P14	Ky/2.5wings level	δa/20WL	A-37	+ψ/100	←			.4000			→
P07	Jxz/5Ix	φ̈/200	A-04	±ψ̈/100	.0939	.0864	.0246	.0218	.0245	.0002	.0150
Q50	5.73Cnδr/Kz	ψ̈/100	A-50	+δr/40	.0629	.0477	.2805	.1300	.1372	.1198	.0903
Q55	57.3 Cnδa/20Kz	ψ̈/100	A-50	+δa/20	.0166	.0123	.1148	.0504	.1608	.0984	.06.4
Q51	57.3 Cnβ/20Kz	ψ̈/100	A-50	-β/20	.0548	.0419	.2521	.0132	.2515	.1908	.0880
Q08	bCnβ̇/4(2V)Kz	ψ̈/100	A-50	±β̇/20	.0010	.0019	.0289	.0049	.0247	.0061	.0129
P32	bCnr/4(2V)Kz	ψ̈/100	A-50	+ψ̇/50	.0489	.0510	.1155	.0382	.0951	.0493	.0512
Q56	5bCnp/2VKz	ψ̈/100	A-50	±φ̇/200	.5441	.5426	.145	.0985	.1304	.0080	.0599
Q58	5Jxz/Iz	ψ̈/100	A-50	±φ̈/200	.7044	.7338	.1656	.1640	.1849	.0918	.1122
P13	V/2292	Ay/100	A-13	Ay/v	.160	.1164	.4433	.3844	.6029	.5068	.8447
Q02	5.73 g1V	β/20	A-24	-sinφ	.5012	.6920	.1818	.2088	.1333	.1583	.0950
P27	qSCy δr v.t.	TL/100K	A-29	+δr/40	.2050	.1110	.9101	.3353	.4146	.3366	.2713
Q29	qSCyδa v.t.	TL/100K	A-29	+δa/20	.0331	.0171	.0849	.0398	.1482	.1114	.0480
P29	qSCyβ v.t.	TL/100K	A-29	-β/20	.0293	.0152	.1953	.0451	.1656	.0647	.1035
P24	bqSCyβ̇ v.t./57.3(2V)	TL/100K	A-29	±β̇/20	.0051	.0036	.0063	.0020	.0089	.0032	.0077
Q24	qSbCyr v.t./57.3(2V)	TL/100K	A-29	+ψ̇/50	.0151	.0108	.0374	.0104	.0250	.0136	.0120
Q28	qSbCyp v.t./57.3(2V)	TL/100K	A-29	+φ̇/200	.1544	.1281	.0069	.0188	.0386	.002.	.0145

OUR POT NUMBER	QUANITY FACTOR	Solving for	AMPL.	INPUT	POT SETINGS FOR LISTED FLIGHT CONDITIONS 0002	0004	0006	0008	0010	0011	0013
P20	127.27qC[illegible]	[illegible]/20K	A-20	-[illegible]/40	.0321	[illegible]	[illegible]	.0642	.3084	.1744	.1782
Q20	127.27qC[illegible]	[illegible]/20K	A-20	-[illegible]/20	.0447	[illegible]	[illegible]	.0091	.4898	.1313	.3945
P21	127.27q[illegible]	[illegible]/20K	A-20	[illegible]	[illegible]	[illegible]	.0128	.0689	.0838	.0178	.0385
Q21	562.5q[illegible]	[illegible]/100K	A-23	[illegible]	[illegible]	[illegible]	.0342	.0328	.1741	.1106	.1295
Q22	562.5[illegible]/27	[illegible]	A-23	[illegible]	.0138	.0113	.0497	.0135	.0363	.0264	.0240
Q23	[illegible]	[illegible]/1000	A-23	[illegible]	.0013	[illegible]	.0347	.0035	.0213	.0296	.0314
Q57	[illegible]	[illegible]/100	A-5[illegible]	[illegible]/250	.5507	.7845	.5369	.6402	.6443	.6422	.6424
Q04											
Q06	(57.3)[illegible]										
Q09	((57.3)^2 P[illegible]	[illegible]/200	[illegible]	[illegible]							
Q05	57.3(200)[illegible]	[illegible]/200	[illegible]	[illegible]	[illegible]	[illegible]	.0286	.0313	.0315	.0317	.0316
Q5[illegible]	(57.3)^2 P[illegible](57.3)	[illegible]	A-30	[illegible]/400	.0052	.0038	.0122	.0094	.0502	.1006	.0604
Q42	Structural	[illegible]/200	A-04	[illegible]/100							
Q45	Structural	-[illegible]/200	A-[illegible]	±[illegible]/100							
P22	[illegible] V.T.	[illegible]/100K	A-29	±[illegible]/400	.0435	.0242	.1730	.0588	.1403	.0877	.1568
Q40	2.5([illegible]) [illegible]	[illegible]/100	A-50	[illegible]/100							
Q44	[illegible]	[illegible]/100	A-50	[illegible]/100							
P34	[illegible]	[illegible]/100	[illegible]	[illegible]				.1260			
P39	[illegible]	[illegible]/100	[illegible]	[illegible]				.1250			
Q41	20([illegible])	Ax/V	A-27	[illegible]							
Q43	20([illegible])	AV/V	A-22	[illegible]							
P38	Hinge Moment Limit	[illegible]	A-39	[illegible]				.7220			
P33	Hinge Moment Limit	[illegible]	A-33	[illegible]				.7820			
P41	(Wx)^2/1000 [illegible]	[illegible]	A-40	-[illegible]				.3090			
P40	Wx/10 [illegible]	[illegible]	A-40	±[illegible]/20							
P42	2[illegible] [illegible]	[illegible]	A-40	[illegible]				.3500			
C25	Scale [illegible]							.1000			
P46	(Wz)^2/100 [illegible]	[illegible]	A-45	[illegible]				.9880			
P45	Wz/50 [illegible]	[illegible]	A-45	±[illegible]/20							

POUR (CT) NUMBER	QUANTITY FACTOR	Solving for	AMPL.	INPUT	0002	0004	0006	0008	0010	0011	0013
P47	2ż Wz Str	+Ż	A-45.	+Ż	←			.5283			→
P26	KZo/100K(g)Wing Heavyness	+L/400K	A-26	+Ay/100							
P31	δa comm(RPO)/20 Input	+δac/20	A-31	M37							
Q84	HmR Limit	δR/400	A-84	+100V	←			.2500			→
Q88	Hmr(-)/20K	-δr/400	A-83	M1	←			.2230			→
Q59	(1-cos)Gust Input W/10	θGust/20	A-05	A-99	←			.6283			→
P15	(1-cos)Gust Input W/10·	θGust/20	A-03	A-05	←			.6283			→
Q14	Wings Level K5/2	δac/20K	A-37	A-00							
Q53	Scale Factor	ψ/100	A-50	ψ/100	←			.1500			→
Q83	HmR Limit	δR/100	A-83	-100V	←			.2500			→
Q86	-Hmr/20K Limit	-δa/100	A-84	M1+	←			.2230			→
P28	T1 Bias		M7J	-100	←			.1050			→
Q34	T2 Bias		M3J	-100	←			.1050			→
P69	T3 Bias		M4J	-100	←			.1110			→
Q80	T/2D δRS(Actuator)	-δR/400	A-82	δR/220	←			.5500			→
P48	δa Elevon Actuator	A-63	A-62	A-90	←			.2670			→
P49	δa Servo	φa/200	A-52	δa/200	←			.2000			→
Q94	Scale $\frac{a-1}{?}$	φal	A-94	δamix/20	←			.3000			→
Q39	Scale	φal	A-63	φa	←			.4000			→
P36	HMa Rate Limit Slope	δr/200	A-33	HMa/100K	←			.2635			→
Q36	HMa Rate Limit Slope	δr/200	A-38	HMa/100K	←			.2635			→
Q90	Scale	φal	A-92	φa/200	←			.5000			→
P12	Scale	-β/20		-β/20	←			.2000			→
Q63	T/50 δa Servo	φal	A-90	φal	←			.5000			→
Q85	Scale	δR/100		δR/40	←			.4000			→
Q26	Lo/400KWing Heavyness	L/400K	A-26	-100V							
P25	K1(an)/400(g)		A-25	δAy/100							
Q4?	δa Servo V.B.	φal	A-49	-100V	←			.0050			→

OUR POT NUMBER	CLARITY FACTOR	Solving for	AMPL.	INPUT	POT SETTINGS FOR LISTED FLIGHT CONDITIONS 0002	0004	0006	0008	0010	0011	0013
Q48	δa Servo Y.B.	φal	A-48	+100V				.0050			
Q27	Scale Wing Heavyness	L/1(10)	A-28	L/400K				.4000			
Q13	tro in RPO	Relay	M6J	-100							
P17	i Gust	M5K	A-11	-100V				.1000			
Q37	δa Command/20	δamix/20	M7J	-100V							
Q15	β Gust WA/20	βGust	A-05	+100V							
P37	δa Command/20	δamix/20	M3J	-100V							
Q74	$\frac{1}{2}$ δR/δa Mech	δRtt/40	A-74	δamix/20				.5000			
Q79	δr Gain	δR/100	A-74	δRtt/40				.2000			
P33	Aileron K.M. Rate Limit	δa/200	A-33	hma/100K							
Q36	Slopes	δa/200	A-38	hma/100K							
Q32	RPO	I.C.	A-31	+100							
Q17	β Gust	I.C.	A-11	+100							
P10	β	I.C.	A-10	+100							
Q10	φ	I.C.	A-10	+100							
P55	ψ	I.C.	A-60	+100							
P51	φ	I.C.	A-15	+100V							
Q07	βGust	I.C.	A-06	-100							
Q31	S.M.	I.C.	A-45	+100		[illegible]					
Q33	Wing Heavyness	I.C.	A-25	-100							
Q35	Cockpit	I.C.	A-35	-100							
Q47	Elevon Actuator	I.C.	A-50	-100							
Q93	Elevon Actuator	I.C.	A-91	-100							

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	2b GROUP
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13. ABSTRACT

Computer analysis of yaw damper systems proposed for modification of the B-58 Aircraft. Results generally verified conclusions and recommendations of contractor studies with the exception of low speed characteristics.

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