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DIRECT LIFT CONTROL

for the LAMS B-52

THESIS

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GGC/EE/68-8

DIRECT LIFT CONTROL

for

LAMS B-52

THESIS

Presented to the Faculty of the School of Engineering of

the Air Force Institute of Technology

Air University

in Partial Fulfillment of the

Requirements for the Degree of

Master of Science

by

Gary L. Nelsen
Captain USAF

and

Robert C. Lorenzetti
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Graduate Guidance and Control

June 1968

Preface

Most theses begin with a short note of thanks to those who aided the author or were helpful above and beyond the call of duty. Ours will begin with a long chapter.

Our thanks first of all to Mr. Robert Johannes and Captain James Lee, our sponsors from the Air Force Flight Dynamics Laboratory, who suggested the project and assisted whenever possible. Mr. Ron Anderson, also of AFFDL, provided some much-needed equipment. Our AFIT faculty advisor, Major Roger Johnson, found time in his crowded schedule to serve as a test pilot, and helped arrange one of the trips to Wichita.

The thesis would not have been possible without the full cooperation of Mr. Paul Burris and Mr. Walt Rohling of The Boeing Company, Wichita Division. They provided equations of motion, aerodynamic coefficients, and sample step elevator responses for the LAMS B-52. They also hosted our trips to Wichita. On the first, they showed us the LAMS aircraft, let us fly their B-52 simulator, and answered innumerable questions. Later, we tested our Direct Lift Controller on their refueling simulator. They also worked hard to bring about a flight test.

Our thanks also to Mr. Vic Falkner of Honeywell Inc. who helped point us in the right direction in the earliest design phase.

Although it is not included in this report, our first LAMS B-52 simulation was done at the Air Force Analog and Hybrid Computation Center in Building 57, Wright-Patterson AFB, Ohio. This simulation was programmed by Mr. Howard Jones and operated by Mr. Keith Jones. It gave us invaluable experience in a major analog simulation, and helped clarify our design ideas. This program, which includes a Honeywell pitch SAS in lieu of our controller, is available through Mr. Howard Jones for further use if required.

From time to time in the text, we mention the limited availability of the piloted six degree of freedom simulation at the AF Flight Dynamics Laboratory. This was due strictly to checkout problems of the new hybrid computer and prior contract commitments. In fact, it was only the complete cooperation of Ray Haas and his staff and their willingness to work overtime that allowed us as much time as we had. The cockpit simulator was ably operated and maintained by Fritz Baker, Harold Hooker, and John Priest of Lear-Siegler, Inc. The analog computer performed faultlessly, thanks to careful and thorough daily checkout and maintenance by Dick Morien and Chuck Donalies of Electronics Associates, Inc. Ray Haas offered practical suggestions on analog computation, and served as a test pilot and general factotum.

The other test pilots provided the variety of experience and opinions necessary to a project of this nature. Our fellow AFIT students, Captains Don Green, Larry Smulczenski, Eric Gruenler, and Harold Brattland took time

off from their own theses and classes, often at inconvenient hours, to help us. Our special thanks to Major Bill Smith of the Royal Canadian Air Force who volunteered his services as a test pilot.

Mr. Paul Blatt, also of the Flight Dynamics Laboratory, offered encouragement and provided several hard-to-get references. He also served as our "advertising agency" on a tour of the major aerospace contractors.

We cannot forget the AFIT faculty: Major Hannen and Captain Kern, our readers, Professor Larsen, head of the Department of Aeronautical Engineering, who created an interest in airplanes through his excellent instruction, and Lt. Burghart, our instructor for three courses in optimization theory, who listened patiently to many sad stories during the early part of the work and aided in choosing the optimization theory to be used.

Last, but not least, we must thank our wives who patiently endured four months of Greek letters, late suppers, and stack after untidy stack of books, paper, computer programs, and strip chart recordings. They also performed feats of magic in translating squiggly lines into neatly typed manuscript.

Robert C. Lorenzetti
Captain USAF

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March 1968

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List of Symbols

$\underline{A}$ - A Vector

A^T - A Matrix Transpose

A^{-1} - A Matrix Inverse

AFFDL - Air Force Flight Dynamics Laboratory

AFIT-SE - Air Force Institute of Technology, School of Engineering

b - Wing Span

$\bar{c}$ - Mean Aerodynamic Wing Chord

DLC - Direct Lift Control

FB - Feedback

g - Gravitational Force

H - Altitude Change

$\dot{H}$ - Rate of Climb

HG - High Gain Amplifier

I - Integrator

IC - Initial Condition

KEAS - Knots Equivalent Airspeed

L - Rolling Moment

LAMS - Load Alleviation and Mode Stabilization

M - Pitching Moment

m - Mass

MAV - Mean Absolute Value

MUL - Multiple Connector

N - Yawing Moment

$N_1(t)$ - Noise (Gust simulation)

N_L - Normal Acceleration of Aircraft Center of Gravity (Positive Up)

P - Roll Angular Velocity

PIO - Pilot Induced Oscillations

Q - Pitch Angular Velocity

q - Dynamic Pressure = $\frac{1}{2} \rho U_o^2$

R - Yaw Angular Velocity

RG - Reduced Gain Amplifier (0.1 and 1.0 inputs)

S - Wing Area, ft^2

S - Summer (on amplifier sheets)

SAS - Stability Augmentation System

SDF - Six Degree of Freedom

TDF - Two Degree of Freedom

TE - Trailing Edge

t_F - Final Time

TR - Test Reference - Only Applied During Static Test

U - Airspeed (X Direction)

U_o - Equilibrium Airspeed

V - Side Velocity (Y Direction)

W - Vertical Velocity (Z Direction)

X,Y,Z - Aerodynamic Force Components

α - Angle of Attack

α_G - α Gust (Vertical)

β - Sideslip Angle

β_G - β Gust (Horizontal)

δ_{ail} - Aileron Deflection

δ_e - Elevator Deflection

δ_{sp} - Spoiler Deflection

θ - Pitch Angle

$\dot{\theta}$ - Pitch Rate

ρ - Air Density

τ - Time Constant

ϕ - Roll Angle

γ - Yaw Angle

ω_o - Gust Frequency

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Abstract

A DLC (Direct Lift Control) system was designed for the LAMS B-52, a test aircraft with a fly-by-wire control system. Spoilers, symmetric ailerons, and elevator respond to the pilot's normal longitudinal column movements, applying lift forces in the desired direction without pitching the aircraft. DLC eliminates the long heave crossover time and normal acceleration reversal associated with large aircraft. The controller permits changes in rates of climb and descent of up to 500 feet per minute from trim condition without a change in aircraft pitch attitude. When larger column movements command greater rates of climb, DLC functions as a conventional control system, causing aircraft rotation.

The pilot's ability to make precise changes in altitude or rate of climb is enhanced in tasks such as approach and landing, in-flight refueling, and terrain following. DLC does not excite the phugoid mode, and visual cues are not disturbed by continually changing pitch attitude.

The closed-loop controller consists of fixed-gain normal acceleration, pitch angle, and pitch rate feedbacks to the three control surface servos. Feedback gains were selected by a digital optimization program based on Kalman's Linear State Regulator Theory.

In addition to virtually eliminating pitch attitude changes, DLC reduces normal acceleration an average of 50%

during climb and descent maneuvers, both in and out of turbulence, and improves the pilot's ability to maintain constant airspeed.

DLC was tested with a piloted moving-base cockpit simulator driven by a six degree of freedom analog program.

Chapter I

Introduction

Background

During the past decade, the production of larger, heavier, higher performance aircraft has required a rapid advance in flight control system technology to maintain acceptable handling qualities. Sophisticated stability augmentation systems are standard in the century series fighters and many other current aircraft. The F-111 has an even more advanced self-adaptive control system. However, precise control of these aircraft is still difficult in certain flight regimes due to excessive pitch rate overshoots while controlling normal (vertical) acceleration. These regimes include approach and landing, in-flight refueling, terrain following, and cargo drop. The pitch rate required for rapid maneuvering can obscure important visual cues during these flight tasks and attempts to make small corrections in flight path angle can result in undesireable transients. These problems exist because both lift and normal acceleration are controlled by pitching the aircraft. Thus, it is desireable to provide the pilot with more direct control of both lift and pitching moment than is available with conventional elevator surfaces.

In transport aircraft, increased size, weight, and airspeed have reduced the ratio of aerodynamic to inertial moments, producing a "sluggishness" of response which reduces the capability of precise control on the approach glide path and in the landing flare.

> To attain satisfactory levels of landing performance with these aircraft, it has been necessary to place great emphasis on optimized operational procedures, and on the presentation of approach guidance information. The next generation of transports will include designs having twice the gross weight of the largest now in operation, and their dynamic characteristics in pitch may be further complicated by several additional factors. Particularly in supersonic designs, proportionately shorter tail lengths together with large pitch inertias will amplify the lift variations accompanying pitch control inputs. With more conventional configurations efficient use of high life devices . . . will result in a reduced sensitivity of lift to angle-of-attack (Ref 5:1).

In-flight refueling, originally restricted to bombers, will also become a cargo aircraft capability with the advent of the C-5A. This task is most demanding for pilots when a heavy airplane is being refueled at high altitude. The response lag following an elevator input requires utmost pilot attention and skill. Figure 1-1, provided by The Boeing Company, illustrates the response due to elevator input during B-52 refueling.

> This condition is present to some degree in all refueling operations, but is aggravated by the long heave crossover time of aircraft with large pitch inertia. With normal visual position cues, the pilot will tend to command elevator deflections about 90° out of phase with the desired optimum.

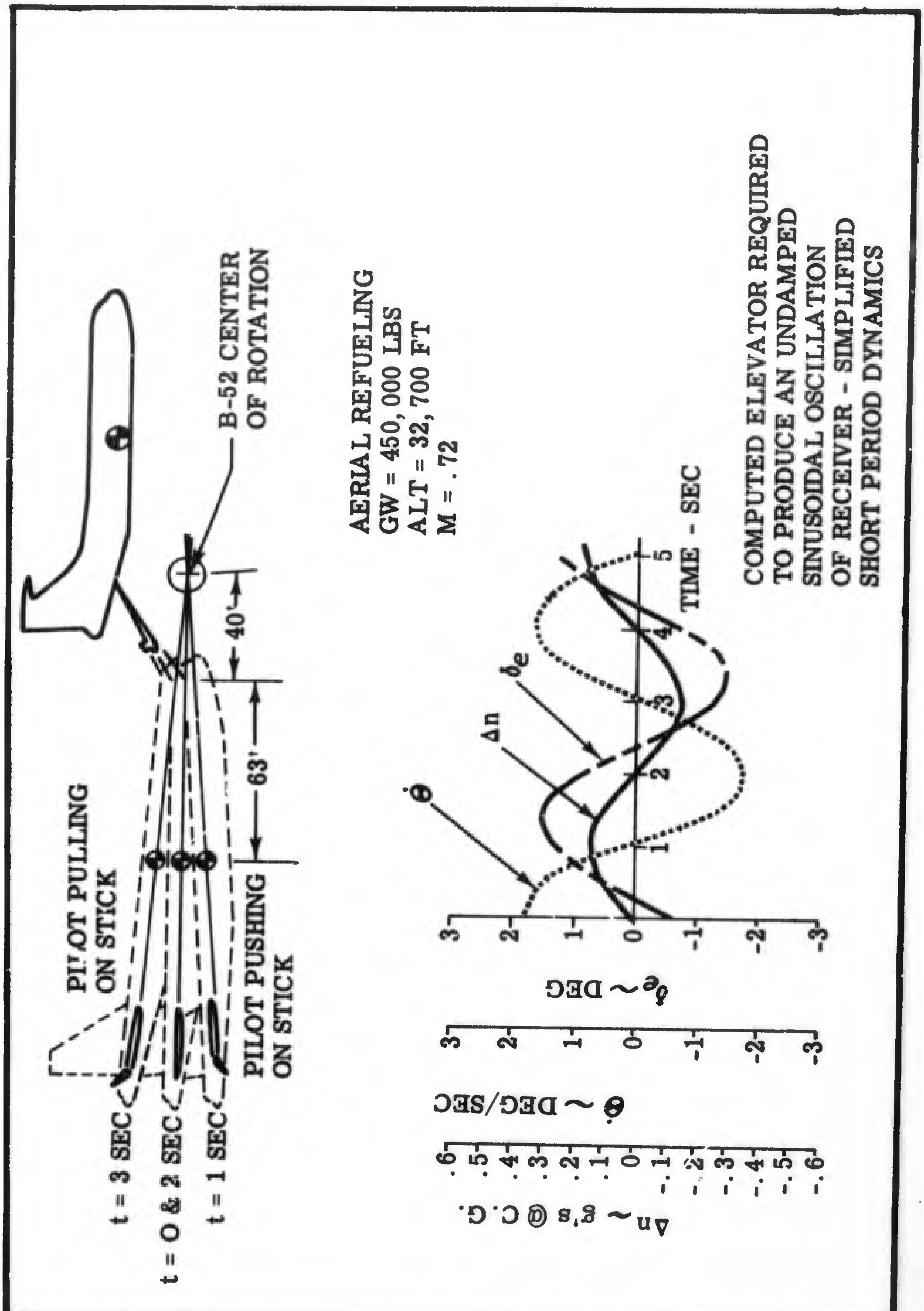


FIG. 1-1. B-52 LONGITUDINAL DYNAMICS DURING REFUELING

> Similar need for improved longitudinal handling qualities exists for other tracking tasks, such as weapon delivery, formation flying, reconnaissance, terrain following, and instrument and GCA approaches. The longer the heave crossover time in relation to the characteristic longitudinal short period frequency, the more compensatory lead the pilot must supply to perform the task satisfactorily. Since there is a practical upper bound on the lead that a pilot can generate effectively, considerable improvement in piloting performance is possible if the lag in airplane response can be alleviated (Ref 7:7).

Direct Lift Control Defined

Within this thesis, Direct Lift Control (DLC) means airplane flight path control by lift forces applied in the direction that the pilot desires to move the airplane. The purpose of DLC is to improve handling characteristics during landing approach, terrain following, aerial refueling, and other precision maneuvering tasks by providing a change in altitude or rate of climb without a change in pitch attitude and without the delay in lift buildup associated with conventional elevator control. The LAMS B-52 Direct Lift Controller will accomplish this by moving the elevator, spoilers, and (symmetrically operated) ailerons in response to the pilot's longitudinal stick commands. Small stick movements must produce lift without pitch to provide the desired precision maneuvering capability. With larger stick commands (for changes in rates of climb or descent in excess of 400-500 feet per minute from trim), the design must function as a conventional control system, providing pitching moments and aircraft rotation. The controller will be closed-loop with normal acceleration, pitch angle, and pitch rate feedbacks.

History of Direct Lift Control

The U.S. Navy, motivated by the requirements of carrier landings, requested Douglas Aircraft Company in 1961 to study means of generating lift without pitch (Ref 8:2). Douglas proved the feasability of the idea, and designed the first direct lift control system. Test pilots who flew this system (via a cockpit simulator tied to an analog computer) were enthusiastic about the improved handling qualities and precise maneuvering capabilities provided by DLC.

Following this study, the Navy and Ling-Temco-Vought conducted wind tunnel, simulator, and flight tests of an F-8C Crusader with a DLC system. The F-8C was chosen because it was particularly difficult to maneuver during carrier approach and landing. DLC was accomplished by rapidly varying the deflection of the ailerons, which were drooped symmetrically as wing trailing edge flaps. The elevator was interconnected with the system to approximately cancel the small pitching moment produced by the ailerons. The pilot commanded DLC by turning a spring-loaded-to-neutral control wheel mounted on top of the normal control stick.

> DLC significantly increases the pilot's ability to control glide slope . . . and reduces average sink speed. It improves overall landing approach characteristics mainly by allowing rapid and precise vertical glide

> path corrections without the necessity for pitch attitude changes . . . A reduction in recommended approach speed may be achieved when using DLC . . . (Ref 6:1).

The Naval Air Test Center recommended incorporation of the DLC system in fleet F-8 airplanes, but this has not yet been done. However, a second recommendation has been carried out. This involved a feasability study of several carrier-based aircraft to determine the suitability of each type for incorporation of DLC.

Of course, DLC offers benefits to commercial airlines as well as military aircraft. Thus, the NASA Ames Research Center conducted flight tests of a Convair 990 jetliner, using biased spoilers (speed brakes) to provide DLC during the landing approach (Ref 5:2-5). Before entering the approach pattern, the spoilers were partially extended (biased up) and the airplane was trimmed to fly in this condition. Lift could then be increased by partially retracting the spoilers, or decreased by extending them still further. This system is admittedly crude, since the pilot must manipulate a DLC (spoiler) control lever which is in an awkward position simultaneously with the throttle and control stick. Nevertheless, the test pilots were again impressed by the improved maneuvering capability provided by DLC. NASA is continuing this project with a DC-8 simulation using more advanced DLC systems.

The Air Force Flight Dynamics Laboratory recently sponsored a one-year study of the applicability of DLC to cargo planes, fighter-bombers, and strategic bombers. With Honeywell Inc. as the contractor, this study was conducted to determine the increased mission effectiveness that can be achieved with control systems using direct lift control. Three aircraft types were investigated: cargo, fighter, and strategic bomber. A pilot-in-the-loop simulator study was performed using the C-5A and F-104 as testbeds. Although the study is complete, the final report has not yet been published. Reference nine is the second of five preliminary reports.

This thesis and the AFFDL study are complimentary in that Honeywell employed conventional control system design theories while the Direct Lift Controller presented here resulted from an application of optimal control theory. In addition, Honeywell's concept of direct lift control is slightly different from the definition on page four.

One of the conclusions in Honeywell's preliminary report (Ref 9:58) was that a closed loop DLC system is superior to an open loop controller, especially in turbulence. In addition, a "blended" system (controlled by normal column movements) was judged superior to a "separate" DLC (separate actuating mechanism as used in the F-8C and Convair 990) (Ref 9:58). Therefore, this study considered blended closed-loop designs exclusively.

Optimal Control Theory

One of the major objectives of this thesis is to demonstrate the applicability of optimal control theory to the design of a Direct Lift Controller. A "linear plant, quadratic cost function" theory developed by R.E. Kalman and his associates was used. This theory designs controllers through digital computer solution of the matrix Riccati equation. There is no theoretical limit on the number of aircraft states and control surfaces that can be used. Since iterative solution of the equations is not required, the digital computer programs are fast-running. Complete details are found in Chapter II.

LAMS B-52

Many of the assumptions and limitations in this thesis are based on the ultimate goal of flight testing the resulting design on the LAMS B-52, although flight test is not part of the thesis itself. LAMS, Load Alleviation and Mode Stabilization, is a current project of the Air Force Flight Dynamics Laboratory. Its objective is to determine methods of increasing the fatigue life of large airplanes by reducing vibrations and flexing of the airframe as a result of wind gusts and/or aircraft maneuvers.

The LAMS B-52 is a modified and heavily instrumented B-52E. The control surfaces themselves (ailerons, spoilers, and elevators) are standard. However, they have been fitted with new fast-responding broad bandwidth actuators. Two transistorized analog computers (Electronics Associates Inc., EAI TR-48's) are installed in the aircraft 41 section to simulate a stability augmentation system (which the B-52 lacks) and implement the LAMS controller (Ref 1:84).

The LAMS controller is only used in three specific flight conditions, which can be described as terrain following, approach and landing, and high altitude cruise (inflight refueling). The Direct Lift Controller is therefore designed for these same three flight conditions. (One of the conclusions, however, is that a single fixed-gain controller would probably be adequate for all three flight conditions.

The LAMS system operates fly-by-wire from the command (left) seat. (Fly-by-wire simply means that the connection between the pilot and the control surface actuators is an electrical one as opposed to a conventional mechanical linkage.) The LAMS aircraft could be used to flight test this Direct Lift Controller by using the existing TR-48 computers, normal acceleration, pitch angle, and pitch rate sensors, and fly-by-wire control system. This demonstrates the flexibility inherent in a fly-by-wire control system.

Actually, direct lift control is one of the techniques used by LAMS to reduce gust loading and dampen structural mode oscillations. The ailerons are operated symmetrically and the spoilers are biased up 20°. Direct lift control, however, is <u>not</u> employed for pilot-induced maneuver commands.

Division of Responsibility

Although both partners are intimately familiar with all portions of this thesis, there is a definite division of responsibilities in accordance with AFIT Department of

Electrical Engineering policy.

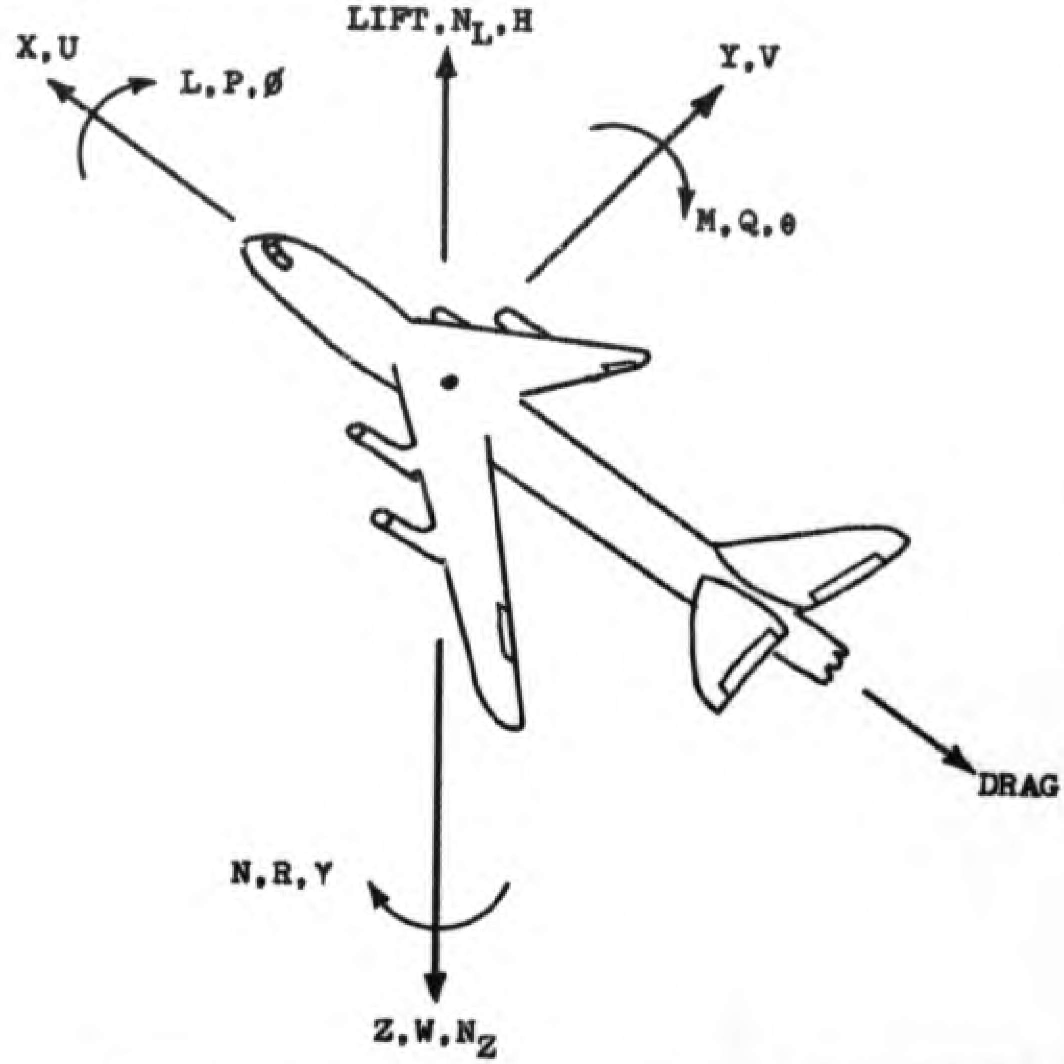
Captain Lorenzetti was primarily responsible for the material in Chapters II and III, the digital optimization programs and the two degree of freedom analog program. He also wrote Chapter I.

Captain Nelsen was primarily responsible for the material in Chapters IV and V, the six degree of freedom analog program, and all details of the cockpit simulation. In addition, he wrote Chapter VI and served as primary test pilot.

Of necessity, both writers collaborated on discussion, conclusions, and recommendations, Chapters VII and VIII. All schematics, drawings, figures, and graphs were done jointly by the authors.

Sign Conventions

Standard NACA sign conventions, used where applicable, are summarized below:



L - Rolling Moment	P - Roll Angular Velocity
M - Pitching Moment	Q - Pitch Angular Velocity
N - Yawing Moment	R - Yaw Angular Velocity
Ø - Roll Angle	X, Y, Z - Aerodynamic Force Components
θ - Pitch Angle	U, V, W - Velocity Components
γ - Yaw Angle	

Stick. Forward stick deflection is positive and causes positive elevator deflection. Full stick deflection equals computer reference voltage in both simulations.

Elevator. Elevator trailing edge deflection down from neutral is positive, producing positive lift and negative pitching moment.

Spoiler. Spoiler trailing edge deflection up from the wing surface is positive, producing negative lift and positive pitching moment. Negative spoiler deflection does not exist.

Symmetric Aileron. Symmetric aileron trailing edge deflection down from neutral is positive, producing positive lift and negative pitching moment.

Wheel. Wheel turn to the right is positive, producing a positive rolling moment.

Conventional Ailerons. Right aileron trailing edge deflection up from neutral is positive, producing positive rolling moment. Left aileron trailing edge deflection down from neutral is positive, also producing positive rolling moment.

Surface Rates and Deflection Limits

A major modification to the LAMS B-52 was the installation of special fast response actuators on all flight control surfaces. The aileron actuators, for example, can provide 120 degrees per second surface rate. (Ref 1:83). Due to these high rates, the analog programs incorporate first order approximations for conventional servos (K/S + K, where K = 1/Time constant). These time constants were varied from 0.1 second to as much as six seconds in some runs. This means that actual control surface movements commanded by the direct lift controller are relatively slow, and success of this scheme does not depend on the LAMS high speed actuators. For flight testing in the LAMS aircraft, control surface movements can be assumed instantaneous, and these servos can be regarded as part of the controller.

Elevator. Elevator limiter was set to 17° for all flight conditions.

Spoilers. Spoilers were biased up 20° for all direct lift control runs. The controller commands were limited to 40° of spoiler in all flight conditions. This provided adequate direct lift while reserving the remaining 20° of spoiler movement for turning control and/or speed brakes as required in an actual flight.

Ailerons. Each aileron was limited to ±17° deflection in all flight conditions.

Chapter II

Direct Lift Controller Design

Three digital quadratic optimal control programs were used to generate direct lift controllers in this study. The theory advanced by R.E. Kalman and his associates was used since it easily handles the dimensionality of the problem. This study used three or four states and three controls, but the numbers of both could be increased with little difficulty. Kalman's linear system theory defines controllers through digital computer solution of the matrix Riccati equation. The plant must be linearized, and the cost function must be quadratic.

Linearized Equations of Motion

As explained in Chapter IV, the LAMS equations are written for perturbations about three specific flight conditions. The digital program is based on a short period approximation of the LAMS B-52 (Ref 3:40-44) with spoiler lift and moment coefficients linearized about the 20° trailing edge up bias position. This reduces the aircraft to a linear system, as required, capable of pitch and vertical movement at constant airspeed.

In addition to the above, the aircraft is assumed trimmed for straight and level flight with spoilers biased up before the direct lift controller begins operation. Thus, the equations of motion reduce to the following:

$$\dot{W} = QU_0 + \frac{qS}{m}\Big[-(C_{L_\alpha}+C_D)\Delta\alpha - C'_{L_q}Q - C'_{L_{\dot\alpha}}\dot\alpha - C_{L_{\delta_e}}\delta_e + C_{L_{\delta_{sp}}}\delta_{sp} - C_{L_{\delta_{ail}}}\delta_{ail}\Big] \quad (2\text{-}1)$$

where

$$q = 1/2\ \rho U_0^2$$

$$C'_{L_q} = \frac{\bar c}{2U_0}\left[\frac{\partial C_L}{\partial(q\bar c/2U_0)}\right]$$

$$C'_{L_{\dot\alpha}} = \frac{\bar c}{2U_0}\left[\frac{\partial C_L}{\partial(\dot\alpha\bar c/2U_0)}\right]$$

$Q = \dot\theta$ (due to absence of lateral motion)

$\dot\alpha = \dot W/U_0$

$$\text{Let } AA = 1 + \frac{qS}{mU_0}C'_{L_{\dot\alpha}}$$

$$\dot\alpha = \{1/AA\}\Big\{Q + \frac{qS}{mU_0}\Big[-(C_{L_\alpha}+C_D)\Delta\alpha - C'_{L_q}Q - C_{L_{\delta_e}}\delta_e + C_{L_{\delta_{sp}}}\delta_{sp} - C_{L_{\delta_{ail}}}\delta_{ail}\Big]\Big\} \quad (2\text{-}2)$$

$$\dot Q = \frac{qS\bar c}{I_{yy}}\Big[C_{M_\alpha}\Delta\alpha + C'_{m_q}Q + C'_{M_{\dot\alpha}}\dot\alpha + C_{M_{\delta_e}}\delta_e + C_{M_{\delta_{sp}}}\delta_{sp} + C_{M_{\delta_{ail}}}\delta_{ail}\Big] \quad (2\text{-}3)$$

where

$$C'_{M_q} = \frac{\bar{c}}{2U_0}\left[\frac{\partial C_M}{\partial(q\bar{c}/2U_0}\right]$$

$$C'_{M_{\dot{\alpha}}} = \frac{\bar{c}}{2U_0}\left[\frac{\partial C_M}{\partial(\dot{\alpha}\bar{c}/2U_0)}\right]$$

Two of the optimization programs use the state N_L, normal acceleration of the aircraft center of gravity, positive in the up direction. An equation for $\dot{N}_L$ was obtained as follows:

$$N_L = U_0\dot{\theta} - \dot{W} \qquad \text{(Ref 3:15-16)} \qquad (2\text{-}4)$$

Since U_0, the airspeed, is considered constant,

$$\dot{N}_L = U_0\ddot{\theta} - \ddot{W} = U_0\dot{Q} - U_0\ddot{\alpha} = U_0(\dot{Q} - \ddot{\alpha})$$

Treating all aerodynamic coefficients as constants,

$$\ddot{\alpha} = \{1/AA\}\left\{\dot{Q} + \frac{qS}{mU_0}\left[-(C_{L_\alpha} + C_D)\dot{\alpha} - C'_{L_q}\dot{Q}\right]\right\}$$

As explained earlier, the LAMS aircraft has high speed actuators on all control surfaces. For practical purposes, commanded control surface movements may be considered instantaneous, so terms containing control surface rates have been dropped. Servo actuator speeds are discussed on page 148.

Let $AB = qS/[mU_0(AA)]$

$$\dot{N}_L = U_0\left\{\dot{Q}(1 - 1/AA) + AB\left[(C_{L_\alpha} + C_D)\dot{\alpha} + C'_{L_q}\dot{Q}\right]\right\} \qquad (2\text{-}5)$$

Optimal Control Theory

Chapter nine of Athans and Falb (Ref 2:750-780) is a comprehensive discussion of Kalman's optimal control theories for a linear plant with quadratic cost function. Portions of the theory actually used in this study are summarized below.

Consider the so-called state regulator problem. Its solution is an optimal feedback system which keeps the components of the state vector $\underline{X}(t)$ near zero without excessive use of the controls $\underline{U}(t)$. The state equation for a completely controllable linear time invariant system is

$$\dot{\underline{X}}(t) = A\underline{X}(t) + B\underline{U}(t) \qquad (2\text{-}6)$$

where A and B are time invariant matrices. For n states and m controls, A is n x n, and B is n x m. The general form of the quadratic cost function is

$$J = \underline{X}^T F \underline{X} + \int_0^{t_F} (\underline{X}^T Q \underline{X} + \underline{U}^T R \underline{U})\, dt \qquad (2\text{-}7)$$

If, with the specified time invariant system, the Q and R matrices are constant and the final time, t_F, is infinity, then the resulting optimal feedback system is linear and time invariant. (Ref 2:751). To insure a finite control, the R matrix, the cost of control, must be positive definite (all eigenvalues positive). The Q matrix, the cost of non-zero states, need only be positive semidefinite (eigenvalues may be zero or positive). The F matrix is the cost of non-zero final states. Since the final time is infinite, an optimal controller will produce zero final state, and F can be the zero matrix.

Under the precise conditions listed above, with $\underline{U}(t)$ not constrained, Kalman has shown (Ref 2:771) that an optimal control exists, is unique, is stable and is given by

$$\underline{U}(t) = -R^{-1}B^{T}C\underline{X}(t) \tag{2-8}$$

where C is the constant n x n positive definite matrix which is the solution of the nonlinear matrix algebraic equation

$$-CA-A^{T}C + CBR^{-1}B^{T}C - Q = 0 \tag{2-9}$$

Algebraic solution is difficult since it involves choosing the proper roots of a quadratic to make C positive definite. In practice, C is readily obtained by integrating the matrix Riccati equation

$$\dot{C}(t) = -C(t)A - A^{T}C(t) + C(t)BR^{-1}B^{T}C(t) - Q \tag{2-10}$$

backwards in time with the boundary condition

$$C(t_F) = 0$$

As backward integration proceeds, the transient due to the "initial condition" $C(t_F) = 0$ dies out, and all elements of the C matrix become constant (See Fig. 2-1 below). C is n x n and symmetric and therefore represents (n x n + n)/2 independent differential equations. Combining the plant and control equations, (2-6) and (2-8), we have

$$\dot{\underline{X}}(t) = G\underline{X}(t) \tag{2-11}$$

where

$$G = A - BR^{-1}B^{T}C \tag{2-12}$$

Kalman has proved that the eigenvalues of G always have negative real parts, so the optimal system is always stable. This is true even if one or more eigenvalues of A have non-negative real parts.

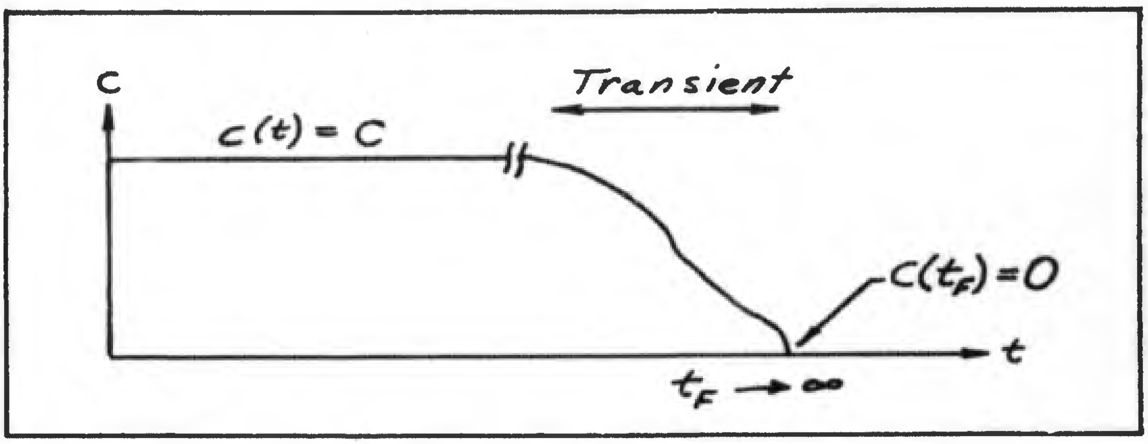


Fig. 2-1. Typical Element of C Matrix

<u>Computer Program A</u>

This first of three optimal control programs uses three states and three controls. The states are:

$X(1) = \alpha$ (Angle of attack)

$X(2) = \theta$ (Pitch angle)

$X(3) = \dot{\theta}$ (Pitch rate)

The controls, which are the same in all three programs, are:

$U(1) = \delta_e$ (Elevator)

$U(2) = \delta_{sp}$ (Spoiler)

$U(3) = \delta_{ail}$ (Symmetric ailerons)

The sequence of operations is identical in the three programs, the statement numbering is similar, and the subroutines are the same except for dimensioning. Thus, the following discussion will suffice for all three programs.

Computer Symbols. The A,B,C,Q, and R matrices and the X and U vectors correspond to the nomenclature of the theory discussed above.

CONT = $-R^{-1}B^{T}C$, establishing the control vector.

N = Number of states

M = Number of controls

TINC = Integration time increment, normally .05 seconds

KTR is a counter used for program control.

TF = Final time. In this program, the C matrix becomes constant after no more than four seconds of real-time backward integration. TF was arbitrarily chosen as ten seconds.

KS = Flight condition

D matrix contains the aerodynamic coefficients for the three flight conditions as listed in Figure 2-2, page 21. These coefficients were furnished by The Boeing Co. (Wichita, Kansas) and are based, in most cases, on B-52 flight test data. The spoilers are nonlinear, and the listed coefficients represent a tangent to the spoiler attenuation curve at the 20° bias point. See Figure 3-1, page 58 for the complete curve.

Program Flow. A listing of computer program A appears on pages 24-27.

ELEMENT	COEFFICIENT	FLT. COND. 1	FLT. COND. 2	FLT. COND. 3
D(1,KS)	C_L (Clean)	.211	.450	.293
D(2,KS)	$C_{L\alpha} + C_D$	3.655	3.890	4.795
D(3,KS)	$C_{L\delta_e}$	.1052	.1578	.1458
D(4,KS)	$C_{M\alpha}$	-.545	-.762	-.758
D(5,KS)	$C_{M\delta_e}$	-.345	-.517	-.484
D(6,KS)	$q S \bar{c} / I_{YY}$	5.4841	2.5750	3.3800
D(7,KS)	qS/M	152.2	71.6	109.84
D(8,KS)	$C'_{L_q} = \frac{\bar{c}}{2U_0}\left[\frac{\partial C_L}{\partial(q\bar{c}/2U_0)}\right]$	.0538	.0876	.0534
D(9,KS)	$C'_{L\dot{\alpha}} = \frac{\bar{c}}{2U_0}\left[\frac{\partial C_L}{\partial(\dot{\alpha}\bar{c}/2U_0)}\right]$	.0130	.0212	.0130
D(10,KS)	$C'_{M_q} = \frac{\bar{c}}{2U_0}\left[\frac{\partial C_M}{\partial(q\bar{c}/2U_0)}\right]$	-.230	-.383	-.233
D(11,KS)	$C'_{M\dot{\alpha}} = \frac{\bar{c}}{2U_0}\left[\frac{\partial C_M}{\partial(\dot{\alpha}\bar{c}/2U_0)}\right]$	-.0372	-.0622	-.0376
D(12,KS)	$C_{L\delta sp}$	.1152	.1524	.1897
D(13,KS)	$C_{M\delta sp}$	.0131	.1496	.1352
D(14,KS)	U_0	626.4	429.4	757.0
D(15,KS)	$C_{L\delta ail}$	.068	.100	.106
D(16,KS)	$C_{M\delta ail}$	-.024	-.036	-.042

Fig. 2-2. D Matrix: Aerodynamic Coefficients

Statements 0 to 99: Zero matrices as required, read data, and write out Q and R matrices for reference.

Statements 99 to 56: Check to insure R matrix is positive definite and Q matrix positive semidefinite as required.

Statements 56 to 29: Compute A and B matrices and write out for reference. Program A uses equation (2-2) above for $\dot{X}(1)$ and equation (2-3) for $\dot{X}(3)$. Of course, $\dot{X}(2) = X(3)$.

Statements 29 to 12: Compute BRIBT ($BR^{-1}B^T$), preparing for backward integration of the matrix Riccati equation.

Statements 12 to 32: Integrate C matrix backwards, write out independent elements of C each TINC, and check for constancy of C matrix each ten TINC (usually 0.5 seconds).

Statements 32 to 42: Compute and write out CONT. This matrix establishes the pot settings to be used in the analog simulations.

Statements 42 to 17: Compute and write out the G matrix. ($G = A - R^{-1}B^TC$ and $\dot{X} = G\underline{X}$). Find characteristic equation and eigenvalues of G to demonstrate stability of the optimal system.

Statements 17 to 45: Read an initial X condition (normally $\theta = 0.1$ radian), integrate the optimal system equations forward in time for five seconds, writing out states and controls used each TINC.

Although every controller produced by this program is optimal for the cost criteria contained in the Q and R matrices, several runs are usually required with different Q and R matrices to produce a controller with good flying qualities. Suitability of a controller can only be established by an analog computer simulation. However, the integration of the optimal system equations gives a basis for comparison of the various controllers, and helps eliminate some unsuitable ones without an analog simulation.

Manipulation of the Q and R matrices is discussed in a later section, and a sample output of computer program A is included in Appendix A.

COMPUTER OPTIMIZATION PROGRAM A

```
$IBJOB          NOMAP
$IBFTC MAIN     XR7
      COMMON A(3,3),B(3,3),D(16,3),Q(3,3),R(3,3),C(3,3),WKA(3,3),
     ZWKB(3,3),WKC(3,3),WKD(3,3),RIBT(3,3),M,N,BRIBT(3,3)
      DIMENSION X(6),COMP(3,3),CHAR(4),EIGNV(3,2),DUMMY(6),CONT(3,3),
     ZU(3,1),WKP(3,3),XX(3,1)
      NAMELIST/INIT/N,KS,M,TF,TINC
 5    FORMAT (5E10.0)
 2    FORMAT(3E10.0)
 3    READ (5,INIT)
      KTR=1
      DO 4 I=1,N
      DO 4 J=1,N
      A(I,J) = 0.
      COMP(I,J)=0.
      C(I,J)=0.
 4    Q(I,J)=0.
      DO 26 I=1,M
      DO 26 J=1,M
      R(I,J)=0.
 26   WKP(I,J)=0.
      DO 6 I=1,N
      DO 6 J=1,M
      U(J,1)=0.
 6    B(I,J)=0.
      READ(5,2) ((Q(I,J),J=1,N),I=1,N)
      READ(5,2) ((R(I,J),J=1,M),I=1,M),((D(I,J),J=1,3),I=1,16)
 46   CONTINUE
 22   FORMAT(1X,8HQ MATRIX,//)
      WRITE(6,22)
 23   FORMAT(5F12.4)
      WRITE(6,25) ((Q(I,J),J=1,N),I=1,N)
 24   FORMAT(//,1X,8HR MATRIX,//)
      WRITE(6,24)
 25   FORMAT(3F12.4)
      WRITE(6,25) ((R(I,J),J=1,M),I=1,M)
C  CHECK ACCEPTABILITY OF Q AND R MATRICES
 99   FORMAT(/,1X,14HQ MATRIX CHECK)
      WRITE(6,99)
      CALL CHRPOL (Q,CHAR,N)
      CALL ROOTS (N,CHAR,EIGNV)
      DO 51 I=1,N
      TNT=EIGNV(I,1)
      IF (TNT.LT. 0.)  GO TO 52
 51   CONTINUE
 100  FORMAT(/,1X,14HR MATRIX CHECK)
      WRITE(6,100)
      CALL CHRPOL(R,CHAR,M)
      CALL ROOTS (M,CHAR,EIGNV)
```

```
      DO 56  I=1,M
      TNT=EIGNV(I,1)
      IF(TNT .LE. 0.)  GO TO 55
 56   CONTINUE
 31   FORMAT(//,1X,17HFLIGHT CONDITION ,I1,//)
 50   WRITE(6,31) KS
C COMPUTE A AND B MATRICES
      AA=1. + D(7,KS)*D(9,KS)/D(14,KS)
      AB=D(7,KS)/(D(14,KS)*AA)
      A(1,1)=-AB*D(2,KS)
      A(1,3)=1./AA-AB*D(8,KS)
      B(1,1)=-AB*D(3,KS)
      B(1,2)=AB*D(12,KS)
      B(1,3)=-AB*D(15,KS)
      A(2,3)=1.
      AC=D(6,KS)
      AD=D(11,KS)
      A(3,1)=AC*(D(4,KS)+AD*A(1,1))
      A(3,3)=AC*(D(10,KS)+AD*A(1,3))
      B(3,1)=AC*(D(5,KS)+AD*B(1,1))
      B(3,2)=AC*(D(13,KS)+AD*B(1,2))
      B(3,3)=AC*(D(16,KS)+A  B(1,3))
 28   FORMAT(//,1X,8HA MATRIX,//)
      WRITE(6,28)
      WRITE(6,25) ((A(I,J),J=1,N),I=1,N)
 29   FORMAT(//,1X,8HB MATRIX,//)
      WRITE(6,29)
      WRITE (6,25) ((B(I,J),J=1,M),I=1,N)
      CALL GMTRA (B,CONT,N,M)
      CALL MTXINV(R,WKP,M)
      CALL GMPRD(WKP,CONT,RIBT,M,M,N)
      CALL GMPRD(B,RIBT,BRIBT,N,M,N)
      TS=TF
      DO 7  I=1,6
 7    X(I)=0.
 30   FORMAT(////,1X,39HELEMENTS OF SYMMETRIC FEEDBACK MATRIX C,//)
      WRITE (6,30)
 13   FORMAT(1X,6HC(1,1),7X,6HC(1,2),6X,6HC(1,3),6X,6HC(2,2),6X,6HC(2,
     Z,6X,6HC(3,3),//)
      WRITE (6,13)
 14   FORMAT(/,1X,7HTIME = ,F6.2)
      DELT =-TINC
 11   IF(TS.GT.0.)  GO TO 12
      STOP
C  INTEGRATE C MATRIX BACKWARDS, 10 STEPS
 12   DO 8  I=1,10
C CALL (NO. EQN, EQN SET, VARIABLE, START TIME, DEL TIME)
      CALL RUNGE(6,1,X,TS,DELT)
 9    FORMAT(1P10E12.4)
      WRITE (6,14) TS
 8    WRITE (6,9)  (X(J),J=1,6)
      CALL FCN(6,2,X,DUMMY)
      CALL MXPM (COMP,C,WKA,N,N,-1)
      SUM =0.
      DO 10 I=1,N
      DO 10 J=1,N
```

```
      COMP(I,J)=C(I,J)
 10   SUM = SUM + ABS(WKA(I,J))
C CHECK FOR CONSTANCY OF C MATRIX
      IF (SUM .GT. 0.3)  GO TO 11
C NOW HAVE CONSTANT FEEDBACK GAIN MATRIX, C.
 32   FORMAT(//,1X,17HFEEDBACK MATRIX C,//)
      WRITE (6,32)
 33   FORMAT (3F12.4)
      WRITE (6,33) ((C(I,J),J=1,N),I=1,N)
      DO 15 I=1,N
      DO 15 J=1,N
 15   WKA(I,J) = 0.
      CALL GMPRD(RIBT,C,CONT,M,N,N)
      DO 16 I=1,M
      DO 16 J=1,N
 16   CONT(I,J) = -CONT(I,J)
 42   FORMAT(//,1X,15HCONTROL= -RIBTC)
      WRITE (6,42)
      WRITE (6,33) ((CONT(I,J),J=1,N),I=1,M)
C CONTROL VECTOR, U = (CONT)(X VECTOR)
C U(1) = ELEVATOR, U(2) = SPOILERS, U(3) = AILERONS.
      CALL GMPRD (B,CONT,WKB,N,M,N)
      CALL MXPM (A,WKB,WKA,N,N,+1)
 40   FORMAT(//,1X,8HA-BRIBTC,//)
      WRITE (6,40)
      WRITE (6,33) ((WKA(I,J),J=1,N),I=1,N)
C X DOT = (WKA)(X)
      CALL CHRPOL(WKA,CHAR,N)
C  FIND CHARACTERITIC EQUATION OF REAL MATRIX (A) BY BOCHER FORMULA. RI
C  234 OF DERUSSO.
      CALL ROOTS(N,CHAR,EIGNV)
C  CHARACTERISTIC EQUATION AND ITS ROOTS ARE FOUND FOR LATER COMPARISOI
C  MODIFIED CONTROLLER.
C READ IN INITIAL X CONDITIONS.
C  X(1)=ALPHA  X(2)=THETA  X(3)=THETA DOT=Q
 17   READ(5,2) (X(I),I=1,3)
      DELT=TINC
 21   FORMAT(//,1X,48HNOTE, CONTROLS IN DEGREES AND ANGLES IN RADIANS.
     Z//,1X,30HSPOILERS BIASED UP 20 DEGREES.,//)
      WRITE(6,21)
 57   FORMAT(/,1X,36HUSING OUTBOARD SPOILERS ONLY (0.56K))
      WRITE (6,57)
 20   FORMAT(4X,4HTIME,8X,5HALPHA,10X,5HTHETA,5X,9HTHETA DOT,5X,
     Z8HELEVATOR,4X,8HSPOILERS,4X,8HAILERONS,//)
      WRITE(6,20)
      TS=0.
      U(2,1)=20.
 19   FORMAT(7F12.4)
      WRITE(6,19) TS,(X(I),I=1,3),(U(J,1),J=1,M)
      TF=.5*TF
 18   CALL RUNGE(3,3,X,TS,DELT)
      DO 41  I=1,N
 41   XX(I,1)=X(I)
      CALL GMPRD(CONT,XX,U,M,N,1)
      U(1,1)=57.3*U(1,1)
      U(2,1)=57.3*U(2,1)+20.
```

```
      U(3,1)=57.3*U(3,1)
      WRITE(6,19) TS,(X(I),I=1,N),(U(J,1),J=1,M)
      IF (TS.LT.TF)  GO TO 18
      TF=2.*TF
      KTR=KTR + 1
      IF(KTR.LT.4)  GO TO 45
      STOP
45    KS=KS+1
      GO TO 56
52    WRITE(6,53)
53    FORMAT(/,1X,47HQ MATRIX NOT POSITIVE SEMIDEFINITE AS REQUIRED.)
      GO TO 50
54    FORMAT(/,1X,43HR MATRIX NOT POSITIVE DEFINITE AS REQUIRED.)
55    WRITE (6,54)
      GO TO 50
      END
```

Computer Program B

This optimal control program uses four states and the usual three controls. The states are:

$X(1) = \alpha$ (Angle of attack)

$X(2) = \theta$ (Pitch angle)

$X(3) = \dot{\theta}$ (Pitch rate)

$X(4) = N_L$ (Normal acceleration, positive in the up direction)

Program B differs from A in dimensioning statements and in the addition of equation (2-5) for X(4). Thus, the optimal controller has four feedbacks instead of three. A listing of this program appears on pages 30-33.

Note that the actual state X(4) used in this program is .01 N_L rather than N_L itself. As shown in the sample output (Appendix A), this makes all elements of the A and B matrices relatively equal in magnitude. When the program was run with N_L, the row four elements of the A and B matrices were two orders of magnitude greater than any other elements, giving an ill-conditioned set of differential equations that was virtually impossible to integrate. Even though this program uses a variable step size Runge-Kutta differential equation solver routine, integration was very slow with repeated overflows and underflows. The results, of course, were useless. However, by simply using .01 N_L, backward integration of the C matrix proceeds rapidly without overflow or underflow.

This is an important point that will eliminate much of the difficulty commonly associated with application of this theory. Simple conditioning of the equations, analogous to amplitude scaling on an analog computer, will save computer time by speeding integration while improving accuracy of the results.

Additional difficulty was experienced with this program in that elements C(1,1) and C(4,4) of the C matrix did not become constant during the backwards integration. This is probably due to the fact that X(4), normal acceleration, is a combination of X(1), alpha, and X(2), theta. (See <u>Linearized Equations of Motion</u> above.) However, a series of special runs demonstrated that although the two C matrix elements did not become constant, the control (CONT) and optimal system (G) matrices computed from C did become essentially constant after approximately ten seconds of backwards integration. Therefore, the matrix Riccati equation was integrated backwards for 20 seconds in each run, and the controllers thus computed gave excellent results on the analog simulation. A sample output of this program is included in Appendix A.

COMPUTER OPTIMIZATION PROGRAM B

```
$IBJOB          NOMAP
$IBFTC MAIN     XR7
      COMMON A(4,4),B(4,3),D(16,3),Q(4,4),R(3,3),C(4,4),WKA(4,4),
     ZWKB(4,4),WKC(4,4),WKD(4,4),RIBT(3,4),M,N,BRIBT(4,4)
      DIMENSION X(10),COMP(4,4),CHAR(5),EIGNV(4,2),DUMMY(10),CONT(3,4),
     ZU(3,1),WKP(3,3),XX(4,1)
      NAMELIST/INIT/N,KS,M,TF,TINC
 5    FORMAT (4E10.0)
 2    FORMAT(3E10.0)
 3    READ (5,INIT)
      KTR=1
      DO 4 I=1,N
      DO 4 J=1,N
      A(I,J) = 0.
      COMP(I,J)=0.
      C(I,J)=0.
 4    Q(I,J)=0.
      DO 26 I=1,M
      DO 26 J=1,M
      R(I,J)=0.
 26   WKP(I,J)=0.
      DO 6 I=1,N
      DO 6 J=1,M
      U(J,1)=0.
 6    B(I,J)=0.
      READ(5,5) ((Q(I,J),J=1,N),I=1,N)
      READ(5,2) ((R(I,J),J=1,M),I=1,M),((D(I,J),J=1,3),I=1,16)
 46   CONTINUE
 22   FORMAT(1X,8HQ MATRIX,//)
 101  WRITE (6,22)
 23   FORMAT(4F12.4)
      WRITE (6,23) ((Q(I,J),J=1,N),I=1,N)
 24   FORMAT(//,1X,8HR MATRIX,//)
      WRITE(6,24)
 25   FORMAT(3F12.4)
      WRITE(6,25) ((R(I,J),J=1,M),I=1,M)
C  CHECK ACCEPTABILITY OF Q AND R MATRICES
 99   FORMAT(/,1X,14HQ MATRIX CHECK)
      WRITE(6,99)
      CALL CHRPOL (Q,CHAR,N)
      CALL ROOTS (N,CHAR,EIGNV)
      DO 51 I=1,N
      TNT=EIGNV(I,1)
      IF (TNT.LT. 0.)  GO TO 52
 51   CONTINUE
 100  FORMAT(/,1X,14HR MATRIX CHECK)
      WRITE(6,100)
      CALL CHRPOL(R,CHAR,M)
      CALL ROOTS (M,CHAR,EIGNV)
```

```
      DO 56  I=1,M
      TNT=EIGNV(I,1)
      IF(TNT .LE. 0.)  GO TO 55
56    CONTINUE
31    FORMAT(//,1X,17HFLIGHT CONDITION ,I1,//)
50    WRITE(6,31) KS
C COMPUTE A AND B MATRICES
      AA=1. + D(7,KS)*D(9,KS)/D(14,KS)
      AB=D(7,KS)/(D(14,KS)*AA)
      A(1,1)=-AB*D(2,KS)
      A(1,3)=1./AA-AB*D(8,KS)
      B(1,1)=-AB*D(3,KS)
      B(1,2)=AB*D(12,KS)
      B(1,3)=-AB*D(15,KS)
      A(2,3)=1.
      AC=D(6,KS)
      AD=D(11,KS)
      A(3,1)=AC*(D(4,KS)+AD*A(1,1))
      A(3,3)=AC*(D(10,KS)+AD*A(1,3))
      B(3,1)=AC*(D(5,KS)+AD*B(1,1))
      B(3,2)=AC*(D(13,KS)+AD*B(1,2))
      B(3,3)=AC*(D(16,KS)+AD*B(1,3))
      AU=D(14,KS)
      AE=(AU-D(7,KS)*D(8,KS))/AA
      AF=(D(7,KS)*D(2,KS))/AA
      A(4,1)=(AU-AE)*A(3,1)+AF*A(1,1)
      A(4,1)=.01*A(4,1)
      A(4,3)=(AU-AE)*A(3,3)+AF*A(1,3)
      A(4,3)=.01*A(4,3)
      B(4,1)=(AU-AE)*B(3,1)+AF*B(1,1)
      B(4,1)=.01*B(4,1)
      B(4,2)=(AU-AE)*B(3,2)+AF*B(1,2)
      B(4,2)=.01*B(4,2)
      B(4,3)=(AU-AE)*B(3,3)+AF*B(1,3)
      B(4,3)=.01*B(4,3)
28    FORMAT(//,1X,8HA MATRIX,//)
      WRITE(6,28)
      WRITE (6,23) ((A(I,J),J=1,N),I=1,N)
29.   FORMAT(//,1X,8HB MATRIX,//)
      WRITE(6,29)
      WRITE (6,25) ((B(I,J),J=1,M),I=1,N)
      CALL GMTRA (B,CONT,N,M)
      CALL MTXINV(R,WKP,M)
      CALL GMPRD(WKP,CONT,RIBT,M,M,N)
      CALL GMPRD(B,RIBT,BRIBT,N,M,N)
      TS=TF
      DO 7  I=1,10
7     X(I)=0.
      DELT =-TINC
11    IF(TS.GT.0.)  GO TO 12
      STOP
C  INTEGRATE C MATRIX BACKWARDS, 10 STEPS
12    DO 8  I=1,10
C CALL (NO. EQN, EQN SET, VARIABLE, START TIME, DEL TIME)
      CALL  RUNGE(10,1,X,TS,DELT)
9     FORMAT(1P10E12.4)
```

```
 8     CONTINUE
       CALL FCN(10,2,X,DUMMY)
       CALL MXPM (COMP,C,WKA,N,N,-1)
       SUM =0.
       DO 10 I=1,N
       DO 10 J=1,N
       COMP(I,J)=C(I,J)
 10    SUM = SUM + ABS(WKA(I,J))
C CHECK FOR CONSTANCY OF C MATRIX
 14    FORMAT(/,1X,7HTIME = ,F6.2)
       WRITE (6,14) TS
 32    FORMAT(//,1X,17HFEEDBACK MATRIX C,//)
 91    WRITE (6,32)
 33    FORMAT (4F12.4)
       WRITE (6,33) ((C(I,J),J=1,N),I=1,N)
       IF(TS.LT.0.5)  GO TO 92
       IF(SUM.GT.2.0)  GO TO 11
C NOW HAVE CONSTANT FEEDBACK GAIN MATRIX, C.
 92    CONTINUE
       DO 15 I=1,N
       DO 15 J=1,N
 15    WKA(I,J) = 0.
       CALL GMPRD(RIBT,C,CONT,M,N,N)
       DO 16 I=1,M
       DO 16 J=1,N
 16    CONT(I,J) = -CONT(I,J)
 42    FORMAT(//,1X,15HCONTROL= -RIBTC)
 57    FORMAT(/,1X,42HNOTE ... ACTUAL STATE X(4) IS .01*N SUB L.)
       WRITE(6,57)
       WRITE (6,42)
       WRITE (6,33) ((CONT(I,J),J=1,N),I=1,M)
C CONTROL VECTOR, U = (CONT)(X VECTOR)
C U(1) = ELEVATOR, U(2) = SPOILERS, U(3) = AILERONS.
       CALL GMPRD (B,CONT,WKB,N,M,N)
       CALL MXPM (A,WKB,WKA,N,N,+1)
 40    FORMAT(//,1X,8HA-BRIBTC,//)
       WRITE (6,40)
       WRITE (6,33) ((WKA(I,J),J=1,N),I=1,N)
C X DOT = (WKA)(X)
       CALL CHRPOL(WKA,CHAR,N)
C  FIND CHARACTERITIC EQUATION OF REAL MATRIX (A) BY BOCHER FORMULA. REF
C  234 OF DERUSSO.
       CALL ROOTS(N,CHAR,EIGNV)
C  CHARACTERISTIC EQUATION AND ITS ROOTS ARE FOUND FOR LATER COMPARISON
C  MODIFIED CONTROLLER.
C READ IN INITIAL X CONDITIONS.
C  X(1)=ALPHA  X(2)=THETA  X(3)=THETA DOT=Q  X(4)= .01*N SUB L  (POSITI\
 17    READ(5,5) (X(I),I=1,N)
       DELT=TINC
 21    FORMAT(//,1X,48HNOTE, CONTROLS IN DEGREES AND ANGLES IN RADIANS.,
     2//,1X,30HSPOILERS BIASED UP 20 DEGREES.,//)
       WRITE(6,21)
 20    FORMAT(4X,4HTIME,8X,5HALPHA,10X,5HTHETA,5X,9HTHETA DOT,5X,5HN ACC:
     2EL,7X,8HELEVATOR,4X,8HSPOILERS,4X,8HAILERONS,//)
       WRITE(6,20)
       TS=0.
```

```
      U(2,1)=20.
      WRITE(6,19) TS,(X(I),I=1,N),(U(J,1),J=1,M)
      TF=.25*TF
18    CALL RUNGE(4,3,X,TS,DELT)
      DO 41  I=1,N
41    XX(I,1)=X(I)
      CALL GMPRD(CONT,XX,U,M,N,1)
      U(1,1)=57.3*U(1,1)
      U(2,1)=57.3*U(2,1)+20.
      U(3,1)=57.3*U(3,1)
      XX(4,1)=100.*XX(4,1)
19    FORMAT(8F12.4)
      WRITE(6,19) TS,(XX(I,1),I=1,N),(U(J,1),J=1,M)
      IF (TS.LT.TF)  GO TO 18
      TF=4.*TF
      KTR=KTR+1
      KS=KS+1
      IF(KTR.EQ.4)  GO TO 93
      GO TO 50
93    CONTINUE
      STOP
45    READ(5,5) ((Q(I,J),J=1,N),I=1,N)
      READ(5,2) ((R(I,J),J=1,M),I=1,M)
      GO TO 46
52    WRITE(6,53)
53    FORMAT(/,1X,47HQ MATRIX NOT POSITIVE SEMIDEFINITE AS REQUIRED.)
      GO TO 50
54    FORMAT(/,1X,43HR MATRIX NOT POSITIVE DEFINITE AS REQUIRED.)
55    WRITE (6,54)
      GO TO 50
      END
```

Computer Program C

The controllers designed by computer program B gave good results when tested on the two degree of freedom analog simulation, and are suitable for use on the LAMS B-52. However, angle of attack is difficult to measure accurately in flight, and most aircraft are not equipped with α sensors. Therefore, α feedback was reduced as much as possible through manipulation of the Q matrix. However, analog simulation still showed noticeable performance degradation when the remaining α feedback was removed. Computer program C was then written to check the roots of the system characteristic equation with α feedback eliminated (flight condition three only). For example, consider the controller below.

Flt. Cond. 3	α FB	θ FB	$\dot{\theta}$ FB	N_L FB
Elevator	3.45	19.89	14.23	-13.97
Spoilers	2.56	-1.78	-2.71	-122.18
Ailerons	-2.57	-0.24	1.09	108.83

The roots of the optimal system characteristic equation are

$S(1) = 0$

$S(2) = -0.054$

$S(3) = -2.0$

$S(4) = -23.65$

Eliminating α feedback, the roots become

$S(1) = -0.122$

$S(2) = -1.49 + J\,0.84$

S(3) = -1.49 - J 0.84

S(4) = -22.95

A listing of this program follows.

```
      DIMENSION A(4,4),B(4,3),CONT(3,4),WKA(4,4),WKB(4,4),CHAR(5),
     ZEIGNV(4,2)
      N=4
      M=3
2     FORMAT(4E10.0)
      DO 3  I=1,N
      DO 3  J=1,N
3     A(I,J)=0.
      DO 4  J=1,M
4     B(2,J)=0.
      A(1,1)=-.6939
      A(3,1)=-2.4739
      A(4,1)=-3.8279
      A(1,3)=.9904
      A(2,3)=1.
      A(3,3)=-.6518
      A(3,4)=5.1590
      B(1,1)=-.0211
      B(1,2)=.0274
      B(1,3)=-.0153
      B(3,1)=-1.6332
      B(3,2)=.4536
      B(3,3)=-.1366
      B(4,1)=-.2298
      B(4,2)=.1773
      B(4,3)=-.0906
      READ(5,2) ((CONT(I,J),J=1,N),I=1,M)
      CALL GMPRD(B,CONT,WKA,N,M,N)
      CALL MXPM(A,WKA,WKB,N,N,+1)
5     FORMAT (4F12.4)
6     FORMAT(//,1X,24HMODIFIED CONTROL, -RIBTC)
      WRITE(6,6)
      WRITE(6,5)  ((CONT(I,J),J=1,N),I=1,M)
7     FORMAT(//,1X,17HMODIFIED A-BRIBTC)
      WRITE(6,7)
      WRITE(6,5) ((WKB(I,J),J=1,N),I=1,N)
      CALL CHRPOL(WKB,CHAR,N)
      CALL ROOTS(N,CHAR,EIGNV)
      STOP
      END

C  THIS PROGRAM USES THE FOLLOWING SUBROUTINES TAKEN FROM THE
C  OPTIMIZATION PROGRAM.

C  SUBROUTINE MXPM(A,B,C,M,N,NSGN)
C  SUBROUTINE GMPRD(A,B,C,NA,NB,NC)
C  SUBROUTINE CHRPOL(A,B,N)
C  SUBROUTINE ROOTS(IDIOT,A,D)
```

Computer Program D

For reasons stated above, angle of attack is not a desireable feedback in most aircraft control systems. However, program C and the analog simulation both show that eliminating α feedback from program B's output results in sub-optimal controllers. Optimization program D was written to overcome these difficulties.

Program D is similar to program A in that it uses three states and three controls. The states are

X(1) = .01 N_L (Normal acceleration, positive in the <u>up</u> direction)

X(2) = θ (Pitch angle)

X(3) = $\dot{\theta}$ (Pitch rate)

Note that .01 N_L is again used for the same reason stated in the program B discussion. The controls are the same as in the previous programs:

U(1) = δ_e (Elevator)

U(2) = δ_{sp} (Spoiler)

U(3) = δ_{ail} (Aileron)

No difficulties were encountered with this program, and the two degree of freedom analog results were good. Since absence of α feedback makes program D controllers more practical than the others, they were used exclusively in the six degree of freedom piloted simulation at the AFFDL computer facility. A program listing follows, and a sample output is included in Appendix A.

COMPUTER OPTIMIZATION PROGRAM D

```
$IBJOB          NOMAP
$IBFTC MAIN     XR7
      COMMON A(3,3),B(3,3),D(16,3),Q(3,3),R(3,3),C(3,3),WKA(3,3),
     ZWKB(3,3),WKC(3,3),WKD(3,3),RIBT(3,3),M,N,BRIBT(3,3)
      DIMENSION X(6),COMP(3,3),CHAR(4),EIGNV(3,2),DUMMY(6),CONT(3,3),
     ZU(3,1),WKP(3,3),XX(3,1)
      NAMELIST/INIT/N,KS,M,TF,TINC
 5    FORMAT (5E10.0)
 2    FORMAT(3E10.0)
 3    READ (5,INIT)
      KTR=1
      DO 4 I=1,N
      DO 4 J=1,N
      A(I,J) = 0.
      COMP(I,J)=0.
      C(I,J)=0.
 4    Q(I,J)=0.
      DO 26 I=1,M
      DO 26 J=1,M
      R(I,J)=0.
 26   WKP(I,J)=0.
      DO 6 I=1,N
      DO 6 J=1,M
      U(J,1)=0.
 6    B(I,J)=0.
      READ(5,2) ((Q(I,J),J=1,N),I=1,N)
      READ(5,2) ((R(I,J),J=1,M),I=1,M),((D(I,J),J=1,3),I=1,16)
 46   CONTINUE
 22   FORMAT(1X,8HQ MATRIX,//)
      WRITE(6,22)
 23   FORMAT(5F12.4)
      WRITE(6,25) ((Q(I,J),J=1,N),I=1,N)
 24   FORMAT(//,1X,8HR MATRIX,//)
      WRITE(6,24)
 25   FORMAT(3F12.4)
      WRITE(6,25) ((R(I,J),J=1,M),I=1,M)
C  CHECK ACCEPTABILITY OF Q AND R MATRICES
 99   FORMAT(/,1X,14HQ MATRIX CHECK)
      WRITE(6,99)
      CALL CHRPOL (Q,CHAR,N)
      CALL ROOTS (N,CHAR,EIGNV)
      DO 51 I=1,N
      TNT=EIGNV(I,1)
      IF (TNT.LT. 0.)  GO TO 52
 51   CONTINUE
 100  FORMAT(/,1X,14HR MATRIX CHECK)
      WRITE(6,100)
      CALL CHRPOL(R,CHAR,M)
      CALL ROOTS (M,CHAR,EIGNV)
```

```
      DO 56  I=1,M
      TNT=EIGNV(I,1)
      IF(TNT .LE. 0.)  GO TO 55
 56   CONTINUE
 31   FORMAT(//,1X,17HFLIGHT CONDITION ,I1,//)
 50   WRITE(6,31) KS
C COMPUTE A AND B MATRICES
      AA=1. + D(7,KS)*D(9,KS)/D(14,KS)
      AB=D(7,KS)/(D(14,KS)*AA)
      A(1,1)=-AB*D(2,KS)
      A(1,3)=1./AA-AB*D(8,KS)
      B(1,1)=-AB*D(3,KS)
      B(1,2)=AB*D(12,KS)
      B(1,3)=-AB*D(15,KS)
      A(2,3)=1.
      AC=D(6,KS)
      AD=D(11,KS)
      A(3,1)=AC*(D(4,KS)+AD*A(1,1))
      A(3,3)=AC*(D(10,KS)+AD*A(1,3))
      B(3,1)=AC*(D(5,KS)+AD*B(1,1))
      B(3,2)=AC*(D(13,KS)+AD*B(1,2))
      B(3,3)=AC*(D(16,KS)+AD*B(1,3))
      AU=D(14,KS)
      AE=(AU-D(7,KS)*D(8,KS))/AA
      AF=(D(7,KS)*D(2,KS))/AA
      A(1,1)=.01*((AU-AE)*A(3,1)+AF*A(1,1))
      A(1,3)=.01*((AU-AE)*A(3,3)+AF*A(1,3))
      B(1,1)=.01*((AU-AE)*B(3,1)+AF*B(1,1))
      B(1,2)=.01*((AU-AE)*B(3,2)+AF*B(1,2))
      B(1,3)=.01*((AU-AE)*B(3,3)+AF*B(1,3))
 28   FORMAT(//,1X,8HA MATRIX,//)
      WRITE(6,28)
      WRITE(6,25) ((A(I,J),J=1,N),I=1,N)
 29   FORMAT(//,1X,8HB MATRIX,//)
      WRITE(6,29)
      WRITE (6,25) ((B(I,J),J=1,M),I=1,N)
      CALL GMTRA (B,CONT,M,M)
      CALL MTXINV(R,WKP,M)
      CALL GMPRD(WKP,CONT,RIBT,M,M,N)
      CALL GMPRD(B,RIBT,BRIBT,N,M,N)
      TS=TF
      DO 7  I=1,6
 7    X(I)=0.
      DELT =-TINC
 11   IF(TS.GT.0.)  GO TO 12
      STOP
C  INTEGRATE C MATRIX BACKWARDS, 10 STEPS
 12   DO 8  I=1,10
C CALL (NO. EQN, EQN SET, VARIABLE, START TIME, DEL TIME)
      CALL RUNGE(6,1,X,TS,DELT)
 9    FORMAT(1P10E12.4)
 8    CONTINUE
      CALL FCN(6,2,X,DUMMY)
      CALL MXPM (COMP,C,WKA,N,N,-1)
      SUM =0.
      DO 10 I=1,N
```

```
      DO 10 J=1,N
      COMP(I,J)=C(I,J)
 10   SUM = SUM + ABS(WKA(I,J))
C CHECK FOR CONSTANCY OF C MATRIX
 14   FORMAT(/,1X,7HTIME = ,F6.2)
      WRITE (6,14) TS
 32   FORMAT(//,1X,17HFEEDBACK MATRIX C,//)
 91   WRITE(6,32)
 33   FORMAT (3F12.4)
      WRITE (6,33) ((C(I,J),J=1,N),I=1,N)
      IF(TS.LT.0.5)  GO TO 92
      IF (SUM .GT. 0.3)  GO TO 11
C NOW HAVE CONSTANT FEEDBACK GAIN MATRIX, C.
 92   CONTINUE
      DO 15 I=1,N
      DO 15 J=1,N
 15   WKA(I,J) = 0.
      CALL GMPRD(RIBT,C,CONT,M,N,N)
      DO 16 I=1,M
      DO 16 J=1,N
 16   CONT(I,J) = -CONT(I,J)
 57   FORMAT(/,1X,42HNOTE ... ACTUAL STATE X(1) IS .01*N SUB L.)
      WRITE(6,57)
 42   FORMAT(//,1X,15HCONTROL= -RIBTC)
      WRITE (6,42)
      WRITE (6,33) ((CONT(I,J),J=1,N),I=1,M)
C CONTROL VECTOR, U = (CONT)(X VECTOR)
C U(1) = ELEVATOR, U(2) = SPOILERS, U(3) = AILERONS.
      CALL GMPRD (B,CONT,WKB,N,M,N)
      CALL MXPM (A,WKB,WKA,N,N,+1)
 40   FORMAT(//,1X,8HA-BRIBTC,//)
      WRITE (6,40)
      WRITE (6,33) ((WKA(I,J),J=1,N),I=1,N)
C X DOT = (WKA)(X)
      CALL CHRPOL(WKA,CHAR,N)
C  FIND CHARACTERITIC EQUATION OF REAL MATRIX (A) BY BOCHER FORMULA. R
C  234 OF DERUSSO.
      CALL ROOTS(N,CHAR,EIGNV)
C  CHARACTERISTIC EQUATION AND ITS ROOTS ARE FOUND FOR LATER COMPARISO
C  MODIFIED CONTROLLER.
C READ IN INITIAL X CONDITIONS.
C  X(1)=.01*N SUB L (POSITIVE UP)  X(2)=THETA  X(3)=THETA DOT=Q
 17   READ(5,2) (X(I),I=1,3)
      DELT=TINC
 21   FORMAT(//,1X,48HNOTE, CONTROLS IN DEGREES AND ANGLES IN RADIANS.
     Z//,1X,30HSPOILERS BIASED UP 20 DEGREES.,//)
      WRITE(6,21)
 20   FORMAT(4X,4HTIME,8X,7HN ACCEL,9X,5HTHETA,5X,9HTHETA DOT,5X,
     Z8HELEVATOR,4X,8HSPOILERS,4X,8HAILERONS,//)
      WRITE(6,20)
      TS=0.
      U(2,1)=20.
 19   FORMAT(7F12.4)
      WRITE(6,19) TS,(X(I),I=1,3),(U(J,1),J=1,M)
      TF=.5*TF
 18   CALL RUNGE(3,3,X,TS,DELT)
```

```
      DO 41  I=1,N
41    XX(I,1)=X(I)
      CALL GMPRD(CONT,XX,U,M,N,1)
      XX(1,1)=100.*XX(1,1)
      U(1,1)=57.3*U(1,1)
      U(2,1)=57.3*U(2,1)+20.
      U(3,1)=57.3*U(3,1)
      WRITE(6,19) TS,(XX(I,1),I=1,N),(U(J,1),J=1,M)
      IF (TS.LT.TF)  GO TO 18
      TF=2.*TF
      KTR=KTR + 1
      IF (KTR.EQ.11)  GO TO 93
      GO TO 45
93    CONTINUE
      STOP
45    READ(5,2) ((Q(I,J),J=1,N),I=1,N),((R(I,J),J=1,M),I=1,M)
      GO TO 46
52    WRITE(6,53)
53    FORMAT(/,1X,47HQ MATRIX NOT POSITIVE SEMIDEFINITE AS REQUIRED.)
      GO TO 50
54    FORMAT(/,1X,43HR MATRIX NOT POSITIVE DEFINITE AS REQUIRED.)
55    WRITE (6,54)
      GO TO 50
      END
```

Standard Subroutines

Except for required dimensioning changes, identical subroutines were used for all the optimization programs. They are:

Sub 1 FCN (KP,J2,X,F). If J2 = 1, derivitives of the matrix Riccati equation are computed for use in Runge-Kutta integration (sub 2). Note the simplification due to C matrix symmetry. If J2 = 2, the C matrix is filled for the "C matrix constancy check" in the main program. If J2 = 3, derivitives of the optimal system equations are computed for use in sub 2.

Sub 2 RUNGE (N,J2,Y,X,DX). This is a variable step size Runge-Kutta digital equation solver provided by the Digital Computation Center (Bldg. 57).

Sub 3 MXPM (A,B,C,M,N,NSGN). Used for addition and subtraction of matrices.

Sub 4 GMTRA (A,B,M,N). Matrix transposition.

Sub 5 GMPRD (A,B,C,NA,NB,NC). Matrix multiplication.

Sub 6 MTXINV (A,B,N). This matrix inversion routine was also supplied by the Digital Computation Center.

Sub 7 CHRPOL (A,B,N). The characteristic polynomial of a real matrix is computed by the Bocher formula (Ref 4:234).

Sub 8 ROOTS (IDIOT,A,D). This subroutine was adapted from an existing AFIT-AID program. It is used after CHRPOL to find the roots of the characteristic polynomials. These roots are the eigenvalues of the input matrix.

All subroutines preserve their input data. Appendix B is a complete listing of all subroutines.

Manipulation of Q and R Matrices

Figure 2-3 expands the integral portion of Kalman's quadratic cost function. A study of these expansions will go far in explaining manipulation of the Q and R matrices to achieve the desired results. Optimization program D states are used for illustration. Two points should be kept in mind throughout the following discussion.

1. Every controller produced by the digital computer programs is optimal for the costs assigned in the Q and R matrices. Manipulation of the cost matrices is only required to find the costs that translate into good flying qualities. Final evaluation of flying qualities must be done on the (two degree of freedom) analog simulation.

2. Since there are no constraints on controls in the optimization programs, the controllers are not necessarily practical. Integration of the optimal system equations with an initial θ of 0.1 radian helps establish the practicality of the controllers.

Clearly, Q_{11} weights the cost of non-zero normal acceleration, Q_{22} weights pitch angle, and Q_{33} weights pitch rate. For the controls, R_{11} weights the cost of elevator usage, R_{22} weights spoiler cost, and R_{33} weights aileron usage. These main diagonal elements of the Q and R matrices are the most important and are often the only ones used. The sample optimization program outputs in

$$\underline{X}^T Q \underline{X} = \begin{bmatrix} N_L & \theta & \dot{\theta} \end{bmatrix} \begin{bmatrix} Q_{11} & Q_{12} & Q_{13} \\ Q_{21} & Q_{22} & Q_{23} \\ Q_{31} & Q_{32} & Q_{33} \end{bmatrix} \begin{bmatrix} N_L \\ \theta \\ \dot{\theta} \end{bmatrix}$$

$$= \begin{bmatrix} N_L & \theta & \dot{\theta} \end{bmatrix} \begin{bmatrix} Q_{11} N_L & + Q_{12}\theta & + Q_{13}\dot{\theta} \\ Q_{21} N_L & + Q_{22}\theta & + Q_{23}\dot{\theta} \\ Q_{31} N_L & + Q_{32}\theta & + Q_{33}\dot{\theta} \end{bmatrix}$$

$$\underline{X}^T Q \underline{X} \quad \begin{matrix} Q_{11} N_L^2 & + Q_{12} N_L \theta & + Q_{13} N_L \dot{\theta} \\ Q_{21} N_L \theta & + Q_{22} \theta^2 & + Q_{23} \theta\dot{\theta} \\ Q_{31} N_L \dot{\theta} & + Q_{32} \theta\dot{\theta} & + Q_{33} \dot{\theta}^2 \end{matrix}$$

Similarly:

$$\underline{U}^T R \underline{U} \quad \begin{matrix} R_{11}\, \delta_e^2 & R_{12}\, \delta_e \delta_{SP} & R_{13}\, \delta_e \delta_{AIL} \\ R_{21}\, \delta_e \delta_{SP} & R_{22}\, \delta_{SP}^2 & R_{23}\, \delta_{SP} \delta_{AIL} \\ R_{31}\, \delta_e \delta_{AIL} & R_{32}\, \delta_{SP} \delta_{AIL} & R_{33}\, \delta_{AIL}^2 \end{matrix}$$

where

$$U_1 = \delta_e \qquad U_2 = \delta_{SP} \qquad U_3 = \delta_{AIL}$$

Fig. 2-3. Expansion of Quadratic Cost Function

Appendix A show Q and R matrices which produced controllers with good flight characteristics.

If the Q's are increased and/or the R's are decreased, the controllers become "stronger." That is, pitch angle, pitch rate, and normal acceleration are kept closer to zero. Note in Figure 2-4 below that θ feedback to the ailerons is positive. That is, if θ is positive, aileron deflection is positive (T.E. down), producing negative pitching moment which tends to drive θ toward zero. However, in earlier runs with $Q_{22} = Q_{23} = 10$, θ feedback to the ailerons was actually negative, and θ and $\dot{\theta}$ feedbacks to the other controls were much smaller. Note also the sign reversal of N_L feedback to spoilers between the 4X (weaker) and 6X (stronger) controllers. Compared to their 4X counterparts, the 6X controllers reduce N_L about five percent when responding to the 0.1 radian θ initial conditions. Performance in driving θ to zero was not measurably affected.

Another interesting point is that as the controllers become stronger they become more similar for the three LAMS flight conditions. Figure 2-5 below shows the 4X series controllers (X indicates flight condition) which were used for most of the testing with the six degree of freedom simulation at the Flight Dynamics Laboratory. These controllers are relatively weak in normal acceleration control (note magnitude of Q_{11} VS Q_{22} and Q_{33}), and N_L feedback varies appreciably in the three flight conditions. However,

Q MATRIX		
1	0	0
0	40	0
0	0	20

CONTROLLER 61* FEEDBACKS

to \ of	N_L	θ	$\dot{\theta}$
δ_e	-.7777	30.35	21.23
δ_{sp}	-.0892	-8.464	-6.200
δ_{ail}	.1791	1.670	1.368

R MATRIX		
.04	0	0
0	.04	0
0	0	.04

CONTROLLER 63* FEEDBACKS

to \ of	N_L	θ	$\dot{\theta}$
δ_e	-.7904	30.51	21.60
δ_{sp}	-.2155	-7.689	-5.884
δ_{ail}	.2074	2.058	1.734

Fig. 2-4. 6X Series Controllers

* Last digit indicates flight condition

they are strong θ and $\dot{\theta}$ controllers, and these feedbacks are essentially the same for all three flight conditions. Since the three LAMS flight conditions vary appreciably in speed, altitude, gross weight (see page 74), there is a possibility that a single fixed gain direct lift controller could be used in the entire B-52 flight envelope.

Somewhat surprising is the fact that zero cost for a state, N_L for instance, does <u>not</u> mean that N_L feedback will be zero. Although the actual figures are not shown, the N_L feedback in this case helps to control $\dot{\theta}$.

<u>Aileron/Elevator Controllers</u>. By increasing R_{22} from .04 to ten, the spoiler feedbacks were reduced essen-

Q MATRIX

0.5	0	0
0	40	0
0	0	20

CONTROLLER 41 FEEDBACKS

to \ of	N_L	θ	$\dot{\theta}$
δ_e	-.9859	30.29	21.24
δ_{sp}	.1192	-8.648	-6.213
δ_{ail}	.0586	1.811	1.379

R MATRIX

.04	0	0
0	.04	0
0	0	.04

CONTROLLER 42 FEEDBACKS

to \ of	N_L	θ	$\dot{\theta}$
δ_e	-1.0055	30.35	21.53
δ_{sp}	.0804	-8.384	-6.118
δ_{ail}	.1056	1.783	1.407

CONTROLLER 43 FEEDBACKS

to \ of	N_L	θ	$\dot{\theta}$
δ_e	-1.0200	30.44	21.61
δ_{sp}	.0461	-7.972	-5.896
δ_{ail}	.0638	2.243	1.742

Fig. 2-5 4X Series Controllers

tially to zero as shown in Fig. 2-6 below. A three-control system with the same Q matrix is shown in Fig. 2-7, page 48. These aileron/elevator controllers provided a measure of DLC benefits, and would be suitable for long-term usage. Due to the increased drag, a controller that uses spoilers biased up 20° can only be used for short periods such as during refueling and landing approach. These aileron/elevator controllers were not pursued extensively due to the limited simulator availability. In addition, the B-52 ailerons are relatively ineffective due to their small size (see D-15 and D-16, Fig. 2-2, page 21). In fact, the fleet G and H model B-52's have no ailerons at all. However, the design method for aileron/elevator controllers has been demonstrated, and this scheme could be very useful on a different aircraft, possibly for gust alleviation during cruise.

Q MATRIX

2	0	0
0	40	0
0	0	20

R MATRIX

.04	0	0
0	10	0
0	0	.04

CONTROLLER FEEDBACKS

to \ of	N_L	θ	$\dot{\theta}$
δ_e	-.2853	31.49	22.08
δ_{sp}	-.0021	-.0337	-.0257
δ_{ail}	.4206	1.472	1.4112

(Flight Condition 1)

Fig. 2-6 Aileron/Elevator Controller

Off-Diagonal Elements. The following discusses use of non-zero off-diagonal elements in the Q and R matrices. All the controllers discussed are variations of controller 61 as shown in Fig. 2-4, page 45 and as included in Appendix

Q MATRIX

$$\begin{matrix} 2 & 0 & 0 \\ 0 & 40 & 0 \\ 0 & 0 & 20 \end{matrix}$$

R MATRIX

$$\begin{matrix} .04 & 0 & 0 \\ 0 & .04 & 0 \\ 0 & 0 & .04 \end{matrix}$$

CONTROLLER 71 FEEDBACKS

to \ of	N_L	θ	$\dot{\theta}$
δ_e	-.3684	30.44	21.21
δ_{sp}	-.4971	-8.107	-6.165
δ_{ail}	.4145	1.398	1.348

Fig. 2-7. Controller 71 Feedbacks

A. All comparisons are to controller 61. They are labeled Y61 for identification, as all incorporate flight condition one data. Computer optimization program D was used exclusively. In each case, all off-diagonal elements except those mentioned are zero.

Controller 161 was designed with $R_{23} = -0.04$. From Fig. 2-3, page 43, this element represents spoilers and ailerons working together. The negative sign of R_{23} means that use of spoilers and ailerons in the positive (or negative) direction simultaneously is encouraged by a negative cost. Compared to controller 61, the elevator and aileron feedbacks (opposite in sign to the aileron feedbacks) are reduced by 20%, and the N_L feedback to the spoilers is reversed in sign, making it the same sign as aileron N_L feedback. In the response to 0.1 radian initial θ, normal acceleration of controller 161 is slightly larger than that of 61. This is logical since the spoiler N_L feedback is more affected by R_{23} cost than by the N_L cost.

Controller 261 was designed with R_{32} = .02. This element also represents spoilers and ailerons working together; the positive value encourages deflections of opposite sign. Due to the sign conventions used (page 12), positive spoiler movement (TE up) and negative aileron movement (TE up) have similar aerodynamic effects. Compared to controller 61, 261's elevator and spoiler feedbacks are unchanged, but the aileron θ and $\dot{\theta}$ feedbacks are tripled, and aileron N_L feedback is increased 22%.

Controller 361 was designed with R_{12} = 0.04, which encourages elevator and spoiler movements of opposite sign. (Normal movements are opposite in sign to cancel pitching moments.) Controllers 61 and 361 are compared below. The response to 0.1 radian of θ is essentially unchanged.

CONTROLLER 61/361 FEEDBACKS

to \ of	N_L	θ	$\dot{\theta}$
δ_e	-.7777/-.4252	30.35/34.59	21.23/24.39
δ_{sp}	-.0892/-.1470	-8.464/-7.511	-6.200/-5.512
δ_{ail}	.1791/ .1907	1.670/ 1.461	1.368/1.218

Fig. 2-8 Controller 61/361 Comparison

Controller 461 was designed with R_{21} = -0.02, <u>discouraging</u> the normal elevator and spoiler movements of opposite sign. Controllers 61 and 461 are compared below. The response of 461 to the 0.1 radian initial θ is slightly degraded, and flying qualities would also be poorer.

CONTROLLER 61/461 FEEDBACKS

to \ of	N_L	θ	$\dot{\theta}$
δ_e	-.7777/-.7847	30.35/32.51	21.23/22.80
δ_{sp}	-.0892/-.4792	-8.464/7.148	-6.200/4.738
δ_{ail}		1.670/1.818	1.368/1.475

Fig. 2-9. Controller 61/461 Comparison

The effect of R_{31} and R_{13} (elevator and ailerons used together) would be similar to the effects of R_{12} and R_{21} on elevator and spoiler feedbacks. This concludes the discussion of the R matrix.

The first Q matrix demonstration controller, 561, was designed with $Q_{23} = 40$. This element weights the cost of θ multiplied by $\dot{\theta}$. If θ is positive and a negative $\dot{\theta}$ is commanded (the normal situation), the cost is negative. This controller is identical to controller 61. If $Q_{23} = Q_{32} = 40$, the controller is still unchanged. However, only one second of real time backward integration is required before the C matrix becomes constant. All other runs of program D required 2.5 seconds of backward integration. Thus, a small saving of computer time is possible. This interesting effect is also illustrated in the program A sample output in Appendix A.

Controller 661 was designed with $Q_{31} = -5$. This element weights N_L multiplied by $\dot{\theta}$ and encourages these states to be of like sign. Since $N_L = U_o\dot{\theta} - \alpha$, a negative Q_{31} does no violence to the controller's normal function, and 661 is identical to 61. On the other hand, when Q_{13}, the other $N_L\dot{\theta}$ weight, is made plus ten, the controller is radically changed and becomes useless for direct lift control.

In summary, the use of the off-diagonal elements is straightforward. Some, such as the positive Q_{13}, specify ludicrous requirements. However, if there are legitimate reasons for restricting certain combinations of controller usage or vehicle motions, the proper elements of the Q and R matrices to be weighted are easily determined. Of course, one must maintain the Q matrix positive semi-definite and the R matrix positive definite. All three optimization programs presented in this chapter automatically check the matrices before proceeding with controller design.

Digital Running Time Comparison

In a normal run, an optimization program was used to design a controller for each of the three LAMS flight conditions using the same Q and R matrices. The average IBM 7094 time required was:

Optimization Program A . . . 90 seconds
" " B . . . 235 seconds
" " D . . . 77 seconds

Program B is severely penalized by the somewhat arbitrary requirement of integrating the C matrix backwards for 20 seconds (see page 29). Program A averaged three seconds of real time C matrix integration, while program D required only 2.5 seconds.

<u>Normal Acceleration Reduction</u>

Although it was not one of the original objectives of this study, Chapter VI shows that normal acceleration of the direct lift controlled aircraft was significantly less than that of the basic aircraft. This is true for the climb and descent maneuvers both with and without turbulence, and raises the possibility of increasing aircraft structural life through DLC. When this effect was noted in the 4X controllers, the 6X controllers were designed with Q_{11}, cost of N_L, increased from 0.5 to 1.0. This reduced N_L mean absolute values another percentage point or two, and made it easier for the pilots to maintain a specified rate of climb or descent. Clearly, the possibilities of DLC in this regard have not been exhausted. Additional controllers should be designed with much larger values of Q_{11}. This is easily done, but unfortunately neither analog computer facility was available for testing these controllers.

Potentiometer Settings

The output of the optimization programs (CONTROL = -RIBTC) translates directly into potentiometer settings for the analog simulations. This is illustrated below for controller 41 and the six degree of freedom simulation.

NOTE . . . ACTUAL STATE X(1) IS .01 * N SUB L

$$\begin{bmatrix} \delta_e \\ \delta_{sp} \\ \delta_{ail} \end{bmatrix} = \begin{bmatrix} -.9859 & 30.2938 & 21.2357 \\ .1192 & -8.6476 & -6.2125 \\ .0586 & 1.8110 & 1.3787 \end{bmatrix} \begin{bmatrix} N_L \\ \theta \\ \dot{\theta} \end{bmatrix}$$

Variable Pot Settings

Pot No.	Page No.	Parameter	Gain	Setting
Q47	11	$5\Delta\alpha$ to δ_e	1	0
Q83	11	$.01\ N_L$ to δ_e	1	.9859
Q55	11	θ to δ_e	10 *	.3029
Q52	11	$2\dot{\theta}$ to δ_e	10 *	.1062
Q49	11	$5\Delta\alpha$ to δ_{sp}	1	0
Q63	11	$.01\ N_L$ to δ_{sp}	10	.0119 #
Q54	11	θ to δ_{sp}	10	.8648
Q53	11	$2\dot{\theta}$ to δ_{sp}	10	.3106
Q85	12	$5\Delta\alpha$ to δ_{ail}	1	0
Q80	12	$.01\ N_L$ to δ_{ail}	1	.0586
Q84	12	θ to δ_{ail}	10	.1811
Q86	12	$2\dot{\theta}$ to δ_{ail}	10	.0689

* An additional 10 gain is provided in the program.

\# To A 45. For 6X controllers where FB is negative, connect to A 64.

Other Design Considerations

The controller schematics are shown on pages 64 and 65 for the short period approximation and on pages 90 and 91 for the six degree of freedom simulation. These controllers consist basically of first order servo approximations, $K/(S + K)$ where $K = 1/\text{Time constant}$, fed by stick output and the feedbacks specified by the digital optimization program. The time constants were varied from 0.1 second to as much as six seconds with good results. The only requirement is that the three control surface actuators have reasonably equal time constants. Direct lift control definitely does not depend on the LAMS high speed actuators.

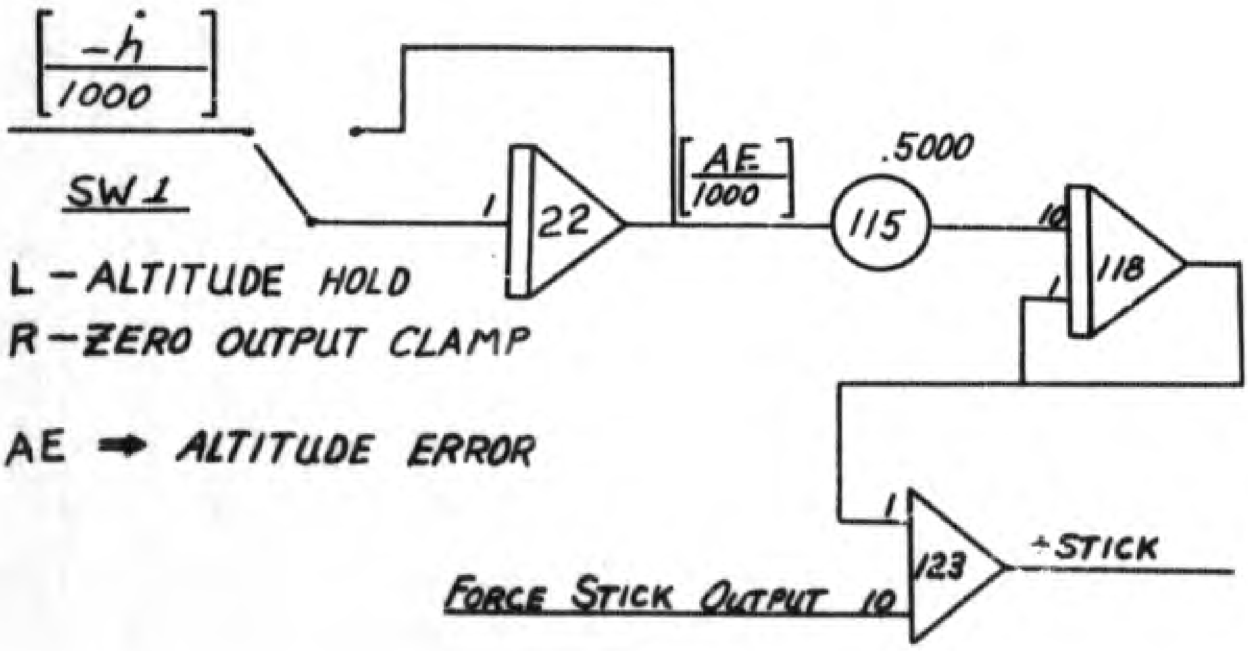
Crossfeeds. The original controller design used crossfeeds between elevators and spoilers, etc., based on a study of the literature concerning conventionally designed direct lift control systems. Experimentation with the two degree of freedom simulation soon showed that these feedbacks degraded system performance. This is logical, since the N_L, θ and $\dot{\theta}$ feedbacks provide an optimal ratio of control surface movements. Any additional items such as crossfeeds only destroy the optimality of the controllers.

Elevator Washout Circuit. The elevator controller shown on pages 64 and 90 incorporates a washout circuit for the θ and $\dot{\theta}$ feedbacks. This circuit was used during the piloted simulation at AFFDL and results were good, as shown in Chapter VI. The purpose of this washout was to assist the pilot in rotating the aircraft if a long climb became necessary. Later tests on the two degree of freedom simulation showed that DLC performance could be improved by eliminating the washout integrator. In other words, the low values of θ shown in the results can be reduced to practically zero. Again, the washout modified the optimal feedback ratios, and degraded performance. Although use of this washout circuit during nearly all of the data runs was unfortunate, it did demonstrate the true optimality of the digitally designed controllers.

Interestingly, the aircraft is also easier to handle on long climbs without the washout circuit. For a climb, the pilot pulls back on the stick, the ailerons go down, the spoilers go down, and the elevator goes up. At a rate of climb of 400-600 feet per minute (depending on flight conditions), the ailerons and spoilers are full down. If the pilot pulls the stick further, only the elevator is able to respond, and the aircraft rotates for a normal climb. When the desired altitude is reached, the pilot releases the stick and the controller feedbacks return the plane to a horizontal attitude. The controller then automatically returns the ailerons and elevator to neutral and the spoilers to their 20° bias position.

Stick-to-Elevator Gain. Pages 90 and 91 of the six degree of freedom simulation show that electrical stick output is fed directly to the spoiler and aileron servos. However, stick output to the elevator servo is fed through a potentiometer set at .3700. This gain can be established by adding the aileron and spoiler pitching moment coefficients and dividing by the elevator pitching moment coefficient (see page 21). Analog experimentation showed this gain was acceptable in all flight conditions. Thus, initial stick movement causes little aircraft pitching. Maximum stick output was adjusted to allow full elevator deflection.

<u>Altitude Hold Mode</u>. An altitude hold circuit was designed which maintains altitude within a few feet, without pitching the aircraft, in the face of relatively severe turbulence. This circuit is only shown on the two degree of freedom schematics since it was designed after completion of the piloted simulation at AFFDL. The complete circuit is reproduced below, showing the ease of adding altitude hold to a DLC system. Results are shown in Fig. 6-19, page 134.



Chapter III

Two Degree of Freedom Analog Program

The purpose of this analog simulation is preliminary testing of controllers designed by the digital optimization programs. Thus, the equations used represent the short period approximation as discussed in Chapter II. The analog equations differ from the digital equations only in the addition of trim terms due to throttle and stabilizer. In addition, the analog program uses a diode function generator to simulate the actual spoiler attenuation constant. (See Fig. 3-1, next page.) Schematics are found on pages 63-68, and Appendix E contains the potentiometer and amplifier assignment sheets.

Equipment Used

Three Electronics Associates Inc. EAI TR-48 transistorized analog computers (Fig. 3-2) located in building 640, AFIT-SE (Math Dept.), were used. Auxiliary equipment included a force stick for "flying" the simulation, a noise generator, strip recorders, an X-Y plotter, and panel meters (Fig. 3-3). The meters displayed all flight parameters normally available to the pilot, plus angle of attack and movements of the three control surfaces. The noise generator, together with the filter on schematic sheet 13 simulated verti-

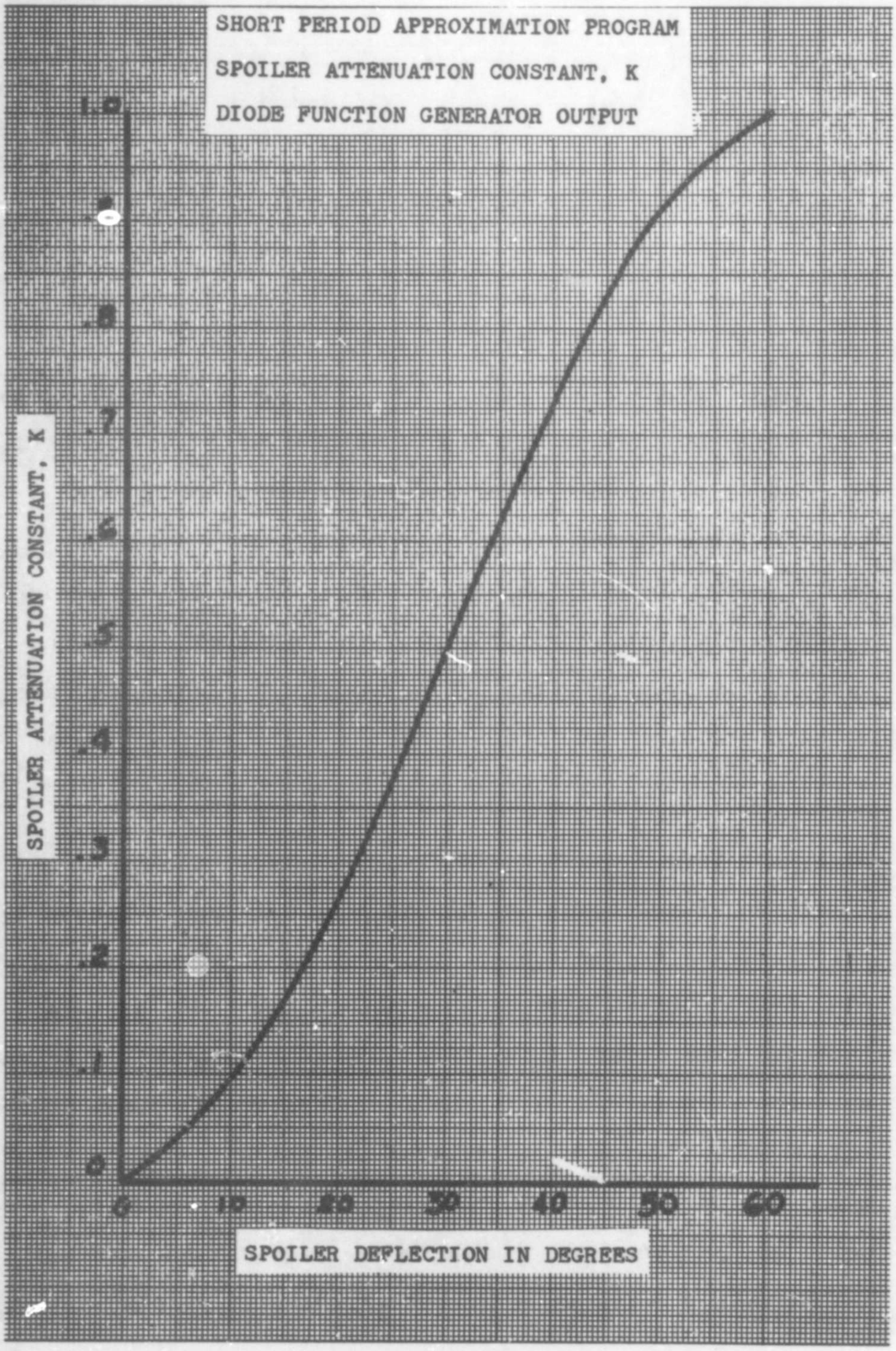


Fig. 3-1. Spoiler Attenuation Constant

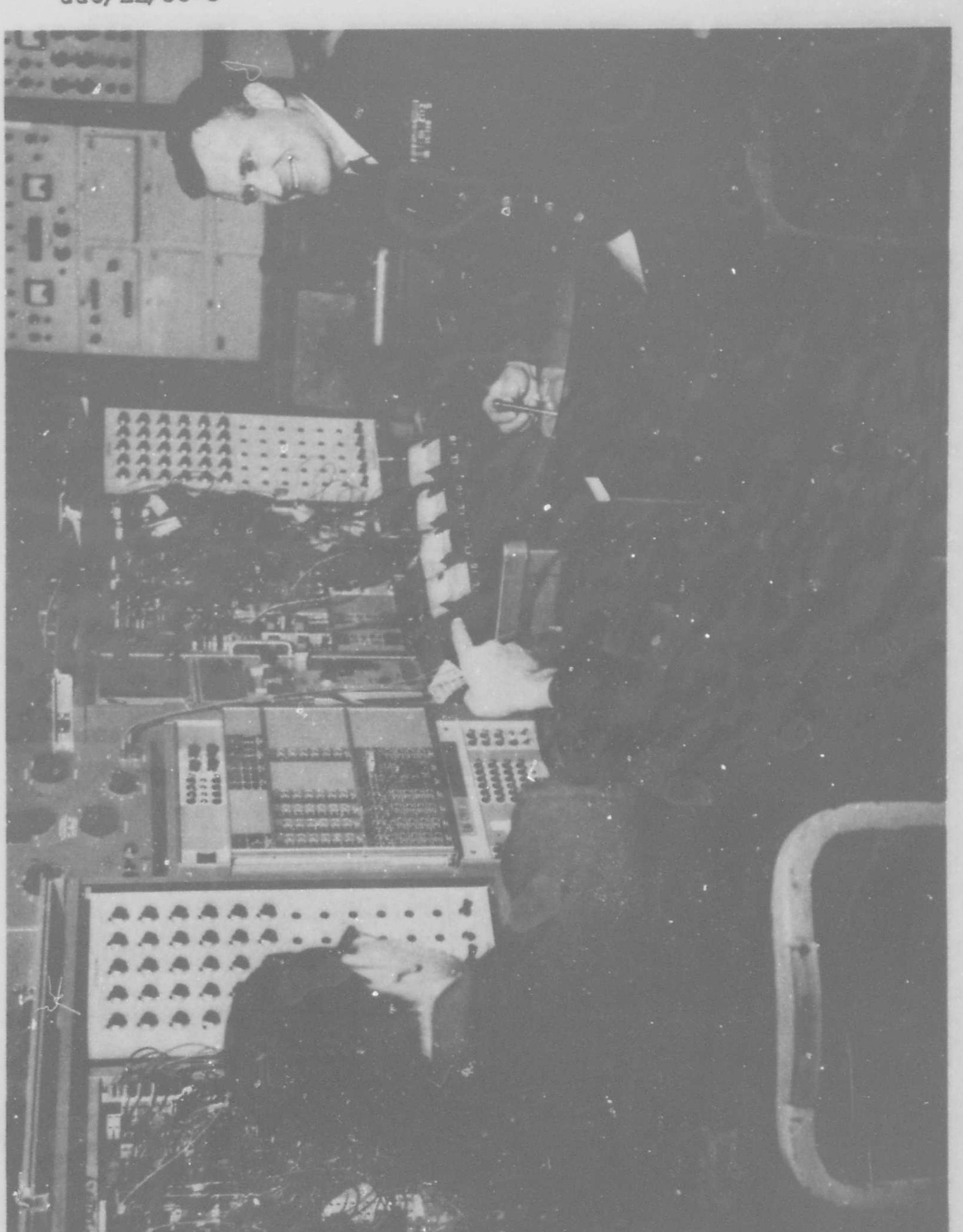


Fig. 3-2. Two Degree of Freedom Simulation

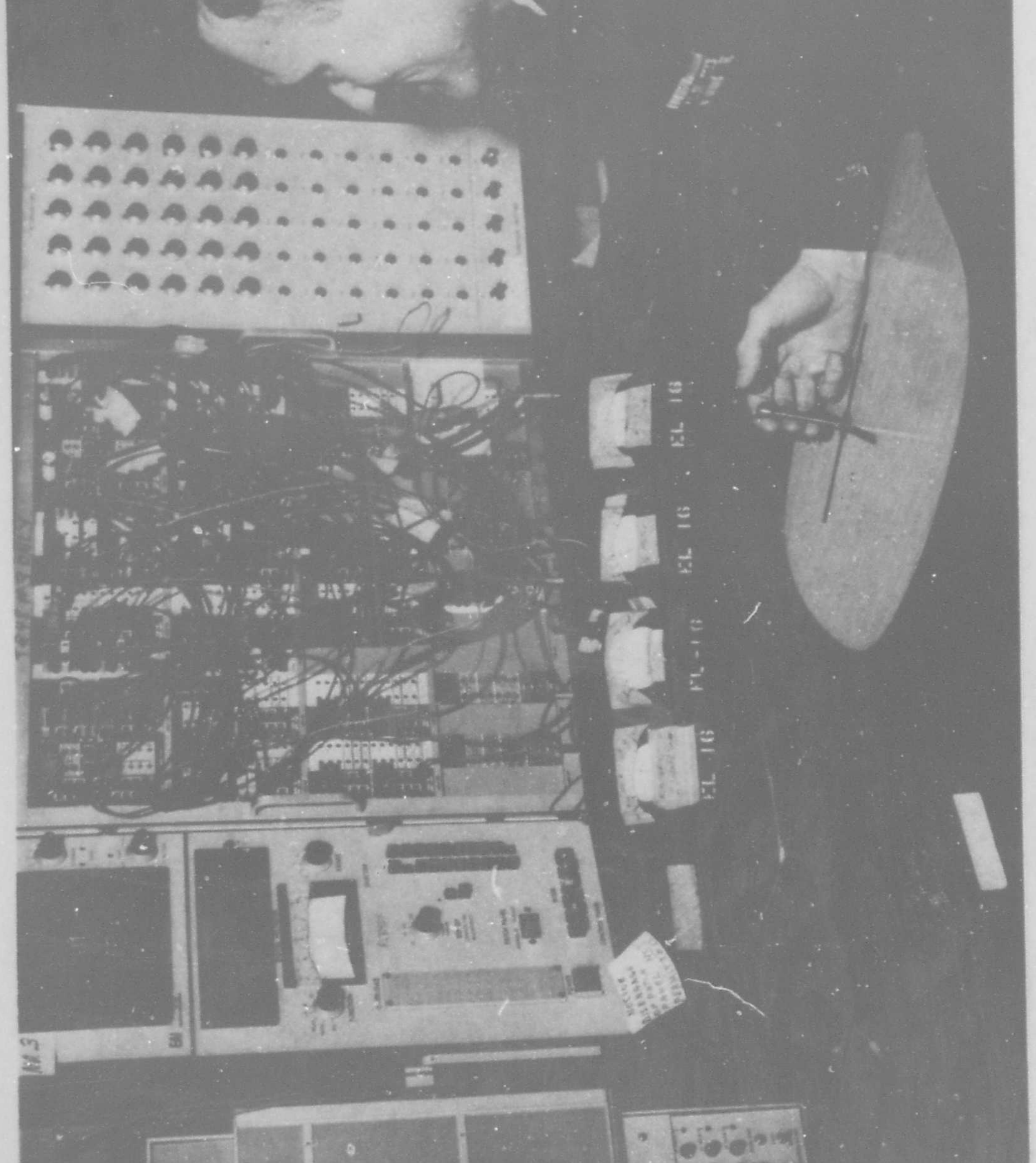


Fig. 3-3. TDF Simulation Force Stick and Meters

cal turbulence. Switches were used for rapid conversion from basic to direct lift controlled aircraft and back again.

Assumptions

The short period approximation equations ignore lateral motion and eliminate the phugoid mode by assuming constant airspeed. Since direct lift control is a longitudinal phenomenon, the first assumption is well-justified. In the six degree of freedom simulation (See Chapters IV and V), the pilots had no difficulty maintaining airspeed within ± ten knots, so the second assumption is also well-justified. Chapter VI, pages 113-121, compares basic aircraft responses (flight condition 1) from the two and six degree of freedom simulations.

Angle of Attack

As explained in Chapter IV, the equations of motion are for perturbations about the three LAMS flight conditions listed on page 74. However, when the spoilers are biased up 20°, an additional angle of attack term must be added to trim the aircraft. The additional angle of attack required in the various flight conditions is:

Flight condition 1 . . . 0.30°
Flight condition 2 . . . 0.36°
Flight condition 3 . . . 0.38°

as verified in the six degree of freedom simulation.

Value of the Simulation

This simulation proved invaluable, especially in view of the limited availability of the Flight Dynamics Laboratory facilities. Digitally designed controllers were tested rapidly, and the results provided a basis for manipulating the Q and R matrices in the quadratic cost function. The controllers eventually chosen (identified as 4X, where X is flight condition) were used without modification for data runs at FDL, and gave good results. Later, the 6X controllers were added to improve gust alleviation. However, the third TR-48 computer, needed for the noise filter and MAV circuits, was not available at AFIT-SE until after completion of the FDL simulation. If the additional machine had been available earlier, this controller improvement would also have been indicated by the two degree of freedom simulation.

In summary, a two degree of freedom simulation can provide a wealth of reliable data on direct lift controllers with a modest amount of equipment. The short period approximation is recommended without hesitation for time and equipment savings in preliminary direct lift controller design and/or in DLC feasability studies for various aircraft types.

Direct Lift Control for LAMS B-52

Simulation of Short Period
Approximation for Preliminary Controller Checkout

<u>at</u> AFIT-SE-MATH DEPT. Building 640

3-EAI-TR-48 Analog Computers (10 V Ref)

Machine 1 - 200 Series Numbers

" 2 - 100 Series Numbers

" 3 - Regular Numbers

U = Constant in each flight condition

$V = P = R = \emptyset = \gamma = 0 \qquad Q = \dot{\theta}$

<u>For Basic Aircraft</u>

SW101-R SW102-C SW103-L SW104-C SW105-L

Disconnect A120 from A111 (Prevent overloads).

Change stick input, A123 to gain 5.

Turn off aileron and spoiler recording.

<u>For Controlled Aircraft</u>

SW101-L SW102-L SW103-R SW104-L SW105-C

Reconnect A120 to A111.

Change stick input, A123 to gain 10.

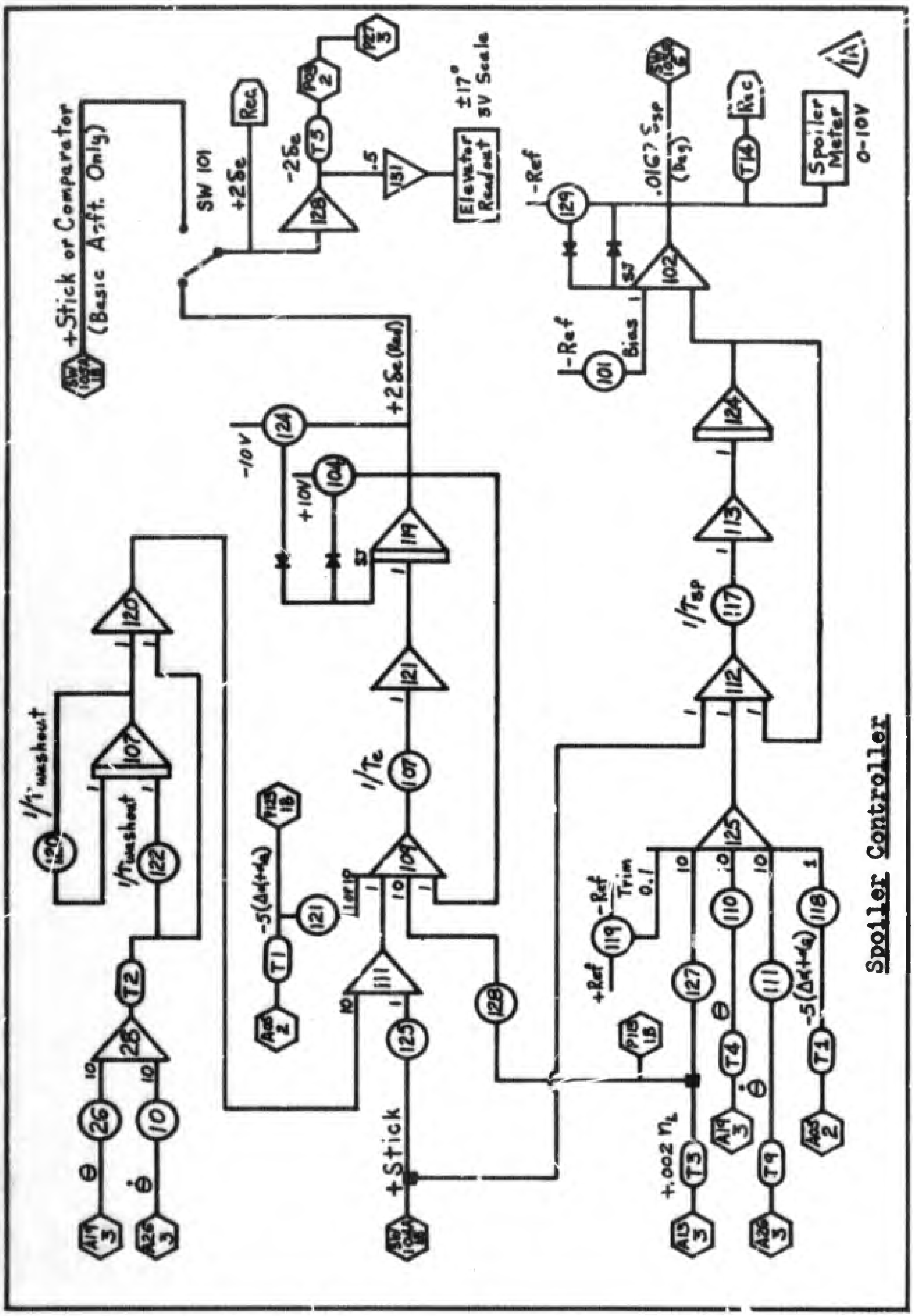
Turn on aileron and spoiler recording.

<u>Symbol Code</u>

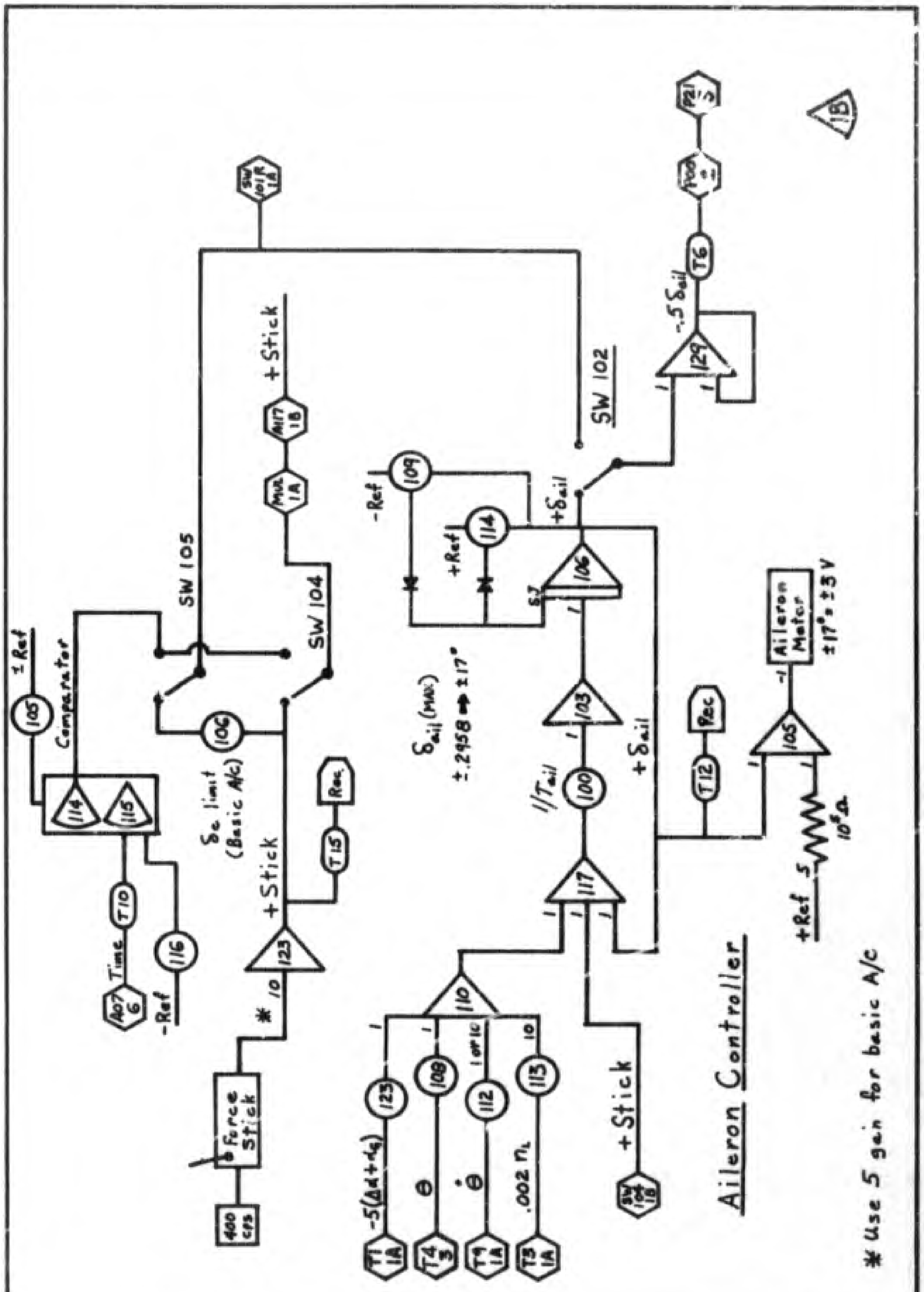
A10 3 — From/to amplifier 10
Sheet number 3

2 — Sheet number

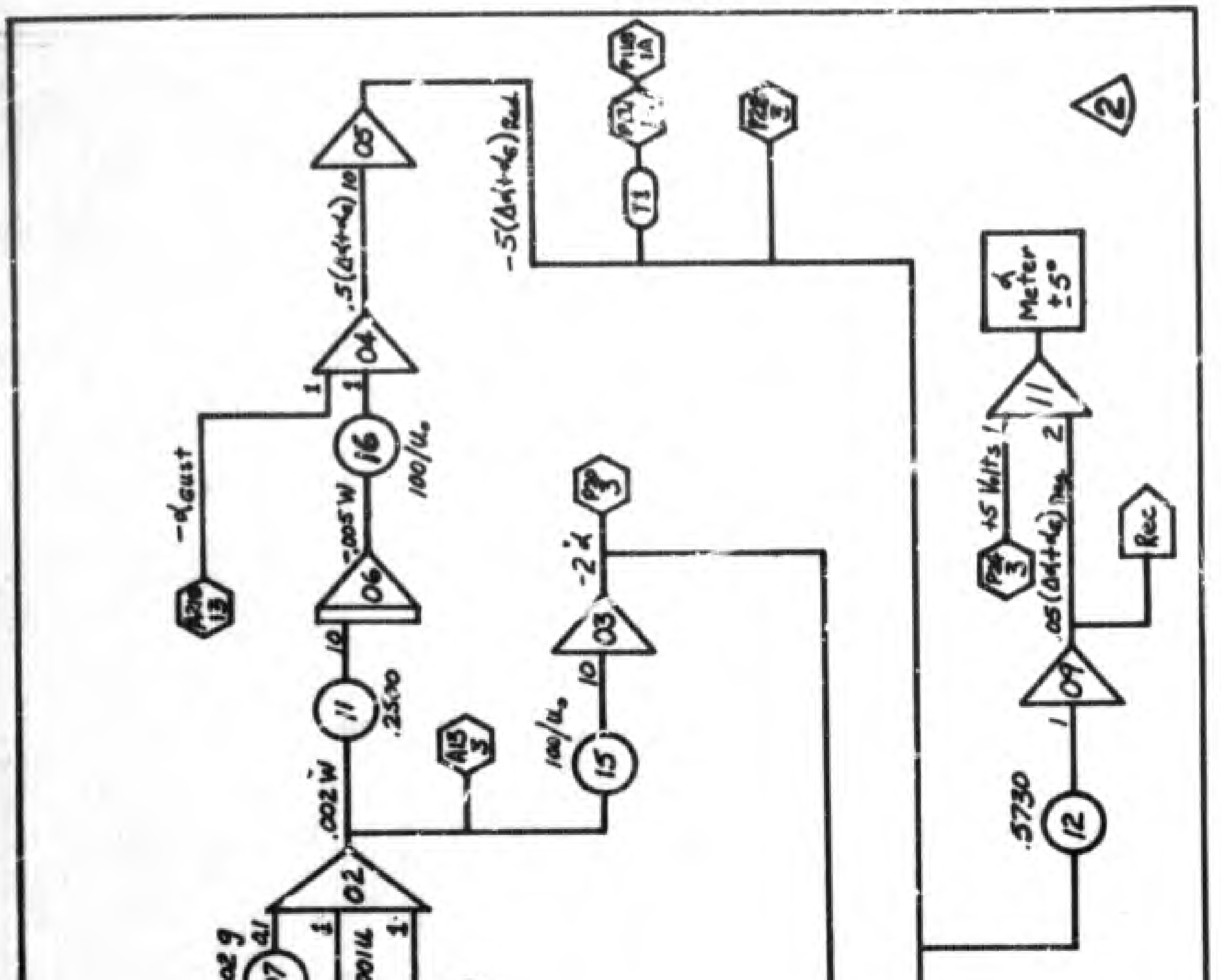
T7 — Trunk number

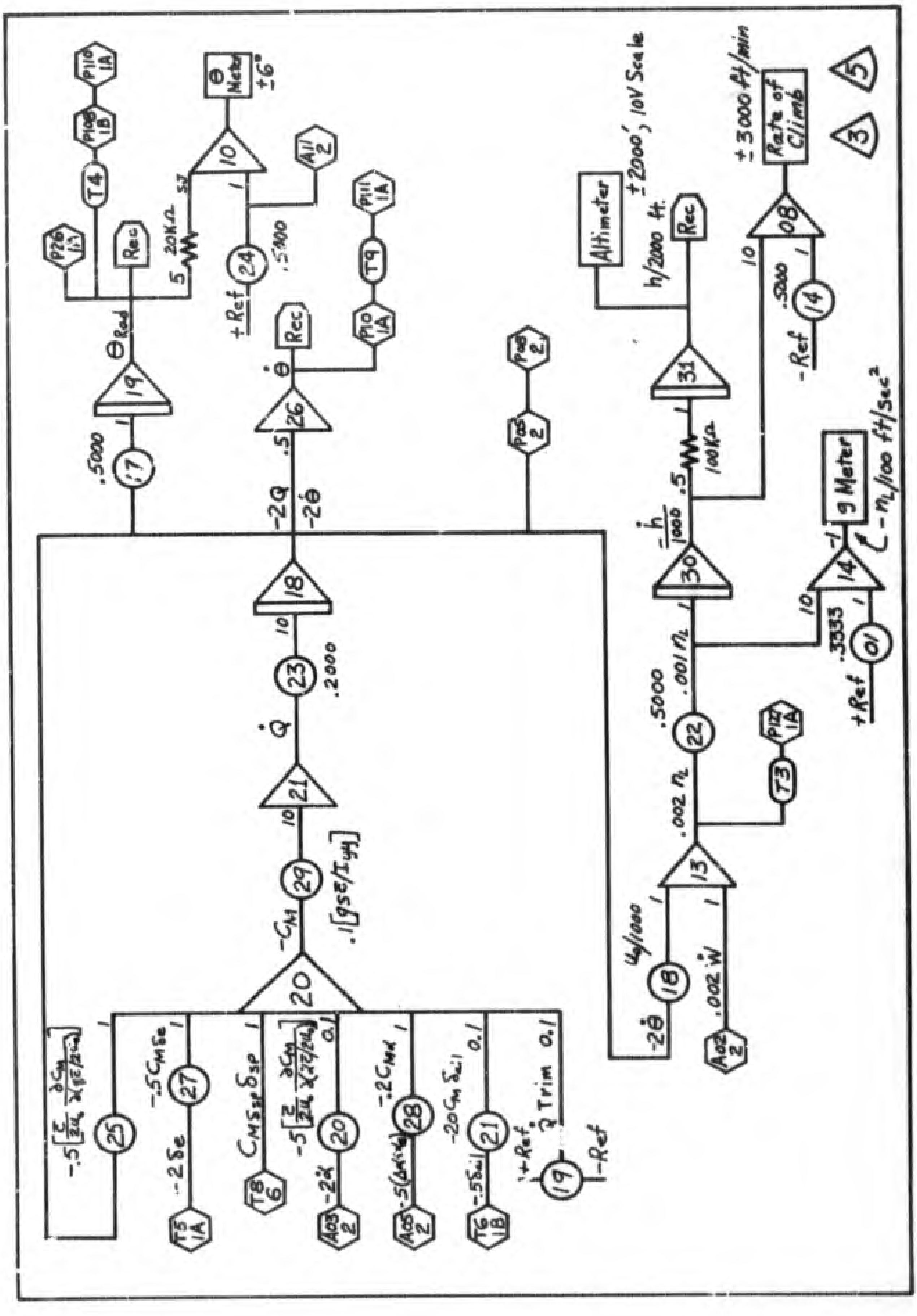


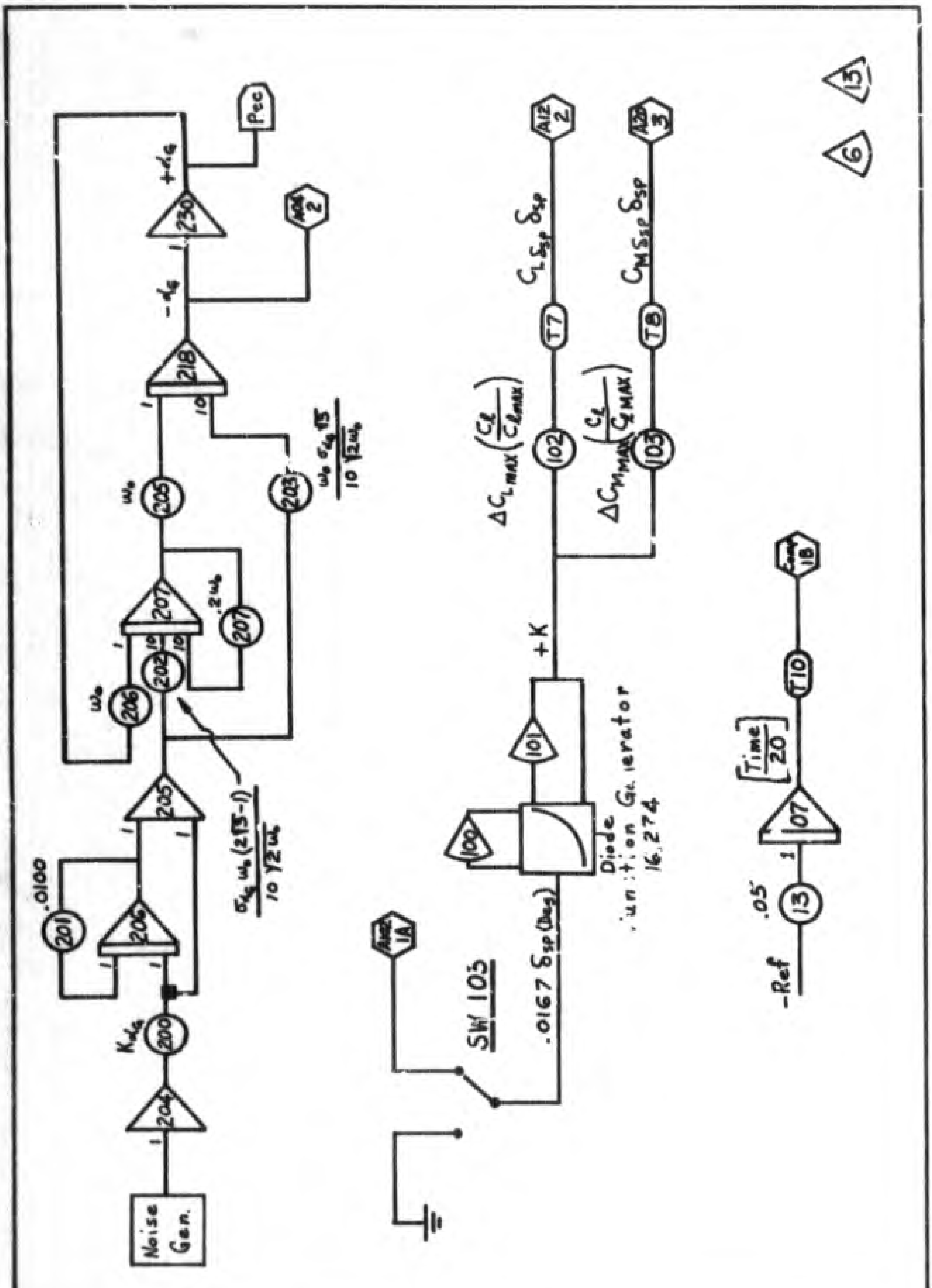
Spoiler Controller



Aileron Controller


Addl. Input Resistors from A00 and A01
Trim
Meter
±5°
Rec
+5 Volts


Altimeter
Rec
Rate of Climb
g Meter
θ Meter
±3000 ft/min
±2000', 10V Scale
h/2000 ft.
-n_L/100 ft/sec^2
±Ref
-Ref
Trim


Noise Gen.
SW 103
Diode Function Generator 16.274
-Ref
.05
+K
.0100
T7
T8
T10

Chapter IV

Six Degree of Freedom Analog Program

The equations of motion (as shown below) and the aerodynamic coefficients for the LAMS B-52E were obtained from The Boeing Company, Wichita Division, Wichita, Kansas. It was from these equations and aerodynamic coefficients that the analog program representing a six degree of freedom simulation was derived and the schematics were drawn.

Airplane Equations of Motion

The following equations of motion were used to simulate airplane dynamics.

X-Equations.

$$\dot{U} = RV - QW - g\theta + \frac{1}{m}\Sigma F_X \tag{4-1}$$

where F_X includes aerodynamic and propulsive forces and can be expressed as

$$\Sigma F_X = q S \Sigma C_X \tag{4-2}$$

where

$$\Sigma C_X = \frac{T}{qS} - C_{D_{CLEAN}} - C_{D_{\delta_{SP}}} \delta_{SP} + \left(C_{L_{CLEAN}} - C_{D_\alpha}\right)\left(\Delta\alpha + \alpha_{GUST}\right) + C_{X_u} \Delta U \tag{4-3}$$

Y-Equations.

$$\dot{V} = PW - RU + g \sin\phi + \frac{1}{m}\Sigma F_Y \qquad (4\text{-}4)$$

where F_Y includes aerodynamic and propulsive forces and can be expressed as

$$\Sigma F_Y = q S \Sigma C_Y \qquad (4\text{-}5)$$

where

$$\Sigma C_Y = C_{Y_\beta}(\beta + \beta_{GUST}) + \frac{185}{2U_o} C_{Y_P} P + C_{Y_{\delta_{SP}}} \delta_{SP} \qquad (4\text{-}6)$$

Z-Equations.

$$\dot{W} = QU - PV + g \cos\phi + \frac{1}{m}\Sigma F_Z \qquad (4\text{-}7)$$

where F_Z includes aerodynamic and propulsive forces and can be expressed as

$$\Sigma F_Z = q S \Sigma C_Z \qquad (4\text{-}8)$$

where

$$\Sigma C_Z = -C_{L_{CLEAN}} - (C_{L_\alpha} + C_D)(\Delta\alpha + \alpha_{GUST}) + C_{Z_U}\Delta U + C_{L_{\delta_{SP}}}\delta_{SP} - C_{L_{\delta_e}}\delta_e - C_{L_{\delta_{STAB}}}\delta_{STAB} - C_{L_{\delta_{AIL}}}\delta_{AIL} - \frac{\bar{c}}{2U_o}\left(C_{L_q} Q + C_{L_{\dot{\alpha}}}\dot{\alpha}\right) \qquad (4\text{-}9)$$

P-Equations.

$$\dot{P} = -QR\frac{(I_{zz} - I_{yy})}{I_{xx}} + \frac{1}{I_{xx}}\Sigma\mathcal{L} \tag{4-10}$$

The cross product of inertia terms and the effect of spinning rotors were assumed to be negligible. $\Sigma\mathcal{L}$ includes aerodynamic and propulsive effects and can be expressed as

$$\Sigma\mathcal{L} = q S b \Sigma C_{\ell} \tag{4-11}$$

where

$$\Sigma C_{\ell} = C_{\ell_{\beta}}(\beta + \beta_{GUST}) + \frac{185}{2U_o}\left(C_{\ell_r} R + C_{\ell_p} P\right) + C_{\ell_{\delta_{AIL}}}\delta_{AIL} + C_{\ell_{\delta_{SP}}}\delta_{SP} \tag{4-12}$$

Q-Equations.

$$\dot{Q} = RP\frac{(I_{zz} - I_{xx})}{I_{yy}} + \frac{1}{I_{yy}}\Sigma\mathcal{M} \tag{4-13}$$

The cross product of inertia terms and the effects of spinning rotors were assumed to be negligible. $\Sigma\mathcal{M}$ includes aerodynamic and propulsive effects and can be expressed as

$$\Sigma\mathcal{M} = q S \bar{c} \Sigma C_M \tag{4-14}$$

where

$$\Sigma C_M = C_{M_o} + C_{M_{STAB}}\Delta_{STAB} + C_{M_{\delta_e}}\delta_e + C_{M_{\delta_{SP}}}\delta_{SP} + C_{M_{\delta_{AIL}}}\delta_{AIL} + \frac{T_{\epsilon_{T_q}}}{qS\bar{c}} + \frac{\bar{c}}{2U_o}\left(C_{M_q}Q + C_{M_{\dot{\alpha}}}\dot{\alpha}\right) + C_{M_{\alpha}}(\Delta\alpha + \alpha_{GUST}) \tag{4-15}$$

R-Equations.

$$\dot{R} = -PQ\,\frac{(I_{YY} - I_{XX})}{I_{ZZ}} + \frac{1}{I_{ZZ}}\,\Sigma\eta \tag{4-16}$$

The cross product of inertia terms and the effect of spinning rotors were assumed to be negligible. $\Sigma\eta$ includes aerodynamic and propulsive effects and can be expressed as

$$\Sigma\eta = qSb\,\Sigma C_N \tag{4-17}$$

where

$$\Sigma C_N = C_{n_\beta}(\beta + \beta_{GUST}) + \frac{185}{2U_0}\left(C_{n_r}R + C_{n_p}P\right) + C_{n_{\delta_{SP}}}\,\delta_{SP} \tag{4-18}$$

Spoiler Equations

The spoilers are actuated by wheel position in the basic aircraft and by wheel position and/or stick position in the direct lift controlled aircraft. The effect of spoiler deflection was included in all six equations of motion as follows:

X-Equation.

$$C_{D_{\delta_{SP}}}\delta_{SP} = \frac{1}{2}\,\frac{\Delta C_D}{K_D}\,\frac{1}{\cos 32.4^\circ}\Big\{K_{D_R}\cos\left[32.4^\circ - (\beta + \beta_G)\right] + K_{D_L}\cos\left[32.4^\circ + (\beta + \beta_G)\right]\Big\} \tag{4-19}$$

Y-Equation.

$$C_{Y_{\delta_{SP}}}\delta_{SP} = \frac{1}{2}\,\frac{\Delta C_D}{K_D}\,\frac{\tan 32.4^\circ}{\cos 32.4^\circ}\Big\{K_{D_L}\cos\left[32.4^\circ + (\beta + \beta_G)\right] - K_{D_R}\cos\left[32.4^\circ - (\beta + \beta_G)\right]\Big\} \tag{4-20}$$

Z-Equation.

$$C_{L_{\delta SP}} \delta_{SP} = \frac{1}{2} \Delta C_{L_{MAX}} \left(\frac{C_l}{C_{l_{MAX}}}\right)\left(K_{D_R} \quad K_{D_L}\right) \tag{4-21}$$

P-Equation.

$$C_{l_{\delta SP}} \delta_{SP} = \left(\beta \quad \beta_G\right)\left(\frac{\partial \Delta C_{l_{MAX}}}{\partial \beta}\right)\left(\frac{C_l}{C_{l_{MAX}}}\right)\left(K_{D_R} \quad K_{D_L}\right) \left(\frac{C_{l_{MAX}}}{F_{CW}}\right) F_{CW} \left(\frac{C_l}{C_{l_{MAX}}}\right)\left(K_{D_R} \quad K_{D_L}\right) \tag{4-22}$$

Q-Equation.

$$C_{M_{\delta SP}} \delta_{SP} = \frac{1}{2} \Delta C_{M_{MAX}} \left(\frac{C_l}{C_{l_{MAX}}}\right)\left(K_{D_R} \quad K_{D_L}\right) \tag{4-23}$$

R-Equation.

$$\begin{aligned} C_{n_{\delta SP}} \delta_{SP} = \frac{1}{2} \frac{TAN\,32.4^\circ}{COS\,32.4^\circ} \frac{\Delta C_D}{K_D} \Big\{ & Cos \left[32.4^\circ - (\beta + \beta_G)\right] \\ & \left[\frac{(X_{RO} K_{D_{RO}} + X_{RI} K_{D_{RI}})}{185} + \left(\frac{59 K_{D_{RO}}}{185} + \frac{48.5\, K_{D_{RI}}}{TAN\ 32.4^\circ}\right)\right] \\ & - Cos\ 32.4^\circ + (\beta + \beta_G) \left[\frac{59 K_{DLO}}{185} + \frac{48.5\, K_{DLI}}{TAN\ 32.4^\circ} \right. \\ & \left. + \frac{(X_{LO} K_{DLO} + X_{LI} K_{DLI})}{185}\right]\Big\} \end{aligned} \tag{4-24}$$

The non-linear spoiler attenuation constant, K, is shown on Fig. 3-1, page 58 as a function of spoiler position.

Quasi-Elastic Perturbation Equations

It should be noted that this is not a rigid body simulation, but rather a "quasi-elastic" simulation. This means that the aerodynamic coefficients have been modified to account for body bending and fin torsion. This was possible because of the hundreds of hours of recorded B-52E flight data available to Boeing for analysis, and their simulation with five degrees of freedom and 65 structural modes. This simulation was used to verify the "corrected" aerodynamic coefficients. A sample derivation of an elastic coefficient was obtained from The Boeing Co. and is found in Appendix C.

The equations of motion with the corresponding constant coefficients are for small perturbations about the same flight conditions used in flight testing the LAMS B-52E. These three conditions provide a variety of airspeeds and altitudes as summarized below.

	ALTITUDE	WEIGHT	SPEED
Flight Condition 1 (Terrain Following)	4,000 ft	350,000 lbs	370 KEAS/.57 Mach 626 ft/sec
Flight Condition 2 (Approach/Landing)	4,000 ft	350,000 lbs	254 KEAS/.39 Mach 429 ft/sec
Flight Condition 3 (Refueling)	32,700 ft	270,000 lbs	448 KEAS/.78 Mach 758 ft/sec

Because of the constant coefficients, the equations of motion are perturbation equations, and it is necessary to remain close to the flight conditions listed above. Changes of plus or minus 2000 feet in altitude and plus or minus 20 knots in airspeed was considered allowable. With these limitations in mind, the tasks listed in Figs. 5-5, 5-6, and 5-7, pages 107, 110, and 111 were selected.

Preliminary Study

It was originally planned to use only the AFFDL (Air Force Flight Dynamics Laboratory) analog computers and moving base cockpit simulator for the program checkout and the pilot-in-the-loop simulation. This was soon found to be infeasible due to a delay in checkout and acceptance by the Laboratory of the new hybrid computer system which in turn caused a backlog of programs to be completed. This also necessitated limiting the simulation to one Electronics Associates Inc. EAI 231R analog computer in order to have the remaining computers available for Government contractors. It was therefore necessary to use the Analog and Hybrid Computation Center located in Building 57, Area B, Wright-Patterson AFB, Ohio, to check out the program and to reduce the number of amplifiers needed to run the simulation. Although the program completed at Bldg. 57 is available for further use (contact Mr. Howard Jones), it is not included here. The aircraft responses obtained were used for general information only and provided data for controller design and insight concerning responses to certain step inputs.

Analog Program

The following deletions were made to further condense the program.

1. <u>Eliminate the Rudder</u>. DLC is primarily concerned with the longitudinal equations while the rudder forces the lateral equations.

2. <u>Eliminate Use of Speed Brakes</u>. The LAMS B-52E is capable of setting the speed brakes (spoilers) in six separate and distinct positions. Spoilers were biased up 20° with all seven panels on each side moving together, rather than inboard and outboard separately.

3. <u>Eliminate the SAS</u> (<u>Stability Augmentation System</u>). The operational B-52's do not have a SAS installed.

4. <u>Eliminate the Autopilot</u>. Autopilot is not required for DLC.

It is with these deletions that the schematics found on pages 80-96 were drawn. With these changes in the analog program completed, a paper static check was made in all three flight conditions. This involved comparing the results obtained from the equations of motion with the derivitives of certain integrators found in the program. The static check values can be found in the appropriate columns of the potentiometer, amplifier, and multiplier assignment sheets in Appendix F, pages F-1 through F-14. Completion of the static check in all three flight conditions proved invaluable later when time in the cockpit simulator was so precious. It also added great confidence to the programmers when changing from one flight condition to another to be able to quickly static check all major amplifier outputs.

At AFFDL, the aircraft responses to step inputs of elevator, ailerons, and spoilers were made in all flight conditions. The results are found on pages 113-121 and in Appendix D. It was during the plotting of these responses with spoilers biased up 20° that a problem of trimming the aircraft developed. After researching the problem, it was determined that with the spoilers biased up 20° the aircraft required an increased angle of attack ($\Delta\alpha$) and therefore the equations of motion must be supplied with the incremental change in α in order to trim the aircraft properly. The actual values of $\Delta\alpha$ were determined for each flight condition and are listed below.

Flight Condition 1 . . . 0.30°
Flight Condition 2 . . . 0.36°
Flight Condition 3 . . . 0.38°

It is important to note that the aircraft responses received from The Boeing Co. and those obtained from this program are nearly identical as seen by comparing pages 113 and 114 for flight condition one, pages D-1 and D-2 for flight condition two, and pages D-7 and D-8 for flight condition three. Although the change in velocity, ΔU, is not found on the Boeing responses, it is included for general information. These responses were instrumental in determining the ratios of elevator to spoiler and elevator to aileron needed to produce zero pitch angle. They will also be used later in comparing the two degrees of freedom simulation and the six degrees of freedom simulation.

Selection of the particular analog console to be used was based solely on the number of amplifiers available and resulted in one basic problem. The console chosen did not have function generators with which to simulate the spoiler attenuation curve as shown on page 58. An alternate function generator (pot padder) was suggested, and although it lacked a little in accuracy, was used. Basically, the pot padder is a servo multiplier cup which has potentiometers set at specific intervals. The value set on each potentiometer is the functional value at that point of the cup. The problems evolved from the fact that it was nearly impossible to create identical functions for the left and right spoilers. In addition, the functions are simply straight lines between potentiometers. The actual curves used are shown on page 79. When a comparison is made with the curve on page 58, the inaccuracies are clearly visible. The small irregularity between the curves caused a slight rolling tendency, but this helped keep the pilots busy and not so preoccupied with only monitoring a constant rate of climb and descent.

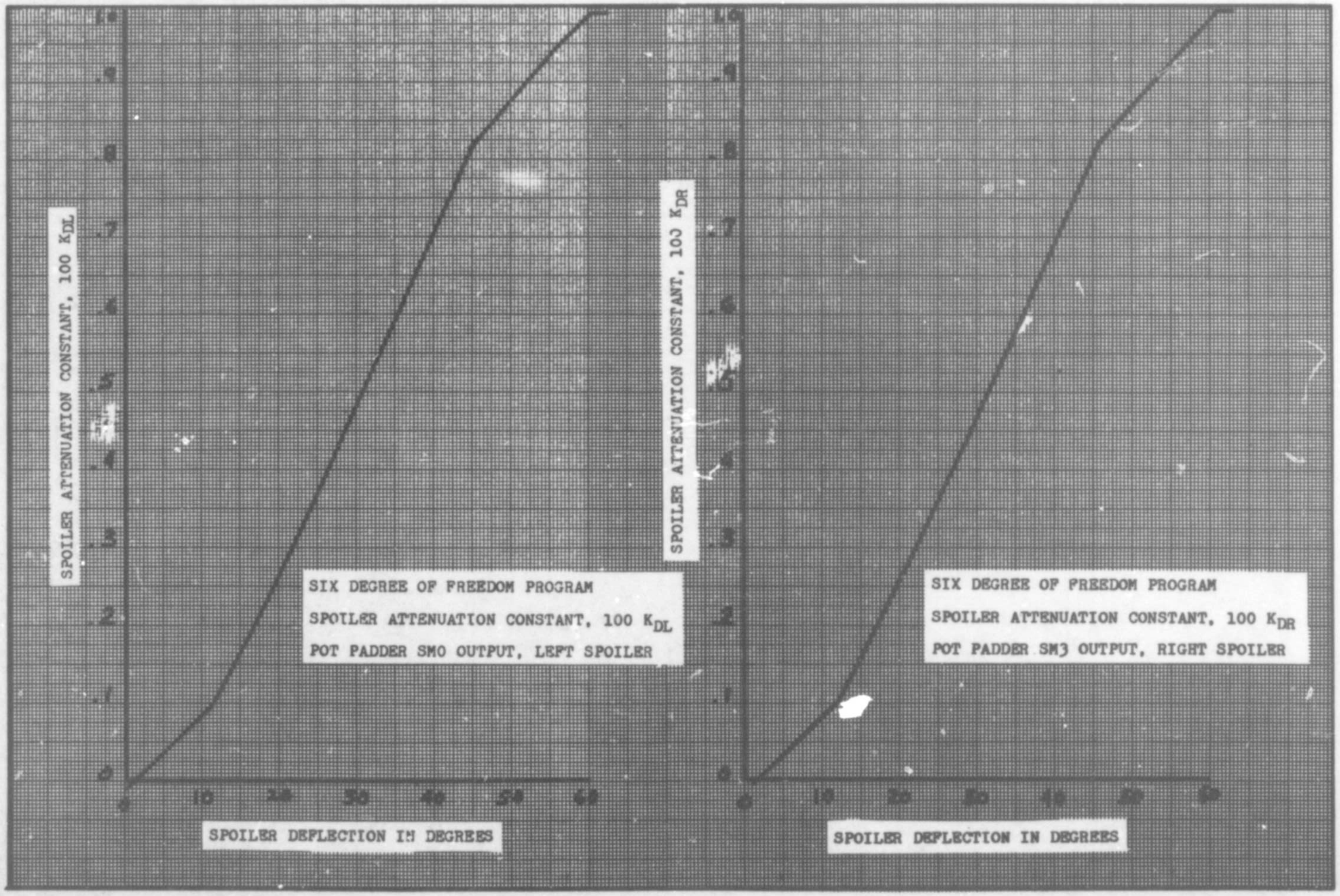
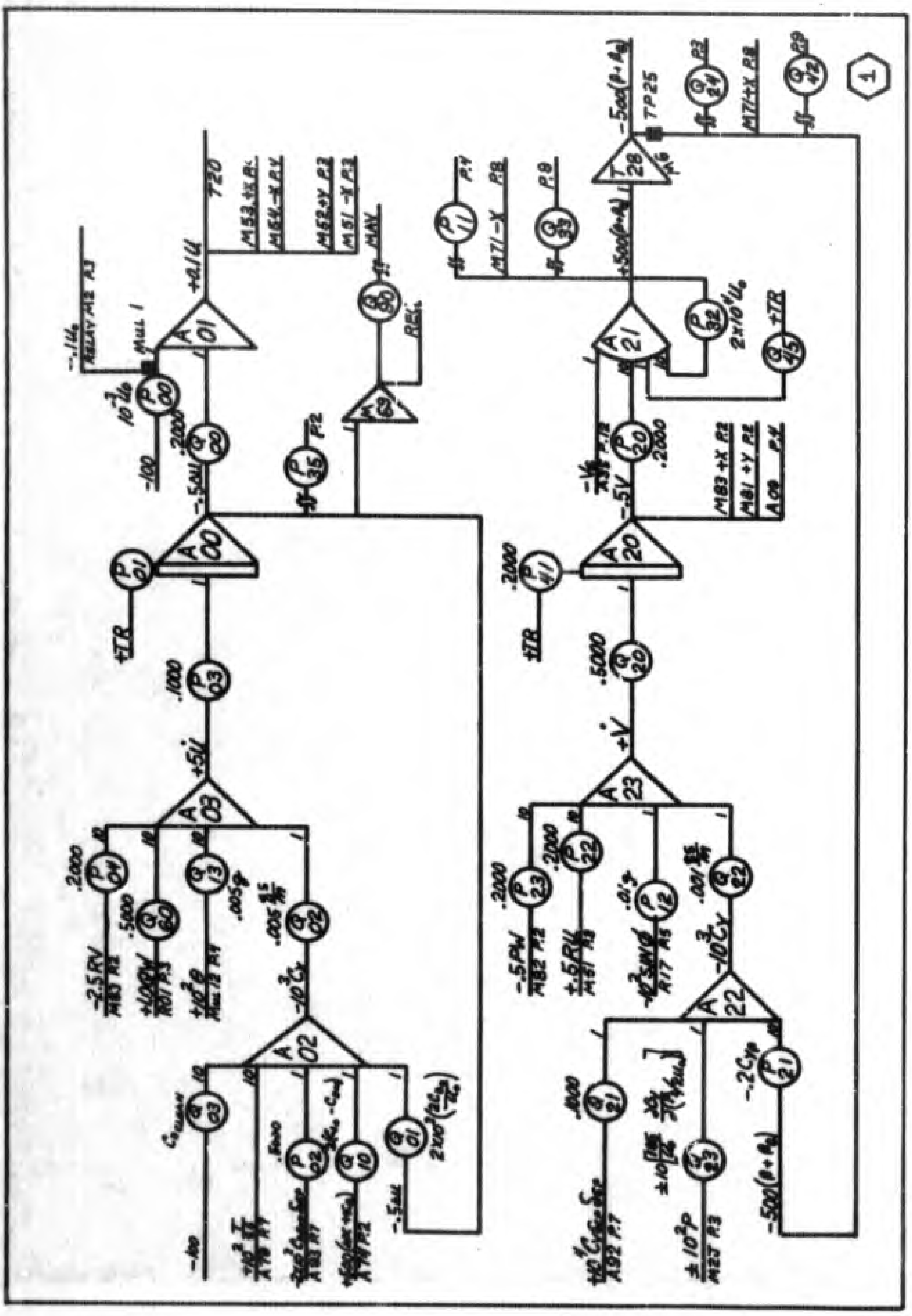
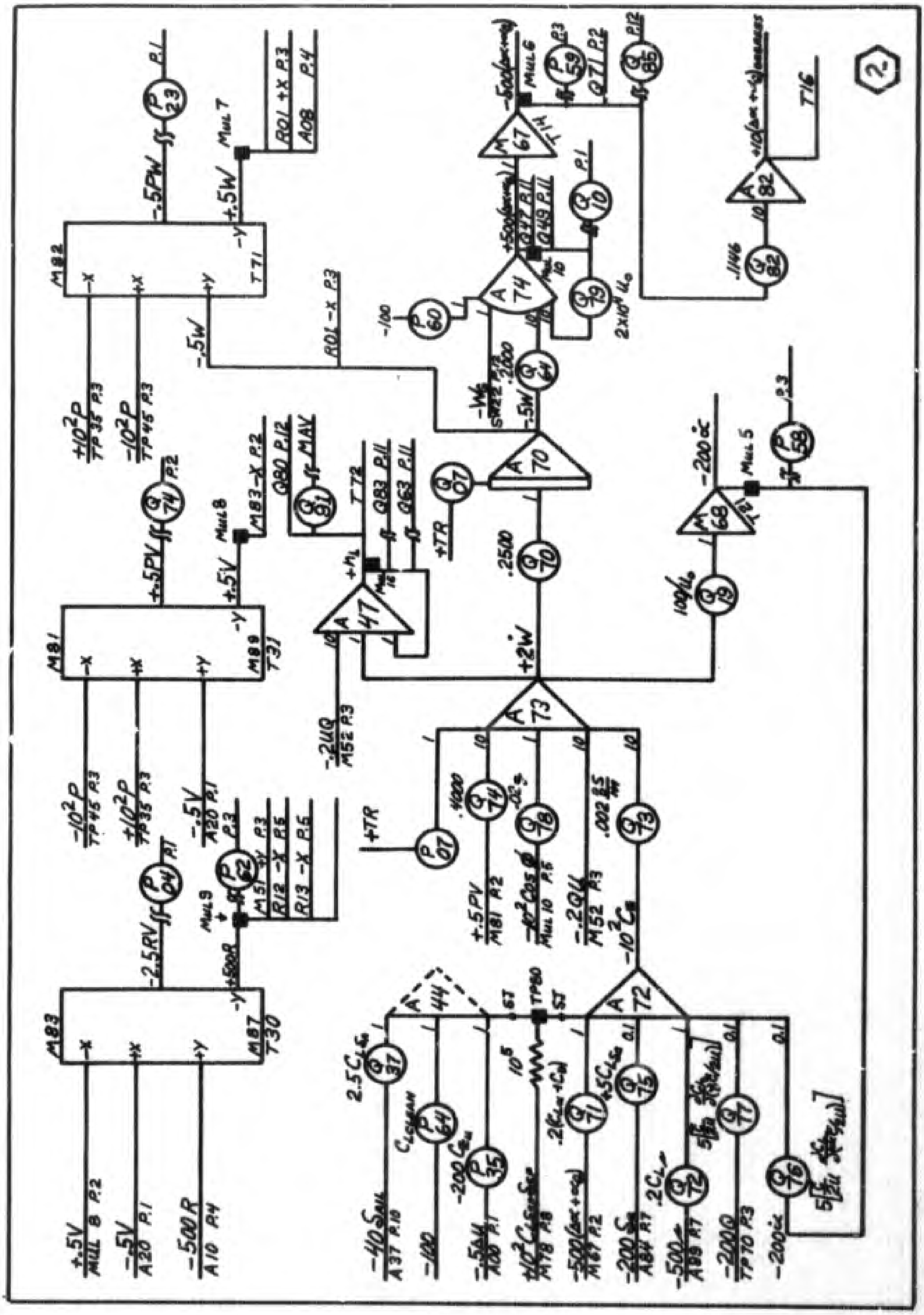
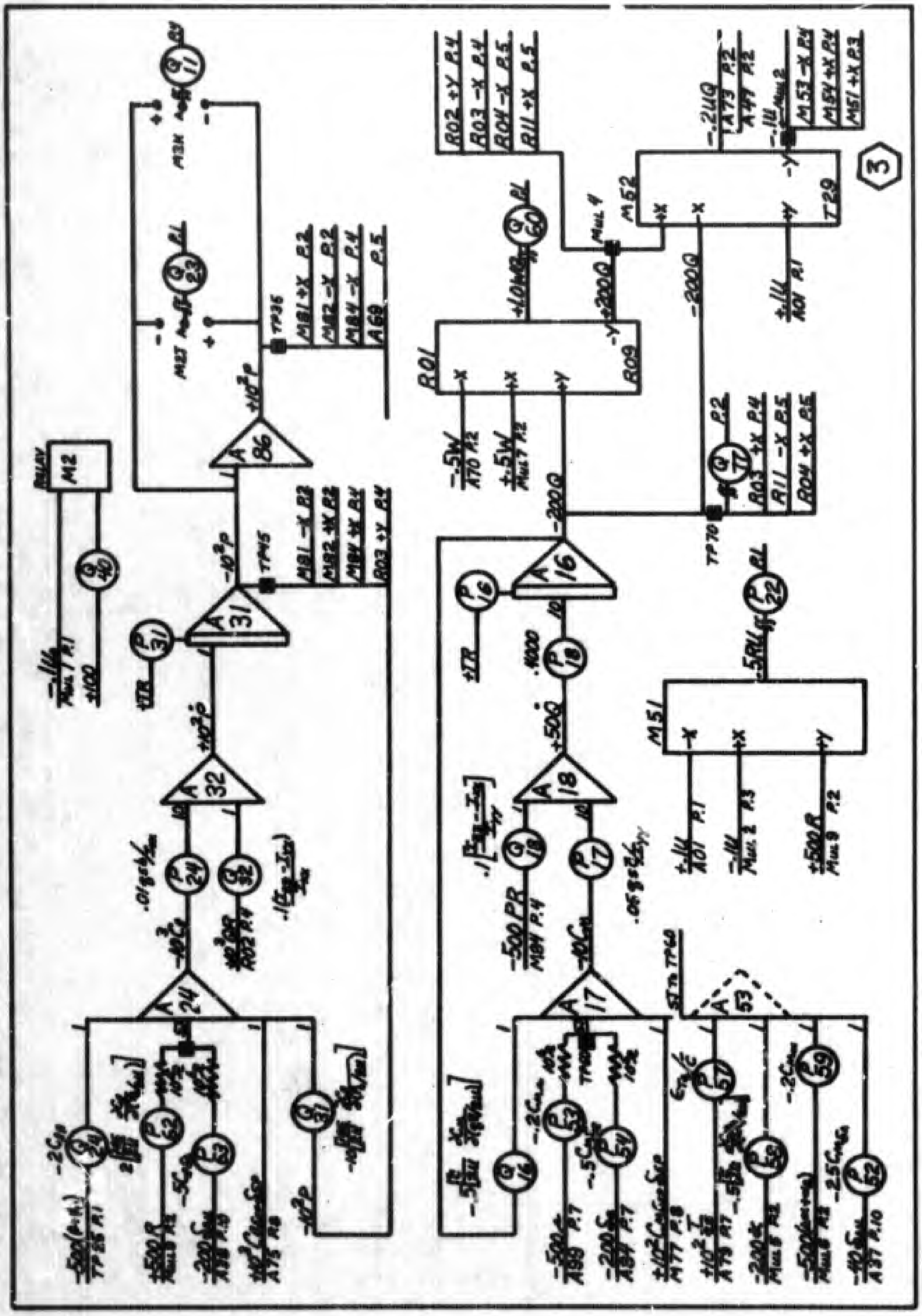
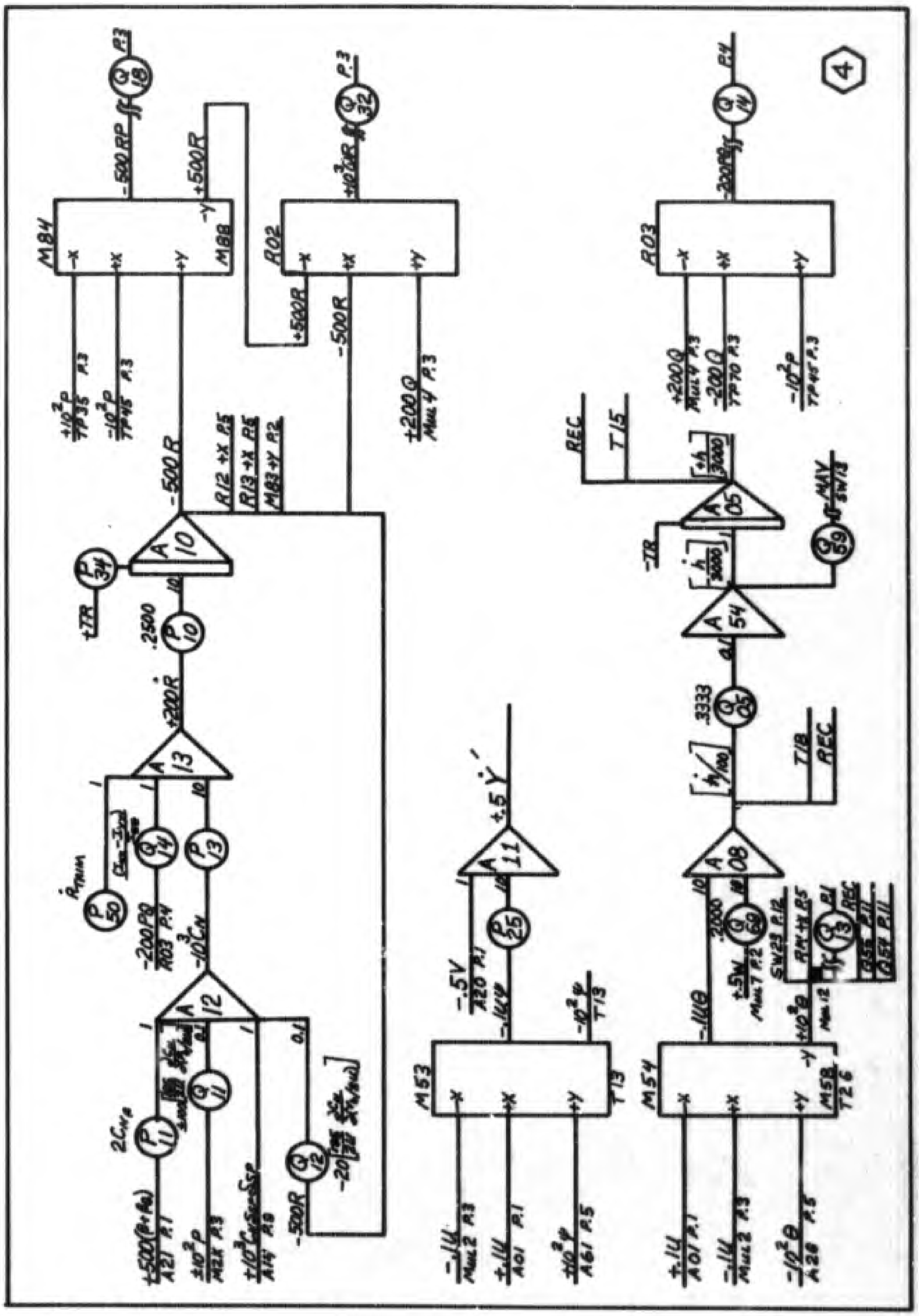


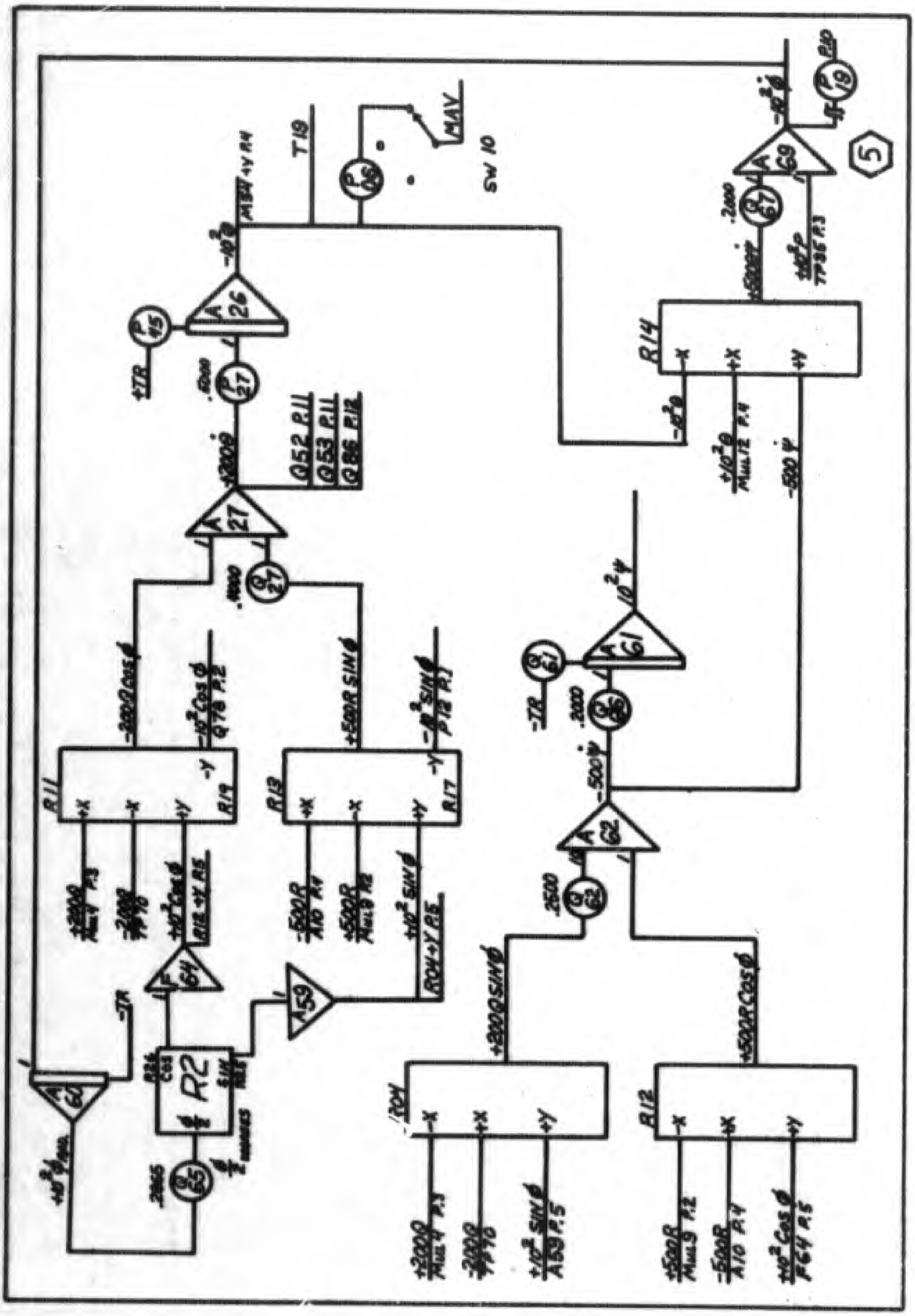
Fig. 4-1. Spoiler Attenuation Constant . . . SDF Pot Padders

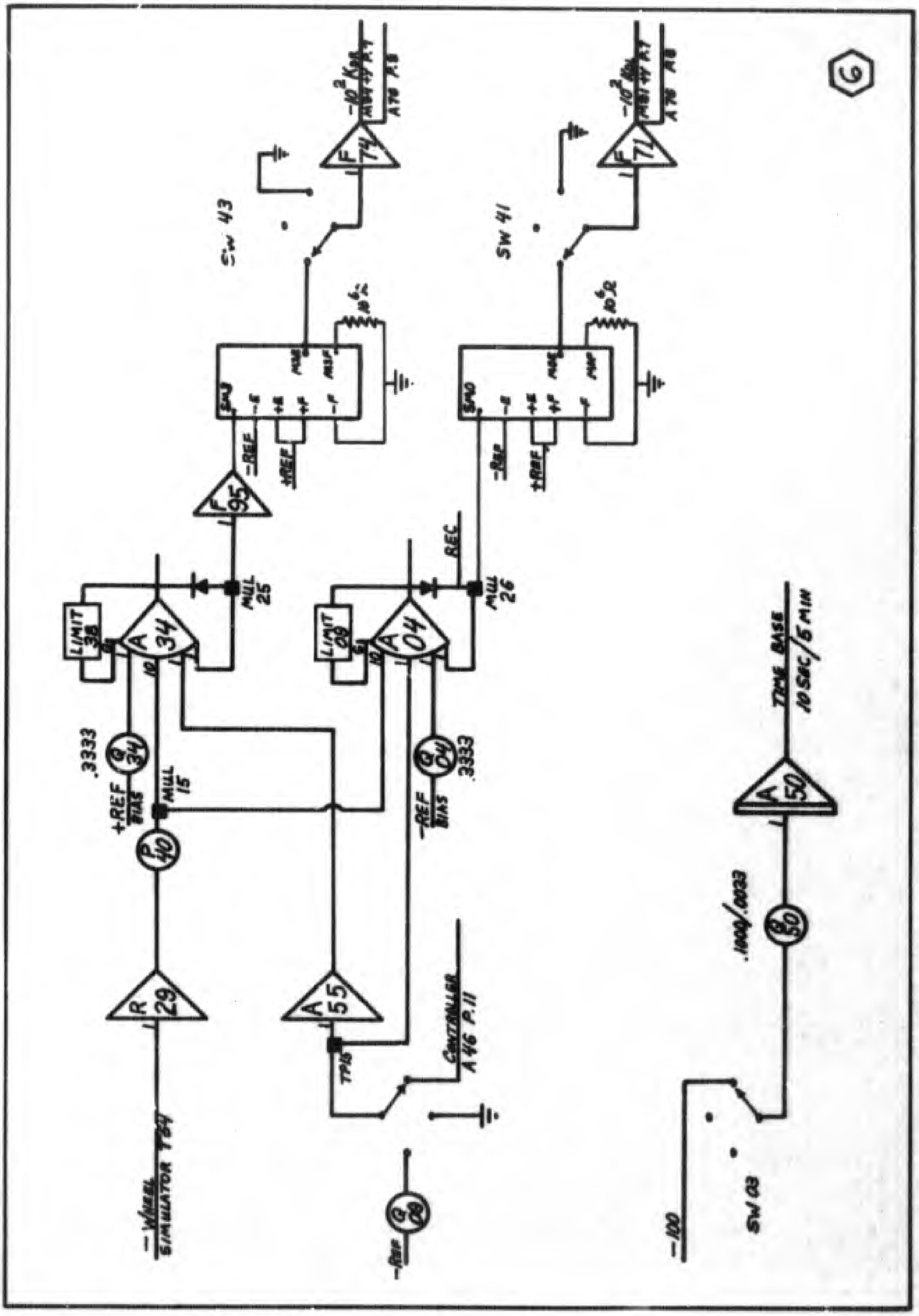


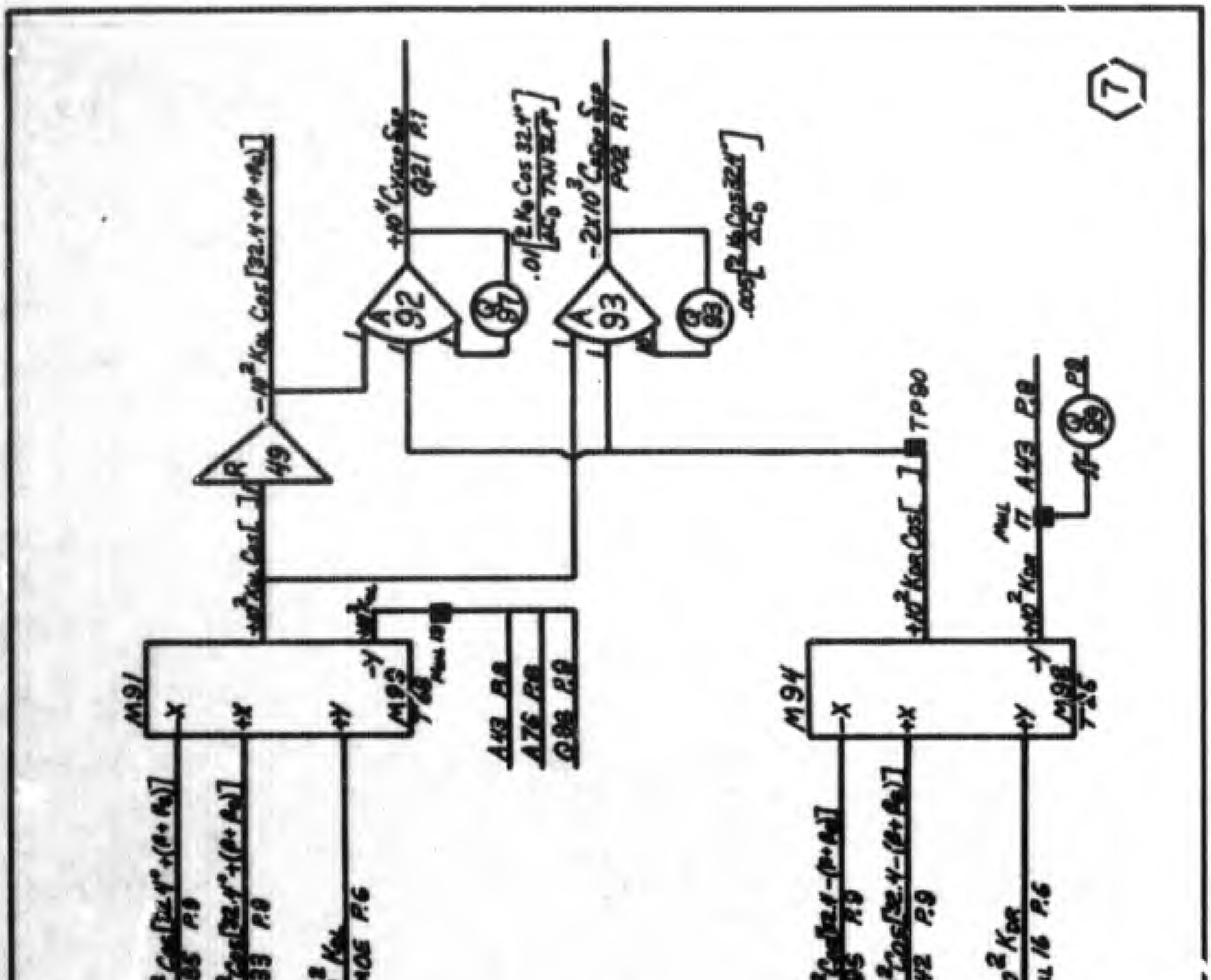


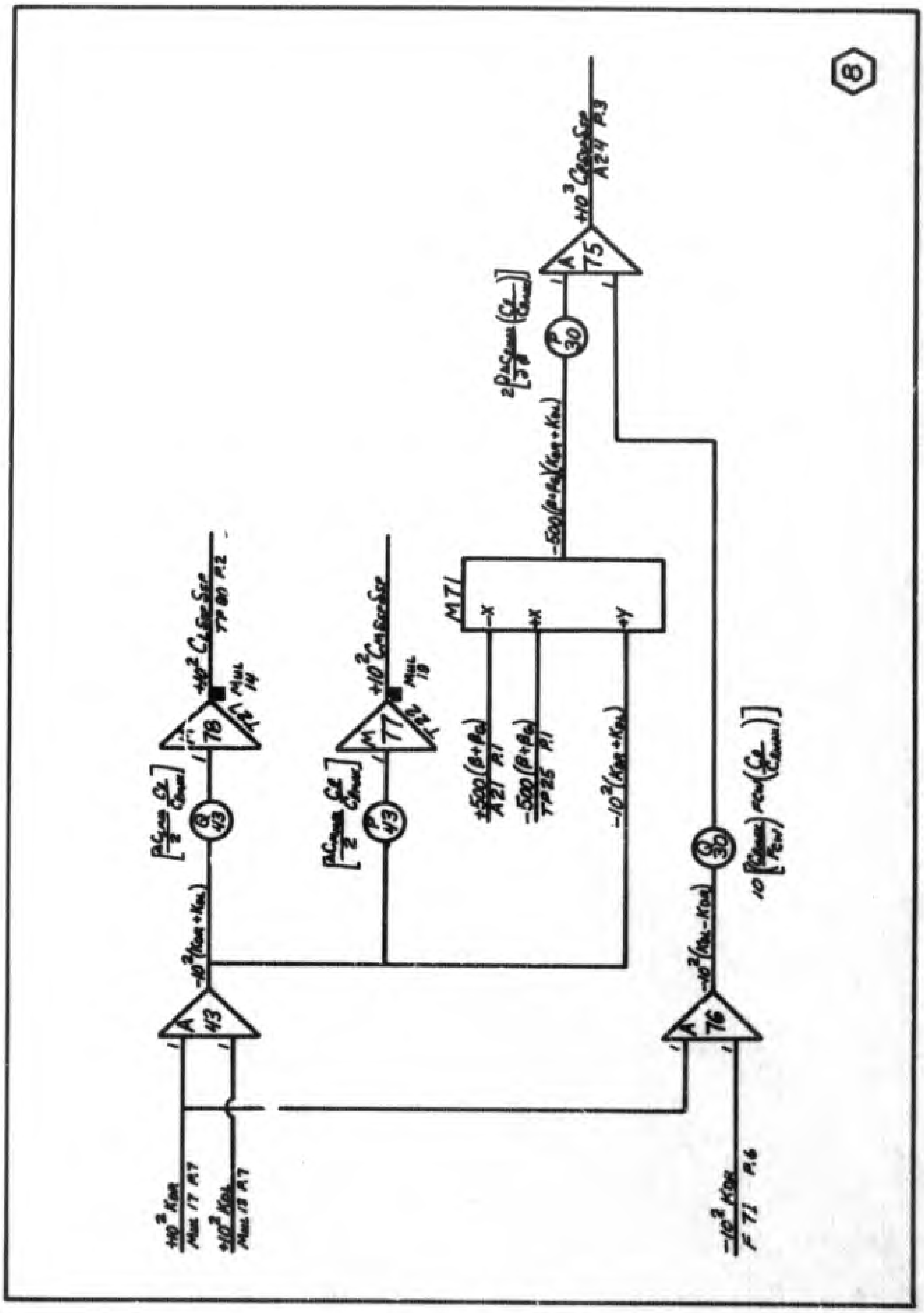


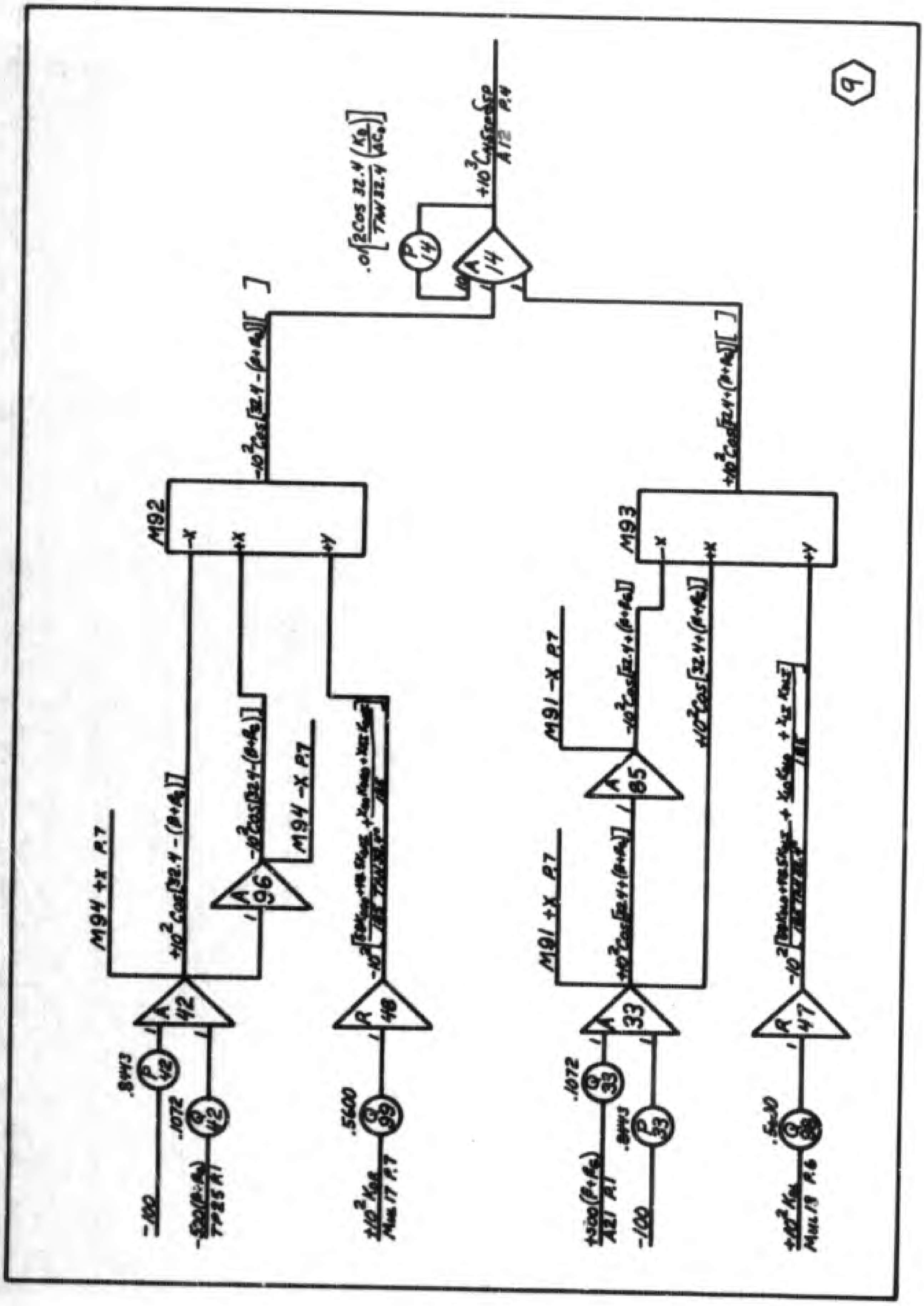

M84
M88
R02
R03
M53
M54
-500R
+500R
-500RP
REC
T15
T18
4

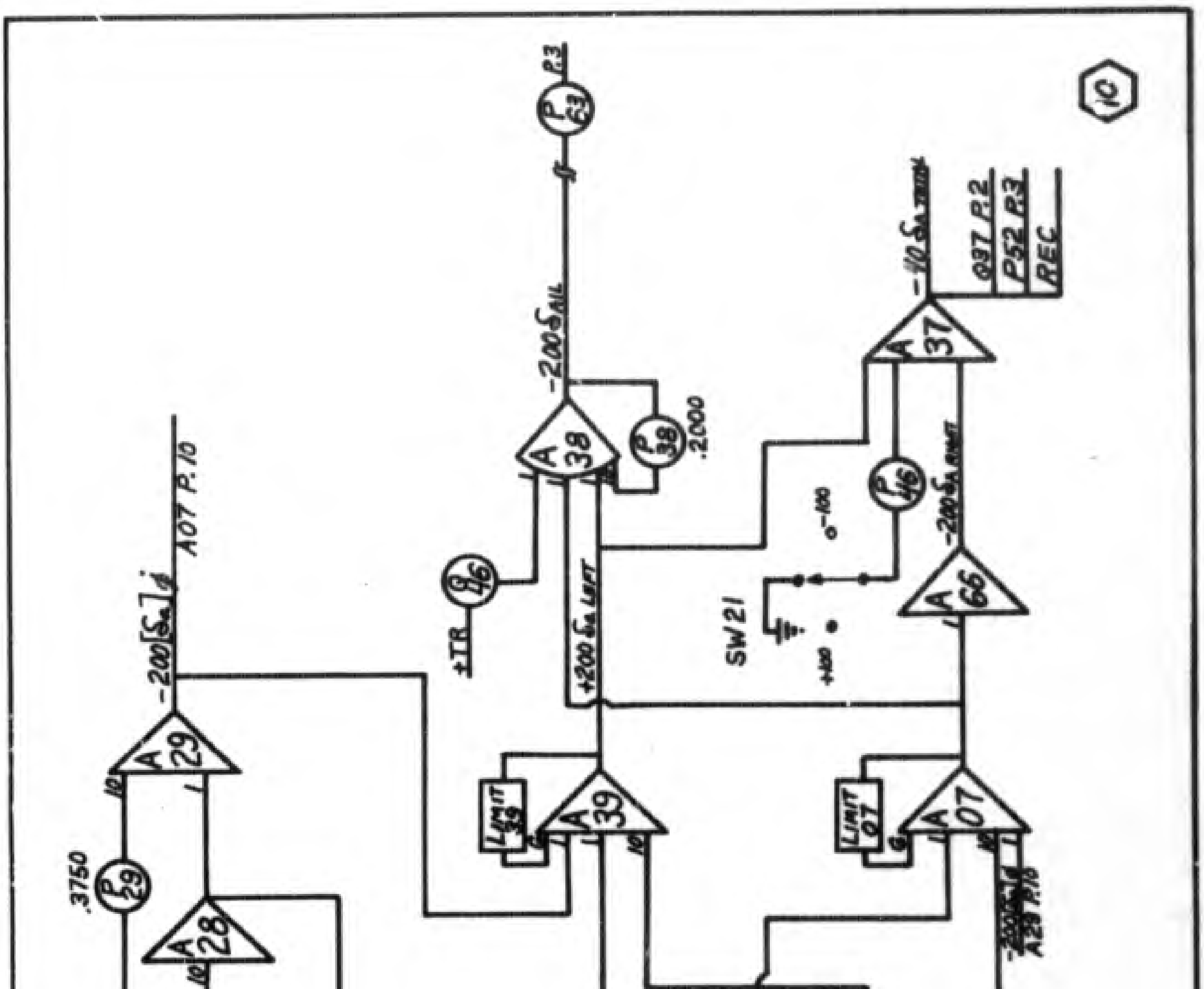

R11
R13
R14
R12
R2
T19
MAV
SW 10
Q52 P.11
Q53 P.11
Q86 P.12
5

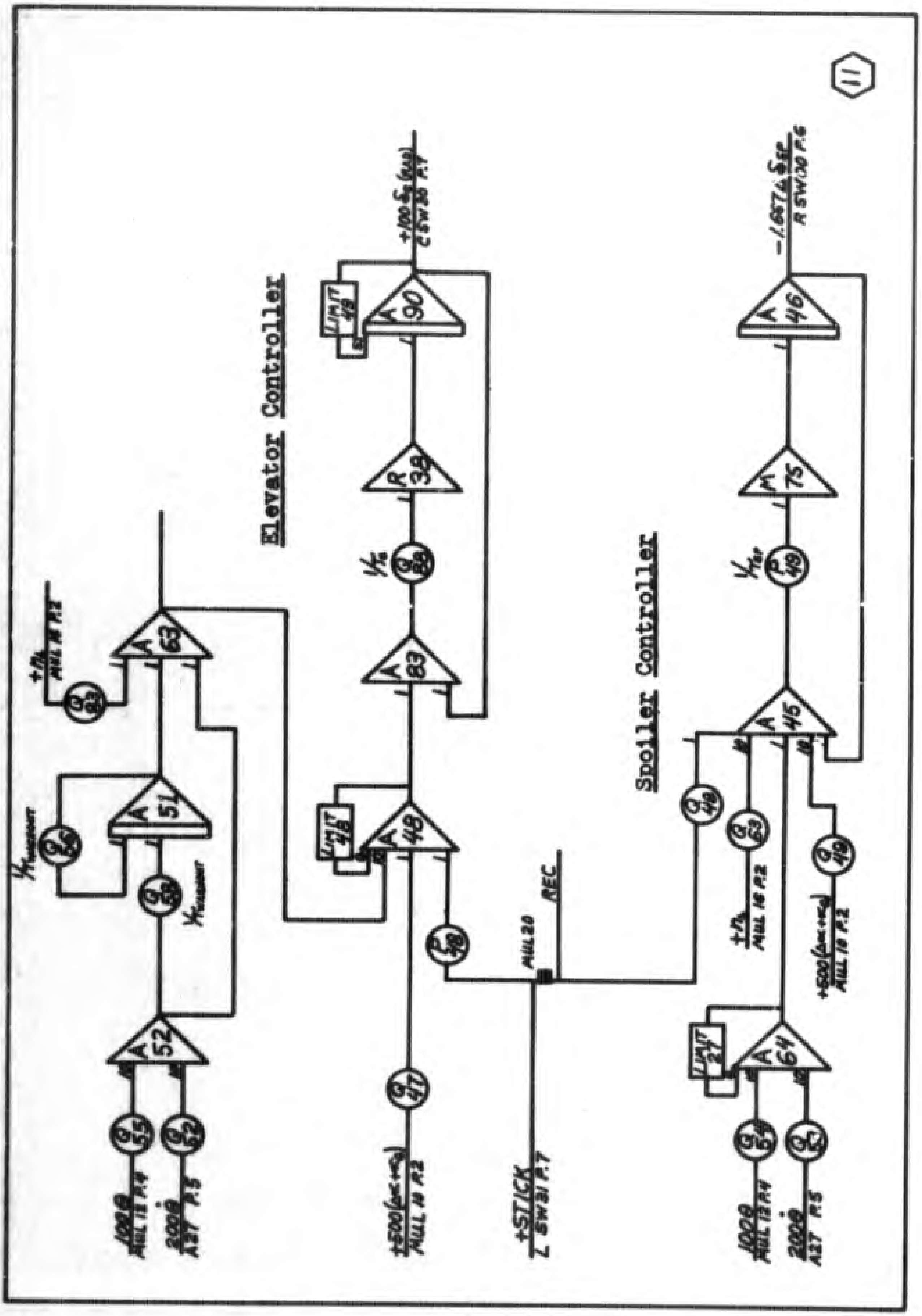

SW 43
SW 41
MULL 25
MULL 26
MULL 15
LIMIT 38
LIMIT 08
-REF
+REF
REC
.3333
TIME BASE
10 SEC/5 MIN
.1000/.0033
SW 03
-100
CONTROLLER A46 P.11

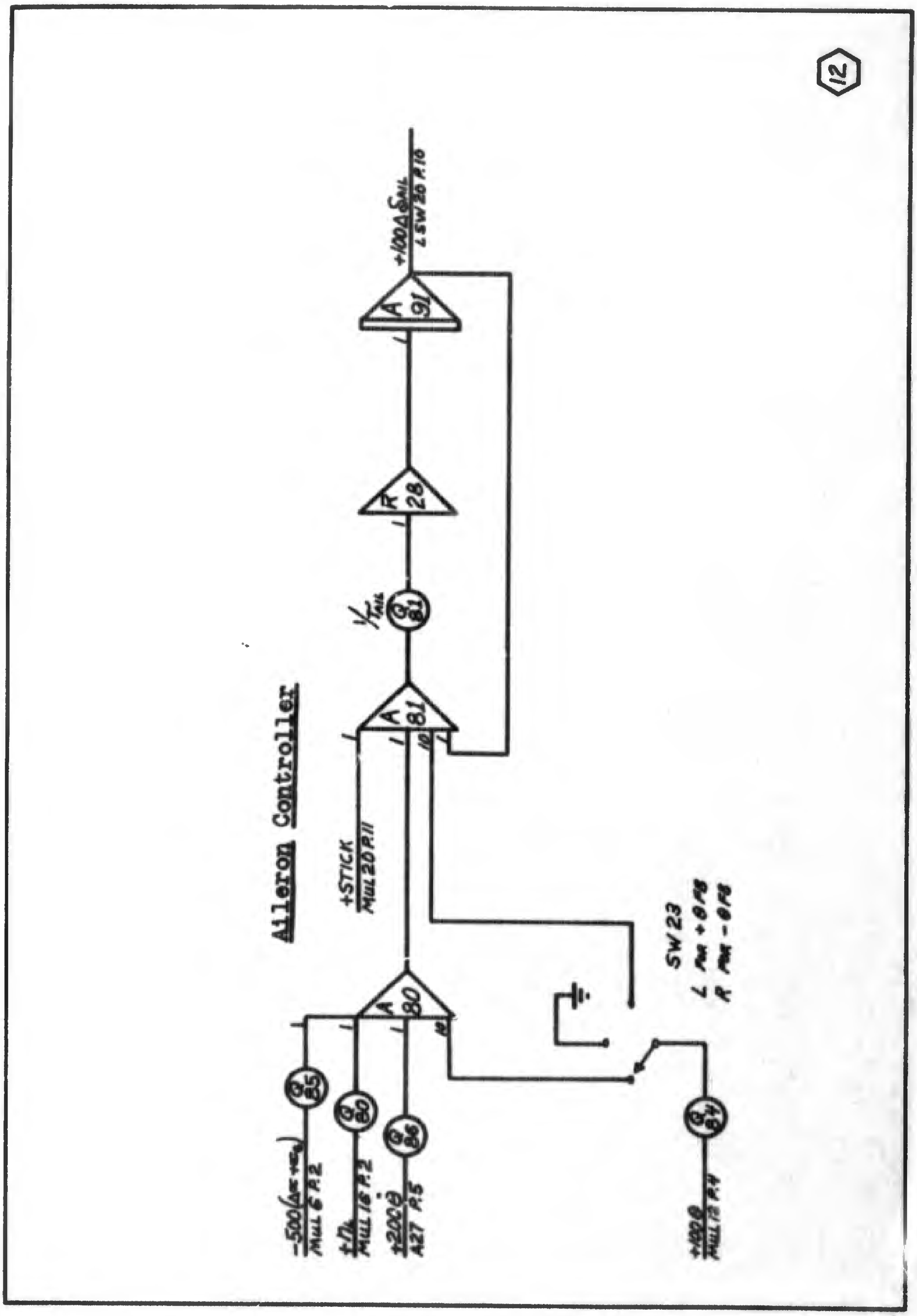

SW33
SW30
SW31
SW32
REC
M91
M94
TP90

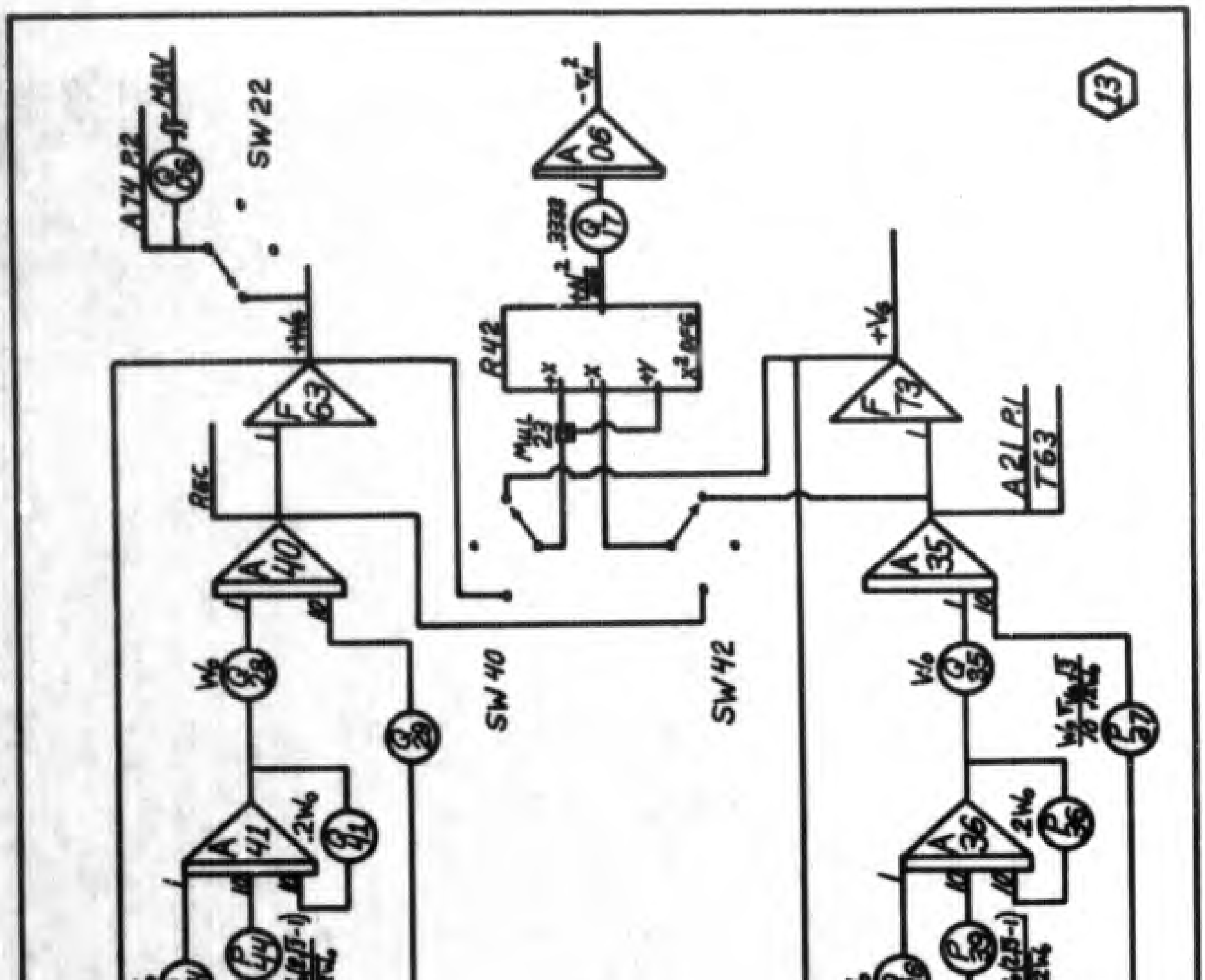

8
M71
75
76
77
78
43

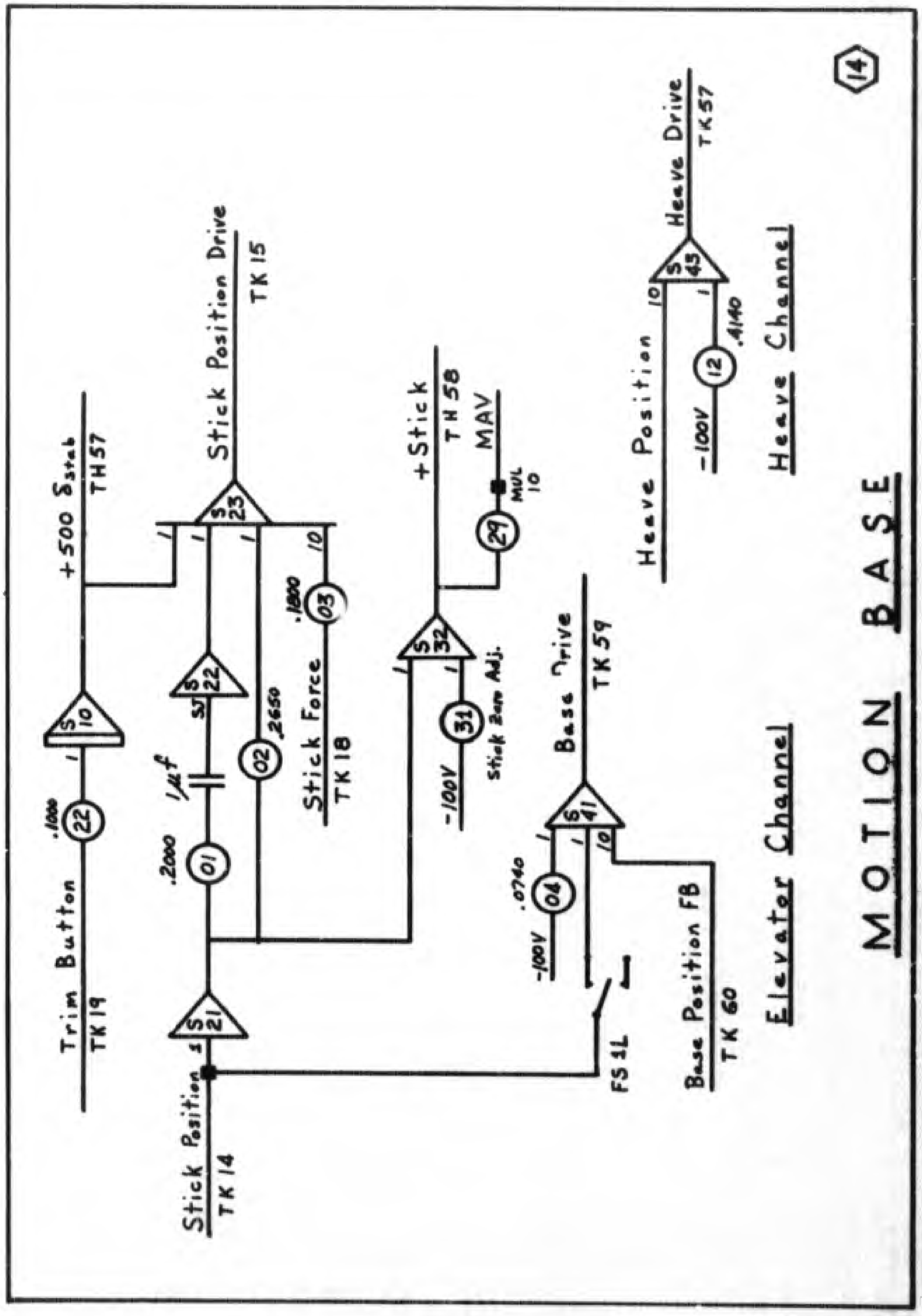

9
M92
M93
M94 +X P7
M94 -X P7
M91 +X P7
M91 -X P7
-100
-100

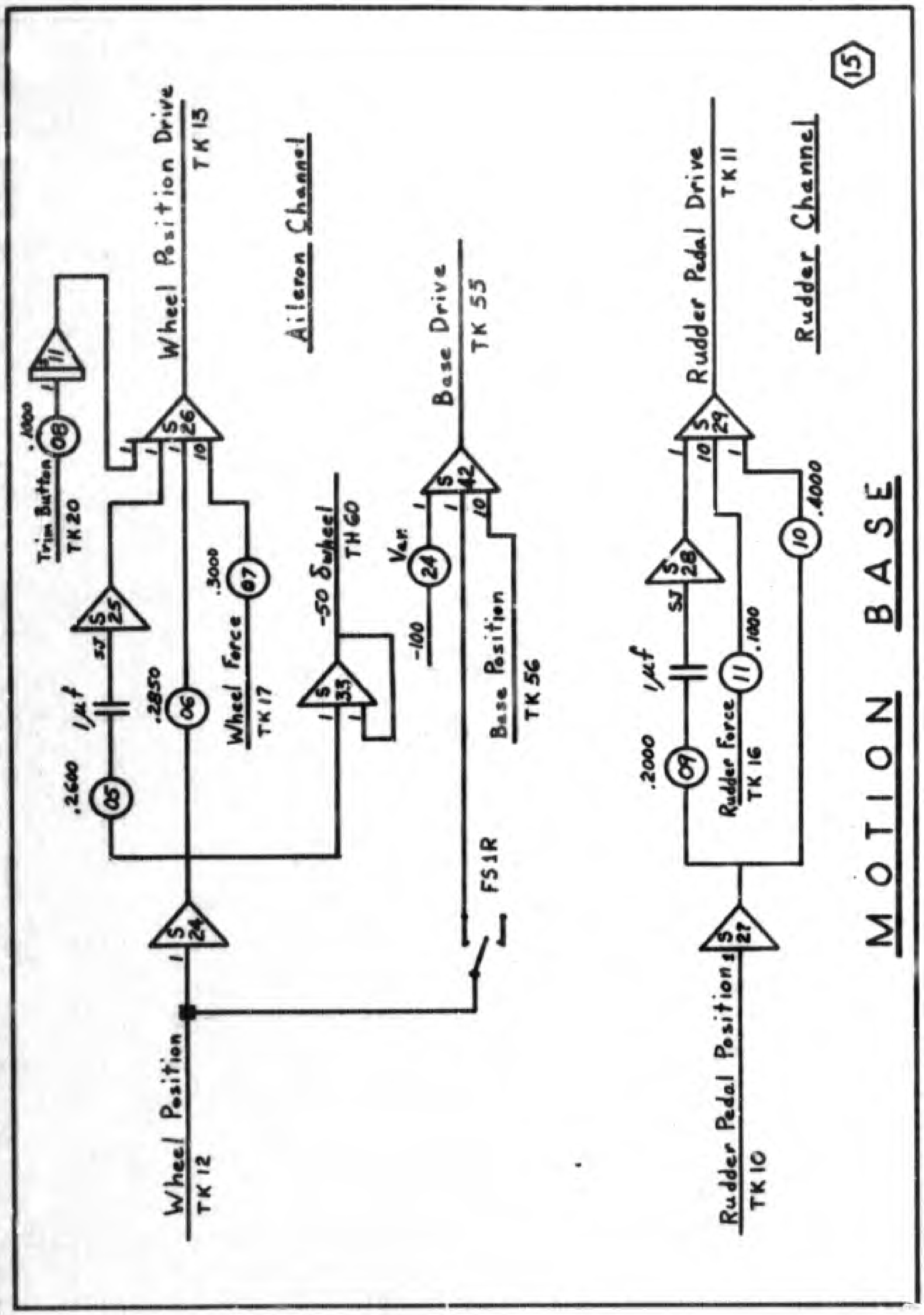

A 19
A 25
A 28
A 29
A 39
A 07
A 38
A 37
A 66
F 60
R 48
LIMIT 39
LIMIT 07
MUL 18
MUL 21
SW 01
SW 20
SW 21
.3750
.2000
A07 P.10
-REF
±TR
-TR
REC

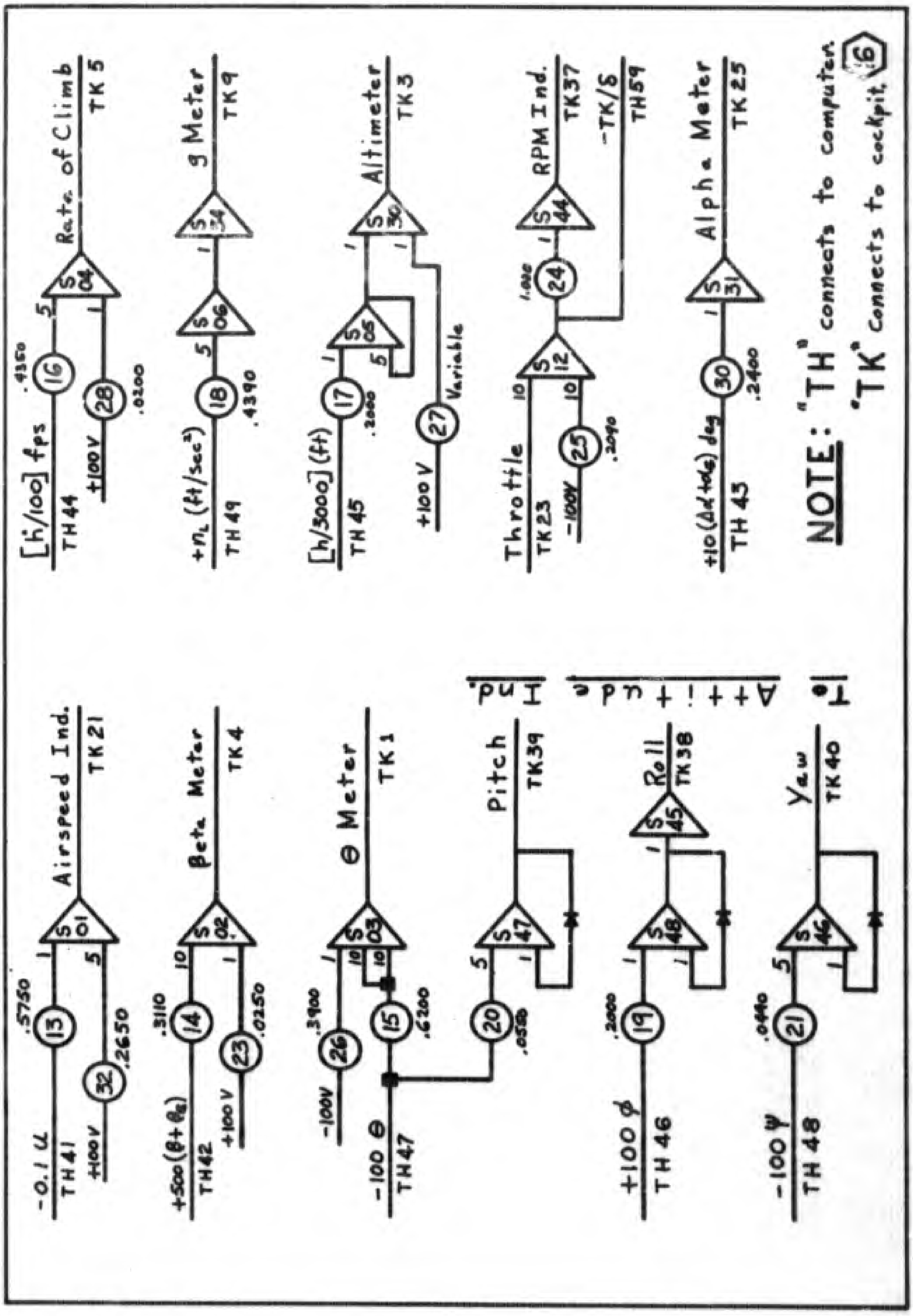

Elevator Controller
Spoiler Controller

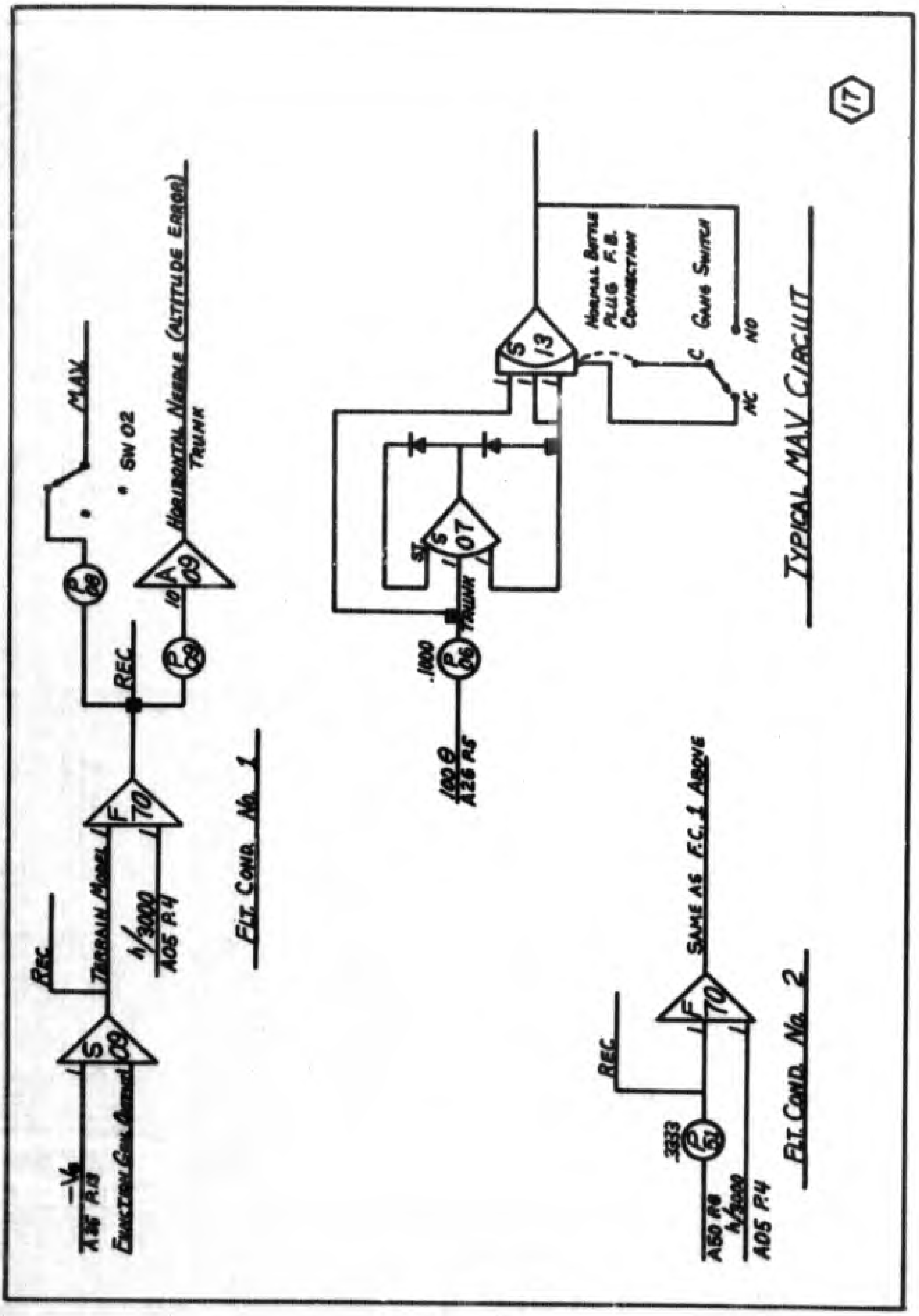

Aileron Controller
+STICK
MUL 20 P.11
SW 23


SW 22
SW 40
SW 42
R42
REC
NOISE GEN.
NOISE GEN.
SUCH THAT
EQUALS 1 VOLT RMS
13


Trim Button
TK19
.1000
+500 δstab
TH57
Stick Position Drive
TK15
Stick Position
TK14
.2000
1µf
.2650
Stick Force
TK18
.1800
+Stick
TH58
MAV
MUL 10
-100V
Stick Zero Adj.
Base Drive
TK59
.0740
-100V
FS 1L
Base Position FB
TK60
Elevator Channel
Heave Position
-100V
.4140
Heave Drive
TK57
Heave Channel
MOTION BASE
14


MOTION BASE
15
Aileron Channel
Rudder Channel
Wheel Position
TK 12
Wheel Position Drive
TK 13
Trim Button
TK 20
.1000
08
11
.2600
05
1μf
.2850
06
Wheel Force
TK 17
.3000
07
-50 δ_{wheel}
TH 60
Var.
24
-100
Base Drive
TK 55
Base Position
TK 56
FS1R
Rudder Pedal Position
TK 10
Rudder Pedal Drive
TK 11
.2000
09
Rudder Force
TK 16
11
.1000
10
.4000


-0.1 u
TH41
+100V
Airspeed Ind.
TK21
+500(β+β_g)
TH42
+100 V
Beta Meter
TK4
-100V
-100 Θ
TH47
Θ Meter
TK1
Pitch
TK39
Ind.
+100 φ
TH 46
Roll
TK38
Attitude
-100 ψ
TH 48
Yaw
TK40
I.
[ḣ/100] fps
TH44
+100V
Rate of Climb
TK 5
+n_L (ft/sec²)
TH 49
g Meter
TK9
[h/3000] (ft)
TH 45
+100 V
Variable
Altimeter
TK3
Throttle
TK23
-100V
RPM Ind.
TK37
-TK/S
TH59
+10(Δα+α_0) deg
TH 43
Alpha Meter
TK25
NOTE: "TH" connects to computer.
"TK" connects to cockpit.



A36 P.13
FUNCTION GEN. OUTPUT
REC
TERRAIN MODEL
h/3000
A05 P.4
S
F
70
REC
MAX
SW 02
A
09
HORIZONTAL NEEDLE (ALTITUDE ERROR)
TRUNK
FLT. COND. NO. 1
1000
A28 P.5
.1000
P
06
TRUNK
S
07
S
13
NORMAL BOTTLE PLUG F.B. CONNECTION
GANG SWITCH
C
NC
NO
TYPICAL MAV CIRCUIT
17
A50 P.8
h/3000
A05 P.4
.3333
P
51
REC
SAME AS F.C. 1 ABOVE
FLT. COND. NO. 2

SIX DEGREE OF FREEDOM SIMULATION

S W I T C H   P O S I T I O N S

PARAMETER	SWITCH NUMBER	STATIC TEST	BASIC ACFT.		CONTROLLED	
			SIM	COMP	SIM	COMP
$-1.6667\ \Delta\ \delta_{sp}$	00	L	C	C	R	R
δ Wheel	01	C	L	R	L	R
X-Y Plotter	02	R=Run	-	-	-	-
Time Base	03	R=Run	-	-	-	-
100 θ MAV	10	R=Run	-	-	-	-
N_L MAV	11	R=Run	-	-	-	-
Δ U MAV	12	R=Run	-	-	-	-
$\dot{H}$ MAV	13	R=Run	-	-	-	-
Symmetric Aileron	20	C	R	R	L	L
Static Test Special	21	L	C	C	C	C
α Gust	22	C	R = Gust		R = Gust	
θ FB, δ_{ail}	23	C	C	C	As Reqd.	
Elevator Command	30	L	R	R	C	C
+ Stick	31	C	R	C	L	C
Stabilizer Trim	32	L	C	L	C	L
-100 T/qS	33	L	C	L	C	L
Noise Generator	40	C	R = V_G		L = W_G	
-100 K_{DL}	41	L	L	R	L	L
Noise Generator	42	C	R = V_G		L = W_G	
-100 K_{DR}	43	L	L	R	L	L

Table IV-1

Chapter V

Simulation

The pilot-in-the-loop simulation study was conducted at the AFFDL, Building 192 and 195, W-P AFB, Ohio, during February, 1968. Figure 5-1 shows the AFFDL moving base simulator used in this experiment. The moving base has limited motion capabilities in three axes (pitch, roll, and heave) and was used to provide realism to the pilots flying the simulation. Figure 5-2 shows the instruments available to the pilot and used to display appropriate parameters from the analog computer simulations of the aircraft dynamics. The instruments included were attitude director indicator, rate of climb indicator, altimeter, airspeed indicator, RPM gage, g meter, angle of attack meter, pitch angle indicators, and a sideslip meter. The throttle was connected, but a time delay between throttle movement and thrust output was not incorporated as the movements were small and the engines were operating at high RPM settings.

Pilot Experience

A total of eight pilots, seven military and one civilian, participated in this simulation with a combined total of 75 years rated experience and over 22,000 flying hours. Their experience ranged from jets (B-47's, B-52's, KC-135's,

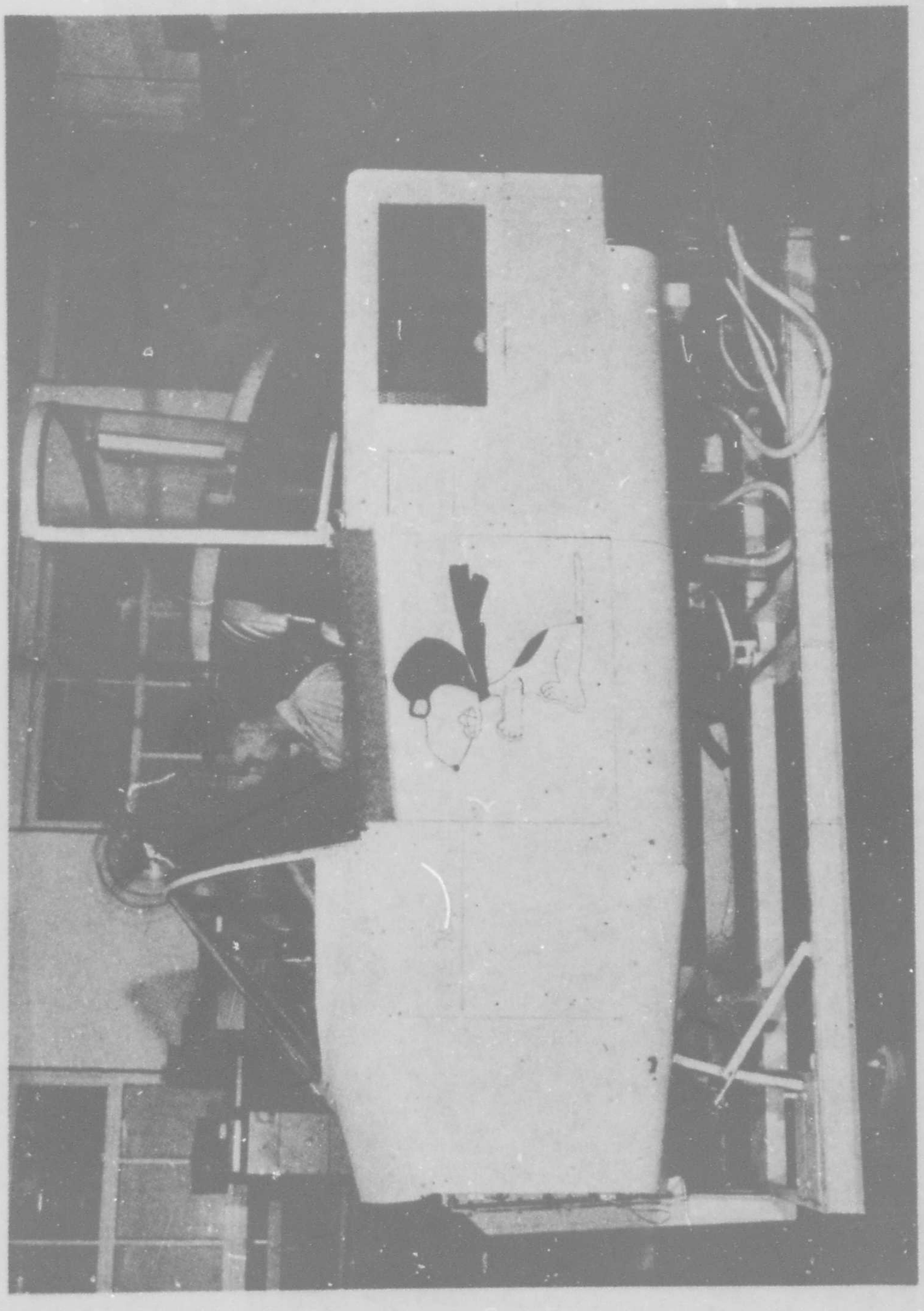


Fig. 5-1. AFFDL Moving Base Simulator

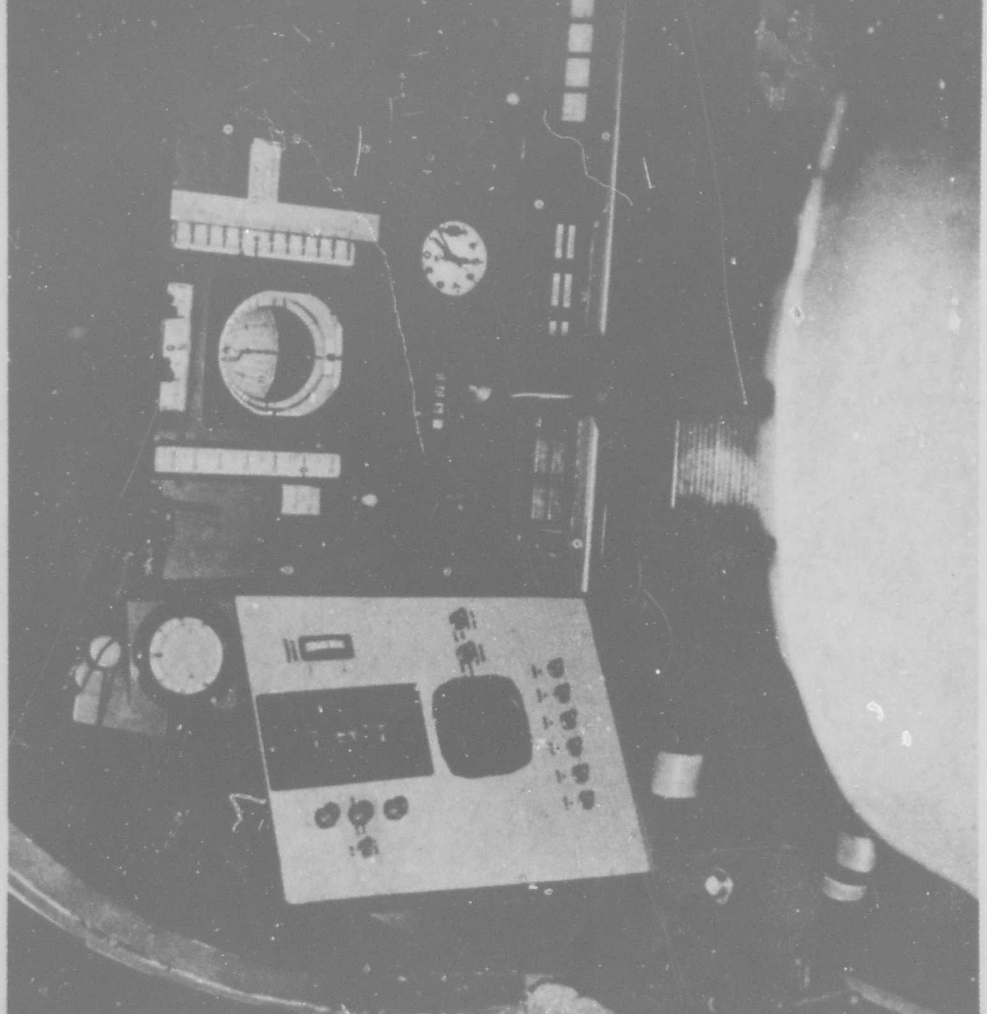


Fig. 5-2. Simulator Cockpit Instruments

F-86's, and T-33's) through reciprocating aircraft (C-124's T-29's, C-131's, C-47's, C-45's) and helicopters (HU-1B's, CH3C's, and HH-43's) to other light aircraft and simulators. This wide range of experience was both desired and needed to give a true evaluation of DLC and the possibilities of its use in present and future aircraft. The specific flight experience of each pilot is presented in Appendix G and Table VI-5 containing a condensation of this information is found on page 162.

Simulator Flight Planning

The original program called for ten pilots to fly the simulation with the two extra pilots coming from the standardization flight crew of the 17th SAC (Strategic Airlift Command) Bomb Wing located at Wright-Patterson Air Force Base, Ohio. This soon proved to be impossible due to a SAC alert and an unscheduled TDY (temporary duty) for the other pilot.

It had been planned that all the pilots would fly each of the three flight conditions, but due to maintenance problems and other scheduling conflicts this was impossible in the few days available. As a result, only four pilots completed all three flight conditions and the last pilot flew only one flight condition.

The three flight conditions listed on page 74 were selected from the LAMS profile and used because they simulate in-flight refueling at altitude, approach and landing, and high speed terrain following. The specific tasks completed by each pilot are listed below.

TASK	PILOTS
Terrain Following	1-2-3-5-6
Approach & Landing	1-2-3-4-5-6-7
Refueling	1-3-4-5-6-7-8

The tasks to be performed by the pilots in each flight condition included a variety of maneuvers such as those expected in actual flight. These maneuvers included rates of climb and descent at constant airspeed while either maintaining heading or making small changes in heading to correct for errors in glide path or alignment to the refueling tanker.

Flight Condition One

In flight condition one, terrain following, rates of climb and descent of 250 fpm and 500 fpm were used without using the maximum deflection of DLC surfaces. It should be noted that rates of climb and descent in excess of those provided by DLC are attained in the conventional manner, i.e., applying more stick or column force will move the elevator more, providing a greater rate of climb or descent. It is important to note that even though the additional elevator resulted in an increase in pitch angle, the angle is less than that experienced in the basic aircraft while both pitch rate overshoot and the initial acceleration reversal were eliminated (See Fig. 1-1, page 3). The terrain following task as described in Fig. 5-3, page 107, gives the maneuver to be evaluated as well as a list of questions pertaining to the completion of the task. A simulated terrain was supplied by summing the output of a noise generator with the output of a function generator capable of providing either a sine or triangular wave form. This "terrain" was then presented to the pilot on the horizontal director needle.

After each flight the pilot was given a Pilot Evaluation Form like the one shown in Fig. 5-4, page 108. With this form and the Cooper Rating Form, (Fig. 5-5, page 109), the pilot rated and compared the basic aircraft and the DLC aircraft. For the pilots inclined to give oral rather than written reports, a tape recorder was available to collect this data.

Flight Condition Two

In flight condition two, approach and landing, rates of climb and descent of 200 fpm and 400 fpm were used, as this was representative of rates needed to correct for glide slope errors. The pilots were also asked to make small heading changes of plus or minus ten degrees to simulate corrections for final approach glide path. An ILS (Instrument Landing System) glide slope was simulated by using the output of an integrator to provide a constant rate of descent, and this was displayed on the horizontal director needle. After each flight, the pilots were again asked to evaluate the simulation and make any comments pertinent to the evaluation. A summary of the approach and landing task is found in Fig. 5-6 on page 110.

Flight Condition Three

In flight condition three, in-flight refueling, it was decided not to simulate a flight path on the horizontal and vertical director indicator needles because of the lack of visual cues, so important to this precise maneuver, such as distance and rate of closure to the tanker. Therefore, the task consisted of making climbs and descents at 300 and 600 feet per minute while maintaining constant airspeed and making plus or minus ten degree changes in heading to simulate alignment to the refueling tanker. The refueling task card is found in Fig. 5-7, page 111.

General Comments

The Cooper Rating Form (Fig. 5-5, page 109) is a standard aeronautical rating form used to evaluate aircraft and/or control systems. Although it is not the only rating system available, it has been used extensively, and provides a good basis for qualitatively comparing the basic and DLC aircraft. The form covers the range of pilot opinions from excellent to completely unacceptable, and defines a certain range in which the system could be used for emergency operation only.

Using the Cooper Rating Form, the pilots evaluated the DLC system as well as the basic aircraft, and a sample of their comments is found in Appendix I.

It was convenient to devise a run number code that would give all the pertinent information at a glance. An explanation of the code used is given next.

Run Number Code

Example 4 - GGC - 68 - R B

B - Basic Aircraft
C - Direct Lift Control

R - Refueling
L - Approach & Landing
T - Terrain Following

Thesis Designator

Pilot Number

The number was then used to identify pilot comment sheets, the strip recordings, and the list of MAV (mean absolute values), numbers which were read after each run was completed. Due to a shortage of integrators on the main console, it was necessary to use the extra integrators located on the simulator console to obtain the MAV. This presented a problem because the simulator console was operating the moving base simulator and therefore had to be kept in the "operate" mode. Thus, prior to each run, the MAV circuits were reset to zero by use of a switch which connected the outputs of the integrators being used to their respective grids. An interphone system, with a "hot mike" on the simulator side, was used to maintain communications between the main console and the moving base simulator.

A total of 109 runs were completed for record and averaged approximately five minutes per run. A short period of practice was given to each pilot to familiarize himself with the simulator and the "feel" of the stick. Although most pilots felt that more practice would be advantageous, it did not greatly affect their performance. A complete series of runs in a particular flight condition took approximately one hour, and a sample list is given below.

15 min. - 30 min.	Practice
4 min. - 6 min.	Climbs & Descents (Basic Aircraft)
4 min. - 6 min.	Climbs & Descents (Basic Aircraft with turbulence)
2 min. - 4 min.	Tracking Task (Basic Aircraft with turbulence, F.C. one and two)
4 min. - 6 min.	Climbs & Descents (DLC Aircraft)
4 min. - 6 min.	Climbs & Descents (DLC Aircraft with turbulence)
2 min. - 4 min.	Tracking Task (DLC Aircraft with turbulence, F.C. one and two)

The order in which the pilots flew the simulation was varied, that is, some pilots flew the basic aircraft first and some flew the DLC aircraft first. But all pilots flew the task without turbulence prior to flying with turbulence. The turbulence was adjusted to give peak gusts of ten feet per second and a MAV of three to four feet per second. Although the turbulence was well behaved, a record was kept to insure that if problems developed during a run, a check could be made to verify whether or not the turbulence was realistic.

Two strip recorders were used to record 15 channels of data necessary to evaluate Direct Lift Control. These 15 channels and the particular parameters are listed on page F-18 of Appendix F.

Evaluation Maneuvers

1. Trim aircraft for straight and level flight, 4000 feet and 370 knots.
2. Climb 300 feet at 250 feet/minute.
3. Descend 600 feet at 250 feet/minute.
4. Climb 600 feet at 500 feet/minute.
5. When simulated terrain is introduced, track it by driving the horizontal director needle to zero.
6. Maintain constant heading and airspeed.

Pilot Comment Checklist

1. Were you able to perform the task easily?
2. Was airspeed difficult to maintain?
3. Was heading difficult to maintain?
4. Was there any tendency towards PIO (Pilot Induced Oscillations)?
5. What effect did turbulence have on performing the task?
6. Was the information presented on the display adequate?
7. How would you rate the Direct Lift Controlled aircraft compared to the basic aircraft?
8. How well does the moving base simulate motion?

Fig. 5-3. Terrain Following Task
(Flight Condition 1)

Pilot Evaluation Form

Date ____________________

Run Number ______________________

Task ____________________

Cooper Rating	How Did You Do?	Work Load
Pitch ______	1. Above Average	1. Light
Lateral ______	2. Average	2. Normal
Overall ______	3. Poor	3. Heavy
	4. Lousy	4. Extreme

Comments: Tape ______ Start ______ Stop ______

____

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Fig. 5-4. Pilot Evaluation Form

Cooper Pilot Rating System

Operating Conditions	Adjective Rating	Numerical Rating	Description	Primary Mission Accomplished
Normal Operation	Satisfactory	1	Excellent, includes optimum	Yes
		2	Good, pleasant to fly	Yes
		3	Satisfactory, but with some mildly unpleasant characteristics	Yes
Emergency Operation	Unsatisfactory	4	Acceptable, but with unpleasant characteristics	Yes
		5	Unacceptable for normal operation	Doubtful
		6	Acceptable for emergency condition only	Doubtful
No Operation	Unacceptable	7	Unacceptable even for emergency condition	No
		8	Unacceptable - dangerous	No
	Catastrophic	9	Unacceptable - uncontrollable	No
		10	Motions possibly violent enough to prevent pilot escape	No

Fig. 5-5. Cooper Rating Form

Evaluation Maneuvers

1. Trim aircraft for straight and level flight, 4000 feet and 254 knots.
2. Climb 200 feet at 200 feet/minute.
3. Descend 400 feet at 200 feet/minute.
4. Climb 400 feet at 400 feet/minute.
5. When glide slope is introduced, track it by driving the horizontal director needle to zero while making plus or minus ten degree heading changes to simulate azimuth correction.
6. When flare is announced, hold the horizontal director needle to zero while maintaining level flight for approximately 30 seconds.
7. Maintain constant airspeed.

Pilot Comment Checklist

1. Were you able to perform the task easily?
2. Was airspeed difficult to maintain?
3. Was there any tendency towards PIO?
4. What effect did turbulence have on performing the task?
5. Was the glideslope displayed adequately?
6. How would you rate the Direct Lift Controlled aircraft compared to the basic aircraft?

Fig. 5-6. Approach and Landing Task
(Flight Condition 2)

Evaluation Maneuvers

1. Trim aircraft for straight and level flight, 32,700 feet and 448 knots.
2. Climb 300 feet at 300 feet/minute.
3. Descend 600 feet at 300 feet.minute.
4. Climb 600 feet at 600 feet/minute.
5. Make small heading changes, plus or minus ten degrees, to simulate alignment to the refueling tanker.
6. Maintain constant airspeed.

Pilot Comment Checklist

1. Were you able to perform the task easily?
2. Was airspeed difficult to maintain?
3. Was there any tendency toward PIO?
4. What effect did turbulence have on performing the task?
5. How would you rate the Direct Lift Controlled aircraft compared to the basic aircraft?
6. Would it have been good to simulate the refueling tanker's position by use of the horizontal needle?
7. How well does the moving base simulate aircraft motion?

Fig. 5-7. In-Flight Refueling Task
(Flight Condition 3)

Chapter VI

Results

Comparison of Six and Two Degree of Freedom Simulations

Figures 6-1 through 6-8 are basic aircraft responses to various step control surface deflections in flight condition one. They are paired to show a comparison between the six and two degree of freedom simulations. The pairs are:

Fig. 6-1 and 6-2 . . . Elevator 1° TE up.

Fig. 6-3 and 6-4 . . . Spoilers 20° TE up.

Fig. 6-5 and 6-6 . . . Ailerons 5° TE up.

Fig. 6-7 and 6-8 . . . Aircraft trimmed with all spoilers biased at 20°, then elevator deflected 1° TE up.

Figure 6-1A (Boeing) also shows the aircraft response to one degree of elevator deflection. The data for this figure was provided by The Boeing Company, Wichita Division. Comparison of Figures 6-1 and 6-1A (Boeing), and similar pairs in Appendix D for flight conditions two and three, demonstrates the validity of the simulation used in this thesis.

The parameters recorded from the SDF (six degree of freedom) simulation are ΔU (change in airspeed), θ (pitch angle), $\dot{\theta}$ (pitch rate), and $\Delta \alpha$ (change in angle of attack).

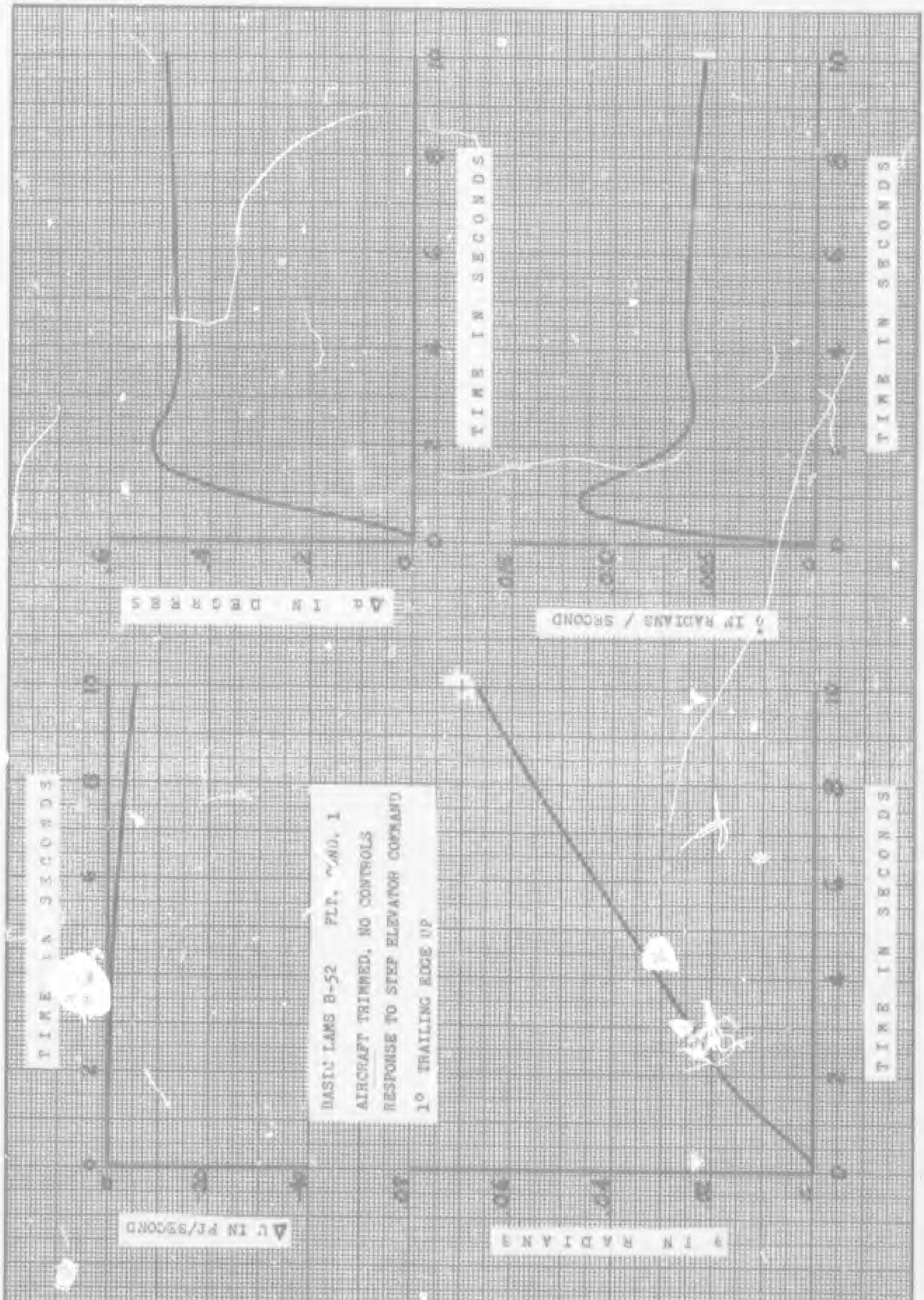


Fig. 6-1. Six Degree of Freedom Elevator Response

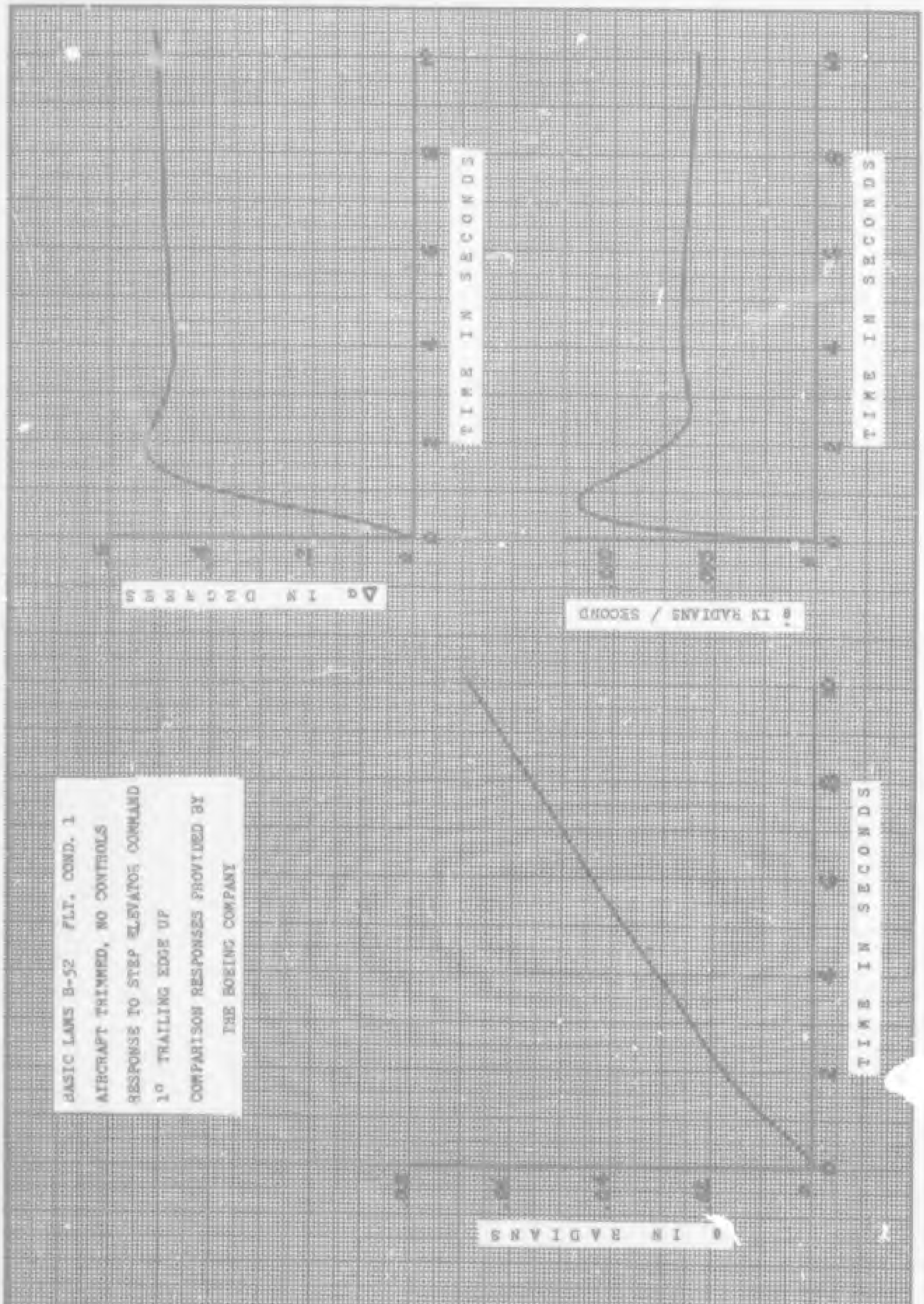


Fig. 6-1A (Boeing). Six Degree of Freedom Elevator Response

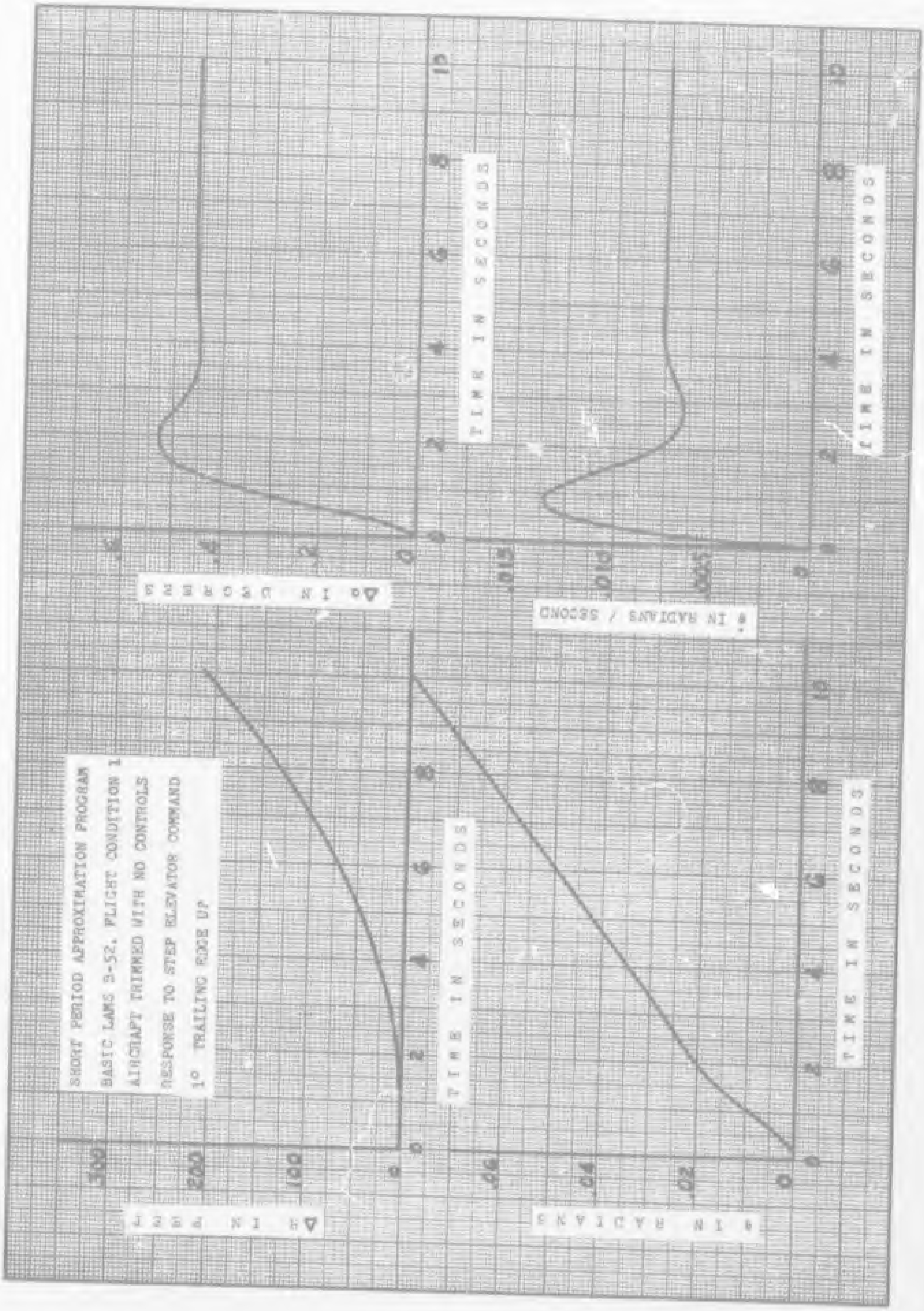


Fig. 6-2. Two Degree of Freedom Elevator Response

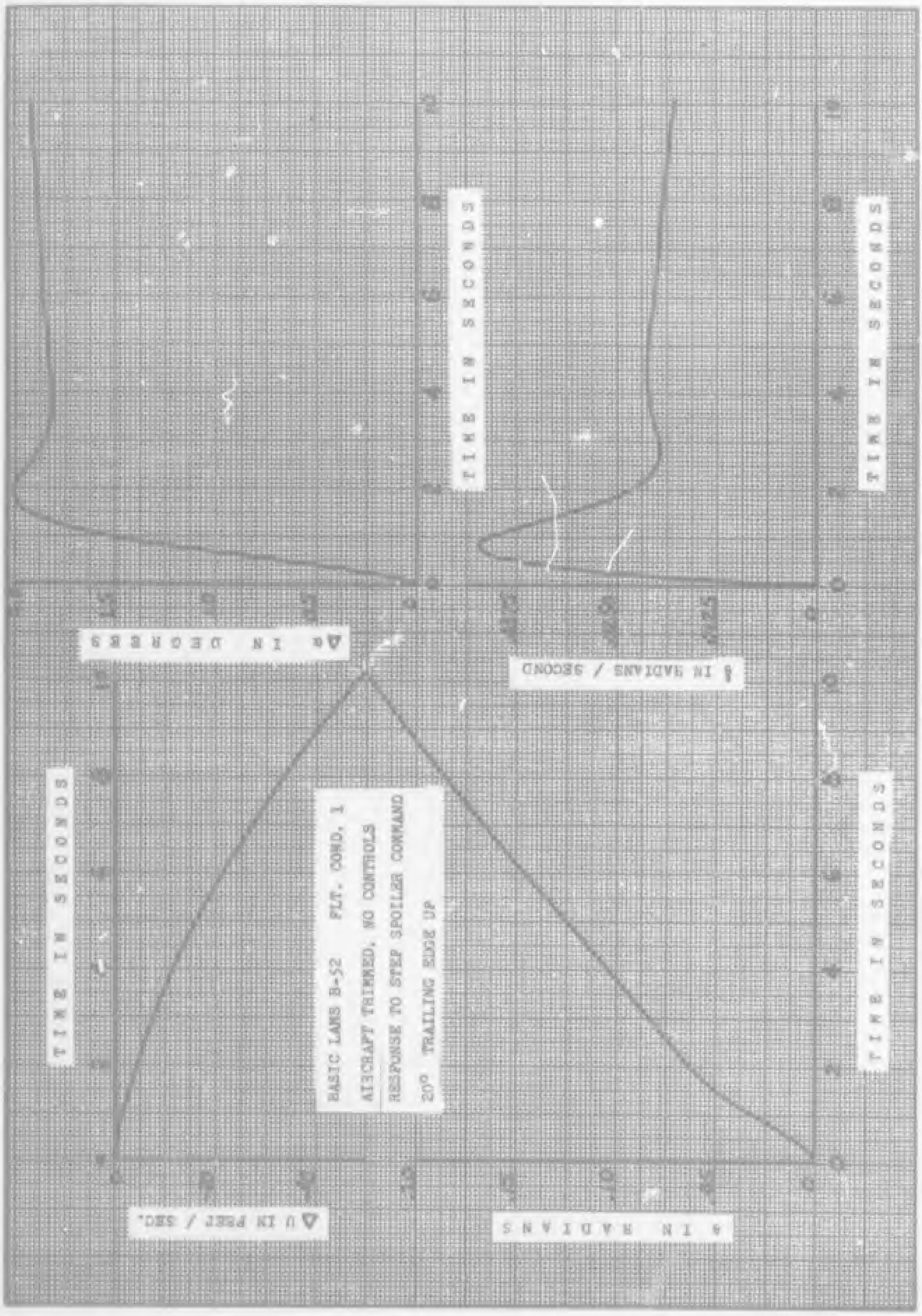


Fig. 6-3. Six Degree of Freedom Spoiler Response

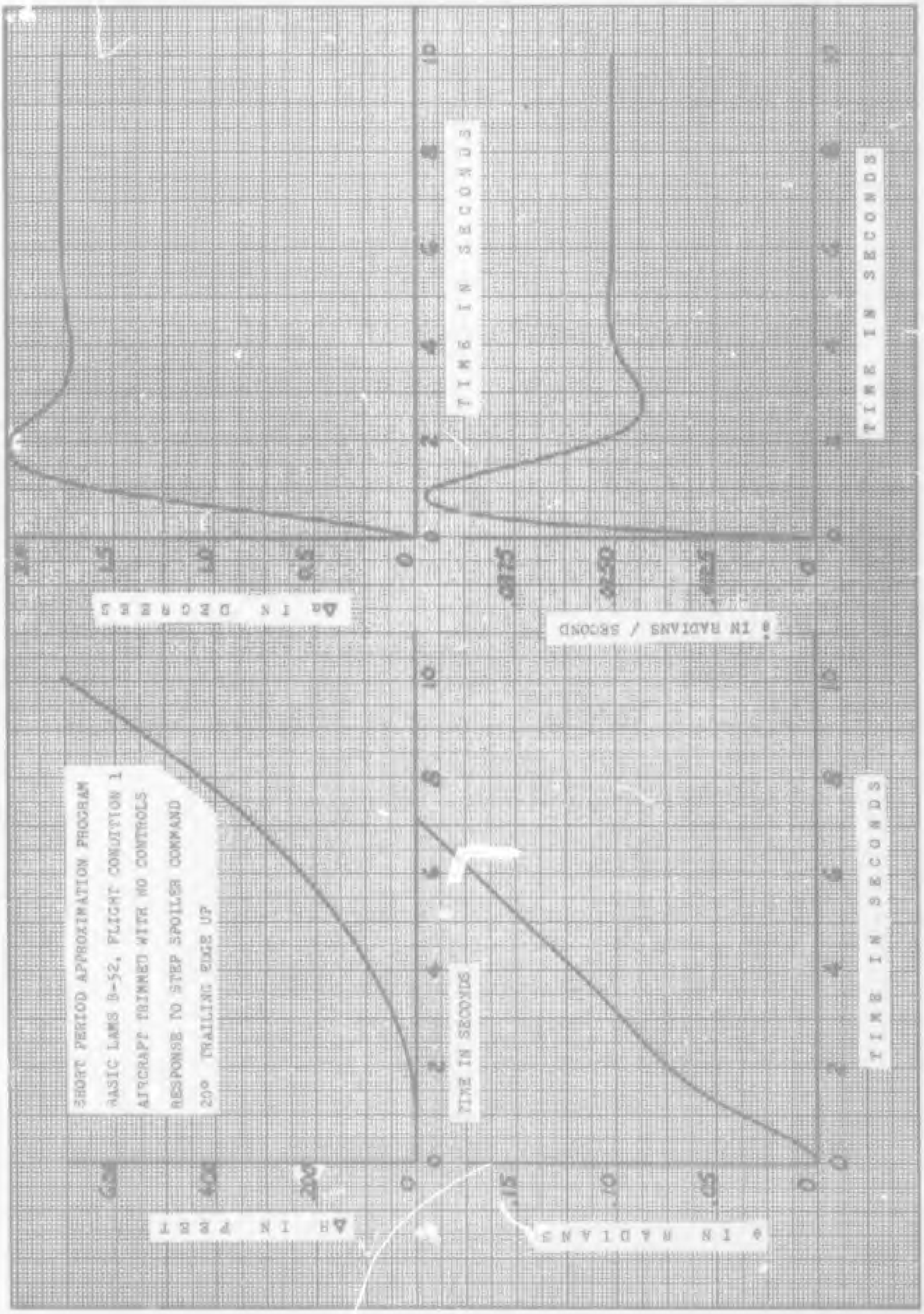


Fig. 6-4. Two Degree of Freedom Spoiler Response

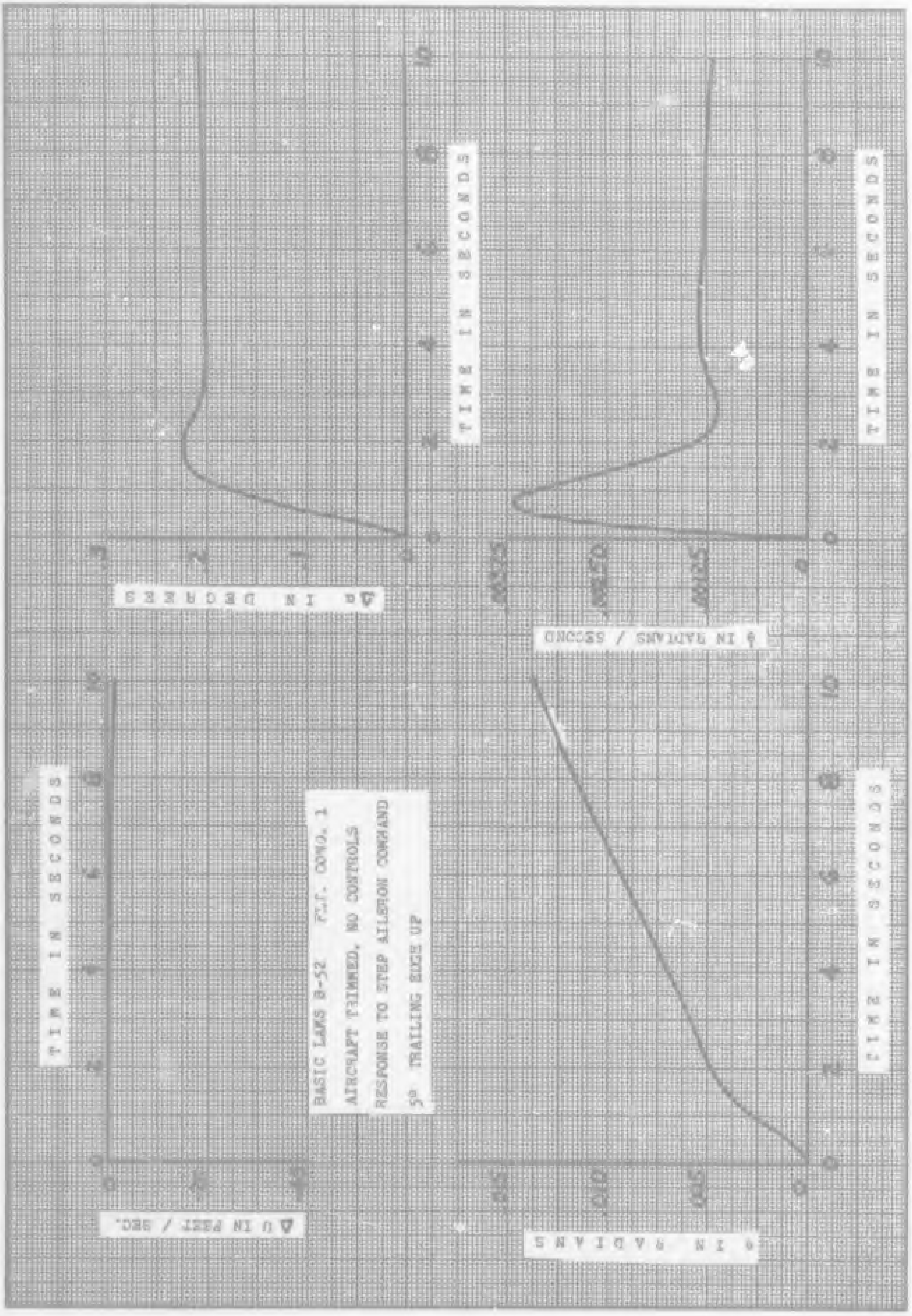


Fig. 6-5. Six Degree of Freedom Aileron Response

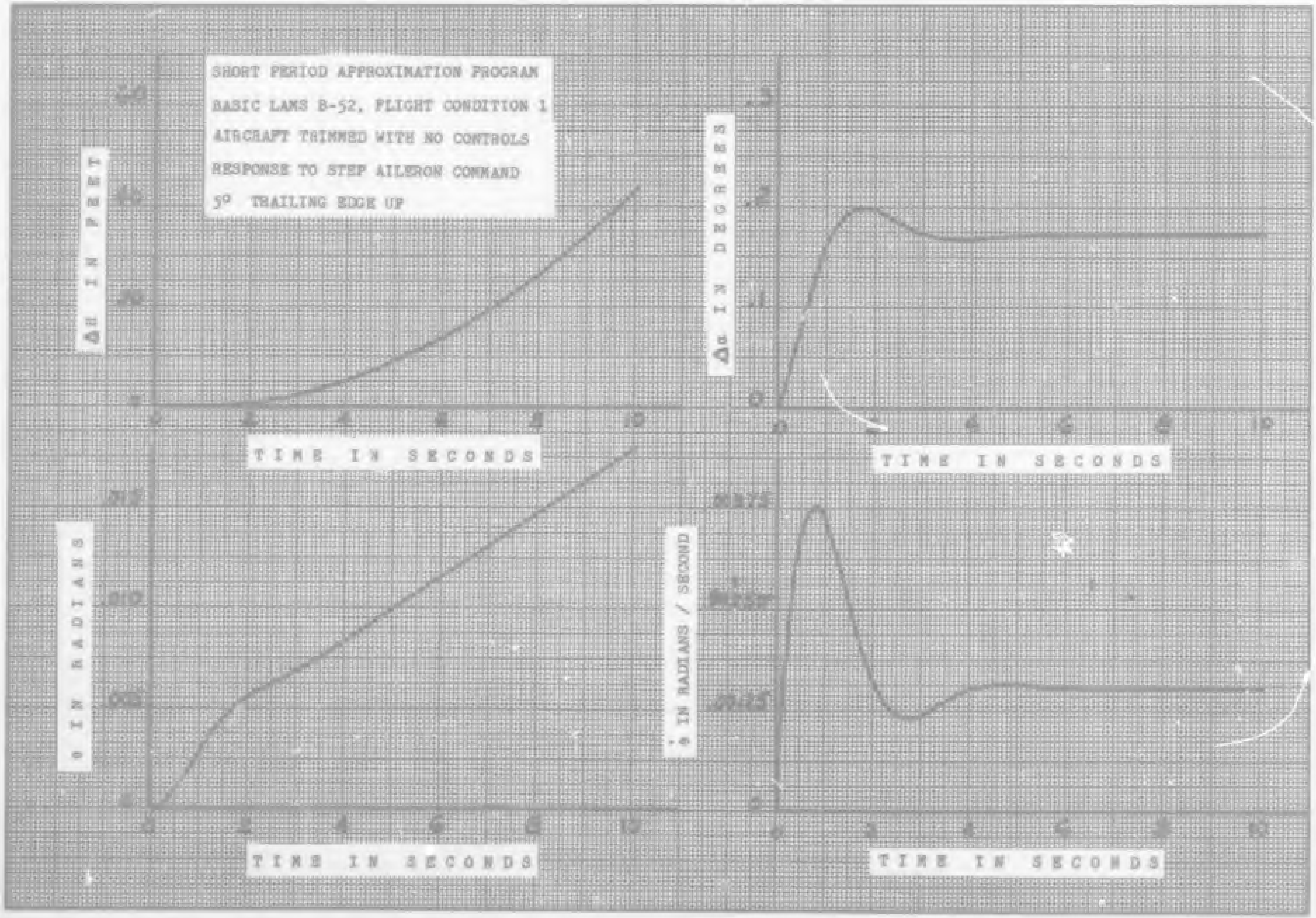


Fig. 6-6. Two Degree of Freedom Aileron Response

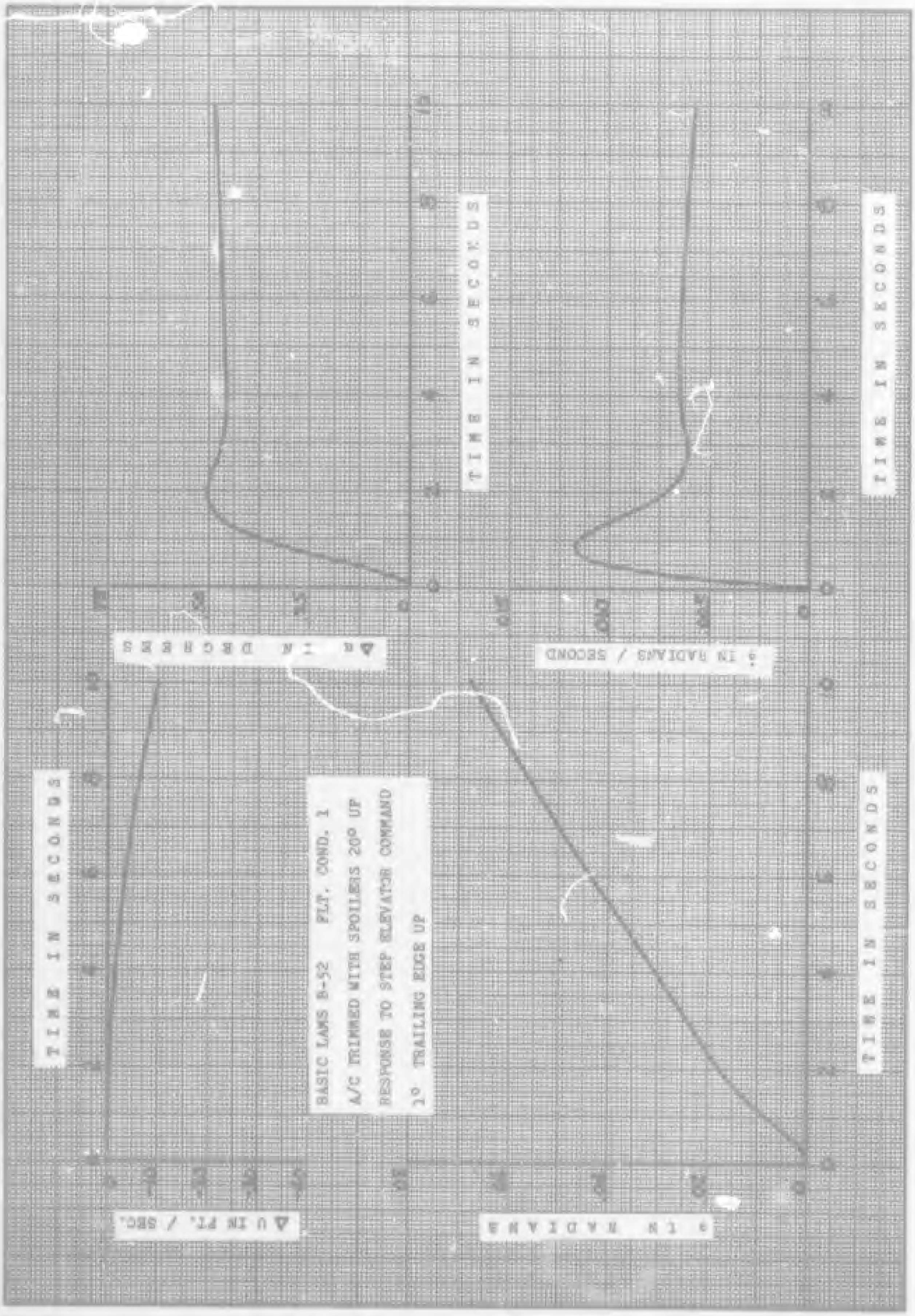


Fig. 6-7. SDF Elevator Response with Spoilers Up

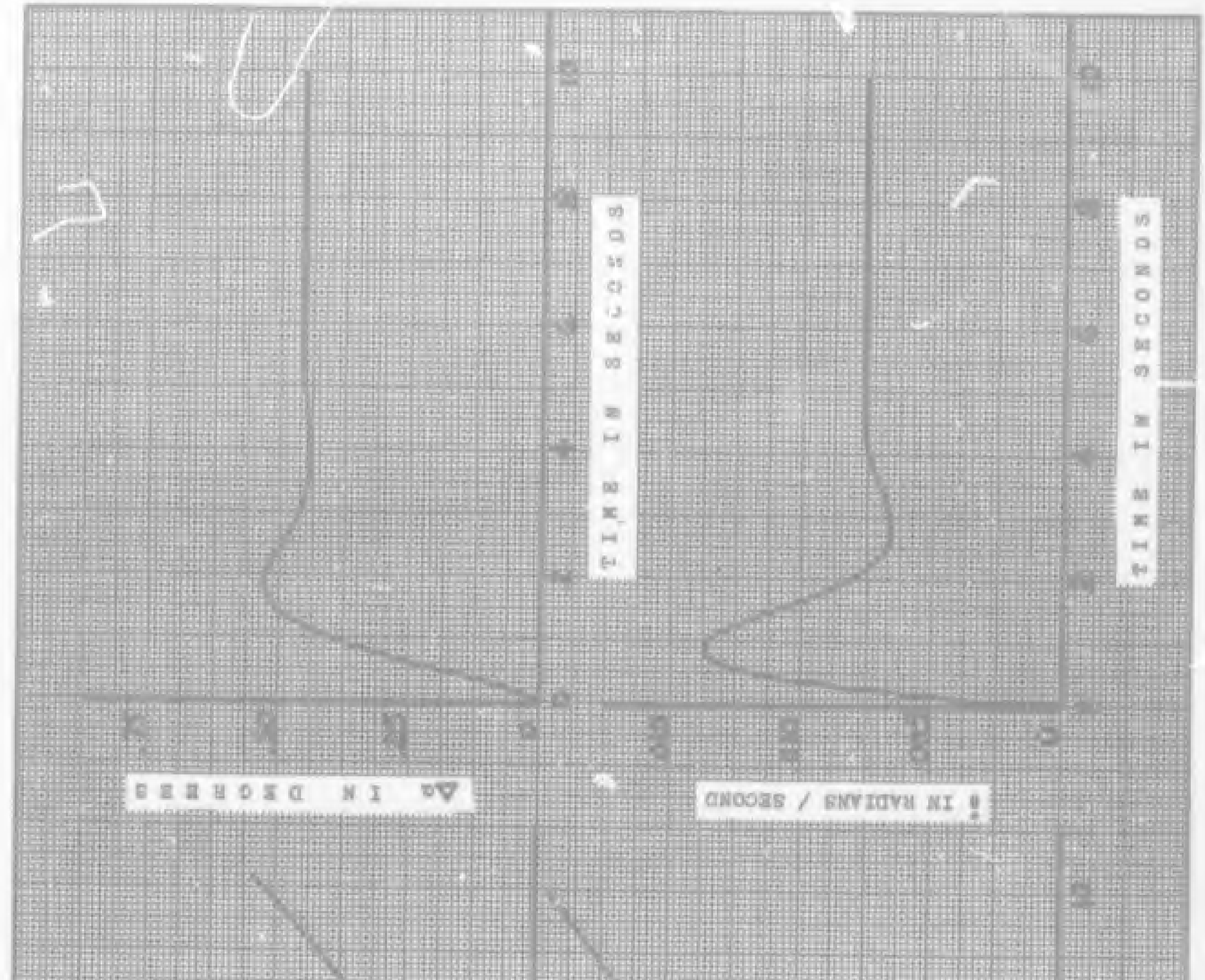


Fig. 6-8. TDF Elevator Response with Spoilers Up

The parameters recorded from the TDF (two degree of freedom) simulation are the same except that ΔH (change in altitude) replaces ΔU. Similar responses in flight conditions two and three, from the six degree of freedom simulation only, are found in Appendix D.

The TDF simulation is based on the aircraft's short period approximation (Ref 3:42-43) which means that airspeed is assumed constant, eliminating the phugoid mode, and lateral motion is completely ignored. A study of the paired figures shows that the principle inaccuracy of the TDF simulation is a too-large pitch angle. However, the Direct Lift Controller uses the elevator to cancel the pitching moment generated by the spoilers and ailerons, so this problem is largely overcome.

In reality, constant airspeed is desireable in the flight conditions tested: terrain following, landing approach, and refueling. Figures 6-32 through 6-34 show that the pilots were able to control airspeed very closely (plus or minus five knots average), and that the DLC aircraft's speed was controlled significantly better than that of the basic model in flight conditions two and three. Thus, the TDF simulation's assumptions are well justified, and its accuracy is acceptable in the preliminary phases of DLC design.

In this thesis, the digital optimization programs were based on the short period approximation and the TDF analog simulation was used to test the resulting controllers and provide a basis for manipulating the Q and R matrices (See pages 42-50). The controllers so selected were used without change during the SDF simulation, and gave good results.

The paired figures 6-9/6-10 and 6-12/6-13 show a "flight" comparison of the SDF and TDF simulations of the basic aircraft. Note that the TDF simulation is much easier to "fly" than the SDF. Pitch angle fluctuations in the SDF were partially due to an over-sensitive stick. This problem was corrected later.

Figures 6-14 and 6-15 are not directly comparable since the controllers and servo time constants used were slightly different. (See Chapter II for listings of controller feedbacks.) However, these DLC results show a much closer correlation between the TDF and SDF simulations than the basic aircraft. In the important θ (pitch angle) channel, the results of the DLC simulations are identical, demonstrating the value of the TDF simulation in preliminary design work.

Results of Direct Lift Control

A comparison between the DLC and basic aircraft shows many interesting results. The most significant of these are shown in Figures 6-32 through 6-34, pages 155 to 157. Although these figures are for all three flight conditions, the results from any one flight condition carry over to the other flight conditions. The important parameters, such as θ (pitch angle), N_L (normal acceleration), and Δ U (change in airspeed), and α_G (turbulence) are recorded in the bar graph form. Each figure represents how well each pilot did compared to all others as well as how the "average" pilot did. In all cases, θ was reduced in the DLC aircraft, with

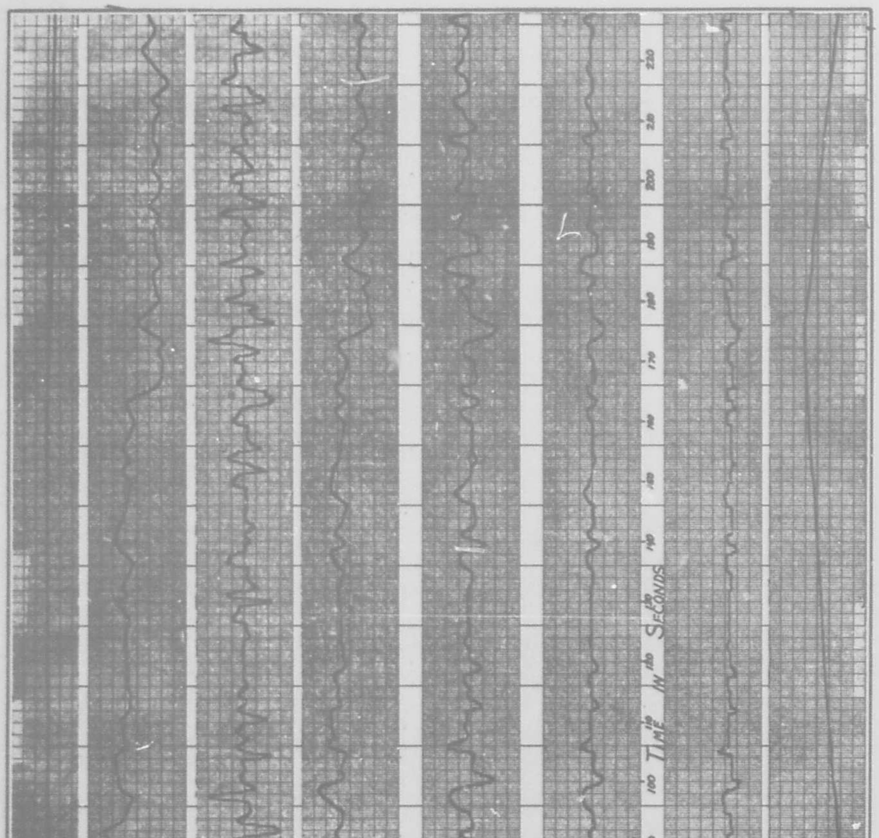


Fig. 6-9. 2-GGC-68-TB (Climb and Descend, Without Turbulence)

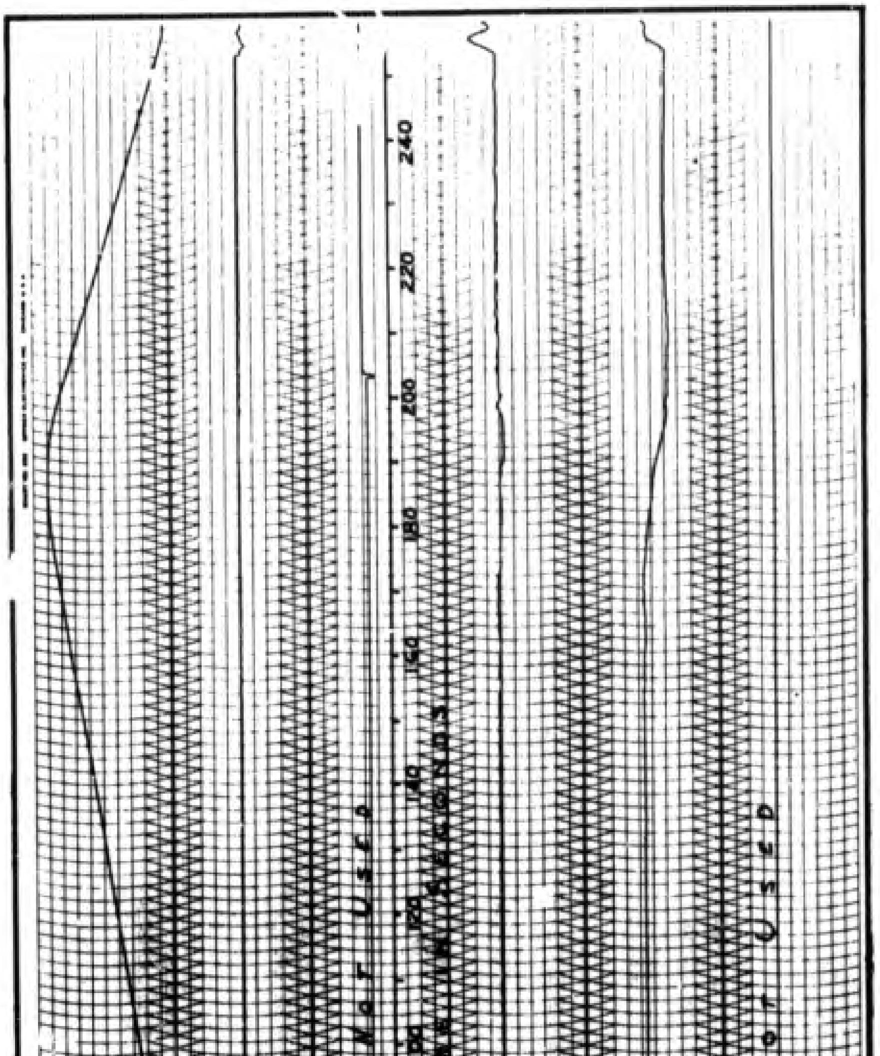


Fig. 6-10. 1-GGC-68-TB (Climb and Descend, No Turbulence, TDF)

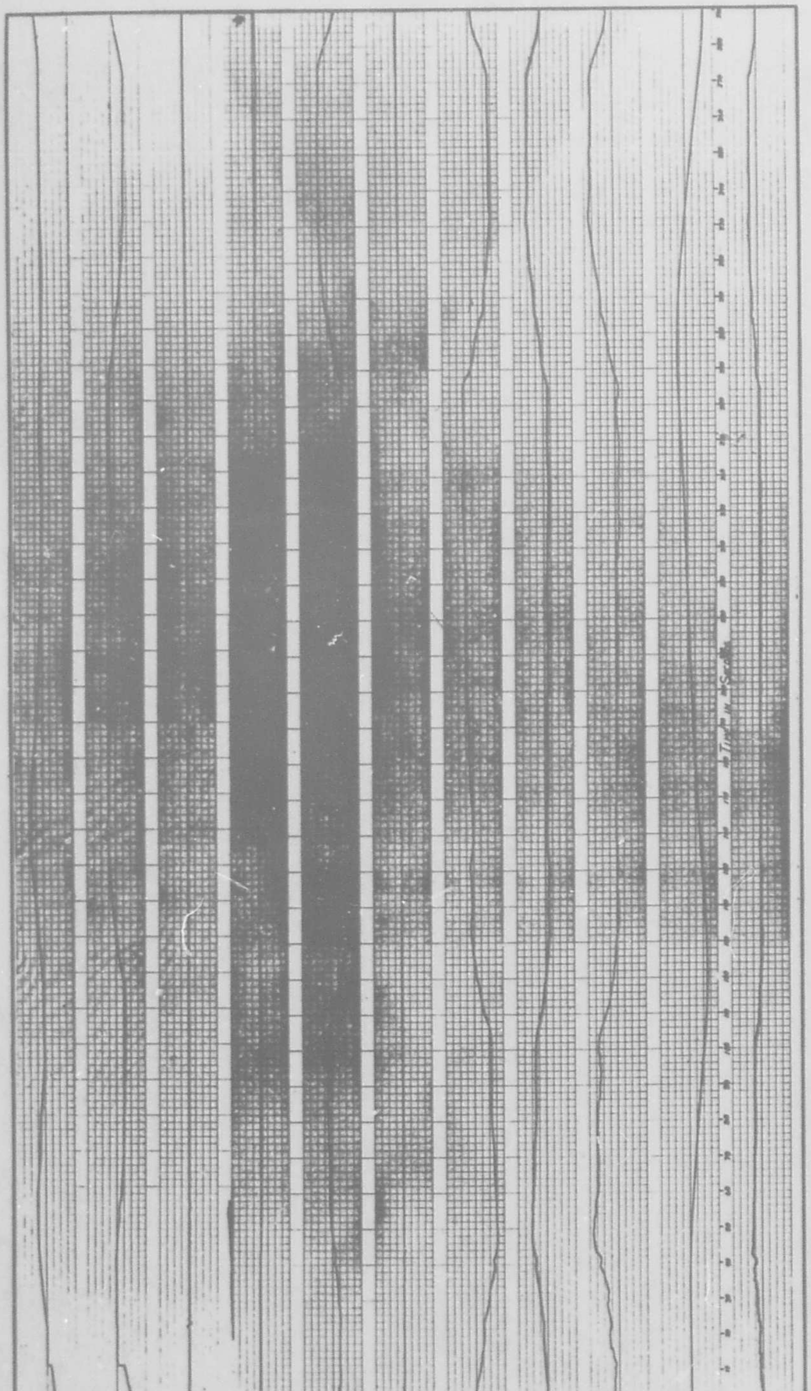


Fig. 6-11. 2-GGC-68-TC (Controller 41, Climb and Descend, Without Turbulence)

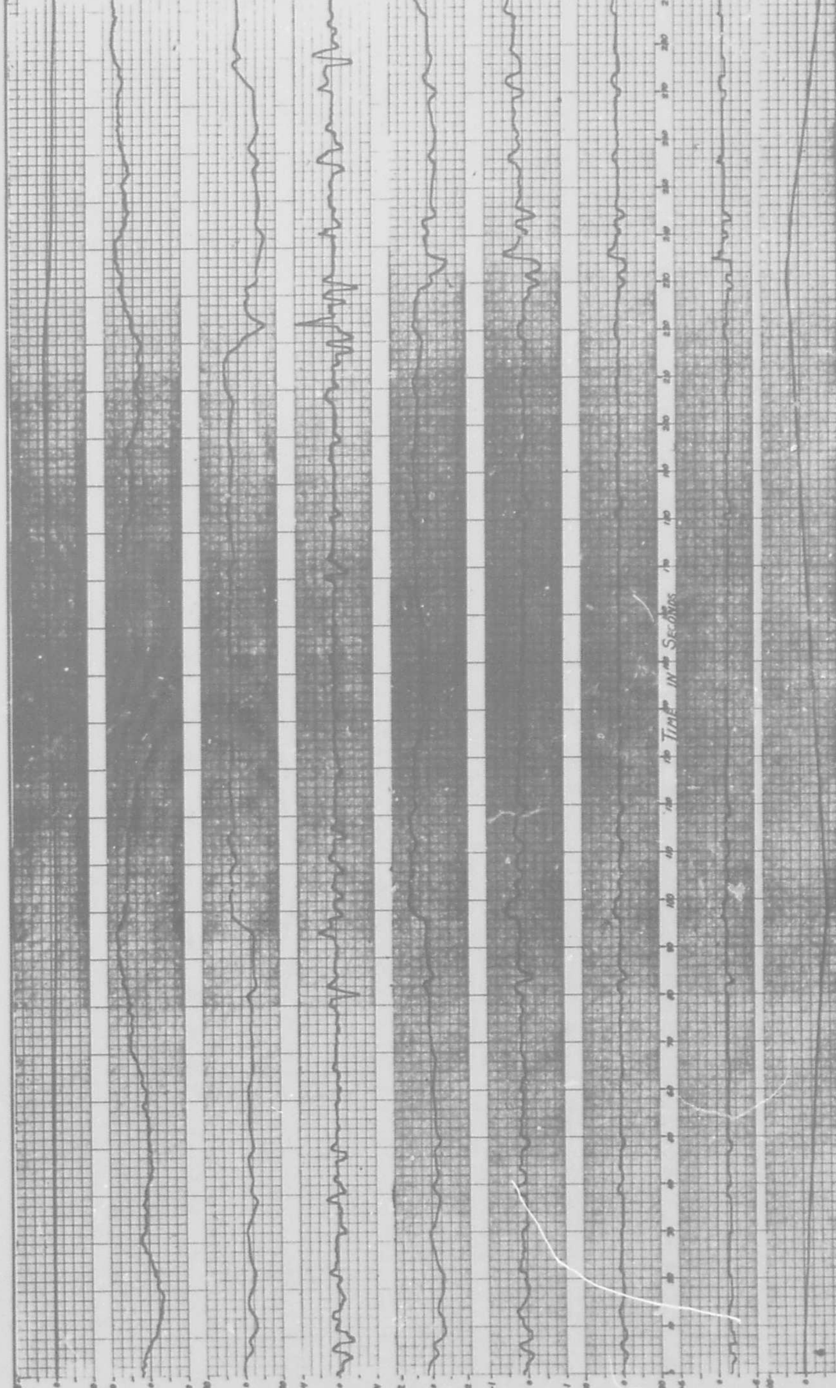


Fig. 6-12. 2-GGC-68-TB (Climb and Descend With Turbulence)

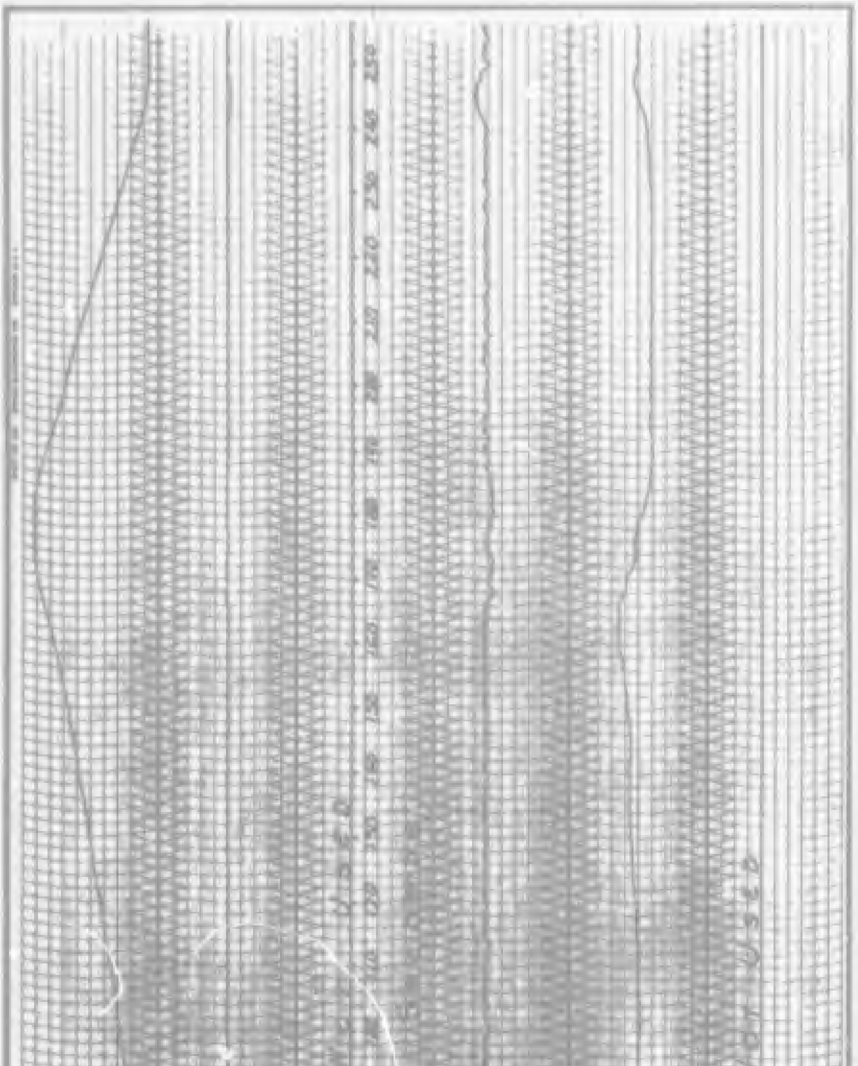


Fig. 6-13. 1-GGC-68-TB (Climb and Descend, With Turbulence, TDP)

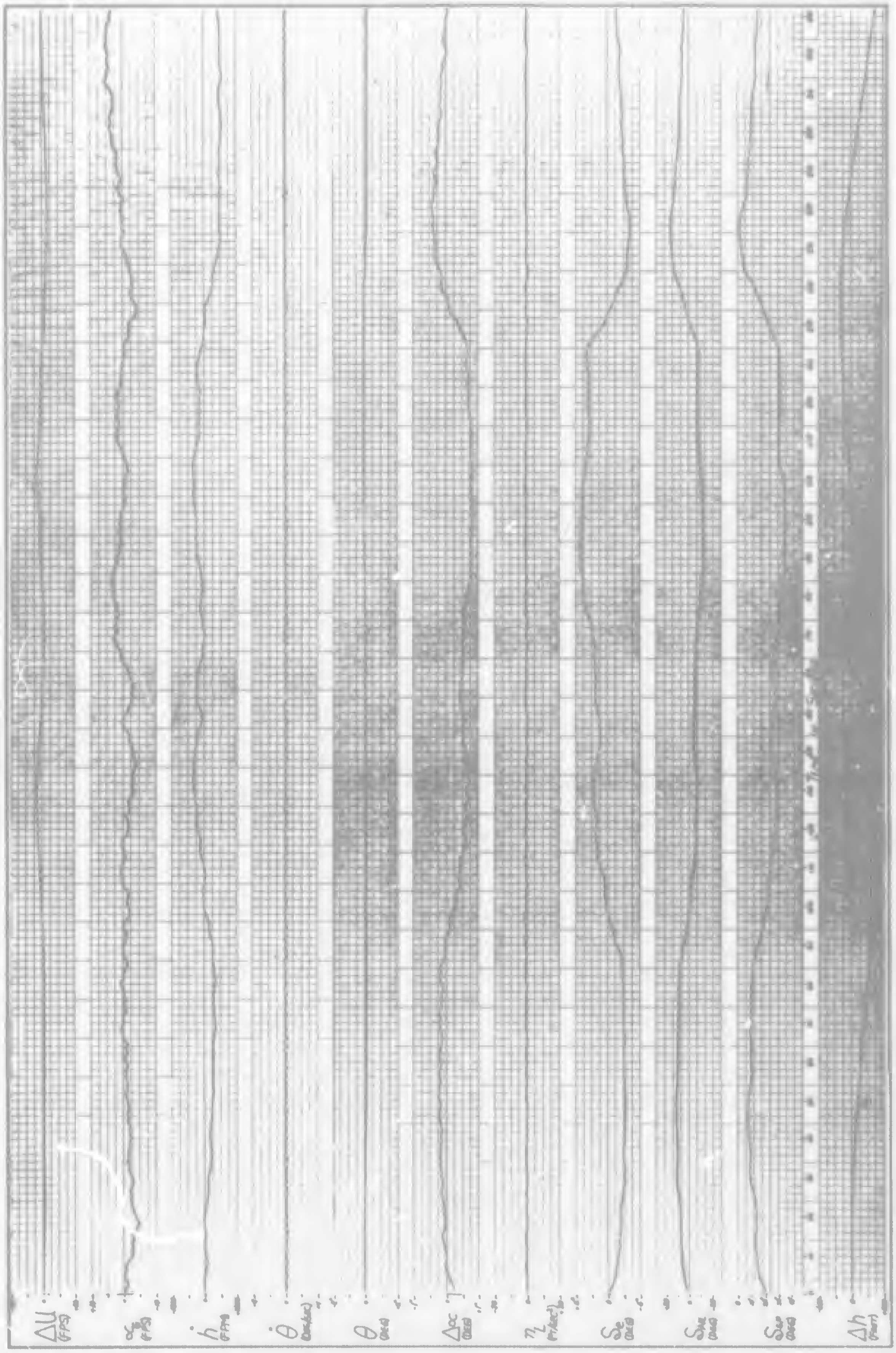


Fig. 6-14. 2-GGC-68-TC (Controller 41, Climb and Descend, With Turbulence)

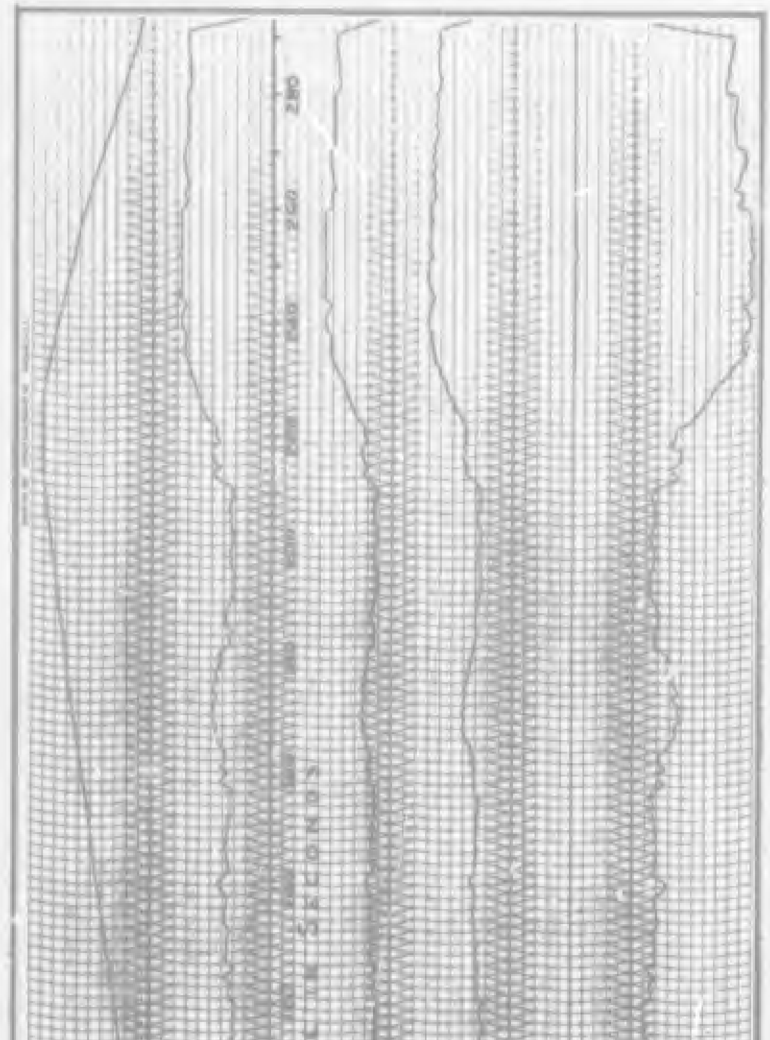


Fig. 6-15. 1-GGC-68-TC (Climb and Descend, With Turbulence Controller 61, One Second Time Constants)

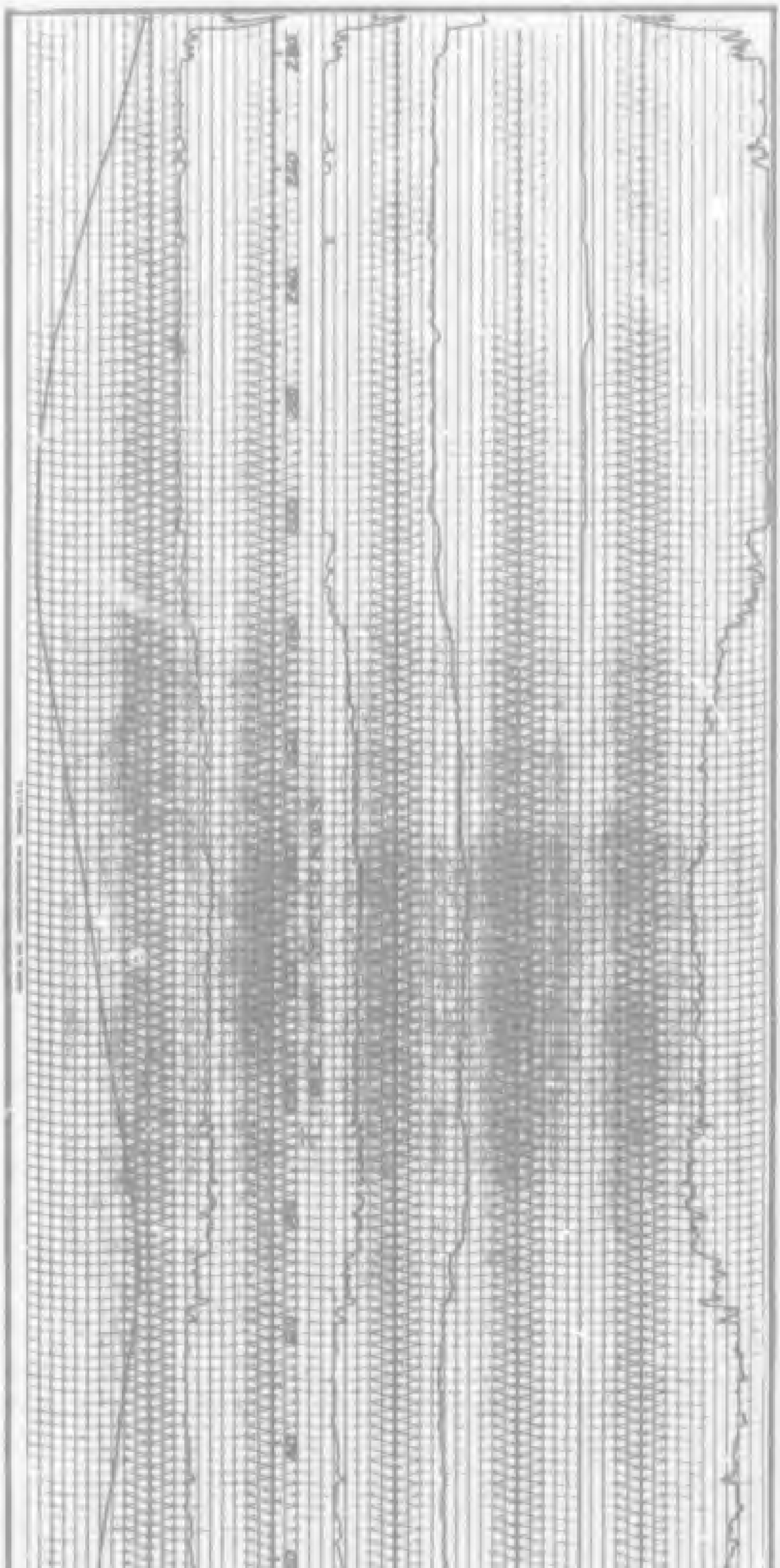


Fig. 6-16. 1-GGC-68-TC (Climb and Descend, With Turbulence, Controller 61, 0.1 Second Time Constants, TDF)

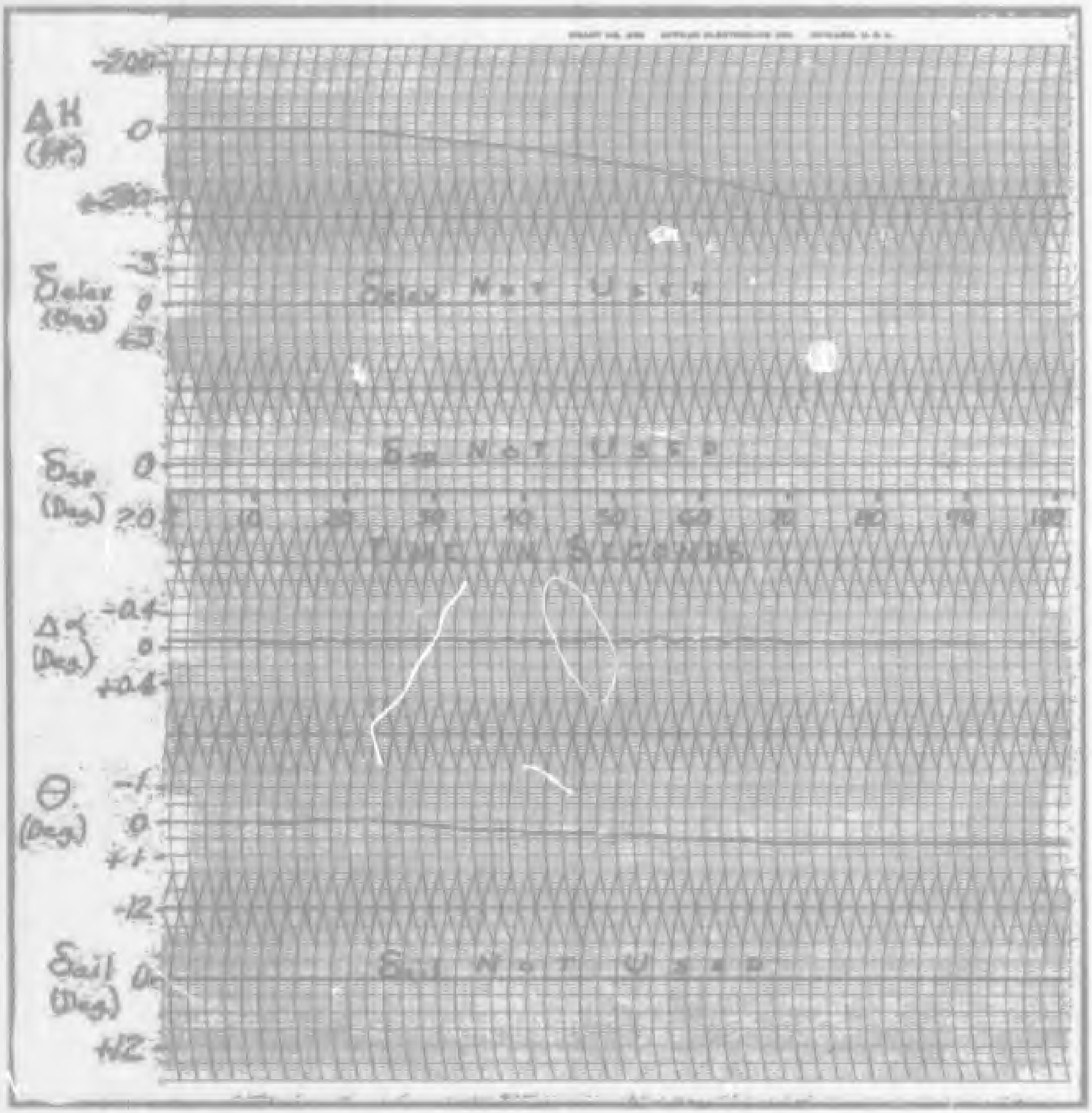


Fig. 6-17. Flight Condition One, Basic Aircraft With Turbulence, Hands Off Controls, TDF Simulation

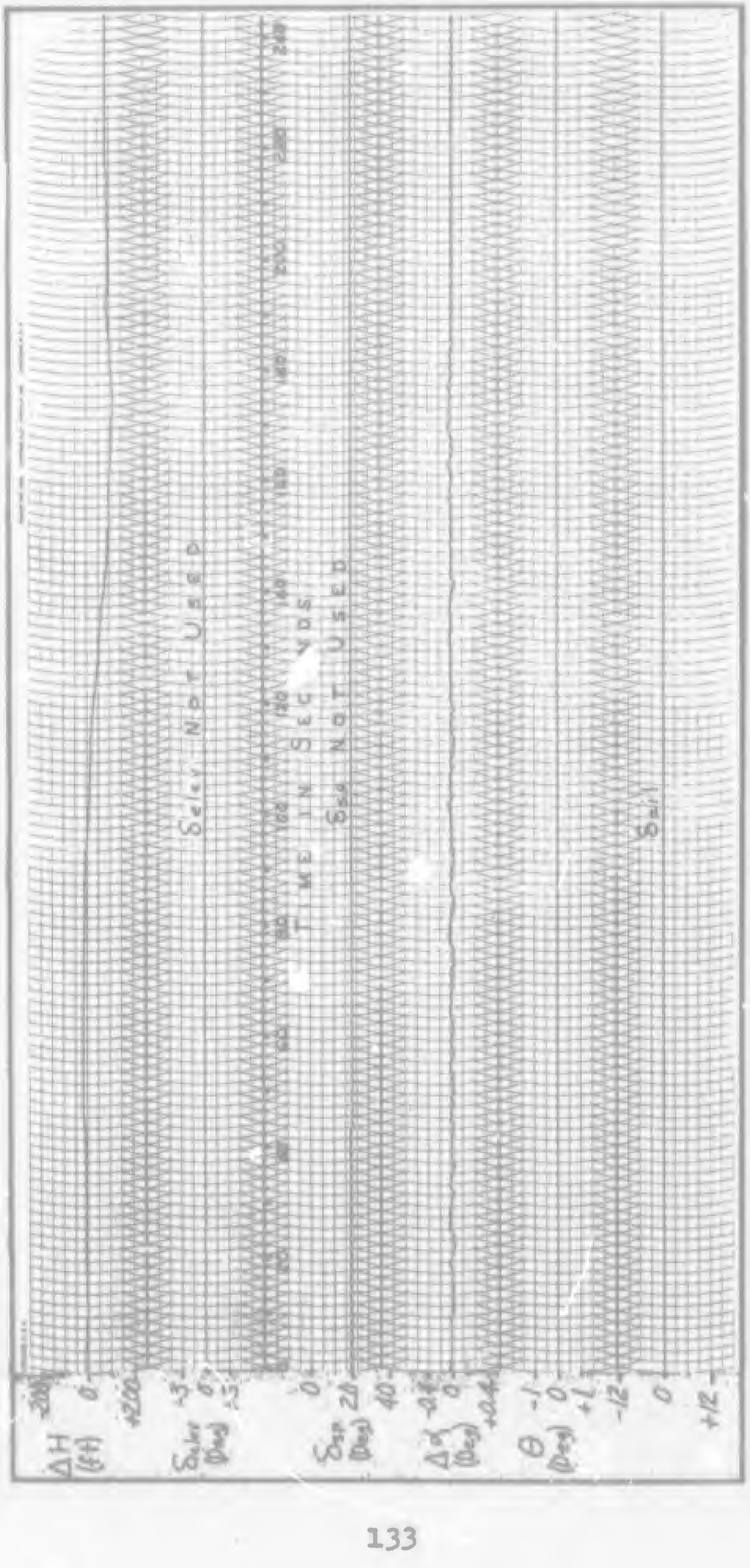


Fig. 6-18. Flight Condition One, Controller 61, With Turbulence, Hands Off Controls, TDF

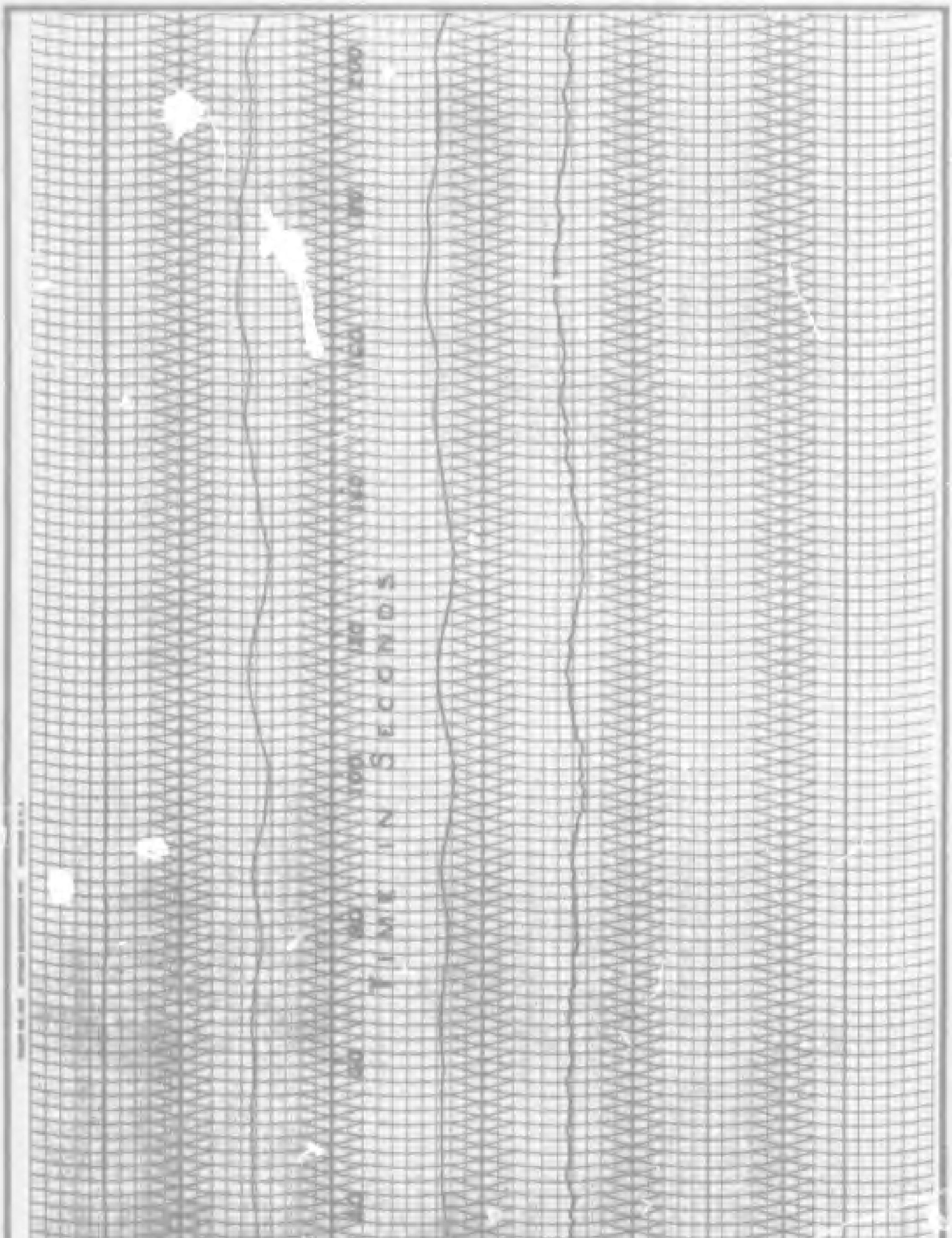


Fig. 6-19. Flight Condition One, Controller 61 With Altitude Hold, Turbulence, Hands Off Controls, TDF Simulation

and without turbulence. The percent reduction is shown in Table VI-1, page 158.

Theta could have been reduced even further had the washout circuit in the elevator controller been eliminated (See page 149). It should also be noted that the last climb in all of the flight conditions was devised to use the maximum "direct lift" available. Any additional rate of climb would of necessity cause some pitching of the aircraft. In other words, the rate of climb or descent without changing pitch attitude is limited by the aerodynamic characteristic of the DLC surfaces.

All of the flight tasks were devised to hold θ as close as possible to zero. With slightly different mechanization, θ could be held to any constant value. For instance, if one wanted to fly a glide path, it would be advantageous to trim the aircraft for the glide path and with use of DLC, make corrections back to the glide path. This can be done easily by feeding back a $\Delta\theta$ in place of θ. Controller gains would not be changed. Recording this $\Delta\theta$ would have given the same type of results as seen above, and the control of the pitch angle about the trim θ would have been similar to that shown on Fig. 6-21 or Fig. 6-23.

After studying Figs. 6-32 to 6-34, it is apparent that N_L (normal acceleration) is also reduced in the DLC aircraft when compared to the basic aircraft. Again, it was less for all the pilots in all flight conditions, with or without turbulence. The average reduction in N_L varied from 52% in flight condition three to 65% in flight condition one, and

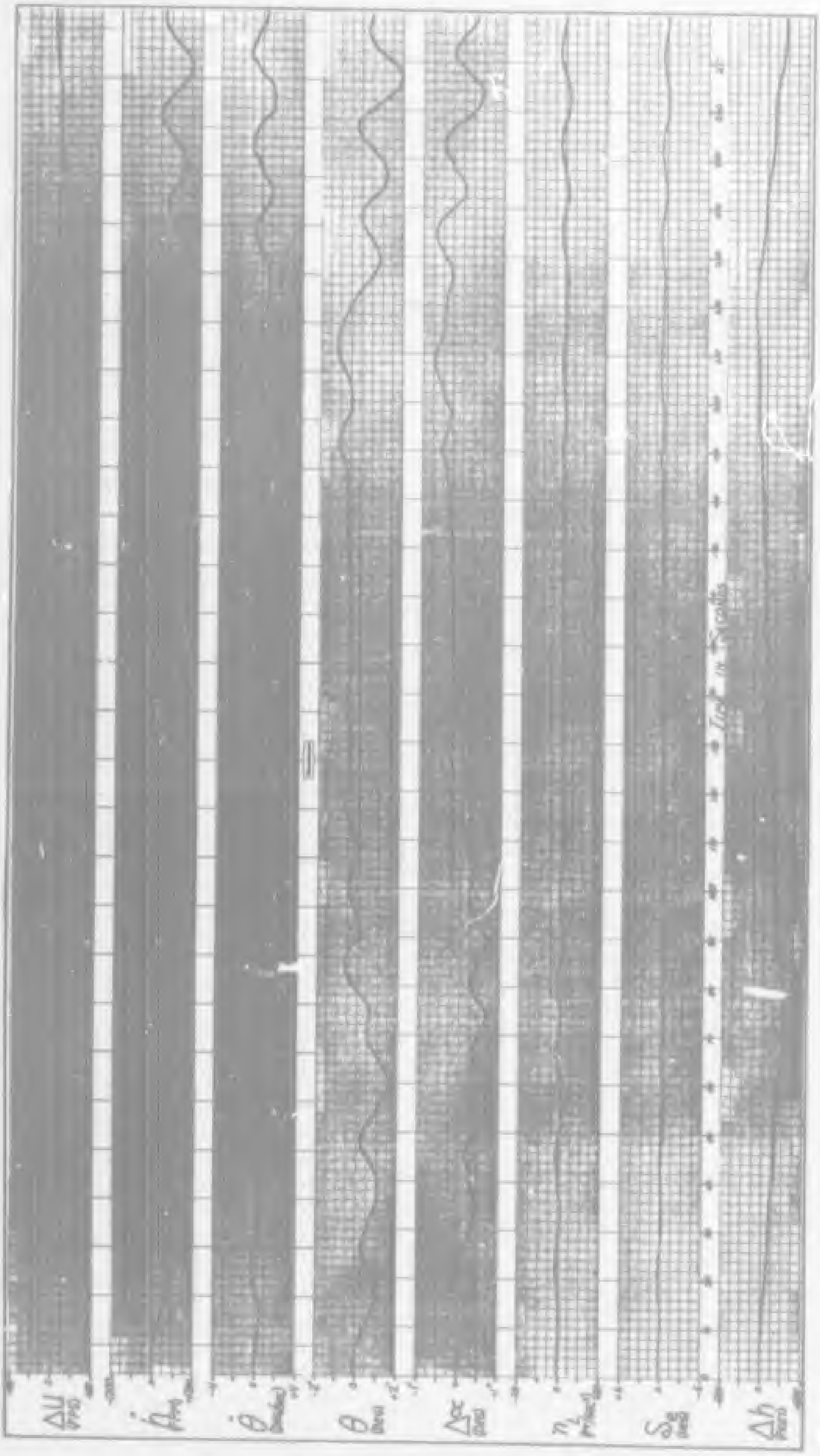


Fig. 6-20. 4-GGC-68-LB (Climb and Descend, No Turbulence)

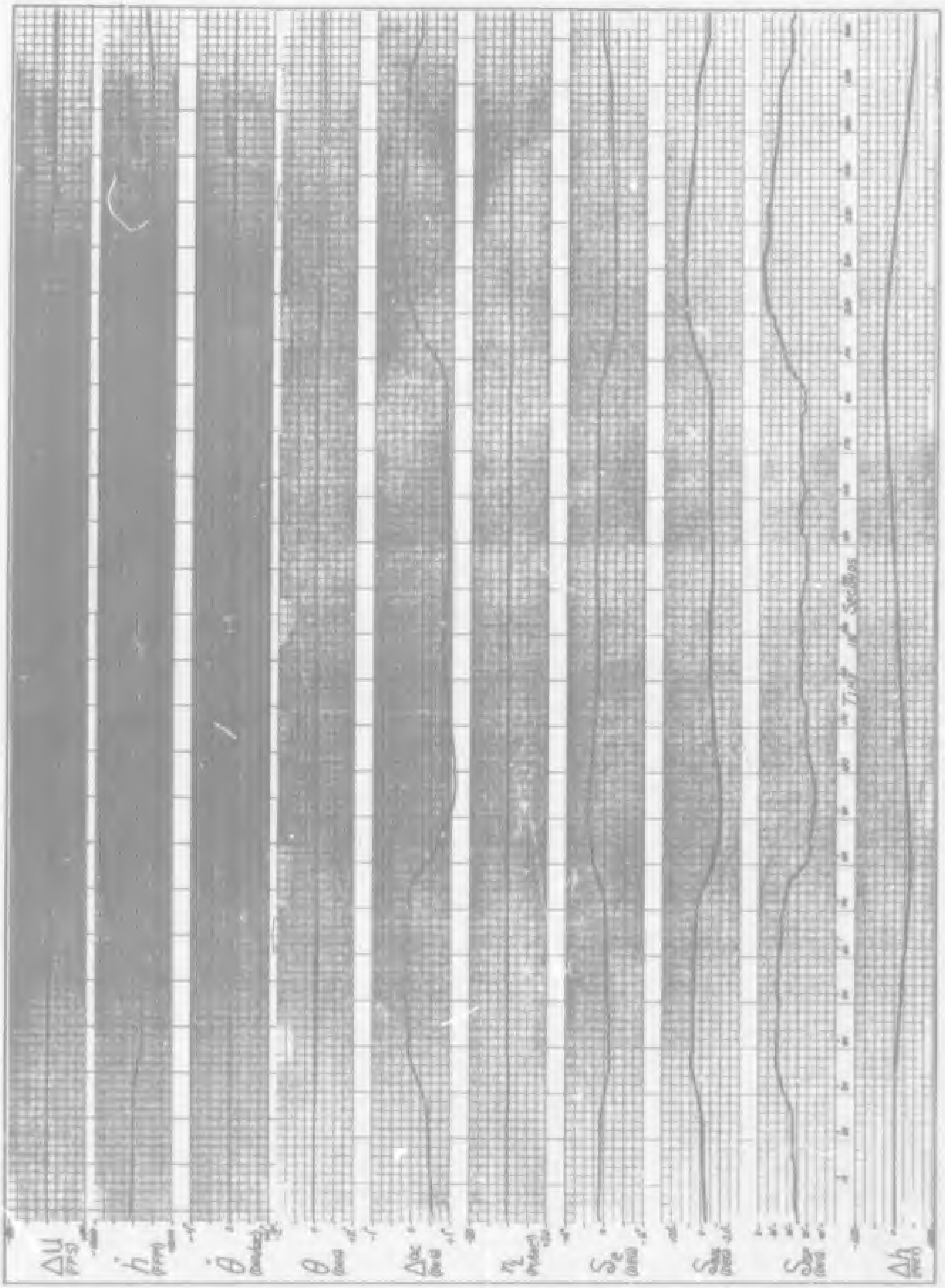


Fig. 6-21. 4-GGC-68-LC (Controller 42, Climb and Descend, Without Turbulence)

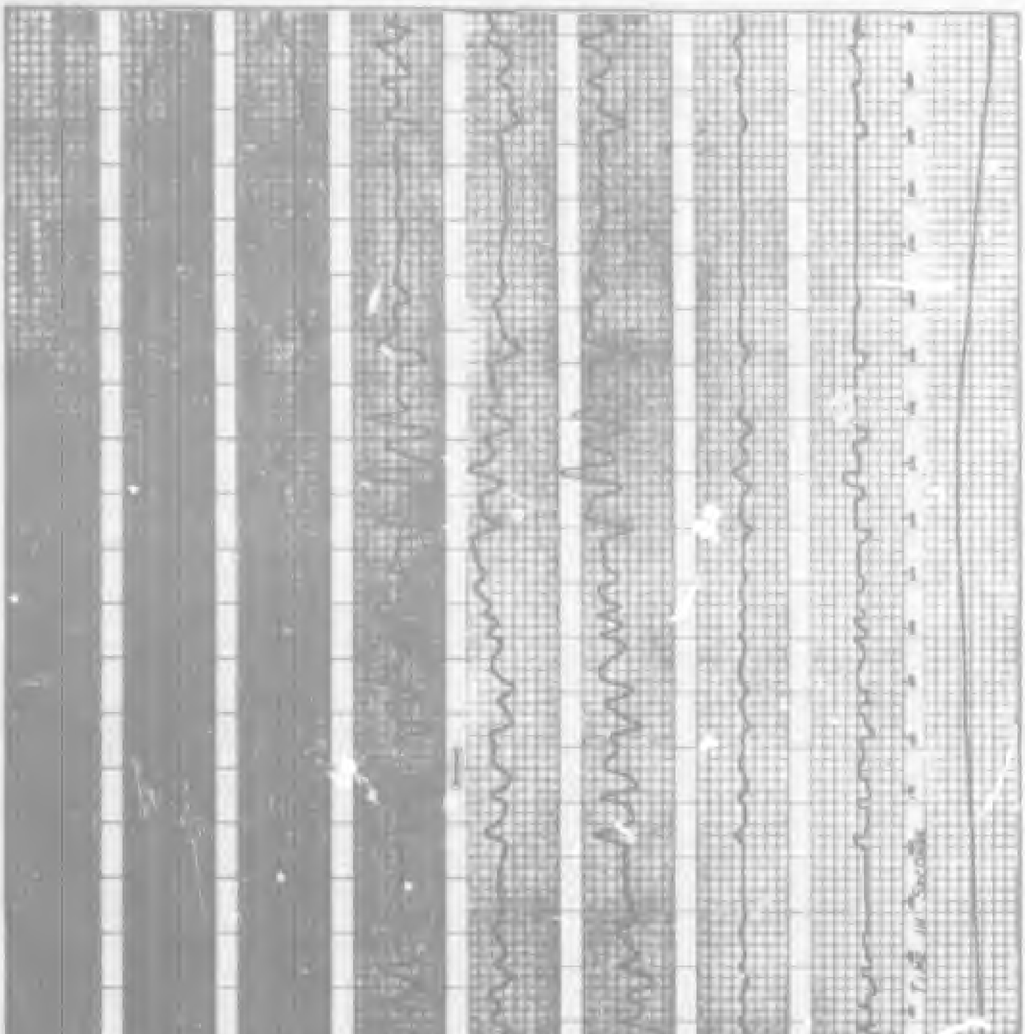


Fig. 6-22. 1-GGC-68-1B (Climb and Descend, With Turbulence)

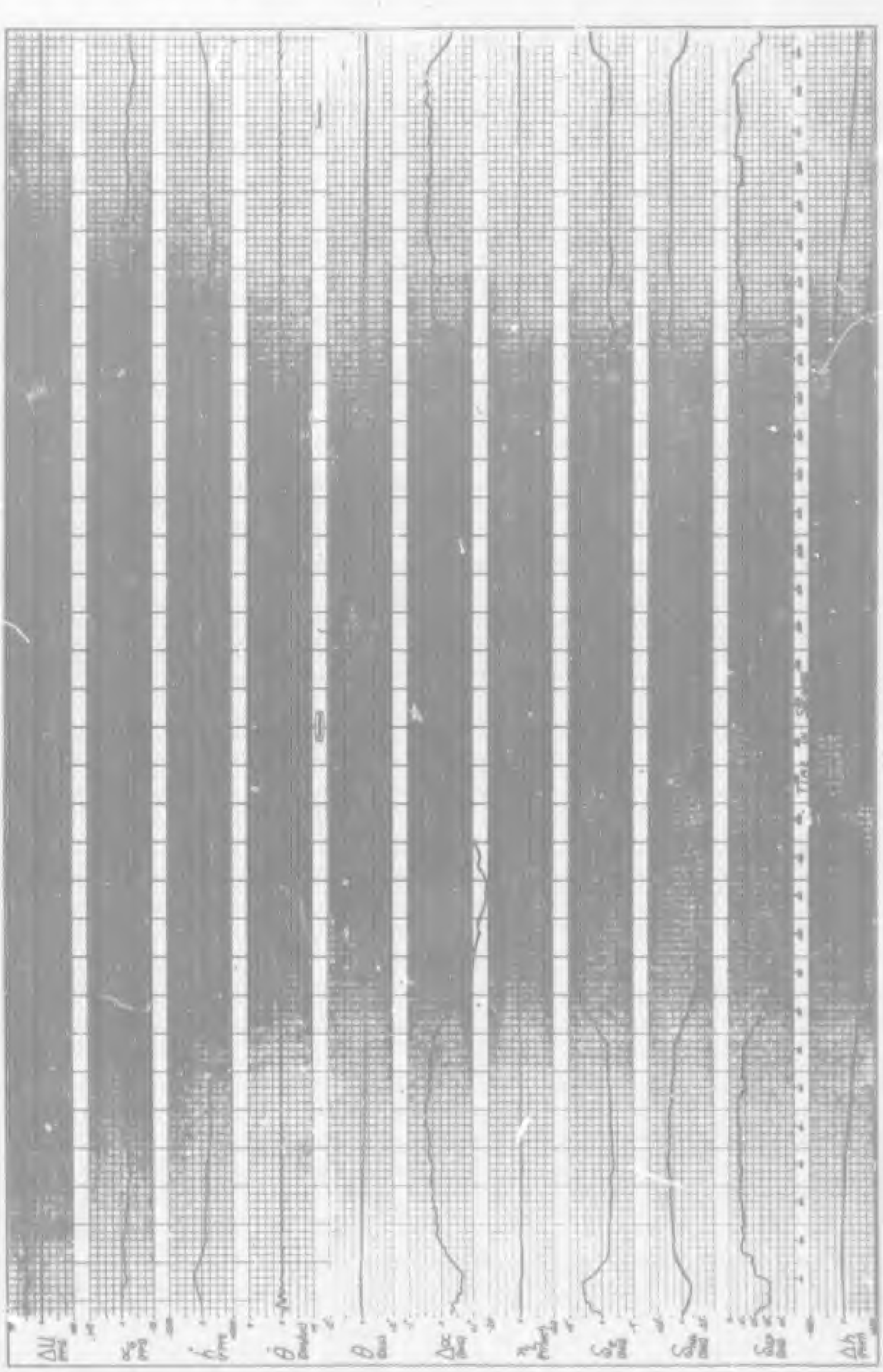


Fig. 6-23. 1-GGC-68-LC (Controller 42, Climb and Descend, With Turbulence)

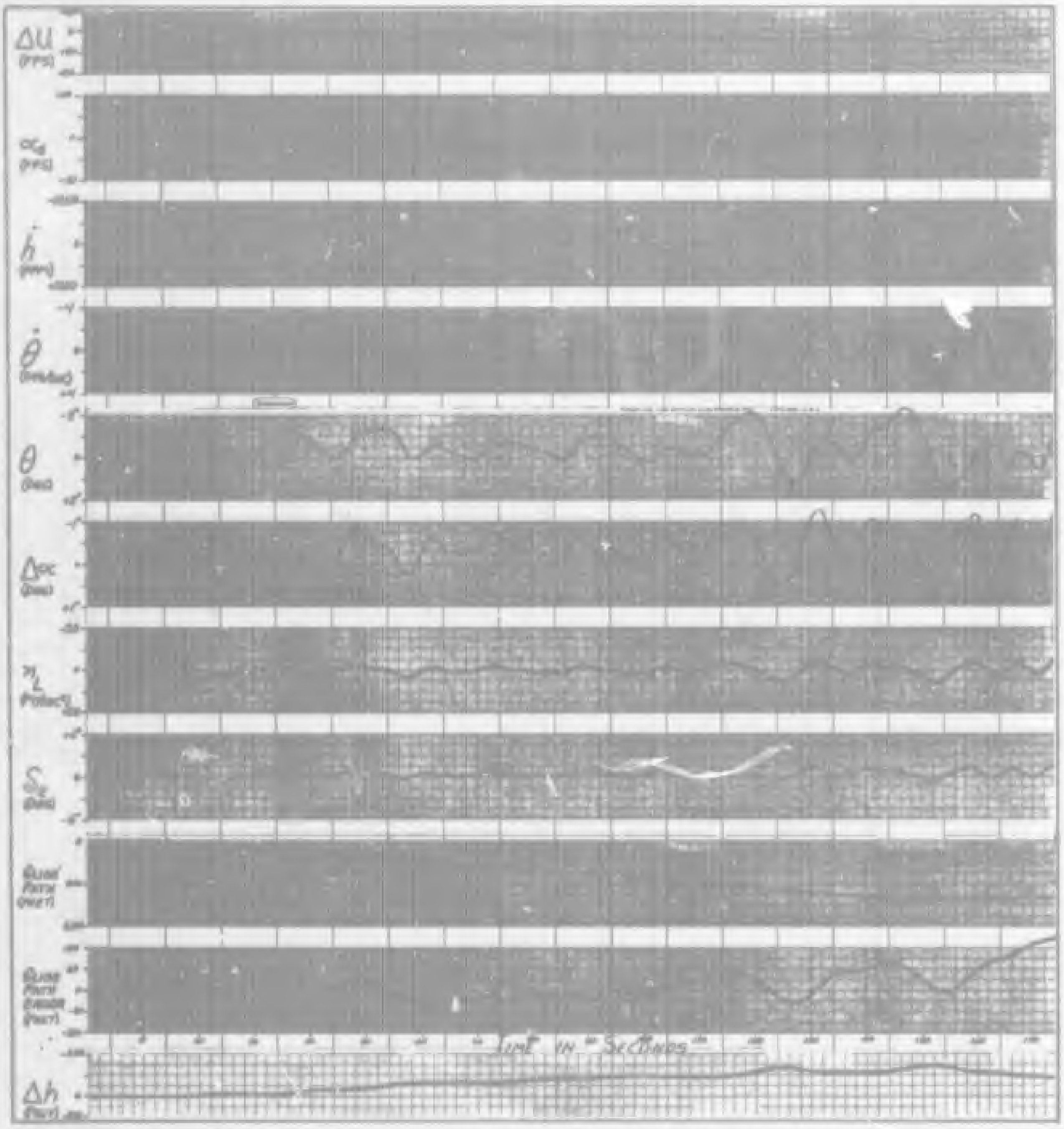


Fig. 6-24. 7-GGC-68-LB (Glideslope With Turbulence)

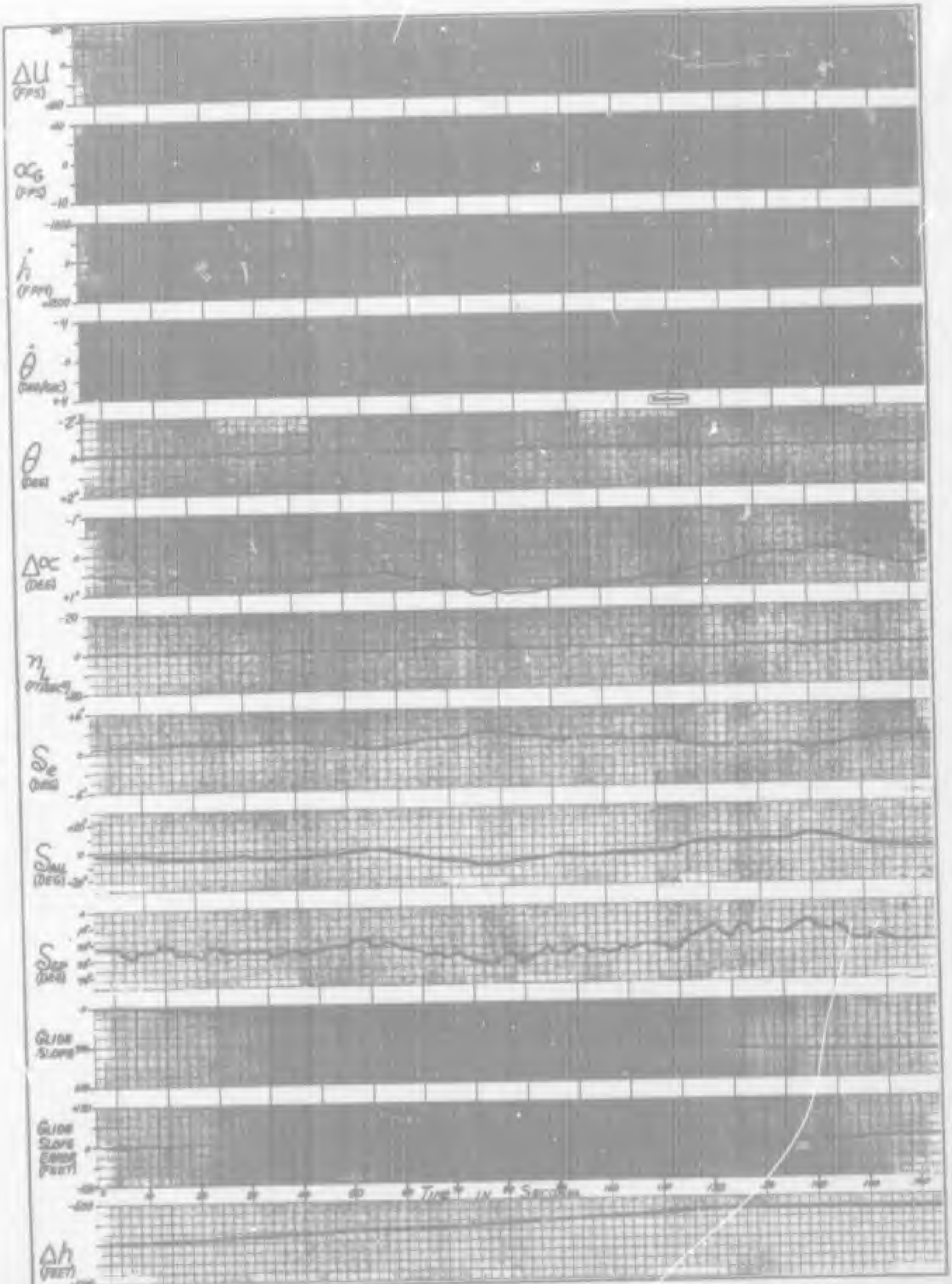


Fig. 6-25. 7-GGC-68-LC (Controller 42, Glideslope, With Turbulence)

is shown in Table VI-1, page 158. As discussed in Chapter II, N_L can be reduced further by properly weighting the Q matrix.

The next parameter to be considered in evaluating DLC is Δ U (change in airspeed) caused while performing the prescribed tasks. The pilots were instructed to maintain constant airspeed. Figure 6-32 shows that while Δ U is relatively small for both the DLC aircraft and the basic B-52, airspeed in the basic aircraft was usually controlled better. This is due to a phenomena peculiar to this airplane with spoilers biased up 20°. When a climb was commanded, the increase in airspeed caused by the decreased drag as the spoilers came down was greater than the decrease in airspeed usually caused by the aircraft's rate of climb. The net result was that the plane would actually gain airspeed in a constant throttle climb and in a like manner lose airspeed in a constant throttle descent. Although this is contrary to popular opinion, the pilots soon accepted it as fact, and were able to control the airspeed better in the DLC aircraft than in the basic aircraft. This is shown in Table VI-1, page 158, as well as in Figs. 6-33 and 6-34.

It was originally planned to have the pilots fly a simulated terrain displayed on the horizontal needle, but due to a lack of time and information about the equipment being used, it was not possible to obtain enough data to evaluate this maneuver. As found in the pilot's comments in Appendix I, the distance that the needle moved vertically as well as the rate at which it moved proved unsatisfactory for pilot evaluation.

These difficulties were soon corrected and a glide slope was presented in flight condition two for evaluation. Although the results shown in Figure 6-33 are not conclusive, it was felt that more time and practice on the part of the pilots would have decreased the glide slope error, due to the pilots' positive control of pitch angle and pitch rate. Also the reversal of airspeed changes invariably had some effect on the results.

Sample runs of all flight conditions and all pilots are found in Figures 6-9 to 6-34. They are included here as general information, providing to the reader some insight into the types of data taken as well as what the actual results looked like. It also points out the necessity of recording mean absolute values of important parameters in order to evaluate such a system. The actual MAV numbers used to draw the bar graphs (Figures 6-32 to 6-34) are found in Appendix H.

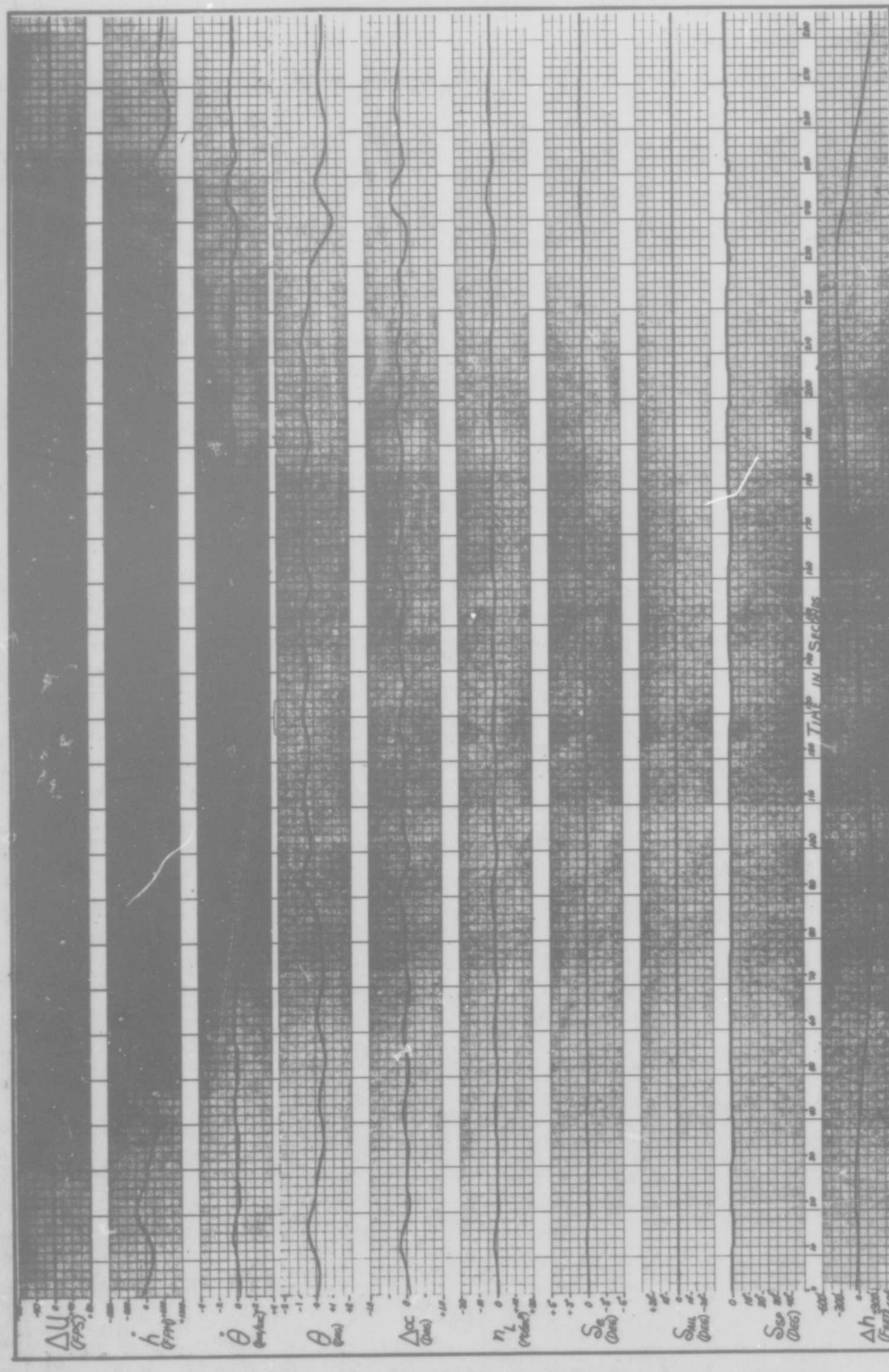


Fig. 6-26. 3-GGC-68-RB (Climb and Descend, No Turbulence)

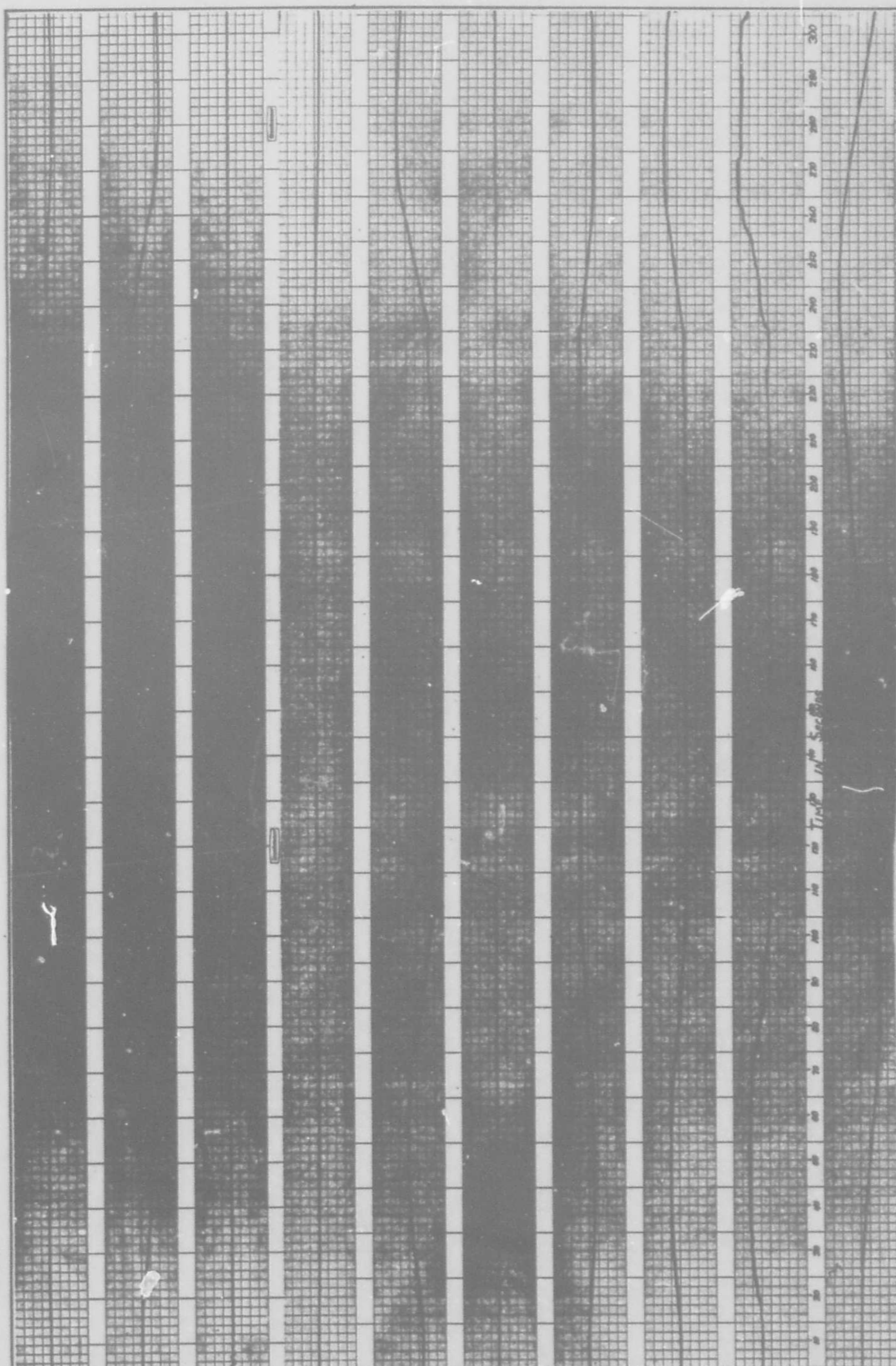


Fig. 6-27. 3-GGC-68-RC (Controller 63, Climb and Descend, Without Turbulence)

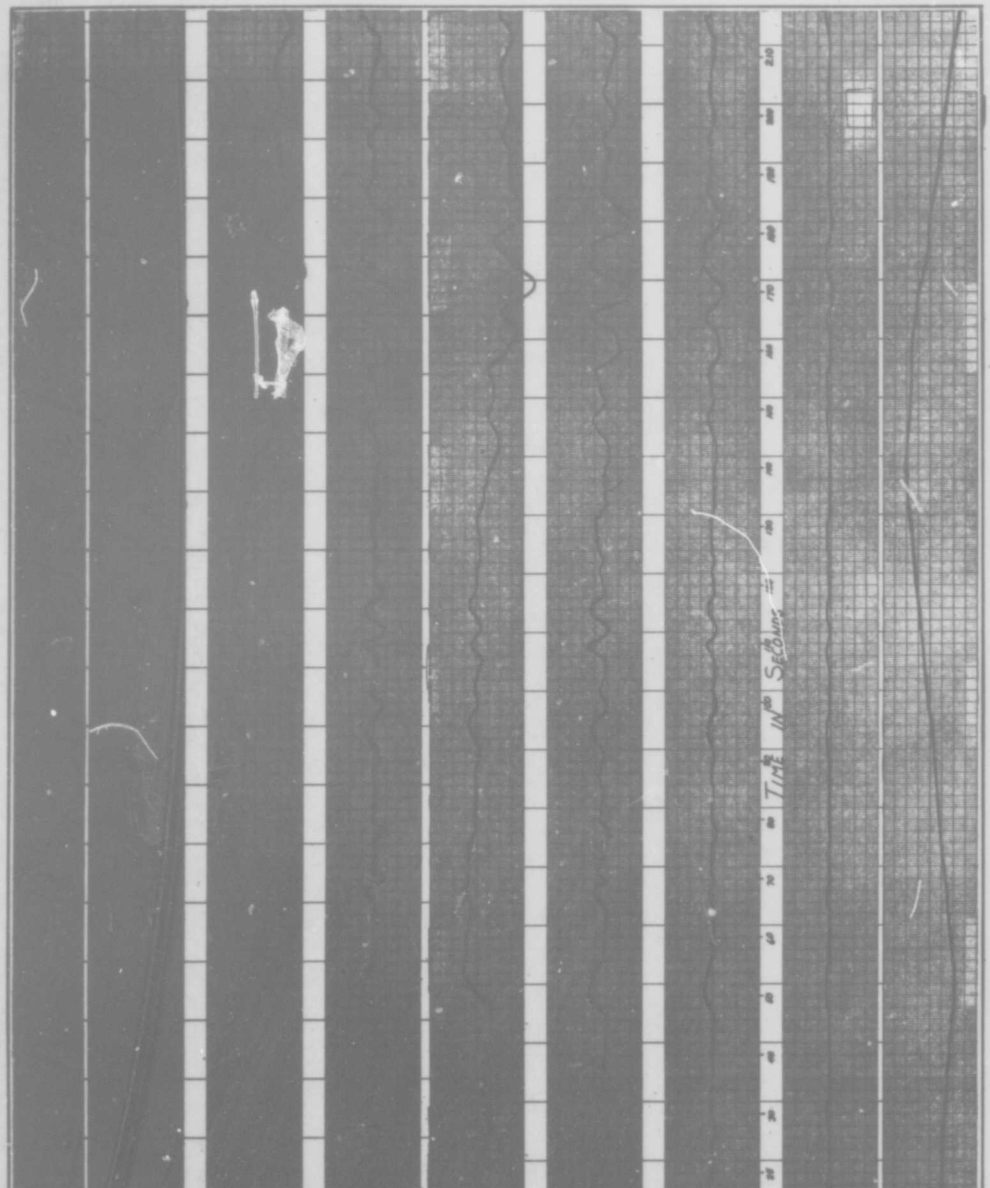


Fig. 6-28. 7-GGC-68-RB (Climb and Descend, With Turbulence)

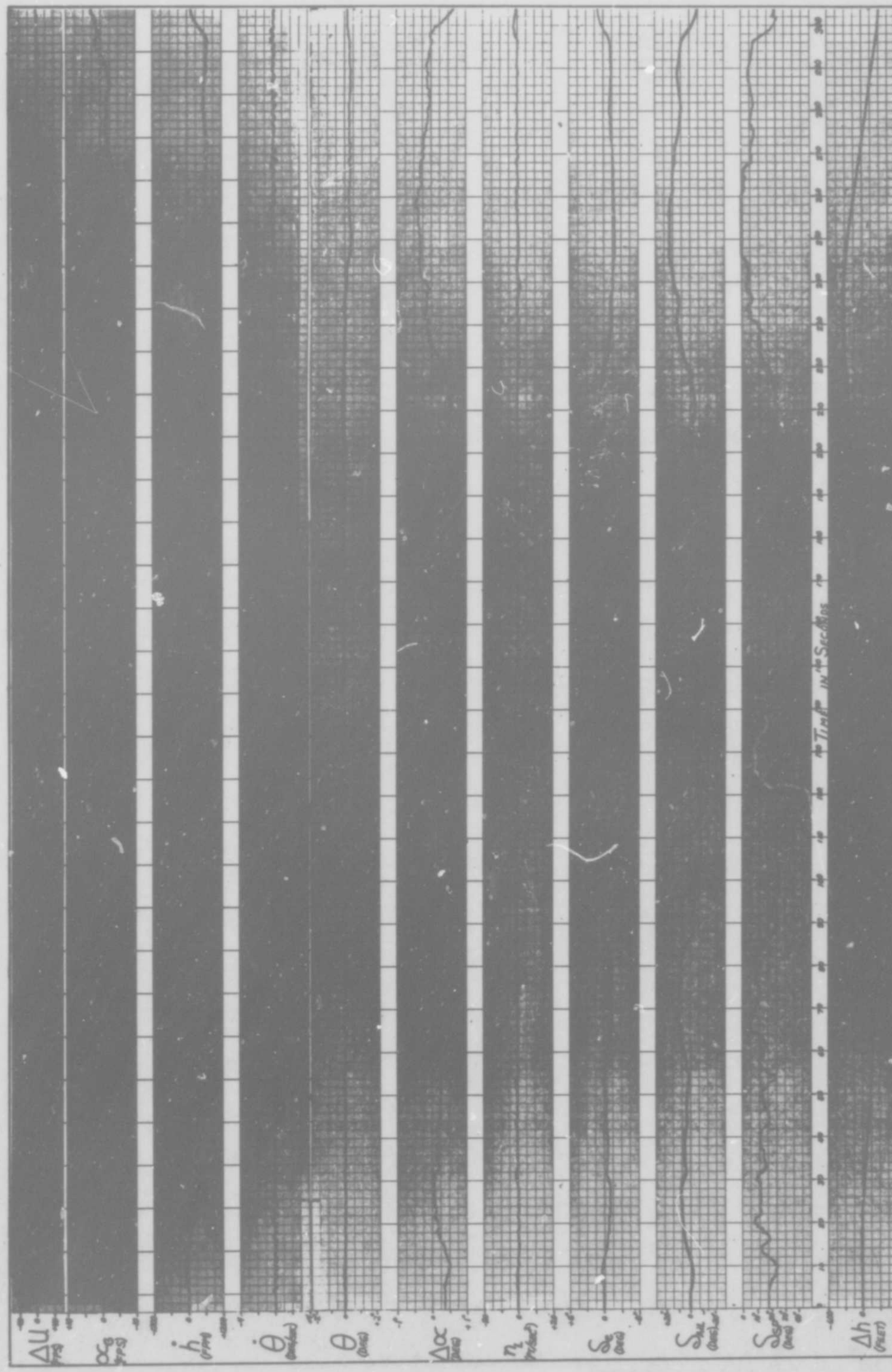


Fig. 6-29. 7-GGC-68-RC (Controller 43, Climb and Descend, With Turbulence)

Effect of Varying Servo Time Constants

The run shown in Fig. 6-14 (SDF) used elevator, aileron, and spoiler servo time constants of four seconds. Fig. 6-15 (TDF) shows a similar flight profile using servo time constants of one second. Fig. 6-16 (TDF) is identical to Fig. 6-15 except that all servo time constants are a tenth of a second. These three figures demonstrate one of the important points of this study: Direct Lift Control does not depend on any particular actuator speed. The only requirement is that the time constants of the various servos be relatively equal. Thus, actuator speeds can be chosen based on considerations such as stability, available hydraulic power, cost, etc. Of course, the DLC mission must also be considered . . . gust alleviation, for instance, can be improved by using faster-acting control surface actuators.

Altitude Hold Mode

Figures 6-17 through 6-19, all from the TDF flight condition one simulation, demonstrate the capability of the altitude hold mode described on page 56. Vertical turbulence averaging three to four feet per second was present in all three runs. With no inputs by the pilot or any controlling device, the basic B-52 quickly pitched up and climbed 200 feet in 70 seconds (Fig. 6-17). In Fig. 6-18, the DLC aircraft (controller 61) was subjected to similar turbulence. It climbed 100 feet in 150 seconds with <u>no change in pitch attitude</u>. The altitude hold mode was then added to controller 61, and the system was again flown in vertical turbulence. Fig. 6-19 shows that the airplane then flew indefinitely with altitude variations not exceeding ten feet. Again, pitch attitude was unchanged.

Effects of Washouts and Control Surface Crossfeeds

The elevator controller shown on pages 64 and 90 incorporates a washout circuit for the θ and $\dot{\theta}$ feedbacks. This circuit was used for all the SDF simulation runs at the AFFDL. The results were good, as summarized in Figures 6-32 through 6-34. The purpose of this washout was to assist the pilot in rotating the aircraft if a rapid conventional climb or descent were required. The TDF simulation runs shown in Figures 6-13, 6-15, 6-16, 6-18, and 6-19 were made later without this washout circuit. Comparison tests on the TDF simulation showed that DLC performance was improved by eliminating the washout integrator. The already small changes in pitch attitude were reduced still further.

Interestingly, the DLC aircraft's performance in conventional climbs (with rotation) was also improved by removing the washout circuit. When a conventional climb is completed and the pilot releases stick pressure, the controller feedbacks return the plane, quickly and with little overshoot, to its original attitude. The controller then automatically returns the ailerons and elevator to neutral, and the spoilers to their 20° bias position. Unfortunately, the equipment for the SDF simulation was not available for further use after this was discovered.

Early controller designs (not shown) used crossfeeds between elevators and spoilers, etc., based on a literature study of conventionally designed direct lift control systems. Experimentation with the TDF simulation soon showed that these feedbacks degraded system performance. This is logical since the digital design program establishes an optimal ratio of N_L, θ, and $\dot{\theta}$ feedbacks to the three control surfaces. Additional items such as control surface crossfeeds and washout circuits only destroy the optimality of the controllers. Although use of the washout circuit, described above, during all the SDF data runs was unfortunate, the improved performance with washout eliminated did serve to demonstrate the true optimality of the digitally designed controller.

Comparison of Responses to Initial θ

Fig. 6-30 shows the response of the basic B-52 to an initial θ of one degree (SDF simulation, flight condition three). As expected, this initial condition excites the phugoid mode. Fig. 6-31 shows the DLC aircraft (controller 43) response to an initial untrimmed pitch angle. Note that the controller essentially uses only the elevator to remove the pitch angle. This response was inadvertantly made with extremely slow servo actuators . . . the three time constants were all six seconds. Faster responding servos would undoubtedly have reduced the oscillations in θ and $\dot{\theta}$. During the system equation integration at the end of the digital

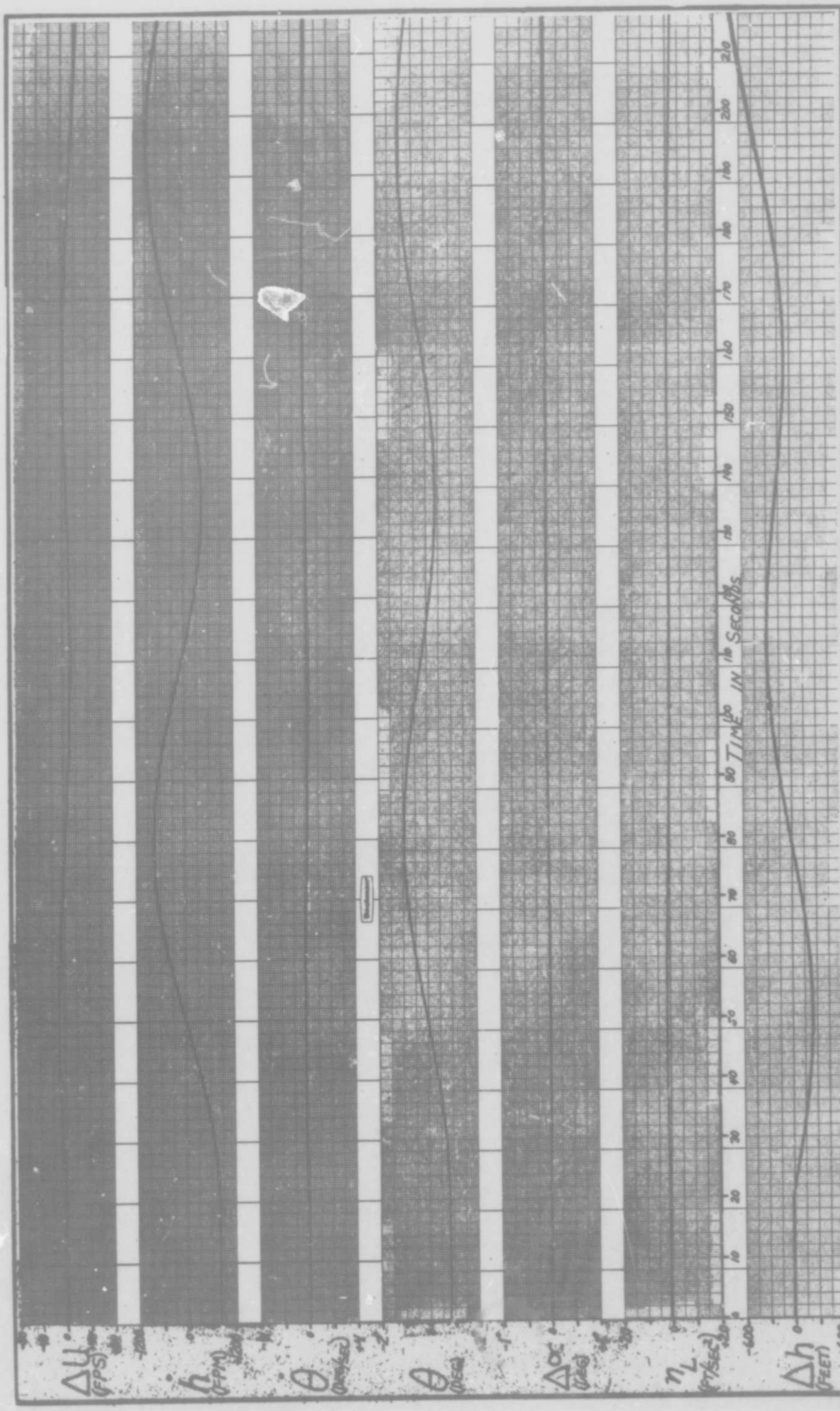


Fig. 6-30. Basic Aircraft Response to Initial θ = 1°, Flight Condition Three

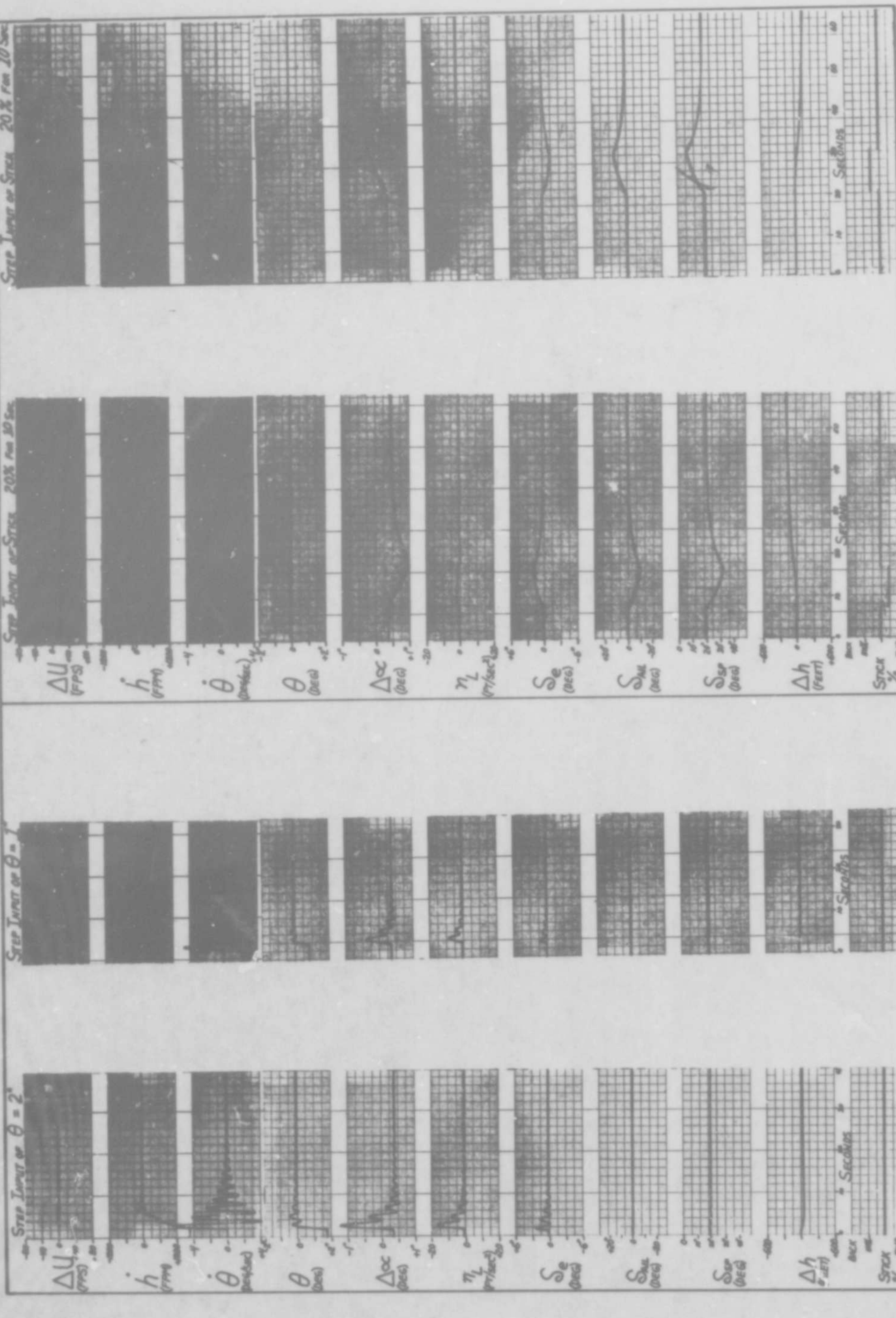


Fig. 6-31. Controller 43 Responses

optimization program (Appendix A, pages A-21 through A-23), this maneuver is accomplished without pitch overshoot.

Figure 6-31 also shows controller 43 responses to ten second stick inputs. Note the slight difference in performance of forward and back stick, caused by non-linearity of the spoilers.

Pilots' Qualitative Evaluation

The simulation would not have been complete without the pilots' evaluation of Direct Lift Control. Pilot comments and evaluations were obtained as explained in Chapter V and the results are presented in tabular form. Table VI-2 lists Cooper ratings for both the DLC and basic aircraft. The DLC aircraft received consistently higher Cooper ratings than the standard B-52 in all three flight conditions. Table VI-3 shows how well the pilots thought they performed the assigned tasks. Every pilot thought he did as well or better with the DLC aircraft. While studying the strip chart recordings and pilot rating forms, the authors noted strong correlation between how well a pilot did and how well he thought he did. If anything, the pilots tended to be modest in their evaluation of how well the tasks were accomplished.

Another important fact is that the majority of the pilots thought the workload in performing a given task was markedly reduced by the addition of the Direct Lift Controller. Workload data is found in Table VI-4, page 161.

In summary, the pilots felt that Direct Lift Control increased their ability to perform the assigned maneuvers and improved aircraft qualities while reducing pilot work-load.

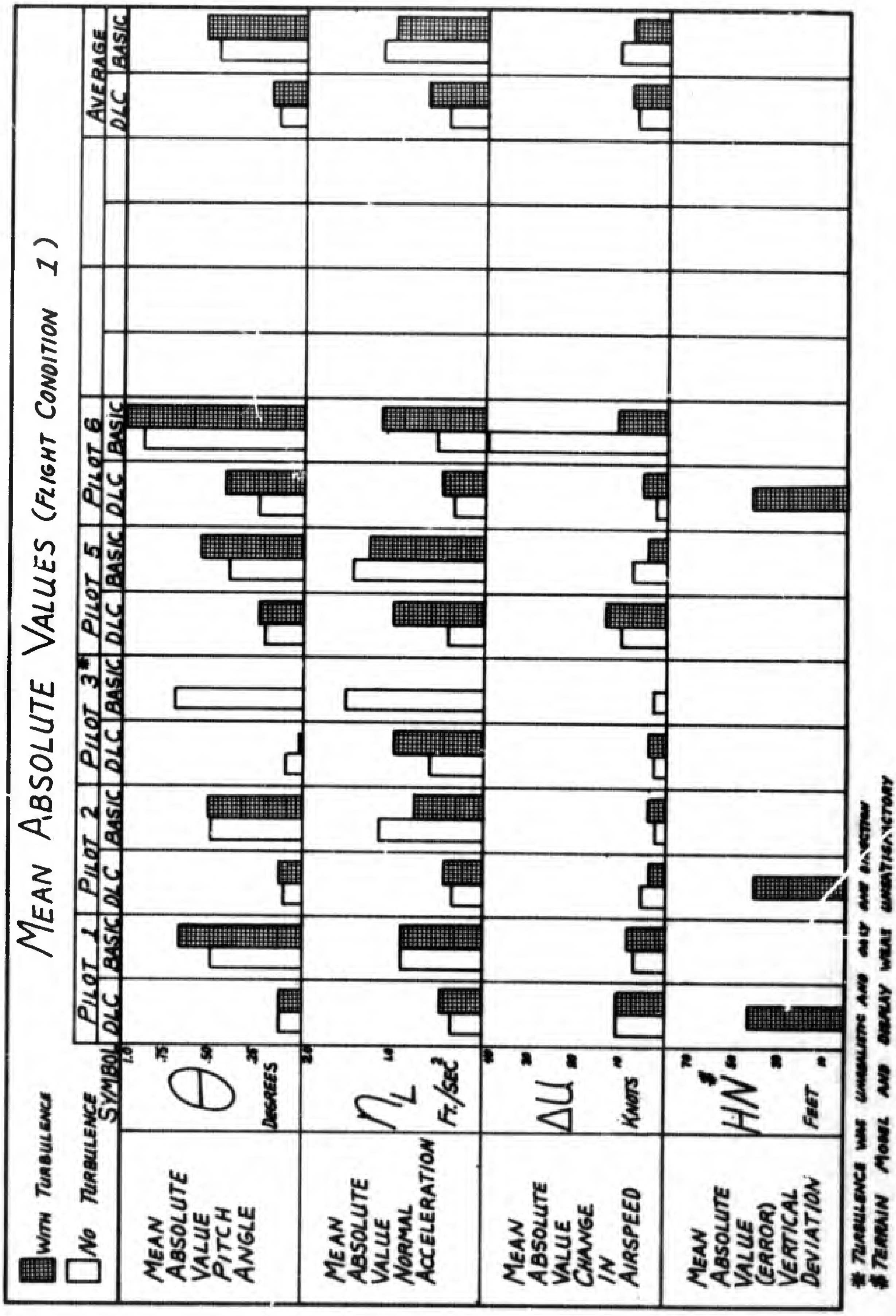
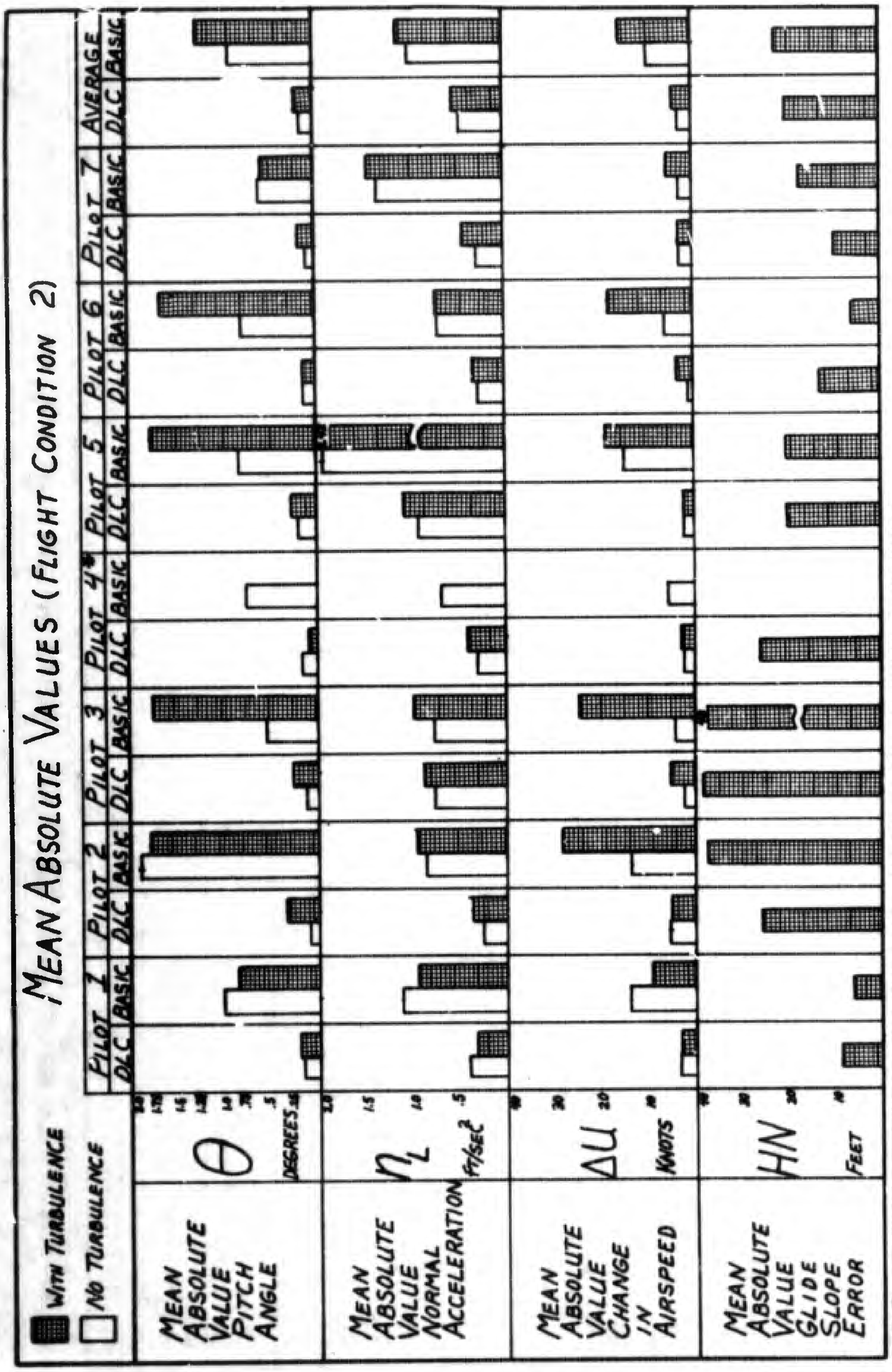


Fig. 6-32. Mean Absolute Values (Flight Condition One)



* PILOT UNABLE TO COMPLETE THE FLIGHT WITH TURBULENCE DUE TO OTHER COMMITMENTS.

Fig. 6-33. Mean Absolute Values (Flight Condition Two)

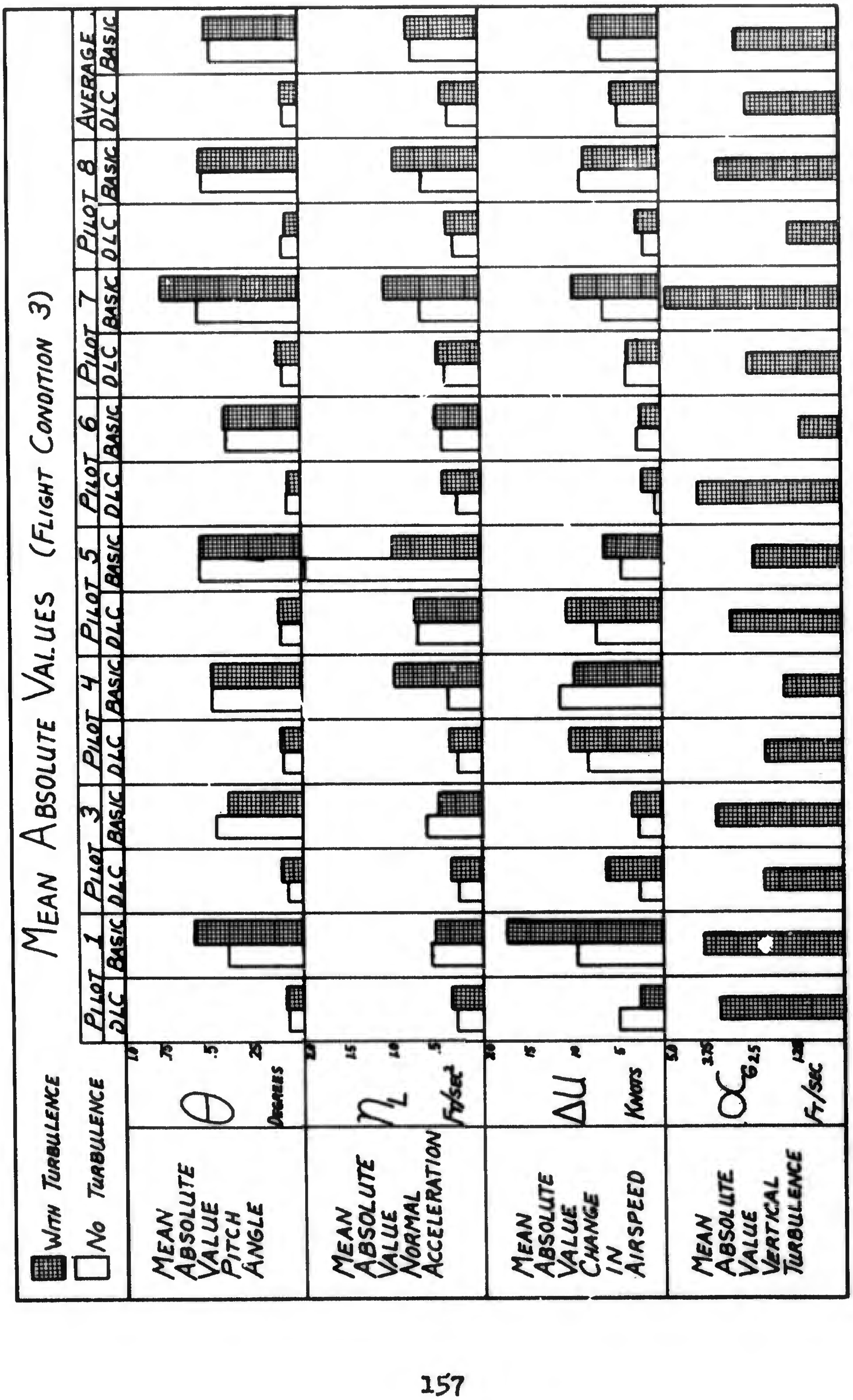


Fig. 6-34. Mean Absolute Values (Flight Condition Three)

PARAMETER REDUCTION WITH DLC

PARAMETER	Flight Condition 1		Flight Condition 2		Flight Condition 3	
	Turbulence		Turbulence		Turbulence	
	No	Yes	No	Yes	No	Yes
Percent Reduction in θ	74.6	73.3	83.3	82.9	81.9	80.9
Percent Reduction in N_L	64.7	36.7	56.9	53.9	52.1	45.8
Percent Reduction in ΔU	44.1	3.0 Increase	70.8	79.3	30.3	26.5

Table VI-1

PILOT	COOPER RATINGS* * See Fig. 5-5, page 109.					
	Flight Condition 1		Flight Condition 2		Flight Condition 3	
	Basic Aircraft	DLC Aircraft	Basic Aircraft	DLC Aircraft	Basic Aircraft	DLC Aircraft
1	3	1.5	3	2	2.5	1
2	2	2	2	2	-	-
3	3.5	2.5	3	2	3	1
4	-	-	3	1	2	1
5	2.5	1	3	1.5	3	1
6	3	3	3	2	3.5	2.5
7	-	-	6	3	4.5	3
8	-	-	-	-	3	1
AVERAGE	2.80	2.00	3.28	1.93	3.07	1.50

Table VI-2

PILOT	HOW DID YOU DO?* * See Fig. 5-4, page 108.					
	Flight Condition 1		Flight Condition 2		Flight Condition 3	
	Basic Aircraft	DLC Aircraft	Basic Aircraft	DLC Aircraft	Basic Aircraft	DLC Aircraft
1	3	2	2	1	2	1
2	2	2	2	1	-	-
3	3	2	3	2	2	1
4	-	-	2	1	2	2
5	3	1	2	1	2	1
6	2	2	2	2	3	2
7	-	-	3	2	3	2
8	-	-	-	-	2	1
AVERAGE	2.60	1.80	2.28	1.43	2.29	1.43

Table VI-3

P I L O T	W O R K L O A D *					
	Flight Condition 1		Flight Condition 2		Flight Condition 3	
	Basic Aircraft	DLC Aircraft	Basic Aircraft	DLC Aircraft	Basic Aircraft	DLC Aircraft
1	3	2	3	2	3	2
2	2	2	2	2	-	-
3	3	2	3	2	3	2
4	-	-	3	1	2	1
5	2	1	2	2	3	2
6	3	3	2	2	2	2
7	-	-	4	2	3	2
8	-	-	-	-	3	2
A V E R A G E	2.60	2.00	2.57	1.87	2.71	1.86

* See Fig. 5-2, page

Table VI-4

PILOT EXPERIENCE

Type of Aircraft	NUMBER OF HOURS							
	Pilot 1	Pilot 2	Pilot 3	Pilot 4	Pilot 5	Pilot 6	Pilot 7	Pilot 8
Jet Fighter/Trainer	235	250	235	280	240	---	2000	250
Jet Bomber/Cargo	---	2200	2475	---	---	---	---	---
Four Engine Recip.	1800	---	---	70	3300	---	---	---
Two Engine Recip.	1200	250	350	2900	200	---	900	130
Helicopter	2	---	---	---	---	---	---	2100
Light Aircraft	200	---	65	50	---	100	100	35
Simulator	100	---	---	---	***	850	---	50
TOTAL	3537	2700	3125	3300	3740	950	3000	2565

*** One year, C-124 simulator instructor.

Table VI-5

Chapter VII

Discussion and Conclusions

LAMS Spoiler Changes

When this study was begun, the LAMS controller operated with all seven spoiler panels on both sides of the airplane biased up 20°. The DLC controller was designed using the same spoiler configuration for compatability with LAMS, due to the possibility of a flight test. The LAMS design was subsequently modified to use only the two outboard spoiler panels on each wing. The designers found that the control power of the entire spoiler bank was not required for LAMS, and the change simplified some actuator problems. However, the LAMS aircraft retains the capability of using the complete spoiler bank with a 20° bias, so a flight test is still possible without major aircraft modifications.

The digital optimization program could be changed to use less than the full bank of spoilers by simply reducing the spoiler lift and pitching moment coefficients appropriately. Of course, the maximum rates of climb and descent without pitch would be reduced if fewer spoiler panels were used. The controller could be designed for a different spoiler bias setting by recomputing the linearized lift and pitching moment coefficients, based on the slope of the spoiler attenuation curve at the selected bias point.

Di. ct Lift Control Feasability

This study, together with the work reported in references five, six, eight, and nine, establishes beyond reasonable doubt the feasability of direct lift control. Control surfaces such as spoilers, symmetric ailerons, flaps, or canards, can be used in conjunction with the elevator to apply lift forces in the direction the pilot desires to move the airplane, without changing the pitch attitude. Direct Lift Control can eliminate the long heave crossover time and normal acceleration reversal associated with large aircraft (Ref 7:7, as quoted on pages 2 and 4).

LAMS B-52 DLC Capability

The Direct Lift Controllers described in this thesis employ the full B-52 spoiler bank with $\pm 20^{\circ}$ authority from the 20° bias position, full $\pm 17^{\circ}$ aileron authority, and full $\pm 17^{\circ}$ elevator authority. The controllers can provide changes in rate of climb and descent of up to 500 feet per minute from trim condition without a change in the aircraft pitch attitude. This improves precision maneuvering capability in tasks such as terrain following, approach and landing, and in-flight refueling. The pilot's ability to make precise corrections in rate of climb or altitude is increased because DLC does not excite the phugoid mode, and visual cues are not disturbed by continually changing pitch attitude.

With large stick movements, DLC functions as a conventional control system, providing pitching moments and aircraft rotation.

In addition, DLC reduced normal acceleration of the airplane center of gravity by an average of 50%, both in and out of turbulence. This suggests an improvement in riding qualities and reduction in aircraft structural damage if DLC is used during periods of turbulence. As an added bonus, the pilots (after a short training period) were able to control airspeed more closely with DLC than with the basic aircraft.

Optimization Theory

Kalman's optimal control theory for a linear plant with quadratic cost function (Ref 2:750-780) proved well-suited to DLC design. The B-52 "plant" was linearized by using the aircraft short period approximation and computing spoiler lift and pitching moment coefficients based on the slope of the spoiler attenuation constant curve (page 58) at the spoiler bias position. Simple magnitude scaling of the equations (page 28) completely eliminated the integration problems sometimes encountered with this theory.

Optimization theories are often criticized for producing complex controllers with an excessive number of feedbacks, some of which may not be readily available. This thesis demonstrates the opposite. The controllers consist simply of fixed-gain feedbacks of normal acceleration, pitch angle, and pitch rate to each of the control surface servos.

The system is operated by the pilot's normal longitudinal stick or column movements. The control surface crossfeeds, washout circuits, etc., normally associated with a conventionally designed DLC system are non-existent. They only serve to degrade DLC performance.

Once written, the digital optimization program is extremely flexible. The goal of the controller described in this thesis is to use spoilers, ailerons, and elevators to prevent change in aircraft pitch attitude. In practice, these controllers would be limited to short-period usage (refueling, landing approach, etc.), by the drag penalties inherent in the 20° spoiler bias. However, by increasing the cost of N_L and spoiler usage, and decreasing the cost of θ and $\dot{\theta}$ (changing four data cards), the program can be used to design a continuous duty controller which uses ailerons and elevators to improve ride and provide gust alleviation during cruise.

Some choice in the number and nature of the feedbacks has also been demonstrated. The N_L, θ, and $\dot{\theta}$ feedback controllers described above were designed by optimization program D (page 36). However, optimization program A, (page 19) uses α (angle of attack), θ, and $\dot{\theta}$ feedbacks, and program B (page 28) provides α, θ, $\dot{\theta}$, and N_L feedbacks. Program D was used for all data runs due to the difficulty of measuring angle of attack in an actual aircraft, desire to reduce N_L, and the obvious preference for three feedbacks rather than four. Nevertheless, the three types of controllers are capable of comparable performance if the specified feedbacks are readily available. A study of the three digital programs will show that little work was required to change from one program to the next. All three programs are fast-running (see page 51).

Pilot Quantitative Evaluation

As described in Chapter VI, pages 153 to 154, the pilots of the six degree of freedom simulation at the Air Force Flight Dynamics Laboratory were uniformly enthusiastic about the benefits of DLC. Cooper Ratings of the DLC aircraft averaged 1.24 better than those of the basic airplane. "How Did You Do?" ratings averaged 0.83 better with DLC, and average workload was reduced 0.72.

Additional Comments

Two Degree of Freedom Analog Simulation. The two degree of freedom analog simulation described in Chapter II proved invaluable for testing digitally designed controllers and provided a basis for manipulating the Q and R matrices. Its merits and deficiencies are discussed in detail on pages 112 to 122. The small amount of equipment required and its simplicity make this simulation especially desireable in the early DLC design stages.

Single Fixed-Gain Controller. Although separate controllers were designed and used for each of the three LAMS flight conditions, controllers designed to the same criteria (identical Q and R matrices) are very similar. This raises the possibility of selecting a single fixed gain controller that would perform adequately throughout the entire B-52 flight envelope. See pages 44 to 45 for details.

<u>Applicability to Other Aircraft</u>. Kalman's optimization theory is unlimited in the number of state variables and controls that can be considered. Thus, the design method and digital optimization programs used should be applicable to other aircraft and other types of control surfaces with only minor modifications.

The authors hope to demonstrate this fact in a follow-on study. The Boeing Company has agreed to provide equations of motion and aerodynamic coefficients for the Boeing 367-80 (707 prototype). This research aircraft has a fly-by-wire control system similar to that of the LAMS B-52, is equipped with high speed trailing edge flaps, and has flap-blowing capability. A dash 80 Direct Lift Controller will be demonstrated with a two degree of freedom simulator.

Chapter VIII

Recommendations for Further Study

As mentioned in Chapter I, the feasability of Direct Lift Control was first demonstrated in 1961. Serious consideration of this principle is just beginning; the first fleetwide application is yet to be made; the opportunities for further study are practically unlimited. The following are just a few of the areas that might be investigated in follow-on AFIT theses.

Additional LAMS B-52 Projects

Much additional study could be done with the LAMS B-52 testbed. For instance, the effect of increasing normal acceleration cost (value of Q_{11} in Optimization Program D) could be investigated, using controller servos that more closely simulate fleet B-52 equipment. Another optimal control theory could be applied to determine the best single controller for use in all three flight conditions. A new computer program could be written using only normal acceleration and pitch angle feedback, or the relatively new C* (Ref 10 and 11) criteria could be applied.

Other Aircraft

Direct Lift Control designs for other aircraft would be of interest to the aviation community. Notable among these are the SST (Supersonic Transport) and the C-5A Galaxy. In a recently announced design refinement, The Boeing Co. has added DLC using wing spoilers and canard surfaces aft of the crew compartment to the SST (Ref 12:16). DLC was added primarily as a landing aid. Since the SST rollout will be delayed by war-dictated funding cutbacks, there should be time for meaningful thesis work with this aircraft.

As mentioned on page 7, Honeywell has designed a direct lift controller for the C-5A using conventional design techniques and slightly different criteria. Interesting comparison studies could be made with a new design based on optimal control theory. Retrofit DLC capability for the C-5A is a distinct possibility due to the longitudinal sluggishness caused by its great size and weight, so a DLC design for this ship could also be of practical as well as theoretical value.

Another comparison between conventional and optimal theory design techniques could be made using the results of NASA's Convair 990/DC-8 studies as the conventional design.

A thesis could also be aimed at specific problem aircraft to better demonstrate the value of DLC. This might include one of the jetliners which is considered difficult to handle in landing approach and a fighter design which has difficulty during in-flight refueling.

Other Control Surfaces

In addition to the canard surfaces mentioned for an SST study, several other direct lift mechanisms could be used. Among these are boundary layer control, blown flaps, and thrust augmented lift.

The Boeing-80 (707 prototype) has been modified for a blown flap test program. This aircraft would be suitable for a DLC study because the results, without blown flaps, would be directly applicable to a large number of current aircraft, and the results with blown flaps would help establish the usefulness of this device. Again, a flight test program would be feasable due to the availability of a test aircraft.

Appendix A

Sample Computer Program Outputs

Sample Output, Computer Program A

The following is a sample output of computer program A, with states alpha, theta, and theta dot. The off-diagonal elements of the Q matrix reduce backward integration time of the C matrix from three seconds to one second, but have no effect on the final controller.

```
Q MATRIX

     0.          -0.          -0.
    -0.          20.0000      20.0000
    -0.          20.0000      20.0000

R MATRIX

     0.0400      -0.          -0.
    -0.           0.0400      -0.
    -0.          -0.           0.0400

COEFFICIENTS OF CHARACTERISTIC EQUATION

A( 1)=  1.000000E 00
A( 2)= -4.000000E 01
A( 3)=  0.
A( 4)= -0.

ROOTS OF CHARACTERISTIC EQUATION

S=  0.                 +J
S=  0.                 +J
S=  4.0000000E 01      +J

COEFFICIENTS OF CHARACTERISTIC EQUATION

A( 1)=  1.000000E 00
A( 2)= -1.200000E-01
A( 3)=  4.800000E-03
A( 4)= -6.399999E-05

ROOTS OF CHARACTERISTIC EQUATION

S=  4.0048018E-02    +J  8.2612640E-05
S=  4.0048018E-02    +J -8.2612640E-05
S=  3.9912091E-02    +J
```

FLIGHT CONDITION 3

A MATRIX

-0.6939	0.	0.9904
0.	0.	1.0000
-2.4739	0.	-0.6518

B MATRIX

-0.0211	0.0145	-0.0153
0.	0.	0.
-1.6332	0.2838	-0.1366

ELEMENTS OF SYMMETRIC FEEDBACK MATRIX C

C(1,1)	C(1,2)	C(1,3)	C(2,2)	C(2,3)
TIME = 9.95				
2.1120E-03	-2.4586E-02	-2.4841E-02	5.1033E-01	5.1341E-01
TIME = 9.90				
6.1484E-03	-3.3156E-02	-3.3768E-02	5.3380E-01	5.3769E-01
TIME = 9.85				
1.0033E-02	-3.4342E-02	-3.5113E-02	5.3427E-01	5.3823E-01
TIME = 9.80				
1.3675E-02	-3.4536E-02	-3.5279E-02	5.3428E-01	5.3824E-01
TIME = 9.75				
1.7084E-02	-3.4723E-02	-3.5273E-02	5.3429E-01	5.3824E-01
TIME = 9.70				
2.0277E-02	-3.5030E-02	-3.5246E-02	5.3432E-01	5.3823E-01
TIME = 9.65				
2.3265E-02	-3.5459E-02	-3.5221E-02	5.3438E-01	5.3823E-01
TIME = 9.60				
2.6064E-02	-3.5992E-02	-3.5201E-02	5.3449E-01	5.3823E-01
TIME = 9.55				
2.8683E-02	-3.6615E-02	-3.5185E-02	5.3464E-01	5.3822E-01
TIME = 9.50				
3.1136E-02	-3.7310E-02	-3.5173E-02	5.3483E-01	5.3822E-01

TIME = 9.45

3.3432E-02	-3.8066E-02	-3.5165E-02	5.3508E-01	5.3822E-01

TIME = 9.40

3.5581E-02	-3.8868E-02	-3.5160E-02	5.3538E-01	5.3821E-01

TIME = 9.35

3.7594E-02	-3.9708E-02	-3.5157E-02	5.3573E-01	5.3821E-01

TIME = 9.30

3.9478E-02	-4.0574E-02	-3.5157E-02	5.3613E-01	5.3821E-01

TIME = 9.25

4.1242E-02	-4.1460E-02	-3.5159E-02	5.3658E-01	5.3821E-01

TIME = 9.20

4.2894E-02	-4.2359E-02	-3.5162E-02	5.3706E-01	5.3822E-01

TIME = 9.15

4.4440E-02	-4.3263E-02	-3.5168E-02	5.3759E-01	5.3822E-01

TIME = 9.10

4.5887E-02	-4.4167E-02	-3.5174E-02	5.3816E-01	5.3822E-01

TIME = 9.05

4.7243E-02	-4.5067E-02	-3.5181E-02	5.3876E-01	5.3823E-01

TIME = 9.00

4.8512E-02	-4.5959E-02	-3.5189E-02	5.3938E-01	5.3823E-01

FEEDBACK MATRIX C

0.0485	-0.0460	-0.0352
-0.0460	0.5394	0.5382
-0.0352	0.5382	0.5423

CONTROL= -RIBTC

-1.4112	21.9524	2.1226
0.2320	-3.8023	-3.8348
-0.1016	1.8209	1.8388

A-BRIBTC

-0.0592	-0.5464	0.4396
0.	0.	1.0000
-0.0892	-37.1814	-38.1229

```
COEFFICIENTS OF CHARACTERISTIC EQUATION

A( 1)=  1.000000E 00
A( 2)=  3.878213E 01
A( 3)=  6.235126E 01
A( 4)=  2.446138E 01

ROOTS OF CHARACTERISTIC EQUATION

S= -1.0086060E 00    +J  0.
S= -6.5335560E-01    +J  0.
S= -3.7120164E 01    +J

NOTE, CONTROLS IN DEGREES AND ANGLES IN RADIANS

SPOILERS BIASED UP 20 DEGREES.

USING OUTBOARD SPOILERS ONLY (0.56K)
```

TIME	ALPHA	THETA	THETA DOT	ELEVATOR	SPOILERS	AILERONS
0.	-0.	0.1000	-0.	0.	20.0000	0.
0.0500	-0.0038	0.0973	-0.0818	19.0096	16.7243	1.5557
0.1000	-0.0082	0.0929	-0.0906	2.6825	19.5569	0.1957
0.1500	-0.0123	0.0884	-0.0882	0.4639	19.9437	0.0085
0.2000	-0.0161	0.0841	-0.0842	0.4238	19.9526	0.0027
0.2500	-0.0195	0.0800	-0.0801	0.6991	19.9067	0.0234
0.3000	-0.0227	0.0761	-0.0762	1.0003	19.8560	0.0464
0.3500	-0.0255	0.0724	-0.0725	1.2836	19.8084	0.0682
0.4000	-0.0281	0.0689	-0.0689	1.5435	19.7646	0.0882
0.4500	-0.0305	0.0655	-0.0655	1.7803	19.7246	0.1065
0.5000	-0.0326	0.0623	-0.0623	1.9955	19.6883	0.1232
0.5500	-0.0345	0.0593	-0.0593	2.1904	19.6554	0.1383
0.6000	-0.0362	0.0564	-0.0564	2.3663	19.6256	0.1520
0.6500	-0.0377	0.0536	-0.0536	2.5244	19.5988	0.1644
0.7000	-0.0390	0.0510	-0.0510	2.6660	19.5748	0.1756
0.7500	-0.0401	0.0485	-0.0485	2.7920	19.5533	0.1855
0.8000	-0.0411	0.0462	-0.0461	2.9036	19.5343	0.1944
0.8500	-0.0420	0.0439	-0.0439	3.0016	19.5175	0.2023
0.9000	-0.0427	0.0418	-0.0417	3.0872	19.5028	0.2092
0.9500	-0.0433	0.0397	-0.0397	3.1610	19.4901	0.2153
1.0000	-0.0437	0.0378	-0.0378	3.2239	19.4792	0.2205
1.0500	-0.0441	0.0360	-0.0359	3.2767	19.4700	0.2249
1.1000	-0.0444	0.0342	-0.0342	3.3201	19.4624	0.2287
1.1500	-0.0446	0.0325	-0.0325	3.3549	19.4562	0.2318
1.2000	-0.0447	0.0310	-0.0309	3.3815	19.4514	0.2342
1.2500	-0.0447	0.0294	-0.0294	3.4007	19.4478	0.2361
1.3000	-0.0446	0.0280	-0.0280	3.4130	19.4454	0.2375
1.3500	-0.0445	0.0266	-0.0266	3.4189	19.4440	0.2384
1.4000	-0.0443	0.0254	-0.0253	3.4189	19.4437	0.2389
1.4500	-0.0441	0.0241	-0.0241	3.4135	19.4442	0.2389
1.5000	-0.0438	0.0229	-0.0229	3.4032	19.4456	0.2385
1.5500	-0.0434	0.0218	-0.0218	3.3883	19.4477	0.2378
1.6000	-0.0431	0.0208	-0.0207	3.3692	19.4506	0.2368
1.6500	-0.0427	0.0198	-0.0197	3.3463	19.4541	0.2355

TIME	ALPHA	THETA	THETA DOT	ELEVATOR	SPOILERS	AILERONS
1.7000	-0.0422	0.0188	-0.0187	3.3200	19.4582	0.2339
1.7500	-0.0417	0.0179	-0.0178	3.2905	19.4628	0.2321
1.8000	-0.0412	0.0170	-0.0169	3.2582	19.4679	0.2301
1.8500	-0.0407	0.0162	-0.0161	3.2232	19.4734	0.2278
1.9000	-0.0401	0.0154	-0.0153	3.1860	19.4793	0.2254
1.9500	-0.0396	0.0147	-0.0146	3.1467	19.4856	0.2228
2.0000	-0.0390	0.0139	-0.0139	3.1056	19.4922	0.2201
2.0500	-0.0384	0.0133	-0.0132	3.0629	19.4990	0.2172
2.1000	-0.0377	0.0126	-0.0126	3.0187	19.5061	0.2142
2.1500	-0.0371	0.0120	-0.0119	2.9732	19.5134	0.2112
2.2000	-0.0365	0.0114	-0.0114	2.9267	19.5209	0.2080
2.2500	-0.0358	0.0109	-0.0108	2.8793	19.5286	0.2047
2.3000	-0.0352	0.0103	-0.0103	2.8311	19.5364	0.2014
2.3500	-0.0345	0.0098	-0.0098	2.7823	19.5443	0.1981
2.4000	-0.0339	0.0094	-0.0093	2.7329	19.5523	0.1947
2.4500	-0.0332	0.0089	-0.0089	2.6832	19.5604	0.1912
2.5000	-0.0326	0.0085	-0.0084	2.6332	19.5685	0.1877
2.5500	-0.0319	0.0081	-0.0080	2.5830	19.5766	0.1843
2.6000	-0.0313	0.0077	-0.0076	2.5328	19.5848	0.1807
2.6500	-0.0306	0.0073	-0.0073	2.4825	19.5930	0.1772
2.7000	-0.0300	0.0070	-0.0069	2.4324	19.6012	0.1737
2.7500	-0.0293	0.0066	-0.0066	2.3823	19.6093	0.1702
2.8000	-0.0287	0.0063	-0.0062	2.3326	19.6174	0.1667
2.8500	-0.0280	0.0060	-0.0059	2.2830	19.6255	0.1632
2.9000	-0.0274	0.0057	-0.0057	2.2339	19.6335	0.1598
2.9500	-0.0268	0.0054	-0.0054	2.1851	19.6415	0.1563
3.0000	-0.0262	0.0052	-0.0051	2.1368	19.6494	0.1529
3.0500	-0.0256	0.0049	-0.0049	2.0889	19.6572	0.1496
3.1000	-0.0250	0.0047	-0.0046	2.0415	19.6649	0.1462
3.1500	-0.0244	0.0045	-0.0044	1.9947	19.6726	0.1429
3.2000	-0.0238	0.0042	-0.0042	1.9485	19.6801	0.1396
3.2500	-0.0232	0.0040	-0.0040	1.9029	19.6876	0.1364
3.3000	-0.0227	0.0038	-0.0038	1.8579	19.6949	0.1332
3.3500	-0.0221	0.0037	-0.0036	1.8136	19.7022	0.1301
3.4000	-0.0216	0.0035	-0.0034	1.7699	19.7093	0.1270
3.4500	-0.0210	0.0033	-0.0033	1.7269	19.7164	0.1239

TIME	ALPHA	THETA	THETA DOT	ELEVATOR	SPOILERS	AILERONS
3.5000	-0.0205	0.0032	-0.0031	1.6846	19.7233	0.1209
3.5500	-0.0200	0.0030	-0.0030	1.6431	19.7301	0.1179
3.6000	-0.0195	0.0029	-0.0028	1.6022	19.7368	0.1150
3.6500	-0.0190	0.0027	-0.0027	1.5621	19.7434	0.1122
3.7000	-0.0185	0.0026	-0.0026	1.5227	19.7498	0.1094
3.7500	-0.0180	0.0025	-0.0024	1.4840	19.7562	0.1066
3.8000	-0.0176	0.0023	-0.0023	1.4461	19.7624	0.1039
3.8500	-0.0171	0.0022	-0.0022	1.4089	19.7685	0.1012
3.9000	-0.0166	0.0021	-0.0021	1.3724	19.7745	0.0986
3.9500	-0.0162	0.0020	-0.0020	1.3367	19.7803	0.0961
4.0000	-0.0158	0.0019	-0.0019	1.3017	19.7861	0.0936
4.0500	-0.0154	0.0018	-0.0018	1.2675	19.7917	0.0911
4.1000	-0.0149	0.0017	-0.0017	1.2340	19.7972	0.0887
4.1500	-0.0145	0.0017	-0.0016	1.2012	19.8026	0.0864
4.2000	-0.0141	0.0016	-0.0016	1.1691	19.8078	0.0841
4.2500	-0.0138	0.0015	-0.0015	1.1377	19.8130	0.0819
4.3000	-0.0134	0.0014	-0.0014	1.1071	19.8180	0.0797
4.3500	-0.0130	0.0014	-0.0013	1.0771	19.8229	0.0775
4.4000	-0.0127	0.0013	-0.0013	1.0478	19.8277	0.0754
4.4500	-0.0123	0.0012	-0.0012	1.0192	19.8324	0.0734
4.5000	-0.0120	0.0012	-0.0012	0.9913	19.8370	0.0714
4.5500	-0.0116	0.0011	-0.0011	0.9640	19.8415	0.0694
4.6000	-0.0113	0.0011	-0.0010	0.9374	19.8459	0.0675
4.6500	-0.0110	0.0010	-0.0010	0.9114	19.8501	0.0656
4.7000	-0.0107	0.0010	-0.0009	0.8860	19.8543	0.0638
4.7500	-0.0104	0.0009	-0.0009	0.8613	19.8583	0.0620
4.8000	-0.0101	0.0009	-0.0009	0.8372	19.8623	0.0603
4.8500	-0.0098	0.0008	-0.0008	0.8137	19.8662	0.0586
4.9000	-0.0095	0.0008	-0.0008	0.7907	19.8699	0.0570
4.9500	-0.0093	0.0008	-0.0007	0.7684	19.8736	0.0554
5.0000	-0.0090	0.0007	-0.0007	0.7466	19.8772	0.0538
5.0500	-0.0087	0.0007	-0.0007	0.7254	19.8807	0.0523

Sample Output, Computer Program B

The following is a sample output of computer program B. The states used are X(1) = α (Angle of attack), X(2) = θ (Pitch angle), X(3) = $\dot{\theta}$ (Pitch rate), and X(4) = N_L (Normal acceleration, positive in the up direction).

```
Q MATRIX

   1.0000       -0.           -C.           -0.
  -C.           20.0000       -0.           -0.
  -0.           -0.           1C.0000       -0.
  -0.           -0.           -0.            0.5000


R MATRIX

   0.0300       -0.           -0.
  -0.            0.C300       -0.
  -0.           -0.            C.0300C

Q MATRIX


COEFFICIENTS OF CHARACTERISTIC EQUATION

A( 1)=   1.000000E 00
A( 2)=  -3.150000E 01
A( 3)=   2.455000E 02
A( 4)=  -3.150000E 02
A( 5)=   9.999997E 01


ROOTS OF CHARACTERISTIC EQUATION

S=  1.0000003E 00   +J  0.
S=  4.9999970E-01   +J  0.
S=  2.0000000E 01   +J  0.
S=  9.9999996E 00   +J  0.
FLIGHT CONDITION 1
A MATRIX

  -0.8850      0.          0.9838       0.
   0.          0.          1.0000       0.
  -2.8067      0.         -1.0414       0.
  -5.1924      0.          5.3502       0.

B MATRIX

  -0.0255      C.0279     -0.0165
   0.          0.          C.
  -1.8858      0.5596     -0.1282
  -0.3324      0.2114     -0.1043
```

FEEDBACK MATRIX C TIME = 19.50

0.6708	0.1238	-0.0009	-0.2328
0.1238	8.9139	0.2396	-0.1418
-0.0009	0.2396	0.2770	-0.0149
-0.2328	-0.1418	-0.0149	0.2268

FEEDBACK MATRIX C TIME = 19.00

2.1957	1.0698	0.0620	-0.7303
1.0698	14.0808	0.4122	-0.6048
0.0620	0.4122	0.2835	-0.0406
-0.7303	-0.6048	-0.0406	0.4110

FEEDBACK MATRIX C TIME = 18.50

4.6869	2.6049	0.1583	-1.3843
2.6049	16.7346	0.5228	-1.1025
0.1583	0.5228	0.2887	-0.0688
-1.3843	-1.1025	-0.0688	0.5902

FEEDBACK MATRIX C TIME = 18.00

8.0129	4.1652	0.2692	-2.1558
4.1652	18.1009	0.5943	-1.5101
0.2692	0.5943	0.2930	-0.0959
-2.1558	-1.5101	-0.0959	0.7727

FEEDBACK MATRIX C TIME = 17.50

11.9757	5.4909	0.3839	-3.0059
5.4909	18.7998	0.6406	-1.8157
0.3839	0.6406	0.2966	-0.1211
-3.0059	-1.8157	-0.1211	0.9569

FEEDBACK MATRIX C TIME = 17.00

16.3916	6.5279	0.4975	-3.9050
6.5279	19.1541	0.6707	-2.0367
0.4975	0.6707	0.2996	-0.1446
-3.9050	-2.0367	-0.1446	1.1408

FEEDBACK MATRIX C TIME = 16.50

21.1210	7.3055	0.6085	-4.8338
7.3055	19.3324	0.6906	-2.1941
0.6085	0.6906	0.3023	-0.1665
-4.8338	-2.1941	-0.1665	1.3237

FEEDBACK MATRIX C TIME = 16.00

26.0657	7.8752	0.7169	-5.7808
7.8752	19.4219	0.7038	-2.3055
0.7169	0.7038	0.3047	-0.1874
-5.7808	-2.3055	-0.1874	1.5053

FEEDBACK MATRIX C TIME = 15.50

31.1585	8.2870	0.8230	-6.7391
8.2870	19.4667	0.7128	-2.3840
0.8230	0.7128	0.3069	-0.2074
-6.7391	-2.3840	-0.2074	1.8857

FEEDBACK MATRIX C TIME = 15.00

36.3534	8.5824	0.9273	-7.7044
8.5824	19.4890	0.7189	-2.4395
0.9273	0.7189	0.3090	-0.2268
-7.7044	-2.4395	-0.2268	1.8651

FEEDBACK MATRIX C TIME = 14.50

41.6194	8.7931	1.0302	-8.6744
8.7931	19.5002	0.7231	-2.4785
1.0302	0.7231	0.3110	-0.2457
-8.6744	-2.4785	-0.2457	2.0438

FEEDBACK MATRIX C TIME = 14.00

46.9348	8.9431	1.1321	-9.6474
8.9431	19.5058	0.7260	-2.5061
1.1321	0.7260	0.3130	-0.2644
-9.6474	-2.5061	-0.2644	2.2219

FEEDBACK MATRIX C TIME = 13.50

52.2848	9.0495	1.2332	-10.6224
9.0495	19.5086	0.7280	-2.5256
1.2332	0.7280	0.3149	-0.2828
-10.6224	-2.5256	-0.2828	2.3996

FEEDBACK MATRIX C TIME = 13.00

57.6591	9.1249	1.3338	-11.5988
9.1249	19.5100	0.7294	-2.5393
1.3338	0.7294	0.3168	-0.3011
-11.5988	-2.5393	-0.3011	2.5770

FEEDBACK MATRIX C TIME = 12.50

63.0505	9.1784	1.4341	-12.5762
9.1784	19.5107	0.7304	-2.5490
1.4341	0.7304	0.3186	-0.3193
-12.5762	-2.5490	-0.3193	2.7542

FEEDBACK MATRIX C TIME =12.00

68.4539	9.2162	1.5340	-13.5543
9.2162	19.5111	0.7311	-2.5559
1.5340	0.7311	0.3205	-0.3374
-13.5543	-2.5559	-0.3374	2.9313

FEEDBACK MATRIX C TIME = 11.50

73.8658	9.2429	1.6338	-14.5328
9.2429	19.5112	0.7316	-2.5607
1.6338	0.7316	0.3223	-0.3554
-14.5328	-2.5607	-0.3554	3.1082

FEEDBACK MATRIX C TIME = 11.00

79.2837	9.2619	1.7334	-15.5116
9.2619	19.5113	0.7320	-2.5641
1.7334	0.7320	0.3242	-0.3734
-15.5116	-2.5641	-0.3734	3.2850

FEEDBACK MATRIX C TIME = 10.50

84.7058	9.2752	1.8329	-16.4907
9.2752	19.5114	0.7322	-2.5666
1.8329	0.7322	0.3260	-0.3914
-16.4907	-2.5666	-0.3914	3.4618

FEEDBACK MATRIX C TIME = 10.00

90.1308	9.2847	1.9323	-17.4699
9.2847	19.5114	0.7324	-2.5683
1.9323	0.7324	0.3278	-0.4093
-17.4699	-2.5683	-0.4093	3.6386

FEEDBACK MATRIX C TIME = 9.50

95.5580	9.2914	2.0317	-18.4492
9.2914	19.5114	0.7325	-2.5695
2.0317	0.7325	0.3296	-0.4273
-18.4492	-2.5695	-0.4273	3.8153

FEEDBACK MATRIX C TIME = 9.00

100.9866	9.2961	2.1311	-19.4286
9.2961	19.5114	0.7326	-2.5703
2.1311	0.7326	0.3314	-0.4452
-19.4286	-2.5703	-0.4452	3.9920

FEEDBACK MATRIX C TIME = 8.50

106.4163	9.2995	2.2305	-20.4081
9.2995	19.5114	0.7327	-2.5709
2.2305	0.7327	0.3333	-0.4631
-20.4081	-2.5709	-0.4631	4.1687

FEEDBACK MATRIX C TIME = 8.00

111.8467	9.3018	2.3298	-21.3875
9.3018	19.5114	0.7327	-2.5714
2.3298	0.7327	0.3351	-0.4810
-21.3875	-2.5714	-0.4810	4.3453

FEEDBACK MATRIX C TIME = 7.50

117.2777	9.3035	2.4291	-22.3671
9.3035	19.5114	0.7327	-2.5717
2.4291	0.7327	0.3369	-0.4989
-22.3671	-2.5717	-0.4989	4.5220

FEEDBACK MATRIX C TIME = 7.00

122.7090	9.3047	2.5284	-23.3466
9.3047	19.5114	0.7328	-2.5719
2.5284	0.7328	0.3387	-0.5168
-23.3466	-2.5719	-0.5168	4.6987

FEEDBACK MATRIX C TIME = 6.50

128.1406	9.3055	2.6277	-24.3262
9.3055	19.5114	0.7328	-2.5720
2.6277	0.7328	0.3405	-0.5347
-24.3262	-2.5720	-0.5347	4.8753

FEEDBACK MATRIX C TIME = 6.00

133.5724	9.3061	2.7270	-25.3057
9.3061	19.5114	0.7328	-2.5721
2.7270	0.7328	0.3423	-0.5527
-25.3057	-2.5721	-0.5527	5.0520

FEEDBACK MATRIX C TIME = 5.50

139.0042	9.3065	2.8263	-26.2853
9.3065	19.5114	0.7328	-2.5722
2.8263	0.7328	0.3442	-0.5706
-26.2853	-2.5722	-0.5706	5.2286

FEEDBACK MATRIX C TIME = 5.00

144.4362	9.3068	2.9256	-27.2649
9.3068	19.5114	0.7328	-2.5722
2.9256	0.7328	0.3460	-0.5885
-27.2649	-2.5722	-0.5885	5.4053

FEEDBACK MATRIX C TIME = 4.50

149.8683	9.3070	3.0248	-28.2445
9.3070	19.5114	0.7328	-2.5723
3.0248	0.7328	0.3478	-0.6064
-28.2445	-2.5723	-0.6064	5.5819

FEEDBACK MATRIX C TIME = 4.00

155.3004	9.3072	3.1241	-29.2241
9.3072	19.5114	0.7328	-2.5723
3.1241	0.7328	0.3496	-0.6243
-29.2241	-2.5723	-0.6243	5.7586

FEEDBACK MATRIX C TIME = 3.50

160.7325	9.3073	3.2234	-30.2036
9.3073	19.5114	0.7328	-2.5723
3.2234	0.7328	0.3514	-0.6422
-30.2036	-2.5723	-0.6422	5.9352

FEEDBACK MATRIX C TIME = 3.00

166.1647	9.3073	3.3227	-31.1832
9.3073	19.5114	0.7328	-2.5723
3.3227	0.7328	0.3532	-0.6601
-31.1832	-2.5723	-0.6601	6.1119

FEEDBACK MATRIX C TIME = 2.50

171.5968	9.3074	3.4220	-32.1628
9.3074	19.5114	0.7328	-2.5724
3.4220	0.7328	0.3550	-0.6780
-32.1628	-2.5724	-0.6780	6.2885

FEEDBACK MATRIX C TIME = 2.00

177.0290	9.3074	3.5212	-33.1424
9.3074	19.5114	0.7328	-2.5724
3.5212	0.7328	0.3569	-0.6959
-33.1424	-2.5724	-0.6959	6.4652

FEEDBACK MATRIX C TIME = 1.50

182.4612	9.3074	3.6205	-34.1220
9.3074	19.5114	0.7328	-2.5724
3.6205	0.7328	0.3587	-0.7138
-34.1220	-2.5724	-0.7138	6.6418

FEEDBACK MATRIX C TIME = 1.00

187.8934	9.3075	3.7198	-35.1016
9.3075	19.5114	0.7328	-2.5724
3.7198	0.7328	0.3605	-0.7317
-35.1016	-2.5724	-0.7317	6.8185

FEEDBACK MATRIX C TIME = 0.50

193.3255	9.3075	3.8191	-36.0811
9.3075	19.5114	0.7328	-2.5724
3.8191	0.7328	0.3623	-0.7496
-36.0811	-2.5724	-0.7496	6.9951

NOTE ... ACTUAL STATE X(4) IS .01*N SUB L.

CONTROL= -RIBTC

4.4618	25.4666	17.7112	-0.2540
3.2558	-4.1983	-5.0272	-1.7607
-3.0162	-0.7036	1.0381	1.3135

A-BRIBTC

-0.8582	-0.7542	0.3754	-0.0643
0.	0.	1.0000	0.
-9.0118	-50.2829	-37.3863	-0.6748
-5.6726	-9.2786	-1.7076	-0.4248

COEFFICIENTS OF CHARACTERISTIC EQUATION

A(1)= 1.000000E 00
A(2)= 3.866978E 01
A(3)= 1.004800E 02
A(4)= 5.145437E 01
A(5)= -2.368927E-03

ROOTS OF CHARACTERISTIC EQUATION

S= -3.5911700E 01 +J
S= -2.0638085E 00 +J 0.
S= -6.9431396E-01 +J 0.
S= 4.6035240E-05 +J

NOTE, CONTROLS IN DEGREES AND ANGLES IN RADIANS.

SPOILERS BIASED UP 20 DEGREES.

TIME	ALPHA	THETA	THETA DOT	N ACCEL	ELEVATOR	SPOILERS	AILERONS
0.	-0.	0.1000	-0.	-0.	0.8111	20.0000	-0.5204
0.0500	-0.0049	0.0964	-0.1105	-3.8461	27.7910	31.6098	-12.5032
0.1000	-0.0102	0.0905	-0.1198	-6.8457	8.8793	37.7462	-14.1687
0.1500	-0.0148	0.0846	-0.1131	-9.3798	6.3364	38.8994	-14.6303
0.2000	-0.0189	0.0792	-0.1044	-11.5468	6.4181	39.1544	-14.8272
0.2500	-0.0224	0.0742	-0.0962	-13.3944	6.8527	39.1986	-14.9232
0.3000	-0.0254	0.0696	-0.0886	-14.9590	7.2723	39.1539	-14.9525
0.3500	-0.0278	0.0653	-0.0817	-16.2720	7.6241	39.0478	-14.9274
0.4000	-0.0299	0.0614	-0.0754	-17.3614	7.9069	38.8911	-14.8554
0.4500	-0.0316	0.0578	-0.0697	-18.2523	8.1271	38.6914	-14.7430
0.5000	-0.0330	0.0544	-0.0645	-18.9675	8.2917	38.4548	-14.5958
0.5500	-0.0341	0.0513	-0.0597	-19.5271	8.4075	38.1870	-14.4187
0.6000	-0.0350	0.0484	-0.0553	-19.9493	8.4802	37.8928	-14.2162
0.6500	-0.0356	0.0458	-0.0514	-20.2503	8.5152	37.5767	-13.9922
0.7000	-0.0360	0.0433	-0.0477	-20.4449	8.5172	37.2425	-13.7503
0.7500	-0.0362	0.0410	-0.0444	-20.5460	8.4904	36.8936	-13.4935
0.8000	-0.0363	0.0388	-0.0413	-20.5653	8.4387	36.5332	-13.2247
0.8500	-0.0363	0.0368	-0.0385	-20.5134	8.3655	36.1640	-12.9462
0.9000	-0.0361	0.0350	-0.0360	-20.3996	8.2736	35.7882	-12.6604
0.9500	-0.0359	0.0332	-0.0336	-20.2321	8.1660	35.4080	-12.3690
1.0000	-0.0355	0.0316	-0.0315	-20.0186	8.0449	35.0253	-12.0738
1.0500	-0.0351	0.0301	-0.0295	-19.7656	7.9124	34.6416	-11.7762
1.1000	-0.0346	0.0287	-0.0276	-19.4791	7.7706	34.2583	-11.4775
1.1500	-0.0341	0.0273	-0.0260	-19.1643	7.6211	33.8768	-11.1790
1.2000	-0.0335	0.0261	-0.0244	-18.8260	7.4653	33.4979	-10.8815
1.2500	-0.0329	0.0249	-0.0230	-18.4683	7.3046	33.1227	-10.5859
1.3000	-0.0322	0.0238	-0.0216	-18.0949	7.1402	32.7520	-10.2930
1.3500	-0.0316	0.0227	-0.0204	-17.7090	6.9731	32.3863	-10.0034
1.4000	-0.0309	0.0217	-0.0193	-17.3137	6.8042	32.0264	-9.7176
1.4500	-0.0302	0.0208	-0.0182	-16.9113	6.6342	31.6725	-9.4362
1.5000	-0.0294	0.0199	-0.0172	-16.5042	6.4640	31.3252	-9.1594
1.5500	-0.0287	0.0191	-0.0163	-16.0944	6.2941	30.9848	-8.8877
1.6000	-0.0280	0.0183	-0.0155	-15.6835	6.1250	30.6515	-8.6212

TIME	ALPHA	THETA	THETA DOT	N ACCEL	ELEVATOR	SPOILERS	AILERONS
1.6500	-0.0273	0.0175	-0.0147	-15.2731	5.9572	30.3254	-8.3602
1.7000	-0.0266	0.0168	-0.0139	-14.8644	5.7911	30.0069	-8.1049
1.7500	-0.0258	0.0161	-0.0132	-14.4587	5.6271	29.6959	-7.8553
1.8000	-0.0251	0.0155	-0.0126	-14.0569	5.4653	29.3925	-7.6117
1.8500	-0.0244	0.0149	-0.0120	-13.6598	5.3061	29.0968	-7.3739
1.9000	-0.0237	0.0143	-0.0114	-13.2682	5.1496	28.8088	-7.1421
1.9500	-0.0230	0.0137	-0.0109	-12.8826	4.9961	28.5283	-6.9163
2.0000	-0.0224	0.0132	-0.0104	-12.5036	4.8457	28.2555	-6.6964
2.0500	-0.0217	0.0127	-0.0099	-12.1316	4.6984	27.9903	-6.4825
2.1000	-0.0210	0.0122	-0.0095	-11.7670	4.5544	27.7324	-6.2744
2.1500	-0.0204	0.0118	-0.0090	-11.4099	4.4137	27.4819	-6.0722
2.2000	-0.0198	0.0113	-0.0087	-11.0607	4.2763	27.2387	-5.8757
2.2500	-0.0192	0.0109	-0.0083	-10.7195	4.1423	27.0026	-5.6848
2.3000	-0.0186	0.0105	-0.0079	-10.3864	4.0118	26.7734	-5.4995
2.3500	-0.0180	0.0101	-0.0076	-10.0614	3.8846	26.5512	-5.3197
2.4000	-0.0174	0.0097	-0.0073	-9.7447	3.7609	26.3356	-5.1453
2.4500	-0.0169	0.0094	-0.0070	-9.4361	3.6405	26.1266	-4.9761
2.5000	-0.0164	0.0090	-0.0067	-9.1358	3.5234	25.9241	-4.8121
2.5500	-0.0158	0.0087	-0.0064	-8.8436	3.4096	25.7278	-4.6531
2.6000	-0.0153	0.0084	-0.0062	-8.5594	3.2991	25.5376	-4.4990
2.6500	-0.0148	0.0081	-0.0059	-8.2832	3.1918	25.3535	-4.3497
2.7000	-0.0143	0.0078	-0.0057	-8.0149	3.0876	25.1751	-4.2051
2.7500	-0.0139	0.0075	-0.0055	-7.7543	2.9865	25.0024	-4.0651
2.8000	-0.0134	0.0072	-0.0053	-7.5014	2.8884	24.8352	-3.9295
2.8500	-0.0130	0.0070	-0.0051	-7.2559	2.7934	24.6734	-3.7982
2.9000	-0.0126	0.0067	-0.0049	-7.0178	2.7012	24.5168	-3.6711
2.9500	-0.0122	0.0065	-0.0047	-6.7869	2.6118	24.3653	-3.5482
3.0000	-0.0118	0.0063	-0.0045	-6.5631	2.5253	24.2186	-3.4291
3.0500	-0.0114	0.0060	-0.0043	-6.3461	2.4414	24.0768	-3.3140
3.1000	-0.0110	0.0058	-0.0042	-6.1358	2.3602	23.9396	-3.2026
3.1500	-0.0106	0.0056	-0.0040	-5.9321	2.2815	23.8069	-3.0948
3.2000	-0.0103	0.0054	-0.0039	-5.7348	2.2054	23.6785	-2.9906
3.2500	-0.0099	0.0052	-0.0037	-5.5438	2.1316	23.5544	-2.8898
3.3000	-0.0096	0.0051	-0.0036	-5.3588	2.0603	23.4344	-2.7923

TIME	ALPHA	THETA	THETA DOT	N ACCEL	ELEVATOR	SPOILERS	AILERONS
3.3500	-0.0093	0.0049	-0.0035	-5.1797	1.9912	23.3183	-2.6980
3.4000	-0.0090	0.0047	-0.0033	-5.0064	1.9244	23.2061	-2.6069
3.4500	-0.0087	0.0045	-0.0032	-4.8386	1.8598	23.0976	-2.5188
3.5000	-0.0084	0.0044	-0.0031	-4.6763	1.7972	22.9928	-2.4336
3.5500	-0.0081	0.0042	-0.0030	-4.5192	1.7367	22.8914	-2.3512
3.6000	-0.0078	0.0041	-0.0029	-4.3672	1.6782	22.7934	-2.2716
3.6500	-0.0076	0.0039	-0.0028	-4.2202	1.6216	22.6987	-2.1946
3.7000	-0.0073	0.0038	-0.0027	-4.0781	1.5669	22.6072	-2.1202
3.7500	-0.0071	0.0037	-0.0026	-3.9406	1.5140	22.5188	-2.0483
3.8000	-0.0068	0.0036	-0.0025	-3.8076	1.4628	22.4333	-1.9788
3.8500	-0.0066	0.0034	-0.0024	-3.6791	1.4133	22.3507	-1.9117
3.9000	-0.0064	0.0033	-0.0023	-3.5547	1.3655	22.2708	-1.8468
3.9500	-0.0062	0.0032	-0.0022	-3.4345	1.3193	22.1937	-1.7840
4.0000	-0.0059	0.0031	-0.0022	-3.3183	1.2746	22.1191	-1.7234
4.0500	-0.0057	0.0030	-0.0021	-3.2060	1.2314	22.0471	-1.6649
4.1000	-0.0055	0.0029	-0.0020	-3.0974	1.1897	21.9775	-1.6083
4.1500	-0.0054	0.0028	-0.0019	-2.9925	1.1493	21.9102	-1.5536
4.2000	-0.0052	0.0027	-0.0019	-2.8910	1.1103	21.8453	-1.5008
4.2500	-0.0050	0.0026	-0.0018	-2.7930	1.0726	21.7825	-1.4497
4.3000	-0.0048	0.0025	-0.0017	-2.6982	1.0362	21.7218	-1.4004
4.3500	-0.0047	0.0024	-0.0017	-2.6066	1.0010	21.6632	-1.3528
4.4000	-0.0045	0.0023	-0.0016	-2.5181	0.9670	21.6066	-1.3067
4.4500	-0.0044	0.0023	-0.0016	-2.4326	0.9341	21.5519	-1.2622
4.5000	-0.0042	0.0022	-0.0015	-2.3500	0.9024	21.4991	-1.2193
4.5500	-0.0041	0.0021	-0.0015	-2.2701	0.8717	21.4480	-1.1778
4.6000	-0.0039	0.0020	-0.0014	-2.1929	0.8421	21.3987	-1.1377
4.6500	-0.0038	0.0020	-0.0014	-2.1184	0.8134	21.3511	-1.0989
4.7000	-0.0037	0.0019	-0.0013	-2.0463	0.7857	21.3050	-1.0615
4.7500	-0.0035	0.0018	-0.0013	-1.9767	0.7590	21.2606	-1.0253
4.8000	-0.0034	0.0018	-0.0012	-1.9095	0.7332	21.2176	-0.9904
4.8500	-0.0033	0.0017	-0.0012	-1.8445	0.7082	21.1761	-0.9566
4.9000	-0.0032	0.0016	-0.0011	-1.7817	0.6841	21.1361	-0.9241
4.9500	-0.0031	0.0016	-0.0011	-1.7211	0.6608	21.0973	-0.8926
5.0000	-0.0030	0.0015	-0.0011	-1.6625	0.6383	21.0599	-0.8621
5.0500	-0.0029	0.0015	-0.0010	-1.6059	0.6166	21.0238	-0.8328

Sample Output, Computer Program D

The following is a sample output of computer program D, with states X(1) = N_L (Normal acceleration), X(2) = θ (Pitch angle), and X(3) = $\dot{\theta}$ (Pitch rate). The feedbacks specified by this run (CONTROL = RIBTC, referred to as controller 61) are similar to those used for flight condition one testing with the six degree of freedom simulation at the Flight Dynamics Laboratory.

```
Q MATRIX

  1.0000      -0.          -0.
 -0.          40.0000      -0.
 -0.          -0.          20.0000

R MATRIX

  0.0400      -0.          -0.
 -0.           0.0400      -0.
 -0.          -0.           0.0400

Q MATRIX CHECK

COEFFICIENTS CF CHARACTERISTIC EQUATION

A( 1)=  1.000000E 00
A( 2)= -6.100000E 01
A( 3)=  8.600000E 02
A( 4)= -8.000000E 02

ROOTS OF CHARACTERISTIC EQUATION

S=  9.9999998E-01   +J
S=  3.9999999E 01   +J  0.
S=  2.0000000E 01   +J  0.

R MATRIX CHECK

COEFFICIENTS CF CHARACTERISTIC EQUATION

A( 1)=  1.000000E 00
A( 2)= -1.200000E-01
A( 3)=  4.800000E-03
A( 4)= -6.399999E-05
```

ROOTS OF CHARACTERISTIC EQUATION

S= 4.0048018E-02 +J 8.2612640E-05
S= 4.0048018E-02 +J -8.2612640E-05
S= 3.9912091E-02 +J

FLIGHT CONDITION 1

A MATRIX

-5.1924	0.	5.3076
0.	0.	1.0000
-2.8067	0.	-1.4612

B MATRIX

-0.3324	0.2114	-0.1043
0.	0.	0.
-1.8858	0.5596	-0.1282

TIME = 7.50

FEEDBACK MATRIX C

0.1117	-0.0825	-0.0339
-0.0825	17.6124	0.3873
-0.0339	0.3873	0.4499

TIME = 7.00

FEEDBACK MATRIX C

0.1132	-0.1582	-0.0357
-0.1582	26.2411	0.5899
-0.0357	0.5899	0.4546

TIME = 6.50

FEEDBACK MATRIX C

0.1135	-0.1834	-0.0363
-0.1834	29.0097	0.6550
-0.0363	0.6550	0.4562

TIME = 6.00

FEEDBACK MATRIX C

0.1135	-0.1904	-0.0365
-0.1904	29.7697	0.6729
-0.0365	0.6729	0.4566

TIME = 5.50

FEEDBACK MATRIX C

0.1135	-0.1922	-0.0365
-0.1922	29.9691	0.6776
-0.0365	0.6776	0.4567

NOTE ... ACTUAL STATE X(1) IS .01*N SUB L.

CONTROL= -RIBTC

-0.7777	30.3457	21.2263
-0.0892	-8.4637	-6.1963
0.1791	1.6701	1.3683

A-BRIBTC

-4.9714	-12.0495	-3.2001
0.	0.	1.0000
-1.4130	-62.1751	-45.1317

COEFFICIENTS OF CHARACTERISTIC EQUATION

A(1)= 1.000000E 00
A(2)= 5.010310E 01
A(3)= 2.820211E 02
A(4)= 2.920704E 02

ROOTS OF CHARACTERISTIC EQUATION

S= -4.9326298E 00 +J 0.
S= -1.3512778E 00 +J 0.
S= -4.3819191E 01 +J

NOTE, CONTROLS IN DEGREES AND ANGLES IN RADIANS.

SPOILERS BIASED UP 20 DEGREES.

TIME	N ACCEL	THETA	THETA DOT	ELEVATOR	SPOILERS	AILERONS
0.	-0.	0.1000	-0.	0.	20.0000	0.
0.0500	-4.0531	0.0959	-0.1196	23.1158	16.1555	-0.6143
0.1000	-6.3444	0.0897	-0.1241	7.8861	20.8761	-1.7951
0.1500	-7.8591	0.0837	-0.1165	7.2960	21.1865	-1.9333
0.2000	-8.8454	0.0781	-0.1083	7.9512	21.0469	-1.9286
0.2500	-9.4387	0.0728	-0.1006	8.4492	20.8910	-1.8888
0.3000	-9.7371	0.0680	-0.0936	8.7072	20.7600	-1.8318
0.3500	-9.8160	0.0635	-0.0871	8.7730	20.6530	-1.7639
0.4000	-9.7335	0.0593	-0.0811	8.6960	20.5654	-1.6891
0.4500	-9.5342	0.0553	-0.0756	8.5155	20.4934	-1.6104
0.5000	-9.2525	0.0517	-0.0705	8.2621	20.4337	-1.5299
0.5500	-8.9147	0.0483	-0.0658	7.9591	20.3840	-1.4492
0.6000	-8.5406	0.0451	-0.0614	7.6241	20.3422	-1.3696
0.6500	-8.1454	0.0421	-0.0573	7.2706	20.3067	-1.2919
0.7000	-7.7405	0.0394	-0.0535	6.9085	20.2765	-1.2167
0.7500	-7.3342	0.0368	-0.0499	6.5455	20.2504	-1.1444
0.8000	-6.9327	0.0344	-0.0466	6.1868	20.2278	-1.0753
0.8500	-6.5405	0.0321	-0.0435	5.8365	20.2080	-1.0094
0.9000	-6.1606	0.0300	-0.0407	5.4973	20.1906	-0.9469
0.9500	-5.7952	0.0281	-0.0380	5.1711	20.1752	-0.8877
1.0000	-5.4456	0.0262	-0.0355	4.8590	20.1614	-0.8318
1.0500	-5.1124	0.0245	-0.0332	4.5616	20.1491	-0.7791
1.1000	-4.7961	0.0229	-0.0310	4.2793	20.1379	-0.7295
1.1500	-4.4966	0.0214	-0.0290	4.0120	20.1278	-0.6829
1.2000	-4.2136	0.0200	-0.0271	.7595	20.1186	-0.6390
1.2500	-3.9468	0.0187	-0.0253	3.5214	20.1101	-0.5979

TIME	N ACCEL	THETA	THETA DOT	ELEVATOR	SPOILERS	AILERONS
1.3000	-3.6955	0.0175	-0.0236	3.2972	20.1024	-0.5593
1.3500	-3.4592	0.0163	-0.0221	3.0863	20.0953	-0.5231
1.4000	-3.2373	0.0153	-0.0206	2.8883	20.0888	-0.4892
1.4500	-3.0289	0.0143	-0.0193	2.7024	20.0827	-0.4575
1.5000	-2.8335	0.0133	-0.0180	2.5280	20.0771	-0.4278
1.5500	-2.6503	0.0125	-0.0168	2.3645	20.0719	-0.4000
1.6000	-2.4787	0.0117	-0.0157	2.2114	20.0671	-0.3740
1.6500	-2.3179	0.0109	-0.0147	2.0680	20.0626	-0.3496
1.7000	-2.1674	0.0102	-0.0138	1.9337	20.0585	-0.3268
1.7500	-2.0265	0.0095	-0.0129	1.8080	20.0546	-0.3055
1.8000	-1.8947	0.0089	-0.0120	1.6904	20.0510	-0.2856
1.8500	-1.7713	0.0083	-0.0112	1.5803	20.0476	-0.2670
1.9000	-1.6559	0.0078	-0.0105	1.4774	20.0445	-0.2496
1.9500	-1.5480	0.0073	-0.0098	1.3811	20.0415	-0.2333
2.0000	-1.4471	0.0068	-0.0092	1.2910	20.0388	-0.2181
2.0500	-1.3527	0.0063	-0.0086	1.2068	20.0363	-0.2038
2.1000	-1.2645	0.0059	-0.0080	1.1281	20.0339	-0.1905
2.1500	-1.1819	0.0055	-0.0075	1.0545	20.0317	-0.1781
2.2000	-1.1048	0.0052	-0.0070	0.9857	20.0296	-0.1665
2.2500	-1.0327	0.0048	-0.0065	0.9213	20.0276	-0.1556
2.3000	-0.9653	0.0045	-0.0061	0.8612	20.0258	-0.1454
2.3500	-0.9022	0.0042	-0.0057	0.8049	20.0241	-0.1359
2.4000	-0.8433	0.0040	-0.0053	0.7524	20.0226	-0.1270
2.4500	-0.7883	0.0037	-0.0050	0.7033	20.0211	-0.1187
2.5000	-0.7368	0.0035	-0.0047	0.6573	20.0197	-0.1110
2.5500	-0.6887	0.0032	-0.0044	0.6144	20.0184	-0.1037
2.6000	-0.6437	0.0030	-0.0041	0.5743	20.0172	-0.0970
2.6500	-0.6016	0.0028	-0.0038	0.5368	20.0161	-0.0906
2.7000	-0.5623	0.0026	-0.0036	0.5017	20.0150	-0.0847
2.7500	-0.5256	0.0025	-0.0033	0.4689	20.0141	-0.0792
2.8000	-0.4913	0.0023	-0.0031	0.4383	20.0131	-0.0740
2.8500	-0.4592	0.0022	-0.0029	0.4097	20.0123	-0.0692
2.9000	-0.4292	0.0020	-0.0027	0.3829	20.0115	-0.0646

TIME	N ACCEL	THETA	THETA DOT	ELEVATOR	SPOILERS	AILERONS
2.9500	-0.4011	0.0019	-0.0025	0.3579	20.0107	-0.0604
3.0000	-0.3749	0.0018	-0.0024	0.3345	20.0100	-0.0565
3.0500	-0.3504	0.0016	-0.0022	0.3126	20.0094	-0.0528
3.1000	-0.3275	0.0015	-0.0021	0.2922	20.0088	-0.0493
3.1500	-0.3061	0.0014	-0.0019	0.2731	20.0082	-0.0461
3.2000	-0.2861	0.0013	-0.0018	0.2553	20.0077	-0.0431
3.2500	-0.2675	0.0013	-0.0017	0.2386	20.0072	-0.0403
3.3000	-0.2500	0.0012	-0.0016	0.2230	20.0067	-0.0377
3.3500	-0.2336	0.0011	-0.0015	0.2085	20.0062	-0.0352
3.4000	-0.2184	0.0010	-0.0014	0.1948	20.0058	-0.0329
3.4500	-0.2041	0.0010	-0.0013	0.1821	20.0055	-0.0307
3.5000	-0.1908	0.0009	-0.0012	0.1702	20.0051	-0.0287
3.5500	-0.1783	0.0008	-0.0011	0.1591	20.0048	-0.0269
3.6000	-0.1667	0.0008	-0.0011	0.1487	20.0045	-0.0251
3.6500	-0.1558	0.0007	-0.0010	0.1390	20.0042	-0.0235
3.7000	-0.1456	0.0007	-0.0009	0.1299	20.0039	-0.0219
3.7500	-0.1361	0.0006	-0.0009	0.1214	20.0036	-0.0205
3.8000	-0.1272	0.0006	-0.0008	0.1135	20.0034	-0.0192
3.8500	-0.1189	0.0006	-0.0008	0.1061	20.0032	-0.0179
3.9000	-0.1111	0.0005	-0.0007	0.0991	20.0030	-0.0167
3.9500	-0.1039	0.0005	-0.0007	0.0927	20.0028	-0.0156
4.0000	-0.0971	0.0005	-0.0006	0.0866	20.0026	-0.0146
4.0500	-0.0907	0.0004	-0.0006	0.0809	20.0024	-0.0137

Appendix B

Digital Computer Subroutines

```
$IBFTC SUB1    DECK,XR7
      SUBROUTINE FCN(KP,J2,X,F)
C     FCN(NO. EQN,EQN SET,VAR. VALUE, VAR. DERIV)
      COMMON A(3,3),B(3,3),D(16,3),Q(3,3),R(3,3),C(3,3),WKA(3,3),
     ZWKB(3,3),WKC(3,3),WKD(3,3),RIBT(3,3),M,N,BRIBT(3,3)
      DIMENSION F(6),X(6),CD(3,3),XX(3,1),DUM(3,1)
      GO TO (1,2,3),J2
C  MATRIX C IS SYMMETRIC PER ATHENS AND FALB
 1    CONTINUE
 2    C(1,1)=X(1)
      C(1,2)=X(2)
      C(1,3)=X(3)
      C(2,1)=X(2)
      C(2,2)=X(4)
      C(2,3)=X(5)
      C(3,1)=X(3)
      C(3,2)=X(5)
      C(3,3)=X(6)
      IF(J2.EQ.2)  GO TO 10
      CALL GMPRD(BRIBT,C,WKB,N,N,N)
      CALL GMPRD(C,WKB,WKC,N,N,N)
      CALL MXPM(WKC,Q,WKB,N,N,-1)
      CALL GMTRA(A,WKC,N,N)
      CALL GMPRD(WKC,C,WKA,N,N,N)
      CALL MXPM(WKB,WKA,WKD,N,N,-1)
      CALL GMPRD(C,A,WKB,N,N,N)
      CALL MXPM(WKD,WKB,CD,N,N,-1)
      F(1)=CD(1,1)
      F(2)=CD(1,2)
      F(3)=CD(1,3)
      F(4)=CD(2,2)
      F(5)=CD(2,3)
      F(6)=CD(3,3)
 10   CONTINUE
      RETURN
 3    DO 4 I=1,N
 4    XX(I,1)=X(I)
      CALL GMPRD(WKA,XX,DUM,N,N,1)
      DO 5  I=1,N
 5    F(I)=DUM(I,1)
      RETURN
      END
```

```
$IBFTC SUB2     DECK,XR7
      SUBROUTINE RUNGE(N,J2,Y,X,DX)
C CALL (NO. EQN, EQN SET, VARIABLE, START TIME, DEL TIME)
      DIMENSION Y(6),YO(6),YT(6),YP(6),PO(6),P1(6),P2(6),P3(6)
 1    XO=X
      X=X+DX
      H=DX
 2    IF(ABS(H).GT.ABS(X-XO)) H=X-XO
      DO 4 I=1,N
 4    YO(I)=Y(I)
      HT=H
      XT=XO
      RMAXP = 1.E37
      DO 5 I=1,N
 5    YT(I)=YO(I)
      ASSIGN 6 TO K
      GO TO 20
 6    DO 7 I=1,N
 7    YP(I)=Y(I)
 8    HT=0.5*H
      ASSIGN 9 TO K
      GO TO 20
 9    DO 10 I=1,N
 10   YT(I)=Y(I)
      XT=XO+HT
      ASSIGN 11 TO K
 20   CALL FCN(N,J2,XT,PO)
      DO 21 I=1,N
 21   Y(I)=YT(I)+0.5*HT*PO(I)
      CALL FCN(N,J2,Y,P1)
      DO 22 I=1,N
 22   Y(I)=YT(I)+.5*HT*P1(I)
      CALL FCN(N,J2,Y,P2)
      DO 23 I=1,N
 23   Y(I)=YT(I)+HT*P2(I)
      CALL FCN(N,J2,Y,P3)
      DO 24 I=1,N
 24   Y(I)=YT(I)+HT*(PO(I)+2.*(P1(I)+P2(I))+P3(I))/6.
      GO TO K,(6,9,11)
 11   RMAX=0.
      DO 12 I=1,N
 12   RMAX=AMAX1(RMAX,.07*ABS((Y(I)-YP(I))/Y(I)))
      IF ((RMAX.GT.1.E-06).AND.(RMAX.LT.RMAXP)) GO TO 17
      XO=XO+H
      IF(XO.EQ.X) RETURN
      IF((RMAX.LT.1.E-7).OR.(RMAX.GT.RMAXP)) H=H+H
      GO TO 2
 17   H=HT
      XT=XO
      DO 19 I=1,N
 18   YP(I)=YT(I)
 19   YT(I)=YO(I)
      RMAXP = RMAX
      GO TO 8
      END
```

```
$IBFTC SUB3     DECK,XR7
      SUBROUTINE MXPM(A,B,C,M,N,NSGN)
C   A + OR - B = C.   M IS NO. ROWS.   N IS NO. COLS.   NSGN INCIDATES + OR
      DIMENSION A(M,N),B(M,N),C(M,N)
      IF(NSGN.LT.0) GO TO 3
      DO1 I=1,M
      DO 1 J=1,N
 1    C(I,J)=A(I,J)+B(I,J)
      RETURN
  3   DO 2 I=1,M
      DO 2 J=1,N
 2    C(I,J)=A(I,J)-B(I,J)
      RETURN
      END
```

```
$IBFTC SUB4     DECK,XR7
      SUBROUTINE GMTRA(A,B,M,N)
C   B=A TRANSPOSE.   M = NO. ROWS IN A = NO. COLS IN B.
C   N = NO. COLS IN A = NO. ROWS IN B.
      DIMENSION A(M,N),B(N,M)
      DO 3 I=1,M
      DO 3 J=1,N
 3    B(J,I)=A(I,J)
      RETURN
      END
```

```
$IBFTC SUB5     DECK,XR7
      SUBROUTINE GMPRD(A,B,C,NA,NB,NC)
C   A*B=C   NA=NO. ROWS IN A=NO. ROWS IN C.   NB=NO. COLS IN A=NO. ROWS IN
C   NC= NO. COLS IN B=NO. COLS IN C.
      DIMENSION A(NA,NB),B(NB,NC),C(NA,NC)
      DO 7 I=1,NA
      DO 7 J=1,NC
      SUM =0.
      DO 5 L=1,NB
 5    SUM=SUM+A(I,L)*B(L,J)
 7    C(I,J)=SUM
      RETURN
      END
```

```
$IBFTC SUB6    DECK,XR7
      SUBROUTINE MTXINV(A,B,N)
C           MATRIX INVERSION SUBROUTINE
C  B=A INVERSE.  N IS ORDER OF MATRICES.
      DIMENSION A(N,N),B(N,N),K(100),P(100)
      NN=N
 20   DO 30 I=1,NN
      K(I)=I
      DO 30 J=1,NN
 30   B(I,J)=A(I,J)
      DO 330 I=1,NN
      I2=NN-I+1
      PE=0.
      DO 120 I1=1,I2
      TPE=B(I1,1)
      IF (ABS(PE)-ABS(TPE)) 100,100,120
 100  PE=TPE
      IP=I1
 120  CONTINUE
      IF (PE) 160,510,160
 160  DO 170 J=2,NN
 170  P(J-1)=B(IP,J)/PE
      P(NN)=1.0/PE
      IP=K(IP)
      I2=0
      DO 310 J=1,NN
      I1=J-I2
      K(I1)=K(J)
      IF (K(J)-IP) 260,250,260
 250  I2=1
      GO TO 310
 260  TPE=-B(J,1)
      DO 300 J1=2,NN
 300  B(I1,J1-1)=B(J,J1)+TPE*P(J1-1)
      B(I1,NN)=TPE*P(NN)
 310  CONTINUE
      DO 320 J=1,NN
 320  B(NN,J)=P(J)
 330  K(NN)=IP
      DO 410 I=1,NN
      DO 400 J=1,NN
      I1=K(J)
 400  P(I1)=B(I,J)
      DO 410 J=1,NN
 410  B(I,J)=P(J)
      RETURN
 510  WRITE (6,1001)
      CALL FXEM(510)
      RETURN
 1001 FORMAT (40H0SUBROUTINE MINV FINDS MATRIX A SINGULAR)
      END
```

```
$IBFTC SUB7    DECK,XR7
      SUBROUTINE CHRPOL (A,B,N)
C  FIND CHARACTERITIC EQUATION OF REAL MATRIX (A) BY BOCHER FORMULA. RE
C  234 OF DERUSSO.
C     A IS GIVEN MATRIX, B HAS COEFFICIENTS OF CHARACTERISTIC EQUATION,
C     N IS ORDER OF A.
C     T(1) = SUM OF MAIN DIAGONAL ELEMENTS OF (A).  T(2) = MAIN DIAGONA
C     (A) SQUARE ...  P(E)=E(N+B(1)*E(N-1 + B(2)*E(N-2 +...+ B(N-1)*E +
      DIMENSION A(3,3),B(4),T(3),C(3,3),D(3,3)
C      COMPUTE T MATRIX
      DO 2 I=1,N
      DO2 J=1,N
 2    C(I,J) = A(I,J)
      SUM = 0.
      DO 3 I=1,N
 3    SUM = SUM + A(I,I)
      T(1) = SUM
      DO 4 K=2,N
      CALL GMPRD (A,C,D,N,N,N)
      SUM = 0.
      DO 5 L= 1,N
 5    SUM = SUM + D(L,L)
      T(K) = SUM
      DO 6 I=1,N
      DO 6 J = 1,N
 6    C(I,J)= D(I,J)
 4    CONTINUE
C     COMPUTE B COEFFICIENTS
      B(1) = 1.
      B(2) = -T(1)
      DO 7 K=2,N
      SUM = T(K)
      NP=K-1
      DO 8  L=1,NP
      NS=K-L
 8    SUM=SUM+B(L+1)*T(NS)
 7    B(K+1) = (-1./FLOAT(K))*SUM
      RETURN
      END


$IBFTC SUB8    DECK,XR7
      SUBROUTINE ROOTS(IDIOT,A,D)
C  IDIOT IS DEGREE OF POLYNOMIAL.  (A) HAS COEFF OF A(1)*X(N + A(2)*X(N
C     (D) HAS COMPLEX ROOTS OF POLYNOMIAL.
      DIMENSION A(4),B(4),C(3),D(3,2)
      M=IDIOT
      NO=1
      KK = 0
      M=M+1
      WRITE (6,111)
 111  FORMAT(//,1X,39HCOEFFICIENTS OF CHARACTERISTIC EQUATION,//)
```

```
      DO 109 I=1,M
109   WRITE(6,110) I,A(I)
110   FORMAT(1X,2HA(,I2,2H)=,1PE14.6)
      WRITE (6,112)
112   FORMAT(//,1X,32HROOTS OF CHARACTERISTIC EQUATION,//)
6     IF(A(M))76,8,76
8     RI=0.
      KK = KK + 1
      D(KK,1) = 0.
      D(KK,2) =0.
      WRITE(6,93) RI
      M=M-1
      GO TO 6
7     SUM=0.
      DO 12 I=1,M
      X=A(I)
      IF(X)9,12,10
9     X=0.-X
10    SUM=SUM+ALOG(X)
12    DOG=M
      DOG=1./DOG
      SUM=DOG*SUM
      CAT=EXP (SUM)
      DO 14 I=1,M
14    A(I)=A(I)/CAT
      X=ABS (A(2)/A(1))
      Y=ABS (A(M-1)/A(M))
18    IF(X-Y)19,22,22
19    DO 20 I=1,M
20    B(I)=A(I)
      J=M
      DO 21 I=1,M
      A(I)=B(J)
21    J=J-1
      NO=0-NO
      GO TO 23
22    NO=NO
23    TOL=1.0E-07
58    P=0.
      Q=0.
      R=0.
      IO=2
      B(1)=A(1)
      C(1)=A(1)
      DO 100 K=1,20
      B(2)=A(2)-(P*B(1))
      DO 25 I=3,M
25    B(I)=A(I)-((P*B(I-1))+(Q*B(I-2)))
      C(2)=B(2)-(P*C(1))
      J=M-2
      IF(J-3)26,27,27
27    DO 28 I=3,J
28    C(I)=B(I)-((P*C(I-1))+(Q*C(I-2)) )
26    C(M-1)=-((P*C(M-2))+(Q*C(M-3)) )
      DENOM=(C(M-2)*C(M-2))-(C(M-1)*C(M-3))
      IF(DENOM)29,30,29
```

```
30    B(M)=ABS(B(M))
40    POW=M-1
      POW=2./POW
      Q=B(M)**POW
      GO TO 44
29    PI=P+(((B(M-1 )*C(M-2))-(B(M)*C(M-3)))/DENOM)
      QI=Q+(((B(M)*C(M-2))-(B(M-1)*C(M-1)))/DENOM)
      IF (A(M-1))32,31,32
31    X=ABS (B(M-1))
34    IF(X-TOL)37,37,38
32    RAT1=ABS (B(M-1)/A(M-1))
41    IF(RAT1-TOL)37,37,38
37    RAT2=ABS (B(M)/A(M))
43    IF(RAT2-TOL)60,60,38
38    P=PI
      Q=QI
44    GO TO (100,50),IO
50    DO 51 I=2,M
51    B(I)=A(I)+R*B(I-1)
      N= M-1
      DO 52 I=2,N
52    C(I)= B(I)+R*C(I-1)
      IF(C(N))54,53,54
53    IO=1
      GO TO 100
54    IO=2
      RI=R-(B(M)/C(N))
      RAT3=ABS (B(M)/A(M))
56    IF(RAT3-TOL)80,80,100
100   R=RI
      TOL=TOL*10.
      IF(TOL-1.0E-03)58,59,59
59    WRITE(6,95) KK
      GO TO 81
60    IF(NO)61,61,62
61    PI=PI/QI
      QI=1.0/QI
62    RR1=-PI/2.
      RR2=RR1
      RAD=((PI*PI)/4.)-QI
      IF(RAD)63,64,65
63    CR1=SQRT (-RAD)
      CR2=-CR1
      GO TO 69
65    IF(PI)66,66,67
66    RR1=RR1+SQRT (RAD)
      GO TO 68
67    RR1=RR1-SQRT (RAD)
68    RR2=QI/RR1
64    CR1=0.
      CR2=0.
69    KK=KK+1
      D(KK,1) = RR1
      D(KK,2) = CR1
      KK = KK + 1
      D(KK,1) = RR2
```

```
      D(KK,2) = CR2
      WRITE (6,93) RR1, CR1
      WRITE (6,93) RR2,CR2
      M=M-2
      IF(M-1)81,81,82
 82   DO 70 I=2,M
 70   A(I)=B(I)
 76   IF(M-3)78,71,7
 71   PI=A(2)/A(1)
      QI=A(3)/A(1)
      GO TO 60
 80   IF(NO)73,73,74
 73   RI=1./RI
 74   KK = KK + 1
      D(KK,1) = RI
      D(KK,2) = 0.
      WRITE(6,93) RI
      M=M-1
      IF(M-1)81,81,83
 83   DO 79 I=2,M
 79   A(I)=B(I)
 78   IF(M-2)81,75,76
 75   RI=-A(2)/A(1)
      GO TO 80
 91   FORMAT(E28.8)
 93   FORMAT(1X,2HS=,1PE15.7,3X,2H+J,1PE15.7)
 95   FORMAT(36HROOTS SUBROUTINE FAILED AFTER ROOT  ,I2)
 81   CONTINUE
      RETURN
      END
```

Appendix C

Sample Derivation

Quasi-Elastic Aerodynamic Coefficients

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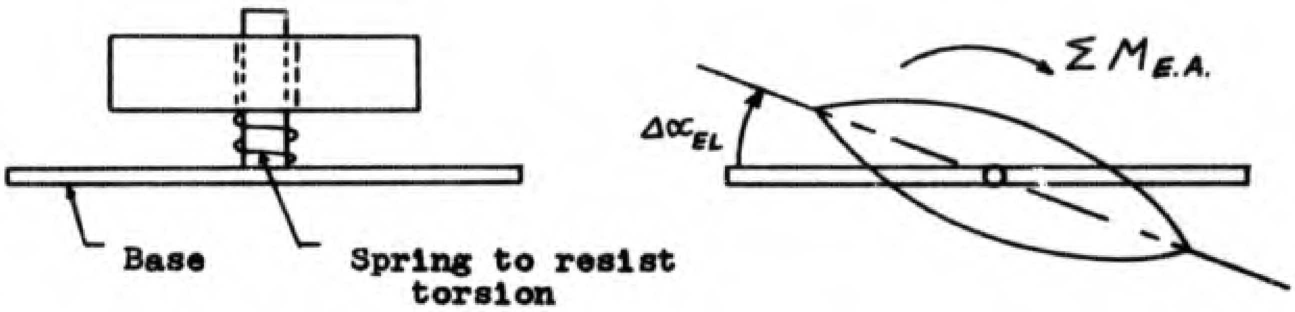
The following sample derivation of a quasi-elastic aerodynamic coefficient was provided by The Boeing Company. In the basic system, the only elastic degree of freedom is fin torsion. When accounting for spanwise variations of lift distribution and surface rigidity, the method becomes more complex. However, the form of the equations is very similar to that shown below.

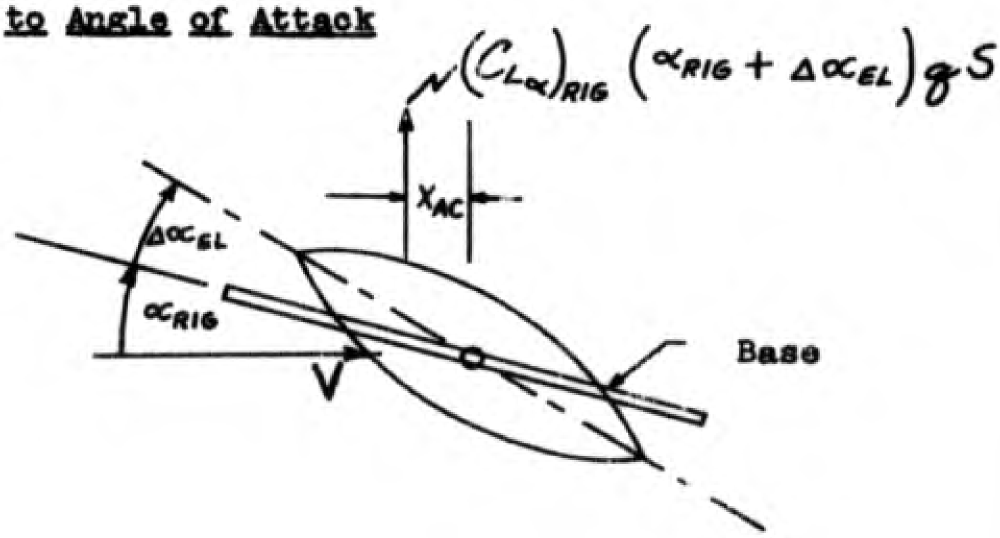


Let $1/K$ = Elastic Deflection in Deg/Ft-Lb

Then: $\sum M_{EA} = K \times \Delta\alpha_{EL}$

Or: $\Delta\alpha_{EL} = \dfrac{\sum M_{EA}}{K}$

<u>Lift Due to Angle of Attack</u>



Taking moments:

$$\Sigma M_{E.A.} = (C_{L\alpha})_{RIG}(\alpha_{RIG} + \Delta\alpha_{EL})\, q\, S\, X_{AC}$$

From the basic definition:

$$\Sigma M_{E.A.} = K\Delta\alpha_{EL}$$

Thus:

$$K\Delta\alpha_{EL} = (C_{L\alpha})_{RIG}(\alpha_{RIG} + \Delta\alpha_{EL})\, q\, S\, X_{AC}$$

Solving for

$$\Delta\alpha_{EL} = \frac{(C_{L\alpha})_{RIG}\,\alpha_{RIG}\, q\, S\, X_{AC}}{K - (C_{L\alpha})_{RIG}\, q\, S\, X_{AC}}$$

Working with lift equations:

By definition:

$$\Sigma L = (C_{L\alpha})_{EL}\,\alpha_{RIG}\, q\, S$$

From lift summation:

$$\Sigma L = (C_{L\alpha})_{RIG}(\alpha_{RIG} + \Delta\alpha_{EL})\, q\, S$$

Thus:

$$(C_{L\alpha})_{EL}\,\alpha_{RIG}\, q\, S = (C_{L\alpha})_{RIG}(\alpha_{RIG} + \Delta\alpha_{EL})\, q\, S$$

Or:

$$(C_{L\alpha})_{EL} = (C_{L\alpha})_{RIG}\frac{(\alpha_{RIG} + \Delta\alpha_{EL})}{\alpha_{RIG}}$$

Or:

$$\left(C_{L\alpha}\right)_{EL} = \left(C_{L\alpha}\right)_{RIG}\left[1 + \frac{\Delta\alpha_{EL}}{\alpha_{RIG}}\right]$$

From the moment equation:

$$\frac{\Delta\alpha_{EL}}{\alpha_{RIG}} = \frac{\left(C_{L\alpha}\right)_{RIG}\, q\, S\, X_{AC}}{K - \left(C_{L\alpha}\right)_{RIG}\, q\, S X_{AC}}$$

Substituting:

$$\left(C_{L\alpha}\right)_{EL} = \left(C_{L\alpha}\right)_{RIG}\left[1 + \frac{\left(C_{L\alpha}\right)_{RIG}\, q\, S X_{AC}}{K - \left(C_{L\alpha}\right)_{RIG}\, q\, S X_{AC}}\right]$$

Rearranging:

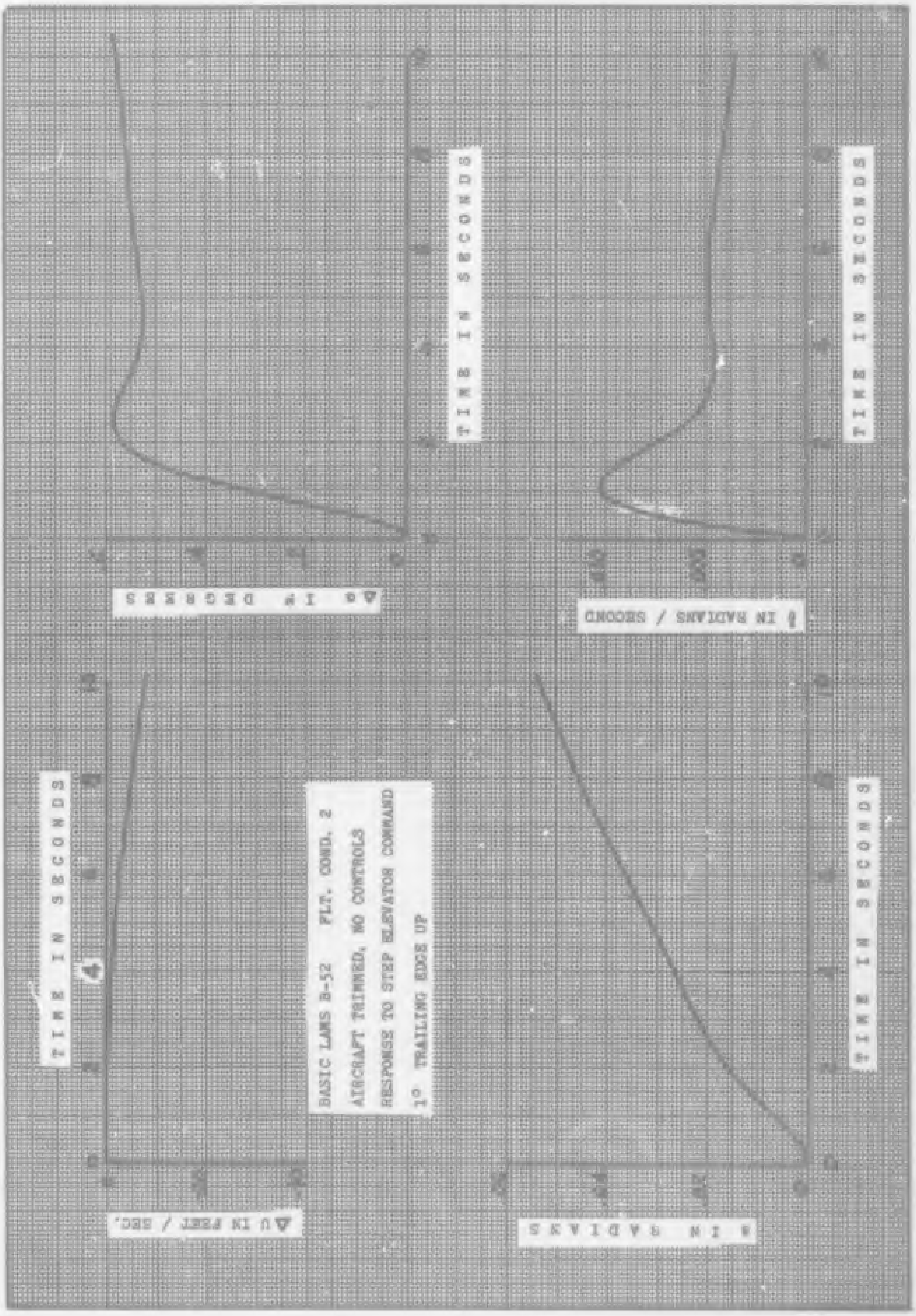
$$\left(C_{L\alpha}\right)_{EL} = \left(C_{L\alpha}\right)_{RIG}\left[\frac{1}{1 - \left(C_{L\alpha}\right)_{RIG}\, q\, S\, \frac{X_{AC}}{K}}\right]$$

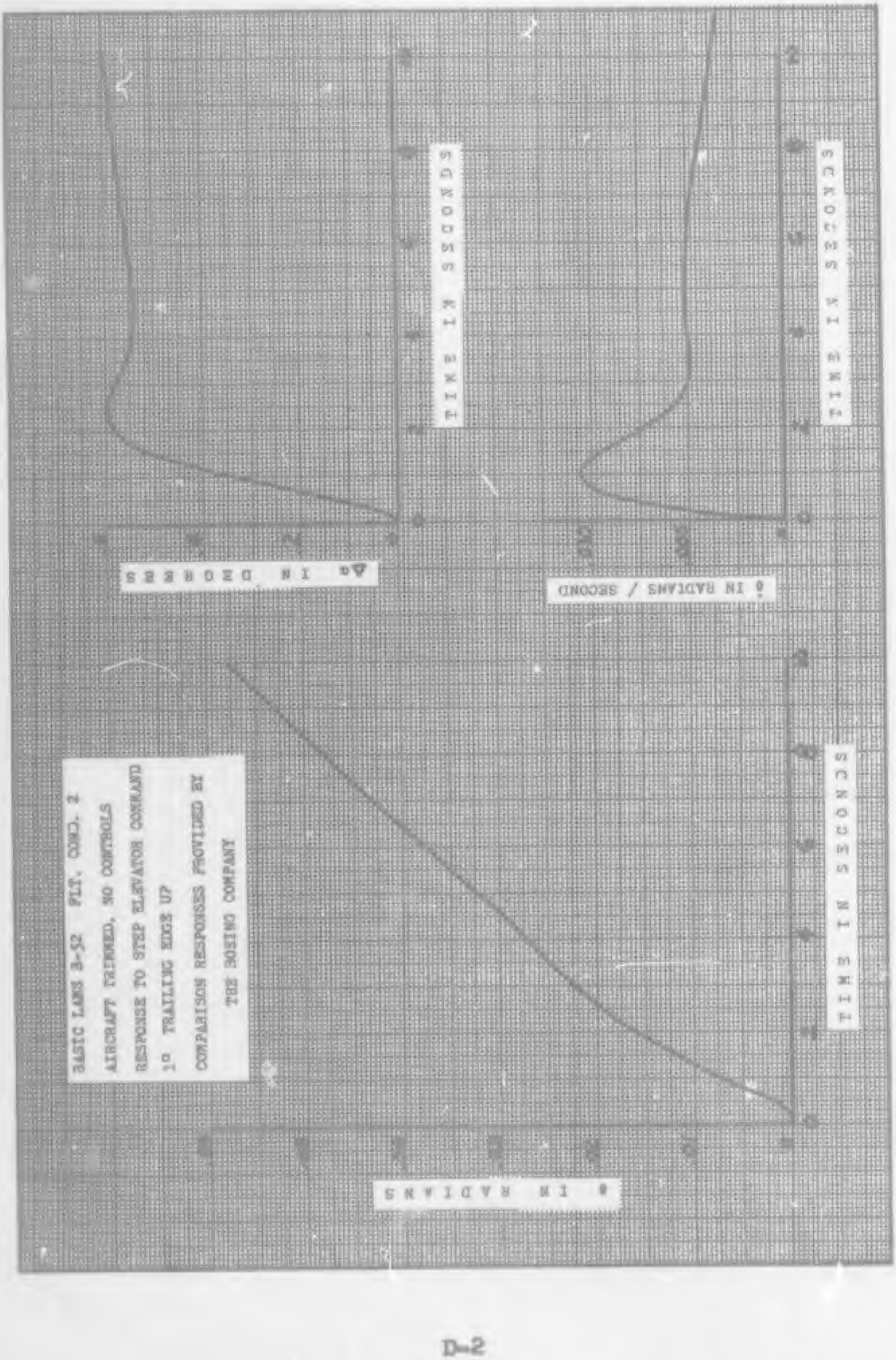
Appendix D

__Basic Aircraft Responses__

__Flight Conditions Two and Three__

__Six Degree of Freedom Simulation__


BASIC LAWS B-52 FLT. COND. 2
AIRCRAFT TRIMMED, NO CONTROLS
RESPONSE TO STEP ELEVATOR COMMAND
1° TRAILING EDGE UP
TIME IN SECONDS
Δα IN DEGREES
θ̇ IN RADIANS / SECOND
Δu IN FEET / SEC.
θ IN RADIANS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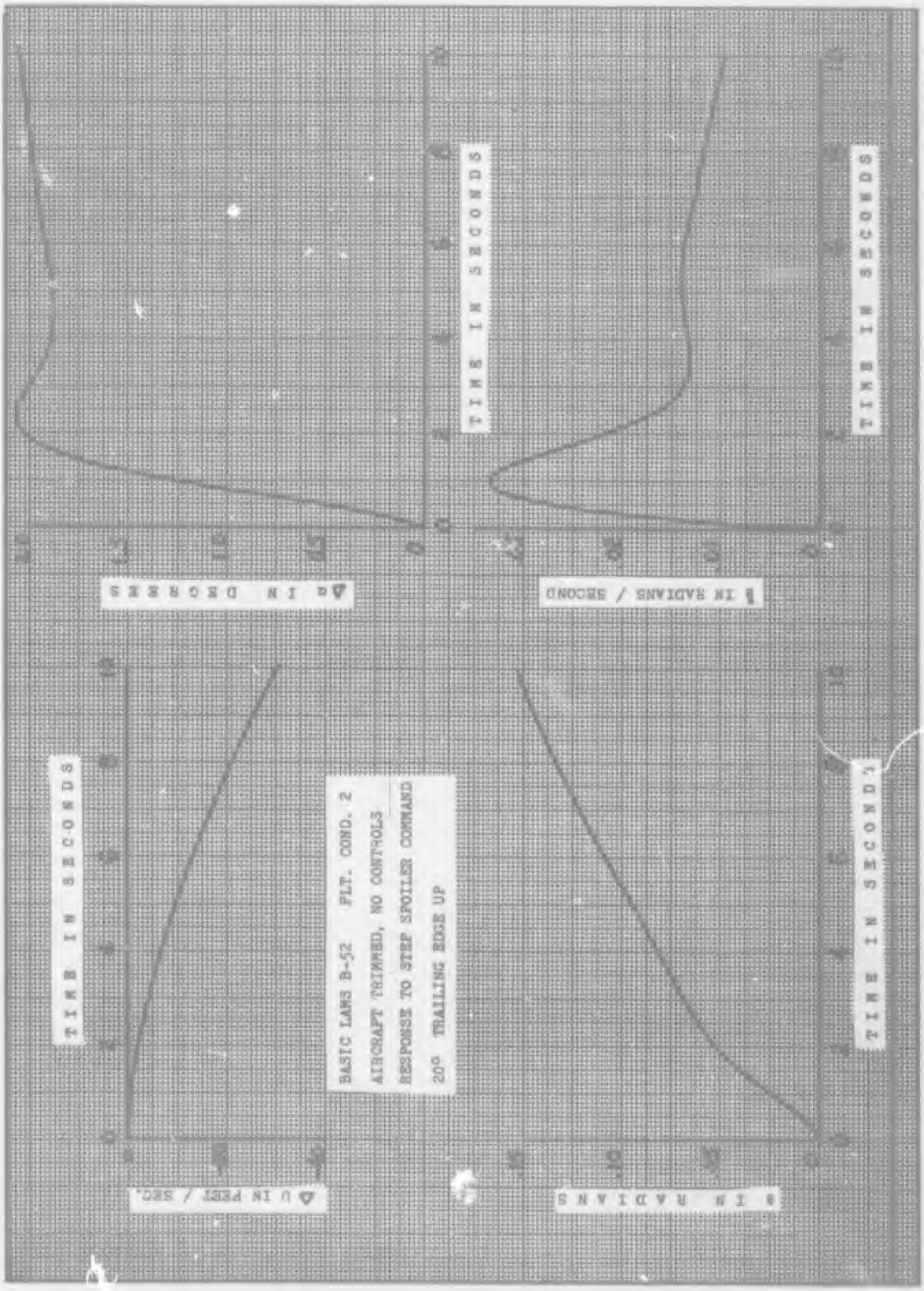
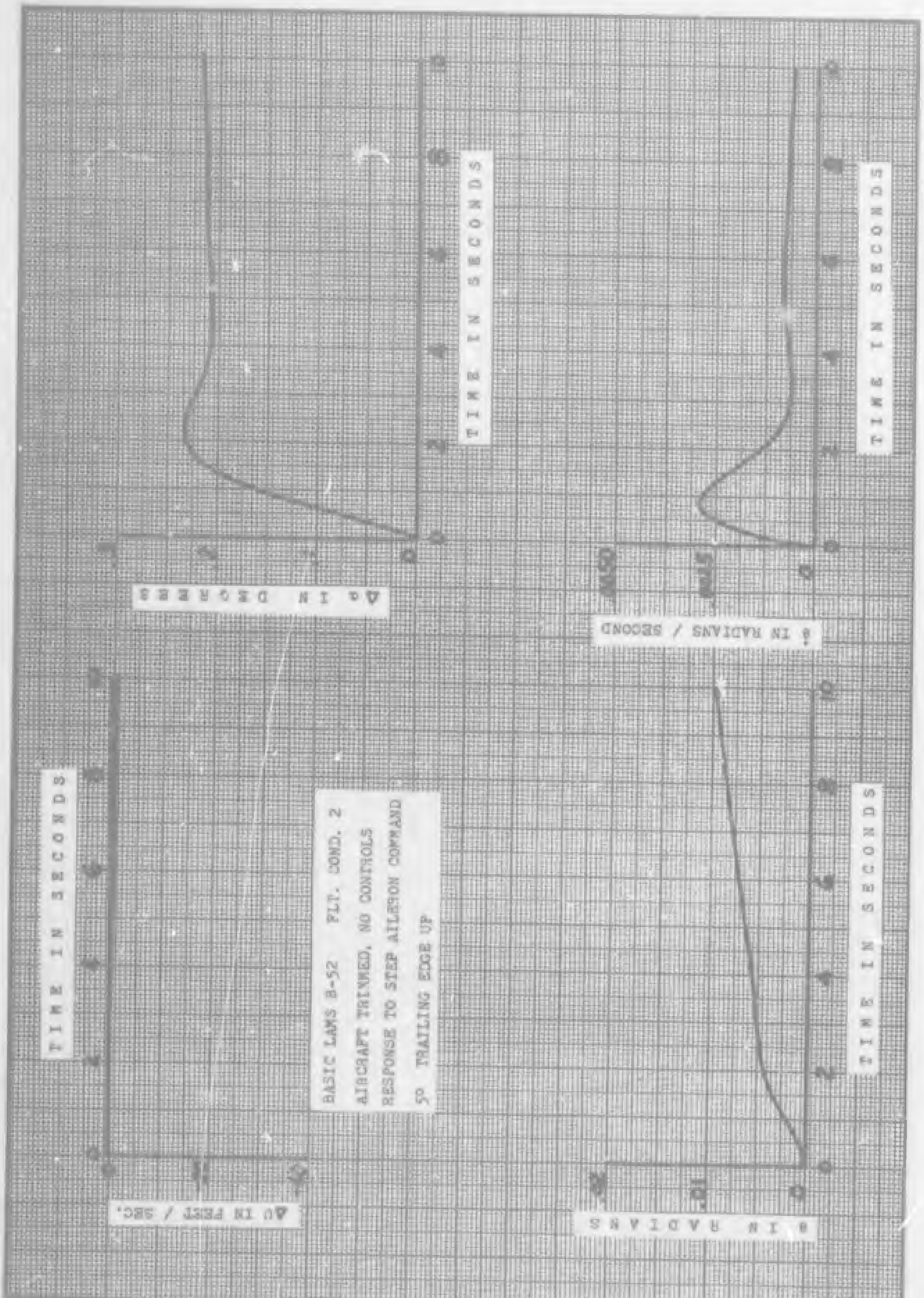
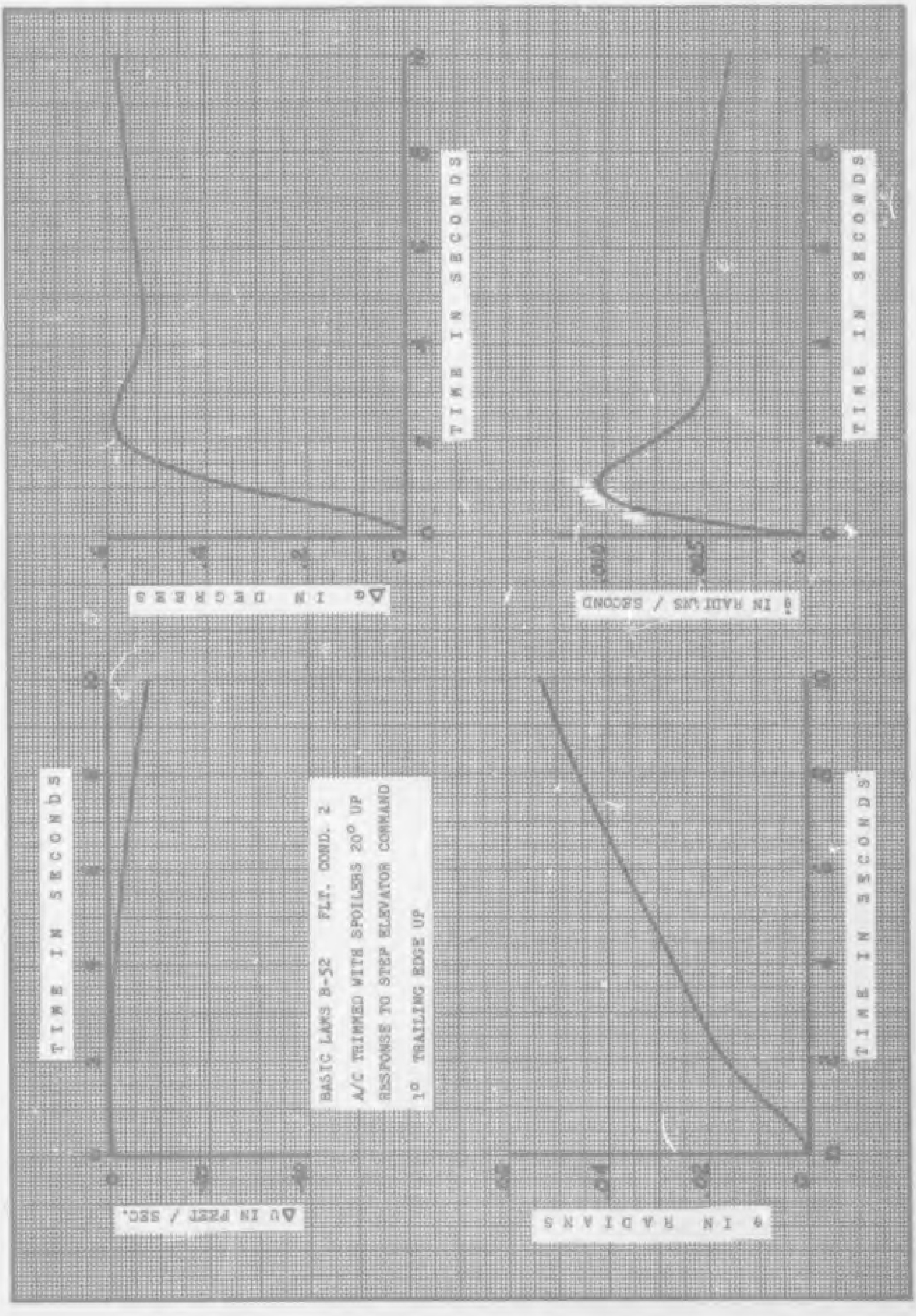
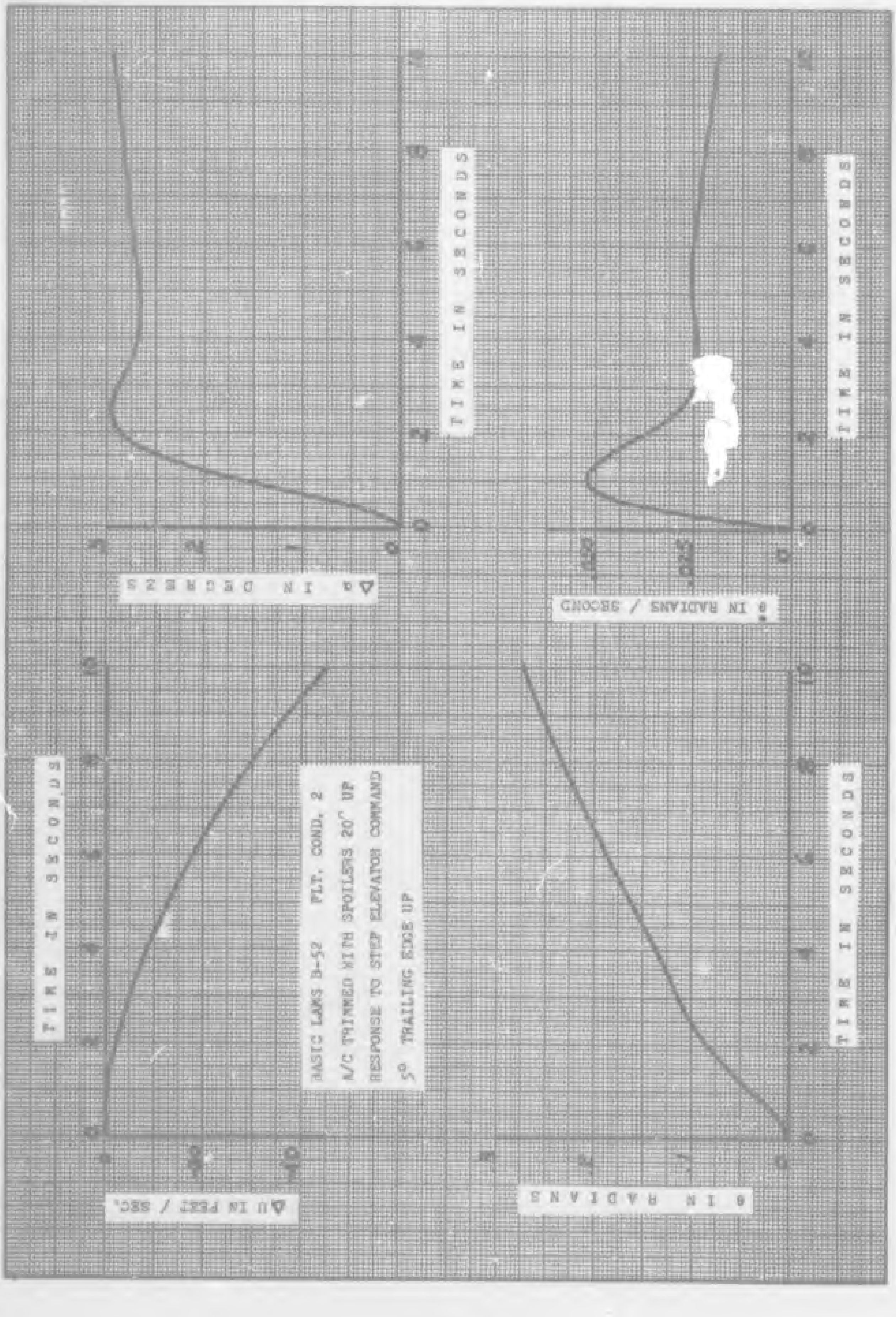
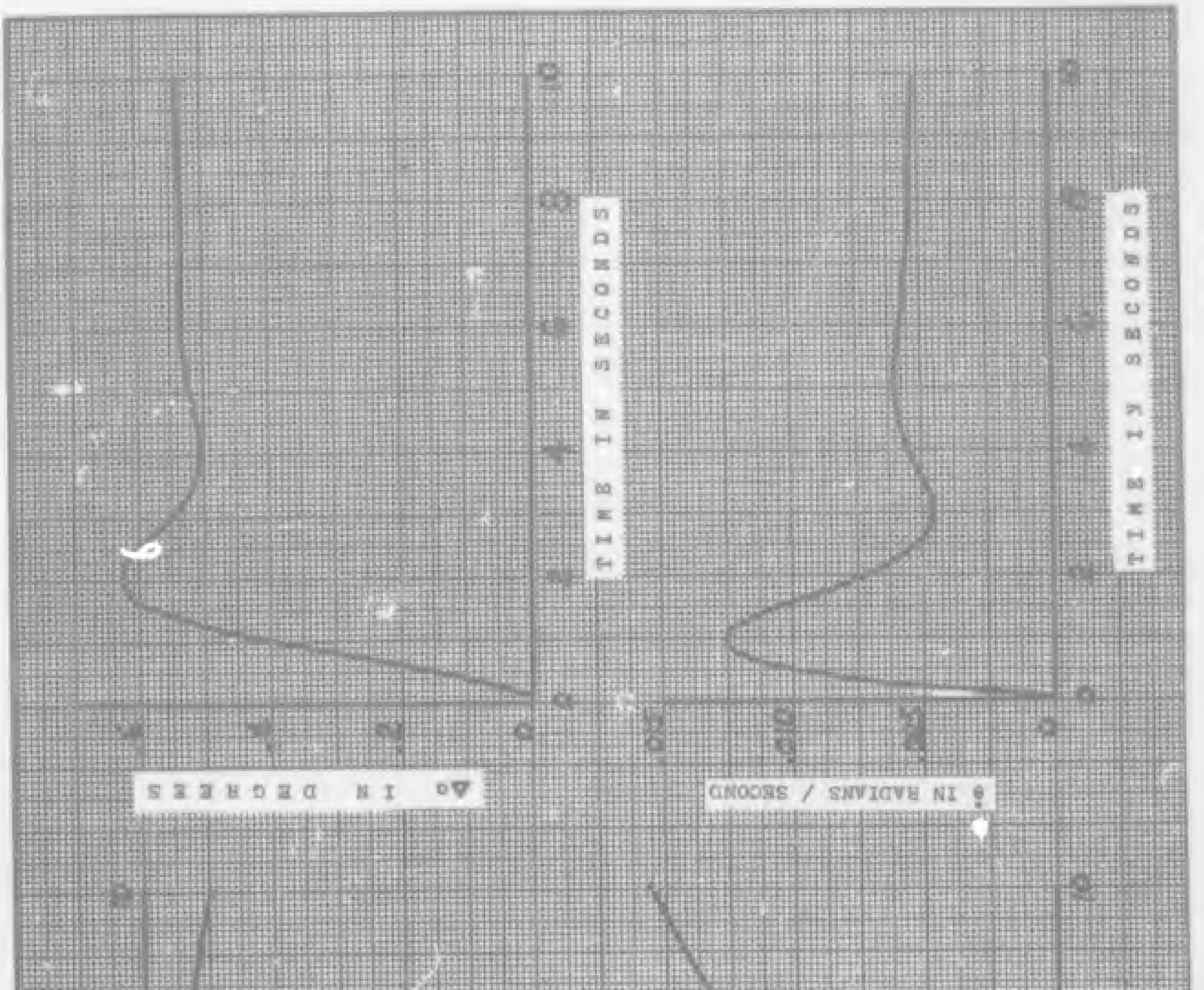
AIRCRAFT TRIMMED, NO CONTROLS
RESPONSE TO STEP ELEVATOR COMMAND
1° TRAILING EDGE UP
COMPARISON RESPONSES PROVIDED BY
THE BOEING COMPANY
Δα IN DEGREES
θ̇ IN RADIANS / SECOND
θ IN RADIANS
TIME IN SECONDS
TIME IN SECONDS
TIME IN SECONDS

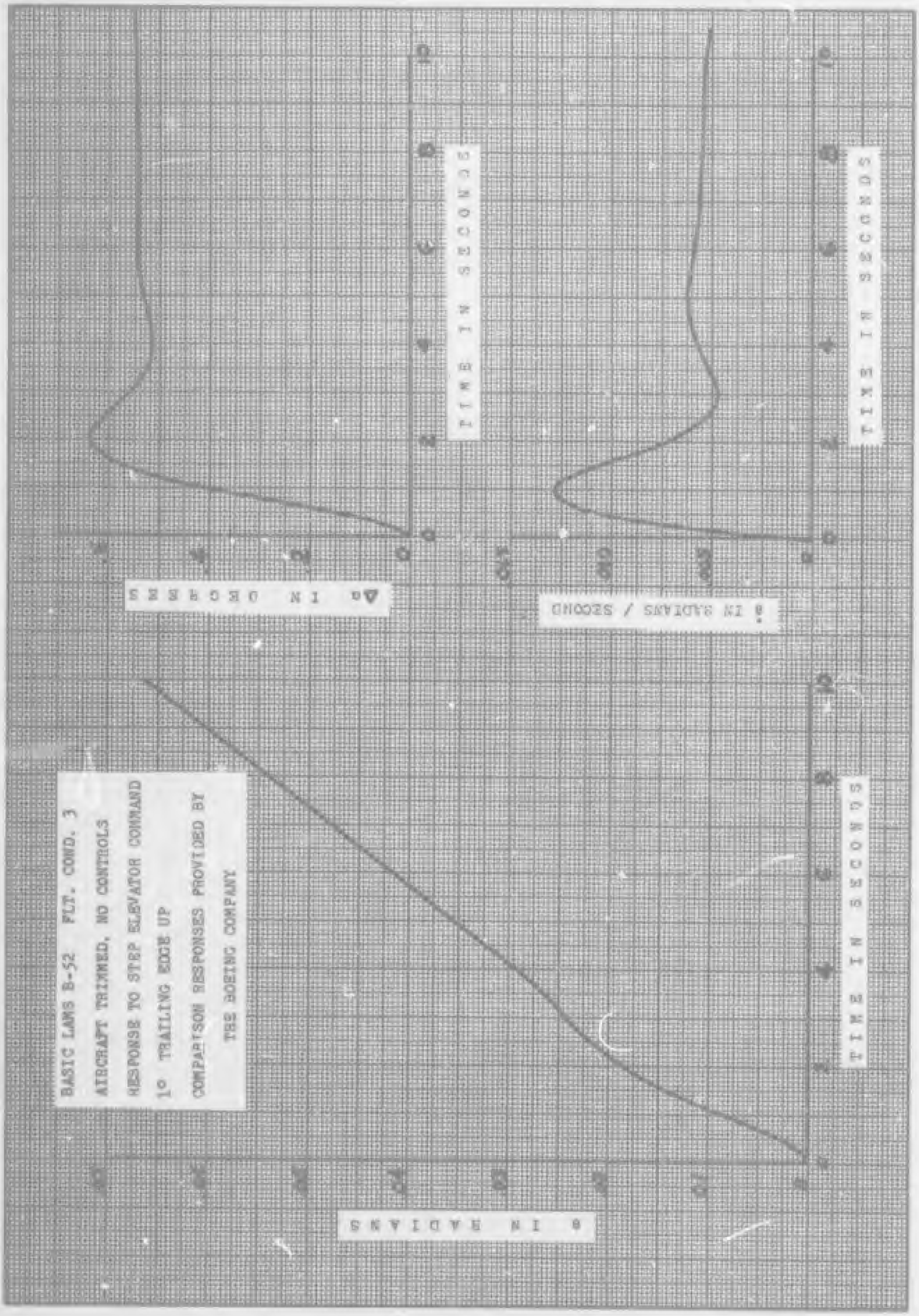

BASIC LAWS B-52 FLT. COND. 2
AIRCRAFT TRIMMED, NO CONTROLS
RESPONSE TO STEP SPOILER COMMAND
20° TRAILING EDGE UP
TIME IN SECONDS
Δα IN DEGREES
θ̇ IN RADIANS / SECOND
ΔU IN FEET / SEC.
θ IN RADIANS

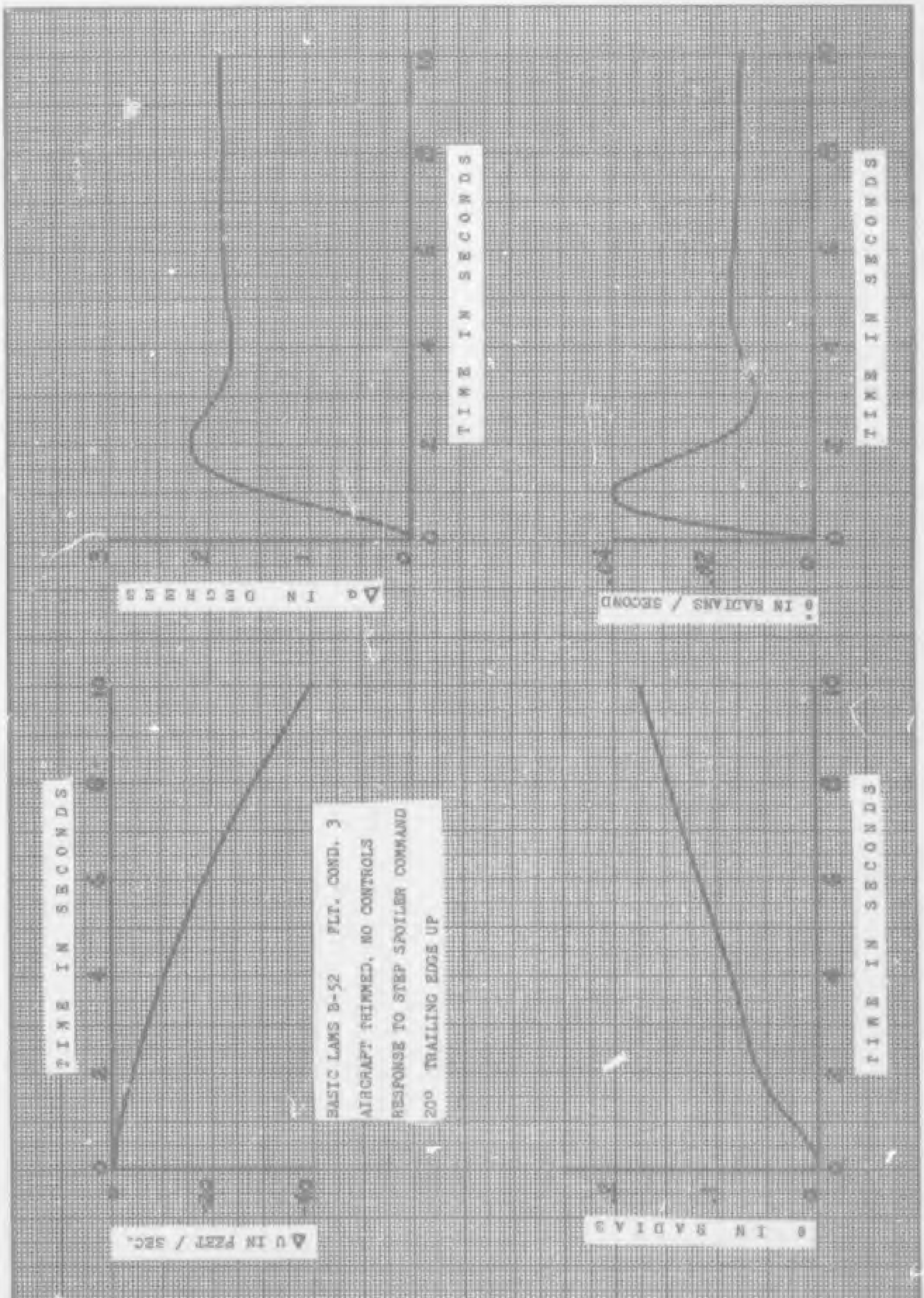

TIME IN SECONDS
Δα IN DEGREES
TIME IN SECONDS
θ̇ IN RADIANS / SECOND
TIME IN SECONDS
Δu IN FEET / SEC.
BASIC LAMS B-52 FLT. COND. 2
AIRCRAFT TRIMMED, NO CONTROLS
RESPONSE TO STEP AILERON COMMAND
5° TRAILING EDGE UP
TIME IN SECONDS
θ IN RADIANS

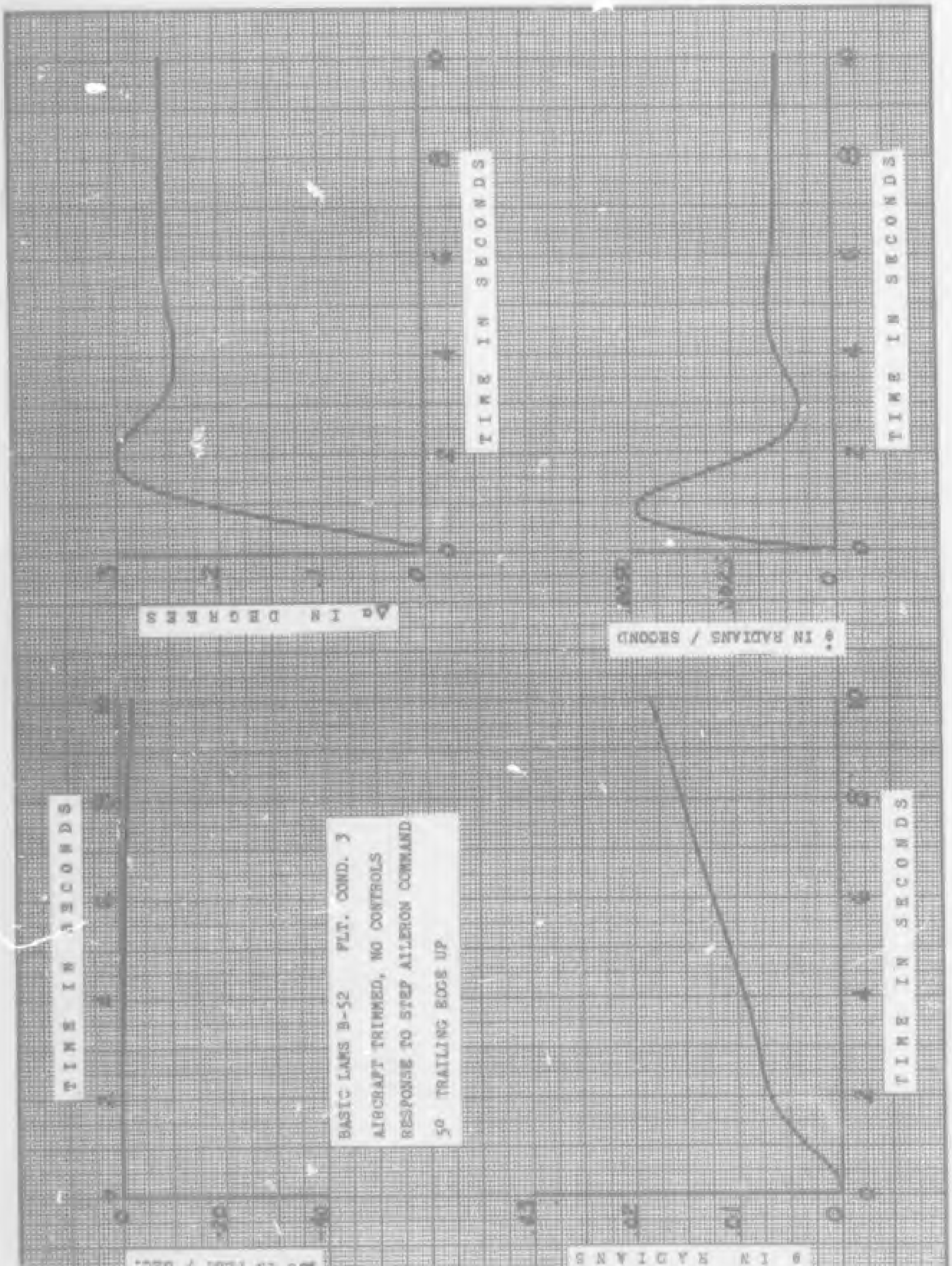

BASIC LAMS B-52 FLT. COND. 2
A/C TRIMMED WITH SPOILERS 20° UP
RESPONSE TO STEP ELEVATOR COMMAND
1° TRAILING EDGE UP
Δα IN DEGREES
θ̇ IN RADIANS / SECOND
Δu IN FEET / SEC.
θ IN RADIANS
TIME IN SECONDS
TIME IN SECONDS
TIME IN SECONDS
TIME IN SECONDS

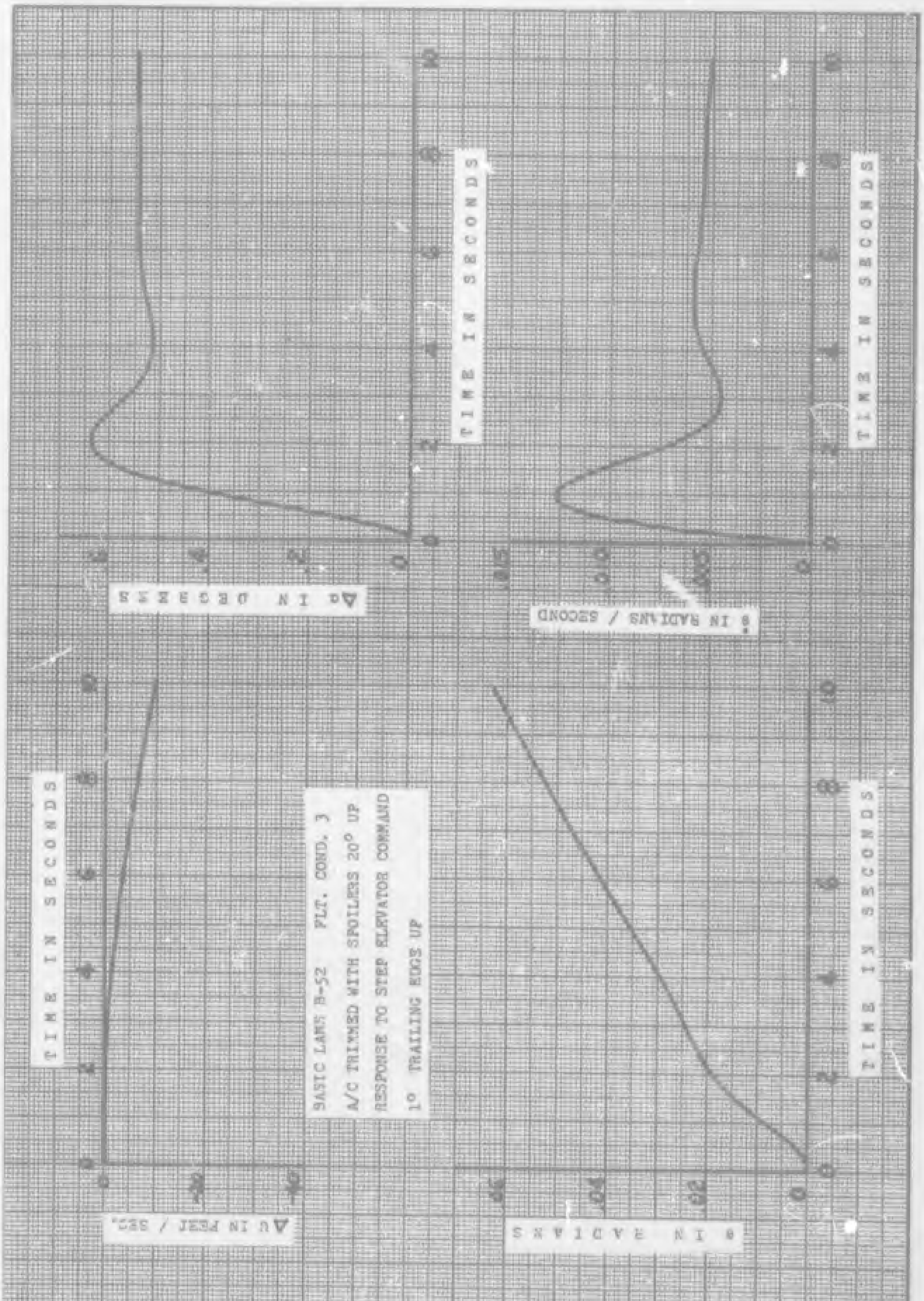

BASIC LAWS B-52 FLT. COND. 2
A/C TRIMMED WITH SPOILERS 20° UP
RESPONSE TO STEP ELEVATOR COMMAND
5° TRAILING EDGE UP
Δα IN DEGREES
θ̇ IN RADIANS / SECOND
Δu IN FEET / SEC.
θ IN RADIANS
TIME IN SECONDS

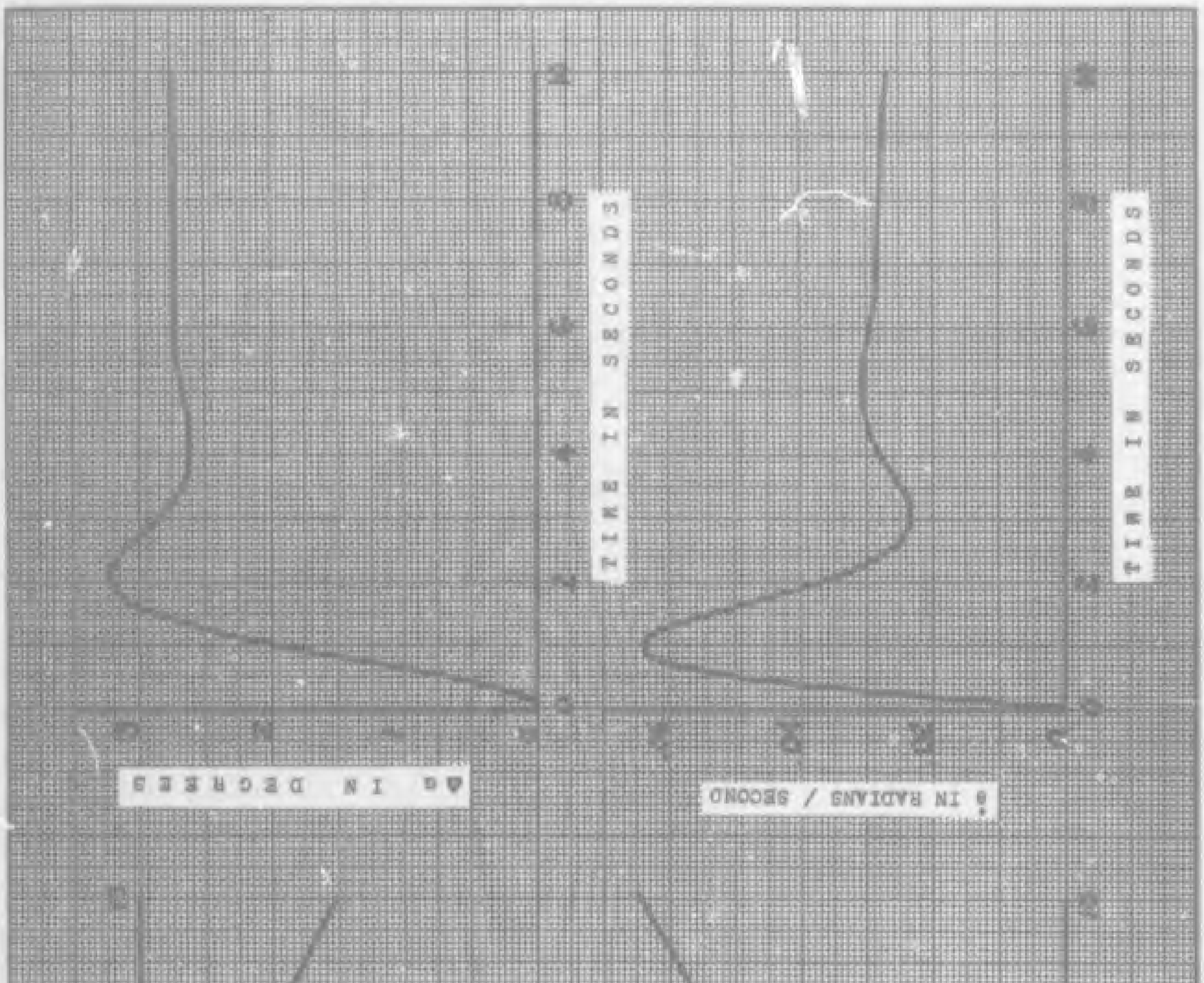

BASIC LAMS B-52 FLT. COND. 3
AIRCRAFT TRIMMED. NO CONTROLS
RESPONSE TO STEP ELEVATOR COMMAND
1° TRAILING EDGE UP
Δα IN DEGREES
θ̇ IN RADIANS / SECOND
Δ U IN FEET / SEC.
θ IN RADIANS
TIME IN SECONDS
TIME IN SECONDS
TIME IN SECONDS
TIME IN SECONDS


BASIC LAMS B-52 FLT. COND. 3
AIRCRAFT TRIMMED, NO CONTROLS
RESPONSE TO STEP ELEVATOR COMMAND
1° TRAILING EDGE UP
COMPARISON RESPONSES PROVIDED BY
THE BOEING COMPANY
Δα IN DEGREES
TIME IN SECONDS
θ̇ IN RADIANS / SECOND
TIME IN SECONDS
θ IN RADIANS
TIME IN SECONDS


BASIC LAMS B-52 FLT. COND. 3
AIRCRAFT TRIMMED, NO CONTROLS
RESPONSE TO STEP SPOILER COMMAND
20° TRAILING EDGE UP
TIME IN SECONDS
Δα IN DEGREES
Δu IN FEET / SEC.
θ̇ IN RADIANS / SECOND
θ IN RADIANS


BASIC LAWS B-52 PLT. COND. 3
AIRCRAFT TRIMMED, NO CONTROLS
RESPONSE TO STEP AILERON COMMAND
5° TRAILING EDGE UP
TIME IN SECONDS
TIME IN SECONDS
TIME IN SECONDS
TIME IN SECONDS
Δa IN DEGREES
φ̇ IN RADIANS / SECOND
ΔU IN FEET / SEC.
φ IN RADIANS


BASIC LAMS B-52 FLT. COND. 3
A/C TRIMMED WITH SPOILERS 20° UP
RESPONSE TO STEP ELEVATOR COMMAND
1° TRAILING EDGE UP
TIME IN SECONDS
Δα IN DEGREES
Δu IN FEET / SEC.
θ IN RADIANS
θ̇ IN RADIANS / SECOND


BASIC LAMS B-52 FLT. COND. 3
A/C TRIMMED WITH SPOILERS 20° UP
RESPONSE TO STEP ELEVATOR COMMAND
5° TRAILING EDGE UP
Δα IN DEGREES
θ̇ IN RADIANS / SECOND
Δ U IN FEET / SEC.
θ IN RADIANS
TIME IN SECONDS
TIME IN SECONDS
TIME IN SECONDS
TIME IN SECONDS

Appendix E

Amplifier and Potentiometer Assignment Sheets

Two Degree of Freedom Analog Program

(Short Period Approximation)

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DIRECT LIFT CONTROL POTENTIOMETER ASSIGNMENT SHEET									
POT. No.	PAGE No.	PARAMETER	GAIN	FLIGHT CONDITION 1		FLIGHT CONDITION 2		FLIGHT CONDITION 3	
				SETTING	OUTPUT	SETTING	OUTPUT	SETTING	OUTPUT
00	2	$2C_{L_{\delta sail}}$	1	.1360	-.0544	.2000	—	.2012	—
01	3	g Meter Bias	1	.3333	—	.3333	—	.3333	—
02	2	$.2(C_{L_\alpha}+C_D)$	1	.7310	-.4664	.7780	—	.9590	—
03	2	$5C_{L_{\delta e}}$	0.1	.5260	-.0420	.7890	—	.7290	—
04	2	C_z Trim	1	— As Required —					
05	2	$5\left[\frac{\bar c}{2u_0}\frac{\partial C_L}{\partial(q\bar c/2u_0)}\right]$	0.1	.2690	-.0810	.4380	—	.2670	—
06	2	$5\left[\frac{\bar c}{2u_0}\frac{\partial C_L}{\partial(\dot\alpha\bar c/2u_0)}\right]$	0.1	.0650	-.0020	.1060	—	.0650	—
07	2	.02 g	0.1	.6440	-.6440	.6440	—	.6440	—
08	2	.001 u	1	.6266	-.1880	.4292	—	.7576	—
09	2	.002 qS/m	1	.3044	.2392	.1482	—	.2195	—
10	1A	$\dot\theta$ FB to δ_e	10	— Variable —					
11	2	Constant	10	.2500	—	.2500	—	.2500	—
12	2	.01 Deg/Rad	1	.5730	—	.5730	—	.5730	—
13	6	Time Slope	1	.0500	—	.0500	—	.0500	—
14	3	Meter Bias	1	.5000	-.5000	.5000	—	.5000	—
15	2	$100/u_0$	10	.1596	—	.2330	—	.1320	—
16	2	$100/u_0$	1	.1596	—	.2330	—	.1320	—
17	3	Constant	1	.5000	—	.5000	—	.5000	—
18	3	$u_0/1000$	1	.6266	—	.4292	—	.7570	—
19	3	$\dot Q$ Trim	0.1	— As Required —					
20	3	$-5\left[\frac{\bar c}{2u_0}\frac{\partial C_m}{\partial(\dot\alpha\bar c/2u_0)}\right]$	0.1	.1860	-.0040	.3110	—	.1880	—
21	3	$-20C_{M_{\delta sail}}$	0.1	.4800	-.1920	.7200	—	.8200	—
22	3	Constant	1	.5000	—	.5000	—	.5000	—
23	3	Constant	10	.2000	—	.2000	—	.2000	—
24	3	Meter Bias	1	.5000	—	.5000	—	.5000	—
25	3	$-.5\left[\frac{\bar c}{2u_0}\frac{\partial C_m}{\partial(q\bar c/2u_0)}\right]$	1	.1150	-.0345	.1915	—	.1165	—
26	1A	θ FB to δ_e	10	— Variable —					
27	3	$-0.5C_{M_{\delta e}}$	1	.1725	-.0138	.2585	—	.2420	—
28	3	$-0.2C_{M_\alpha}$	1	.1090	-.0695	.1524	—	.1516	—
29	3	$0.1[qS\bar c/I_{yy}]$	10	.5481	—	.2575	—	.3380	—
100	1B	$1/T_{aileron}$	1	.2500	—	.2500	—	.2500	—
101	1A	Spoiler Bias	1	.3333	—	.3333	—	.3333	—
102	6	$\Delta C_{L_{max}}(C_L/C_{L_{max}})$	1	.0922	.0235	.1230	—	.1518	—

DIRECT LIFT CONTROL
POTENTIOMETER ASSIGNMENT SHEET

POT. No.	PAGE No.	PARAMETER	GAIN	FLIGHT CONDITION 1		FLIGHT CONDITION 2		FLIGHT CONDITION 3	
				SETTING	OUTPUT	SETTING	OUTPUT	SETTING	OUTPUT
103	6	$\Delta C_{m_{max}}(C_L/C_{L_{max}})$	1	.0873	.0223	.1195	—	.1080	—
104	1A	$-\delta_e$ limit	SJ	−.5934 Output → −17°					—
105	1B	Comparator	—	— As Required					—
106	1B	δ_e limit, basic	1	.5934	—	.5934	—	.5934	—
107	1A	$1/T_e$	1	.2000	—	.2000	—	.2000	—
108	1B	θ FB to δ_{ail}	1	— Variable —					
109	1B	$+\delta_{ail}$ limit	SJ	+.1927 ⇒ +17° (T.E. Down)					
110	1A	θ FB to δ_{sp}	10	— Variable —					
111	1A	$\dot{\theta}$ FB to δ_{sp}	10	— Variable —					
112	1B	$\dot{\theta}$ FB to δ_{ail}	Var.	— Variable —					
113	1B	n_z FB to δ_{ail}	10	— Variable —					
114	1B	$-\delta_{ail}$ limit	SJ	−.1927 ⇒ −17° (T.E. Up)					
116	1B	Comparator	1	— As Required —					
117	1A	$1/T_{sp}$	1	.2500	—	.2500	—	.2500	—
118	1A	α FB to δ_{sp}	1	— Variable —					
119	1A	A125 Trim	0.1	— As Required —					
120	1A	$1/T_{washout}$	1	.1000	—	.1000	—	.1000	—
121	1A	α FB to δ_{sp}	Var.	— Variable —					
122	1A	$1/T_{washout}$	1	.1000	—	.1000	—	.1000	—
123	1B	α FB to δ_{ail}	1	— Variable —					
124	1A	$+\delta_e$ limit	SJ	+.5934 Output ⇒ +17°					—
125	1A	Stick to $2\delta_e$	1	.7400	—	.7400	—	.7400	—
127	1A	n_z FB to δ_{sp}	10	— Variable —					
128	1A	n_z FB to δ_e	10	— Variable —					
129	1A	δ_{sp} limit	—	Output	.5000 ⇒ 30°	1.0000 ⇒ 60°		.8333 ⇒ 50°	
200	13	Noise Adjust	1	— As Required —					
201	13	Constant	1	.0100	—	.0100	--	.0100	—
202	13	$\frac{\sigma_{u_g}\,\omega_0(2\sqrt{3}-1)}{10\sqrt{2\omega_0}}$	10	.6900	—	.5710	—	.7580	—
203	13	$\frac{\sigma_{u_g}\,\omega_0(\sqrt{3}-1)}{10\sqrt{2\omega_0}}$	10	.6900	—	.5710	—	.7580	—
205	13	ω_0	1	.6264	—	.4300	—	.7570	—
206	13	ω_0	1	.6264	—	.4300	—	.7570	—
207	13	$0.2\,\omega_0$	10	.1253	—	.0860	—	.1518	—

DIRECT LIFT CONTROL
AMPLIFIER ASSIGNMENT SHEET

FB	AMPL. No.	PAGE No.	OUTPUT VARIABLE	STATIC CHECK VALUES F.C. 1 CALC. OUTPUT	F.C. 1 CALC. DERIV.	F.C. 1 MEAS. OUTPUT	F.C. 1 MEAS. DERIV.	F.C. 2 CALC. OUTPUT	F.C. 2 CALC. DERIV.	F.C. 2 MEAS. OUTPUT	F.C. 2 MEAS. DERIV.	F.C. 3 CALC. OUTPUT	F.C. 3 CALC. DERIV.	F.C. 3 MEAS. OUTPUT	F.C. 3 MEAS. DERIV.
—	00	2	Input resistors to A12	—	—	—	—								
—	01	2	Input resistors to A12	—	—	—	—								
RG	02	2	$.002\ \dot{W}$	.1320	—	.1307	—								
S	03	2	$-2\dot{\alpha}$	-.0211	—	-.0208	—								
S	04	2	$0.5(\Delta\alpha+\alpha_G)$	.0638	—	.0638	—								
S	05	2	$-5(\Delta\alpha+\alpha_G)_{RAD}$	-.6380	—	-.6375	—								
I	06	2	$-.005\ W$	-.4000	-.0330	-.4000	-.0326								
I	07	6	Time Base	—	—	—	—								
S	08	3	Rate of Climb Meter Dr.	—	—	—	—								
S	09	2	$.05(\Delta\alpha+\alpha_G)_{Deg}$	.3656	—	.3651	—								
S	10	3	θ Meter Drive	—	—	—	—								
S	11	2	α Meter Drive	—	—	—	—								
RG	12	2	$-C_z$	.7854	—	.7854	—								
S	13	3	$.002\ n_L$	—	—	—	—								
S	14	3	g Meter Drive	—	—	—	—								
	15														
	16														
	17														
I	18	3	$-2Q = -2\dot{\theta}$	-.3000	1.6190	-.3000	1.605								
I	19	3	θ (Rad)	1.0000	.1500	1.0005	.1499								

NO STATIC CHECKS
FLIGHT CONDITIONS 2 & 3

DIRECT LIFT CONTROL
AMPLIFIER ASSIGNMENT SHEET

FB	AMPL. NO.	PAGE NO.	OUTPUT VARIABLE	STATIC CHECK VALUES: F.C. 1 CALC. OUTPUT	F.C. 1 CALC. DERIV.	F.C. 1 MEAS. OUTPUT	F.C. 1 MEAS. DERIV.	F.C. 2 CALC. OUTPUT	F.C. 2 CALC. DERIV.	F.C. 2 MEAS. OUTPUT	F.C. 2 MEAS. DERIV.	F.C. 3 CALC. OUTPUT	F.C. 3 CALC. DERIV.	F.C. 3 MEAS. OUTPUT	F.C. 3 MEAS. DERIV.
RG	20	3	$-C_M$	.1477	—	.1467	—								
S	21	3	$\dot{Q}$	-.8095	—	.8035	—								
	22														
	23														
	24														
	25														
S	26	3	$\dot{\Theta}$	—	—	—	—								
	27														
S	28	1A	Elevator Controller	—	—	—	—								
	29														
I	30	3	$-\dot{h}/1000$ (ft/sec)	—	—	—	—								
I	31	3	$h/2000$ (ft)	—	—	—	—								
HG	100	6	Function Generator	—	—	—	—								
HG	101	6	+K (Spoilers)	.2550	—	.2550	—								
S	102	1A	.0167 δ_{sp} (Deg)	.3340	—	.3341	—								
S	103	1B	Aileron Time Constant	—	—	—	—								
	104														
S	105	1B	Aileron Meter Drive	—	—	—	—								
I	106	1B	$+\delta_{ail}$	—	—	—	—								
I	107	1A	Elevator Washout	—	—	—	—								

NO STATIC CHECKS
FLIGHT CONDITIONS 2 & 3

DIRECT LIFT CONTROL
AMPLIFIER ASSIGNMENT SHEET

FB	AMPL. NO.	PAGE NO.	OUTPUT VARIABLE	STATIC CHECK VALUES: F.C. 1 CALC.		F.C. 1 MEAS.		F.C. 2 CALC.		F.C. 2 MEAS.		F.C. 3 CALC.		F.C. 3 MEAS.	
				OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.
	108														
S	109	1A	Elevator Time Constant	—	—	—	—								
S	110	1B	Aileron Controller	—	—	—	—								
S	111	1A	Elevator Controller	—	—	—	—								
S	112	1A	Elevator Time Constant	—	—	—	—								
S	113	1A	Elevator Time Constant	—	—	—	—								
HG	114	1B	Comparator	—	—	—	—								
HG	115	1B	Comparator	—	—	—	—								
	116														
S	117	1B	Aileron Time Constant	—	—	—	—								
	118														
I	119	1A	$+2\delta_e$ (Rad)	.0800	—	.0800	—								
S	120	1A	Elevator Controller	—	—	—	—								
S	121	1A	Elevator Time Constant	—	—	—	—								
	122														
S	123	1B	+ Stick	—	—	—	—								
I	124	1A	$-.0167\,\Delta\delta_{sp}$ (Deg)	—	—	—	—								
S	125	1A	Spoiler Controller	—	—	—	—								
	126														
	127														

NO STATIC CHECKS
FLIGHT CONDITIONS 2 & 3

Appendix F

<u>Six Degree of Freedom Analog Program</u>

* Potentiometer Sheets
* Amplifier Sheets
* Multiplier Sheet
* Computer Interconnect Trunking
* MAV Circuits
* Recorder Assignment Sheet
* Limiter List

DIRECT LIFT CONTROL POTENTIOMETER ASSIGNMENT SHEET									
POT. No.	PAGE No.	PARAMETER	GAIN	FLIGHT CONDITION 1		FLIGHT CONDITION 2		FLIGHT CONDITION 3	
				SETTING	OUTPUT	SETTING	OUTPUT	SETTING	OUTPUT
P00	1	$10^3 u_0$	1	.6266	−.6266	.4292	−.4292	.7570	−.7570
P01	1	I.C.	—	.4000	+.4000	.4000	+.4000	.4000	+.4000
P02	1	CONSTANT	1	.5000	−.1266	.5000	−.1350	.5000	−.1350
P03	1	CONSTANT	1	.1000	−.0132	.1000	−.0420	.1000	−.0336
P04	1	CONSTANT	10	.2000	−.0120	.2000	−.0120	.2000	−.0120
P05	10	I.C.	—	.7000	+.7000	.7000	+.7000	.7000	+.7000
P06	5	MAV.	1	.1000	—	.1000	—	.1000	—
P07	2	I.C. (TR)	1	.0000	.0000	.0000	.0000	1.0000	+1.0000
P08	17	M.A.V.		.1000	—	.1000	—	.1000	—
P09	17	CONSTANT	10	.5000		.5000		.5000	
P10	4	CONSTANT	10	.2500	+.0140	.2500	−.0005	.2500	+.0174
P11	4	$2 C_{n\beta}$	1	.2286	+.0073	.2654	+.0124	.2898	+.0077
P12	1	$.01q$	1	.3220	−.2711	.3220	−.2711	.3220	−.2711
P13	4	$2\times10^{-2}\frac{\bar{q}Sb}{I_{zz}}$	10	.3680	−.0003	.1729	+.0055	.2140	−.0002
P14	9	.01 [illegible]	10	.8197	+.0014	.7692	+.0022	.7692	+.0013
P15	13	K_{V_6}	1	NOISE	ADJUSTMENT				
P16	3	I.C.	—	.5000	+.5000	.5000	+.5000	.5000	+.5000
P17	3	$.05\left[\frac{\bar{q}S\bar{c}}{I_{yy}}\right]$	10	.2741	+.0409	.1288	+.0324	.1690	+.0293
P18	3	CONSTANT	10	.4000	−.1533	.4000	−.1226	.4000	−.1102
P19	10	$.2 K_{\dot\theta}$	10	.1700	−.1028	.1700	−.1028	.1940	−.1173
P20	1	CONSTANT	10	.2000	−.0400	.2000	−.0400	.2000	−.0400
P21	1	$-0.2 C_{y\beta}$	10	.1478	−.0047	.1497	−.0070	.1758	−.0046
P22	1	CONSTANT	10	.2000	+.0424	.2000	+.0306	.2000	+.0502
P23	1	CONSTANT	10	.2000	−.0360	.2000	−.0360	.2000	−.0360
P24	3	$.01\frac{\bar{q}Sb}{I_{xx}}$	10	.3098	+.0516	.1460	+.0504	.1730	+.0361
P25	4	CONSTANT	10	.5000	−.0354	.5000	−.0255	.5000	−.0419
P26									
P27	5	CONSTANT	1	.5000	+.0845	.5000	+.0845	.5000	+.0845
P28	10	CONSTANT	10	.2000	−.1400	.2000	−.1400	.2000	−.1400
P29	10	CONSTANT	10	.3750	−.1395	.3750	−.1395	.3750	−.0853
P30	8	[illegible]	1	.0734	−.0018	.0734	−.0027	.0734	−.0015
P31	3	I.C.	—	.6000	+.6000	.6000	+.6000	.6000	+.6000
P32	1	$2\times10^{-4}u_0$	10	.1253	+.0040	.0859	+.0040	.1515	+.0040

DIRECT LIFT CONTROL POTENTIOMETER ASSIGNMENT SHEET									
POT. NO.	PAGE NO.	PARAMETER	GAIN	FLIGHT CONDITION 1		FLIGHT CONDITION 2		FLIGHT CONDITION 3	
				SETTING	OUTPUT	SETTING	OUTPUT	SETTING	OUTPUT
P33	9	Cos 32.4°	1	.8443	−.8443	.8443	−.8443	.8443	−.8443
P34	4	I.C.	—	.3000	+.3000	.3000	+.3000	.3000	+.3000
P35	2	$-200C_{Zu}$	1	.1350	−.0540	.4190	−.1676	.1549	−.0620
P36	13	$.2W_0$	10	.1253	—	.0860	—	.1518	—
P37	13	[illegible]	10	.7740	—	.9340	—	.7029	—
P38	10	CONSTANT	10	.2000	+.0210	.2000	+.0210	.2000	+.0197
P39	13	[illegible]	10	.6900	—	.5710	—	.7580	—
P40	6	CONSTANT	10	.1348	—	.1348	—	.1348	—
P41	1	I.C.	—	.2000	+.2000	.2000	+.2000	.2000	+.2000
P42	9	Cos 32.4°	1	.8443	−.8443	.8443	−.8443	.8443	−.8443
P43	8	[illegible]	1	.0437	−.0341	.0598	−.0466	.0541	−.0422
P44	13	[illegible]	10	.6900	—	.5710	—	.7580	—
P45	5	I.C.	—	.0845	+.0845	.0845	+.0845	0845	+.0845
P46	10	I.C. TRIM	1	.2349	+.2349	.2349	+.2349	.2349	+.2349
P47									
P48	11	STICK TO [illegible]	1	.3700	—	.3700	—	.3700	—
P49	11	$1/T_{SP}$	1	.2500	—	.2500	—	.2500	—
P50	4	$\dot{R}_{TRIM}$	1	.0000	.0000	.0000	.0000	.0000	.0000
P51									
P52	3	$-2.5C_{m\delta_{SAIL}}$	1	.0600	−.0141	.0900	−.0211	.1050	−.0247
P53	3	$-.2C_{m\alpha}$	1	.2880	−.0288	.3164	−.0316	.3440	−.0344
P54	3	$-.5C_{m\dot{\alpha}}$	1	.1725	−.0301	.2585	−.0451	.2420	−.0422
P55									
P56									
P57	3	$\epsilon_{TE}/\bar{c}$	1	.0673	+.0067	.0673	+.0067	.0725	+.0073
P58	3	[illegible]	1	.0186	−.0043	.0311	−.0048	.0188	−.0034
P59	3	$-.2C_{m_q}$	1	.1090	−.0522	.1524	−.1064	.1516	−.0600
P60	2	α_{TRIM} FOR S[illegible]	1	.0324	.0000	.0271	.0000	.0505	.0000
P62	3	[illegible]	1	.0231	+.0069	.0580	+.0174	.0262	+.0079
P63	3	$-5C_{D\delta_{SAIL}}$	1	.1003	+.0105	.1003	+.0105	.1003	+.0105
P64	2	$C_{L_{CLEAN}}$	1	.2110	−.2110	.4600	−.4500	.2930	−.2930
P65	7	STABILIZER TRIM	1	.5500	+.1000	.5500	+.1000	.5500	+.1000
P66	7	MANUAL THROTTLE	1	.1000	−.1000	.1000	−.1000	.1000	−.1000

DIRECT LIFT CONTROL
POTENTIOMETER ASSIGNMENT SHEET

POT NO.	PAGE NO.	PARAMETER	GAIN	FLIGHT CONDITION 1		FLIGHT CONDITION 2		FLIGHT CONDITION 3	
				SETTING	OUTPUT	SETTING	OUTPUT	SETTING	OUTPUT
Q00	1	CONSTANT	1	.2000	−.0800	.2000	−.0800	.2000	−.0800
Q01	1	$2\times10^3\left(\frac{2C_{D_\alpha}}{U_0}\right)$	1	.0920	−.0368	.2000	−.0800	.0850	−.0340
Q02	1	$.005\frac{\rho S}{M}$	1	.7610	−.6344	.3580	−.3463	.5492	−.4307
Q03	1	$C_{D_{CLEAN}}$	10	.0143	−.0143	.0215	−.0215	.0162	−.0162
Q04	6	CONSTANT	1	.3333	−.3333	.3333	−.3333	.3333	−.3333
Q05	4	CONSTANT	1	.3333	−.0010	.3333	−.0567	.3333	+.0357
Q06	13	M.A.V.	1	.0500	—	.0500	—	.0500	—
Q07	2	I.C.	—	.3000	+.3000	.3000	+.3000	.3000	+.3000
Q08	10	$\frac{17}{85}\times\frac{200}{50}$	1	.8000	−.8000	.8000	−.8000	.8000	−.8000
Q09	6	CONSTANT	—	.1000	−.1000	.1000	−.1000	.1000	−.1000
Q10	1	$2(C_{L_{CLEAN}}-C_{D_{\alpha}})$	1	.2920	+.1398	.5690	+.3974	.2910	+.1152
Q11	4	$\pm10^2\left[\frac{185}{2U_0}\frac{\partial C_N}{\partial(r b/2U)}\right]$	.1	.0260	+.0156	.5030	−.3018	.0260	+.0156
Q12	4	$-20\left[\frac{185}{2U_0}\frac{\partial C_N}{\partial(p b/2U)}\right]$	.1	.2810	−.0843	.4750	−.1425	.2910	−.0873
Q13	1	$.005g$	10	.1610	+.0136	.1610	+.0136	.1610	+.0136
Q14	4	$\frac{(I_{XX}-I_{YY})}{I_{ZZ}}$	1	.1773	−.0532	.1773	−.0532	.2257	−.0677
Q15	13	CONSTANT	1	.0100	—	.0100	—	.0100	—
Q16	3	$-.5\left[\frac{c}{2U_0}\frac{\partial C_M}{\partial(\dot{\theta}c/2U)}\right]$	1	.1150	−.0575	.1915	−.0958	.1165	−.0583
Q17	13	100/5 MINUTES	1	.3333	—	.3333	—	.3333	—
Q18	3	$.1\frac{(I_{ZZ}-I_{XX})}{I_{YY}}$	1	.0974	−.0175	.0974	−.0175	.0974	−.0175
Q19	2	$100/U_0$	1	.1596	+.2261	.2330	+.1558	.1320	+.1782
Q20	1	CONSTANT	1	.5000	+.0995	.5000	+.1635	.5000	+.0619
Q21	1	CONSTANT	1	.1000	−.0003	.1000	−.0005	.1000	−.0003
Q22	1	$.001\frac{\rho S}{M}$	1	.1522	+.0082	.0716	−.0018	.1098	+.0053
Q23	1	$\pm10\left[\frac{185}{U_0}\frac{\partial C_Y}{\partial(r b/2U)}\right]$	1	.0105	−.0063	.1595	+.0957	.0025	−.0015
Q24	3	$-2C_{Y_\beta}$	1	.2164	−.0069	.2933	−.0137	.2870	−.0076
Q25	13	K_{W_G}	1	NOISE	ADJUSTMENT				
Q26	13	CONSTANT	1	.0100	—	.0100	—	.0100	—
Q27	5	CONSTANT	1	.4000	+.1010	.4000	+.1010	.4000	+.1010
Q28	13	W_0	1	.6264	—	.4300	—	.7590	—
Q29	13	[illegible]	10	.6900	—	.5710	—	.7580	—
Q30	8	[illegible]	1	.1210	.0000	.1653	.0000	.1770	.0000
Q31	3	$-10\left[\frac{185}{2U_0}\frac{\partial C_\ell}{\partial(r b/2U)}\right]$	1	.2980	−.1788	.6040	−.3624	.3680	−.2208
Q32	3	$.1\frac{(I_{ZZ}-I_{YY})}{I_{XX}}$	1	.0984	+.0147	.0984	+.0147	.0984	+.0147

Pot. No.	Page No.	Parameter	Gain	Flight Condition 1 Setting	Flight Condition 1 Output	Flight Condition 2 Setting	Flight Condition 2 Output	Flight Condition 3 Setting	Flight Condition 3 Output
				DIRECT LIFT CONTROL POTENTIOMETER ASSIGNMENT SHEET					
Q33	9	(SIN 32.4)/5	1	.1072	+.0034	.1072	+.0050	.1072	+.0028
Q34	6	Spoiler Bias	1	.3333	+.3333	.3333	+.3333	.3333	+.3333
Q35	13	W_0	1	.6264	—	.4300	—	.7590	—
Q36	13	W_0	1	.6264	—	.4300	—	.7590	—
Q37	2	$2.5 C_{L_{SAIL}}$	1	.1700	−.0399	.2500	−.0587	.2650	−.0622
Q38	10	Controller	1	.2000	−.2000	.2000	−.2000	.2000	−.2000
Q39	7	Constant	10	.0175	+.0175	.0175	+.0175	.0175	+.0175
Q40	3	Constant	1	.5000	+.5000	.5000	+.5000	.5000	+.5000
Q41	13	$.2 W_0$	10	.1253	—	.0860	—	.1518	—
Q42	9	(SIN 32.4)/5	1	.1072	−.0034	.1072	−.0060	.1072	−.0028
Q43	8	$\left[\frac{\Delta C_{L_{MAX}}}{2} \frac{C_L}{C_{L_{MAX}}}\right]$	1	.0460	−.0359	.0615	−.0479	.0759	−.0592
Q44	13	W_0	1	.6264	—	.4300	—	.7590	—
Q45	1	I.C. (+TR)	1	.3598	+.3598	.3598	+.3598	.3597	+.3597
Q46	10	+TR (TRIM)	1	.9700	+.9700	.9700	+.9700	.9826	+.9826
Q47	11	$5\Delta\alpha$ To δ_e	1	CONTROLLER VARIABLE					
Q48	11	Stick To δ_{SP}	1	1.0000	—	1.0000	—	1.0000	—
Q49	11	$5\Delta\alpha$ To δ_{SP}	10	CONTROLLER VARIABLE					
Q50	6	Time Base Slope	1	.0033 / .1000	—	.0033 / .1000	—	.0033 / .1000	—
Q51									
Q52	11	$2\dot{\theta}$ To δ_e	10	CONTROLLER VARIABLE					
Q53	11	$2\dot{\theta}$ To δ_{SP}	10	CONTROLLER VARIABLE					
Q54	11	θ To δ_{SP}	10	CONTROLLER VARIABLE					
Q55	11	θ To δ_e	10	CONTROLLER VARIABLE					
Q56	11	$1/\tau$ washout	1	.1000	—	.1000	—	.1000	—
Q57									
Q58	11	$1/\tau$ washout	1	.1000	—	.1000	—	.1000	—
Q59	4	M.A.V.	1	1.0000	—	1.0000	—	1.0000	—
Q60	1	Constant	10	.5000	+.0750	.5000	+.0750	.5000	+.0750
Q61	5	I.C.	—	.1000	−.1000	.1000	−.1000	.1000	−.1000
Q62	5	Constant	10	.2500	−.1052	.2500	−.1052	.2500	−.1052
Q63	11	$.01 n_L$ To δ_{SP}	10	CONTROLLER VARIABLE					
Q64	2	Constant	10	.2000	−.0600	.2000	−.0600	.2000	−.0600
Q65	5	Constant	SIN COS	.2865	+.2865	.2865	+.2865	.2865	+.2865

DIRECT LIFT CONTROL POTENTIOMETER ASSIGNMENT SHEET									
POT. NO.	PAGE NO.	PARAMETER	GAIN	FLIGHT CONDITION 1		FLIGHT CONDITION 2		FLIGHT CONDITION 3	
				SETTING	OUTPUT	SETTING	OUTPUT	SETTING	OUTPUT
Q66	5	CONSTANT	1	.2000	-.0534	.2000	-.0534	.2000	-.0534
Q67	5	CONSTANT	1	.2000	+.0045	.2000	+.0045	.2000	+.0045
Q68	5	I.C. $(-T_R)$	1	.9472	-.9472	.9472	-.9472	.9472	-.9472
Q69	4	CONSTANT	10	.2000	+.0600	.2000	+.0600	.2000	+.0600
Q70	2	CONSTANT	1	.2500	+.3532	.2500	+.1672	.2500	+.3375
Q71	2	$.2(C_{L_\alpha}+C_0)$	1	.7310	-.3500	.7780	-.5434	.9590	-.3798
Q72	2	$.2C_L$	1	.0878	-.0088	.0964	-.0096	.1038	-.0104
Q73	2	$.002\frac{\rho S}{M}$	10	.3044	+.1984	.1432	+.1745	.2196	+.1703
Q74	2	CONSTANT	10	.4000	+.0480	.4000	+.0480	.4000	+.0480
Q75	2	$+5C_{L_{\delta e}}$	0.1	.5260	-.0918	.7890	-.1377	.7290	-.1272
Q76	2	$5\left[\frac{c}{2U_0}\frac{\partial C_L}{\partial(\dot{\alpha}c/2U)}\right]$	0.1	.0650	-.0147	.1060	-.0166	.0650	-.0120
Q77	2	$5\left[\frac{c}{2U_0}\frac{\partial C_L}{\partial(qc/2U)}\right]$	0.1	.2690	-.1345	.4380	-.2190	.2670	-.1335
Q78	2	$.02g$	1	.6440	-.3477	.6440	-.3477	.6440	-.3477
Q79	2	$2\times10^{-4}U_0$	10	.1253	+.0600	.0859	+.0600	.1515	+.0600
Q80	12	$.01 n_L$ TO S_{AIL}	10	CON	TRO	LLER	VA	RIA	BLE
Q81	12	$1/T_{AIL}$	1	.2500	—	.2500	—	.2500	—
Q82	2	CONSTANT	10	.1146	-.0599	.1146	-.0800	.1146	-.0464
Q83	11	$.01 n_L$ TO S_e	10	CON	TRO	LLER	VA	RIA	BLE
Q84	12	θ TO S_{AIL}	1	CON	TRO	LLER	VA	RIA	BLE
Q85	12	$5\Delta\alpha$ TO S_{AIL}	1	CON	TRO	LLER	VA	RIA	BLE
Q86	12	$2\dot{\theta}$ TO S_{AIL}	1	CON	TRO	LLER	VA	RIA	BLE
Q87									
Q88	11	$1/T_e$	1	.2000	—	.2000	—	.2000	—
Q89									
Q90	1	M.A.V.	1	.0100	—	.0100	—	.0100	—
Q91	2	M.A.V.	1	.1000	—	.1000	—	.1000	—
Q93	7	$.005\left[\frac{2K_0\cos 32.4°}{\Delta C_D}\right]$	10	.2601	-.0659	.2439	-.0659	.2439	-.0659
Q94	7	$4\left(\frac{100d}{SSK}\right)$	1	.0572	—	.1236	—	.0480	—
Q95									
Q96									
Q97	7	$.01\left[\frac{2K_0\cos 32.5°}{\Delta C_D \tan 32.5°}\right]$	1	.8191	-.0026	.7692	-.0039	.7692	-.0022
Q98	9	CONSTANT	1	.5600	+.2184	.5600	+.2184	.5600	+.2184
Q99	9	CONSTANT	1	.5600	+.2184	.5600	+.2184	.5600	+.2184

DIRECT LIFT CONTROL
AMPLIFIER ASSIGNMENT SHEET

FB	AMPL. No.	PAGE No.	OUTPUT VARIABLE	STATIC CHECK VALUES											
				F.C. 1 CALC.		F.C. 1 MEAS.		F.C. 2 CALC.		F.C. 2 MEAS.		F.C. 3 CALC.		F.C. 3 MEAS.	
				OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.
I	A00	1	$-0.5\,\Delta u$	-.4000	+.0132	-.4000	+.0130	-.4000	+.0420	-.4000	+.0424	-.4000	+.0335	-.4000	+.0337
S	A01	1	$+0.1\,u$	+.7066	—	+.7068	—	+.5092	—	+.5083	—	+.8370	—	+.8369	—
S	A02	1	$-10^3 C_x$	-.8334	—	-.8334	—	-.9674	—	-.9658	—	-.7842	—	-.7830	—
S	A03	1	$+5\dot{u}$	-.1316	—	-.1320	—	-.4197	—	-.4204	—	-.3353	—	-.3344	—
S	A04	6	$+1.667\,\delta_{SP}$ (DEG)	+.4333	—	+.4305	—	+.4333	—	+.4303	--	+.4333	—	+.4333	—
I	A05	4	$+h/3000$	1.0000	+.0001	1.0000	+.0002	1.0000	-.0059	1.0000	-.0055	1.0000	+.0036	1.0000	+.0040
I	A06	13	TURBULENCE FILTER	1V RMS	—	—	—	1V RMS	—	—	—	1V. RMS	—	—	—
S	A07	10	$+200[\delta_a]_{RIGHT}$	-.5900	—	-.5903	—	-.5900	—	-.5899	—	-.5900	—	-.5894	—
RG	A08	4	$\dot{h}$	-.0030	—	-.0037	—	-.1700	—	-.1715	—	+.1070	—	+.1076	—
S	A09	17	HORIZONTAL NEEDLE ERROR	—	—	—	—	—	—	—	—	—	—	—	—
I	A10	4	$-500\,R$	-.3000	-.1400	-.3000	-.1380	-.3000	+.0045	-.3000	+.0046	-.3000	-.1735	-.3000	-.1728
S	A11	4	$+0.5\dot{Y}'$	+.5535	—	+.5549	—	+.4550	—	+.4545	—	+.6190	—	+.6197	—
RG	A12	4	$-10^3 C_N$	-.0007	—	-.0006	—	+.0318	—	+.0318	—	-.0008	—	-.0007	—
S	A13	4	$+200\dot{R}$	+.0558	—	+.0555	—	-.0018	—	-.0017	—	+.0694	—	+.0693	—
HG	A14	9	$+10^3 C_{n\delta_{SP}}\,\delta_{SP}$	+.0002	—	+.0002	—	+.0003	—	+.0003	—	+.0002	—	+.0002	—
I	A15	13	V_G FILTER	—	—	—	—	—	—	—	—	—	—	—	—
I	A16	3	$-200\,Q$	-.5000	+1.5328	-.5000	+1.5354	-.5000	1.2256	-.5000	1.2277	-.5000	+1.1020	-.4999	+1.1027
S	A17	3	$-10^2 C_M$	+.1462	—	+.1463	—	+.2515	—	+.2520	—	+.1732	—	+.1732	—
S	A18	3	$+50\dot{Q}$	-.3832	—	-.3835	—	-.3064	—	-.3070	—	-.2755	—	-.2750	—
S	A19	10	ROLL SAS	-.3720	—	-.3739	—	-.3720	—	-.3730	—	-.2270	—	-.2278	—

DIRECT LIFT CONTROL
AMPLIFIER ASSIGNMENT SHEET

FB	AMPL. NO.	PAGE NO.	OUTPUT VARIABLE	STATIC CHECK VALUES											
				F.C. 1 CALC.		F.C. 1 MEAS.		F.C. 2 CALC.		F.C. 2 MEAS.		F.C. 3 CALC.		F.C. 3 MEAS.	
				OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.
I	A20	1	$-.5V$	-.2000	-.0895	-.2000	-.0990	-.2000	-.1635	-.2000	-.1635	-.2000	-.0619	-.2000	-.0610
HG	A21	1	$+500(\beta+\beta_G)$	+.0319	—	+.0319	—	+.0466	—	+.0466	—	+.0264	—	+.0264	—
S	A22	1	$-10^3 C_Y$	+.0537	—	+.0538	—	-.0254	—	-.0257	—	+.0482	—	+.0483	—
S	A23	1	$+\dot{V}$	+.1989	—	+.1982	—	+.3269	—	+.3275	—	+.1238	—	+.1225	—
S	A24	3	$-10^3 C_\ell$	+.1665	—	+.1667	—	+.3455	—	+.3458	—	+.2085	—	+.2088	—
I	A25	10	ROLL SAS	-.7000	3.720	-.7000	3.739	-.7000	3.720	-.7004	3.730	-.7000	+2.270	-.7000	+2.274
I	A26	5	$-10^2\theta$	-.0845	-.0845	-.0845	-.0842	-.0846	-.0846	-.0845	-.0842	-.0846	-.0845	-.0845	-.0844
S	A27	5	$+200\dot{\theta}$	+.1690	—	+.1690	—	+.1690	—	+.1690	—	+.1690	—	+.1690	—
S	A28	10	ROLL SAS	1.4000	—	1.4009	—	1.4000	—	1.3999	—	+1.400	—	+1.3996	—
S	A29	10	$-200[\delta_{AIL}]_{\dot{\phi}}$	-.0050	—	-.0010	—	-.0050	—	-.0010	—	-.5467	—	-.5470	—
I	A30	13	W_G FILTER	—	—	—	—	—	—	—	—	—	—	—	—
I	A31	3	$-10^2 P$	-.6000	+.5305	-.6000	+.5307	-.6000	+.5191	-.6000	+.5199	-.6000	+.3754	-.6000	+.3764
S	A32	3	$+10^2\dot{P}$	-.5305	—	-.5304	—	-.5191	—	-.5196	—	-.3754	—	-.3762	—
S	A33	9	$+10^2\cos[32.4°+(\beta+\beta_G)]$	+.8409	—	+.8410	—	+.8393	—	+.8395	—	+.8415	—	+.8414	—
S	A34	6	$-1.667\,\delta_{SP}$ (DEG)	-.4333	—	-.4288	—	-.4333	—	-.4274	—	-.4333	—	-.4333	—
I	A35	13	$-V_G$	—	—	—	—	—	—	—	—	—	—	—	—
I	A36	13	V_G FILTER	—	—	—	—	—	—	—	—	—	—	—	—
RG	A37	10	$-40\,\delta_{AIL}$ (TOTAL)	-.2349	—	-.2352	—	-.2349	—	-.2351	—	-.2349	—	-.2345	—
HG	A38	10	$-200\,\delta_{AIL}$	+.1050	—	+.1049	—	+.1050	—	+.1042	—	+.0987	—	+.0988	—
S	A39	10	$+200[\delta_{AIL}]_{LEFT}$	-.5900	—	-.5900	—	-.5900	—	-.5884	—	-.5900	—	-.5884	—

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FB	AMPL. No.	PAGE NO.	OUTPUT VARIABLE	STATIC CHECK VALUES: F.C. 1 CALC. OUTPUT	F.C. 1 CALC. DERIV.	F.C. 1 MEAS. OUTPUT	F.C. 1 MEAS. DERIV.	F.C. 2 CALC. OUTPUT	F.C. 2 CALC. DERIV.	F.C. 2 MEAS. OUTPUT	F.C. 2 MEAS. DERIV.	F.C. 3 CALC. OUTPUT	F.C. 3 CALC. DERIV.	F.C. 3 MEAS. OUTPUT	F.C. 3 MEAS. DERIV.
I	A40	13	$-W_G$	—	—	—	—	—	—	—	—	—	—	—	—
I	A41	13	W_G FILTER	—	—	—	—	—	—	—	—	—	—	—	—
S	A42	9	$+10^2 \cos[32.4° - (\beta + \beta_0)]$	+.8477	—	+.8479	—	+.8493	—	+.8495	—	+.8471	—	+.8471	—
S	A43	8	$-10^2 (K_{DR} + K_{DL})$	-.7800	—	-.7768	—	-.7800	—	-.7777	—	-.7800	—	-.7795	—
—	A44	2	INPUT RESISTORS TO A72	—	—	—	—	—	—	—	—	—	—	—	—
S	A45	11	SPOILER CONTROLLER	—	—	—	—	—	—	—	—	—	—	—	—
I	A46	11	$-1.667 \Delta \delta_{SP}$	—	—	—	—	—	—	—	—	—	—	—	—
RG	A47	2	$+n_L$ ft/sec^2	+1.0584	—	+1.0583	—	+.9386	—	+.9387	—	+1.4176	—	+1.4170	—
S	A48	11	ELEVATOR CONTROLLER	—	—	—	—	—	—	—	—	—	—	—	—
S	A49	9	-10^2 [[illegible] / 185 TAN 32.4 + [illegible] / 185]	-.2184	—	-.2180	—	-.2184	—	-.2172	—	-.2184	—	-.2186	—
I	A50	6	TIME BASE	—	—	—	—	—	—	—	—	—	—	—	—
I	A51	11	ELEVATOR CONTROLLER	—	—	—	—	—	—	—	—	—	—	—	—
S	A52	11	ELEVATOR CONTROLLER	—	—	—	—	—	—	—	—	—	—	—	—
—	A53	3	INPUT RESISTORS TO A17	—	—	—	—	—	—	—	—	—	—	—	—
RG	A54	4	$-.0333\, h$	+.0001	—	+.0002	—	+.0057	—	+.0058	—	-.0036	—	-.0036	—
S	A55	6	-CONTROLLER δ_{SP} COMM.	+.1000	—	+.0993	—	+.1000	—	+.1002	—	+.1000	—	+.1000	—
S	A56														
S	A59	5	$+10^2 \sin \phi$	+.8418	—	+.8410	—	+.8418	—	+.8411	—	+.8418	—	+.8409	—
I	A60	5	$+10^2 \phi$ RAD.	+1.0000	+.6045	+1.0000	+.6050	+1.0000	+.6045	+1.0000	+.6048	+1.0000	+.6045	+1.0000	+.6049
I	A61	5	$+10^2 \psi$ RAD	+.1000	+.0534	+.1000	+.0538	+.1000	+.0534	+.1001	+.0536	+.1000	+.0534	+.1000	+.0537

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FB	AMPL. NO.	PAGE NO.	OUTPUT VARIABLE	STATIC CHECK VALUES											
				F.C. 1 CALC.		F.C. 1 MEAS.		F.C. 2 CALC.		F.C. 2 MEAS.		F.C. 3 CALC.		F.C. 3 MEAS.	
				OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.
S	A62	5	$-.500\dot{\psi}$ RAD/SEC.	-.2672	—	-.2667	—	-.2672	—	-.2664	—	-.2672	—	-.2665	—
S	A63	11	ELEVATOR CONTROLLER	—	—	—	—	—	—	—	—	—	—	—	—
S	A64	11	SPOILER CONTROLLER	—	—	—	—	—	—	—	—	—	—	—	—
S	A65	13	$+N_2(t)$	—	—	—	—	—	—	—	—	—	—	—	—
S	A66	10	$-200[\delta_{AIL}]_{RIGHT}$	+.5900	—	+.5903	—	+.5900	—	+.5904	—	+.5900	—	+.5894	—
S	A69	5	$-10^2\dot{\phi}$ RAD/SEC.	-.6045	—	-.6046	—	-.6045	—	-.6045	—	-.6045	—	-.6047	—
I	A70	2	$-0.5\,W$	-.3000	-.3532	-.3000	-.3537	-.3000	-.1672	-.3000	-.1640	-.3000	-.3375	-.3000	-.3374
RG	A72	2	$-10^2 C_{\theta}$	+.6519	—	+.6528	—	+1.2185	—	+1.2180	—	+.7755	—	+.7755	—
S	A73	2	$+2\dot{W}$	+1.4163	—	+1.4170	—	+.6688	—	+.6611	—	+1.3498	—	+1.3500	—
HG	A74	2	$+500(\Delta\alpha+\alpha_G)$	+.4788	—	+.4800	—	+.6984	—	+.6992	—	+.3960	—	+.3966	—
S	A75	8	$+10^3 C_{\delta SP}\,\delta_{SP}$	+.0018	—	+.0018	—	+.0027	—	+.0024	—	+.0016	—	+.0014	—
S	A76	8	$-10^2(K_{DL}-K_{DR})$	.0000	—	-.0001	—	.0000	—	+.0019	—	.0000	—	+.0014	—
S	A79	7	$+100\,T/qS$	+.1000	—	+.1000	—	+.1000	—	+.0998	—	+.1000	—	+.0999	—
S	A80	12	AILERON CONTROLLER	—	—	—	—	—	—	—	—	—	—	—	—
S	A81	12	AILERON CONTROLLER	—	—	—	—	—	—	—	—	—	—	—	—
S	A82	2	$+10(\Delta\alpha+\alpha_G)$ DEG.	+.5490	—	+.5502	—	+.8000	—	+.8012	—	+.4640	—	+.4643	—
S	A83	11	ELEVATOR CONTROLLER	—	—	—	—	—	—	—	—	—	—	—	—
S	A84	7	$-200\delta_e$	-.1745	—	-.1743	—	-.1745	—	-.1745	—	-.1745	—	-.1746	—
S	A85	9	$-10^2\cos[32.4^\circ+(\beta+\beta_c)]$	-.8409	—	-.8410	—	-.8393	—	-.8396	—	-.8415	—	-.8415	—
S	A86	3	$+10^2 P$	+.6000	—	+.6001	—	+.6000	—	+.6000	—	+.6000	—	+.6000	—

				DIRECT LIFT CONTROL AMPLIFIER ASSIGNMENT SHEET — STATIC CHECK VALUES											
FB	AMPL. NO.	PAGE NO.	OUTPUT VARIABLE	F.C. 1 CALC.		F.C. 1 MEAS.		F.C. 2 CALC.		F.C. 2 MEAS.		F.C. 3 CALC.		F.C. 3 MEAS.	
				OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.
I	A90	11	$+100\ \delta_e$ (RAD)	—	—	—	—	—	—	—	—	—	—	—	—
I	A91	12	$+100\ \Delta\delta_{AIL}$	—	—	—	—	—	—	—	—	—	—	—	—
HG	A92	7	$+10^4 C_{Y\delta SP}\ \delta_{SP}$	-.0032	—	-.0031	—	-.0051	—	-.0046	—	-.0029	—	-.0034	—
HG	A93	7	$-2\times10^3 C_{D\delta SP}\ \delta_{SP}$	-.2532	—	-.2530	—	-.2700	—	-.2695	—	-.2700	—	-.2701	—
S	A94	9	-10^2 [illegible]	-.2184	—	-.2181	—	-.2184	—	-.2178	—	-.2184	—	-.2185	—
S	F95/A96	6	$+1.667\ \delta_{SP}$ (DEG)	+.4333	—	+.4221	—	+.4333	—	+.4212	—	+.4333	—	+.4300	—
S	A96	9	$-10^2 \cos[32.4° - (\beta + \beta_0)]$	-.8477	—	-.8479	—	-.8493	—	-.8495	—	-.8471	—	-.8469	—
S	A99	7	-500α	-.1000	—	-.1000	—	-.1000	—	-.1000	—	-.1000	—	-.1002	—
S	R09	3	$+200\,Q$	+.5000	—	+.5000	—	+.5000	—	+.5000	—	+.5000	—	+.4999	—
S	R17	5	$-10^2 \sin\phi$	-.8418	—	-.8410	—	-.8418	—	-.8410	—	-.8418	—	-.8409	—
S	R19	5	$-10^2 \cos\phi$	-.5400	—	-.5402	—	-.5400	—	-.5402	—	-.5400	—	-.5402	—
SIN	R25	5	$-10^2 \sin\phi$	-.8418	—	-.8411	—	-.8418	—	-.8411	—	-.8418	—	-.8411	—
COS	R26	5	$-10^2 \cos\phi$	-.5400	—	-.5402	—	-.5400	—	-.5402	—	-.5400	—	-.5402	—
S	R28	12	AILERON CONTROLLER	—	—	—	—	—	—	—	—	—	—	—	—
S	R29	6	$+50\ \delta_{WHEEL}$	.0000	—	.0000	—	.0000	—	.0000	—	.0000	—	.0000	—
S	R38	11	ELEVATOR CONTROLLER	—	—	—	—	—	—	—	—	—	—	—	—
S	R48	10	$-20\ \Delta\delta_{AIL}$	+.2000	—	+.2000	—	+.2000	--	+.2000	—	+.2000	—	+.2000	—
S	R49	7	$-10^2 K_{DL} \cos[32.4° + (\beta + \beta_0)]$	-.3280	—	-.3275	—	-.3273	—	-.3270	—	-.3282	—	-.3277	—
S	M5/T13	4	$-10^2\psi$	-.1000	—	-.1000	—	-.1000	—	-.1000	—	-.1000	—	-.1000	—
S	M67/T14	2	$-500(\Delta\alpha + \alpha_G)$	-.4788	—	-.4800	—	-.6984	—	-.6992	—	-.3960	—	-.3966	—

DIRECT LIFT CONTROL AMPLIFIER ASSIGNMENT SHEET															
				STATIC CHECK VALUES											
				F.C. 1 CALC.		F.C. 1 MEAS.		F.C. 2 CALC.		F.C. 2 MEAS.		F.C. 3 CALC.		F.C. 3 MEAS.	
FB	AMPL. NO.	PAGE NO.	OUTPUT VARIABLE	OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.	OUTPUT	DERIV.
S	M68/T21	2	$-200\,\dot{\alpha}$	-.2261	—	-.2267	—	-.1558	—	-.1532	—	-.1782	—	-.1786	—
S	M77/T22	8	$+10^2 C_{M\delta SP}\,\delta_{SP}$	+.0341	—	+.0342	—	+.0466	—	+.0467	—	+.0422	—	+.0486	—
S	M98/T25	7	$+10^2 K_{DR}$	+.3900	—	+.3894	—	+.3900	—	+.3880	—	+.3900	—	+.3907	—
S	M58/T26	4	$+10^2\,\theta$	+.0845	—	+.0845	—	+.0845	—	+.0846	—	+.0845	—	+.0845	—
S	M78/T27	8	$+10^2 C_{L\delta SP}\,\delta_{SP}$	+.0359	—	+.0361	—	+.0479	—	+.0482	—	+.0592	—	+.0597	—
S	M6/T28	1	$-500(\beta+\beta_G)$	-.0319	—	-.0319	—	-.0466	—	-.0465	—	-.0264	—	-.0264	—
S	M5/T29	3	$-.1u$	-.7066	—	-.7066	—	-.5092	—	-5080	—	-.8370	—	-.8368	—
S	M87/T30	2	$+500\,R$	+.3000	—	+.3001	—	+.3000	—	+3000	—	+.3000	—	+.3000	—
S	M89/T31	2	$+.5\,V$	+.2000	—	+.2000	—	+.2000	—	+.2000	—	+.2000	—	+.2000	—
S	M99/T68	7	$+10^2 K_{DL}$	+.3900	—	+.3895	—	+.3900	—	+.3896	—	+.3900	—	+.3895	—
S	M8/T71	2	$+.5\,W$	+.3000	—	+.3004	—	+.3000	—	+.3003	—	+.3000	—	+.3004	—
S	M69	1	$+.5\,\Delta u$	+.4000	—	+.4000	—	+.4000	—	+.4000	—	+.4000	—	+.4000	—
S	M75	11	SPOILER CONTROLLER	—	—	—	—	—	—	—	—	—	—	—	—
S	M88	4	$+500\,R$	+.3000	—	+.3000	—	+.3000	—	+.3000	—	+.3000	—	+.3000	—
S	F60	10	$+200\,\delta_{AIL}$ (WHEEL)	+.8000	—	+.8000	—	+.8000	—	+.7999	—	+.8000	—	+.8000	—
S	F61	13	V_G FILTER	—	—	—	—	—	—	—	—	—	—	—	—
S	F62	13	W_G FILTER	—	—	—	—	—	—	—	—	—	—	—	—
S	F63	13	$+W_G$	—	—	—	—	—	—	—	—	—	—	—	—
S	F64	5	$+10^2 \cos\phi$	+.5400	—	+.5400	—	+.5400	—	+.5402	—	+.5400	—	+.5402	—
S	F71	6	$-10^2 K_{DL}$	-.3900	—	-.3900	—	-.3900	—	-.3900	—	-.3900	—	-.3900	—

DIRECT LIFT CONTROL
AMPLIFIER ASSIGNMENT SHEET

FB	AMPL. NO.	PAGE NO.	OUTPUT VARIABLE	STATIC CHECK VALUES: F.C.1 CALC. OUTPUT	F.C.1 CALC. DERIV.	F.C.1 MEAS. OUTPUT	F.C.1 MEAS. DERIV.	F.C.2 CALC. OUTPUT	F.C.2 CALC. DERIV.	F.C.2 MEAS. OUTPUT	F.C.2 MEAS. DERIV.	F.C.3 CALC. OUTPUT	F.C.3 CALC. DERIV.	F.C.3 MEAS. OUTPUT	F.C.3 MEAS. DERIV.
S	F72	13	$N_1(t)$	—	—	—	—	—	—	—	—	—	—	—	—
S	F73	13	$+V_G$	—	—	—	—	—	—	—	—	—	—	—	—
S	F74	6	$-10^2 K_{DR}$	-.3900	—	-.3897	—	-.3900	—	-.3884	—	-.3900	—	-.3900	—

DIRECT LIFT CONTROL
Amplifier Assignment Sheet

CONSOLE "S"
Cockpit Simulator

FB	AMPL. No.	PAGE NO.	OUTPUT VARIABLE	FB	AMPL. No.	PAGE NO.	OUTPUT VARIABLE	FB	AMPL. No.	PAGE NO.	OUTPUT VARIABLE
S	S01	16	Airspeed (Knots)	S	S21	14	Stick Drive Ckt.	S	S41	14	Base Drive (Elev)
S	S02	16	Beta Meter Drive	S	S22	14	Stick Drive Ckt.	S	S42	15	Base Drive (Aileron)
S	S03	16	θ Meter Drive	S	S23	14	Stick Position Drive	S	S43	14	Heave Drive
S	S04	16	Rate of Climb (fpm)	S	S24	15	Wheel Drive Ckt.	S	S44	16	RPM Ind. Drive
S	S05	16	-Altitude (ft)	S	S25	15	Wheel Drive Ckt.	S	S45	16	Roll (Attitude Ind.)
S	S05	16	g Meter Ckt.	S	S26	15	Wheel Position Drive	S	S46	16	Yaw (Attitude Ind.)
HG	S07	17	HG, θ MAV	S	S27	15	Rudder Drive Ckt.	S	S47	16	Pitch (Attitude Ind.)
HG	S08	17	HG, n_L MAV	S	S28	15	Rudder Drive Ckt.	S	S48	16	+Roll
S	S09	17	Terrain Model	S	S29	15	Rudder Drive Ckt.				
I	S10	14	+500 δ_{STAB}	S	S30	16	Altimeter Drive				
I	S11	15	Roll Trim	S	S31	16	Alpha Meter Drive				
S	S12	16	Throttle Ckt.	S	S32	14	+ Stick				
I	S13	17	θ MAV	S	S33	15	-50 δ_{wheel}				
I	S14	17	n_L MAV	S	S34	16	g Meter Drive				
I	S15	17	Horiz. Needle MAV	HG	S35	17	HG, Horiz. Needle MAV				
I	S16	17	Δu MAV	HG	S36	17	HG, Δu MAV				
	S17				S37						
I	S18	17	$\dot{h}$ MAV	HG	S38	17	HG, $\dot{h}$ MAV				
I	S19	17	α_{Gust} MAV	HG	S39	17	HG, α_{Gust} MAV				
I	S20	17	+ Stick MAV	HG	S40	17	HG, +Stick MAV				

DIRECT LIFT CONTROL
MULTIPLIER ASSIGNMENT SHEET

MULTIPLIER	PAGE	+X INPUT	FROM	+Y INPUT	FROM	OUTPUT VARIABLE	STATIC CHECK VALUES FLT. COND. 1	FLT. COND. 2	FLT. COND. 3
R 01	3	$+.5W$	MUL 7	$-200Q$	A 16	$+1.0\,WQ$	+.1500	+.1500	+.1500
R 02	4	$-500R$	A 10	$+200Q$	MUL 4	$+10^3 RQ$	+.1500	+.1500	+.1500
R 03	4	$-200Q$	TP 70	$-10^2 P$	TP 45	$-200PQ$	-.3000	-.3000	-.3000
R 04	5	$-200Q$	TP 70	$+10^2 \sin\phi$	R 17	$+200Q \sin\phi$	+.4209	+.4209	+.4209
R 11	5	$+200Q$	MUL 4	$-10^2 \cos\phi$	MUL 10	$-200Q \cos\phi$	-.2700	-.2700	-.2700
R 12	5	$-500R$	A 10	$+10^2 \cos\phi$	R 19	$+500R \cos\phi$	+.1620	+.1620	+.1620
R 13	5	$-500R$	A 10	$-10^2 \sin\phi$	MUL 11	$+500R \sin\phi$	+.2525	+.2525	+.2525
R 14	5	$+10^2\theta$	MUL 12	$-500\dot{\psi}$	A 62	$+500\theta\dot{\psi}$	+.0226	+.0226	+.0226
R 42	13	$+V_G$	F 73	$+V_G$	F 73	$N^2/100$ NOISE GENERATOR ADJUSTMENT	—	—	—
M 51	3	$-.1U$	MUL 2	$+500R$	MUL 9	$+.5RU$	+.2120	+.1528	+.2511
M 52	3	$+200Q$	MUL 4	$+.1U$	A 01	$-.2QU$	-.3533	-.2546	-.4185
M 53	4	$+.1U$	A 01	$+10^2\psi$	A 61	$-.1U\psi$	-.0707	-.0509	-.0837
M 54	4	$-.1U$	MUL 2	$-10^2\theta$	A 26	$-.1U\theta$	-.0597	-.0430	-.0707
M 71	8	$-500(\beta+\beta_G)$	TP 25	$-10^2(K_{DR}+K_{DL})$	A 43	$-500(\beta+\beta_G)(K_{DR}+K_{DL})$	-.0249	-.0363	-.0206
M 81	2	$+10^2 P$	TP 35	$-.5V$	A 20	$+.5PV$	+.1200	+.1200	+.1200
M 82	2	$-10^2 P$	TP 45	$-.5W$	A 70	$-.5PW$	-.1800	-.1800	-.1800
M 83	2	$-.5V$	A 20	$-500R$	A 10	$-2.5RV$	-.0600	-.0600	-.0600
M 84	4	$-10^2 P$	TP 45	$-500R$	A 10	$-500PR$	-.1800	-.1800	-.1800
M 91	7	$+10^2\cos[32.4°+(\beta+\beta_G)]$	A 33	$-10^2 K_{DL}$	SMOE	$+10^2 K_{DL}\cos[32.4°+(\beta+\beta_G)]$	+.3280	+.3273	+.3282
M 92	9	$-10^2\cos[32.4°-(\beta+\beta_G)]$	A 98	$-10^2[\quad]$	R 48	$-10^2\cos[32.4°-(\beta+\beta_G)][\quad]$	-.1851	-.1855	-.1850
M 93	9	$+10^2\cos[32.4°+(\beta+\beta_G)]$	A 33	$-10^2[\quad]$	R 47	$+10^2\cos[32.4°+(\beta+\beta_G)][\quad]$	+.1837	+.1833	+.1838
M 94	7	$+10^2\cos[32.4°-(\beta+\beta_G)]$	A 42	$-10^2 K_{DR}$	MUL 16	$+10^2 K_{DR}\cos[32.4°-(\beta+\beta_G)]$	+.3306	+.3312	+.3304

Six Degree of Freedom Simulation

COMPUTER INTERCONNECT TRUNKING

PARAMETER	Origin		Console 4	Console 5	Console 5	Simulator Console	
	Cons.	Page	Trunk No.	Input No.	Output No.	Input No.	Readout No.
0.1 U	4	1	T 20	760	80	TH 41	00
$-500\ (\beta + \beta_G)$	4	1	T 28	768	81	TH 42	01
$10\ (\Delta\alpha + \alpha_G)$ Deg	4	2	T 16	676	82	TH 43	02
$\dot{H}/50$ (ft/sec)	4	4	T 18	678	83	TH 44	03
H/3000 (ft)	4	4	T 15	675	84	TH 45	04
$100\ \phi$ (Rad)	4	5	T 17	677	85	TH 46	05
$-100\ \theta$ (Rad)	4	5	T 19	679	86	TH 47	06
$-100\ \gamma$ (Rad)	4	4	T 13	673	87	TH 48	07
N_L (g Meter)	4	2	T 72	872	88	TH 49	08
$-V_{Gust}$	4	13	T 63	863	89	TH 50	09
Function Generator	5	-	--	-	90	TH 51	10
Terrain Model	S	17	T 65	865	91	TH 52	11
Horiz. Needle Error	4	17	T 01	661	92	TH 53	12
500 Stabilizer Trim	S	7	T 73	873	96	TH 57	16
Stick	S	7	T 75	875	97	TH 58	17

Six Degree of Freedom Simulation

COMPUTER INTERCONNECT TRUNKING, CONT.

PARAMETER	Origin		Console 4 Trunk No.	Console 5 Input No.	Console 5 Output No.	Simulator Console	
	Cons.	Page				Input No.	Readout No.
$-T/qS$ (Throttle)	S	7	T 74	874	98	TH 59	18
50 δ_{Wheel} (to δ_{sp})	S	6	T 64	864	99	TH 60	19
50 δ_{Wheel} (to δ_{ail})	S	10	T 12	672	99	TH 60	19
Noise Generator	1	13	T 40	-	--	From T 140, Cons. 1	

Six Degree of Freedom Simulation

MEAN ABSOLUTE VALUE CIRCUITS

PARAMETER	ORIGIN			Cons 5	SIMULATOR				Cons 5	Console 4	Switch #
	Ampl # Page #	Pot Trunk	Pot Setting	Input Output	HG Ampl #	Integ Ampl #	Mul #	Input Output	Input Output	Readout	(Cons 4)
100 θ	A 26 5	P 06 T 33	.1000	773 480	S 07	S 13	S 1	LH 41 LH 53	492 961	T 81	10
$\overset{N}{N_L}$	A 47 2	Q 91 T 24	.1000	764 481	S 08	S 14	S 2	LE 42 LH 54	493 962	T 82	11
Horiz. Needle	F 70 17	P 08 T 02	.1000	662 482	S 35	S 15	S 3	LH 43 LH 55	494 963	T 83	02
U	M 69 1	Q 90 T 32	.0100	772 483	S 36	S 16	S 6	LH 44 LH 56	495 964	T 84	12
α_{Gust}	A 40 13	Q 06 T 06	.0500	666 486	S 39	S 19	S 9	LH 47 LH 59	498 965	T 85	22
$\dot{H}$	A 54 4	Q 59 T 35	.0100	775 485	S 38	S 18	S 8	LH 46 LH 58	497 966	T 86	13
Stick	S 32 14	29 --	.0100	--- ---	S 40	S 20	S10	--- LH 60	499 967	T 87	--
Running Time	A 50 3	Q 50 --	.0033 (5 Min)	--- ---	--	--	-	--- ---	--- ---	A 50	03

RECORDER ASSIGNMENT SHEET

PARAMETER	CHANNEL	V/CM	FROM	PAGE	NOTES
$100\ \theta$	1 A	2	MUL 12	4	
$10(\Delta\alpha+\alpha_G)$ DEG	1 B	5	A 82	2	
N_L (CG) ft/sec^2	1 C	10	A 47	2	
$-2\ \delta_e$	1 D	20	A 84	7	
$-0.4\ \delta_{ail}$ (Total)	1 E	10	A 37	10	$5^\circ = 3.5V$
$1.667\ \delta_{sp}$ DEG	1 F	20	A 04	6	
H/3000 ft	1 G	10	A 05	4	
Stick	1 H	50	MUL 20	11	
$0.5\ \Delta U$	2 A	20	M 69	1	
α Gust	2 B	5	SW 22	13	
$\dot{H}/100$	2 C	10	A 08	4	
$200\ \dot{\theta}$	2 D	2	A 27	5	
$200\ \delta_{ail}$ (Left)	2 E	50	A 39	10	$20^\circ = 60V$
Terrain Model	2 F	2	A 07	10	
Horiz. Needle Error	2 G	2	A 37	10	

DIRECT LIFT CONTROL
LIMITER ASSIGNMENT SHEET

Limiter No.	Page No.	Parameter	Flight Condition 1		Flight Condition 2		Flight Condition 3	
			Low Setting	High Setting	Low Setting	High Setting	Low Setting	High Setting
L07	10	Right Aileron	-.5900	.5900	-.5900	.5900	-.5900	.5900
L09	6	Left Spoiler	.0000	.6666	.0000	.6666	.0000	.6666
L27	11	Spoiler Controller	-1.3000	1.3000	-1.3000	1.3000	-1.3000	1.3000
L38	6	Right Spoiler	-.6666	.0000	-.6666	.0000	-.6666	.0000
L39	10	Left Aileron	-.5900	.5900	-.5900	.5900	-.5900	.5900
L48	11	Elevator Controller	-1.3000	1.3000	-1.3000	1.3000	-1.3000	1.3000
L49	11	Elevator Controller	-.2950	.2950	-.2950	.2950	-.2950	.2950

Appendix G

Pilot Experience

Pilot One

Pilot one is a senior pilot in the USAF with seven years of rated experience. He has flown 1800 hours in C-124's, 1000 hours in the HU-16 (Albatross), 200 in C-47's, 200 in light aircraft, and has spent approximately 100 hours flying in simulators.

Pilot Two

Pilot two has eight years rated experience in the USAF and is a senior pilot. He has flown 2200 hours in KC-135 tankers, 250 hours in C-47's in addition to pilot training, and some light aircraft time.

Pilot Three

Pilot three is also a senior pilot in the USAF with eight years of rated experience. He has flown 2475 hours in B-47's and B-52's in the capacity of aircraft commander. He also flew 350 hours in T-29's/C-131's as well as his pilot training and some light aircraft time.

Pilot Four

Pilot four has 15 years of rated experience and is a command pilot in the USAF. He is an aircraft commander with 2900 hours in T-29's/C-131's and C-119's, and 70 hours in C-118's and C-97's, plus light aircraft time.

Pilot Five

Pilot five is a senior pilot with seven years of rated experience in the USAF. His time includes 3300 hours in C-124's, 200 hours in T-29's, and 240 hours in pilot training.

In addition, he was an instructor pilot in the C-124 simulator at Hickam AFB, Hawaii, for a year prior to attending AFIT.

Pilot Six

Although pilot six is not an Air Force pilot, he has over 100 hours in light aircraft and a wealth of information concerning aircraft simulations and simulator flight tests. He has flown approximately 850 hours in this and other cockpit simulators used in the past four years at AFFDL.

Pilot Seven

Pilot seven is a member of the Royal Canadian Air Force with 18 years of rated experience. He has flown 2000 hours in F-86's and T-33's, and 1000 hours in Dakota's, C-45's, and other light aircraft.

Pilot Eight

Pilot eight has been rated for eight years and is a senior pilot in the USAF. Since pilot training, he has flown 2100 hours in four different helicopters, 130 hours in C-47's, and has other light aircraft and simulator flight time.

Appendix H

Tabular MAV Data

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MEAN ABSOLUTE VALUES
(FLIGHT CONDITION 1)

	θ (Deg.)				N_L (Ft./Sec.^{2})				U (Knots)			
	DLC Aircraft		Basic Aircraft		DLC Aircraft		Basic Aircraft		DLC Aircraft		Basic Aircraft	
	Turbulence		Turbulence		Turbulence		Turbulence		Turbulence		Turbulence	
PILOT	No	Yes	No	Yes	No	Yes	No	Yes	No	Yes	No	Yes
1	.127	.122	.510	.673	.356	.489	.908	.880	11.6	11.0	6.5	8.0
2	.092	.111	.508	.506	.314	.389	1.180	.738	5.7	3.2	2.5	3.8
3	.092	.011	.713	-	.637	.947	1.610	-	3.1	3.5	2.9	-
4 **												
5	.211	.238	.396	.561	.368	1.000	1.480	1.260	10.0	13.0	7.3	3.8
6	.250	.434	.908	1.000	.343	.459	.544	1.140	2.4	5.1	40.0	12.1
7 **												
8 **												
AVERAGE	.154	.183	.606	.685	.403	.658	1.144	1.004	6.6	7.2	11.8	7.0

** Pilots 4,7,and 8 did not fly this flight condition.

MEAN ABSOLUTE VALUES
(FLIGHT CONDITION 2)

PILOT	θ (Deg.)				N_L (Ft./Sec.^{2})				ΔU (Knots)			
	DLC Aircraft		Basic Aircraft		DLC Aircraft		Basic Aircraft		DLC Aircraft		Basic Aircraft	
	Turbulence		Turbulence		Turbulence		Turbulence		Turbulence		Turbulence	
	No	Yes	No	Yes	No	Yes	No	Yes	No	Yes	No	Yes
1	.232	.284	1.040	.834	.390	.561	1.172	.902	3.1	2.8	13.8	8.1
2	.108	.368	1.930	1.800	.255	.322	.868	.938	5.1	4.9	13.8	28.1
3	.125	.277	.532	1.760	.786	.845	.766	.985	2.4	4.9	4.0	24.8
4	.141	.062	.750	-	.291	.371	.684	-	2.6	2.8	6.2	-
5	.193	.267	.845	1.780	.914	1.130	1.910	2,420	2.1	2.2	14.8	19.2
6	.173	.126	.806	1.530	.286	.353	.710	.729	1.0	3.1	6.3	18.1
7	.101	.192	.611	.556	.283	.370	1.355	1.442	2.8	2.8	2.8	5.1
8 **												
1 Extra Run	.155	.168	.820	.650	.388	.307	.858	.672	2.9	3.6	14.7	11.1
AVERAGE	.153	.218	.917	1.274	.449	.532	1.040	1.155	2.8	3.4	9.6	16.4

** Pilot 8 did not fly this flight condition.

MEAN ABSOLUTE VALUES
(FLIGHT CONDITION 3)

PILOT	θ (Deg.)				N_L (Ft./Sec.^{2})				ΔU (Knots)			
	Turbulence		Turbulence		Turbulence		Turbulence		Turbulence		Turbulence	
	No	Yes	No	Yes	No	Yes	No	Yes	No	Yes	No	Yes
1	.078	.090	.414	.602	.308	.357	.603	.536	7.1	6.7	9.6	17.6
2 **												
3	.081	.124	.480	.391	.281	.334	.652	.493	2.6	6.8	2.9	3.5
4	.109	.112	.505	.500	.303	.357	.359	.988	8.3	10.9	11.4	9.3
5	.111	.123	.562	.548	.701	.741	1.960	.950	7.2	11.2	4.2	6.4
6	.078	.074	.398	.412	.287	.419	.416	.534	1.0	2.1	2.7	2.1
7	.092	.130	.572	.775	.396	.492	.661	1.093	3.7	3.6	6.7	9.8
8	.096	.062	.545	.548	.297	.342	.643	.932	2.0	2.7	8.9	6.2
1 Extra Run	.077	.111	-	-	.318	.370	-	-	5.0	2.5	-	-
AVERAGE	.090	.103	.497	.539	.361	.427	.753	.790	4.6	5.8	6.6	7.9

** Pilot 2 did not fly flight condition 3

MEAN ABSOLUTE VALUES
(FLIGHT CONDITIONS 2 and 3)

PILOT	HN (Feet) (F.C. 2)				α_G (Ft./Sec.) (F.C. 3)							
	DLC Aircraft		Basic Aircraft		DLC Aircraft		Basic Aircraft					
	Turbulence		Turbulence		Turbulence		Turbulence					
	No	Yes	No	Yes	No	Yes	No	Yes				
1	-	8.3	-	6.2	-	3.34	-	3.87				
2*	-	24.0	-	37.5	-	-	-	-				
3	-	38.0	-	48.0	-	2.16	-	3.55				
4	-	26.8	-	-	-	2.08	-	1.63				
5	-	21.4	-	20.4	-	3.11	-	2.43				
6	-	13.5	-	6.5	-	3.93	-	1.14				
7	-	10.2	-	17.7	-	2.57	-	4.83				
8**	-	—	-	-	-	1.48	-	3.43				
1 Extra Run	-	-	-	-	-	2.75	-	-				
AVERAGE	-	20.3	-	22.7	-	2.68	-	2.97				

* Pilot 2 did not fly flight condition 3.
** Pilot 8 did not fly flight condition 2.

Appendix I

<u>Pilot Tape-Recorded Comments</u>

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Pilot Tape Recorded Comments

Most pilots tape-recorded their comments and answers to the questions on the task sheets. A few preferred written comments, and some made no comments at all due to lack of time. These comments have been reproduced with a minimum of editing.

Run 3-GGC-68-Terrain Following

The most annoying thing was lateral control ... every once in a while the aircraft would run away in a right bank and one must hold almost full left aileron. This is particularly noticable in high rates of climb. The controlled aircraft handles very satisfactorily ... much better than the basic aircraft. I believe the basic aircraft is way too sensitive and most of the time it is caused by slop in the deadspace around the stick neutral. I personally know the actual aircraft is not that sensitive. If I tried to hold 200-300 fpm rate of climb, just touching the stick might make it vary from 0-600 fpm, and it would jump just by touching the stick. Some tendency to roll also in basic aircraft.

(Author's comments: Pilot three is an ex-B-52 pilot, so stick sensitivity of the basic aircraft was reduced based on his comments above. He then flew the simulation again, and his comments are shown below. The right bank problem was eventually traced to a bad amplifier. The deadspace slop was corrected the following day by Lear-Siegler maintenance personnel.)

Run 3-GGC-68-Terrain Following

Performing the task: Difficult to control in gust. Once the gust was turned off, it was easier to maintain desired task. Airspeed easily maintained? Not bad ... within ±5 knots. I think it was harder to maintain in the basic aircraft. Aircraft is still sensitive in this mode. Heading fairly difficult to maintain ... still wanted to roll. The roll is rather slow, even with 10° bank it is very slow. PIO? In basic aircraft very much so. No PIO in the controlled aircraft. Effect of turbulence? Task practically impossible with turbulence in basic aircraft. Displayed information adequately? Not too bad, but should increase deviation (Ed. On the horizontal needle which displayed simulated terrain.) because in actual aircraft with terrain avoidance radar, if you need a 200-300 fpm you will get about ½ inch displacement on the indicator and you need only to pull back the stick to put airplane on the indicator to get the rate of climb necessary to take care of it. I would personally like to see a greater amplitude depicted on the horizontal needle indicator.

(Author's comments: Basic airplane stick sensitivity was reduced further until pilot three said it felt about right. The roll problem was corrected by adding the lateral trim button. Horizontal needle amplitude was increased. However, this method of presenting terrain was never entirely satisfactory.)

Run 5-GGC-68-Terrain Following

Basic aircraft: I found it was difficult to make small corrections in my rate of climb or descent and there was a tendency to oscillate about the desired rate. However, with concentration I could stop the oscillations, but not necessarily at the rate desired.

Controlled aircraft: I found it very easy, light work load, to achieve rate of climb and descent I wanted. However I found the control movements different than I had expected, particularly at climbs and descents of 300 fpm. Once established, I could let go of the stick and maintain that rate of climb. This was not true at 500 fpm. I had to keep in pressure, but to level off I expected to release the stick and it would return to neutral and the aircraft would level off, but I had to push forward on the stick and then return it to the neutral position.

(Author's comments: Later analysis on the two degree of freedom simulation showed this problem was due to the $\theta/\dot{\theta}$ washout circuit in the elevator controller. Eliminating the washout integrator eliminated the problem.)

Run 1-GGC-68-Terrain Following

In the basic aircraft I found there was a great tendency toward PIO. I think most of it is due to the deadzone in the stick. The sensitivity of the stick might be high yet. The airspeed and heading required extra concentration and due to PIO it was probably ignored a little too much. The task is difficult due to PIO and stick sensitivity. Turbulence was very noticable and made the task even harder to complete.

Controlled aircraft: The aircraft is not nearly as sensitive in the DLC mode and the addition of turbulence did not significantly affect the mission. It is possible to hold almost any rate of climb or descent desired.

The terrain model is not the best ... the change in the horizontal needle is so small that it is almost impossible to see the change as it occurs. The period of oscillations of the terrain seems a little rapid but then it is taking place at high speed so maybe it is logical. In the basic aircraft the task is nearly impossible due to PIO.

Heading and airspeed are not difficult to maintain but it is strange to see airspeed increase in a climb without the addition of throttle. (Ed. Referring to the controlled aircraft. In a climb, the spoilers move full down from the 20° bias position, and the decrease in drag causes speed to increase.)

Run 5-GGC-68-Landing

First the basic aircraft. In the basic aircraft, I was able to perform the task, however, I felt my accuracy was poor due to PIO around my desired rate of climb or descent rather than holding a constant rate. As for airspeed: I wasn't paying too much attention to airspeed and wasn't trying to maintain heading, but rather made turns with 10^o of bank. PIO was very prevalent.

Turbulence: I felt in the basic aircraft that this made the task a real challenge although not impossible. I concentrated on the attitude indicator and crosschecked the rate of climb. I felt the vertical velocity instrument varied too rapidly to use as primary aircraft control, so tried to use the attitude indicator as the primary instrument and cross checked with vertical velocity.

Controlled Aircraft: Task was easy. Airspeed was not watched very closely, so can't say too much. As far as heading was concerned, I was shooting for small turns, keeping in mind the $\pm 90^o$ limitation on the simulation. I found no tendency to PIO. I enjoyed flying with turbulence more than without in the controlled aircraft because once I had established a turn or climb, I almost became bored without turbulence. Without PIO being present, it was very easy to use the rate of climb meter as the primary instrument and I could set any rate of climb or descent I wanted.

Run 2-GGC-68-Landing

Aircraft was easy to control in both basic and control mode. Basic aircraft was extremely sensitive in pitch and roll but not difficult to perform task.

Controlled aircraft extremely stable and easy to hold rates of climb. There was some tendency to roll . . . a little bit more than in the basic airplane. Overall, no trouble in holding airspeed, heading, or rate of climb.

Turbulence in basic aircraft: I really didn't think it had much effect because simulator is so sensitive anyway in that mode. Can't tell if it is the simulator or turbulence. With the controller in, the aircraft is just like a "rock". . . very stable and easy to fly.

Run 7-GGC-68-Landing

Basic aircraft: I didn't do much trimming as it appeared to stay in trim rather well for the changes necessary. The task was rather difficult, especially to settle down on a desired rate of climb or descent. The airspeed was not difficult to maintain as it stayed pretty close to the desired value. The heading was not particularly easy to maintain, but was not too difficult either.

I found some tendency to PIO in trying to settle it on desired rates of descent or climb. The turbulence made the task appreciably more difficult.

The presentation of information was adequate . . . I was trying to set the desired rate of descent by pitch attitude and I found the attitude indicator was such that a very small adjustment gave a greater rate of descent than I desired and hence PIO.

Controlled aircraft: I was able to perform the task very easily and was able to settle the aircraft on the rate of descent I desired. The airspeed was not difficult to maintain as it stayed close to the desired value. I found it a little strange to gain airspeed in a climb and lose it in descent but this was the only real problem. The heading was about the same as in the basic aircraft but now I could put more attention to it as less attention was needed to establish rates of climb or descent so it was actually easier to maintain desired results. Turbulence had a very minor effect and did not degrade the task appreciably.

The information presented was quite adequate in this case.

Run 1-GGC-68-Landing

Basic aircraft: The basic aircraft is less sensitive in this flight condition, and after a little practice it is easier to maintain an almost constant rate of climb or descent. The stick sensitivity is better but could maybe be reduced further. Still some PIO and the turbulence gives an increase in workload.

The controlled aircraft is very easy to fly. I can set a desired rate of climb exactly and hold it for as long as I like. It helps make the airspeed and heading easier to achieve also.

The glide slope presentation is pretty good although I think a greater rate of descent would simulate more effectively the actual glideslope presented on an ILS. The turbulence was not very noticeable in the controlled aircraft.

Run 4-GGC-68-Landing

(Written comments from pilot evaluation sheet.)

The basic aircraft is much more sensitive than I am used to. This is possibly due to the high airspeed that is carried. I don't feel I did very well and the workload is rather heavy.

In the controlled aircraft, the complete task was very easy to complete even in turbulence. The workload is very light and I was able to do much better.

Run 8-GGC-68-Refueling

With the controller on, performing the task was very easy while with it off it was difficult to hold pitch at high speeds because of the sensitivity of the aircraft. Airspeed was not hard to maintain ... my personal cross-check was a little slow here and this caused it to get off some. I don't think the airspeed is difficult to maintain. Heading was not difficult to maintain either with or without the controller, but I did notice that the lateral mode seemed a little more sensitive with the controller on. With turbulence, the controlled aircraft was no problem at all but without the controller it made the task more difficult.

As for the information presented ... it was good. The attitude indicator was a little sticky, otherwise it was good.

General comments: With the controller on, the aircraft was very easy to fly and turbulence was no problem at all. All in all, I would rate the controller very high as a needed item for future aircraft.

Run 3-GGC-68-Refueling

I thought the aircraft improved immensely over the previous flights that I made, especially in the controlled aircraft. I thought it was very easy to fly today, and I could hold any rate of climb or descent with no sweat at all.

In the basic aircraft it is still rather sensitive, it lags the control input, just about like in the B-52. It has improved also as you have gotten rid of all the slop in the neutral position of the stick. It was quite a pleasure to fly today.

Run 5-GGC-68 Refueling

With the controlled aircraft it was very easy to perform the maneuver with no tendency toward PIO ... very enjoyable to fly in this mode. The addition of turbulence had no derogatory effects on performing the task. It increased the workload slightly as would be expected, however, in the basic aircraft it was much more noticeable with turbulence.

In order to avoid merely holding a position when I was stabalized in bank angle and rate of climb, I would initiate a turn in the opposite direction to give me a little more to do to possibly simulate an actual situation more adequately.

In the basic aircraft, the performance of this maneuver at 33,000 feet was sort of wild because of the reduced lateral control at this altitude and the extreme sensitivity that I found in controlling the rate of climb. I was oscillating anywhere from 100-200fpm. This oscillation was primarily due to getting out of phase with the aircraft's own oscillation and the indication on the vertical velocity indicator.

I found I could bracket the desired rate of climb a little easier by going over to the pitch angle indicator (θ meter) rather than the attitude indicator itself. I think some sort of pitch angle indicator is essential here because the change in attitude indication was so small that it didn't really enable one to control the aircraft adequately using just the attitude indicator as a reference. The motion base made it very realistic for turns especially.

Run 1-GGC-68-Refueling

The controlled aircraft is so stable that I almost forgot what I was supposed to be doing. The turbulence has little effect on the performance of the task and it is very easy to keep airspeed and heading at the desired values.

The basic aircraft is much better now that the stick sensitivity is reduced. With only a little practice, it is possible to establish desired rates of climb or descent. The turbulence adds quite a bit to the workload. There is still a slight tendency towards PIO and I found that there is a little sticking in the vertical velocity indicator.

I think the moving base simulator added a lot to the simulation in roll and pitch. It would be interesting to apply the turbulence to the moving base as well as the equations of motion.

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13. ABSTRACT
A DLC (Direct Lift Control) system was designed for the LAMS B-52 test aircraft using optimization theory. Spoilers, symmetric ailerons, and elevator respond to normal longitudinal column movements, achieving incremental rates of climb up to 500 fpm from trim without pitching the aircraft. DLC reduces normal acceleration 50%, both in and out of turbulence, and improves airspeed control in precision maneuvering tasks: Approach and landing, in-flight refueling, and terrain following. The closed-loop DLC controller, consisting of fixed-gain normal acceleration, pitch angle, and pitch rate feedbacks to the three control surface servos, was demonstrated by a piloted six degree of freedom analog simulation.

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