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TITLE DESIGN OF A POWERED ELEVATOR CONTROL SYSTEM -
FINAL REPORT

MODEL _____

ISSUE NO. _____ TO: _____ _____ (DATE)

NASA Contract NAS 2-7293

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1.0 SUMMARY

In October 1972, The Boeing Company was awarded a contract by NASA Ames Research Center that led to the design, fabrication and flight testing of a powered elevator system for the Augmentor Wing Jet STOL Research Aircraft (AWJSRA or Mod C-8A). The system replaces a manual spring tab elevator control system that was unsatisfactory in the STOL flight regime.

Pitch control on the AWJSRA is by means of a single elevator control surface. The elevator is used for both maneuver and trim control as the stabilizer is fixed. A fully powered, irreversible flight control system powered by dual hydraulic sources was designed. The existing control columns and single mechanical cable system of the AWJSRA have been retained as has been the basic elevator surface, except that the elevator spring tab is modified into a geared balance tab. The control surface is directly actuated by a dual tandem moving body actuator. Control signals are transmitted from the elevator aft quadrant to the actuator by a linkage system that includes a limited authority series servo actuator. Artificial feel forces are provided by a dualized feel system that schedules the feel gradient as a function of impact pressure. Trim control is obtained by shifting the neutral point of the feel system. Manual reversion control of the elevator is possible following the loss of both hydraulic systems.

The design was implemented on a government furnished C-8A tail assembly at Boeing, Seattle. The modified tail assembly was subsequently shipped to AMES RESEARCH CENTER for mating with the AWJSRA airframe. Following the functional and engineering ground tests, the AWJSRA was flight tested. The powered elevator control system has been demonstrated to be airworthy and to meet its design requirements. No adverse system characteristics were noted by the test pilots. Control problems encountered in the STOL flight regime with the spring tab elevator system were eliminated by the powered elevator installation.

Elevator deflection capability above 136 knots is higher than desirable due to lower-than-predicted hinge moments. This can produce tail loads in excess of design loads. The following recommendations are therefore made:

1. Pilots should be cautioned not to make large elevator inputs above 120 knots.

2. The powered elevator system should be flight tested with the actuator supply pressure reduced to 1500 psi. This change should be permanently incorporated if system performance is found acceptable.

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NOMENCLATURE

AWJSRA	--	Augmentor Wing Jet STOL Research Aircraft
C_{He}	--	Elevator hinge moment coefficient
$C_{He_{\delta_t}}$	--	$\partial C_{He}/\partial \delta_t$, elevator hinge moment coefficient variation with geared tab deflection
F_p, F_s	--	Column force
G/S	--	Glide slope
h	--	Altitude
L_T	--	Horizontal tail lift
M_T	--	Horizontal tail bending moment @ 25% MAC
N_H	--	High pressure engine shaft speed
n_z	--	normal load factor
PCU	--	Power control unit
PLF	--	Power for level flight
PIO	--	Pilot induced oscillation
qc	--	impact pressure
SAS	--	Stability augmentation system
V_A	--	Design maneuvering speed
V_B	--	Design speed for maximum gust intensity
V_c	--	Design cruise speed
V_D	--	Design dive speed
V_e	--	Equivalent airspeed
V_f	--	Design flap speed
β	--	Sideslip angle
γ	--	Flight path angle

δ_c -- Column position

δ_e -- Elevator position

δ_F -- Flap position

$\delta_{N,\nu}$ -- Pegasus nozzle position

δ_R -- Rudder position

δ_t -- Geared balance tab position

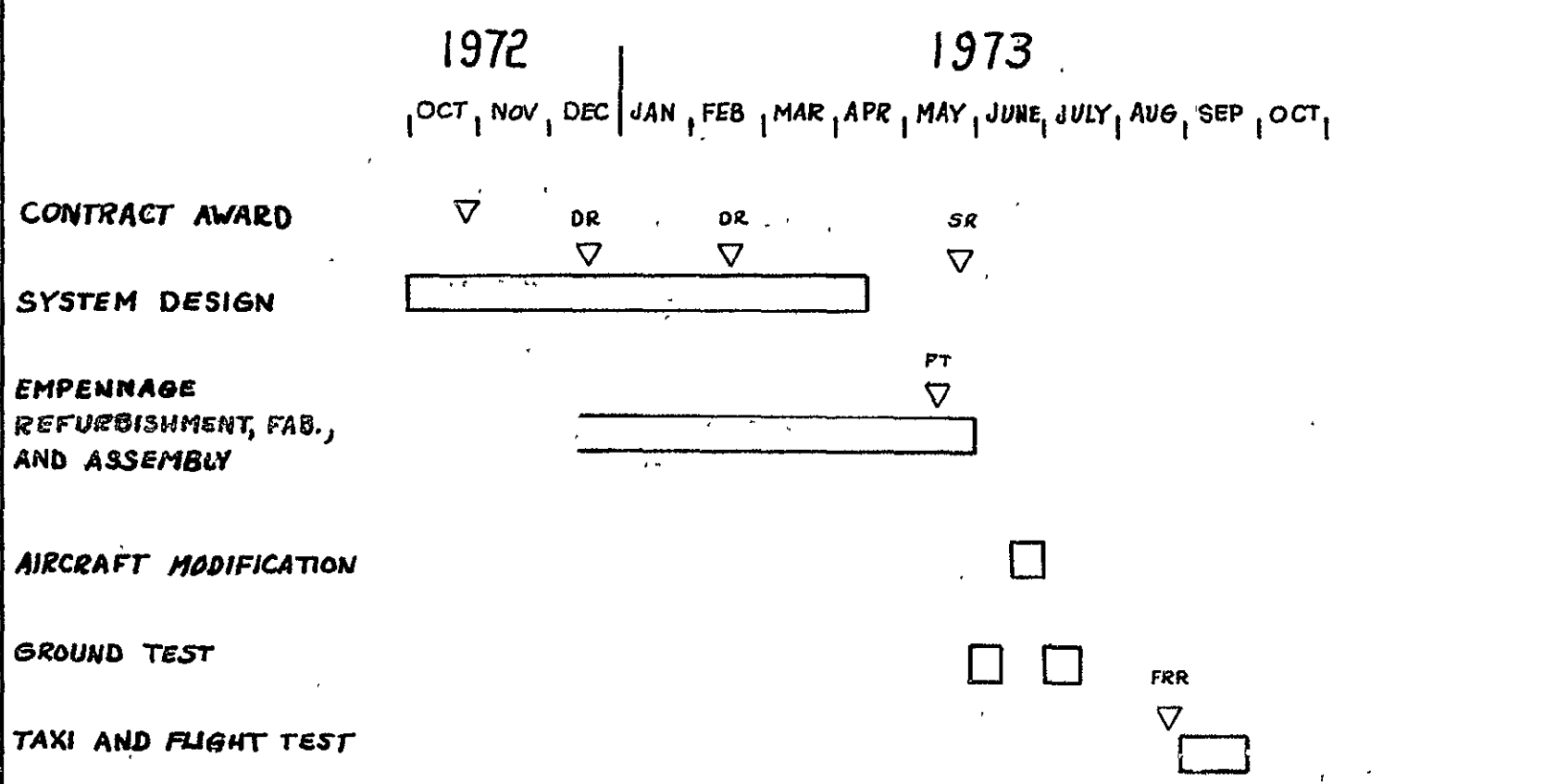
δ_{TT} -- Trim tab position

2.0 INTRODUCTION

In October 1972, The Boeing Company was awarded Contract NAS2-7293 to design a powered elevator control system for the Augmentor Wing Jet STOL Research Airplane (AWJSRA). The contract included component fabrication, refurbishment and modification of a government-furnished tail assembly.

The contract was subsequently enlarged to cover installation of the modified tail assembly and related components on the AWJSRA, as well as ground and flight testing to verify airworthiness of the powered elevator installation.

This document summarizes the modification program for the powered elevator system. The schedule and significant milestones of the program are summarized in figure 1.



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PROGRAM SCHEDULE
POWERED ELEVATOR SYSTEM
BOEING
MOD C-8A
FIG. 2.0-1
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3.0 AIRPLANE DESCRIPTION

3.1 GENERAL FEATURES

The AWJSRA is a 45,000 pound, 50 psf wing loading, turbo-fan powered airplane designed for research in the STOL terminal flight regime. The airplane is derived from a deHavilland C-8A "Buffalo" airframe. Augmentor-wing jet flaps, blown and drooped ailerons and leading edge slats have been added to produce high lift capability for STOL research. Wing span has been shortened to increase wing loading. Two Rolls Royce Spey 801SF jet engines provide blowing air to the flaps and ailerons via ducts as well as direct hot thrust through vectorable Pegasus nozzles. A three view of the airplane is shown in figure 3.1-1. Airplane operational data are presented in table 3-1.

The AWJSRA has two independent, equal capacity 3000 psi hydraulic systems using MIL-H-5606 fluid. The electrical system is powered by two engine driven 115/200 volt, three phase, 400 Hz brushless generators that are connected to normally isolated left and right a.c. busses and through rectifiers to the left and right d.c. busses. A 24 volt battery is connected to the left d.c. bus.

Lateral control is achieved through three control surfaces: ailerons, spoilers and outboard flap augmentor chokes, each contributing approximately a third of the roll control. The control surfaces are fully powered. Spoilers and chokes have single hydraulic sources, the ailerons are powered by dual hydraulics. A limited authority series SAS actuator is included in the lateral axis. Manual reversion control of the ailerons is possible following the loss of both hydraulic systems. The lateral control system is shown in figure 3.1-2.

Directional control is via a single double-hinged rudder. The surface is fully powered by a dual tandem actuator. A limited

authority series SAS actuator is included in the directional axis. No manual reversion is possible. Figure 3.1-3 shows the directional control system.

Longitudinal control is via a single elevator surface which provides both maneuver and trim control functions as the stabilizer incidence is fixed. Prior to the powered elevator modification, longitudinal control was entirely manual. A spring tab on the right side of the elevator was used to reduce stick forces. A tab on the left side manually operated by the pilot and automatically by a flap-trim interconnect, was used for trimming. The manual elevator control system is shown in figure 3.1-4.

3.2 ELEVATOR CONTROL PROBLEM

The AWJSRA was flown by Boeing and NASA-ARC 67 times with the manual spring tab elevator control system. Certain shortcomings of this system below 80 knots became almost immediately apparent.

Maximum elevator deflection was not possible without extremely high stick forces. Stick forces in excess of 100 pounds were required for full elevator deflection at 60 knots, as shown in figure 3.2-1. The elevator up float angle (figure 3.2-2) changed considerably below 80 knots due to mass and aerodynamic overbalance effects. This had an adverse effect on control feel and airplane response. The elevator dynamic response was poor. Figure 3.2-3 shows elevator position to lag pilot force by approximately 130 degrees at 0.5 Hz. This high control system phase lag made tight attitude control extremely difficult.

These shortcomings of the manual elevator control system had a detrimental effect on the evaluation of the AWJSRA in the STOL regime and led to the design of a fully powered elevator control system.

3.3 EXTENT OF MODIFICATION

Airplane modifications made to incorporate the powered elevator system were as follows:

1. A dual tandem moving body power control unit which directly powers the elevator surface was installed in the horizontal stabilizer.

2. The elevator spring tab was replaced by a geared balance tab with ratio $\delta_t/\delta_e = -0.7$.

3. The flap-trim tab interconnect was deleted.

4. A feel system was located in the vertical stabilizer to provide pilot feel forces.

5. An electric trim motor that moves the neutral point of the feel system was installed.

6. The hydraulic system was modified to provide the power to the elevator PCU, feel system and SAS actuator.

7. Structural modifications were made to the vertical and horizontal stabilizers as well as to the forward segment of the rudder surface.

TABLE 3-1

OPERATIONAL DATA.

I. PLACARDS

DESIGN FLAP SPEEDS

F	V_f Knots
75°	90
50°	90
30°	120
5.6°(UP)	160

FLAPS UP DESIGN SPEEDS

Gust V_B	= 140 knots
Maneuvering V_A	= 136
Design Cruise $V_C = V_{MO}$	= 160
Design Dive Speed	= 180

DESIGN LOADS

Flaps down $0\ g \leq n_Z \leq 2.0\ g$

Flaps up (5.6°) $-.5\ g \leq n_Z \leq 2.25\ g$

MAXIMUM ALTITUDE

15,000 feet

FLIGHT RULES

VFR

II. TYPICAL DESIGN FLIGHT CONDITIONS

TAKEOFF

δ_F = 30°, γ = 6°, V_e = 75 kts, γ_0 = 15° (2 engines, takeoff power, 45,000 lbs)

CRUISE

δ_F = up, γ = 6°, V_e = 140 to 160 kts, γ_0 = 0

LANDING

δ_F = 65°, γ = 70°, V_e = 60 kts, γ_0 = -7.5°, Δnz_{margin} = .35 g

ΔV_{margin} = 16 kts

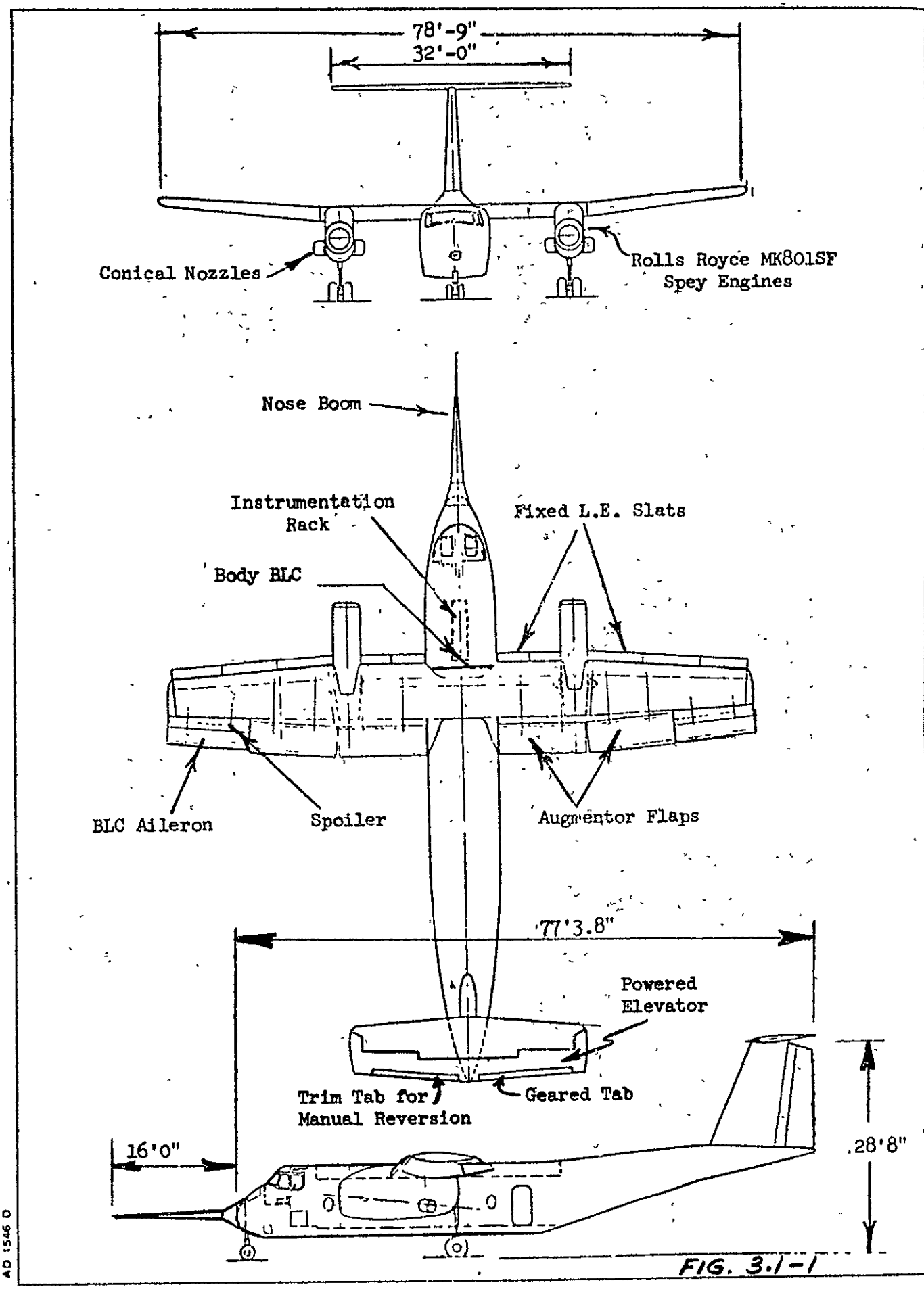

FIG. 3.1-1

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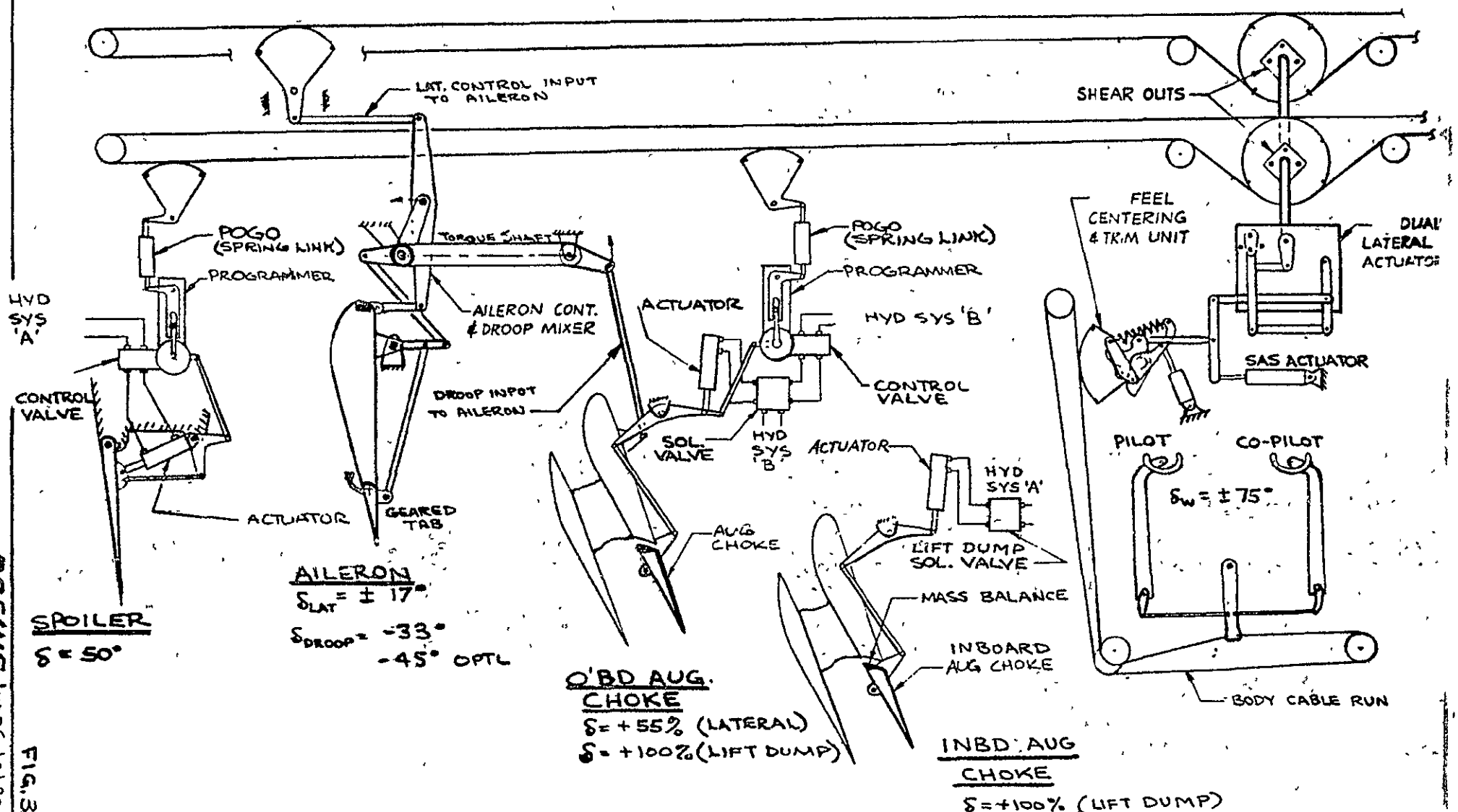
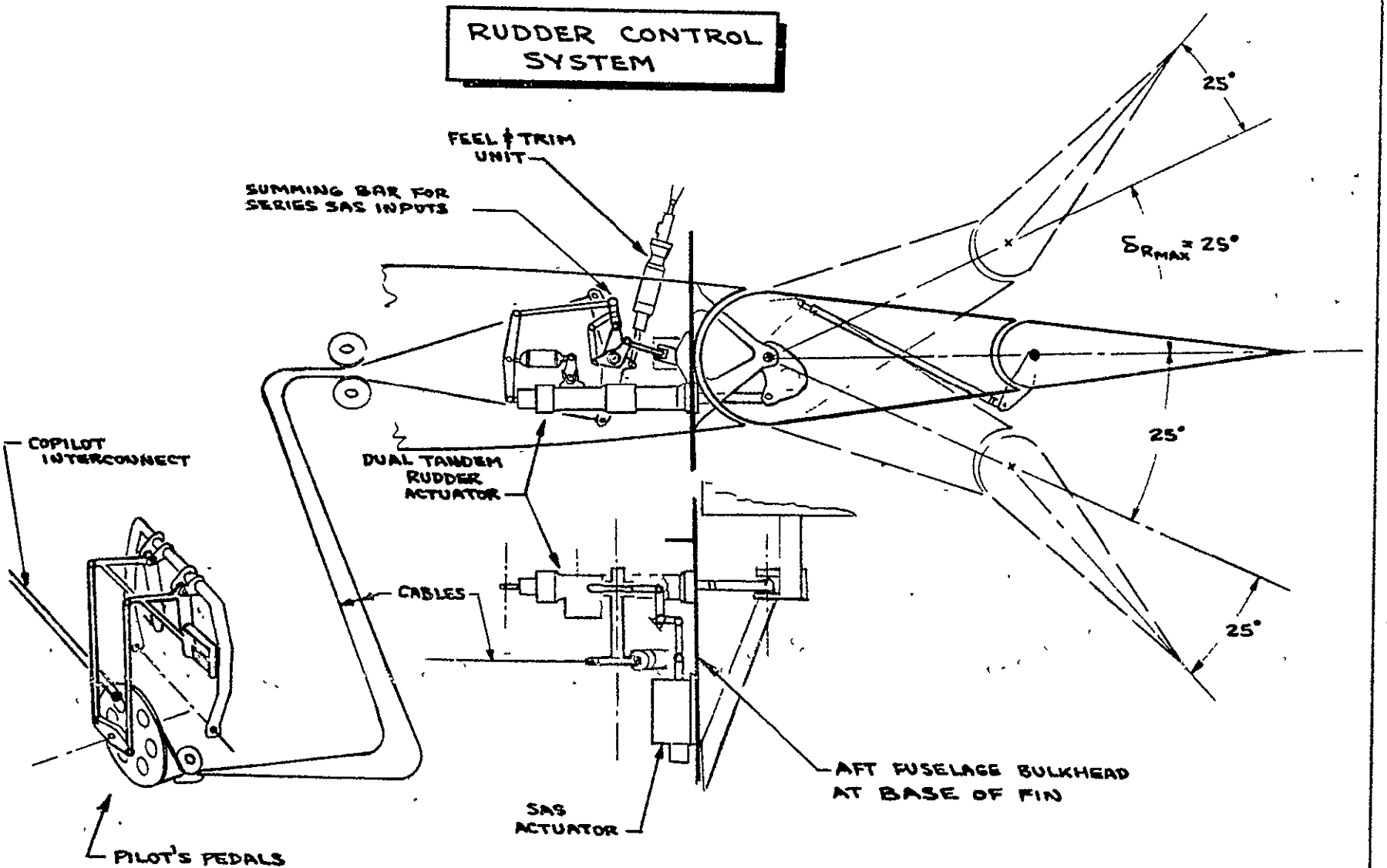
LATERAL CONTROL SYSTEM



NOTE:
ELECTRIC TRIM AT RATE OF $\dot{\delta}_w = \pm 4.2$ DEG/SEC

FIG. 3.1-2

RUDDER CONTROL SYSTEM



NOTE:
ELECTRIC TRIM AT RATE OF $\dot{\delta}_R \approx \pm .8 \frac{DEG}{SEC}$

MOD C-8A Fig 8.1-3

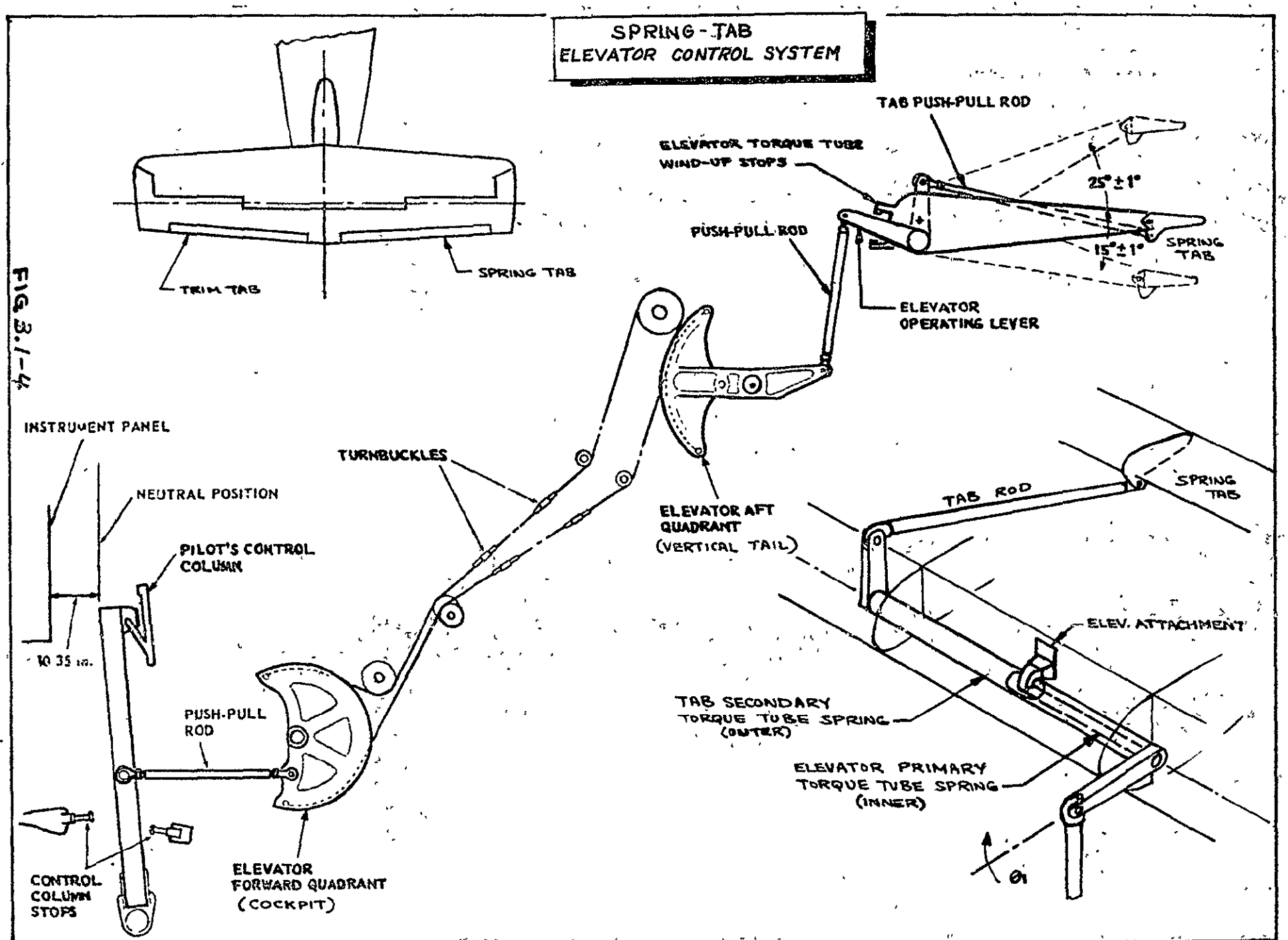

SPRING-TAB
ELEVATOR CONTROL SYSTEM
TRIM TAB
SPRING TAB
TAB PUSH-PULL ROD
ELEVATOR TORQUE TUBE
WIND-UP STOPS
25° ± 1°
15° ± 1°
SPRING TAB
PUSH-PULL ROD
ELEVATOR
OPERATING LEVER
INSTRUMENT PANEL
NEUTRAL POSITION
10 35 in.
PILOT'S CONTROL
COLUMN
TURNBUCKLES
ELEVATOR AFT
QUADRANT
(VERTICAL TAIL)
TAB ROD
SPRING TAB
ELEV. ATTACHMENT
PUSH-PULL
ROD
TAB SECONDARY
TORQUE TUBE SPRING
(OUTER)
ELEVATOR PRIMARY
TORQUE TUBE SPRING
(INNER)
CONTROL
COLUMN
STOPS
ELEVATOR
FORWARD QUADRANT
(COCKPIT)

FIG. 3.1-4

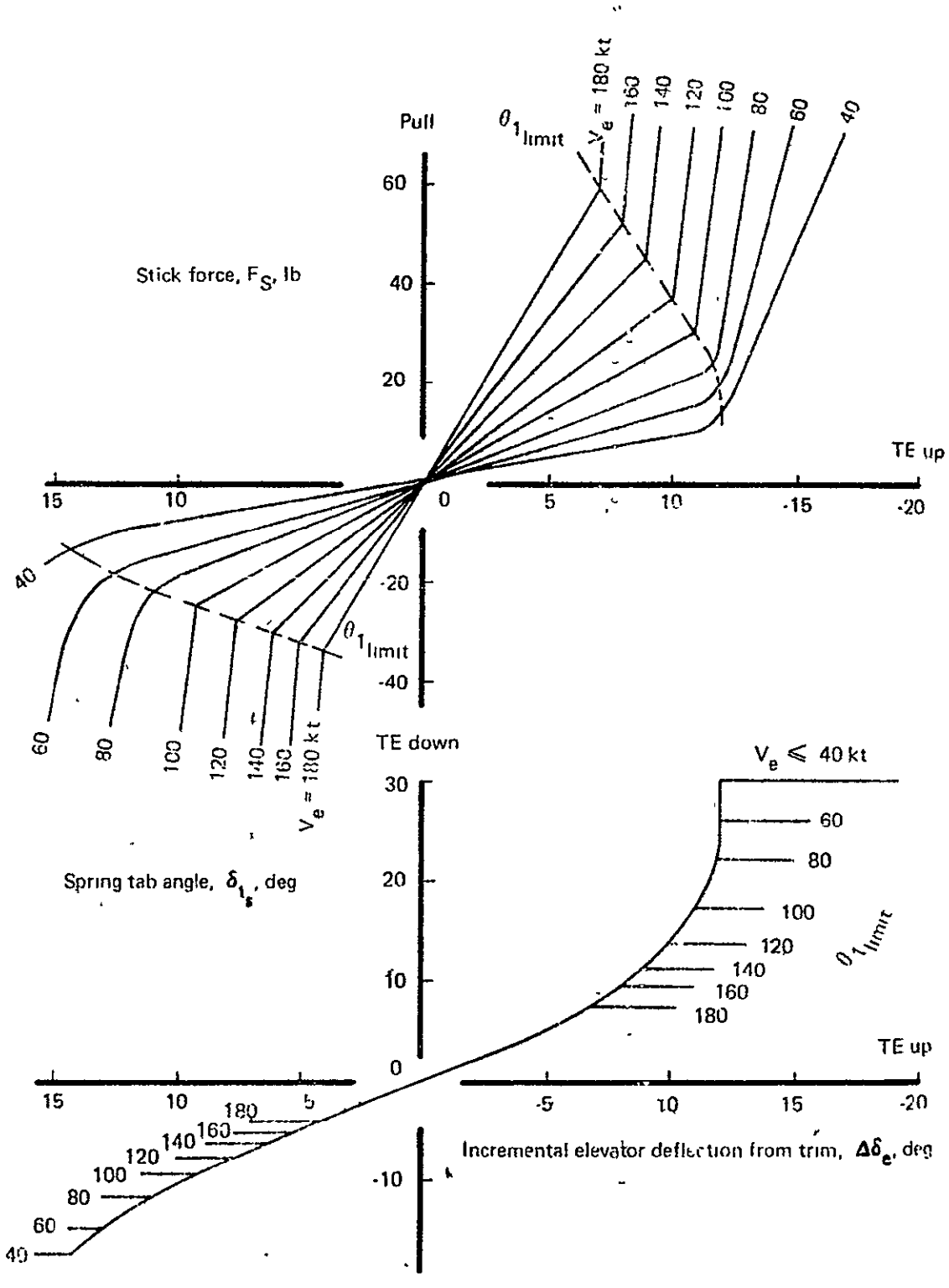


FIGURE 3.2-1 –STICK FORCE AND TAB EFFECTIVENESS BASED ON TAXI AND FLIGHT TESTING

SPRING TAB ELEVATOR CONTROL SYSTEM

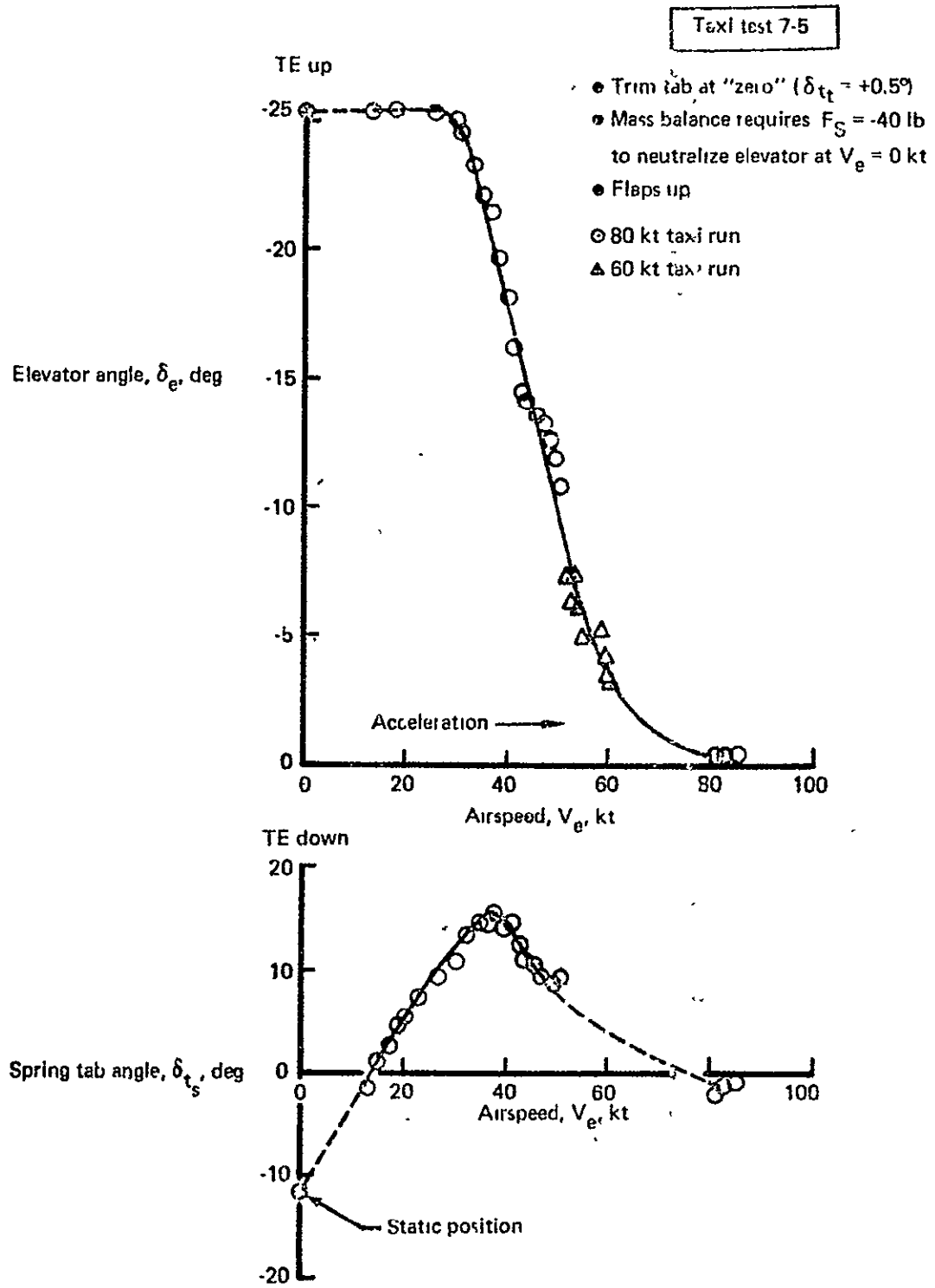


FIGURE 3.2-2 –ELEVATOR FLOAT AT ZERO STICK FORCE DURING TAXI

SPRING TAB ELEVATOR CONTROL SYSTEM

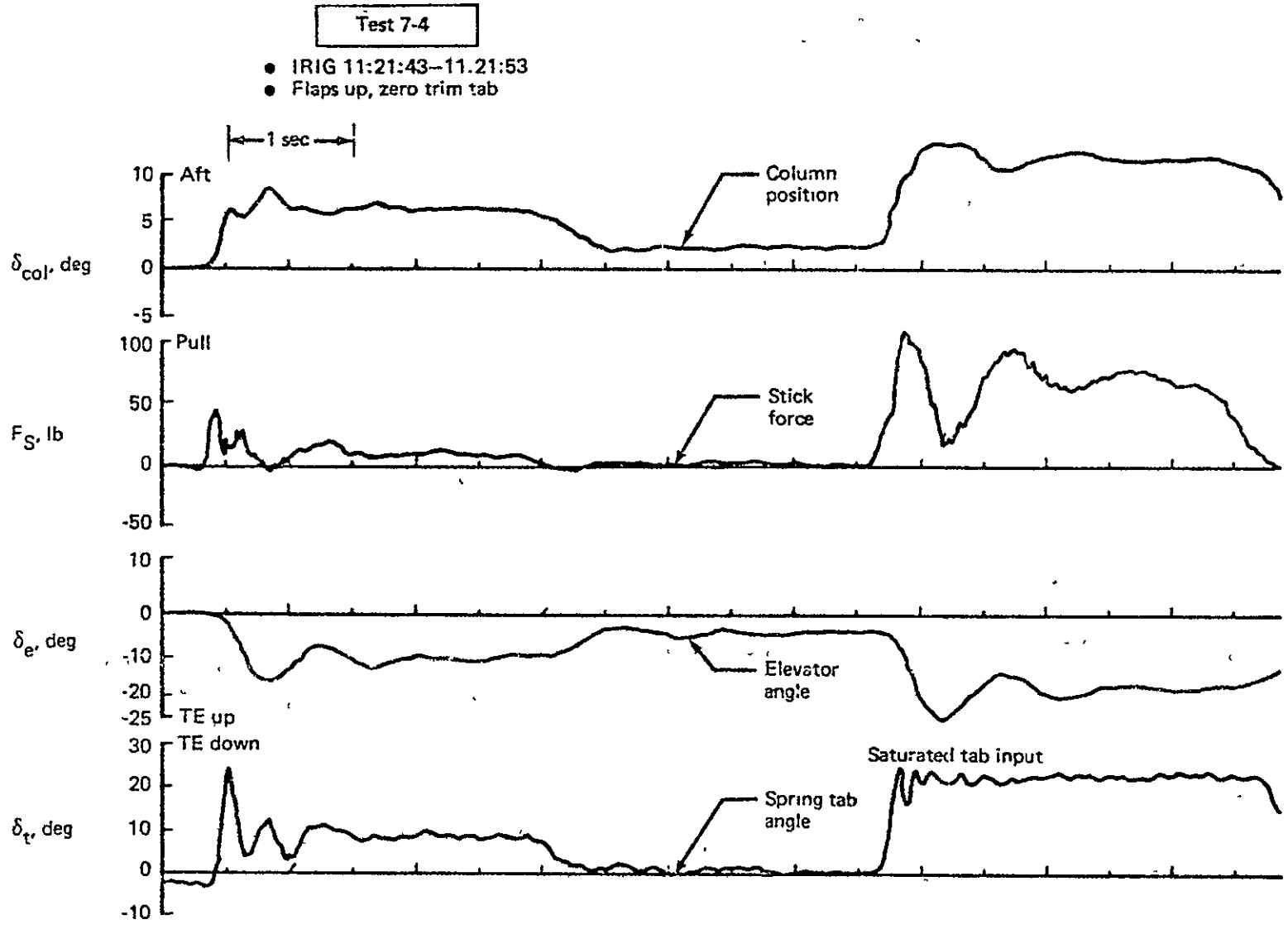


FIGURE 3.2-3 –ELEVATOR DYNAMICS AT 60 KT TAXI

SPRING TAB ELEVATOR CONTROL SYSTEM

4.0 POWERED ELEVATOR SYSTEM DESIGN AND ANALYSIS

4.1 DESIGN REQUIREMENTS

The powered elevator system has been designed to comply with the intent of the Federal Aviation Regulations, Part 25, except for deviations with respect to mechanical failure conditions. These deviations were considered acceptable for a research airplane. This allowed use of the existing single load path cable system and the single control surface and thereby minimized the cost of the modification.

Specific design requirements which determined the system configuration are as follows:

1. The powered elevator system shall provide stick forces compatible with one hand operation.

2. Loss of one hydraulic system shall not cause unsafe stick force per 'g' characteristics.

3. Elevator manual reversion shall be available for safe flight to a landing.

4. The control system shall be capable of accepting series SAS commands.

5. The control system shall be safely operable over the total flight envelope of the AWJSRA.

4.2 DATA BASE

The data base for the design of a powered elevator control system came from actual flight test results of the airplane, from the original DHC Buffalo, and from a piloted simulator study done at NASA-Ames in May 1971 (reference 1).

A fundamental assumption was that the existing elevator control surface had sufficient control power to meet maneuver and trim

requirements throughout the flight envelope. This was in fact verified by flight test and therefore allowed use of the existing control surface.

Elevator hinge moment data was generated by combining the tab elevator effectiveness ratio (figure 4.2-1) obtained from Boeing flight test results with the tab hinge moment coefficient $C_{H_{\delta_t}}$ of the basic Buffalo. A linear coefficient was assumed for tab deflections below 10 degrees. Elevator hinge moment data for deflections greater than 15 degrees were estimated from stick force data of the spring tab control system with the tab stops contracted. Figure 4.2-2 shows the estimated elevator hinge moment coefficient with zero tab deflection as well as the coefficient for the proposed surface configuration.

The elevator per 'g' characteristic used in the calculations is shown in figure 4.2-3. This characteristic is basically unchanged from the existing configuration and therefore requires the same airplane gross weight - c.g. schedule to be maintained as with the manual elevator control system.

The piloted simulator study was used to evaluate handling quality improvements with a powered elevator as well as to determine basic system parameters such as elevator/column gearing, minimum elevator surface rates, SAS authority, trim rates. Major simulator results were as follows:

1. A 30 degree per second elevator rate is a minimum acceptable lower level.

2. At the approach condition a linear gradient with 30 pounds stick force at maximum column deflection results in good control feel. (This was later revised to 40 pounds at maximum deflection).

3. A flap trim interconnect is not necessary. (This was also confirmed by pilot opinion in a flight test of the airplane prior to the powered elevator modification).

4. An elevator trim rate of 2 degrees elevator per second represents an acceptable compromise between high speed and low speed trim rate requirements.

5. A control system gearing of -3.28 degrees elevator per inch of column deflection is satisfactory, although -4 was preferred. A gearing above -5 tended to cause PIOs, especially with flaps up.

6. A SAS authority of ±5 degrees elevator represents a satisfactory lower limit in the STOL flight regime.

4.3 POWERED ELEVATOR SYSTEM DESCRIPTION

Pitch control on the AWJSRA is by means of a single elevator control surface located on a fixed stabilizer. No load elevator position limits are +15 degrees, -25 degrees. The elevator surface (figure 4.3-1) includes a full time geared tab with a balance ratio $\delta_t / \delta_e = -0.7$, as well as a trim tab used during manual reversion.

The longitudinal control system schematic is shown in 4.3-2. Pilot and copilot signals are transmitted by a single cable run from dual control columns (figure 4.3-3) to the aft elevator control quadrant in the vertical stabilizer. The control columns and cable run are essentially unchanged from the basic C-8A with the exception of two bias springs connected to the control column torque tube to counter the column mass unbalance in the vertical direction. From the aft quadrant, the pilot signals are transmitted to a dual tandem, moving body power control unit directly connected to the elevator surface. With one or two hydraulic systems operable, the PCU is irreversible and positions the elevator in response to pilot commands. The control system becomes reversible following the loss of both hydraulic systems. Elevator control for this condition is through a slop link in parallel with the PCU.

The control linkage between the aft quadrant and the elevator PCU is non-redundant. It includes a limited authority, series SAS actuator originally designed by Moog for the F-4 and subsequently modified for the AWJSRA application. The actuator includes a mechanical locking device that centers and locks the actuator when hydraulic power is lost or shut off. The control linkage and actuator installations are shown in detail in figures 4.3-4 and 4.3-5. A picture of the actual installation is shown in figure 4.3-6.

The elevator power control unit is a modified Grumman Gulfstream II elevator/aileron actuator. The actuator was one of several existing actuators investigated during the preliminary design phase. New actuators were not considered due to cost and schedule limitations. The actuator is a dual tandem, moving-body type, as shown in figure 4.3-7. Actuator modifications consisted of a new flow control valve and a strengthened head end output rod.

The installation has several noteworthy features. The actuator is specifically installed in a horizontal position to sense only motion in the fore-aft direction. The actuator input is thereby effectively decoupled from vertical deflections of the elevator hinge line. These represent a spurious input and can cause instability, particularly where high structural feedback is used.

The actuator and its input linkage (links 1 and 2) are mounted in and hence referenced to the horizontal stabilizer, while pilot signals transmitted by bellcrank 4 are referenced to the rear spar of the vertical stabilizer. Relative motion between the horizontal and vertical stabilizers could therefore induce inputs to the actuator. The horizontal pushrod (link 3) was therefore included to decouple the actuator from relative motion in the vertical plane. Structural feedback for actuator stability is obtained by mounting the input lever 2 on the vertical swing link 5. Deflections of the actuator support structure are thereby fed back to the valve in a stabilizing sense.

Pilot feel during powered operation is provided by a variable gradient feel system. The feel system comprises a feel computer, figure 4.3-8 and a feel control unit, figure 4.3-9. Both units are modified 727 components. The feel computer generates hydraulic pressure proportional to impact pressure as sensed by a pitot probe in the leading edge of the vertical stabilizer. The control unit utilizes the hydraulic feel pressure in conjunction with a fixed mechanical spring gradient to produce the artificial feel force. The hydraulic feel force generation is completely dualized. A differential pressure switch in the feel computer monitors the two feel pressure computations and generates a warning signal if a significant difference exists.

The feel system is connected to the aft elevator control quadrant via a preloaded spring push-pull rod (pogo). This pogo allows feel system overtravel during an out-of-trim condition, as well as permitting elevator control with a jammed feel system.

During manual reversion, pilot stick force is the sum of the aerodynamic forces, actuator and control system friction and damping forces, and the feel system mechanical gradient.

Pilot trim during powered operation is accomplished by shifting the neutral point of the feel system, and hence the control system. Trim authority is +9, -14 degrees elevator with a trim rate of 2 degrees/second. Trim authority is fixed for all flight conditions. Trim control is electrical only. Powered trim indication is via the elevator surface position indicator.

Trim control during manual reversion is via a tab on the elevator surface. The trim tab control system is shown in figure 4.3-10. Pilot control of the trim tab is mechanical only. A locking and indexing feature on the trim wheel prevents inadvertent mistrim during powered operation. Positive action is required by the pilot to unlock the trim wheel and allow its use for manual reversion trim.

Hydraulic power for the elevator control system is supplied by hydraulic systems 'A' and 'B'. The elevator PCU and feel system are supplied by both 'A' and 'B' systems; the SAS servo by system 'A' only. Supply pressure to the elevator PCU is reduced to 2440 psi. A relief valve and a low pressure warning switch provide protection against overpressure and underpressure. The hydraulic system schematic is shown in figure 4.3-11.

4.4 PREDICTED LONGITUDINAL CONTROL CHARACTERISTICS

Elevator trim requirements for 1 'g' flight are shown in figure 4.4-1. No trim change due to the powered elevator modification was predicted. Available elevator deflection during powered operation is shown on figure 4.4-2. Elevator deflection is limited at higher airspeeds by actuator blowdown. With both hydraulic systems operating, blow down from -25 degrees elevator starts at 100 knots and from +15 degrees at 118 knots. The 100 knot blow down speed allows maximum elevator deflection at the STOL approach speed with on hydraulic system lost as well as for dual hydraulics with loss of the geared tab.

Feel force characteristics, excluding friction, are shown in figure 4.4-3. The feel system provides a nearly constant 42 lbs/'g' for all flight conditions. The feel gradient is reduced by the pogo between the elevator aft quadrant and the feel control unit for force levels greater than 37 pounds column force.

The elevator power control unit has a no-load time constant of .04 second and a maximum no-load rate of 55 degrees/second. Both of these parameters are load sensitive and hence will vary with elevator position and airspeed, as shown on figure 4.4-4 and figure 4.4-5. Load effects at all trim conditions are small enough to ensure satisfactory PCU performance.

The control system is designed for a maximum SAS authority of $\pm$ 8 degree elevator deflection. This SAS authority is a varying

percentage of pilot authority. The worst case occurs with trim at the maximum aircraft nose down trim limit (+9 degrees elevator), where +8 degrees SAS authority is equivalent to 104 percent pilot authority in the aircraft nose down direction (figure 4.4-6). A SAS hardover at this condition in the aircraft nose up direction will require SAS disconnect rather than override.

The power control unit input lever travel capability is limited to $\pm$ 2 degrees elevator deflection to minimize the amount of lost motion in the control system during the manual reversion mode. This, however, produces an undesirable effect during SAS operation, where the SAS actuator can backdrive the column $\pm$ 3.5°δ_e if the pilot and the SAS both command maximum elevator in the same direction. This can be avoided during normal operation by automatically reducing the SAS command to zero as the pilot input approaches the maximum.

Trim control is electrical only, with no redundancy or mechanical backup. Trim runaways are a possibility, therefore, and must be countered by a corresponding pilot force. The force required to maintain airplane trim due to a maximum trim runaway at any one flight condition is shown on figure 4.4-7. Also shown is the force required for maximum elevator deflection with the most adverse trim runaway.

Manual reversion is automatically provided following the loss of both hydraulic systems. Airplane controllability during manual reversion covers three stages:

1. Hydraulic power on/off transient
2. Trim control
3. Maneuver control

Following the loss of two hydraulic systems, the elevator surface will automatically float to -4 degrees. If the airplane is not at a flight condition that requires -4 degrees trim, then a maneuver will result. The magnitude of the maneuver is dependent on the

difference between elevator float point and elevator trim prior to hydraulic power loss. The load factor transient for various conditions is shown on figure 4.4-8 while pilot force required to prevent this transient is shown on figure 4.4-9. Because of the magnitude of the possible transients, it is important that the pilot divert to 90 knots, flaps 30, as soon as possible if complete loss of hydraulic power can occur with one additional failure.

Trim for hands-off flight during manual reversion is possible through the existing trim tab. Trim control is only mechanical through the trim wheel. Authority is limited to ± 10.5 degrees elevator.

Maneuver control forces are considerably higher than in the powered mode. Aerodynamic loads are shown on figure 4.4-10. Control system friction, damping and the feel system centering spring will further increase the manual reversion loads.

The powered elevator system was analyzed to assure a flutter free design. In addition to the normal operating condition, certain failure cases were also investigated. These were

1. One and two hydraulic system failures
2. Geared balance tab rod failure

The analysis showed the elevator system to be flutter-free for all conditions investigated.

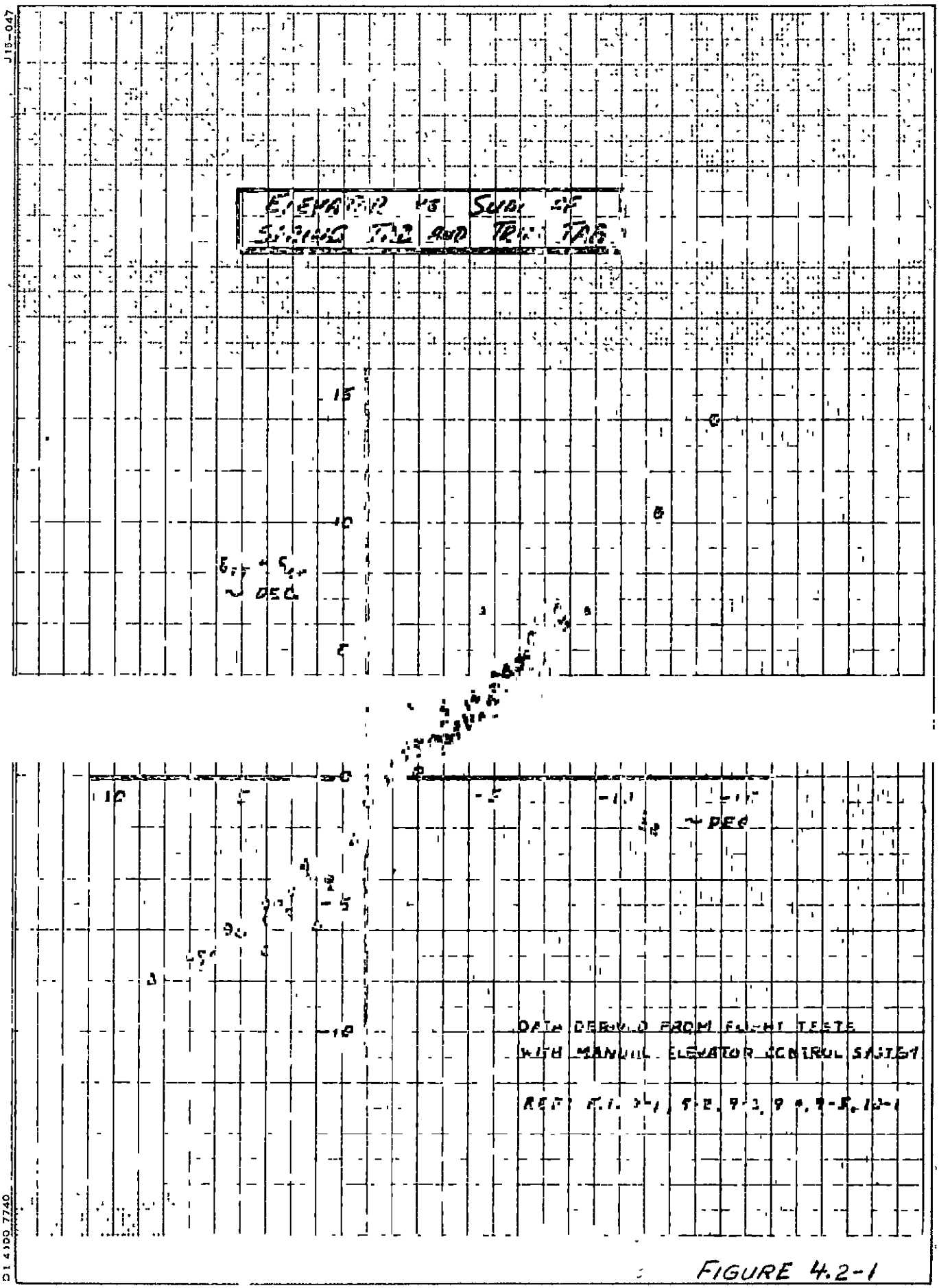

ELEVATOR VS SUM OF
SPRING TAB AND TRIM TAB
15
10
5
0
-5
-10
~ DEG
DATA DERIVED FROM FLIGHT TESTS
WITH MANUAL ELEVATOR CONTROL SYSTEM

FIGURE 4.2-1

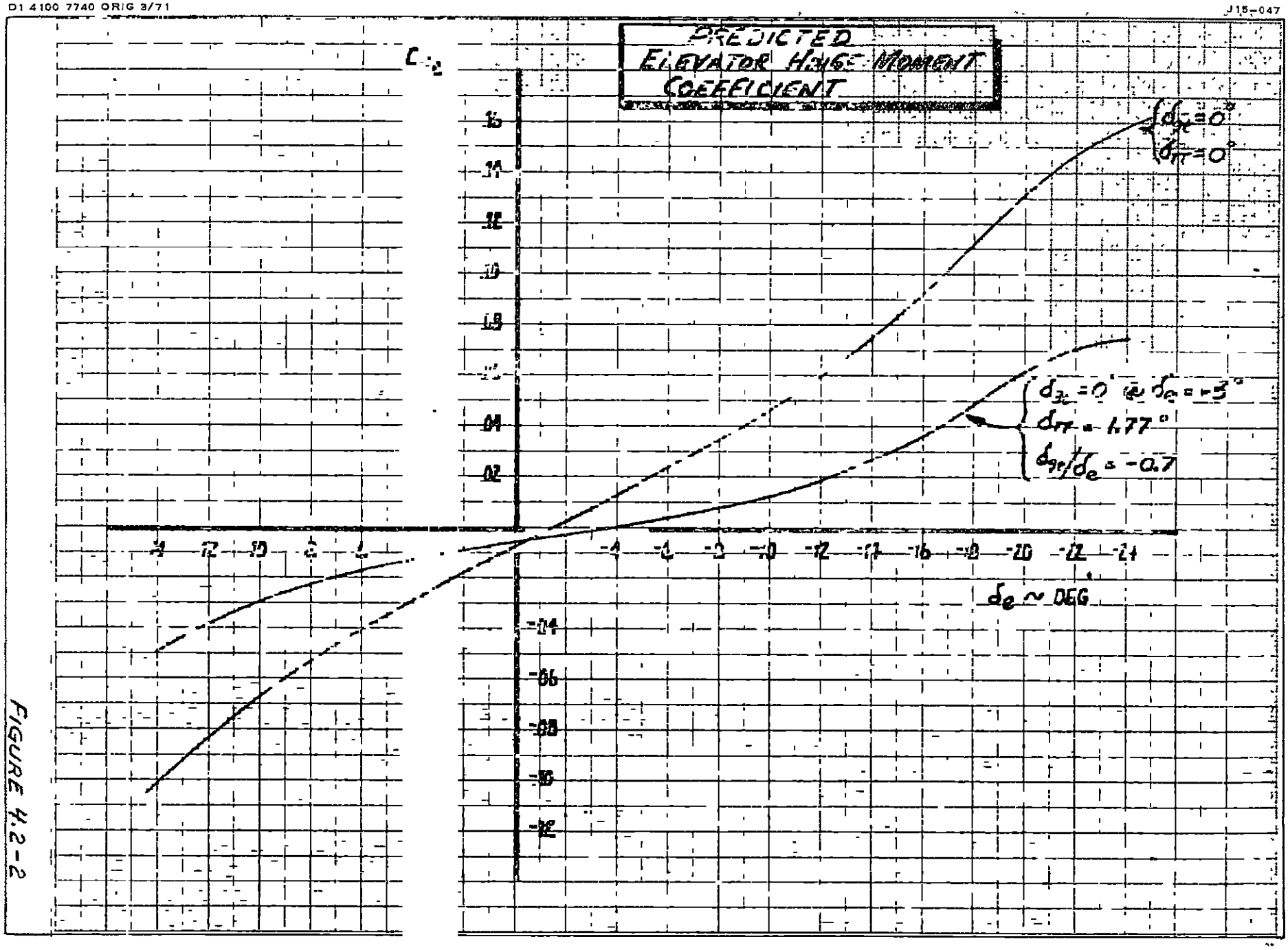

PREDICTED
ELEVATOR HINGE MOMENT
COEFFICIENT
C_{h_e}
$\delta_{3c} = 0°$
$\delta_{TT} = 0°$
$\delta_{3c} = 0°$ @ $\delta_e = -3°$
$\delta_{TT} = 1.77°$
$\delta_{3c}/\delta_e = -0.7$
$\delta_e \sim$ DEG

FIGURE 4.2-2

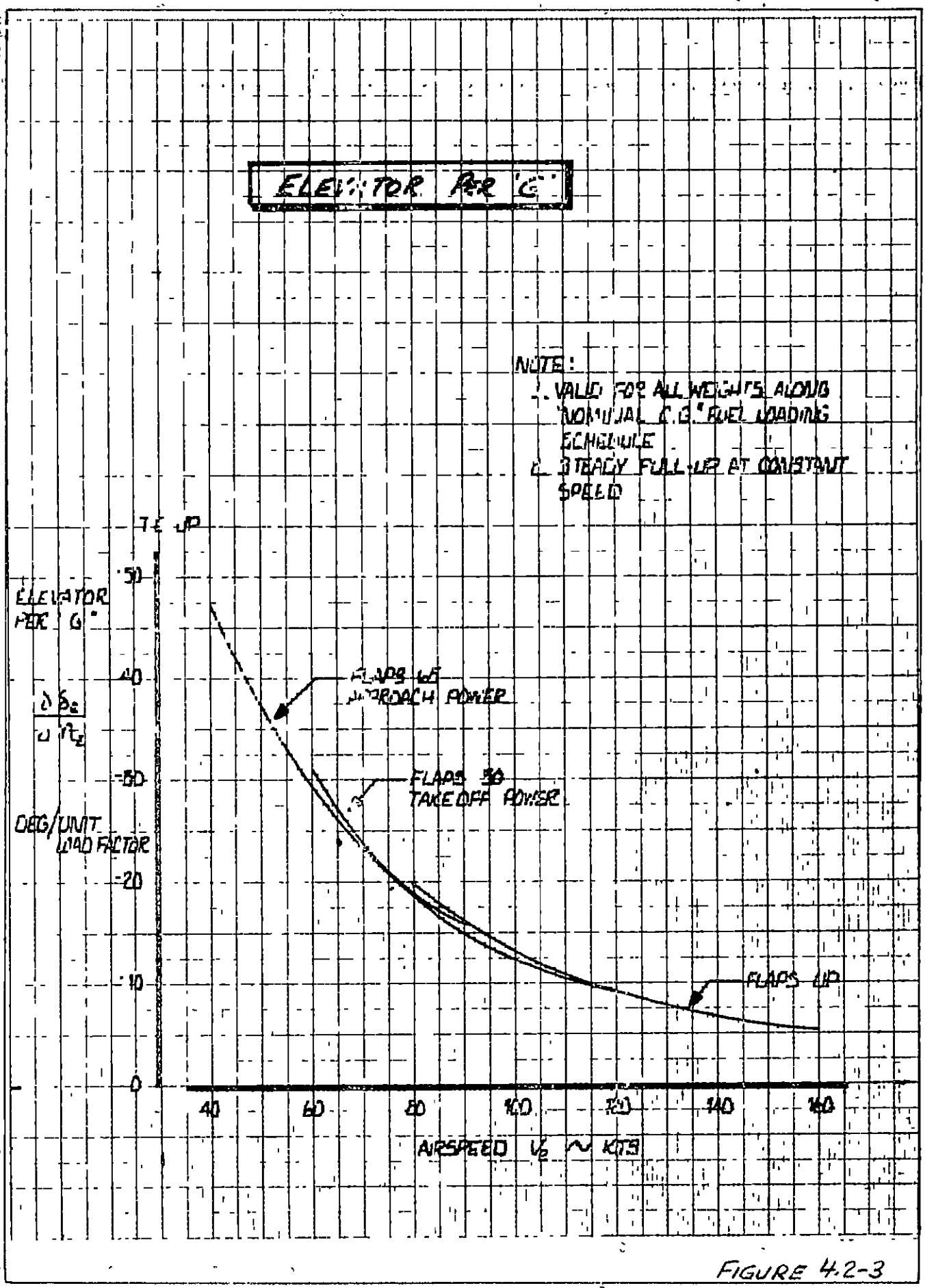

ELEVATOR PER 'G'
NOTE:
1. VALID FOR ALL WEIGHTS ALONG NOMINAL C.G. FUEL LOADING SCHEDULE
2. STEADY PULL-UP AT CONSTANT SPEED
TO UP
ELEVATOR PER G
∂δe / ∂nz
DEG/UNIT LOAD FACTOR
50
40
30
20
10
0
FLAPS 40 APPROACH POWER
FLAPS 30 TAKE OFF POWER
FLAPS UP
40
60
80
100
120
140
160
AIRSPEED Ve ~ KTS

FIGURE 4.2-3

POWERED ELEVATOR SURFACE CONFIGURATION

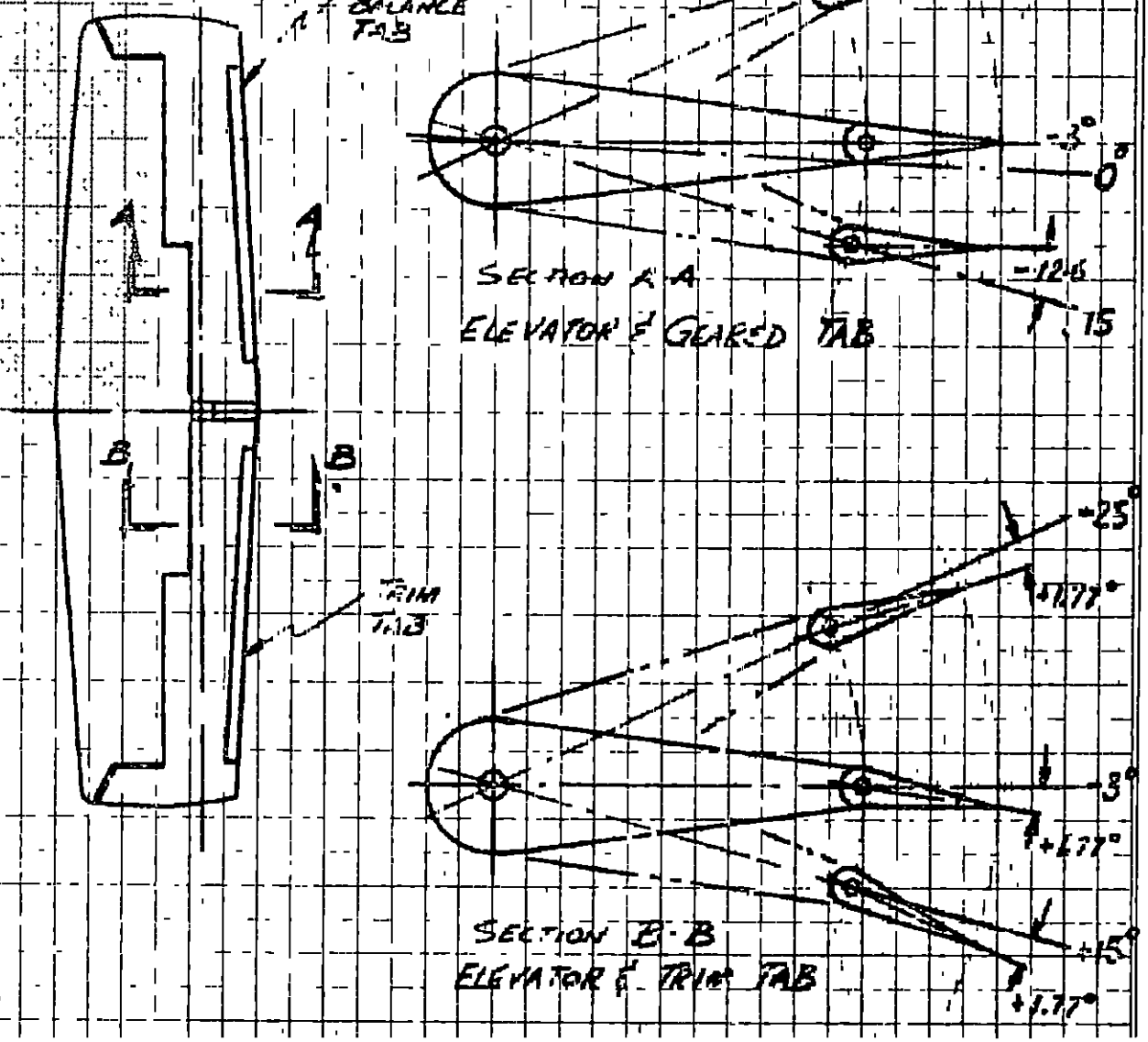


FIGURE 4.3-1

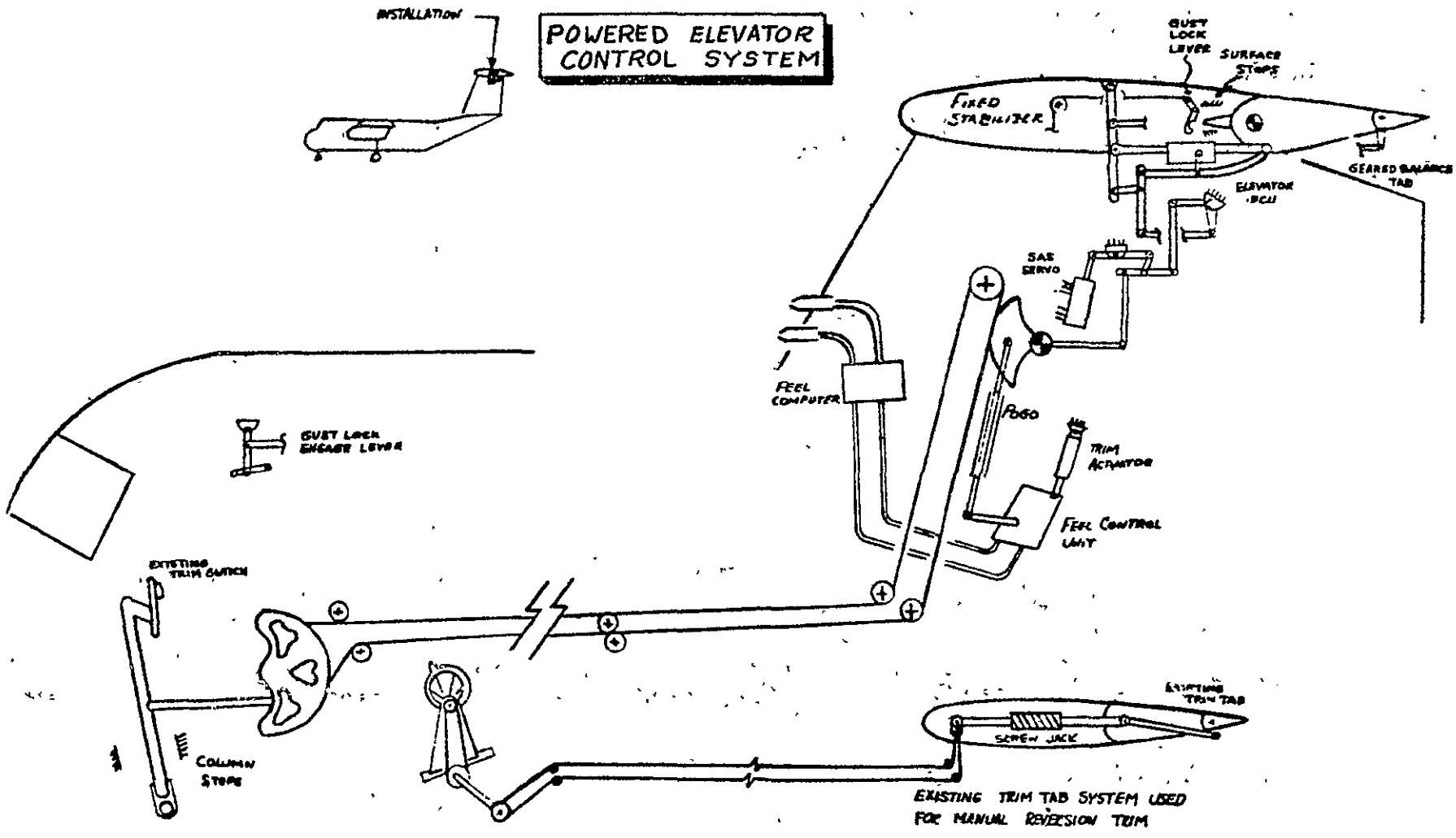

INSTALLATION
POWERED ELEVATOR CONTROL SYSTEM
GUST LOCK LEVER
SURFACE STOPS
FIXED STABILIZER
GEARED BALANCE TAB
ELEVATOR (PCU)
SAS SERVO
FEEL COMPUTER
POGO
TRIM ACTUATOR
FEEL CONTROL UNIT
GUST LOCK ENGAGE LEVER
EXISTING TRIM CLUTCH
COLUMN STOPS
EXISTING TRIM TAB
SCREW JACK
EXISTING TRIM TAB SYSTEM USED FOR MANUAL REVERSION TRIM

FIGURE 4.3-2.

CONTROL COLUMN INSTALLATION

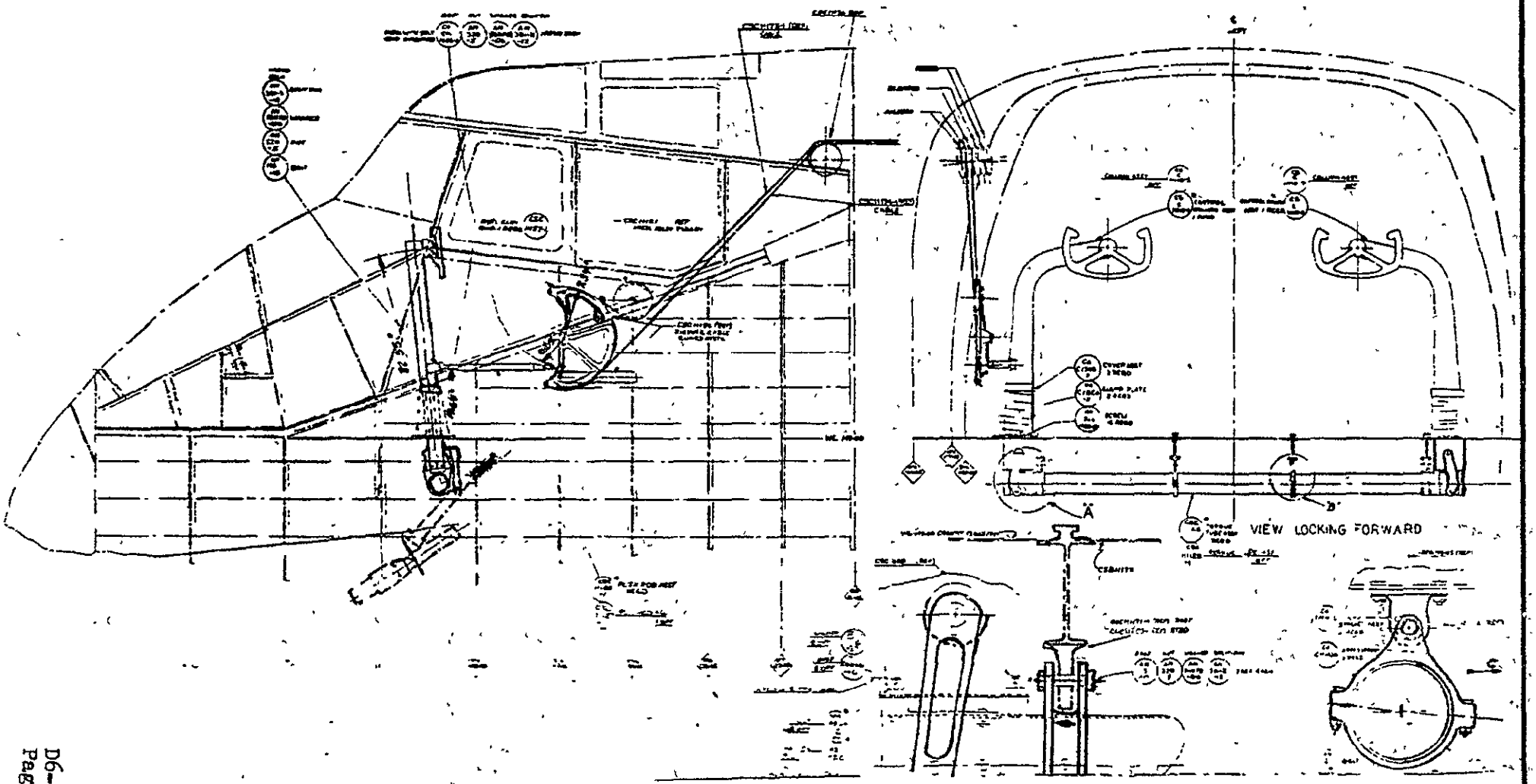
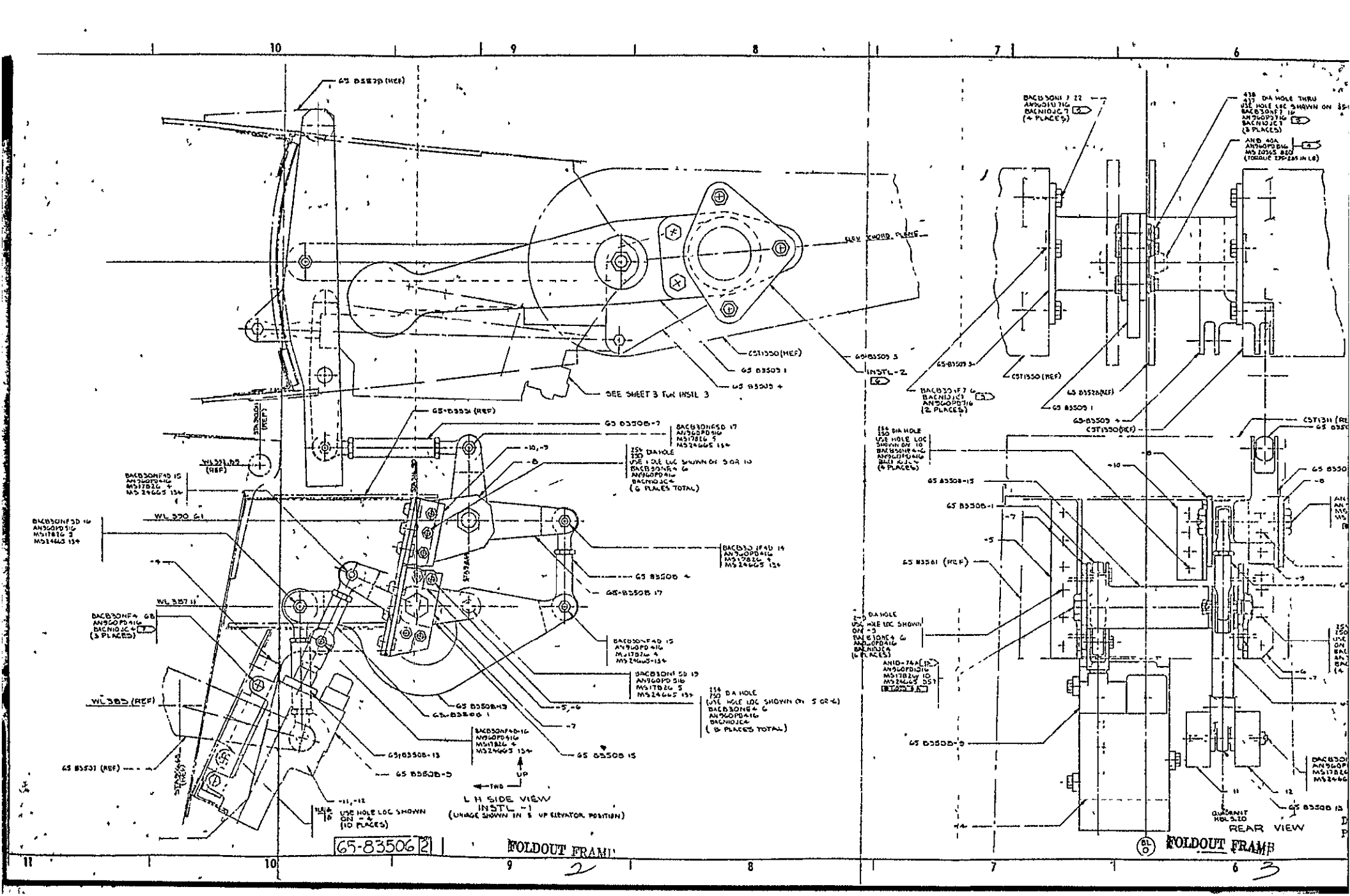
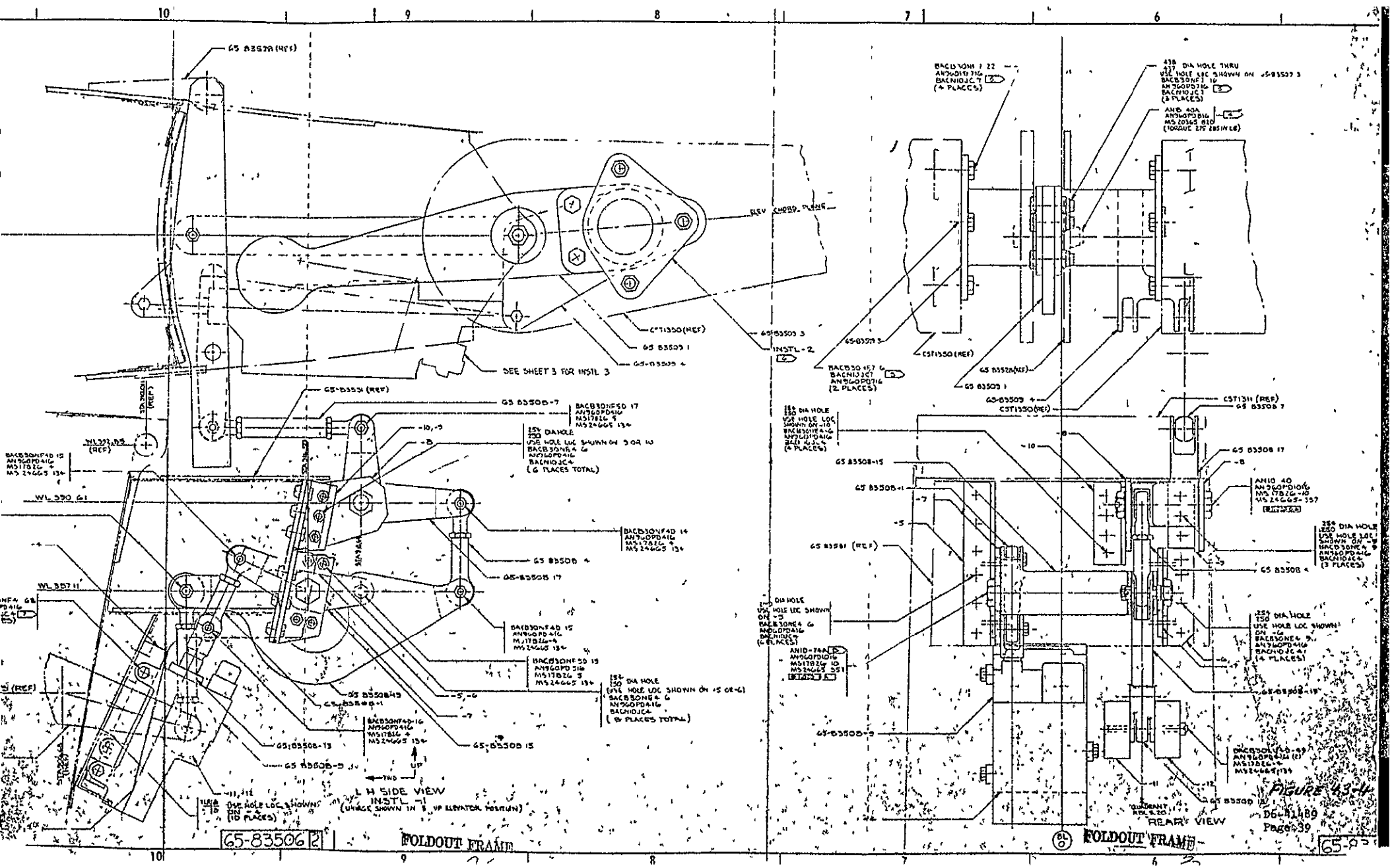


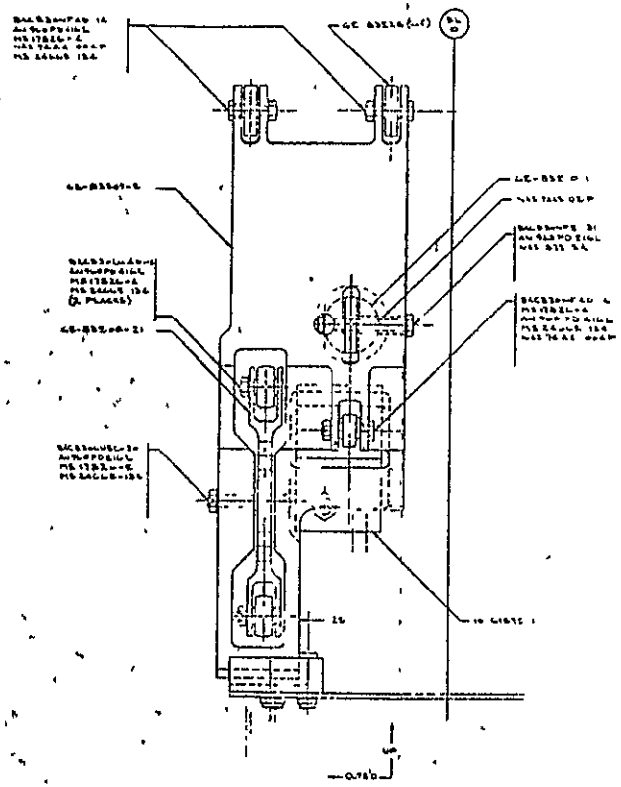
FIGURE 4.3-3


65-83506 2
FOLDOUT FRAME 2
FOLDOUT FRAME 3
L H SIDE VIEW
INSTL - 1
REAR VIEW
INSTL - 2
SEE SHEET 3 FOR INSTL 3

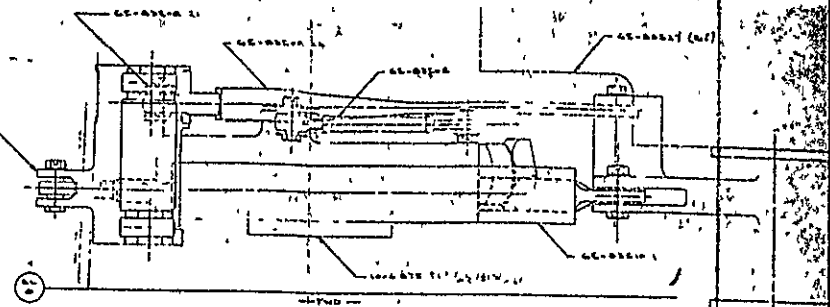


L H SIDE VIEW
INSTL -1
(UNIAGE SHOWN IN 3 UP ELEVATOR POSITION)
REAR VIEW
SEE SHEET 3 FOR INSTL 3
ELEV CHORD PLANE
65-83506 2
FOLDOUT FRAME
FOLDOUT FRAME

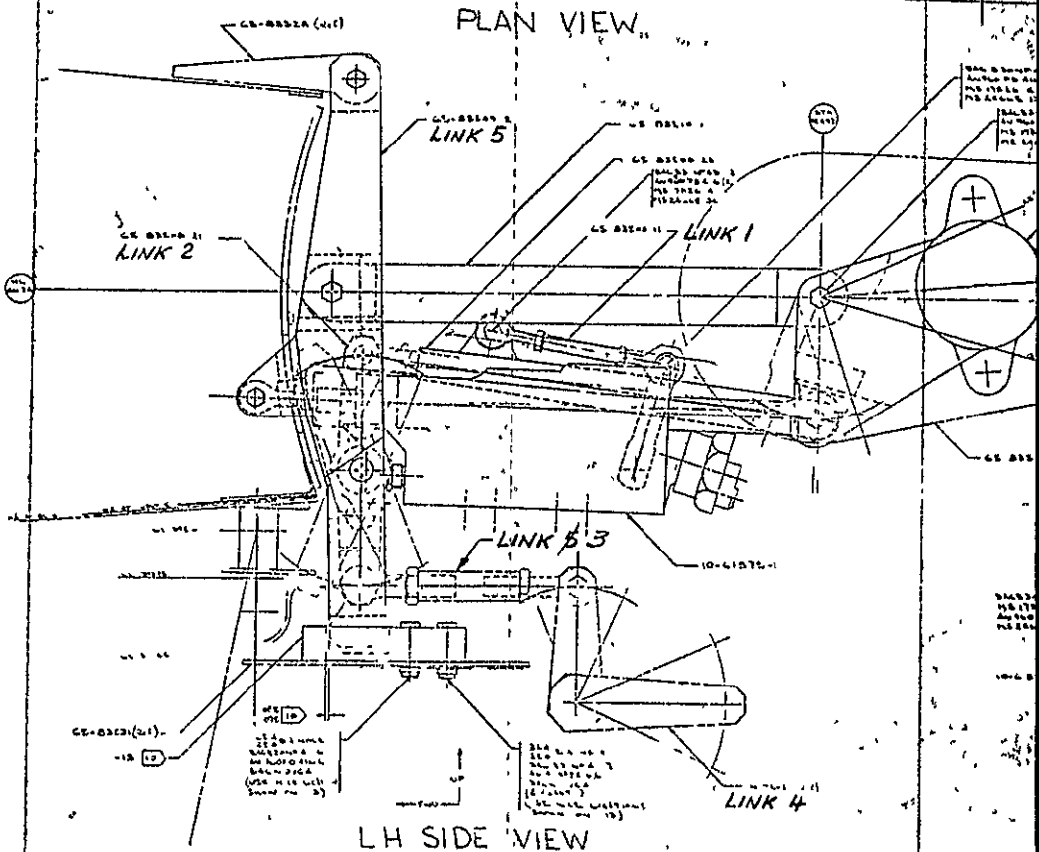
FIGURE 43-4



FRONT VIEW



PLAN VIEW



LH SIDE VIEW

INSTL -3

FOLDOUT FRAME 8 7 6

FOLDOUT FRAME 5 4

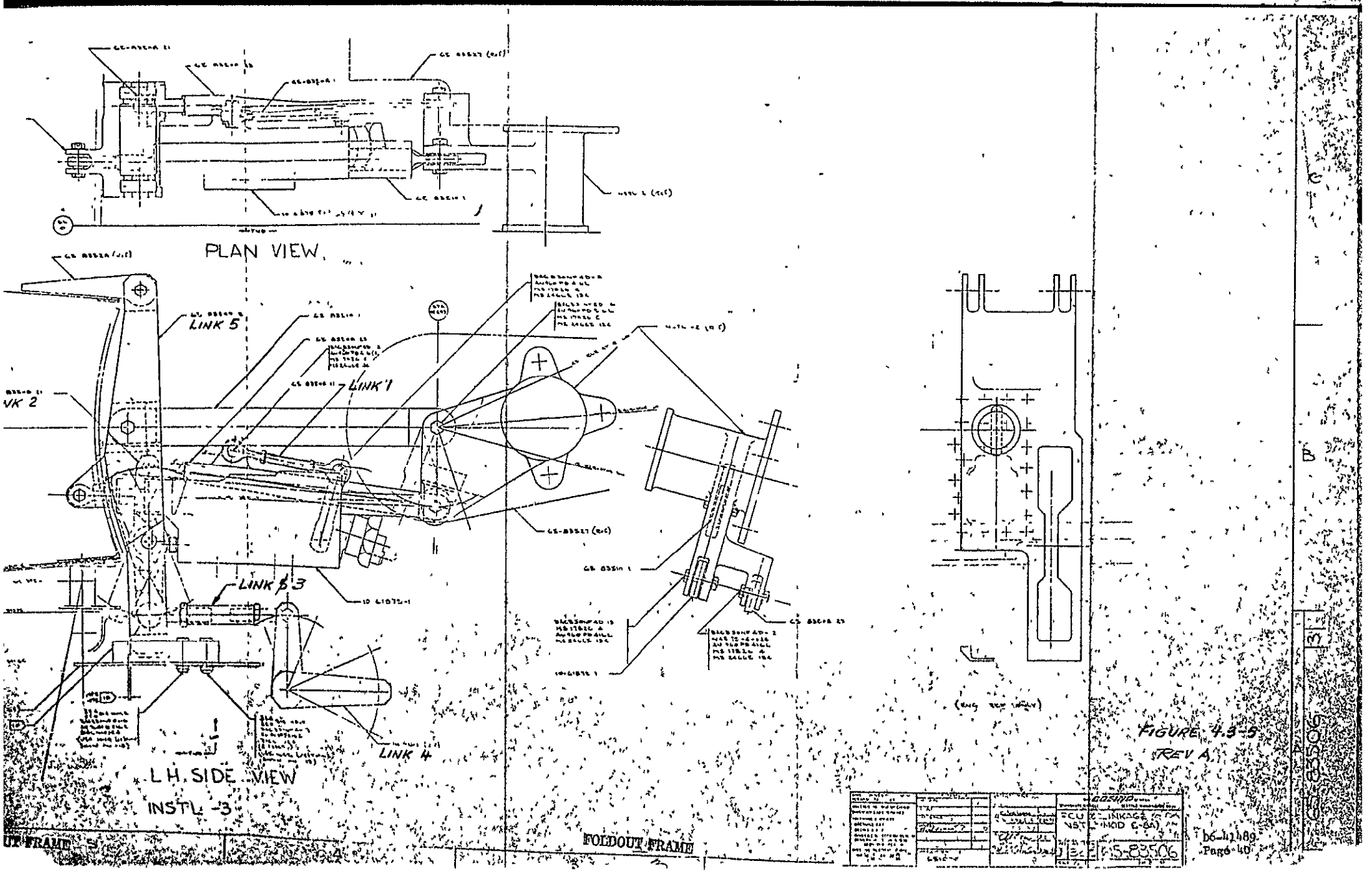
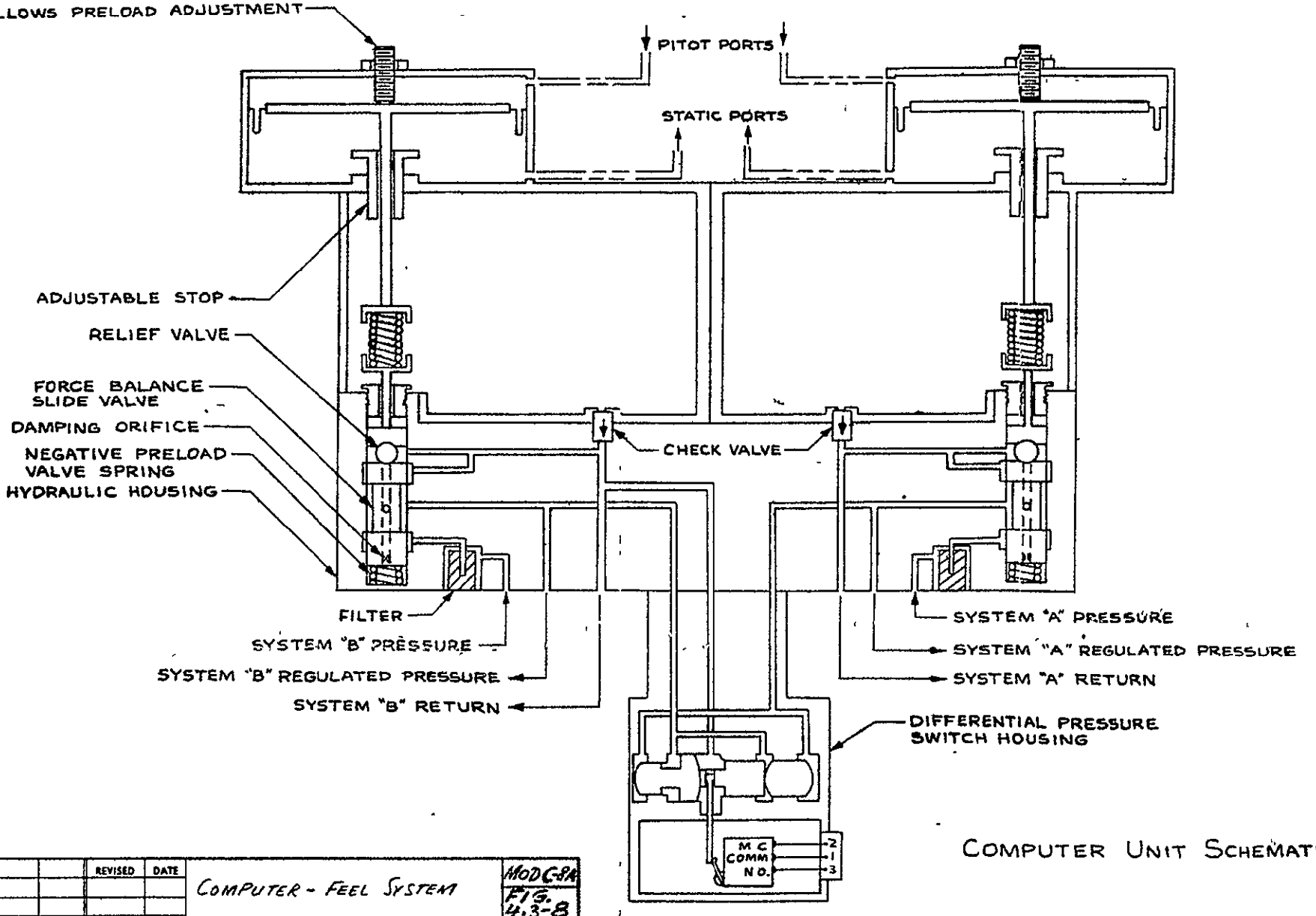

PLAN VIEW
LINK 5
LINK 1
LINK 2
LINK 3
LINK 4
L.H. SIDE VIEW
INSTL -3
FOLDOUT FRAME
FIGURE 4.3-5
REV A
65-83506
b6-41489
Page 40

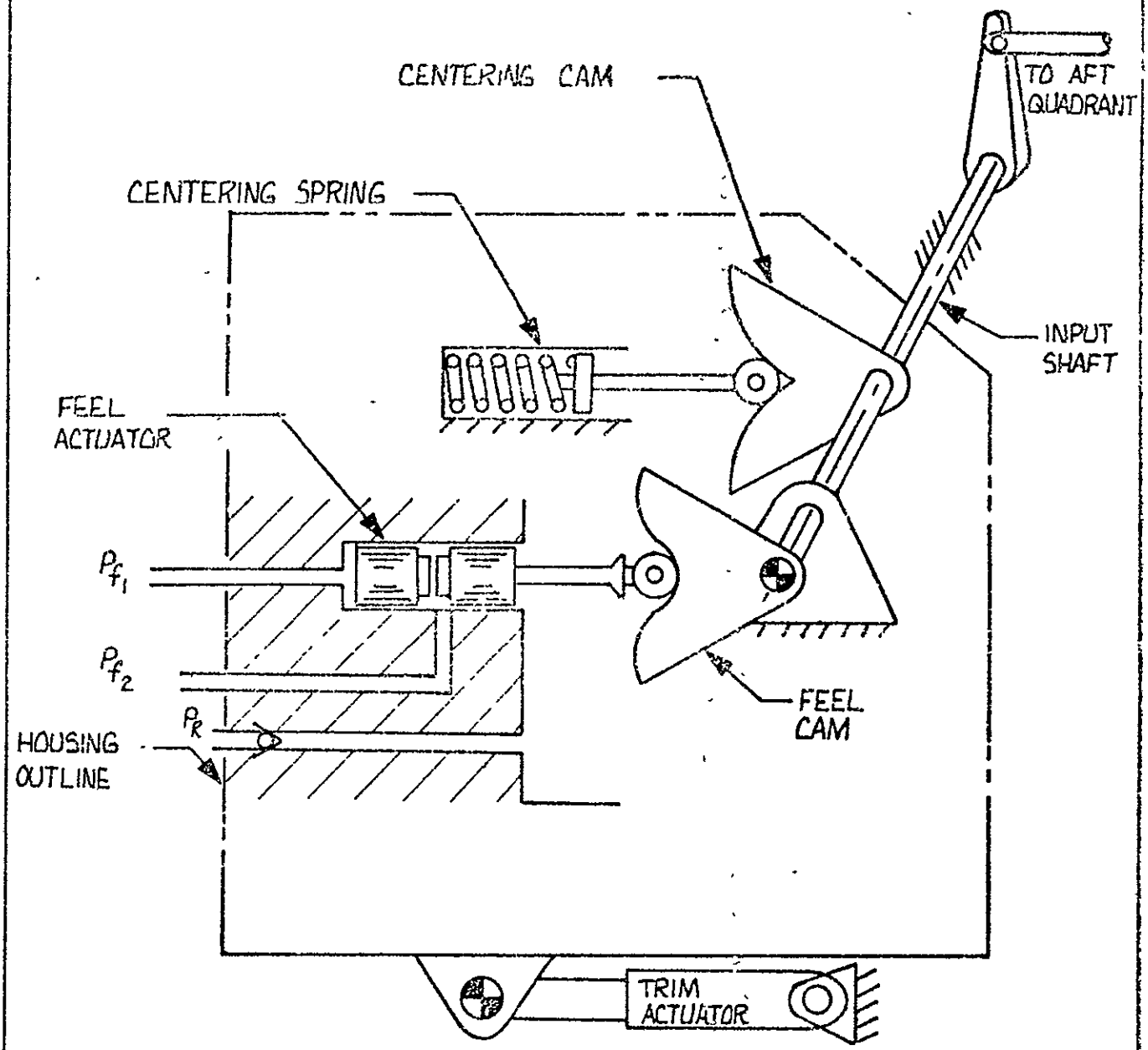
FIGURE 4.3-6
D6-11489
Page 41



COMPUTER UNIT SCHEMATIC

CALC.			REVISED	DATE	COMPUTER - FEEL SYSTEM	MOD C-8R FIG. 4.3-8
CHECK						
APPD						
APPD					THE BOEING COMPANY RENTON WASH	PAGE 43

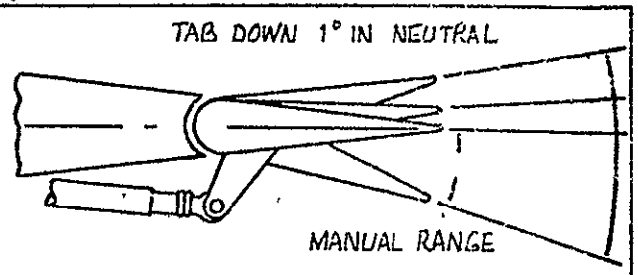
D1 6103 8180 ONIC 3/71



FEEL CONTROL UNIT

FIG. 4.3-9

ELEVATOR TRIM TAB SYSTEM



NOMINAL TAB RANGES OF MOVEMENT

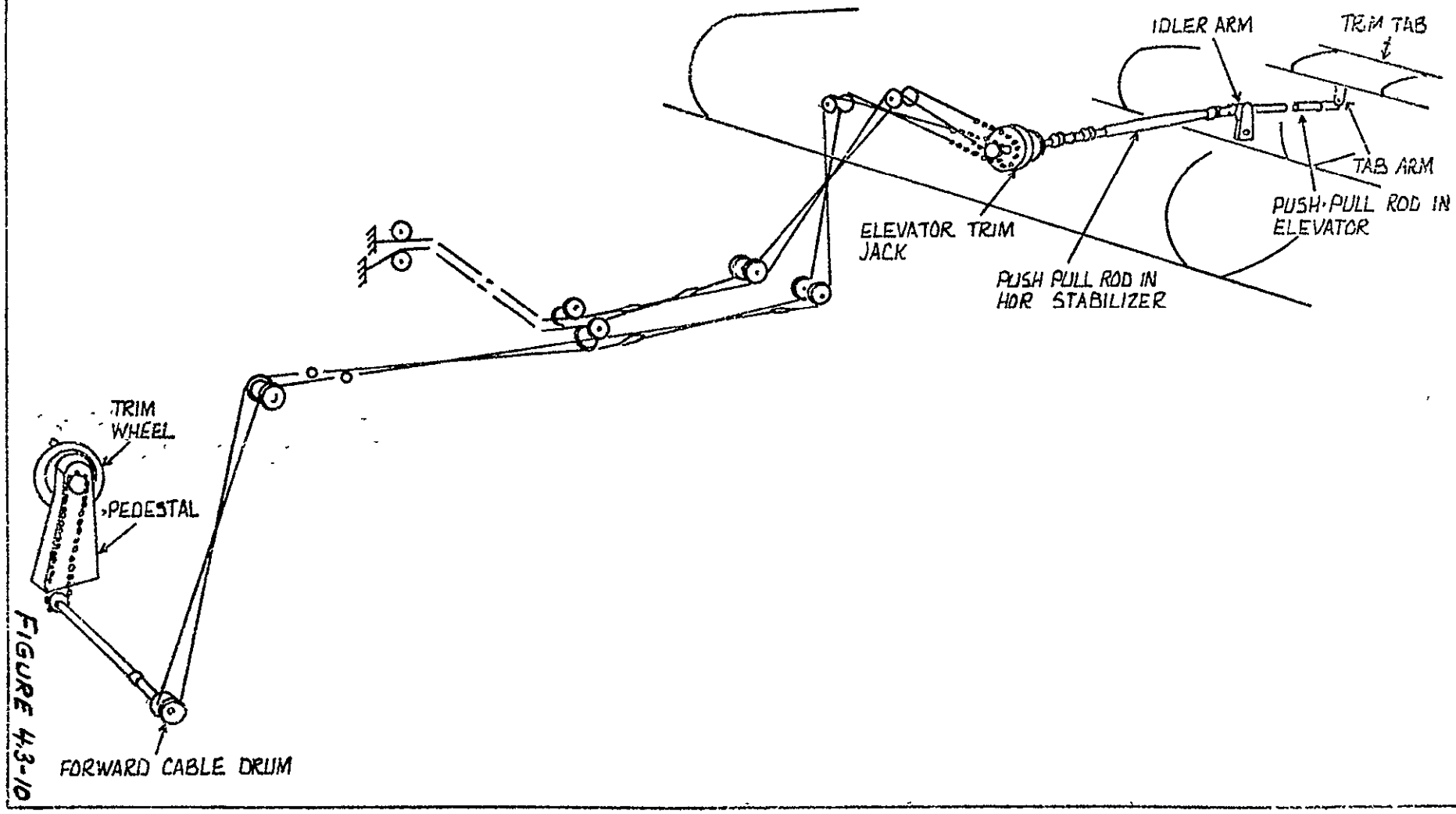
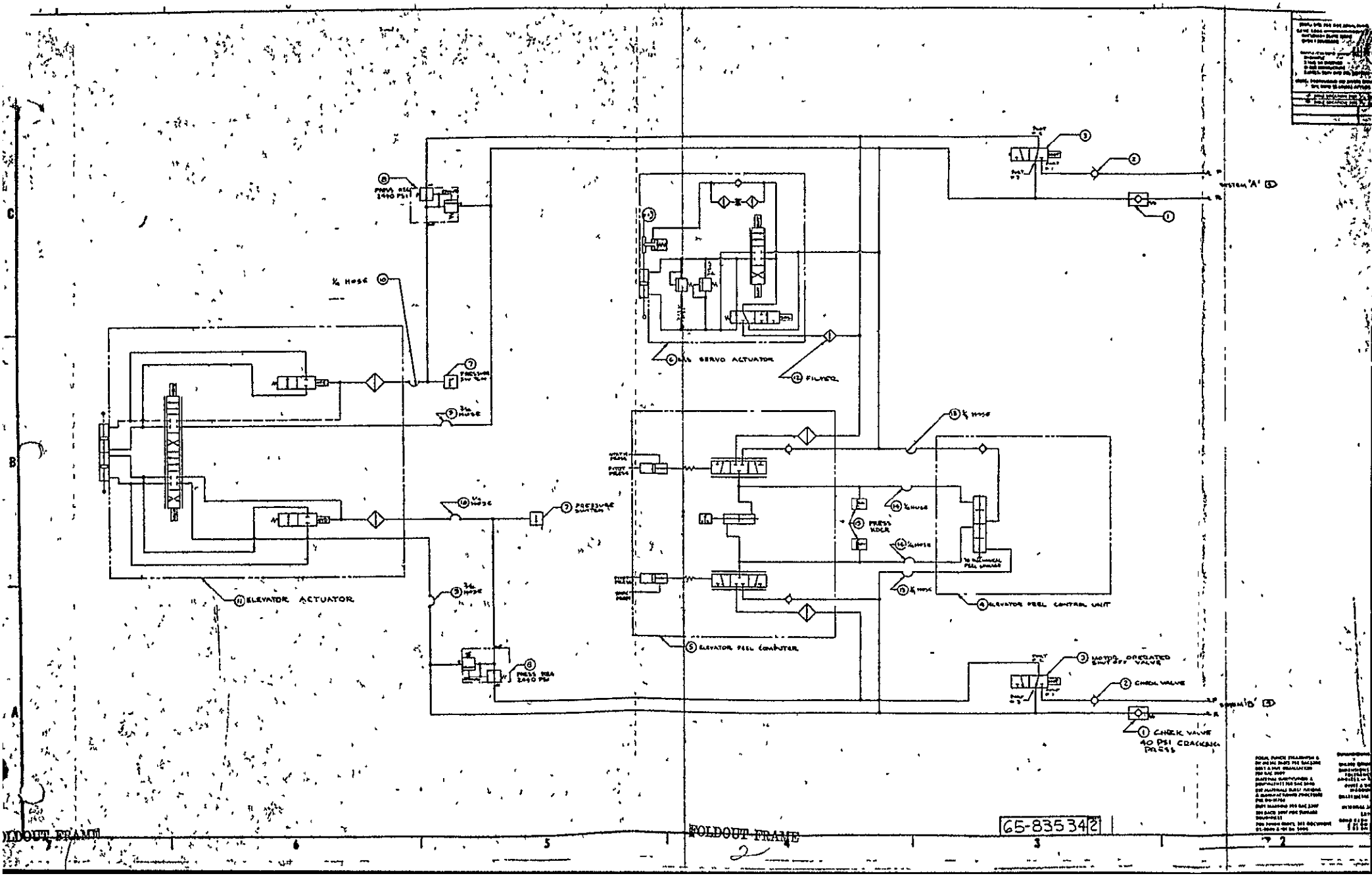


FIGURE 43-10


SYSTEM 'A'
SYSTEM 'B'
ELEVATOR ACTUATOR
SERVO ACTUATOR
FILTER
ELEVATOR FEEL COMPUTER
ELEVATOR FEEL CONTROL UNIT
CHECK VALVE
40 PSI CRACKING PRESS
FOLDOUT FRAME
FOLDOUT FRAME
65-83534 2

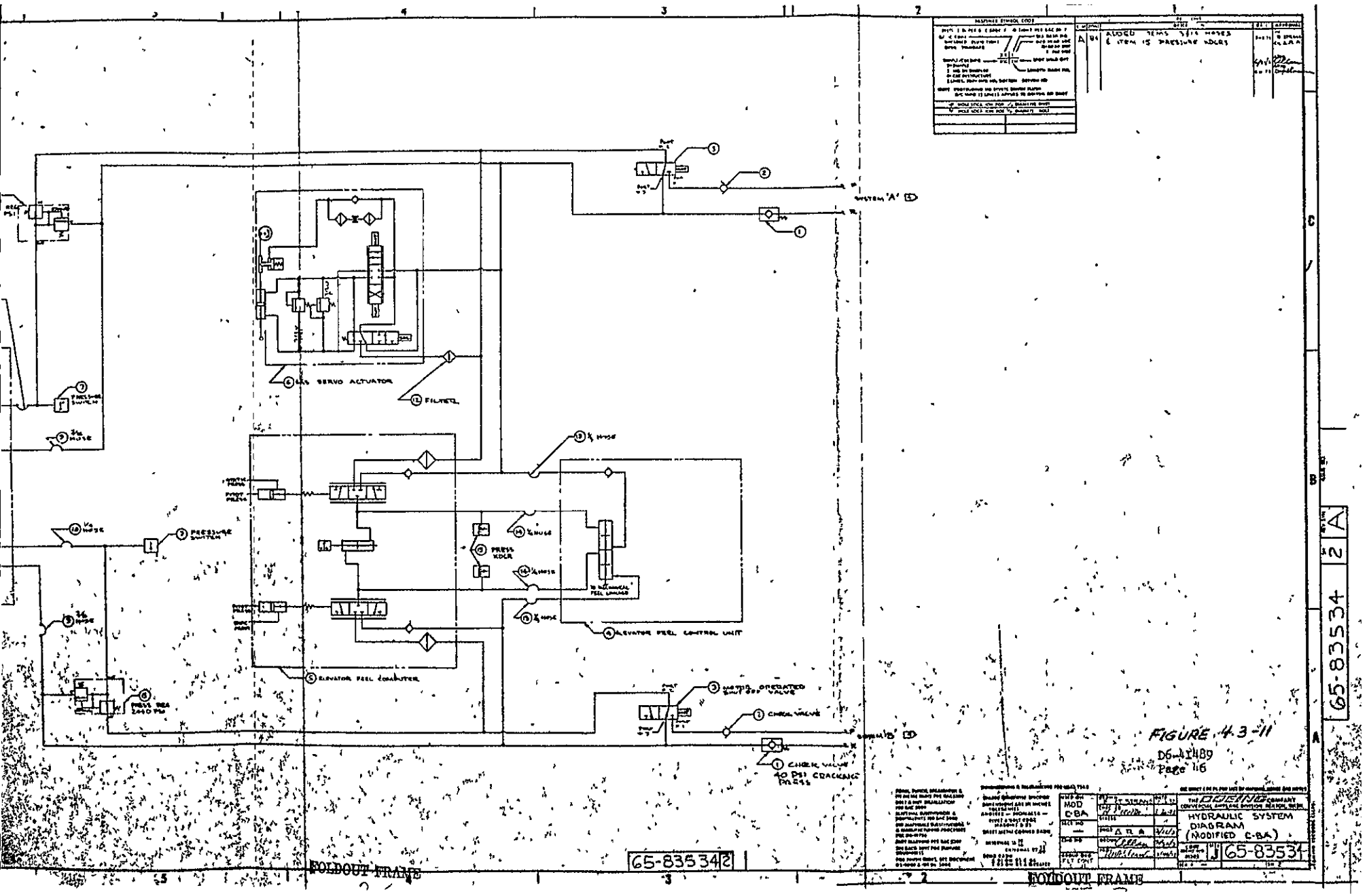


FIGURE 4.3-11
D6-41489
Page 116

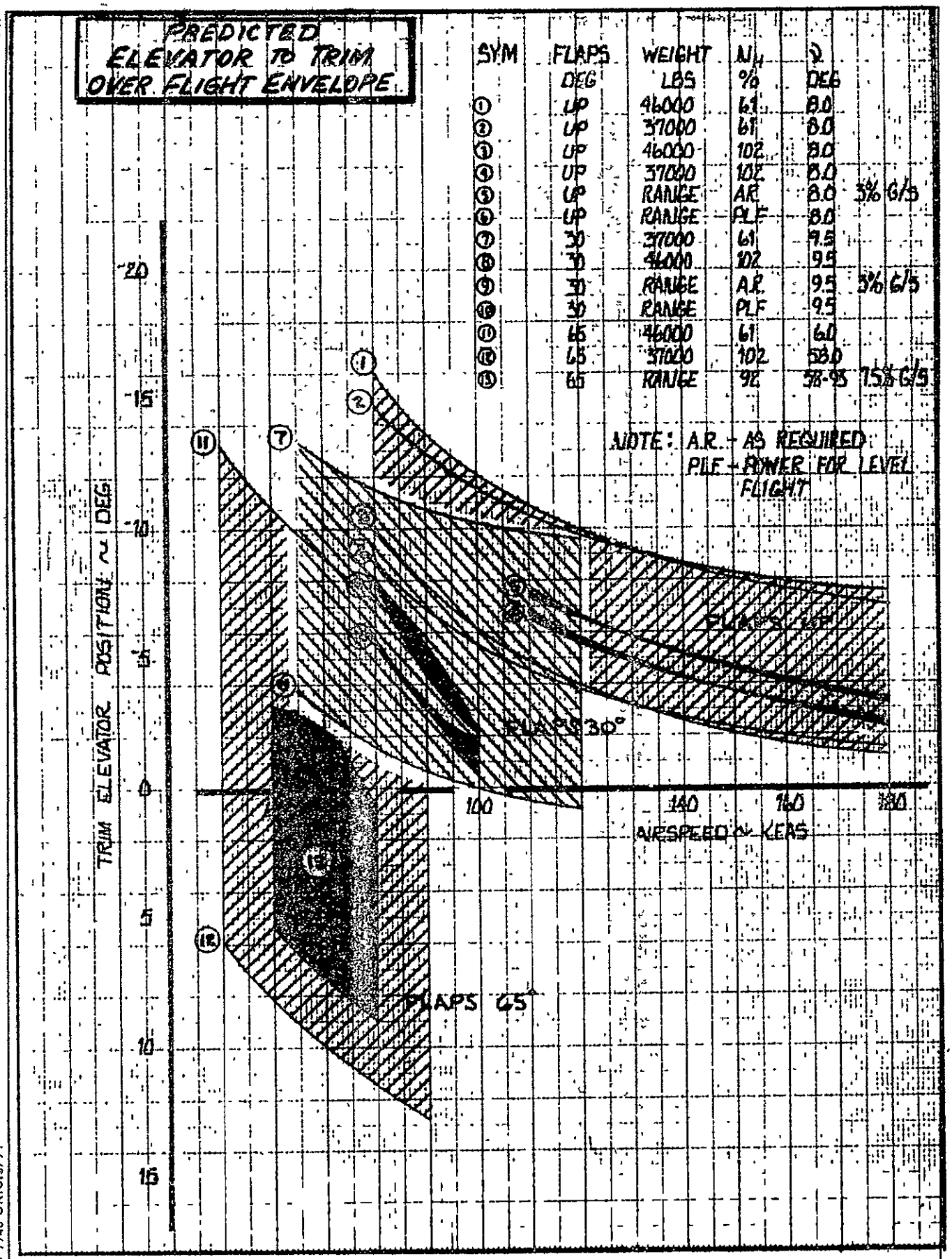

PREDICTED
ELEVATOR TO TRIM
OVER FLIGHT ENVELOPE
SYM
FLAPS DEG
WEIGHT LBS
N_1 %
Ω DEG
① UP 46000 61 8.0
② UP 37000 61 8.0
③ UP 46000 102 8.0
④ UP 37000 102 8.0
⑤ UP RANGE A.R. 8.0 3% G/S
⑥ UP RANGE PLF 8.0
⑦ 30 37000 61 9.5
⑧ 30 46000 102 9.5
⑨ 30 RANGE A.R. 9.5 3% G/S
⑩ 30 RANGE PLF 9.5
⑪ 45 46000 61 6.0
⑫ 45 37000 102 58.0
⑬ 45 RANGE 92 58-95 7.5% G/S
NOTE: A.R. - AS REQUIRED
PLF - POWER FOR LEVEL FLIGHT
FLAPS 30°
FLAPS 45°
TRIM ELEVATOR POSITION ∿ DEG
-20
-15
-10
-5
0
5
10
15
100
140
160
180
AIRSPEED ∿ KEAS

FIGURE 4.4-1

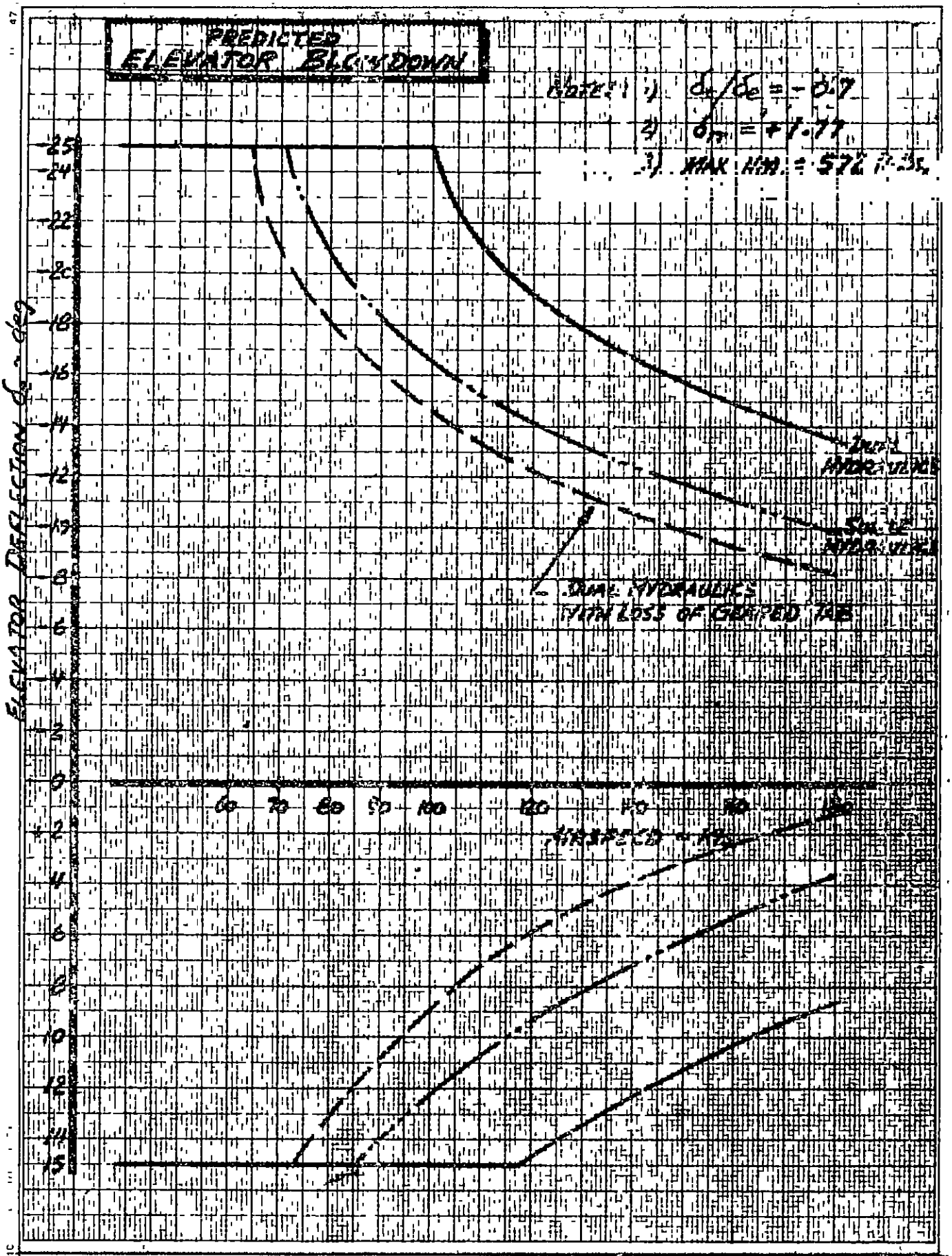

PREDICTED
ELEVATOR BLOWDOWN
NOTE: 1) $\delta_t/\delta_e = -0.7$
2) $\delta_{tr} = +1.77$
3) MAX MACH = .572
ELEVATOR DEFLECTION δ_e, deg
DUAL HYDRAULICS
SINGLE HYDRAULICS
DUAL HYDRAULICS
WITH LOSS OF GEARED TAB
AIRSPEED, KEAS

FIGURE 4.4-2

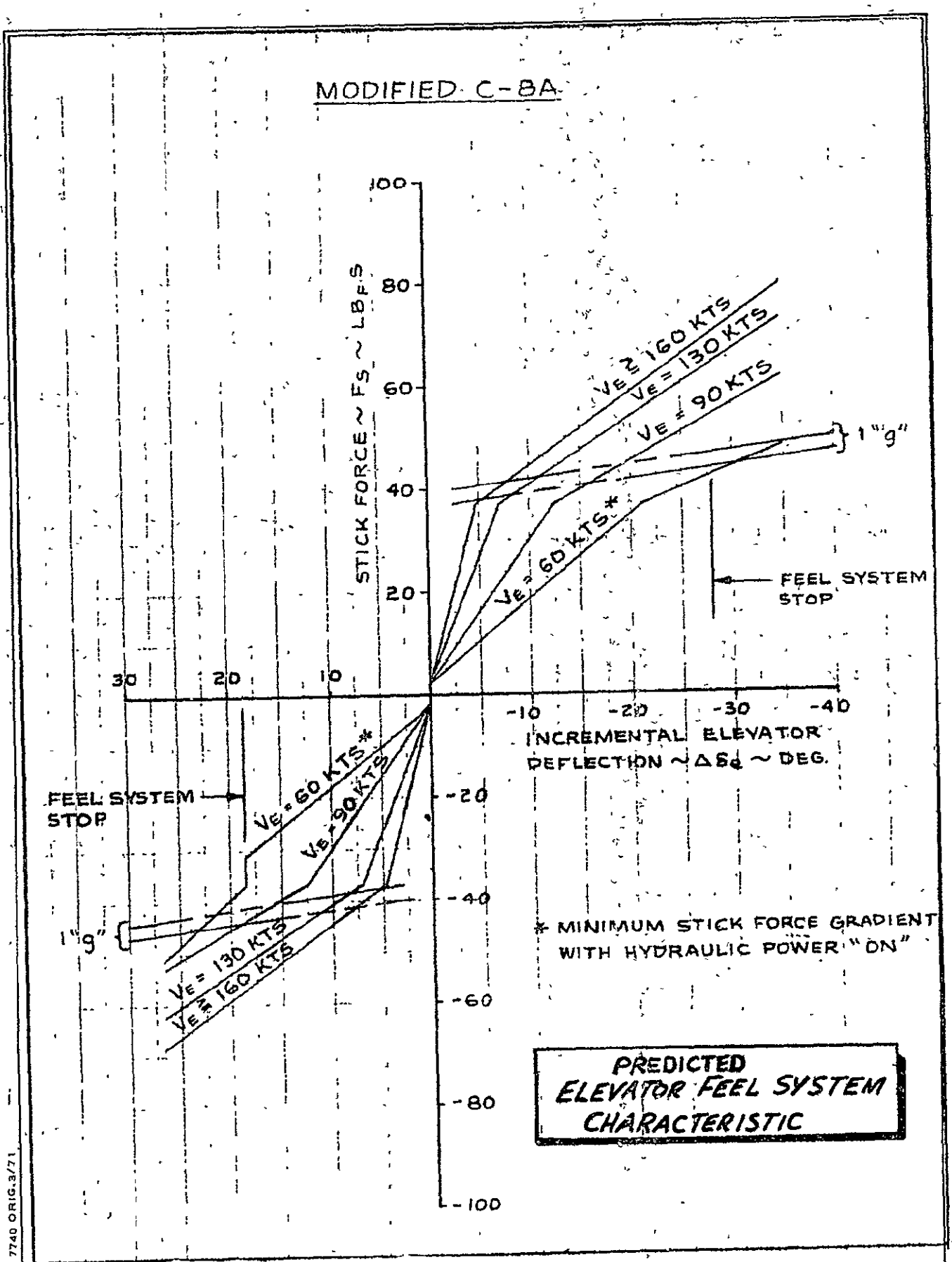

MODIFIED C-8A
STICK FORCE ~ Fs ~ LBFS
100
80
60
40
20
-20
-40
-60
-80
-100
30
20
10
-10
-20
-30
-40
INCREMENTAL ELEVATOR DEFLECTION ~ Δδe ~ DEG.
VE = 160 KTS
VE = 130 KTS
VE = 90 KTS
VE = 60 KTS*
1 "g"
FEEL SYSTEM STOP
FEEL SYSTEM STOP
VE = 60 KTS*
VE = 90 KTS
VE = 130 KTS
VE = 160 KTS
1 "g"
* MINIMUM STICK FORCE GRADIENT WITH HYDRAULIC POWER "ON"
PREDICTED ELEVATOR FEEL SYSTEM CHARACTERISTIC

FIGURE 4.4-3

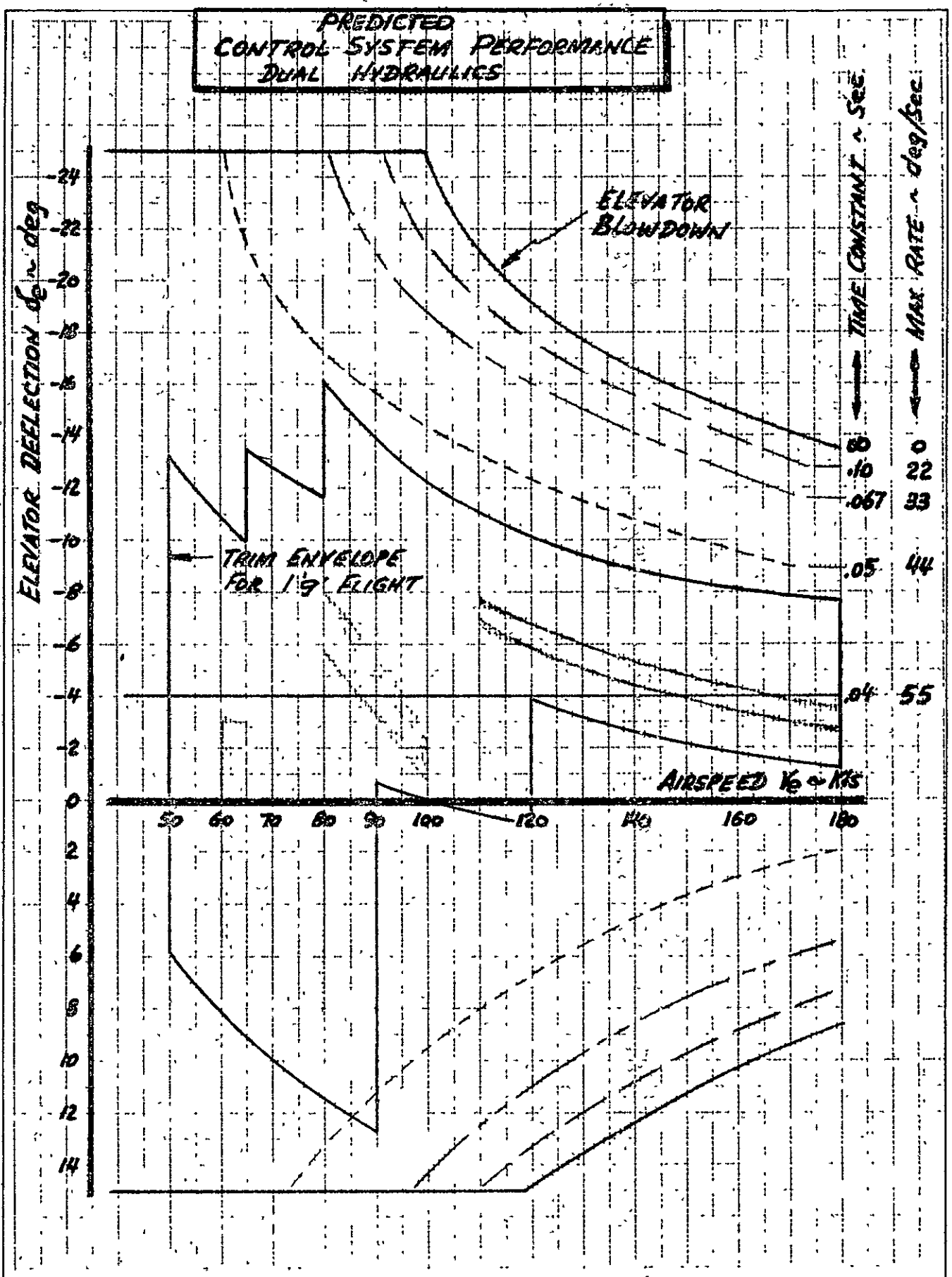

PREDICTED
CONTROL SYSTEM PERFORMANCE
DUAL HYDRAULICS
ELEVATOR DEFLECTION δe, deg
-24
-22
-20
-18
-16
-14
-12
-10
-8
-6
-4
-2
0
2
4
6
8
10
12
14
ELEVATOR BLOWDOWN
TRIM ENVELOPE FOR 1g FLIGHT
TIME CONSTANT τ, SEC
MAX. RATE, deg/sec
.08 0
.10 22
.067 33
.05 44
.04 55
AIRSPEED Ve, KTS
50
60
70
80
90
100
120
140
160
180

FIGURE 4.4-4

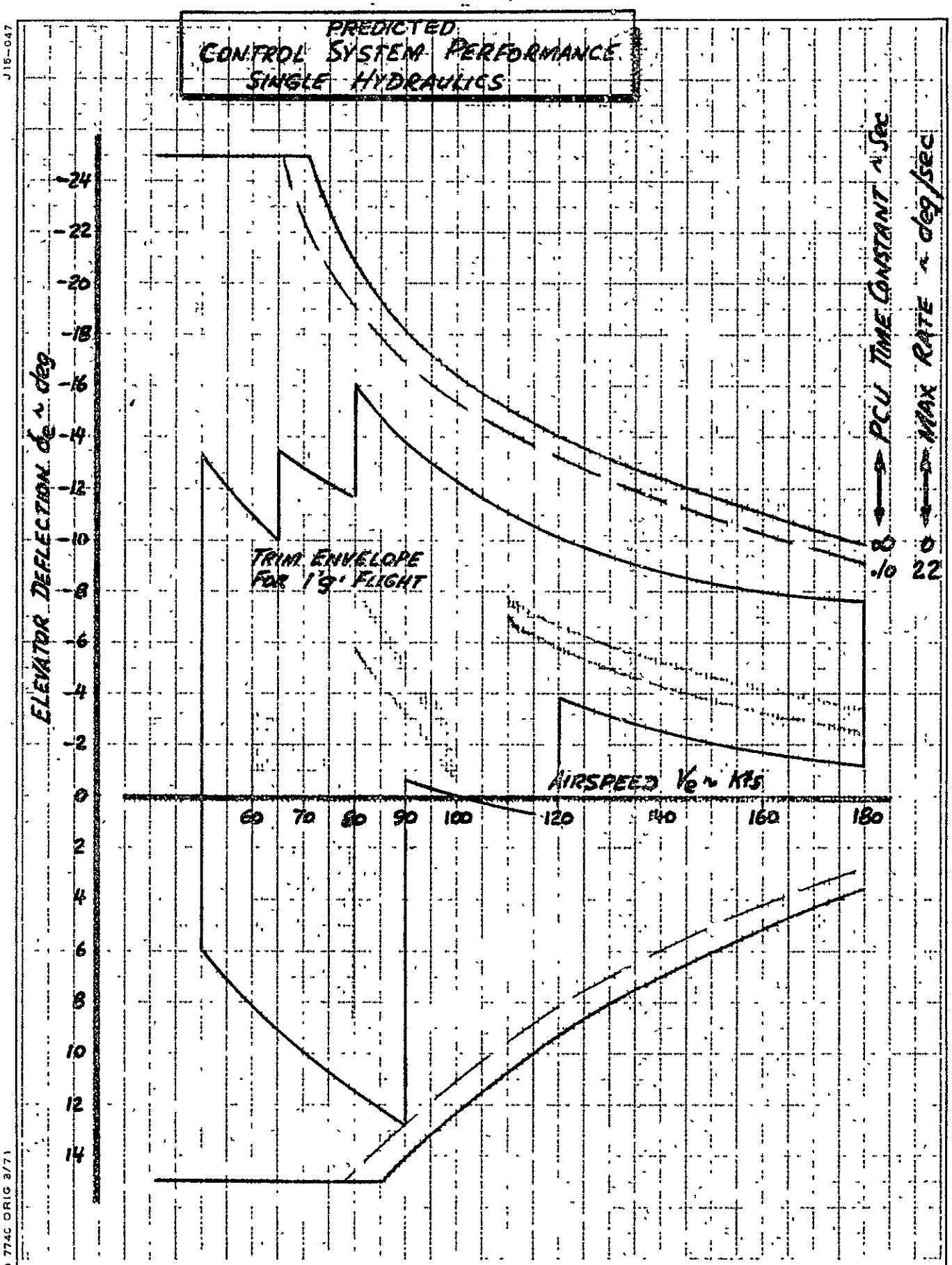
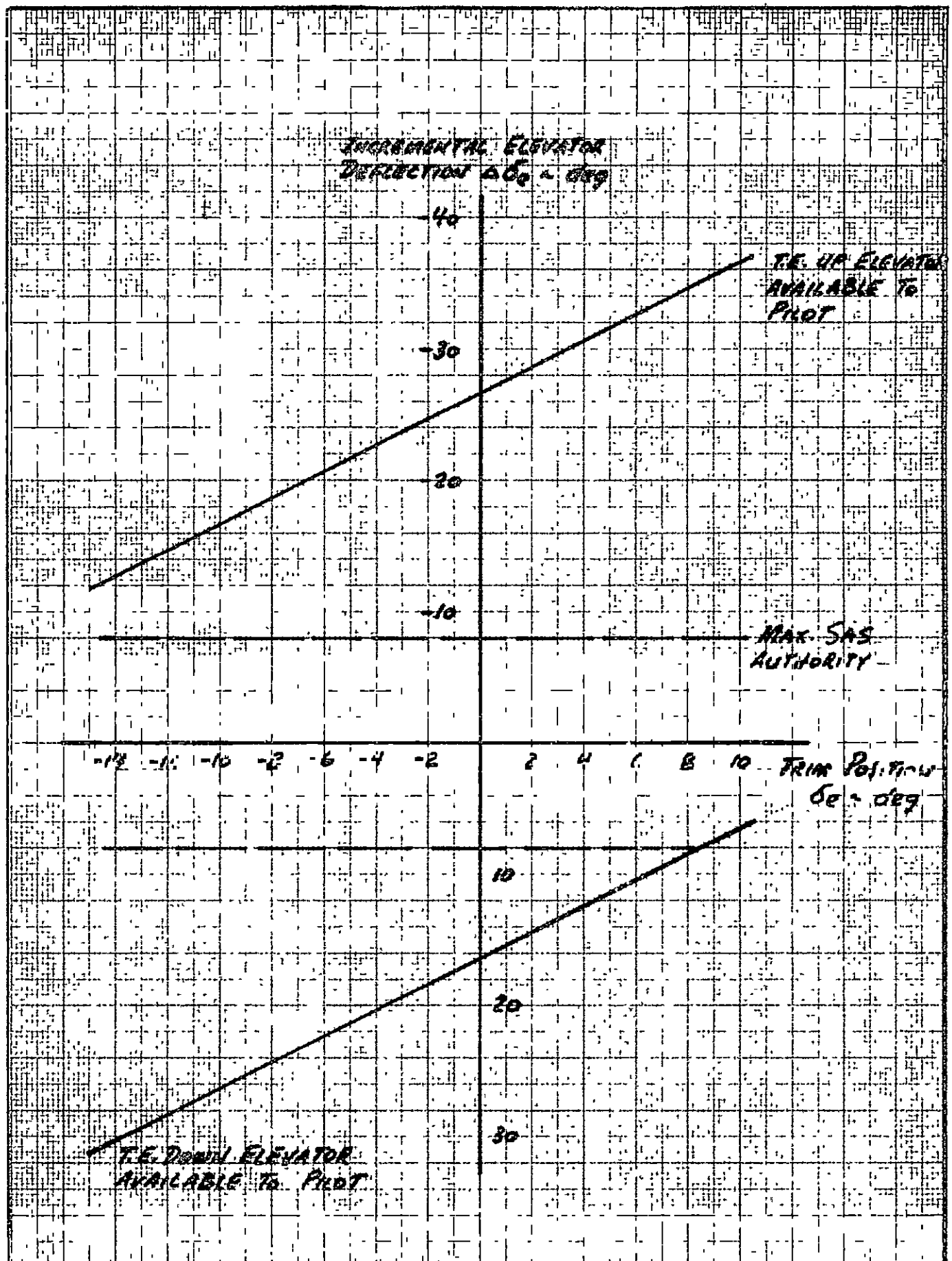

PREDICTED
CONTROL SYSTEM PERFORMANCE
SINGLE HYDRAULICS
ELEVATOR DEFLECTION δe ~ deg
-24
-22
-20
-18
-16
-14
-12
-10
-8
-6
-4
-2
0
2
4
6
8
10
12
14
TRIM ENVELOPE
FOR 1'g' FLIGHT
AIRSPEED Ve ~ KTS
60
70
80
90
100
120
140
160
180
PCU TIME CONSTANT ~ Sec
.08
.10
MAX RATE ~ deg/sec
0
22

FIGURE 4.4-5

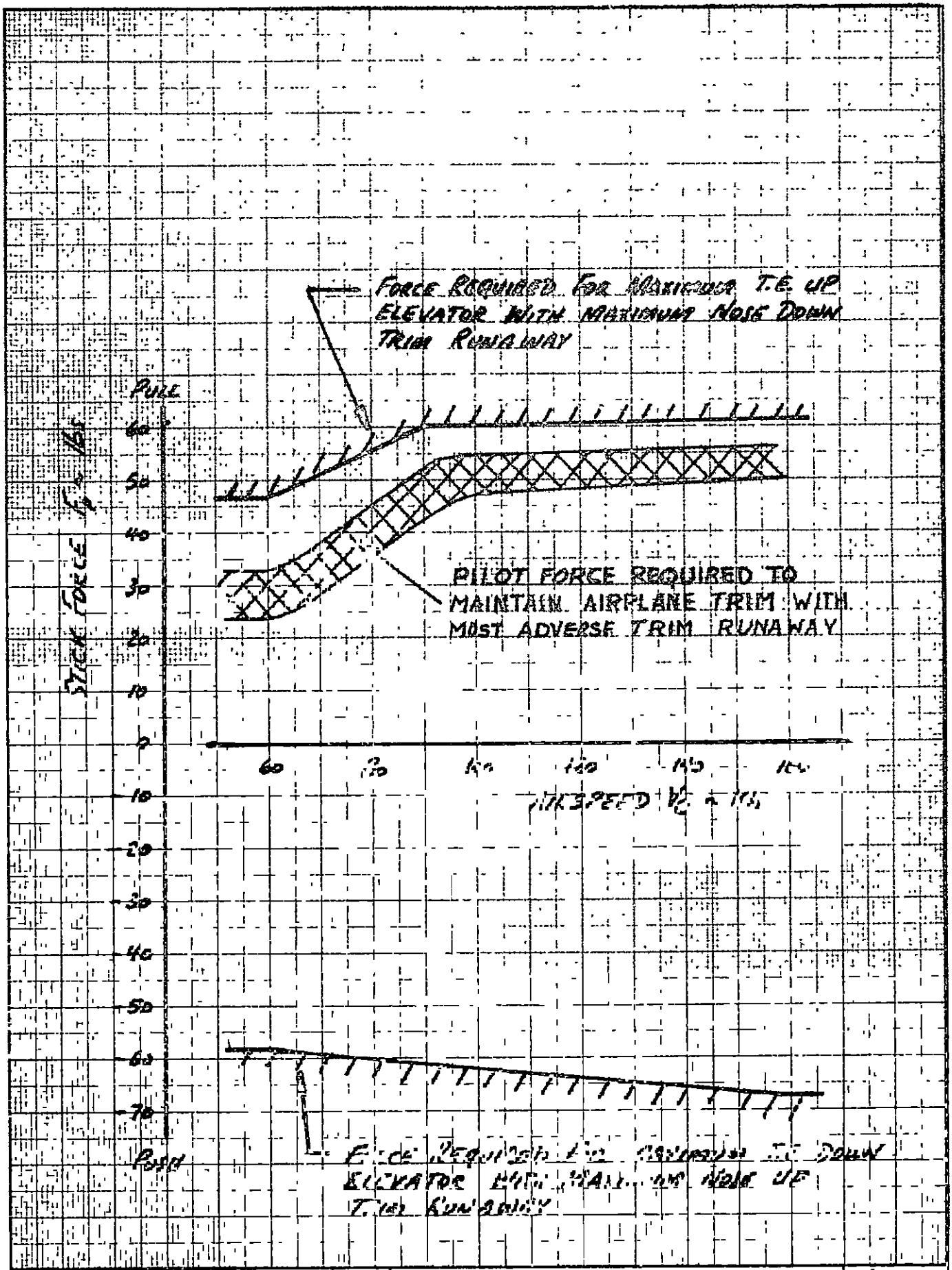


CALC			REVISED	DATE	PREDICTED ELEVATOR AUTHORITY VS TRIM
CHECK					
APR					
APR					
					THE BOEING COMPANY

MOD C-8A

FIG 4.4-6

U.E. ALBANENE 1051



CALC			REVISED	DATE	PREDICTED TRIM RUNAWAY OVERRIDE FORCES	NIDD C-BA
CHECK						FIG 4.4-7
APR						
APR					THE BOEING COMPANY	PAGE 53 D6-41830

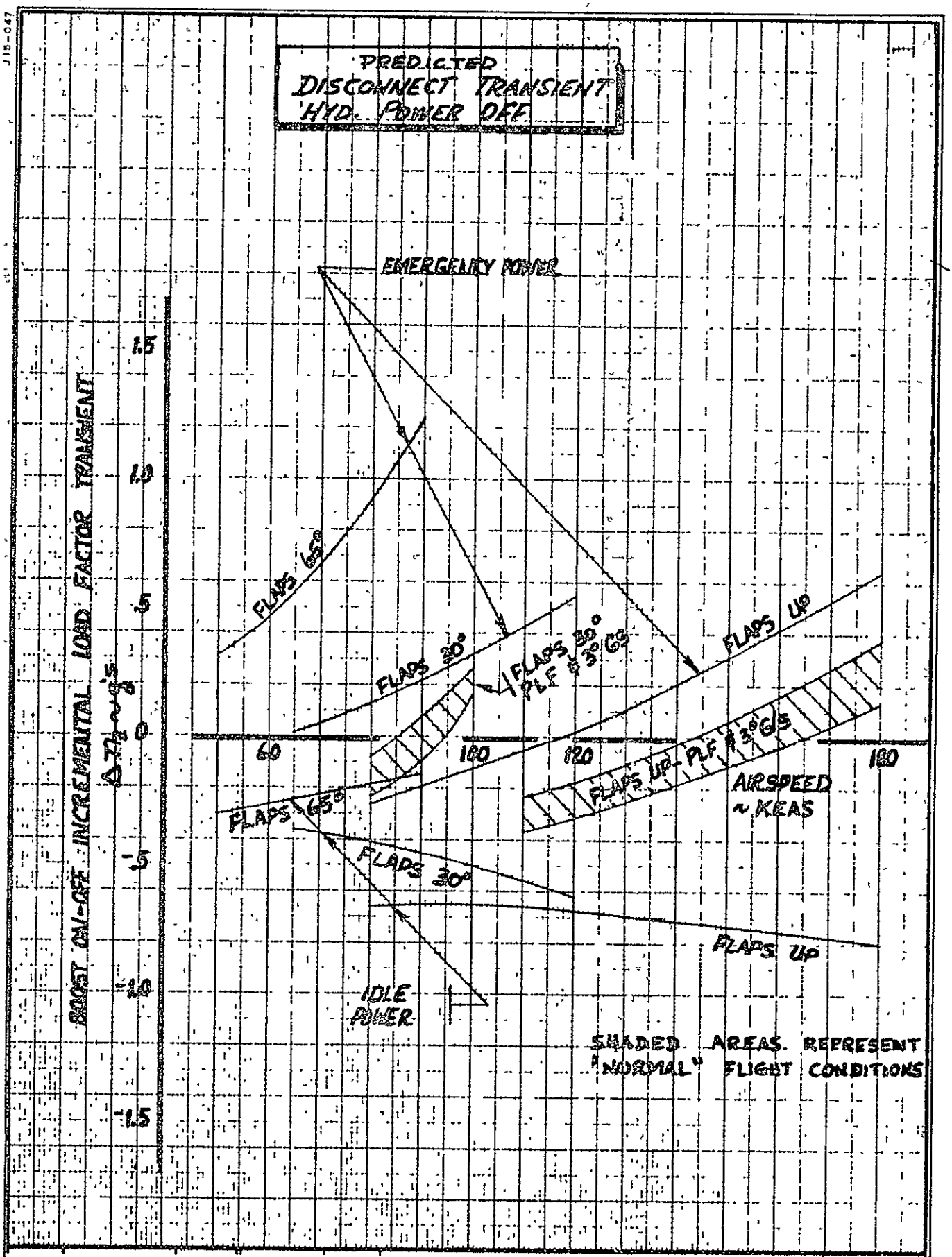
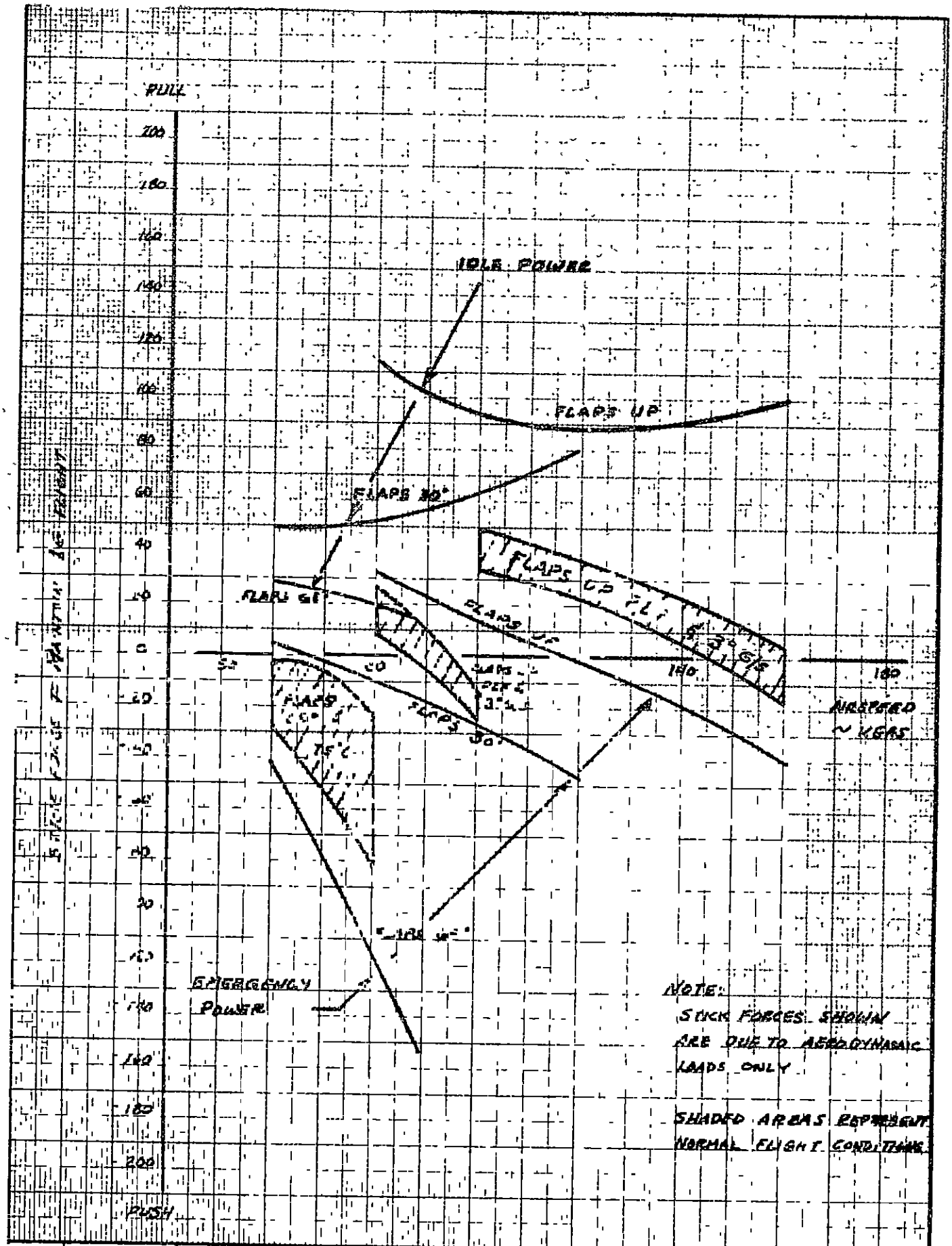

PREDICTED
DISCONNECT TRANSIENT
HYD. POWER OFF
EMERGENCY POWER
FLAPS 65°
FLAPS 30°
FLAPS 30° PLF & 3°G/S
FLAPS UP
FLAPS UP - PLF & 3°G/S
FLAPS 65°
FLAPS 30°
FLAPS UP
IDLE POWER
SHADED AREAS REPRESENT "NORMAL" FLIGHT CONDITIONS
BOOST ON-OFF INCREMENTAL LOAD FACTOR TRANSIENT
Δ n_z ~ g's
1.5
1.0
.5
0
-.5
-1.0
-1.5
60
100
120
180
AIRSPEED ~ KEAS

FIGURE 4.4-B



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PREDICTED
BOOST OFF STICK FORCES
TO MAINTAIN 1 G FLIGHT

MOD C-BA

FIG 4.4-9

THE BOEING COMPANY

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D1 4100 4240

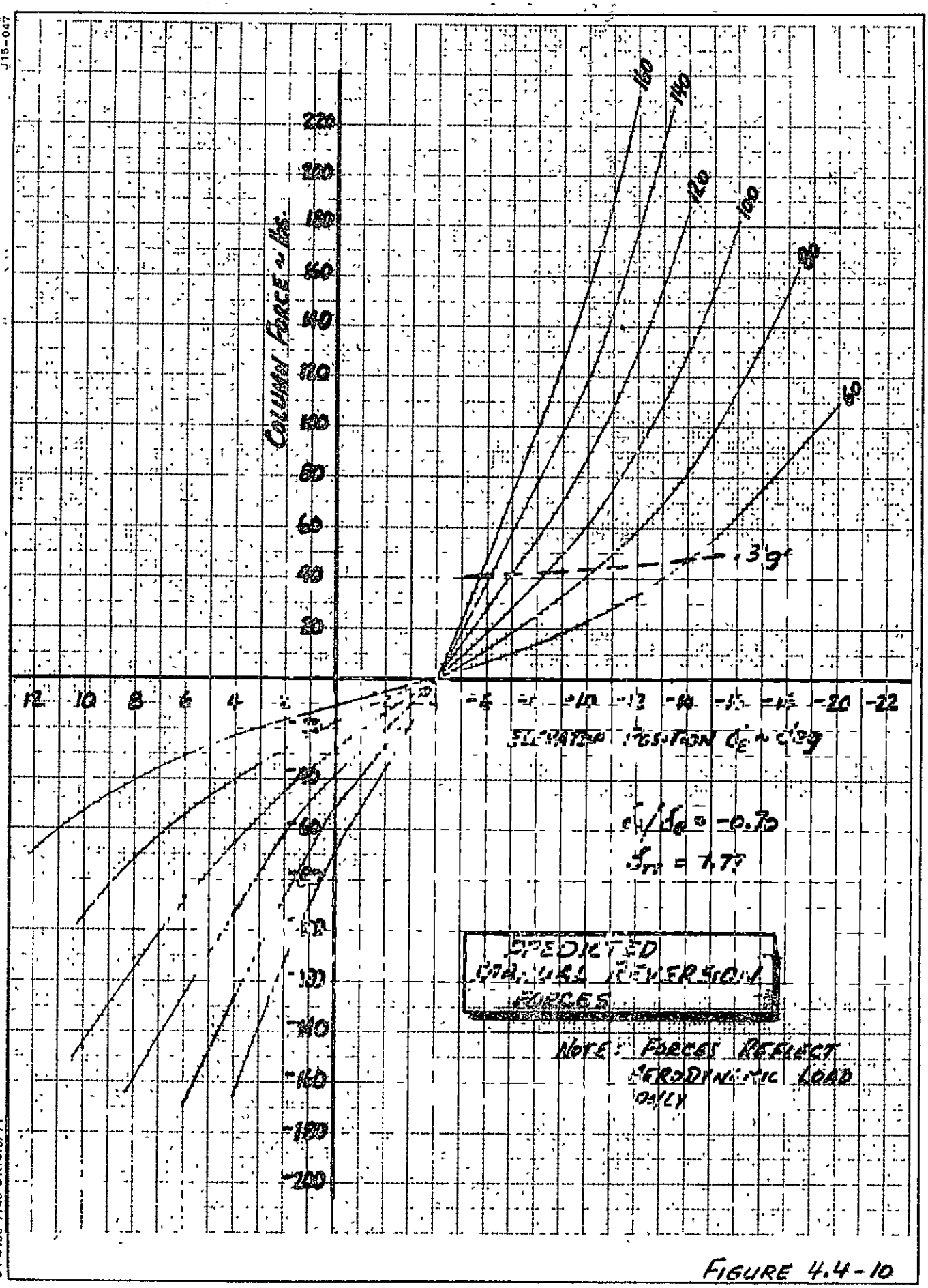
COLUMN FORCE ~ lbs.
ELEVATOR POSITION δ_E ~ deg
160
140
120
100
80
60
.3g
$\delta_e/\delta_e = -0.70$
$\delta_m = 1.77$
PREDICTED MANUAL REVERSION FORCES
NOTE: FORCES REFLECT AERODYNAMIC LOAD ONLY

FIGURE 4.4-10

5.0 ENGINEERING GROUND TESTS

Engineering ground tests on the powered elevator system were conducted in two phases. Initial tests were made on the modified empennage at Boeing, Seattle between May 30, 1973 and June 11, 1973. The horizontal and vertical stabilizers were mated and installed in their normal position for these tests, as shown in figure 5.0-1. The empennage was subsequently shipped to NASA-Ames Research Center for mating with the Mod C-8A airframe. Final ground tests were performed on the airplane between July 2, 1973 and July 16, 1973.

The purpose of the ground tests was to determine the operating characteristics of the powered elevator system and to verify that it was satisfactory for flight. All tests were conducted with the elevator surface unloaded. Tests conducted fall into two categories:

1. Control system proof and operations test
2. Control system performance tests

Test results are described below.

5.1 PROOF AND OPERATIONS TEST

The elevator control system was shown to withstand limit load with no permanent deformation or tendency to jam.

The control system was incrementally loaded through the control columns to a maximum combined pilot effort of 450 pounds in the push and 400 pounds in the pull direction. Hydraulic power to the elevator PCU was engaged for the test. The elevator surface was blocked at -25 degrees for the push test and +10 degrees for the pull test. The lower force in the pull direction was due to a testing error, rather than by intent. With the maximum load applied to the column, the control system was inspected and found free from potential jams. Following the test, the system was visually inspected and the rigging checked to verify that no permanent deformation had occurred.

5.2 CONTROL SYSTEM PERFORMANCE TESTS

5.2.1 Control System Gearing

Elevator versus Control Column - The elevator-to-column gearing in the airplane nose up direction was lower than predicted due to

a reduced aft quadrant-to-column deflection. The control column aft travel limit was therefore increased to 14 degrees to allow full trailing edge up elevator to be commanded.

The PCU installation was found to have a lower than predicted structural stiffness. This will cause a reduction in the elevator PCU steady state gain with increasing airspeed and hence a reduction in elevator-column gearing as shown in figure 5.2.1. For airspeeds below 110 knots, the maximum nose up elevator will be limited by the available aft column travel, as shown in figure 5.2-2. This reduction in elevator authority is not expected to cause a problem.

Elevator versus Geared Tab Deflection - The predicted and actual geared tab-to-elevator deflection is shown in figure 5.2-3. The actual gear ratio δ_t/δ_e is -0.6 for elevator deflections between +10 degrees and -14 degrees, compared to a predicted ratio of -0.7. In addition, the geared tab neutral rig point is at -4. degrees elevator instead of -3 degrees. These deviations were considered acceptable for flight test.

5.2.2 Control System Feel Characteristics

Feel characteristics were determined for varying airspeeds and trim conditions by moving the pilot's control column and recording column force versus elevator position on an x-y plotter.

Initial tests showed the control system to have 4.1 pounds friction, of which 3.1 pounds were due to the control column and control cables. Total system friction was subsequently reduced to 3.3 pounds by reducing the cable rig load from 118 pounds to 80 pounds and realigning three fairleads. All ground test results are for this configuration unless noted otherwise. Typical test data at various simulated airspeeds are shown in figures 5.2-4 through 5.2-8.

Breakout and friction forces as a function of trim and airspeed are shown in figure 5.2-9. The forces generally increase with increasing nose down trim and increasing airspeed, and are symmetric with respect to push and pull. The average breakout force is 6.0 pounds, the average friction force 3.9 pounds. Positive centering, but not absolute centering was found for all conditions. Some reduction in friction is expected in flight due to increased vibration. This will reduce the breakout forces and improve system centering.

The theoretical and actual feel gradient (F_s/δ_e) as a function of airspeed is shown in figure 5.2-10. Test results are valid for $\pm$ 5 degrees elevator deflection about zero degrees elevator trim. The ground test results deviate from theoretical in two respects:

a. The feel gradient in the column aft direction is approximately 20% lower than in the column forward direction.

b. The feel gradient error (theoretical gradient - actual gradient) becomes increasingly more negative with increasing airspeed for both column forward and aft.

The feel gradient variation between push and pull is attributed primarily to non-linear gearing. The feel gradient did not vary appreciably with either hydraulic system A, B, or A and B engaged.

5.2.3 Elevator Feel System Computer

The elevator feel computer static gain was checked by applying varying pitot pressures corresponding to the full airspeed range and recording the resultant feel pressures. Final ground test results agree fairly well with the theoretical results, although an inconsistency was noted in the 'A' channel results between the final test results and earlier tests. Figures 5.2-11 and 5.2-12 show theoretical and actual results for systems A and B.

Both feel computers A and B were found to be sensitive to feel pressure flow demands. Figure 5.2-13 shows a typical response. Feel pressure A and B variations are 146 psi and 85 psi respectively

about a 240 psi nominal feel pressure, with a maximum elevator rate of approximately 34 degrees per second.

The feel computer performance is dependent on the vibration environment of the computer. With no vibration, the feel computer has up to 100 psi feel pressure hysteresis. At 60 knots, this would result in a 30% variation in feel gradient. With a small amount of vibration, hysteresis is less than 30 psi, reducing the feel gradient variation to less than 10%. Improved performance will require an increase in the pressure control valve flow gain and reduced valve spool friction.

The feel computer pitot system contains two small drain holes. There was some indication that these drain holes resulted in a lower pitot pressure at the feel computer than at the pitot source. The ground test was therefore conducted with the drain holes plugged. The effect of these drain holes on feel pressure will be evaluated during flight test.

It was possible to trigger a sustained oscillation in the 'B' system feel computer by applying a simulated airspeed of 100 knots or greater and then switching the 'B' hydraulic system supply pressure off and then back on. The 'B' system would immediately oscillate at approximately 7 Hz and + 200 psi. This oscillation could be stopped by reducing airspeed. It was <u>not</u> possible to induce the oscillation by moving the control column. The oscillation could also not be induced in the 'A' system feel computer. Although the oscillation could be felt in the control column, it did not couple with the elevator PCU. The oscillation was further investigated during flight test to determine if the oscillation is aided by the pitot pressure ground test equipment.

5.2.4 <u>Control System Resolution</u>

The control column-elevator resolution was found to be good, with no apparent deadzones. Elevator position hysteresis for full column travel is 0.3 degree elevator,

5.2.6 Trim System

Trim authority for the powered elevator system was found to be +8.2 degrees, -14.8 degrees elevator. The no-load trim rate was 2.1 degrees elevator per second nose up and 2.3 degrees per second nose down. These rates were achieved after inserting a 3 ohm resistor in series with the motor armature circuit. Without the resistor the trim rate was 2.9 degrees per second nose up and 3.1 degrees per second nose down. The trim rate appeared to be insensitive to aiding loads, but decreases approximately 20 percent for opposing loads

5.2.7 Elevator SAS Actuator

The SAS actuator has a maximum authority of ± 5 degrees elevator, corresponding to an actuator stroke of ± .32 inch. This authority is independent of elevator position provided that the total elevator command does not exceed + 15 degrees or -25 degrees. The SAS authority may be increased to a maximum of -7.5 degrees, +7.9 degrees elevator (corresponding to ± .5 inch SAS actuator stroke) by removing the internal actuator stops.

The elevator to SAS actuator static gain was found to be 16 degrees/inch. This gain is essentially independent of elevator trim. (Figure 5.2-14).

The SAS actuator has a maximum no-load rate of ± 33 degrees elevator per second. Actuator centering and locking following disengagement was positive under all conditions tested. Time to center from maximum stroke is approximately 1.5 seconds. System resolution for SAS actuator inputs was found to be excellent. The hysteresis between SAS command and elevator surface position due to control system non-linearities was found to be .07 degree elevator.

5.2.5 Control System Dynamic Response

The overall control system response to pilot inputs was evaluated by "sinusoidally" moving the control column and recording column force, column position and elevator position. Test results show the elevator to closely follow the control column inputs. Typically, the elevator position lags the column force by approximately 30 degrees and column position by approximately 10 degrees at a frequency of 1.4 radians/second. The damping of the mechanical control system is satisfactory, with the column undergoing only one overshoot when displaced and allowed to center.

The dynamic response of the elevator power control unit (PCU) was determined by sinusoidally driving the PCU input through the SAS actuator. Test results showed the damping to be close to the predicted value, but the natural frequency to be 9.0 rad/sec lower. Figures 5.2-15 and -16 show the frequency response for both one and two hydraulic systems. Static load tests showed the low natural frequency to be due to a considerably lower than predicted structural stiffness. The reduction in natural frequency should have no effect on pilot or autopilot control of the airplane, since the gain and phase shift below 10 radians/second is relatively unchanged between the predicted and actual response.

The maximum no-load elevator rate was found to be 54 degrees per second trailing edge up and 61 degrees per second trailing edge down.

Ground test results showed the power control unit installation to be stable with either hydraulic system A, B, or A and B engaged. The installation is however dependent on structural feedback for stability, as the PCU could be made to oscillate by locking out this feedback in either one or both directions. The maximum structural feedback is limited by stops in order to control the magnitude of the load seen by the PCU reaction link. Because of the lower structural stiffness, these stops were adjusted to prevent bottoming under full aerodynamic load.

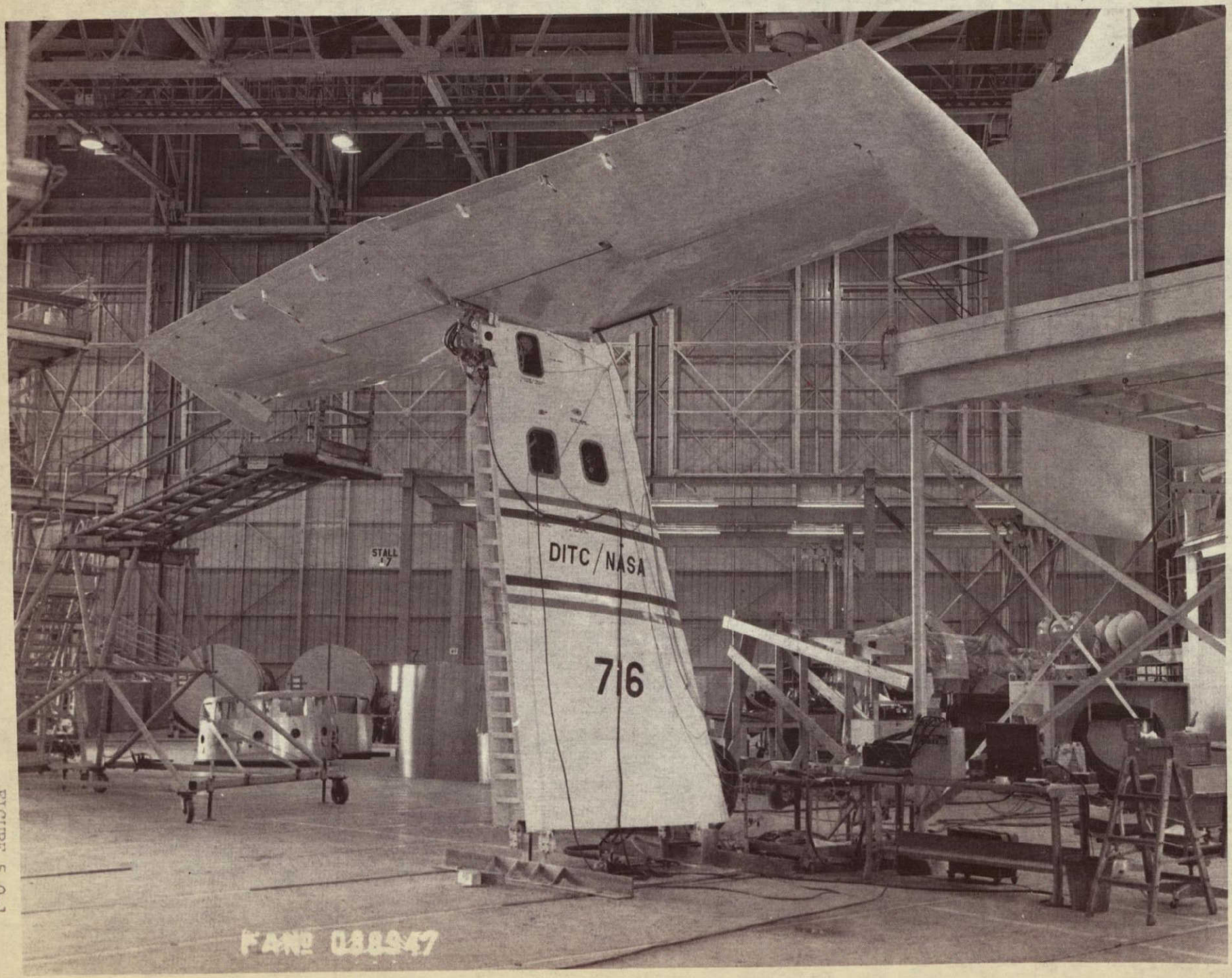

FIGURE 5.0-1
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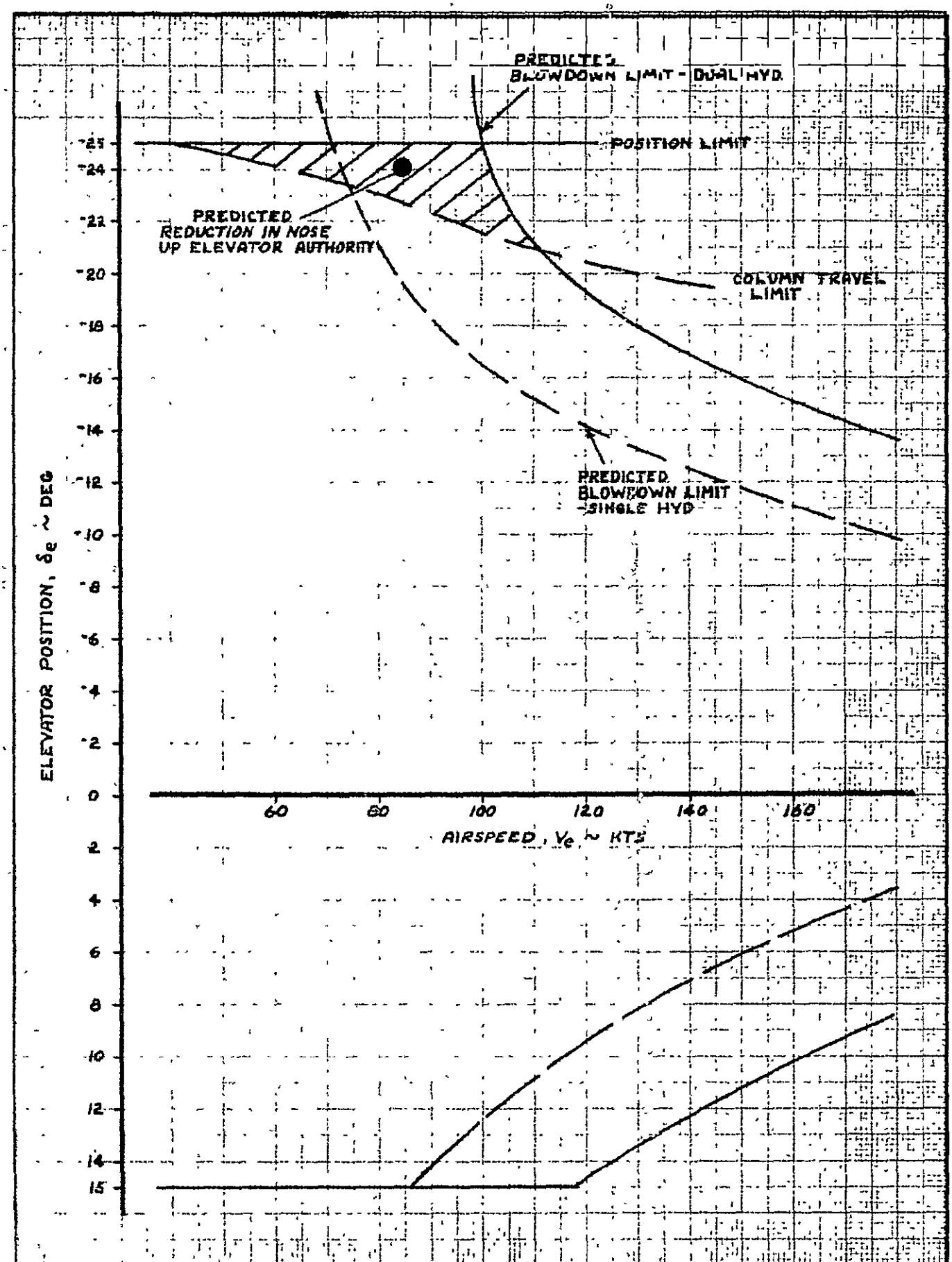

CALC		REVISED	DATE	PREDICTED ELEVATOR AUTHORITY
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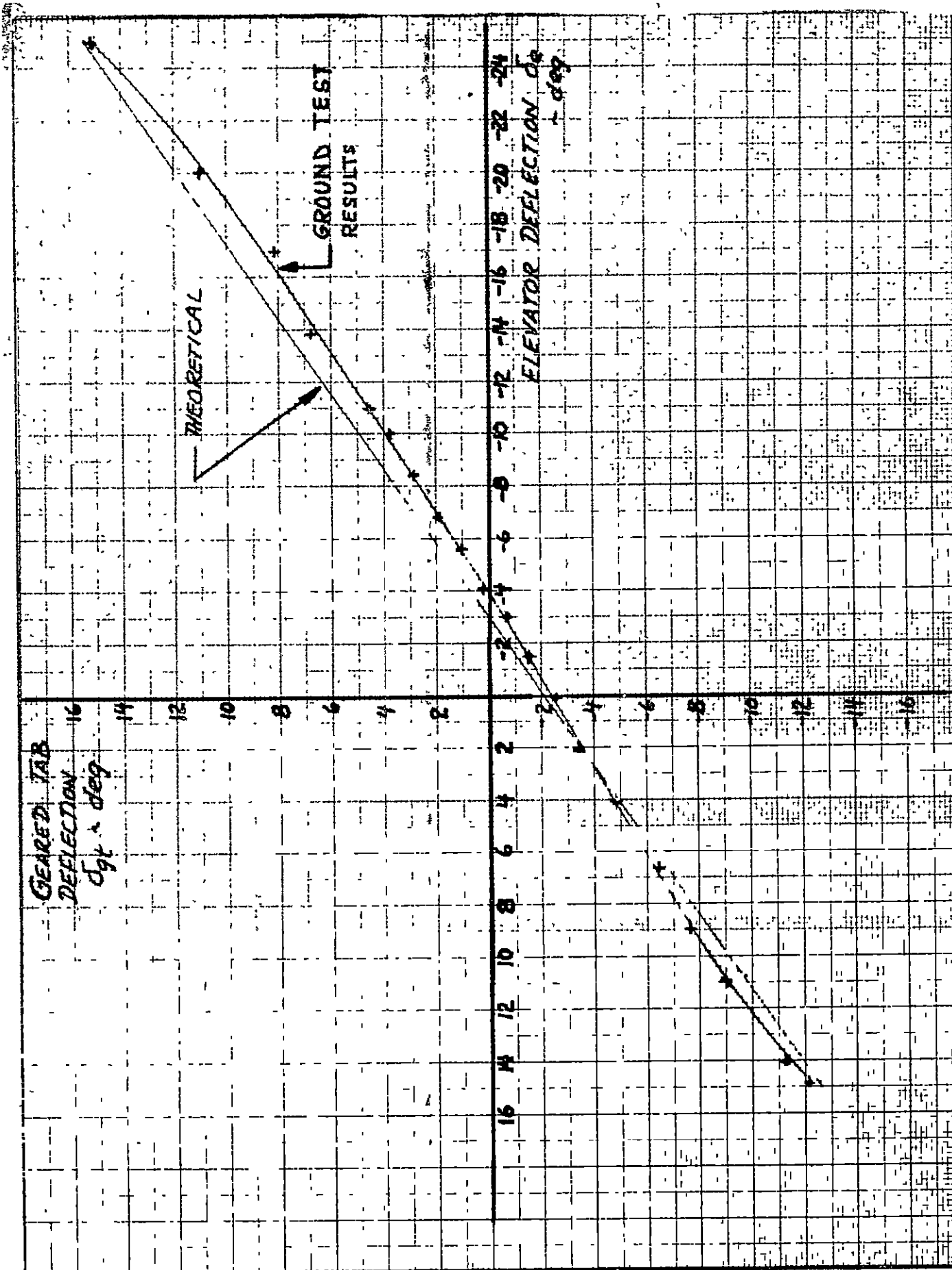
INK	HEMBREE 1-21-74			THE BOEING COMPANY

MOD C-8A

FIG 5-2-2



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APR				
INK	J. L. EINSEN			

GEARED TAB vs. ELEVATOR DEFLECTION

THE BOEING COMPANY

MOD C-8A

FIG 5-2-3

D1-11189

K-E ALBANENE 1961

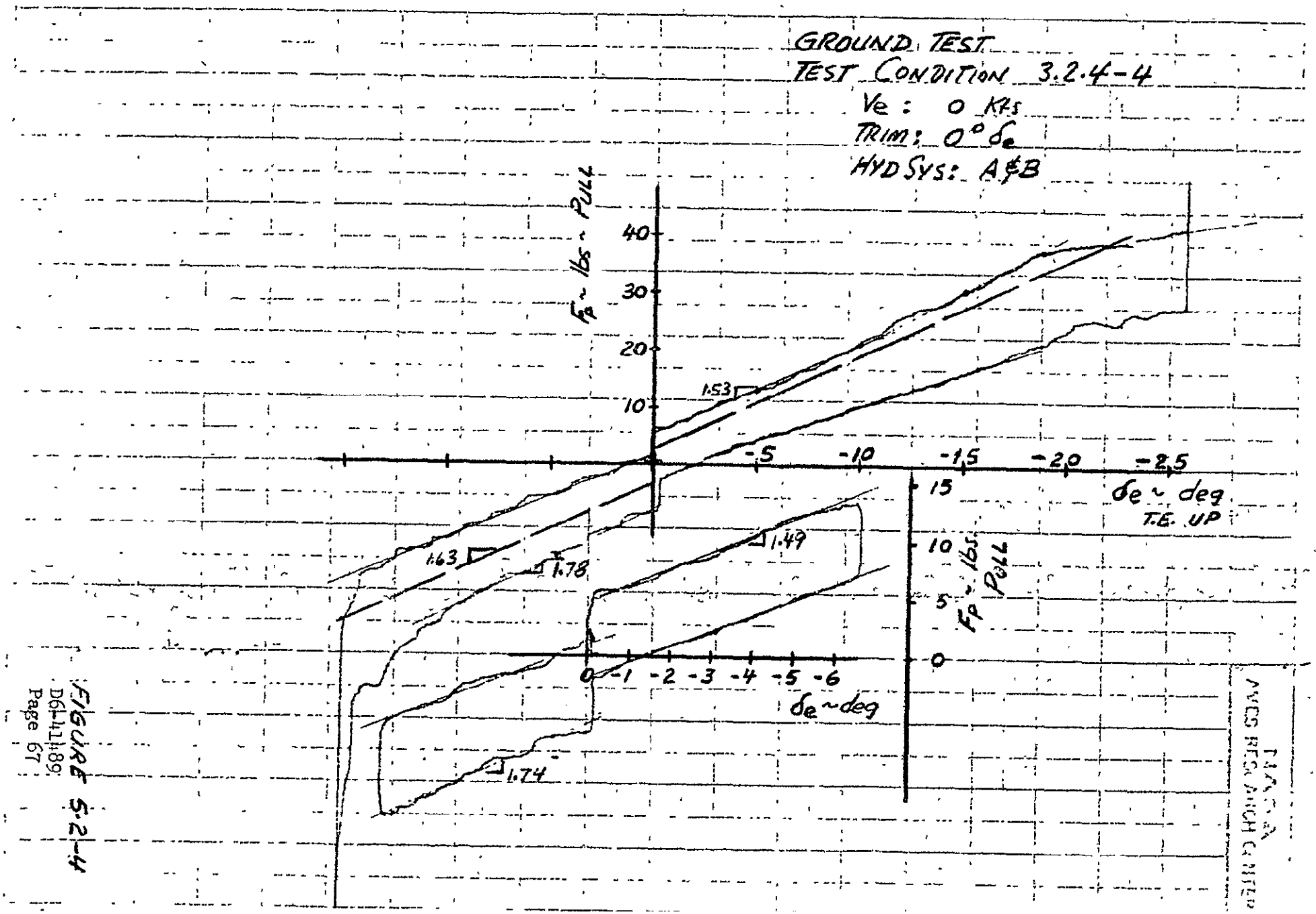
GROUND TEST
TEST CONDITION 3.2.4-4
Ve : 0 KTS
TRIM: 0° δe
HYD SYS: A&B
Fp ~ lbs ~ PULL
40
30
20
10
1.53
-5
-10
-15
-20
-25
δe ~ deg
T.E. UP
15
10
5
0
Fp ~ lbs PULL
1.63
1.78
1.49
1.74
0 -1 -2 -3 -4 -5 -6
δe ~ deg

FIGURE 5.2-4

FIGURE 5.2-5

NASA
AMES RESEARCH CENTER

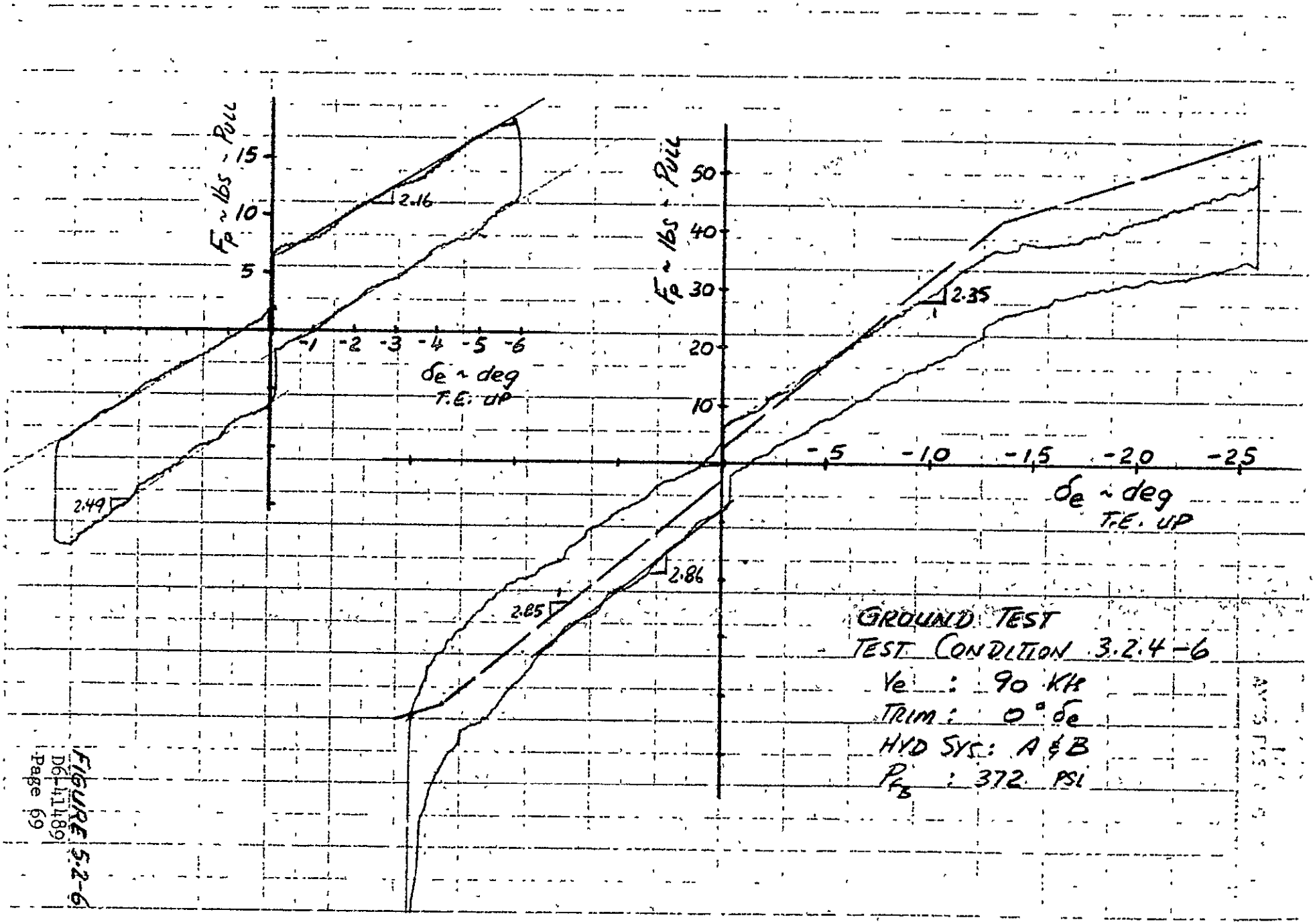
F_p ~ lbs - Pull
15
10
5
2.16
-1 -2 -3 -4 -5 -6
δ_e ~ deg
T.E. UP
2.49
F_p ~ lbs - Pull
50
40
30
20
10
2.35
-5 -10 -15 -20 -25
δ_e ~ deg
T.E. UP
2.86
2.65
GROUND TEST
TEST CONDITION 3.2.4-6
Ve : 90 KIAS
TRIM : 0° δ_e
HYD SYS: A & B
P_{H_B} : 372 PSI

FIGURE 5.2-6

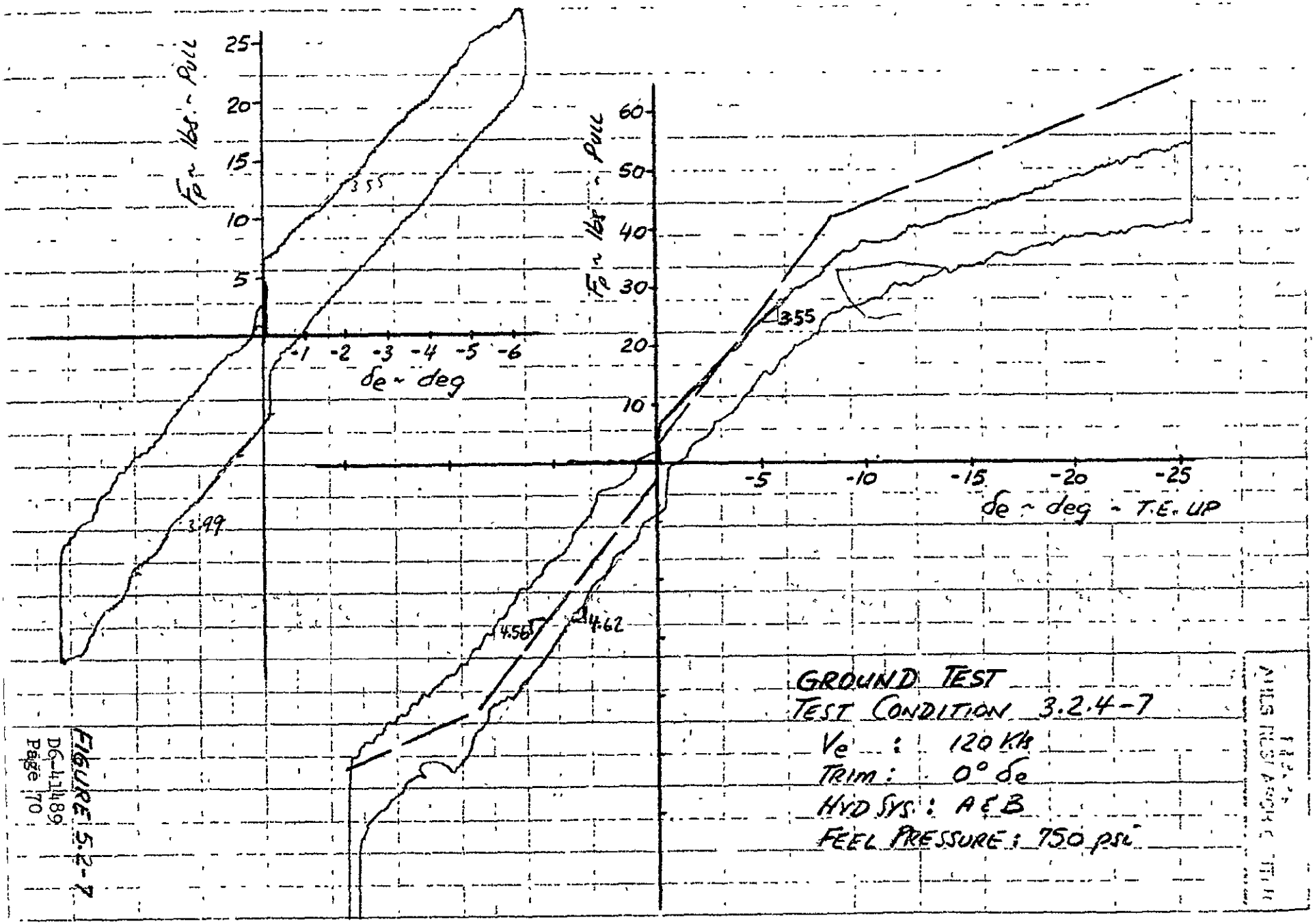
F_e ~ lbs ~ pull
25
20
15
10
5
3.55
3.99
-1 -2 -3 -4 -5 -6
δ_e ~ deg
F_e ~ lbs ~ pull
60
50
40
30
20
10
3.55
4.56
4.62
-5 -10 -15 -20 -25
δ_e ~ deg ~ T.E. UP
GROUND TEST
TEST CONDITION 3.2.4-7
Ve : 120 KIAS
TRIM : 0° δ_e
HYD SYS : A & B
FEEL PRESSURE : 750 psi

FIGURE 5.2-7

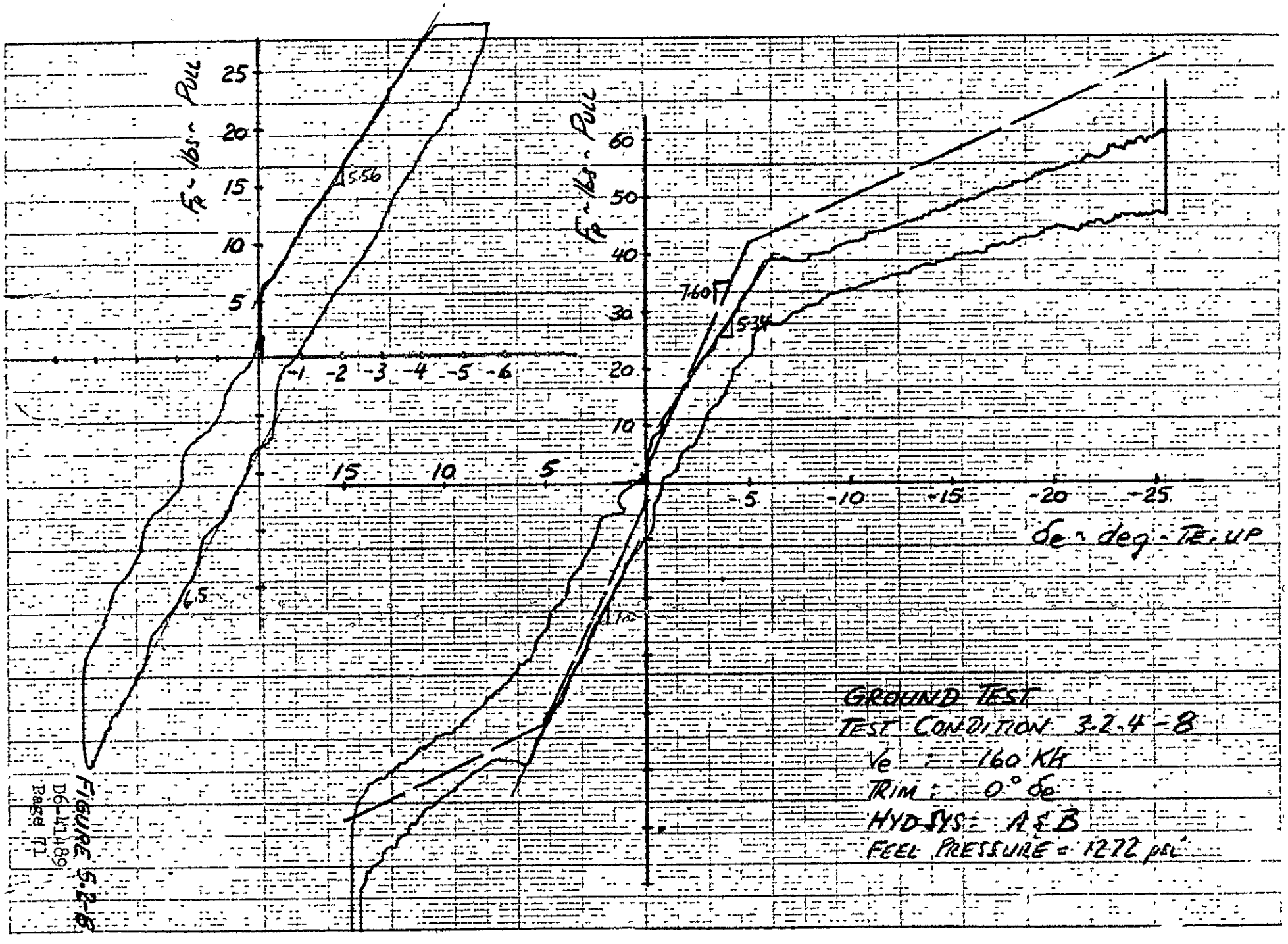

FIGURE 5.2-8
D6-41189
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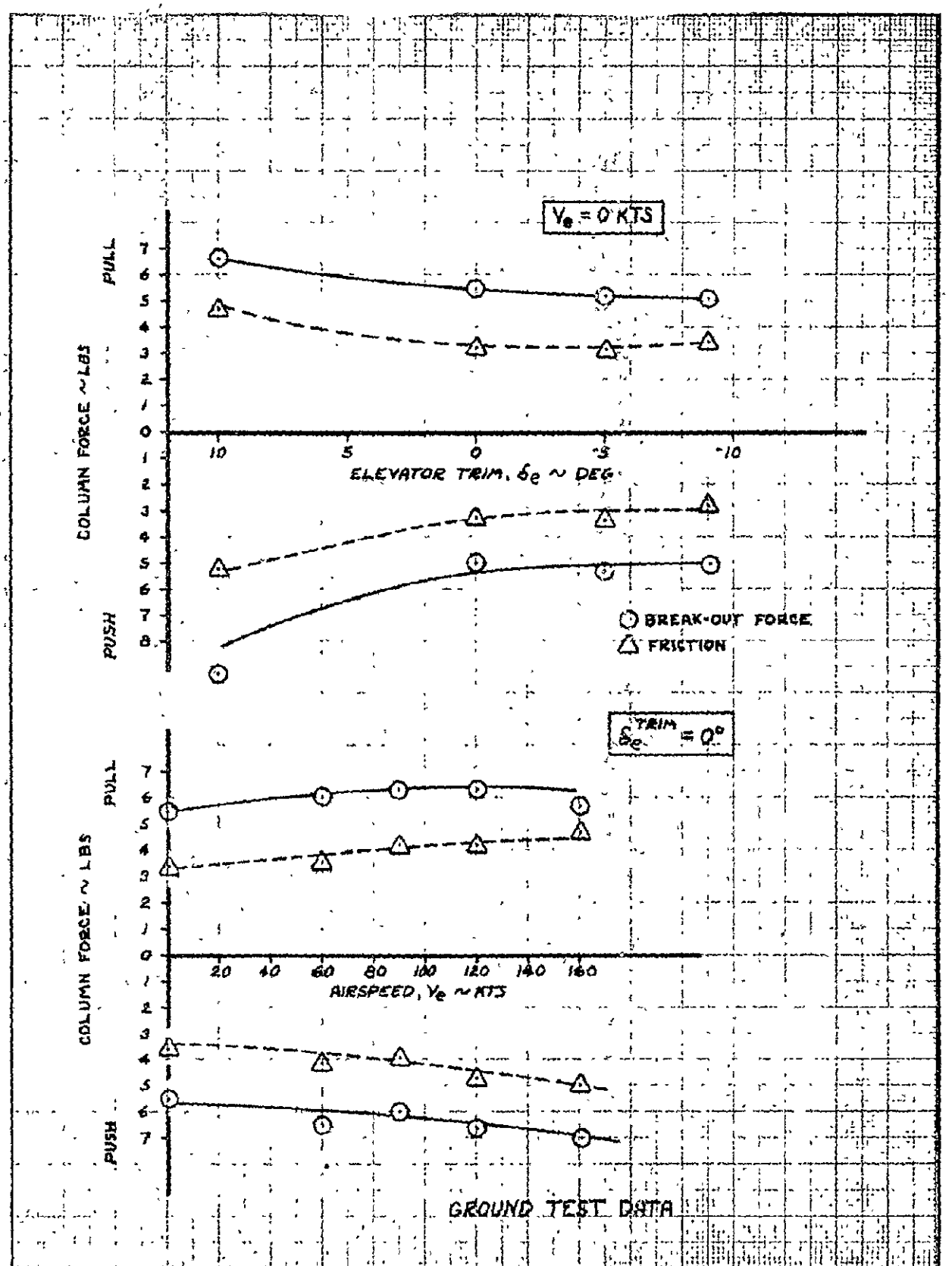

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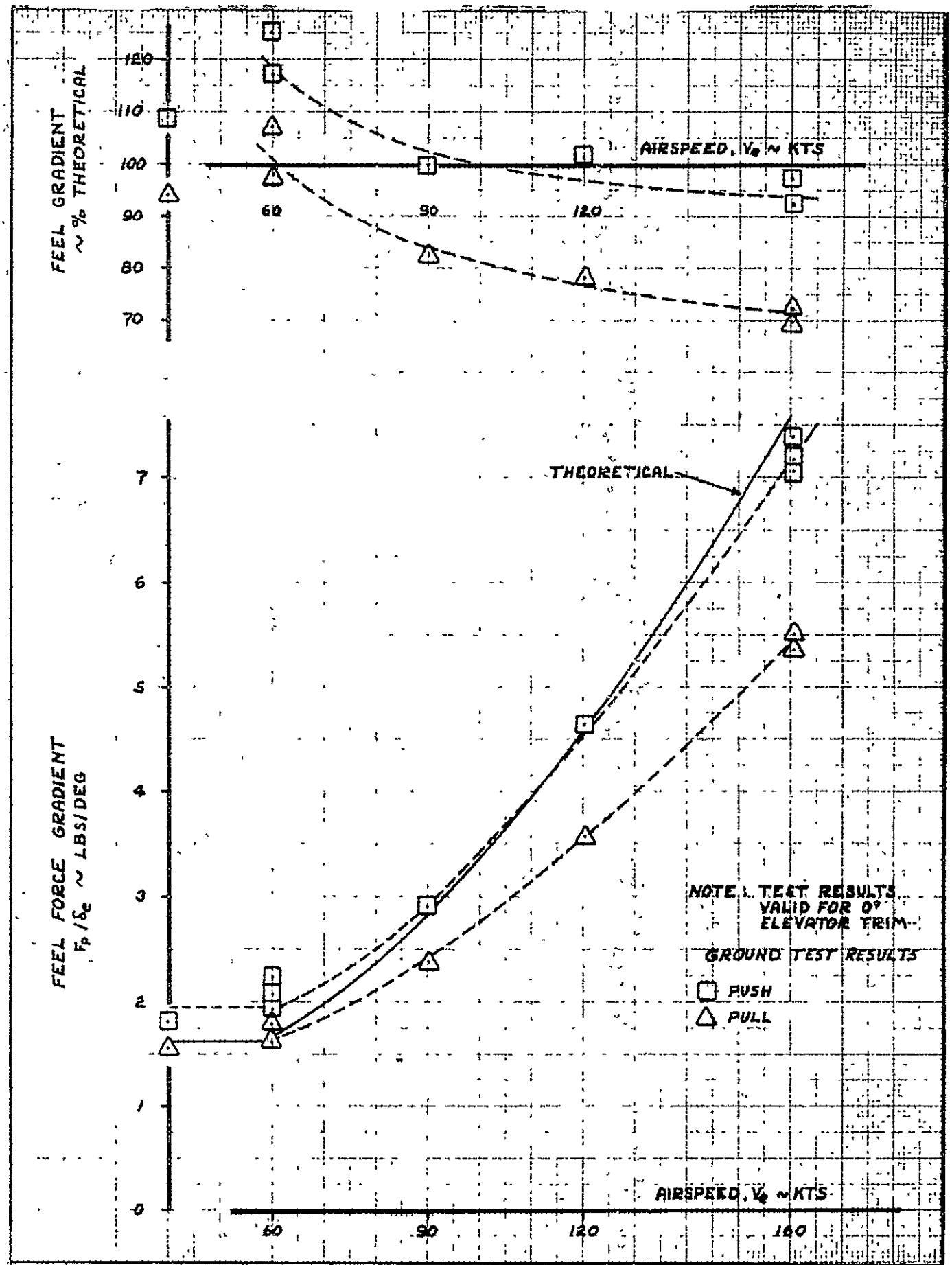
CONTROL SYSTEM BREAK-OUT AND FRICTION FORCES

MOD C-8A

FIG 5.2-9

THE BOEING COMPANY



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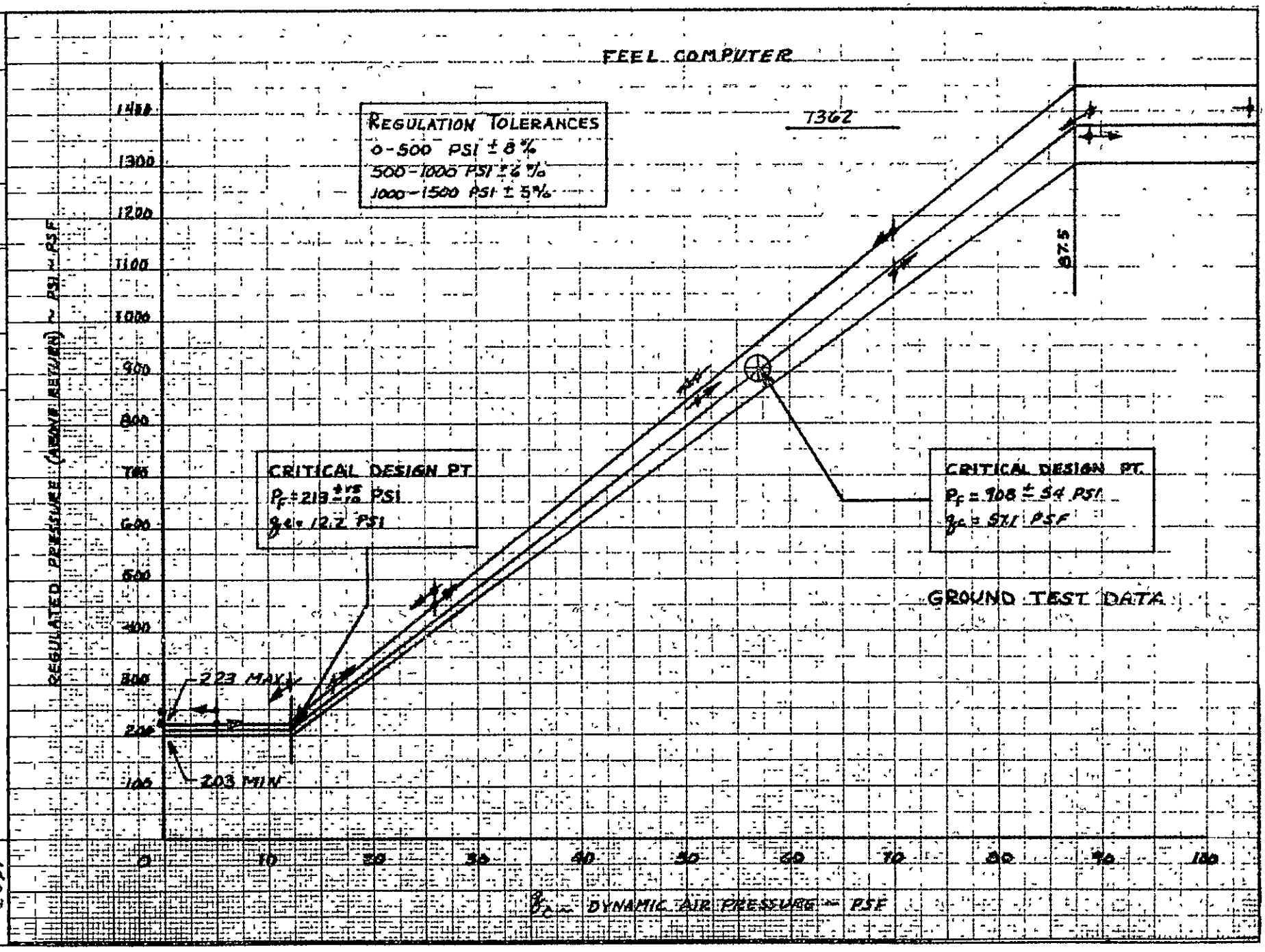
FEEL FORCE GRADIENT
VS
AIRSPEED

THE BOEING COMPANY

MOD C-8A

FIG 5.2-10

K+E ALBANENE 1951



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FEEL COMPUTER PERFORMANCE
'A' SYSTEM

THE BOEING COMPANY

MOD C-8A

FIG. 5.2-11

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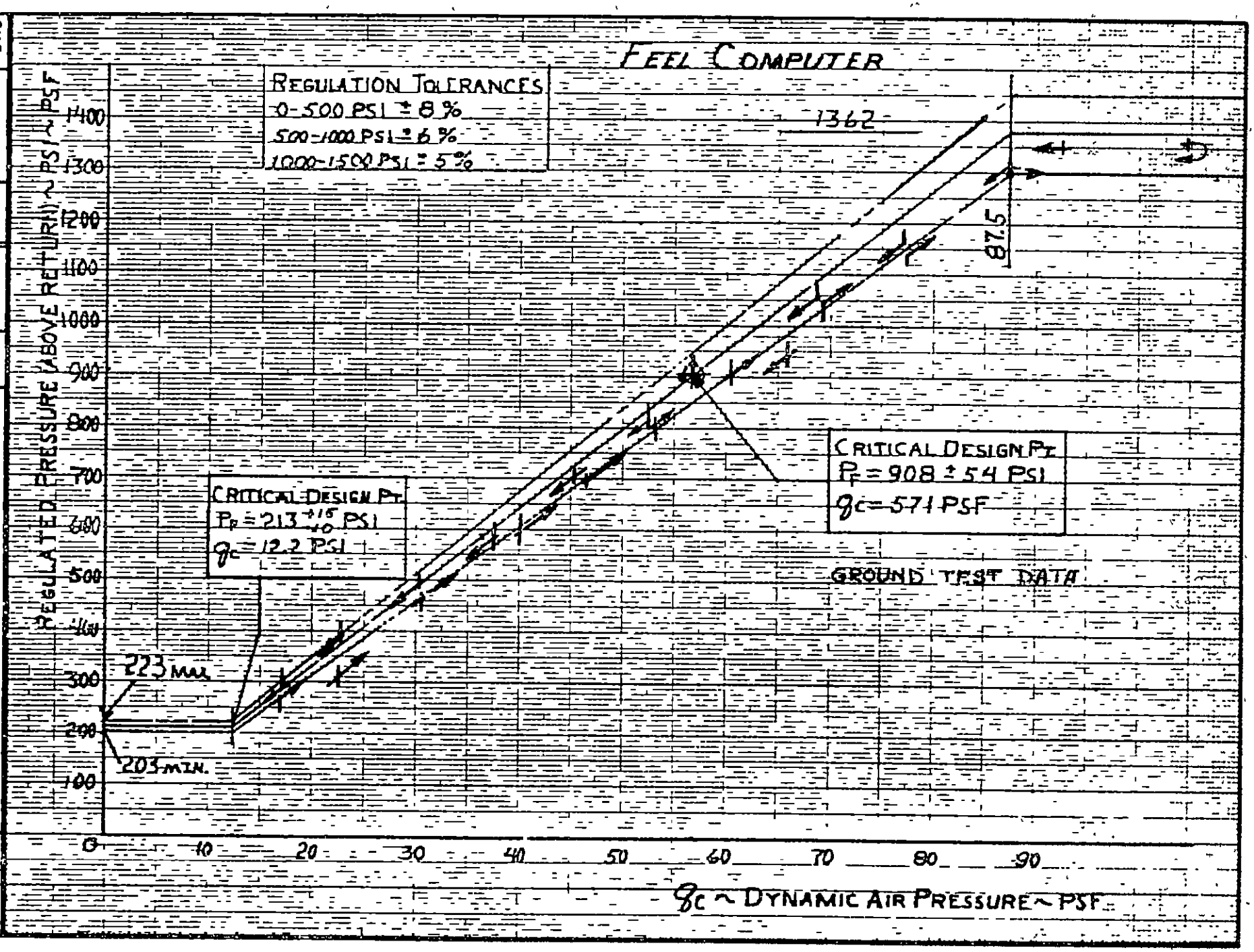
D6-41489



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INK	J. JENSEN		

FEEL COMPUTER PERFORMANCE

'B' SYSTEM

THE BOEING COMPANY

MOD C-8A

FIG. 5.2-12

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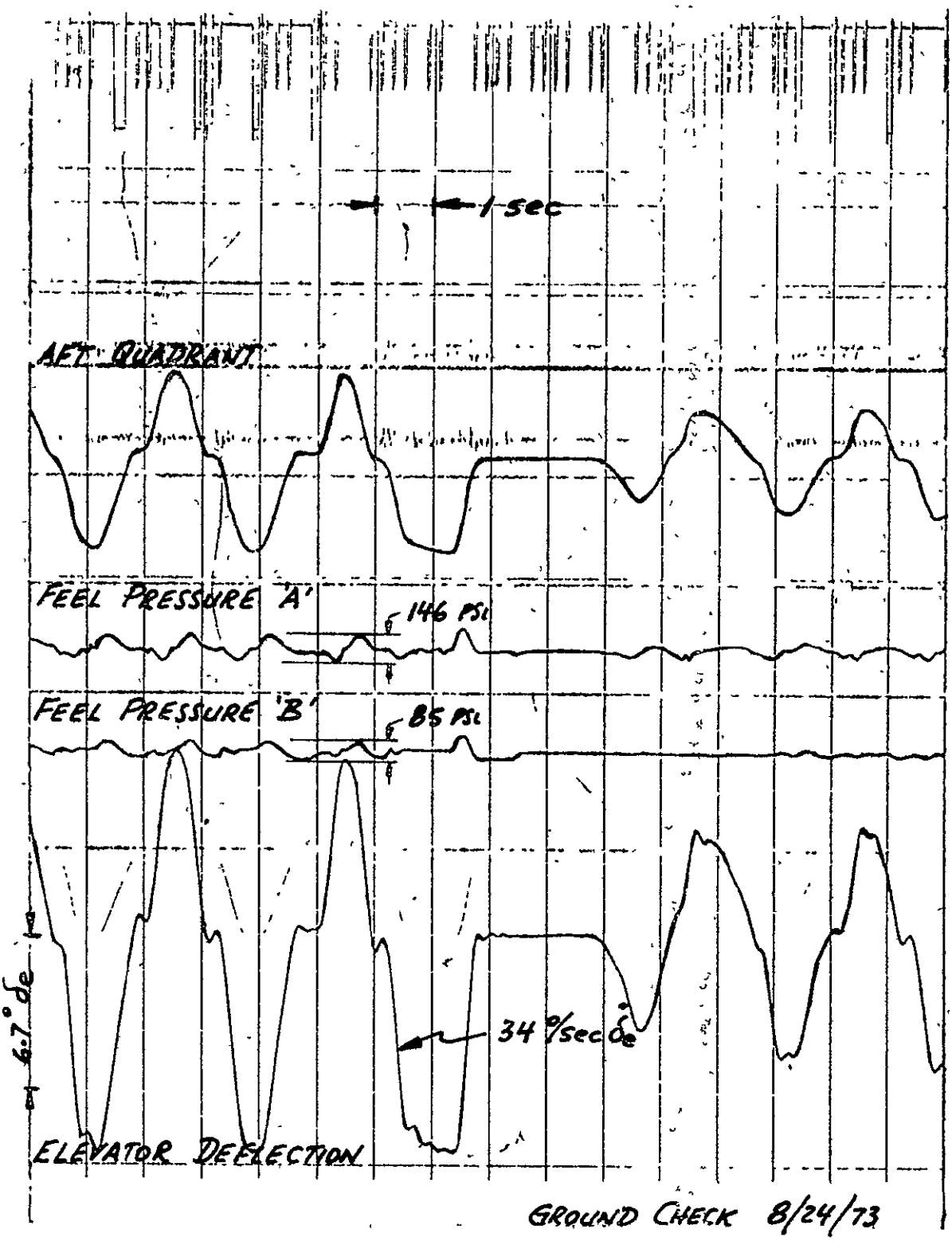
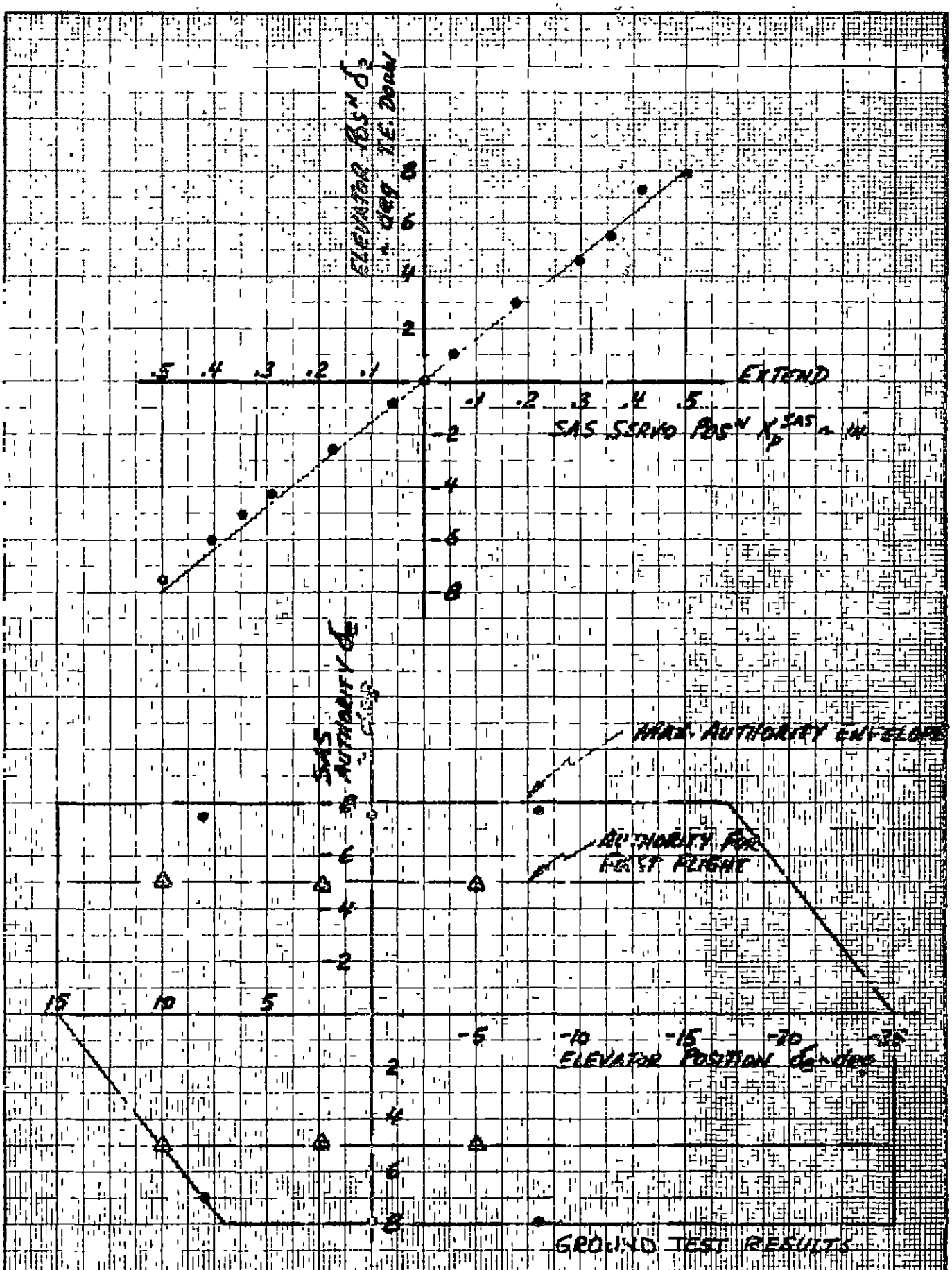


GROUND TEST

FEEL COMPUTER DYNAMIC RESPONSE

FIGURE 5.2-13



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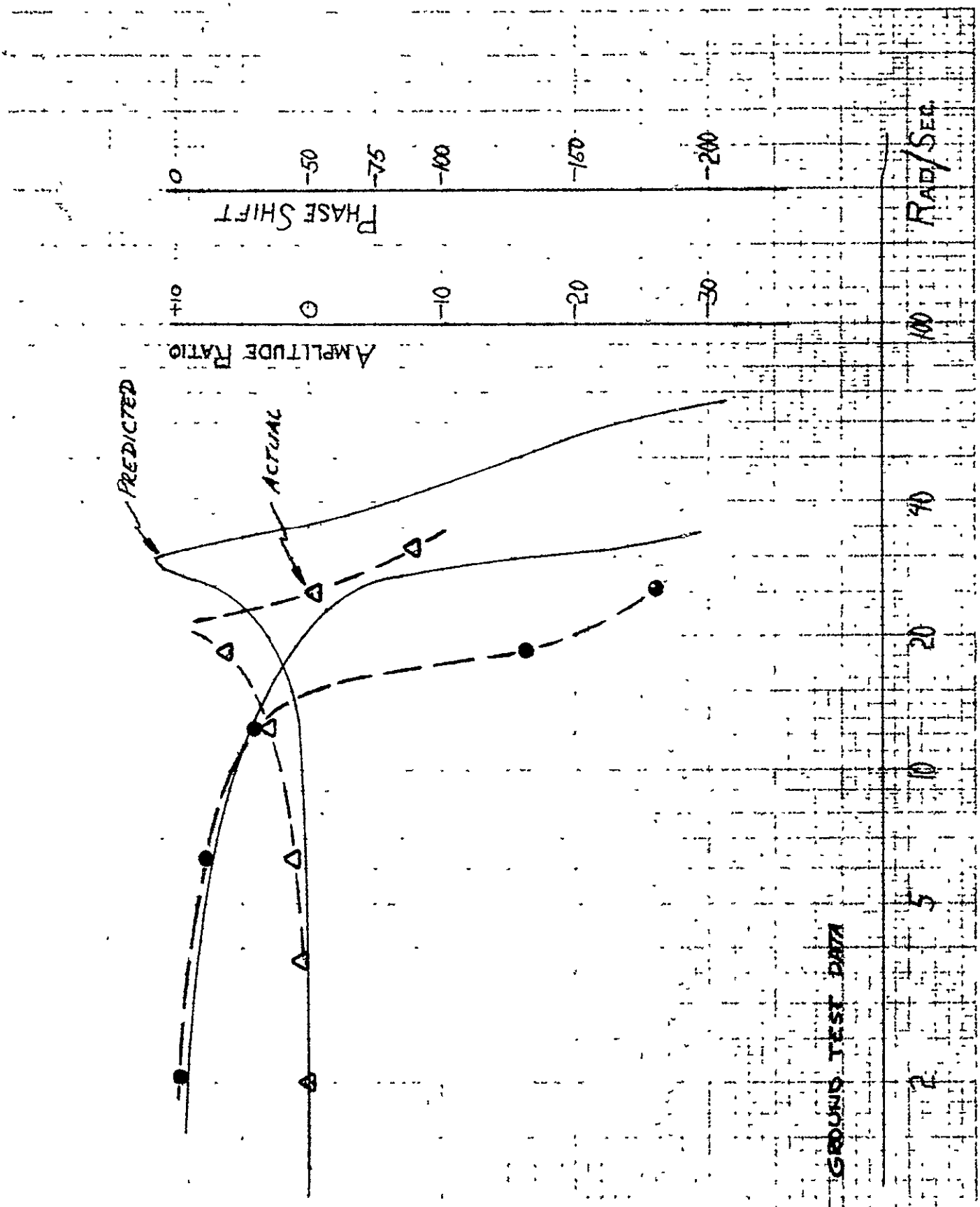
ELEVATOR SAS SERVO POSITION AUTHORITY AND GAIN

THE BOEING COMPANY

MOD C-8A

FIG 5.2-14



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APR					
CONTRACT NO.					

MOD C-8A

FIG 5.2-15

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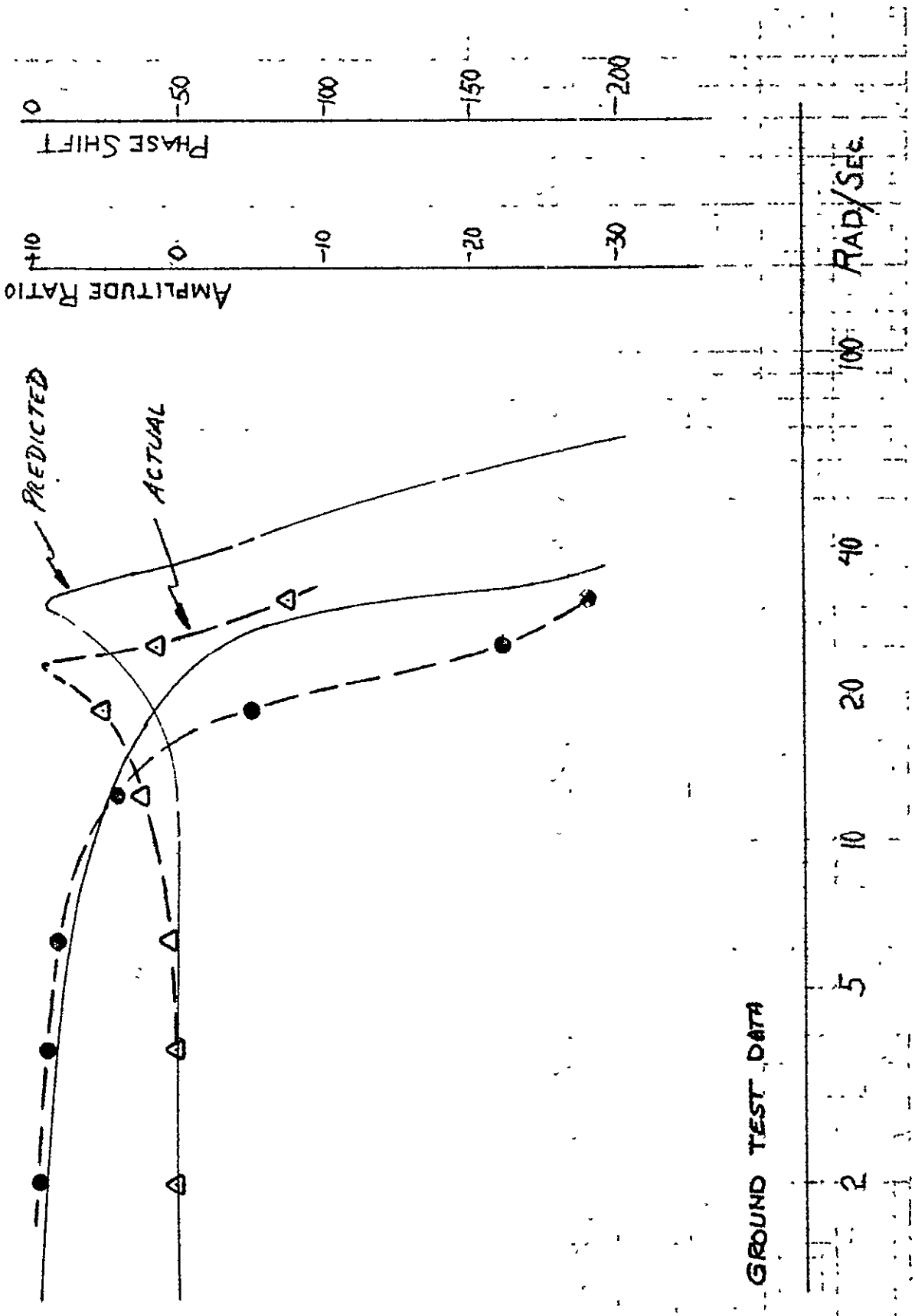

CALC			REVISED	DATE	ELEVATOR FREQUENCY RESPONSE DUAL HYDRAULICS THE BOEING COMPANY	MOD C-8A FIG 5.2-16 PAGE 79 D6-41489
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6.0 TAXI AND FLIGHT TESTS

The flight test program for the powered elevator modification on the Mod C-8A airplane was conducted by NASA-ARC with Boeing support at Moffett Field, California. Purpose of the tests was to demonstrate airworthiness and to determine the operating characteristics of the powered elevator system.

Flight testing consisted of one taxi test and five flight tests. Testing was conducted on the following dates:

Flight	Date	Purpose
Taxi	Aug 28, 1973	Taxi tests to 110 knots for flutter checks and control evaluation.
No. 68	Sept 19 "	Flight test to 120 knots
69	" 21 "	Flight test to 140 knots
70	" 21 "	Flight test to 160 knots
71	" 25 "	Flight test to 160 knots
72	" 26 "	Flight test to 180 knots

A three-week delay occurred between the taxi test and first flight (flight 68) due to an electrical system problem unrelated to the powered elevator installation. The flight test was planned for a minimum of four flights to allow flutter clearance in 20-knot speed increments. The test sequence generally followed on each flight was to initially conduct the flutter checks up to the maximum airspeed designated. Following this the control system evaluation was conducted for the remainder of the flight. Recorded flight test data were analyzed for critical parameters prior to permitting the next flight.

During the flight test program, the airplane was flown at airspeeds ranging between 49 knots (flaps 65 stall) and 180 knots IAS. Flight weights ranged from 46,000 pounds at takeoff to a minimum of 36,000 pounds at landing. The airplane center of gravity varied between 29% and 31% MAC. Variations in load factor from 0.1 g to 1.9 g were attained. The airplane was sideslipped to 12 degrees. Angle of attack varied between approximately -3 degrees and +26

degrees (flaps up stall). All tests were conducted below 10,000 foot altitude.

Pilot reports showed the powered elevator system to perform very satisfactorily throughout the flight envelope, with no adverse characteristics noted.

6.1 FLUTTER TESTS

The flight envelope was flutter cleared in 20 knot speed increments. On each flight, a range of speed points was tested in five knot increments. The airplane was excited by abrupt pilot inputs to the rudder and elevator, with pilot feel being relied on to detect excessive oscillations. Recorded data were analyzed after each flight to determine the damping levels and trends of the various modes of vibration versus airspeed. Clearance to higher speeds for the next flight was given when the results were found satisfactory.

The vertical tail, horizontal tail, rudder and elevator were instrumented to measure their response. A diagram showing the instrumentation is presented in figure 6.1-1. After the third flight, the accelerometers at the upper and lower portions of the vertical tail were relocated in the cockpit because of pilot comments about the vibration level. No problem was found.

The results of the flight flutter tests showed that the Modified C-8A airplane with powered elevator control system satisfied the flutter clearance speeds with adequate damping. The responses of the rudder, elevator and tabs are heavily damped for all the speed points. The two important empennage modes are antisymmetric modes at about 2.8 Hz (stabilizer vibrating in a roll sense), and 6.2 Hz (stabilizer vibrating in a yaw sense). Figures 6.1-2 and -3 show examples of the response to a rudder and an elevator kick respectively at 180 KIAS.

6.2 LONGITUDINAL CONTROL EVALUATION

The longitudinal control system was evaluated at the following flight conditions to verify satisfactory operation:

1. Takeoff and climb at 90 knots, flaps 30
2. Cruise at 120 knots, flaps up
3. Cruise at 160 knots, flaps up
4. Descent at 120 knots, flaps up
5. Holding at 90 knots, flaps 30
6. Approach at 90 knots, flaps 30
7. Approach at 60 knots, flaps 65

In addition to routine flight maneuvers, specific evaluation tasks consisted of sinusoidal column inputs, column step changes, rapid pitch attitude changes, sideslips, wind-up turns, pushover/pullups and stalls. The evaluation was conducted for both one and two hydraulic systems.

6.2.1 Elevator Control Power

Prior to first flight, the airplane was ballasted to obtain the same airplane gross weight/center of gravity schedule as previously flown with the spring tab elevator control system. This schedule is shown in figure 6.2-1.

The elevator required for steady 1 'g' trim points conducted during the flight test program is presented in figure 6.2-2. The trim points correspond to typical climb, level flight and descent conditions at flaps 5.6 flaps 30 and flaps 65. Major parameters are listed in table 1. As expected, elevator trim requirements were unchanged by the powered elevator modification.

Elevator maneuver requirements are well within the capabilities of the powered elevator control system. The maximum trailing edge up elevator occurred during the flare, where -19.1 degrees elevator trailing edge up was reached (flight 70, IRIG 23:55:47). Maximum trailing edge down elevator of +9.0 degrees was attained during ± 9.5 degrees per second pitch rate maneuvers at the STOL approach

TABLE 1

Flight No.	Time	Weight x 10^3	Flaps	Percent N_H	δ_N	h	γ	V_e	δ_e
72	21:57:57	45.2	6	97.2	6	5210	+4.4	127	-4.3
70	23:21:00	41.5	6	92.4	6	5777	+2.4	96	-7.2
70	23:21:39	41.4	6	92.4	6	6058	+2.5	88	-8.9
69	16:15:20	42.9	6	92.7	6	4920	0.6	123	-5.4
	16:54:20	42.6	6	92.7	6	5079	-0.4	133	-4.9
	17:00:40	41.8	6	94.0	6	5156	0.4	144	-3.8
70	23:22:14	41.4	6	92.4	6	6329	0.9	75	-10.2
	23:22:55	41.3	6	92.4	6	6385	-0.3	69	-12.8
72	22:01:07	44.8	6	94.8	6	6990	-0.4	164	-2.9
	22:03:52	44.5	6	96.6	6	6412	0.8	172	-3.1
	22:06:30	44.0	6	96.6	6	6835	0.6	175	-2.8
	22:12:46	43.1	6	99.7	6	7408	0.2	191	-2.5
70	23:23:35	41.2	6	92.4	6	6345	-2.3	66	-17.0
72	22:16:46	42.6	6	86.9	6	5996	-8.2	179	-3.4
69	16:46:56	43.7	30	97.0	6	2978	5.1	95	-0.6
	16:48:20	43.2	30	97.2	6	4384	6.3	82	-1.2
	16:49:40	43.1	30	97.2	6	5171	3.4	106	-0.6
70	22:59:35	43.9	30	93.8	6	5870	1.6	97	-1.2
72	21:55:07	45.6	30	97.0	6	2615	2.2	107	-1.2
	21:55:43	45.5	30	97.0	6	3187	5.1	86	-2.4

Flight No.	Time	Weight x 10^3	Flaps	Percent N_H	δ_N	h	γ	V_e	δ_e
69	17:06:18	41.3	30	94.1	6	5423	+1.3	105	-1.5
	17:07:28	41.2	30	93.3	6	5590	0.6	97	-2.7
	17:09:11	41.1	30	92.6	6	5541	-0.1	84	-4.4
72	22:31:10	40.8	30	93.8	6	7118	-0.8	98	-2.7
69	17:10:28	40.8	30	90.2	6	5150	-3.3	84	-6.2
	17:12:34	40.6	30	90.2	6	4404	-3.8	95	-4.7
	17:13:54	40.4	30	91.9	6	3880	-1.6	107	-2.4
72	22:22:28	41.9	30	94.5	6	7043	-1.9	106	-2.2
	22:33:50	40.5	80	87.6	6	5342	-5.6	96	-5.9
72	22:38:20	40.0	65	99.9	6	6985	+1.5	63	3.1
69	17:47:39	36.6	65	93.5	89	4003	-10.3	72	1.8
72	22:37:30	40.0	65	97.0	6	6838	-1.3	63	1.8
	22:40:32	39.7	65	93.8	70	6446	-8.1	67	0.7
	22:40:57	39.6	65	93.7	70	5891	-7.2	81	3.4
	22:41:48	39.5	65	93.7	70	5110	-10.7	56	-0.8
	22:42:32	39.4	65	93.5	70	4213	-7.0	78	3.0

condition (flight 69, IRIG 17:27:10). Typical elevator maneuver requirement time histories are shown in figures 6.2-3, -4, -5, and -6. A summary of maneuver requirements is shown in figure 6.2-7.

Elevator per 'g' data obtained from wind-up turns at constant airspeed is shown in figure 6.2-8. Test results show no change due to the powered elevator modification.

6.2.2 Elevator Authority

The elevator is position limited to -25 degrees and +15 degrees surface deflection at no-load. During the taxi tests, deflections of -24.5 degrees and +14 degrees were demonstrated at a speed of 62 knots (IRIG 21:26:16).

Elevator reversals with one hydraulic system off were conducted at 106 knots in order to verify the predicted elevator authority at high speed; blowdown was not reached, however. The elevator hinge moment coefficient was therefore derived from elevator actuator loads measured during flight test and used to calculate a blowdown curve. The predicted and calculated hinge moment coefficient is shown in figure 6.2-9. The curve shows a considerably lower coefficient than predicted for large trailing edge up deflections. This results in an elevator blowdown speed of 170 knots instead of 100 knots, as shown on figure 6.2-10. It should be noted that the blowdown speed is further increased by any pilot effort transmitted to the elevator surface after elevator PCU blowdown is reached.

The increased elevator deflection capability at high speed could produce excessive tail loads. Figure 6.2-11 shows the horizontal tail load design envelope. Loads due to an abrupt elevator input of -25 degrees are shown for various airspeeds up to 170 knots. The structural design of the empennage is adequate for abrupt elevator maneuvers up to the design maneuver speed V_A (136 knots for the Mod C-8A). This will satisfy the requirements of FAR 25. However, full trailing edge up elevator above this speed would result in tail loads in excess of design loads.

Because of the relatively low pilot effort required to attain full trailing edge up elevator above 136 knots two recommendations are made:

1. Pilots should be cautioned not to make large and rapid elevator control inputs above 120 knots. Control forces should be kept below 40 pounds pilot effort.

2. The effect of reducing the elevator PCU supply pressure from its present value of 2440 psi to 1500 psi should be investigated and, if found acceptable, implemented. This would limit elevator deflection due to actuator output to a safe value, as shown in figure 6.2-10. Maximum elevator no-load rate would however be reduced from 54 degrees per second to 42 degrees per second.

The reduced supply pressure will only limit elevator deflections to a safe value if no pilot effort is transmitted to the elevator surface. Since this is a possibility once actuator blowdown is reached, the cautionary note should be retained regardless of any reduction in hydraulic supply pressure.

6.2.3 Control System Gearing

The elevator-to-column deflection at different airspeeds is shown in figure 6.2-12. The gearing decreases with increasing airspeed due to increased cable stretch and a reduction in the steady-state gain of the elevator PCU. At 62 knots the elevator-to-column gain between 0 and -10 degrees elevator is 1.65 deg/deg. This gain is reduced to 1.35 deg/deg at 180 knots. A greater reduction had been anticipated as a result of the engineering ground test data (figure 5.2-2) but did not occur due to the lower elevator hinge moment.

The geared tab-to-elevator deflection is shown in figure 6.2-13. The gearing is reduced from a nominal δ_t/δ_e = -0.6 at 62 knots to δ_t/δ_e = -0.45 at 180 knots. The reduction in balance ratio is of little significance since elevator control power and manual reversion forces are satisfactory.

The series SAS actuator was not activated during the flight test program. SAS actuator gearing could therefore not be checked.

6.2.4 Control System Feel Characteristics

Feel and centering characteristics were considered satisfactory by the pilots. Specific comments were that at 90 knots and 120 knots,

breakout forces were not noticeable and system centering was good. Feel gradients were found compatible with one-handed operation. Figures 6.2-14 and -15 show typical column force-elevator characteristics. Breakout forces are approximately 7.5 pounds, compared to 5.5 pounds obtained during the engineering ground tests. This difference is primarily due to a misaligned cable fairlead near body station 510.

The feel gradient characteristic obtained in flight is lower than predicted but is similar to ground test results. The variation in feel gradient between push and pull noted during the ground tests was not evident in the flight test data at the higher speeds primarily because the elevator is trimmed in a more linear region and elevator deflection are necessarily limited in magnitude. Figure 6.2-16 summarizes the feel gradient characteristic. The figure also shows stick force per 'g' data obtained from wind-up turns to vary between 30 pounds/'g' and 40 pounds/'g'. Calculated feel force characteristics based on the feel gradient data of figure 6.2-16 are shown in figure 6.2-17.

Feel computer operation from a pilot viewpoint was found to be satisfactory in that no feel force anomalies were noted. During the engineering ground tests, two discrepancies were noted:

1. A sustained feel computer oscillation could be induced in the "B" computer by cycling the hydraulic supply pressure at simulated airspeeds above 100 knots.
2. The elevator feel pressure warning light would light momentarily when cycling the elevator.

Neither of these items was found to occur during the flight test program.

The feel computer static gain is shown in figures 6.2-18 and -19. Both computer gains are on the high side, with 17.7 psi/psf and 16.3 psi/psf for computer A and computer B respectively compared to a nominal gain of 15.2 psi/psf.

In the steady state computers A and B track each other adequately, with the 'A' system feel pressures nominally 40 psi higher than the 'B' system pressures.

No feel pressure variations with pegasus nozzle position, flaps, one or two hydraulic systems, angle of attack or sideslip were noted. Any variation is small enough to be masked by the normal computer repeatability. For computer 'A' this is approximately 70 psi, for computer 'B', 40 psi. The better performance of the 'B' system appears to be due to lower valve spool friction.

6.2.5 Control System Dynamics

Control system dynamics with one or two hydraulic systems were judged satisfactory by the pilots. Release from control column steps produced one column overshoot at low speeds (60-90 knots) and two overshoots at higher airspeeds. The overshoots were not considered objectionable as the airplane did not appear to respond to them. Also, normal pilot activity added sufficient damping to eliminate overshoots.

The control system response to a column release at 127 knots and a "sinusoidal" column input at 60 knots is shown in figure 6.2-20. The control column at 127 knots has a natural frequency of 18.2 rad/sec and a damping ratio of approximately 0.35. The "sinusoidal" response at 60 knots shows the elevator position to lag column force by 13 degrees at a frequency of 1.1 radians/second. The phase lag did not vary significantly with airspeed. Control system phase lag at the "short period" frequency (approximately 0.8 radians/second at 60 knots and 3.0 radians/second at 160 knots) is therefore small enough to not affect airplane controllability.

6.2.6 Elevator Rate

The elevator rate was found to be satisfactory throughout the flight envelope. Rate saturation appears to have occurred for only one condition. This was an elevator reversal at 100 knots with one hydraulic system shut off.

Maximum elevator rates commanded during large maneuvers were as follows:

Condition	Elevator Rate (Deg/Sec)
Takeoff rotation	27°/S
Ten degree pitch attitude changes at 60 kts	34°/S
Landing flare	38°/S

6.2.7 Elevator Powered Trim System

The powered trim system has sufficient authority to trim the airplane for hands-off flight over the total flight envelope. The single speed trim rate was considered an acceptable compromise between high speed and low speed trim requirements. Trim runaways at any airspeed are easily overridden by the pilot and represent no safety problem.

6.2.8 Hydraulic Power Off Transients

Disconnect transients due to loss of hydraulic power were determined by monitoring elevator PCU loads during climb, level flight and descent conditions at various airspeeds.

Taxi test results showed the elevator float angle to be more positive than predicted. Since this directly affected the potential disconnect transients, a new trim tab setting was determined. The initial trim tab setting for the taxi tests and flight 68 was +1 degree trailing edge down. This was subsequently changed to +4 degrees for flights 69 and 70 and to +3.5 degrees for flight 72. All data has been corrected for this trim tab setting.

Results show that pilot forces between 37 pounds pull and 30 pounds push are required to maintain 1 'g' flight at any "normal" flight condition (figure 6.2-21). For flaps 30 flight conditions between 80 knots and 100 knots IAS, disconnect transients can vary between 30 pounds push for climb conditions and 20 pounds pull for descent conditions. During the manual reversion evaluation at 90 knots, flaps 30, a 16 pound push force was required to maintain trim. The force transient was gradual (approximately two seconds) and was considered mild by the pilot. This disconnect transient was zero when the trim tab was changed from +3.35 degrees (+3.5 degrees measured on the ground) to +2.95 degrees.

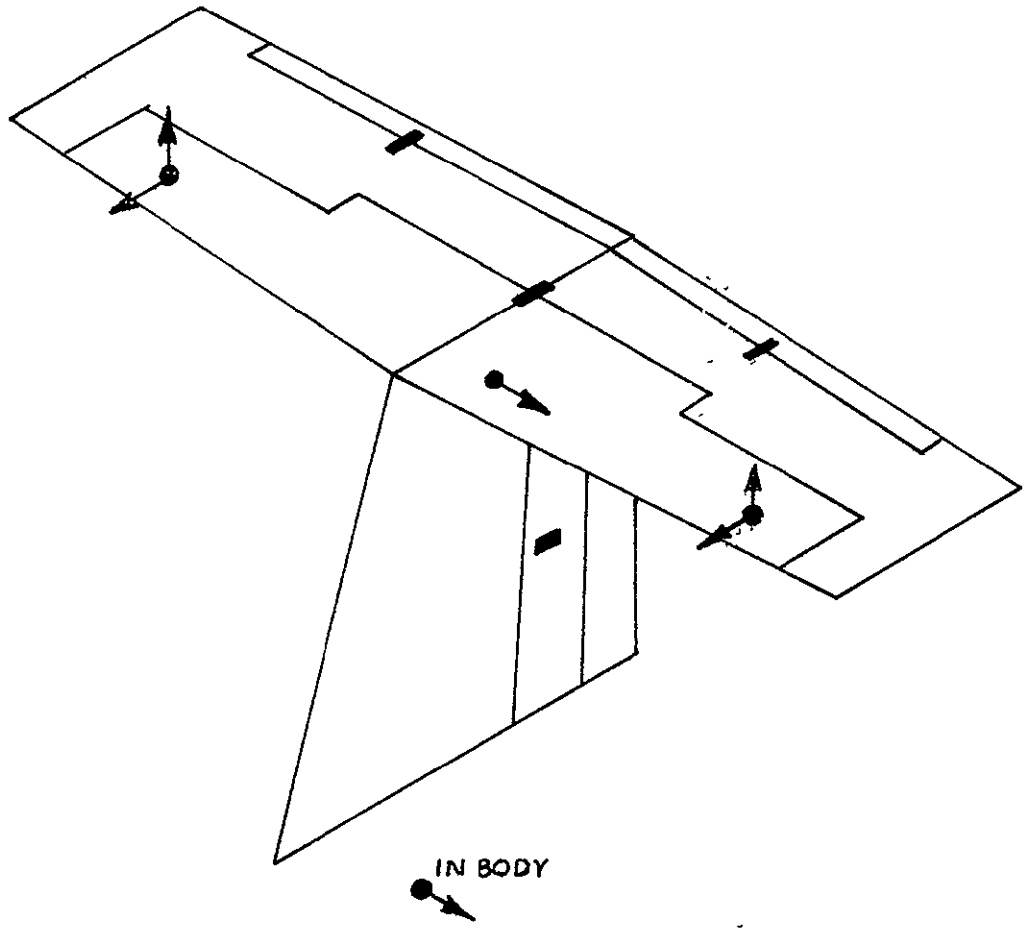
The present trim tab rigging (+3.5 degrees) is considered satisfactory. Disconnect transients over the total flight envelope are relatively small and should result in no hazardous condition.

6.2.9 Manual Reversion

Manual reversion was evaluated for approximately five minutes at a nominal 90 knots, flaps 30 flight condition. Control was deemed adequate for continued safe flight to a landing. The most prominent characteristic is a +3.2 degree elevator deadzone (equivalent to ±1.2 inches column travel). Stick forces are higher than for powered operation but not objectionably so. Calculated manual reversion forces, based on the hinge moment coefficient of figure 6.2-9 are shown in figure 6.2-22. For "small" elevator deflections, forces agree closely with predicted values as can be seen by comparison with figure 4.4-10. Control forces are lighter than predicted for large elevator deflections. Satisfactory operation of the trim tab was also demonstrated.

6.3 RUDDER CUT-OUT EFFECT

The effect of the 1.5 ft^2 rudder cut-out on rudder control power was investigated by doing sideslips at 65 knots F65 and 90 knots F30. No detrimental effect was noted when compared against flight test results made prior to the modification, as shown in figures 6.2-23 .



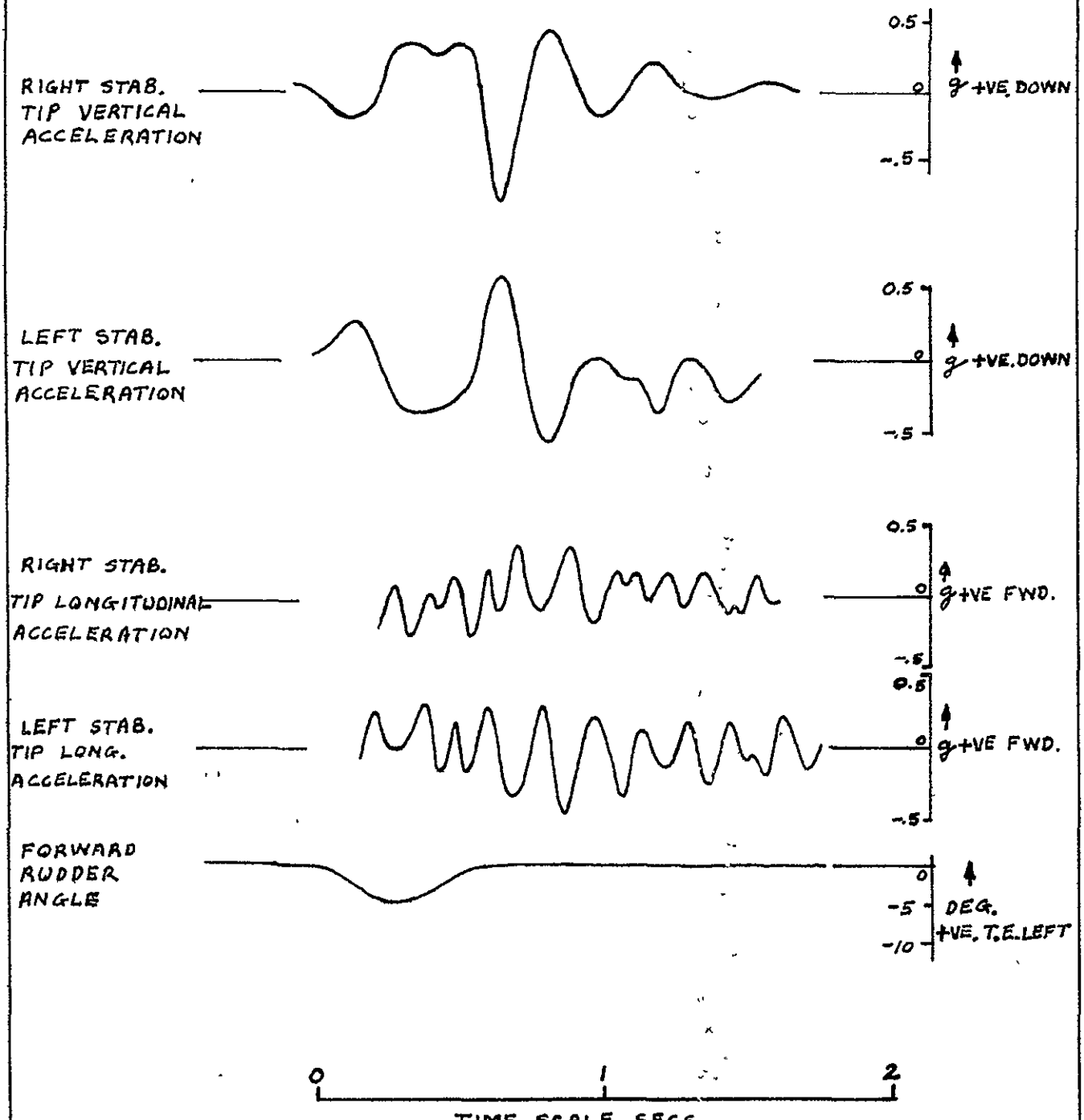
ACCELEROMETER READING IN DIRECTION OF ARROW

CONTROL SURFACE ROTATIONAL DISPLACEMENT

CALC	R.J.F.	10/73	REVISED	DATE	BUFFALO, C-8A MODIFIED WITH POWERED ELEVATOR FLIGHT FLUTTER TEST INSTRUMENTATION	MOD C-8A.
CHECK						FIG 6.1-1
APPD					THE BOEING COMPANY	PAGE 01 D6-41489
APPD						

BOTH HYDRAULIC SYSTEMS OPERATING

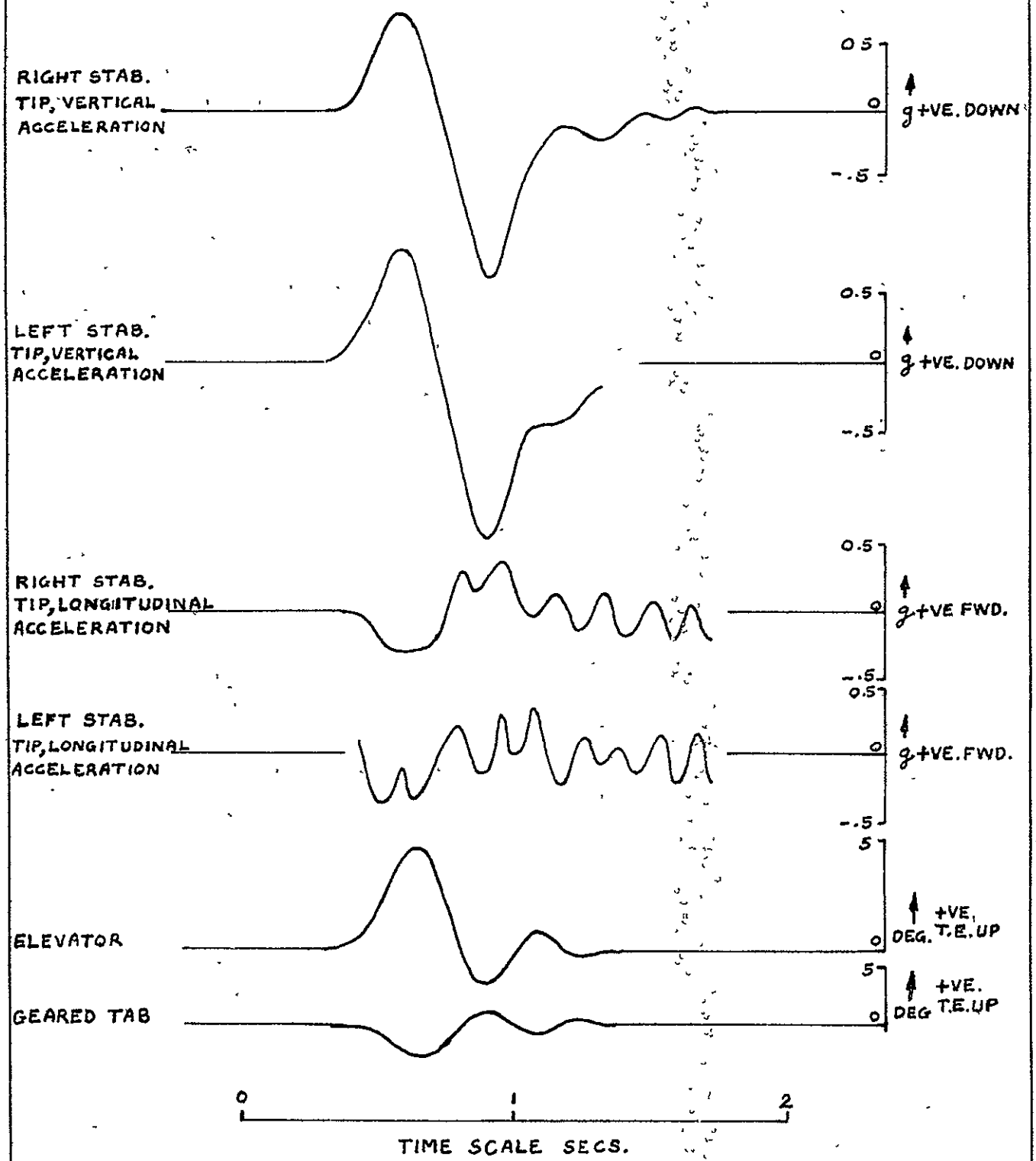
VELOCITY = 180 KNOTS INDICATED



CALC	R.J.F.	5/11/73	REVISED	DATE	BUFFALO C-8A MODIFIED WITH POWERED ELEVATOR FLIGHT FLUTTER TEST RESULTS	MOD C-8A
CHECK						FIG 6.1-2
APPD						
APPD					THE BOEING COMPANY	PAGE 92 D6-41489

BOTH HYDRAULIC SYSTEMS OPERATING

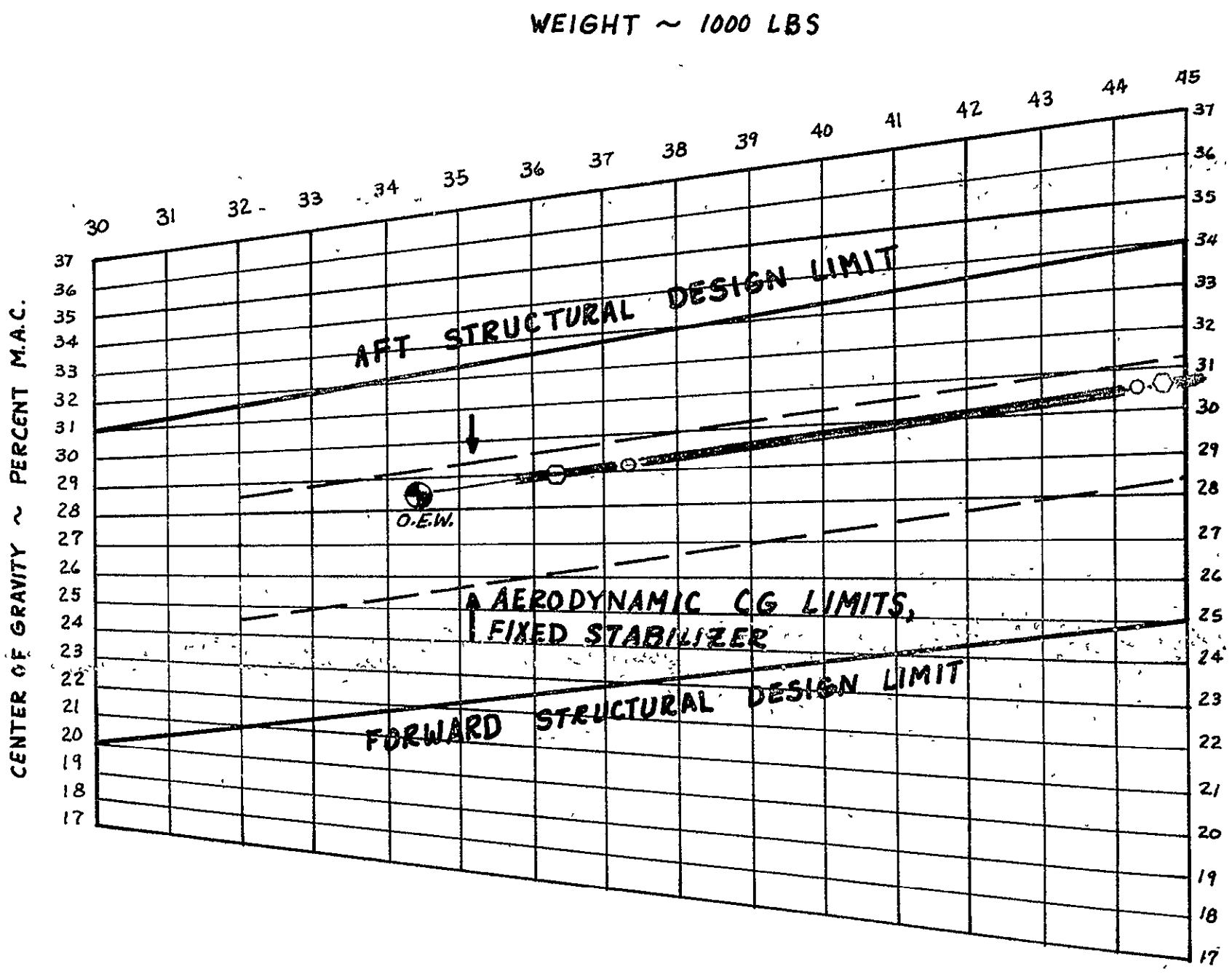
VELOCITY = 180 KNOTS INDICATED



CALC	R.J.F.	6/11/73	REVISED	DATE	BUFFALO C-8A MODIFIED WITH POWERED ELEVATOR FLIGHT FLUTTER TEST RESULTS	MOD C-8A
CHECK						FIG 6-1-3
APPD						
APPD					THE BOEING COMPANY	PAGE 93 D6-41489

D1 4100 5040 ORIG 9/71

J15-047



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DATE			

BOEING

GROSS WEIGHT VS C.G.

MOD C-8A

FIG. 6.2-1

D6-41189
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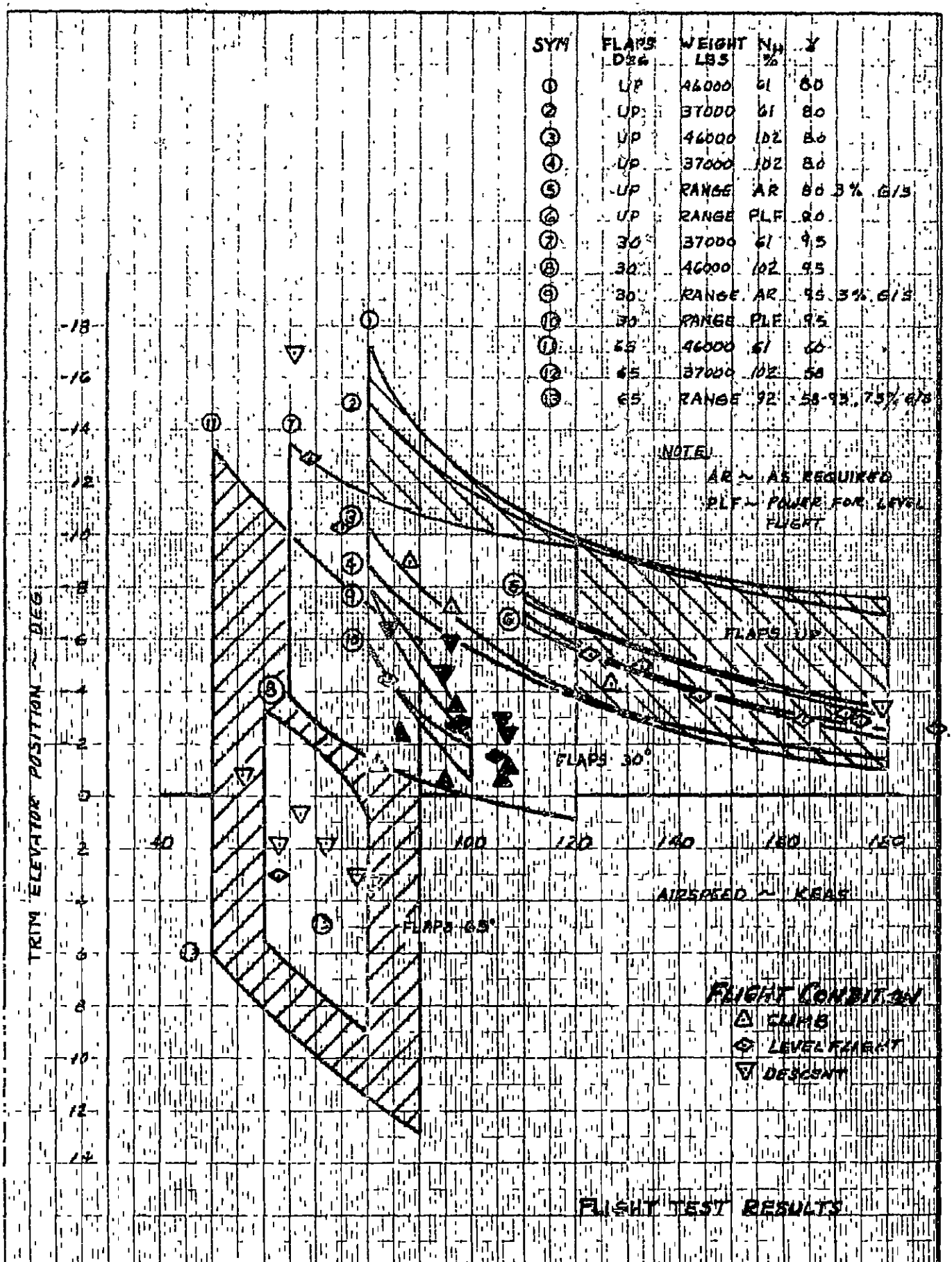

CALC			REVISED	DATE	ELEVATOR TO TRIM OVER FLIGHT ENVELOPE
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APR					
APR					
					THE BOEING COMPANY

MOD C-6A

FIG 6-2-2

U.C. ALBANENE 1961 8 5 6 8000

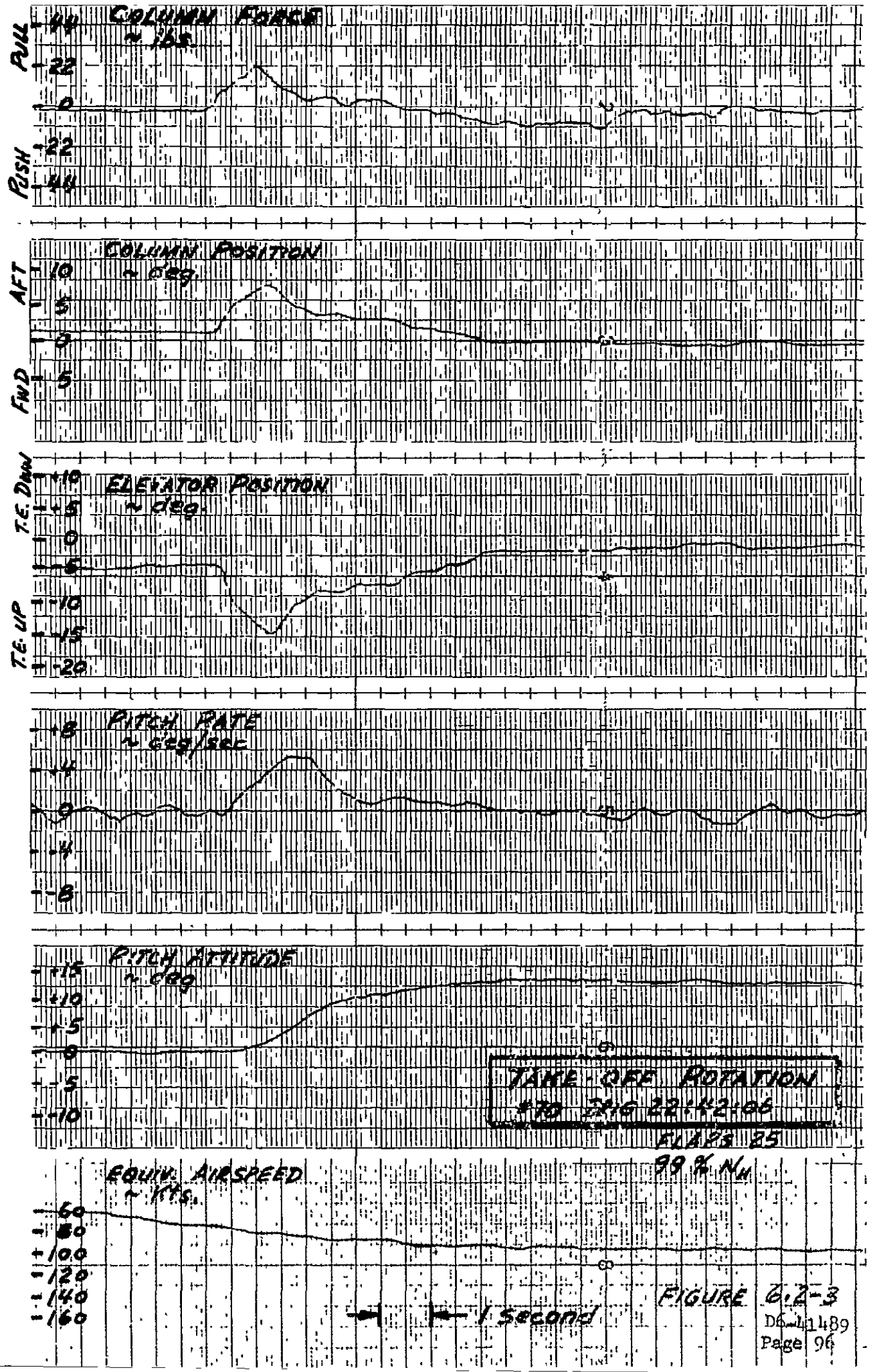

COLUMN FORCE
~ lbs.
PULL
PUSH
44
22
0
-22
-44
COLUMN POSITION
~ deg.
AFT
FWD
10
5
0
5
ELEVATOR POSITION
~ deg.
T.E. DWN
T.E. UP
+10
+5
0
-5
-10
-15
-20
PITCH RATE
~ deg/sec
+8
+4
0
-4
-8
PITCH ATTITUDE
~ deg
+15
+10
+5
0
-5
-10
TAKE-OFF ROTATION
FLAPS 25
99 % N_H
EQUIV. AIRSPEED
~ Kts.
60
80
100
120
140
160
1 second

FIGURE 6.2-3

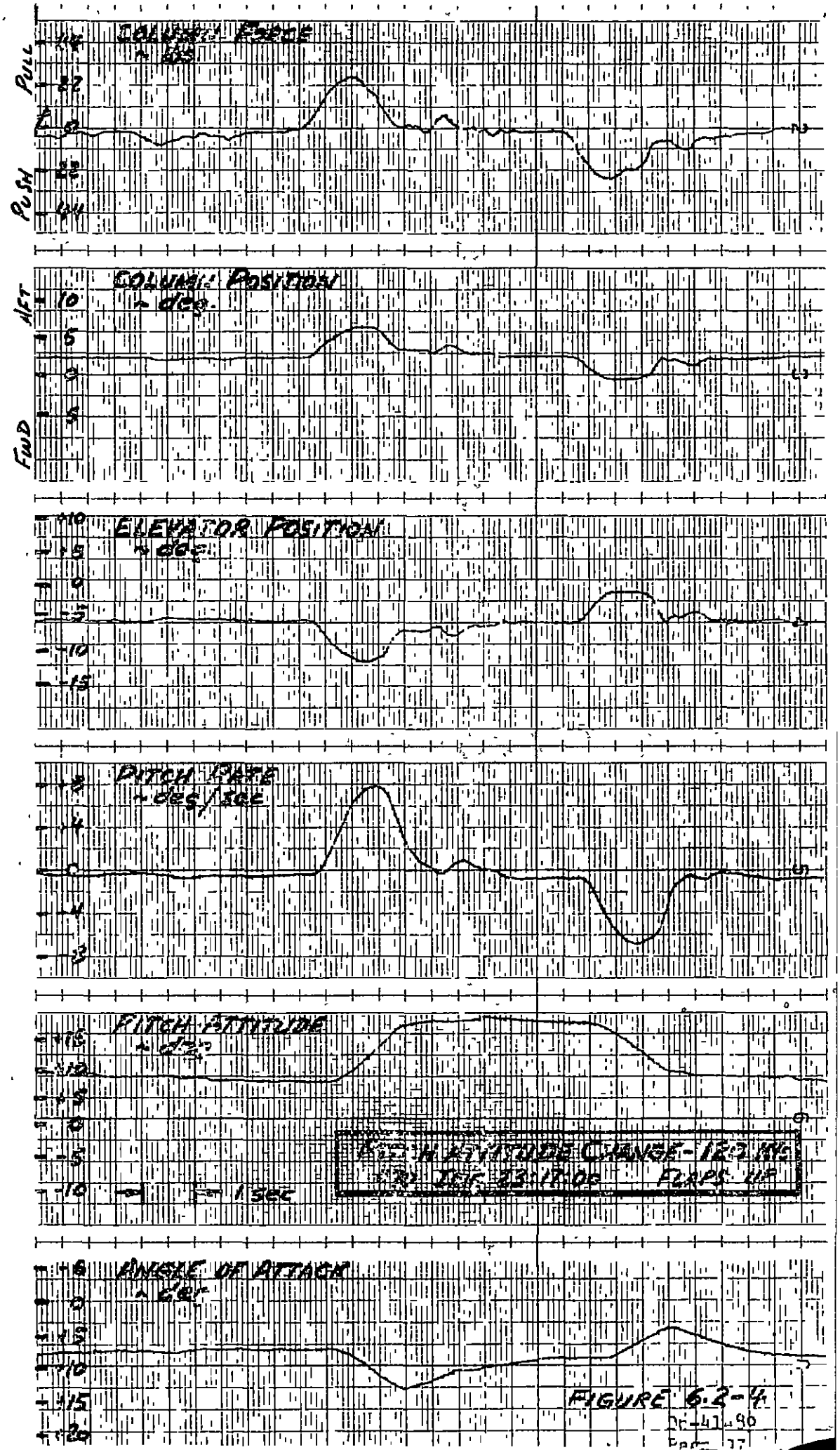

COLUMN FORCE
~ lbs
PULL
PUSH
COLUMN POSITION
~ deg.
AFT
FWD
ELEVATOR POSITION
~ deg.
PITCH RATE
~ deg/sec
PITCH ATTITUDE
~ deg
1 sec
ANGLE OF ATTACK
~ deg

FIGURE 6.2-4

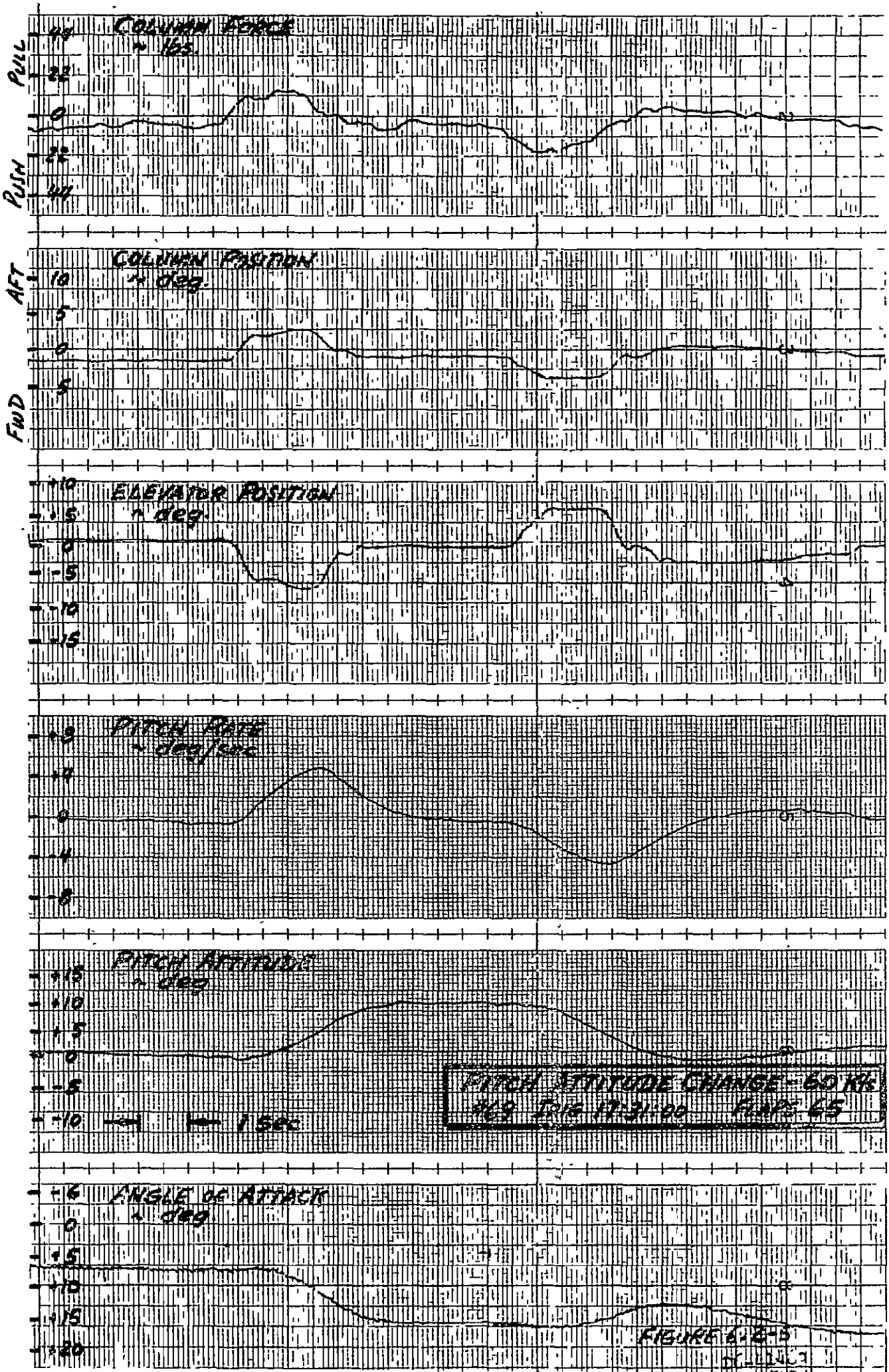

COLUMN FORCE
± lbs.
PULL
PUSH
COLUMN POSITION
± deg.
AFT
FWD
ELEVATOR POSITION
± deg.
PITCH RATE
± deg/sec
PITCH ATTITUDE
± deg
1 SEC
PITCH ATTITUDE CHANGE - 60 KTS
ANGLE OF ATTACK
± deg

FIGURE 6.2-5

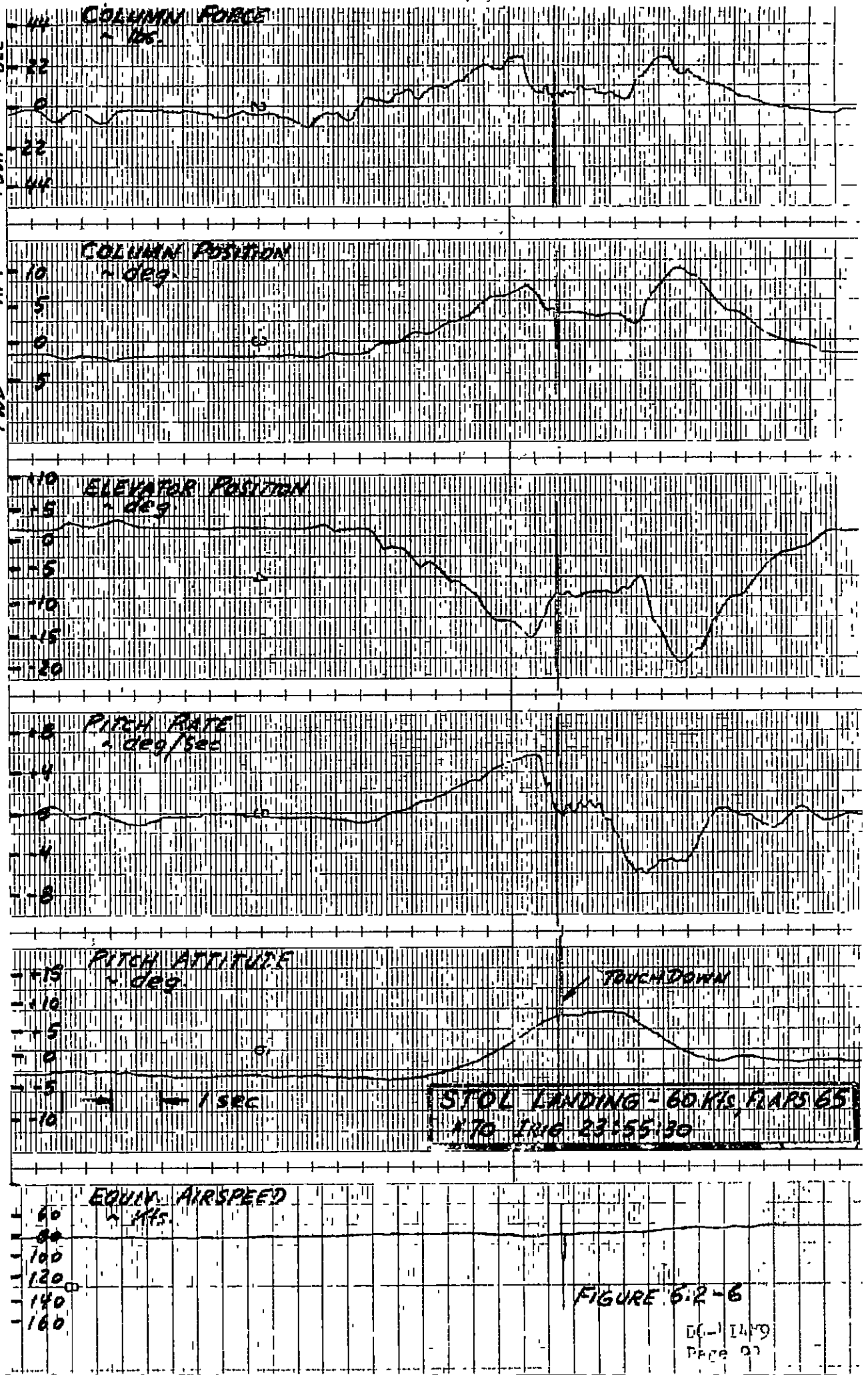
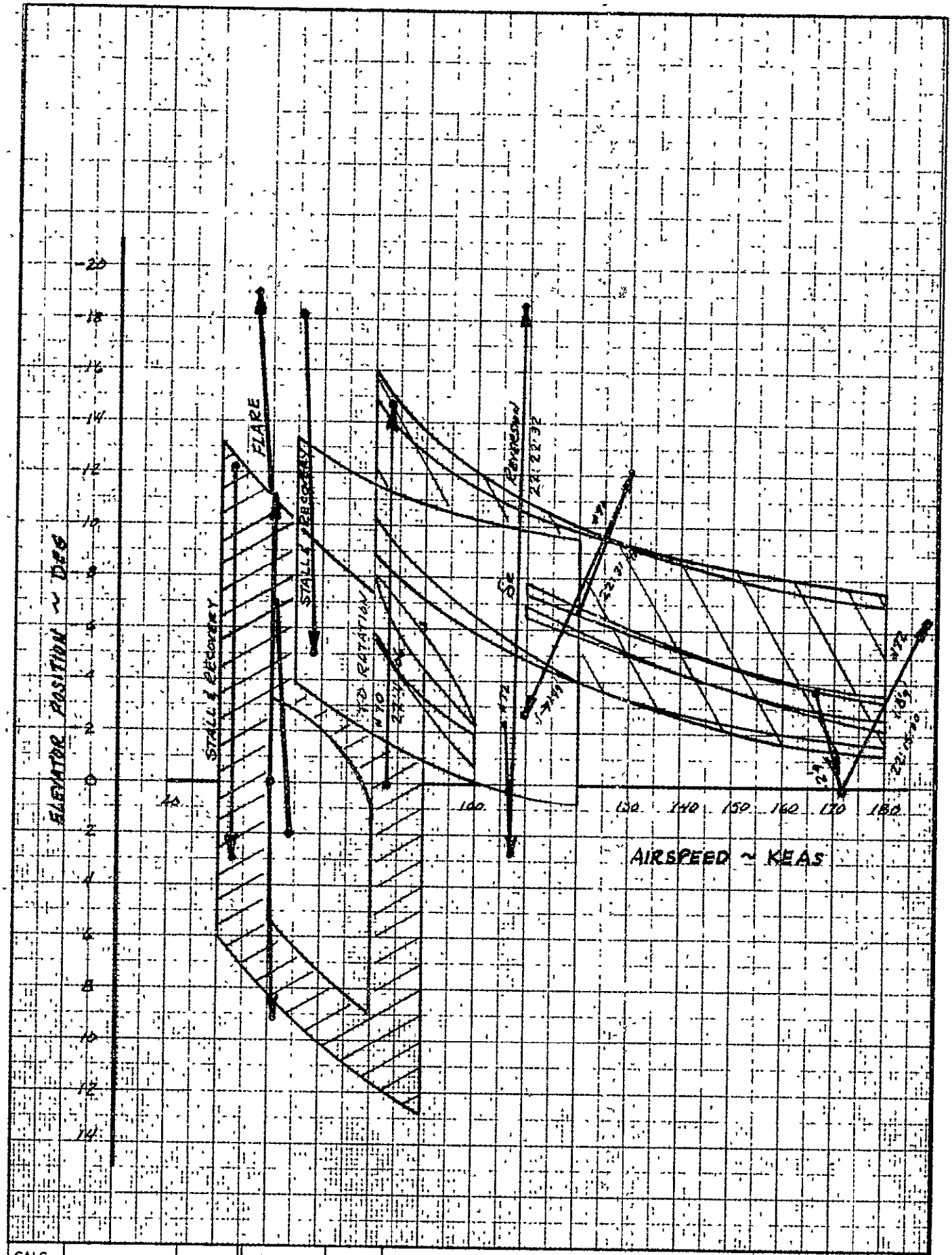

COLUMN FORCE
~ lbs.
PULL
PUSH
44
22
0
22
44
COLUMN POSITION
~ deg.
AFT
FWD
10
5
0
5
ELEVATOR POSITION
~ deg.
+10
+5
0
-5
-10
-15
-20
PITCH RATE
~ deg/sec
+8
+4
0
-4
-8
PITCH ATTITUDE
~ deg.
+15
+10
+5
0
-5
-10
TOUCHDOWN
1 sec
STOL LANDING - 60 Kts, FLAPS 65
TO IRIG 23:55:30
EQUIV. AIRSPEED
~ Kts.
60
80
100
120
140
160

FIGURE 6.2-6



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MANEUVER REQUIREMENTS
FLIGHT TEST RESULTS

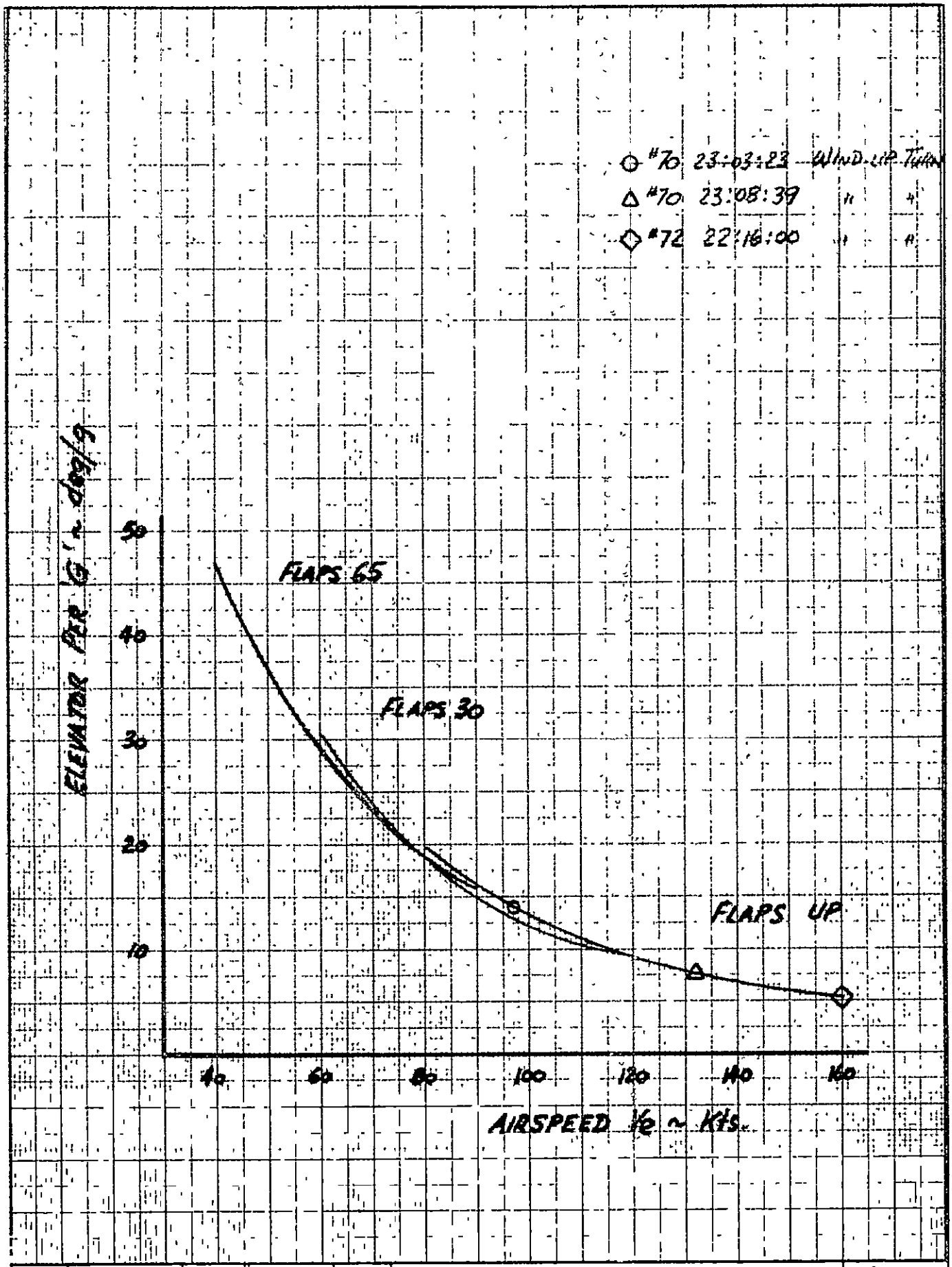
THE BOEING COMPANY

MOD. C-8A

FIG. 6.2-7

PAGE 130

AD 161 C 86



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ELEVATOR PER 'G'

THE BOEING COMPANY

MOD C-8A

FIG 6-2-8

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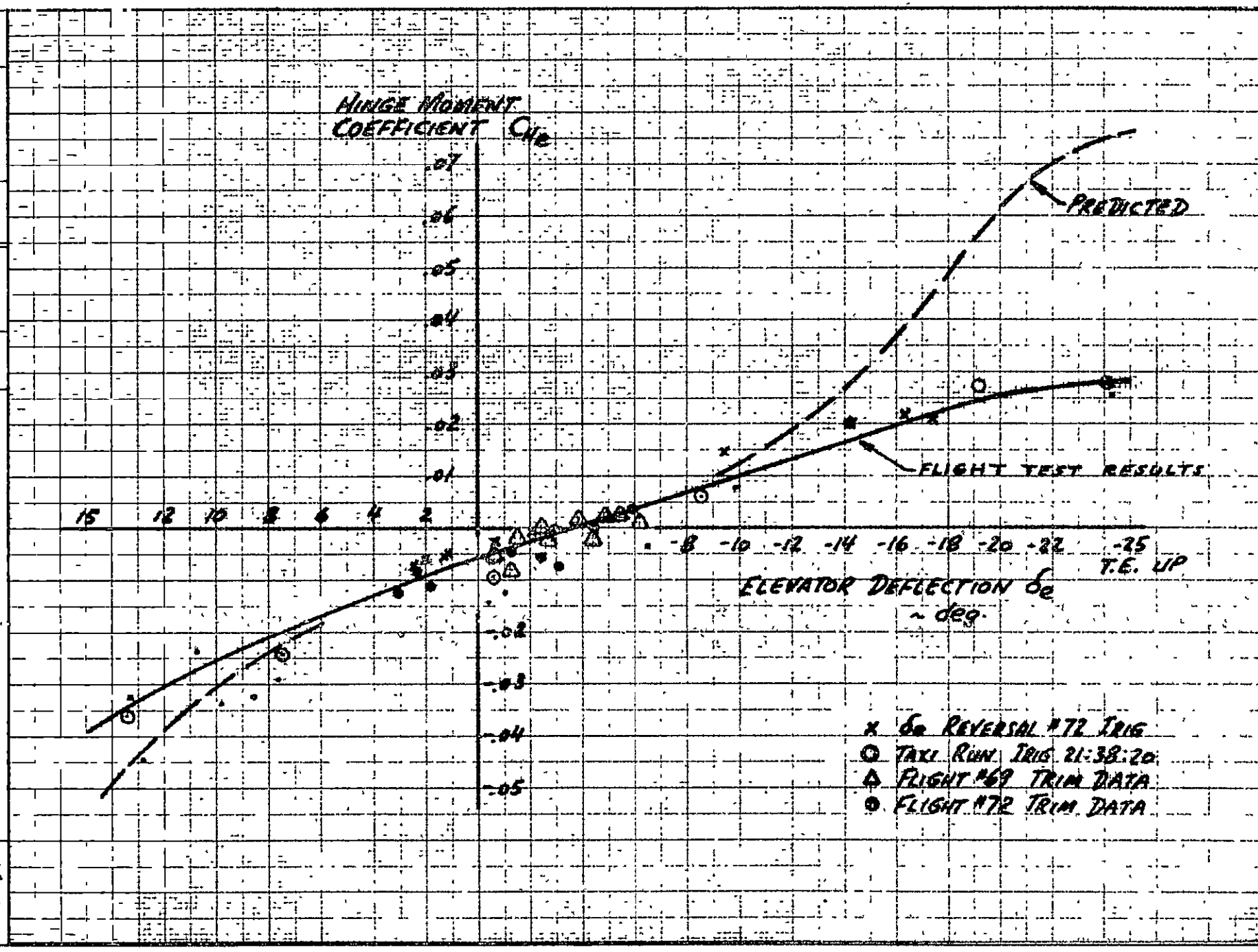

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ELEVATOR HINGE MOMENT COEFFICIENT

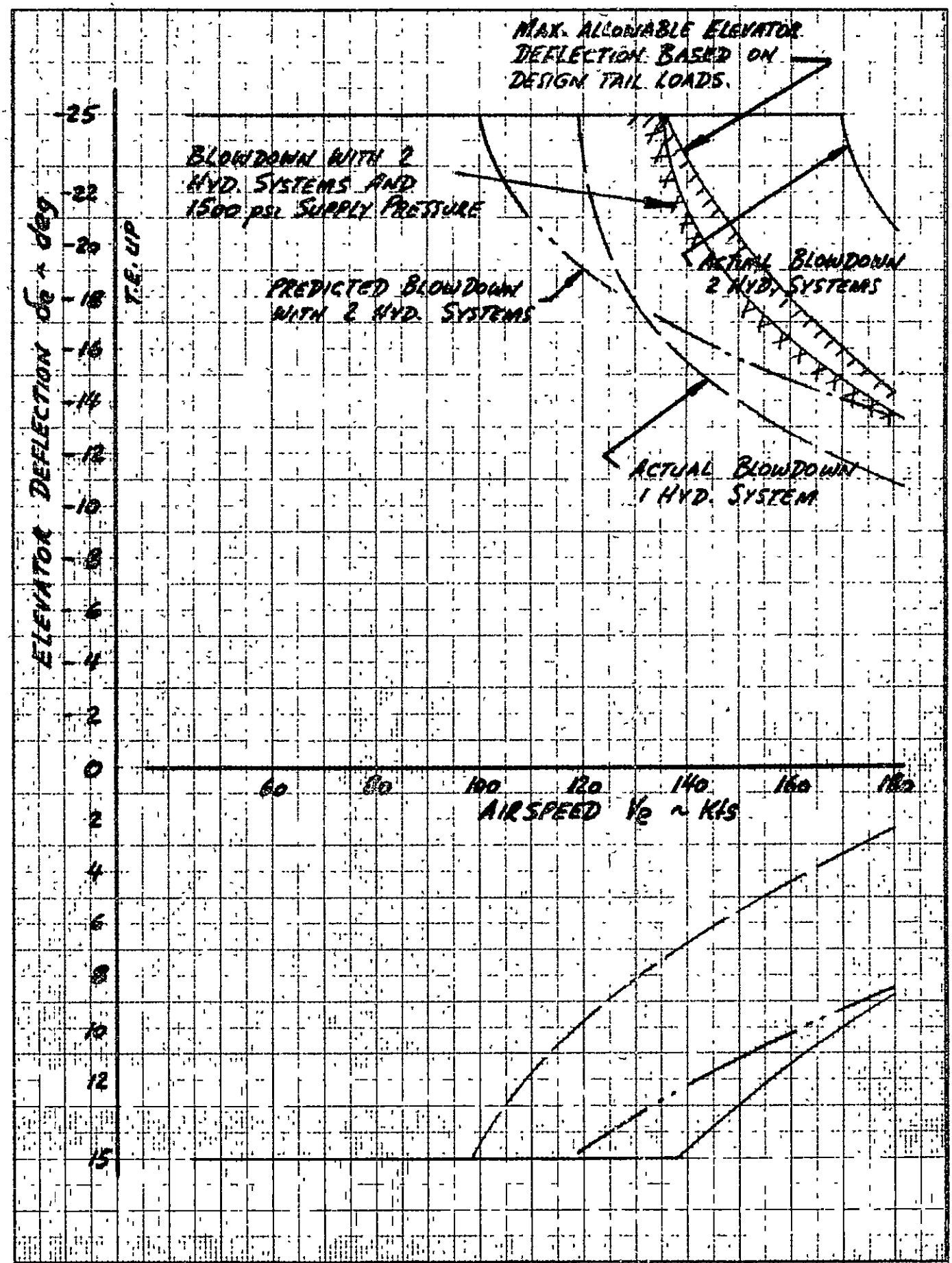
THE BOEING COMPANY

MOD C-8A

FIG 6.2-9

PAGE 102

AD 461 C R6
K+E ALBANENE 195L 8 5 G 8000



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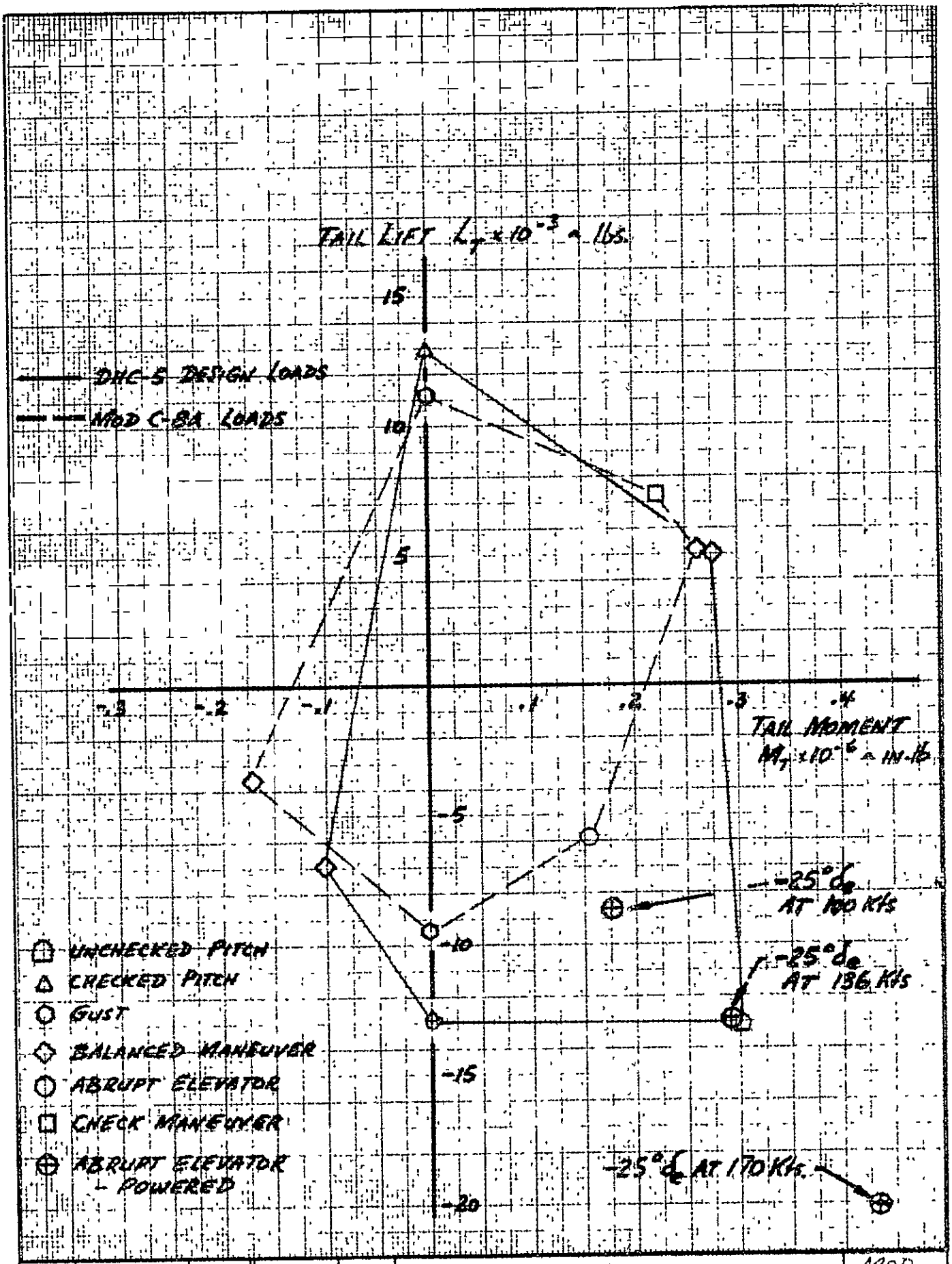
ELEVATOR BLOWDOWN

MOD C-8A

FIG 6.2-10

THE BOEING COMPANY



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HORIZONTAL TAIL LOADS

MOD C-8A

FIG. 6-2-11

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THE BOEING COMPANY

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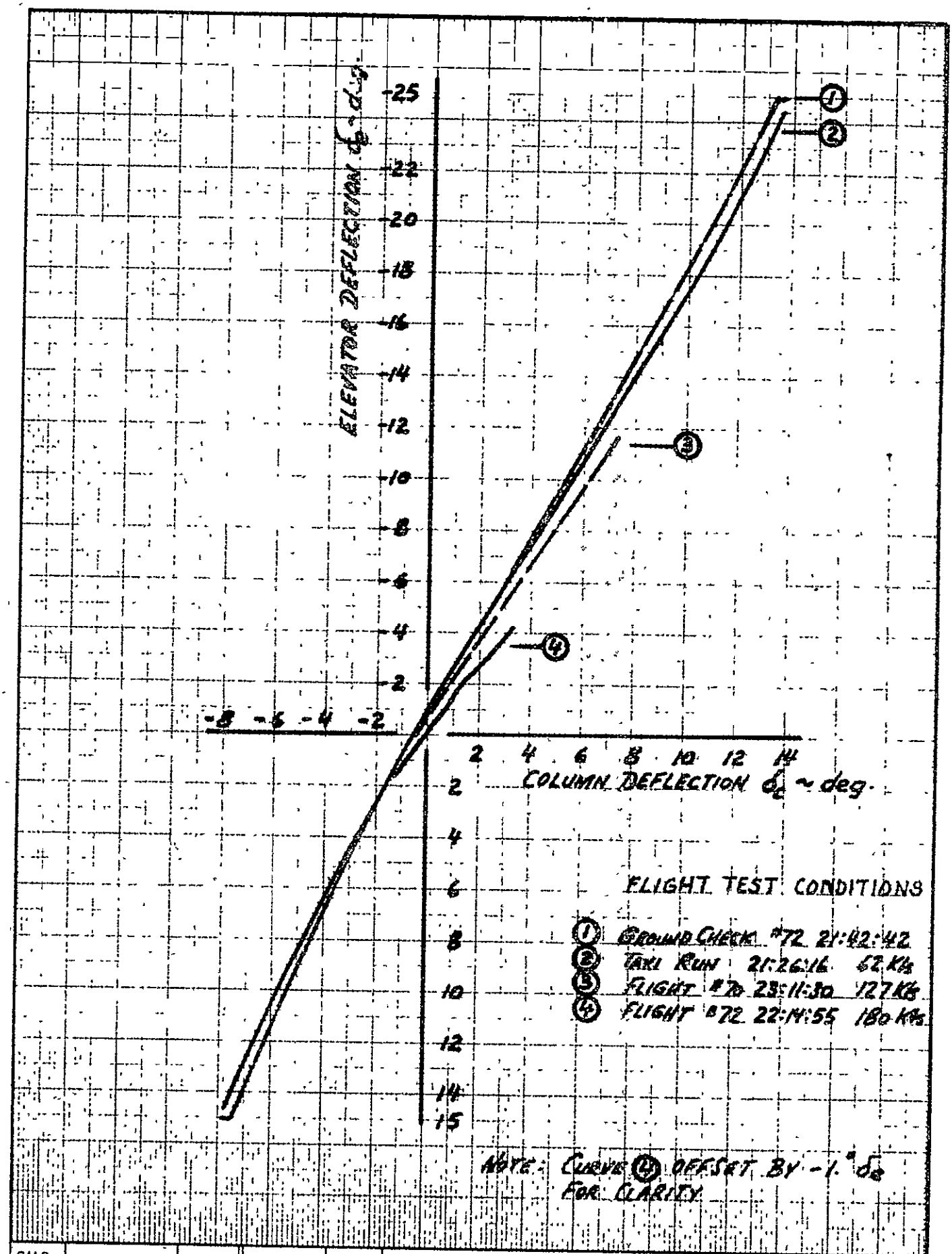
K-E ALBANENE 1951 TRACING PAPER



FLIGHT TEST CONDITIONS

1. GROUND CHECK #72 21:42:42
2. TAXI RUN 21:26:16 62 KIAS
3. FLIGHT #70 23:11:30 127 KIAS
4. FLIGHT #72 22:14:55 180 KIAS

NOTE: CURVE (4) OFFSET BY -1.° δ_e FOR CLARITY

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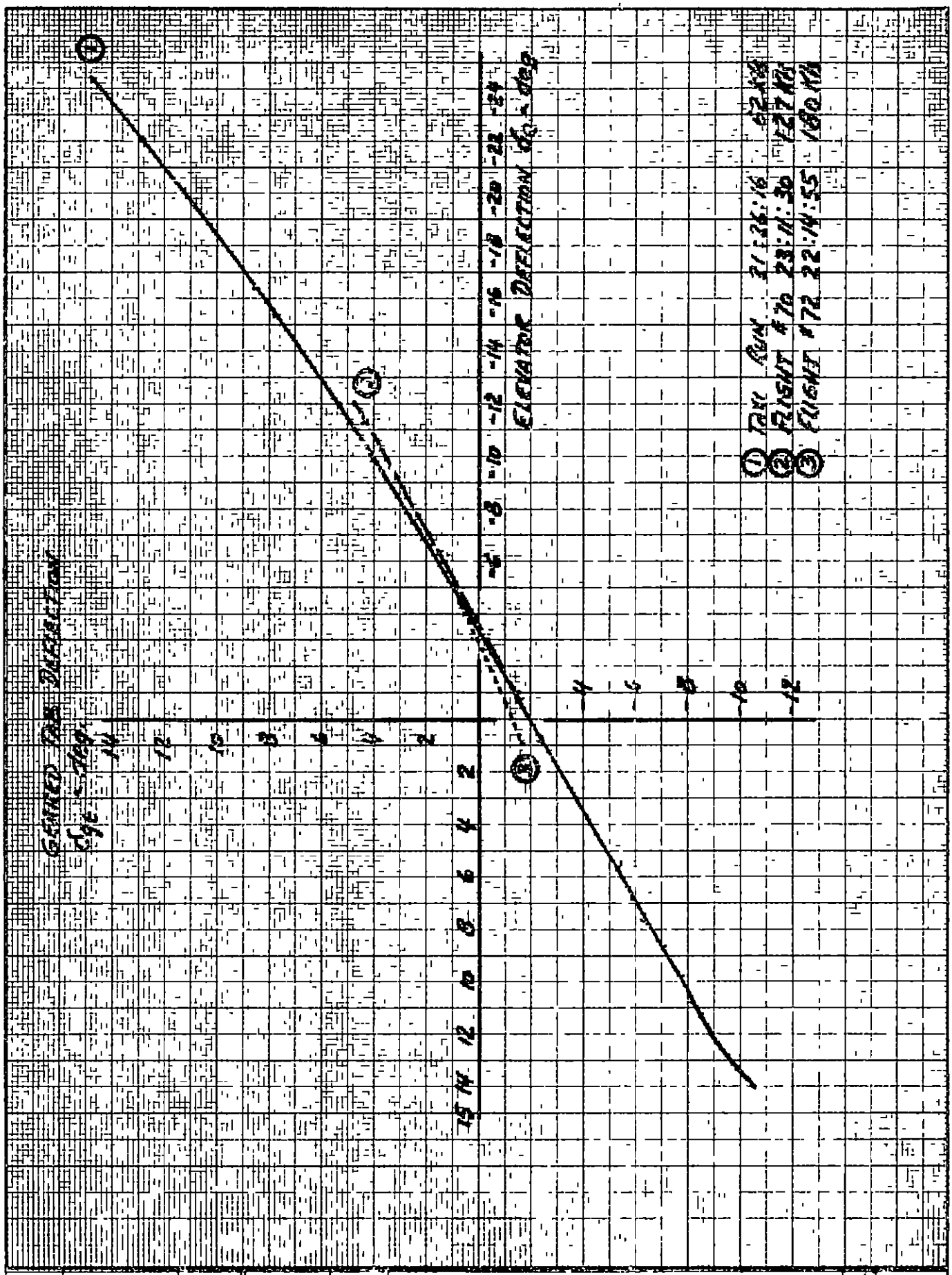
ELEVATOR - COLUMN GEARING

THE BOEING COMPANY

MOD C-8A

FIG. 6.2-12



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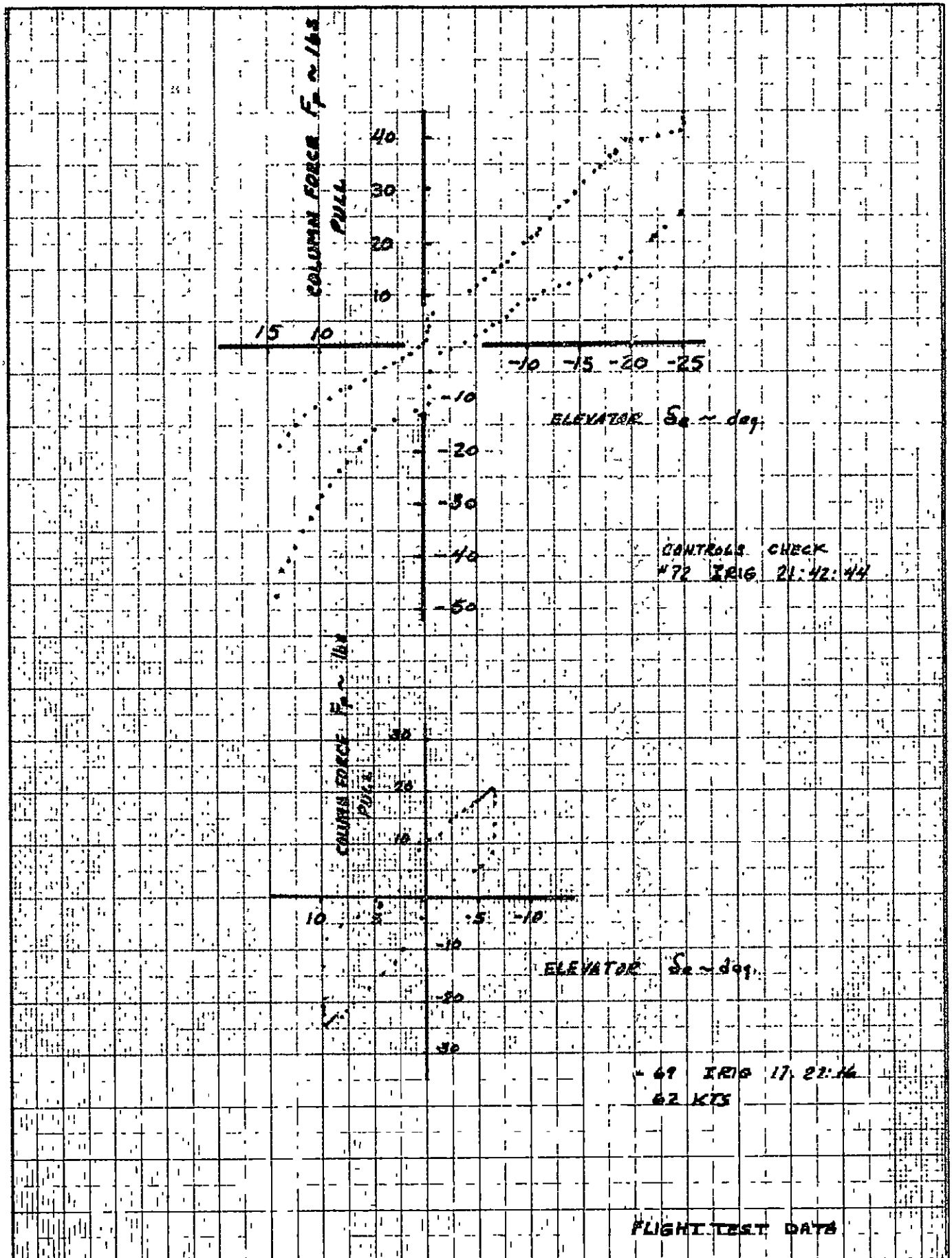
ELEVATOR BALANCE TAB GEARING - FLIGHT TEST DATA

THE BOEING COMPANY

C-E-A

FIG 6.2-13



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APR				
APR				

CONTROL SYSTEM FEEL CHARACTERISTIC

THE BOEING COMPANY

D-OD
C-8A

FIG 6-2-14

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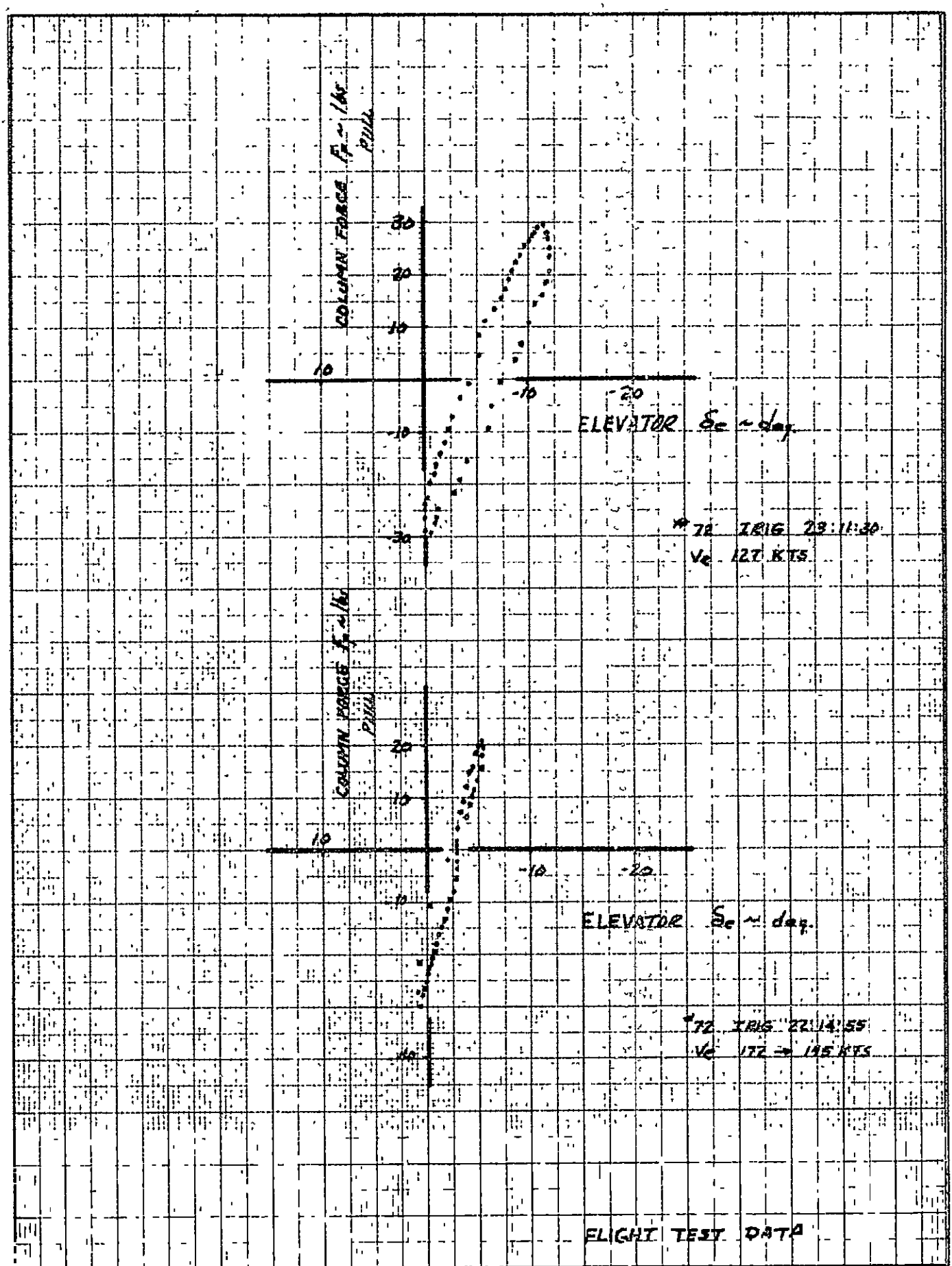
K+E ALBANENE 1861 8 5 G 8000



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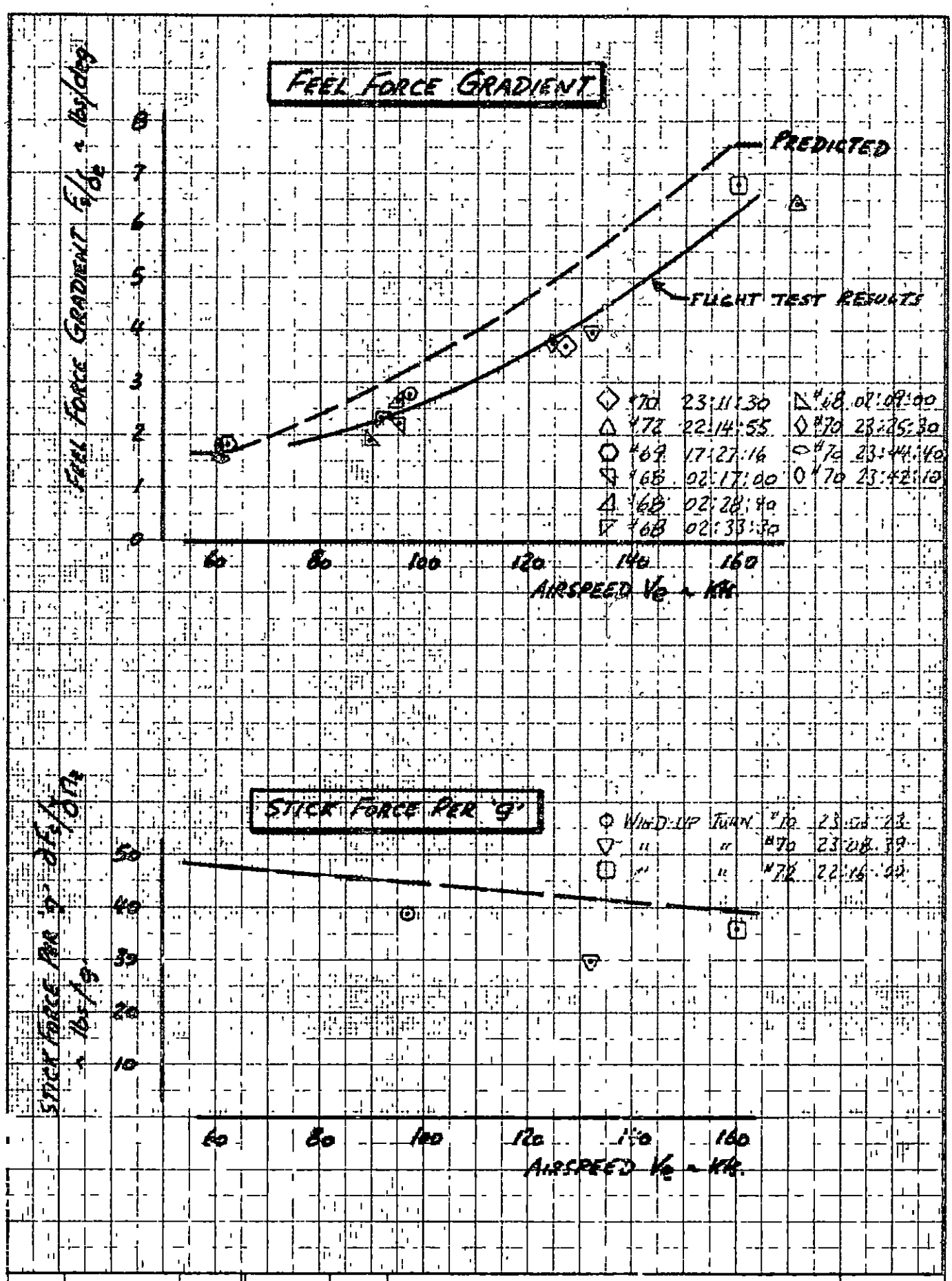
APR				

CONTROL SYSTEM
FEEL CHARACTERISTIC

THE BOEING COMPANY

MOD C-8A

FIG. 6.2-15



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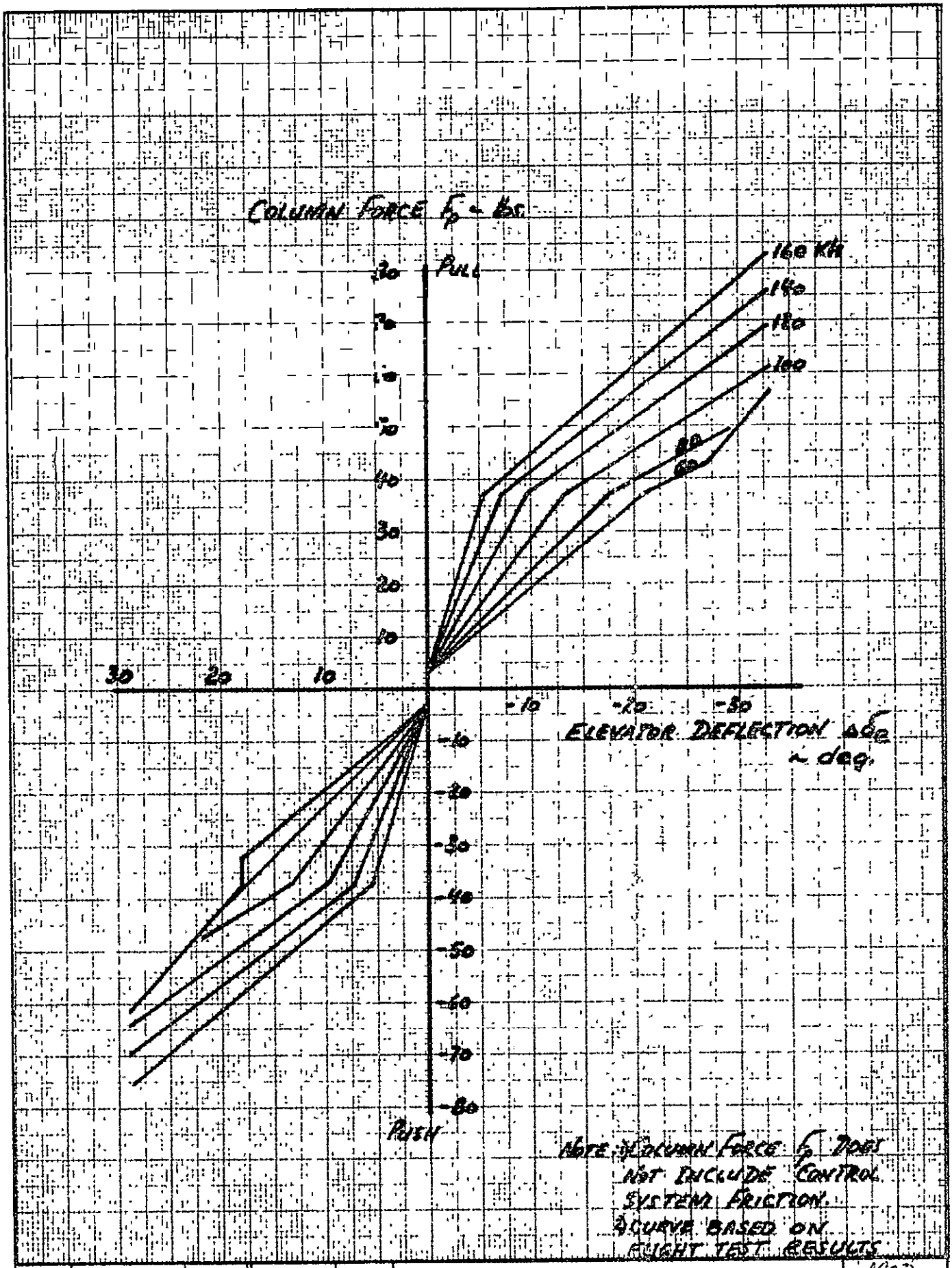
FEEL CHARACTERISTICS

THE BOEING COMPANY

MOD C-B'A

FIG 6.2-16



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CALCULATED
ELEVATOR FEEL FORCE

MOD
C-8A

FIG 6-2-17

THE BOEING COMPANY

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D6-14430

FEEL COMPUTER A

CALC	CHECK	APR	APR	INK
				J.J.JENSEN

REVISED	DATE

REGULATION TOLERANCES
0-500 PSI ± 8%
500-1000 PSI ± 6%
1000-1500 PSI ± 5%

1362

87.5

CRITICAL DESIGN P_T
P_T = 213 $^{+15}_{-10}$ PSI
q_c = 12.2 PSI

CRITICAL DESIGN P_T
P_T = 908 ± 54 PSI
q_c = 57.1 PSF

223 MAX.

203 MIN.

REGULATED PRESSURE (ABOVE RETURN) ~ PSI / PSF

1400 1300 1200 1100 1000 900 800 700 600 500 400 300 200 100

0 10 20 30 40 50 60 70 80 90 100

q_c ~ DYNAMIC AIR PRESSURE ~ PSF

FLIGHT 59 △
FLIGHT 70 ▽
FLIGHT 72 ○

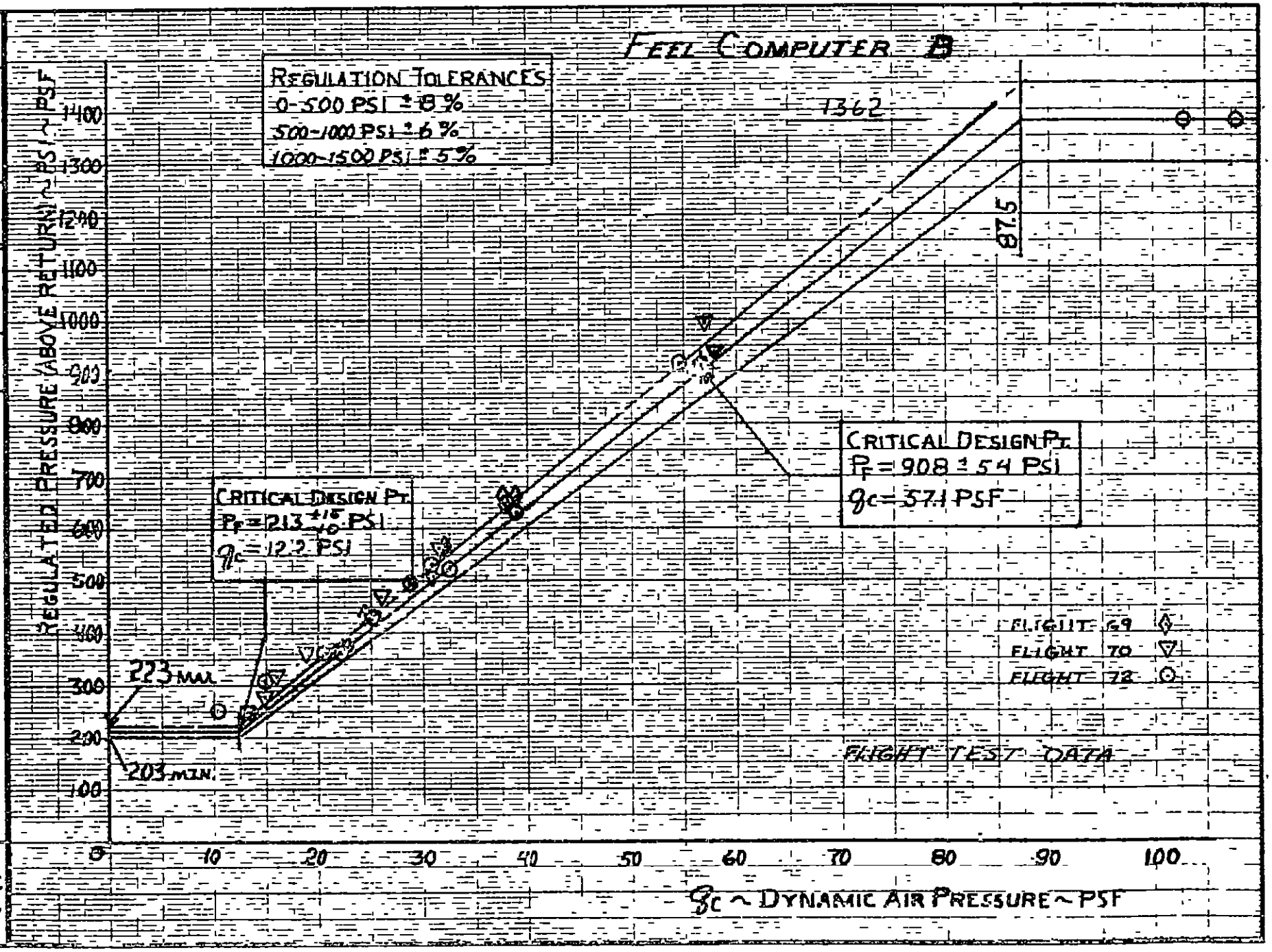
FLIGHT TEST DATA

THE BOEING COMPANY

FEEL PRESSURE
'A' COMPUTER

MOD	FIG.
C-8A	6.2-18

PAGE	
D6-11489	111



CALC		REVISED	DATE
CHECK			
APR			
APR			
INK	J. JENSEN		

FEEL PRESSURE
'B' COMPUTER

THE BOEING COMPANY

MOD C-8A

FIG. 6.2-19

PAGE 112
D6-41489

D1 4100 3240

K&E ALBANENE 1981

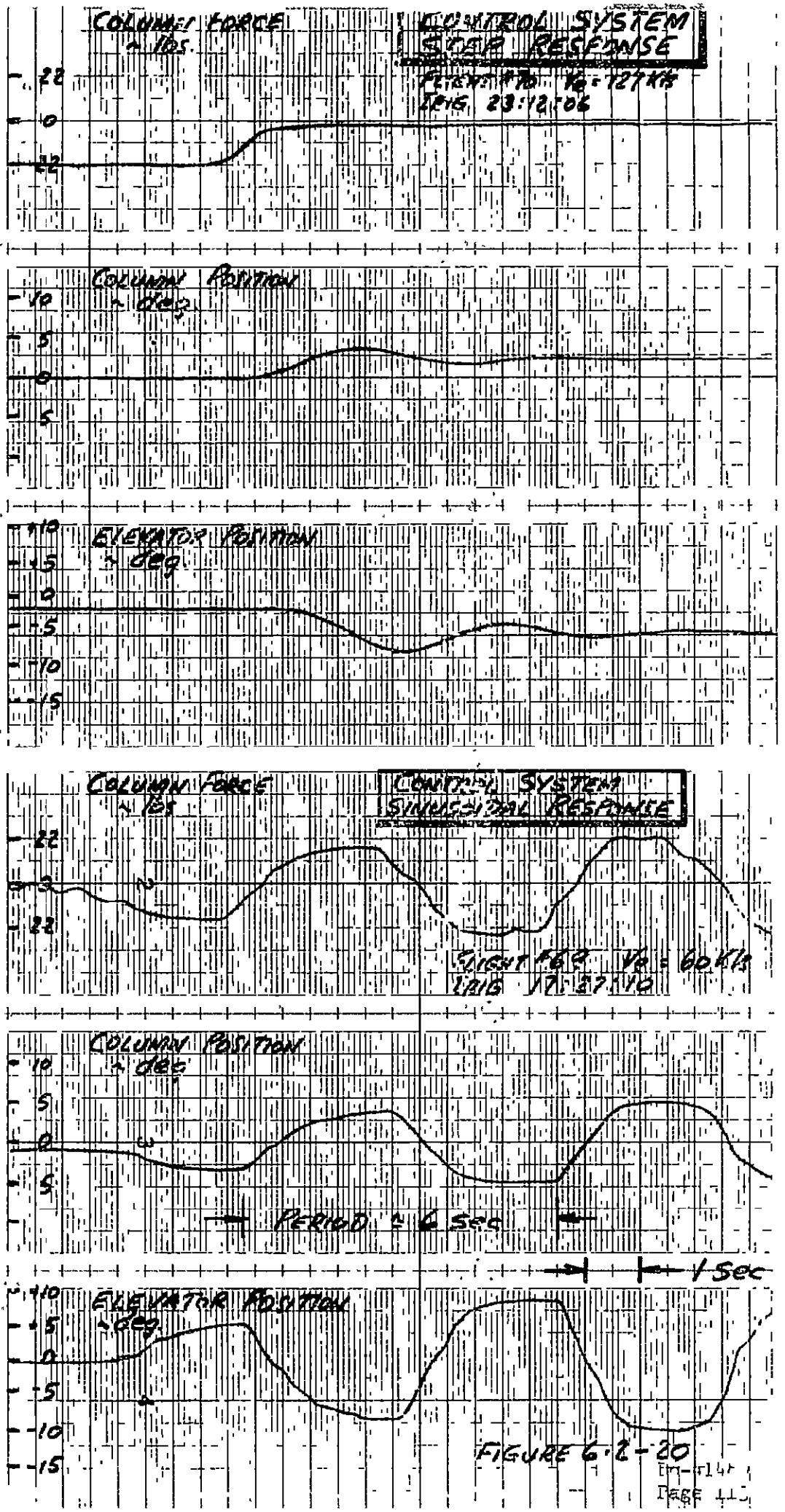
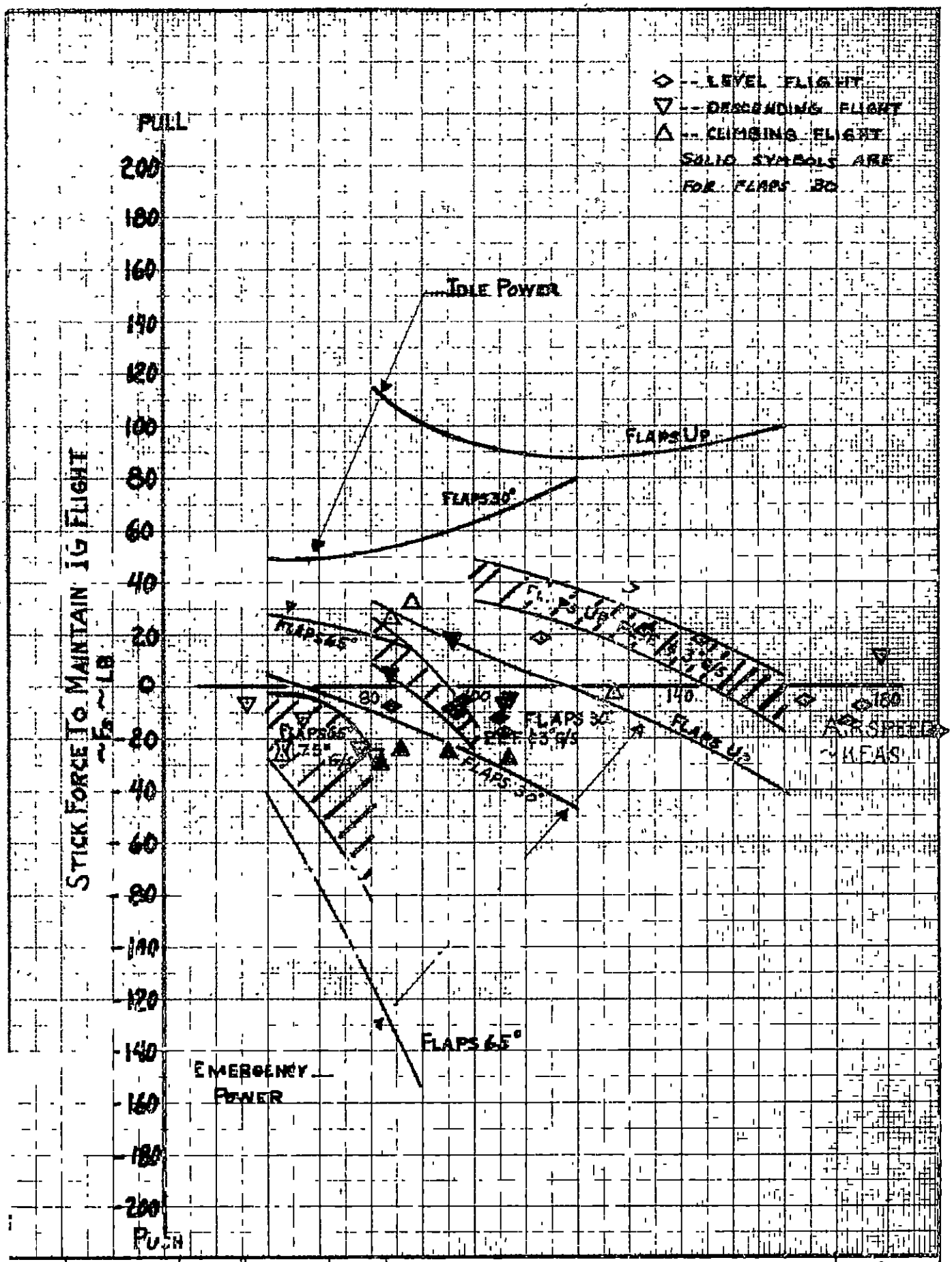

CONTROL SYSTEM STEP RESPONSE
COLUMN FORCE ~ lbs
COLUMN POSITION ~ deg
ELEVATOR POSITION ~ deg
CONTROL SYSTEM SINUSOIDAL RESPONSE
COLUMN FORCE ~ lbs
COLUMN POSITION ~ deg
PERIOD ≈ 4 sec
1 Sec
ELEVATOR POSITION ~ deg

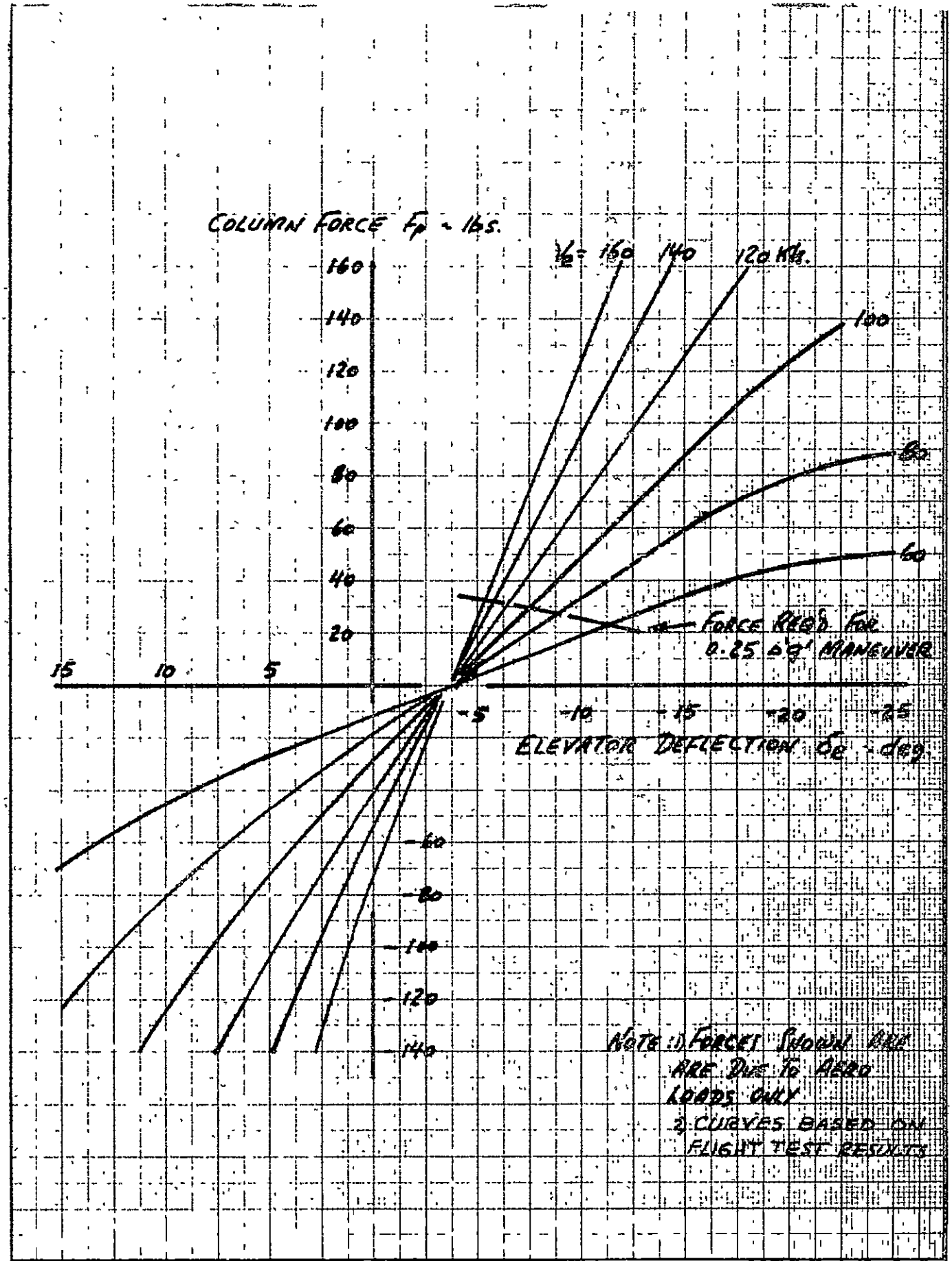
FIGURE 6.2-20



CALC			REVISED	DATE	BOOST OFF STICK FORCES TO MAINTAIN 1G FLIGHT
CHECK					
APR					
APR					THE BOEING COMPANY
INK	J. JENSEN				

MOD C-8A

FG6-2-21



CALC			REVISED	DATE
CHECK				
APR				
APR				

CALCULATED
MANUAL REVERSION

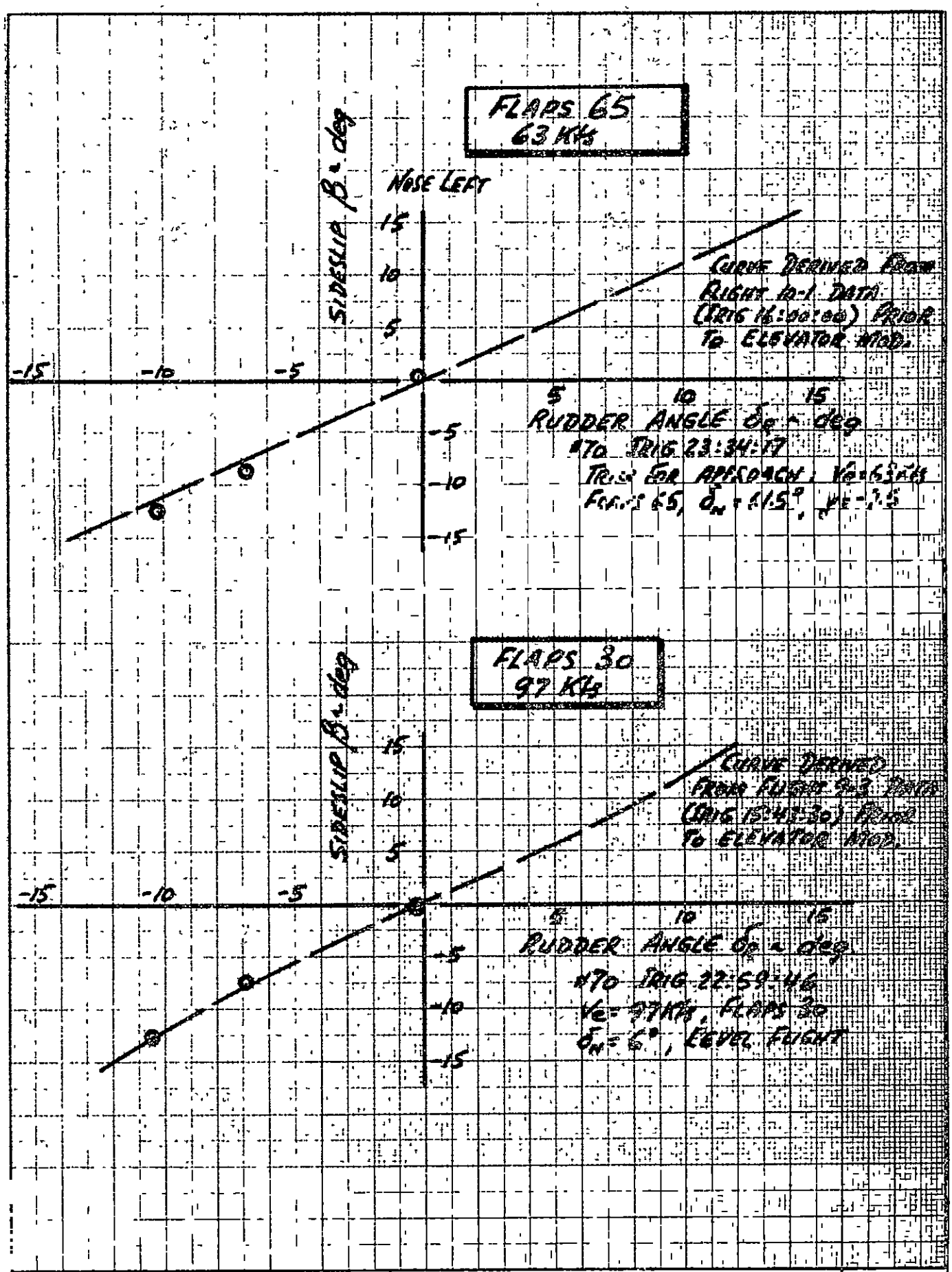
THE BOEING COMPANY

MOD C-8A

FIG 6.2-22

PAGE 115
D6-24780

K•E ALBANENE 1951 8 5 G 8000 TRACING PAPER



CALC			REVISED	DATE
CHECK				
APR				
APR				

STEADY SIDESLIP

THE BOEING COMPANY

MOD C-8A

FIG. 6.2-23

M-F ALBANENE 1961 8 5 G 8000

7.0 CONCLUSIONS AND RECOMMENDATIONS

The design, fabrication and testing of a powered elevator system for the Augmentor Wing Jet STOL Research Airplane was successfully conducted within a 12 month time period. The control system was demonstrated by flight test to be airworthy and to meet its design requirements. No adverse system characteristics were noted by the test pilots.

Control problems encountered in the STOL flight regime with the spring tab elevator system have been eliminated by the powered elevator system. Maximum elevator deflection is possible without excessive pilot effort; control feel has been improved; control system response is rapid and precise. This has allowed the pilot to investigate the low speed characteristics of the AWJSRA with greater precision and increased confidence.

The powered elevator has a blowdown speed of 170 knots instead of 100 knots as predicted. The increased elevator deflection capability at high speed could produce excessive tail loads. Because of the relatively low pilot effort required to attain full trailing edge up elevator, the following recommendations are made:

1. Pilots should be cautioned not to make large elevator control inputs above 120 knots. Control forces should be kept below 40 pounds pilot effort.

2. The powered elevator system should be flight tested with the hydraulic supply pressure reduced from its present value of 2440 psi to 1500 psi. This change should be permanently incorporated if system performance is found acceptable.

REFERENCES

1. D6-26057 Longitudinal Control and Airspeed Control System Study Report

2. D6-40720 The Development of an Augmentor Wing Jet STOL Research Aircraft (Modified C-8A)

3. D6-40947 Powered Elevator with SAS Provisions - Design Requirement, Configuration Definition and System Characteristics

APPENDIX A

PILOT FLIGHT REPORT

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FLIGHT REPORT

DEL NO	NO.	DATE OF TEST
AWJSRA	716	9/19/73
ORDER	PROJ ENG.	PILOT
T-4488	Vomaske	Innis/Grossmith
GHT NO	T.O GROSS WT/C G	FLIGHT TIME
68	44,800#/30.7%	

RPOSE Flutter Test

Preliminary evaluation of powered elevator.

1. At 90 Kts, A/C seems quite responsive to column inputs. Step attitude changes can be made quickly and precisely. Release from column steps produce 1 or 2 overshoots but are not objectionable because A/C doesn't appear to respond to them and normal pilot activity (hand on wheel) adds sufficient damping to eliminate overshoots. Elevator deflection rate seems satisfactory. Breakout was not noticeable during maneuvering and centering was satisfactory. Force gradient was satisfactory - easily manageable with one hand. With elevator fixed, full nose up and nose down trim produced forces which could be held with one hand. A/C could be controlled with both hands. These same comments would apply up to 120 Kts.

2. Immediately after lift off, aileron forces seemed rather light with some tendency to overcontrol. This didn't bother me after I got used to it, but will have to be evaluated further.

3. Approach to stall was accomplished with flaps 30°, PLF, trim at 90 Kts. One α indicator went off scale (20°) at 66 Kts. The other one was hooked up backwards. Very light buffet was detected at 58 Kts. Stall was at 56 Kts with light buffet, slight nose down pitch and perhaps a little left roll. Recovery seemed to occur with very little α reduction and little or no hysteresis. A/C was quite responsive to forward column input.

4. Landing with flaps 65°, nozzles 70° at 65 Kts. Flare seemed to require more back force than I was used to, but control seemed good. Toughdown was firm but not hard. A/C bounced back in air, however, even with lift dump active. (Throttles were not retarded)

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FLIGHT REPORT

L NO	NO.	DATE OF TEST
C-8A	N716NA	9/21/73
ORDER	PROJ. ENG.	PILOT
T-4488	Vomaske	Grossmith/Innis
IT NO	T.O. GROSS WT/C.G.	FLIGHT TIME
69	44,640/30.7	1.3

OSE

Item 1 Control sweeps

Item 2 Take-off δF=25, 99% N_H, Noz=6° SAS-off.
Precise control of elevator throughout.

Item 3 10 sec. steady-state. 90 KIAS, δF=30, 96% N_H, Noz=6° SAS-off.

Item 4 As per item 3. 80 KIAS SAS-off.

Item 5 As per item 3. 100 KIAS SAS-off.

Item 6 110 KIAS. δF=5.6, PFLF, Noz=6°, SAS-off
Elevator and rudder pulses, "B" Hyd. Sys. off.

Item 7 120 KIAS. as per item 6.

Item 8 120 KIAS. δF=5.6, PFLF, Noz=6° SAS-off
Elevator and rudder pulses, both Hyd. Sys. on.

Item 9 125 KIAS. As per item 8

Item 10 130 KIAS as per item 8

Item 12 135 KIAS as per item 8

Item 13 140 KIAS as per item 8
No peculiarities noted. At 140 KIAS, some longitudinal oscillation noted.
Fuselage bending mode?

Item 15 100 KIAS. δF=30°, PFLF, Noz=6° SAS-on.
5 sec. trim shot.

Item 16 90 KIAS as per item 15

Item 17 80 KIAS as per item 15

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FLIGHT REPORT

L NO	NO.	DATE OF TEST
2-24	711	7-21-73
ORDER	PROJ ENG	PILOT
		SMOLSMITH
T NO.	T.O GROSS WT/C G.	FLIGHT TIME
69	44610	

OSE

Item 18 90 KIAS δF=30°, Noz=6°, Flt Path Angle -3°, SAS on
5 sec. trim shot.

Item 19 90 KIAS as per item 18

Item 20 100 KIAS as per item 18

Item 21 60 KIAS, δF=65°, Noz=70°, Flt Path Angle =-7 1/2°, 93% N_H sinusoidals, displacements, full nose up and nose down trim changes, sideslips to 15°, pitch attitude changes ±5° and ±10°, approach to stall.
Notes:- in sideslips max. bank angle required for 15° s.s. approximately 5°.
Wanders in yaw.
Approach to stall - 58 KIAS α =14°
53 KIAS α = 20°
50 KIAS α = 24°
48 KIAS α = 26°
At α = 26° some laterally wallowing - lateral control adequate to control.
At approximately α= 30° slight "g" break experienced, no roll-off tendency.

Item 22 90 KIAS, δF=30°, Noz=6°, PFLF.
Pull-up and push overs, 0.4 - 1.6"g" "A" hyd. sys. off. A slight bump in the elevator control during the 1.6g maneuver.

Item 23 As per item 22. with "B" hyd. sys. off.
No bump experienced this time during the 1.6g maneuver.

Item 25 65 KIAS, δF=65°, Noz=70°, 93% N_H - normal approach and landing. Elevator control precise during the approach, flare and landing forces seemed higher during flare and landing when compared to original elevator system.

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FLIGHT REPORT

DEL NO	NO.	DATE OF TEST
AWLISRA	716	9/21/73
B ORDER	PROJ. ENG	PILOT
T-4488	Vomaske	Innis/Grossmith
GHT NO.	T O. GROSS WT/C G.	FLIGHT TIME
70	46,800#/30.0%	

RPOSE

Powered Elevator Flight Test

1. Approach to stall. Flaps 5.6°, trim at 120 Kts. Terminated at 66 Kts, α=27° where moderate buffet (tail, I think) was encountered. Nose high attitude was quite extreme. Copilot reported 17° up elevator required at min. speed.

   Steady state sideslip at 60 KTS, Flaps 65° nozzles 70°, SAS ON. Seemed difficult to stabilize sideslip due to tendency for nose to wander in yaw. SAS doesn't seem to help much. Also noted quite a bit of bottom rudder required in turns.

3. Longitudinal control at 60 Kts. A/C is quite sluggish and quite a bit of opposite control is required to stop pitch rate once established.

4. Approach and landing, 65 Kts. Fairly high back force required to flare A/C but control didn't seem too bad. Landing was OK.

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FLIGHT REPORT

ODEL NO	NO	DATE OF TEST
AWJSRA	716	9/25/73
OB ORDER	**PROJ. ENG**	**PILOT**
T-4488	Vomaske	Innis/Hardy
LIGHT NO	**T.O. GROSS WT/C.G.**	**FLIGHT TIME**
71		

URPOSE
Powered Elevator Tests

1. At 160 Kts, Flaps 5.6° - Push over-pull ups. Noticeable increase in buffet as A/C is pushed over to 2° α
   Longitudinal stability and control are excellent.
   Full nose up and nose down trim can be held with one hand on column.

2. At 120 Kts, no change in longitudinal trim was noticed with $\pm 8°$ β.

3. Landing approach was made at 60 Kts, $\gamma < 6°$. Flare was more gentle than previous. Control was good.

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FLIGHT REPORT

DEL NO	NO.	DATE OF TEST
C-8A	716	~~11/29/73~~ 9/26/73
ORDER	PROJ ENG	PILOT
	Vomaske	Hardy/Innis
HT NO	T.O GROSS WT/C.G	FLIGHT TIME
72	46550#	1:14

POSE

Final Powered Elevator Acceptance Test Flight

This was my first left seat flight with the powered elevator and I was quite pleased with the system. Stick force gradients, breakouts, etc., are very nice from 60-180 kts.

Manual reversion was examined at 90 kts and 30° flaps. Airplane was slightly out of trim (Fs<20#) when the hydraulics were shut off. No problem maneuvering the aircraft in this mode and feel a successful landing could be made. Large deadband is noticeable but acceptable.

Lateral control was better than on my previous flights (March 73)