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AFWAL-TR-86-3019

TEST EVALUATION OF A RECONFIGURABLE FAULT-TOLERANT FLY-BY-WIRE ACTUATION SYSTEM

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July 1986

terim Report for Period May 1933 to October 1983

Approved for public release; distribution unlimited.

FLIGHT DYNAMICS LABORATORY
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NOTICE

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UNCLASSIFIED

SECURITY CLASSIFICATION OF THIS PAGE

ADA172455

REPORT DOCUMENTATION PAGE

1a. REPORT SECURITY CLASSIFICATION UNCLASSIFIED	1b. RESTRICTIVE MARKINGS None
2a. SECURITY CLASSIFICATION AUTHORITY N/A	3. DISTRIBUTION/AVAILABILITY OF REPORT Approved for Public Release; distribution unlimited
2b. DECLASSIFICATION/DOWNGRADING SCHEDULE N/A	
4. PERFORMING ORGANIZATION REPORT NUMBER(S)	5. MONITORING ORGANIZATION REPORT NUMBER(S) AFWAL-TR-86-3019
6a. NAME OF PERFORMING ORGANIZATION Dynamic Controls, Inc. 6b. OFFICE SYMBOL (If applicable)	7a. NAME OF MONITORING ORGANIZATION Flight Dynamics Laboratory (AFWAL/FIGL)
6c. ADDRESS (City, State and ZIP Code) 7060 Cliffwood Place Dayton, OH 45424	7b. ADDRESS (City, State and ZIP Code) Air Force Wright Aeronautical Laboratories Air Force Systems Command Wright-Patterson AFB, OH 45433-6553
8a. NAME OF FUNDING/SPONSORING ORGANIZATION Flight Dynamics Laboratory 8b. OFFICE SYMBOL (If applicable) AFWAL/FIGL	9. PROCUREMENT INSTRUMENT IDENTIFICATION NUMBER F33615-83-C-3600
8c. ADDRESS (City, State and ZIP Code) Air Force Wright Aeronautical Laboratories Air Force Systems Command Wright-Patterson AFB, OH 45433	10. SOURCE OF FUNDING NOS. PROGRAM ELEMENT NO.: 62201F PROJECT NO.: 2403 TASK NO.: 02 WORK UNIT NO.: 91
11. TITLE (Include Security Classification) Test Evaluation of a Reconfigurable Fault-Tolerant Fly-by-Wire Actuator System	
12. PERSONAL AUTHOR(S) William G. Talley, Julianne Minch, Gavin D. Jenney	
13a. TYPE OF REPORT Interim 13b. TIME COVERED FROM 5/83 TO 10/83	14. DATE OF REPORT (Yr., Mo., Day) 86/07/08 15. PAGE COUNT 184
16. SUPPLEMENTARY NOTATION None	

17. COSATI CODES			18. SUBJECT TERMS (Continue on reverse if necessary and identify by block number)
FIELD	GROUP	SUB. GR.	Smart Actuator, Digital Flight Controls, Fly-by-Wire, Control Reconfiguration, Servoactuator
01	03		

19. ABSTRACT (Continue on reverse if necessary and identify by block number)

This report describes the test evaluation of a redundant fly-by-wire actuator system designed by the Boeing Military Airplane Company, Seattle, Washington. The testing was conducted from May to October 1983. The system was designed to be two-fail-operate for electromechanical failures and single-fail-operate for hydromechanical failures. The system used a single microprocessor to close the position control loop, monitor the operation of the actuator and reconfigure the system upon component failures of the actuator. The microprocessor was not designed to be failure tolerant. The testing conducted was a measurement of the input-to-output characteristics of the system. The system was unusual in the use of a microprocessor for failure detection and control and in that electro-hydraulic channels were operated together in a nominal force fight configuration. The test evaluation included operating the system in both loaded and unloaded configurations.

(continued on reverse)

20. DISTRIBUTION/AVAILABILITY OF ABSTRACT UNCLASSIFIED/UNLIMITED ☐ SAME AS RPT. ☒ DTIC USERS ☐	21. ABSTRACT SECURITY CLASSIFICATION UNCLASSIFIED
22a. NAME OF RESPONSIBLE INDIVIDUAL Mr. Gregory Cecere	22b. TELEPHONE NUMBER (Include Area Code) 513-255-2831 22c. OFFICE SYMBOL AFWAL/FIGL

DD FORM 1473, 83 APR EDITION OF 1 JAN 73 IS OBSOLETE

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SECURITY CLASSIFICATION OF THIS PAGE

CONT. The system operated successfully with input/output characteristics similar to other fly-by-wire systems. Areas where improvements or careful design should be implemented are presented in the report.

FOREWORD

The effort described in this document was performed by Dynamic Controls, Inc. of Dayton, Ohio under Air Force Contract F33615-83-C-3600. The work under the contract was carried out in the Air Force Flight Dynamics Laboratory of the Air Force Wright Aeronautical Laboratories, Wright-Patterson Air Force Base. Ohio. The work was administered by Mr. Gregory Cecere, AFWAL/FICL Program Manager.

This report covers work performed between May 1983 and October 1983. The technical report was submitted by the authors in September 1985.

The authors express their appreciation to Dynamic Controls, Inc. engineer Harry Schreadley for his contributions in defining and implementing the test system setup. Appreciation is also expressed to Boeing engineer I. J. Takats for his assistance in the inital setup and operation of the microprocessor-controlled fly-by-wire actuator.

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SECTION I. INTRODUCTION AND SUMMARY

INTRODUCTION

This document describes the test of a redundant Fly-By-Wire actuator system designed by the Boeing Military Airplane Company, Seattle, Washington. The system is designed to be two-fail-operate for electromechanical failures and single-fail-operate for hydromechanical failures. Testing of the system by DCI at Wright-Patterson AFB, Ohio, occurred during the period from May to October 1983.

The design of the system is based on using a microprocessor to control and monitor the operation of a tandem actuator and reconfigure the system upon component failure in the actuator. The microprocessor was not designed to be failure tolerant. The testing conducted was a measurement of the input-to-output characteristics of the system. The system was unusual in the use of a microprocessor for failure detection and control, and in that electro-hydraulic channels were run together in a nominal force fight configuration. The test evaluation included operating the system in both loaded and unloaded configuration.

SUMMARY

The system operated successfully with input/output characteristics consistent with other Fly-By-Wire systems. The microprocessor was able to identify failures and reconfigure the system successfully. However, there are several characteristics of the mechanism for which improvement or careful design is recommended. These are: (a) the piston seal should be designed to accept the stresses resulting from the increased force fight in the presence of digital noise, (b) the failure logic threshold is frequency dependent and should be set up to match the failure response requirements of the actuator, (c) without tracking equalization (as was the test system), bias mis-matches of the servovalves degrade the threshold and signal fidelity, (d) the technique of sampling and reusing failed channels after a failure has occurred caused incorrect failure voting (it is recommended that it not be used).

SECTION II. SYSTEM DESCRIPTION

The Boeing Microprocessor controlled actuator which was evaluated is based upon using a tandem hydraulic actuator with two drive pistons. Each piston is connected through a solenoid operated bypass valve to a servovalve. Each servovalve contains two input coils and two LVDT's which measure spool position. In the normal mode operation, both servovalves are used to drive the actuator.

The actuator system is designed to be two-fail-operate for electronic failures and single-fail-operate for hydraulic failures. Loss of supply pressure or two electronic failures in the section used to drive one servovalve cause the failure logic to bypass that servovalve. To provide the dual-fail-operate characteristics, electronic servovalve models of the spool position are used for comparison with the actual spool position measured by the LVDT's. Two spool position LVDT's and two servovalve models are used for each servovalve. Only one servovalve LVDT and one servovalve model pair's output is connected to the failure detection logic at a time. Four position signals (a model and an LVDT output for each servovalve) are used for failure detection. When the failure detection logic determines there is a disagreement between any one of the four signals and the other three, an action in the electronics associated with the "failed" signal is initiated. If the failure is associated with an "active" channel, a transfer in initiated. This transfer is a switching of the input coils used to control the servovalve associated with the "failed" signal and a simultaneous transfer of the spool LVDT and model position signal output to the alternate pair. If the failure is associated with a "model" channel, no transfer occurs but a failure is declared and the output signal of the failed model channel is no longer voted with the other remaining channels.

The two bypass valves (one for each servovalve) are electrically controlled and pressure operated. Either loss of hydraulic pressure to the servovalve or two voted failures in the electronics associated with the servovalve cause the bypass valve to operate.

The failure detection logic design provides for automatically changing the failure status of a channel after it is voted "failed". The output signal of the "failed" electronics is continuously sampled to determine if it should return to a "good" state again. If a "failed" channel's output is correct for a specified number of consecutive samples and comparisons, the channel status is changed from a "failed" to a "good" status. No change of assignment of the active and model channel operation results from the change in the "failed" channel status, however the channel is used for failure monitoring.

Figure 1 is a schematic of the Boeing microprocessor control actuator system. Figure 2 shows the microprocessor control equipment with its failure display panel and digital input sources. Figure 3 shows the actuator with the interface panel mounted on top by DCI. In Figure 3 the actuator is clamped to a test plate in preparation for the unloaded tests.

The central processing unit used by Boeing is an Intel 80/05 microprocessor. DATEL analog I/O units provided 16 channels of analog input and 16 channels of analog output.

The actuator used for the system was a tandem electrohydraulic actuator, Part Number HR 41004890, manufactured by HR Textron, Valencia, California. The

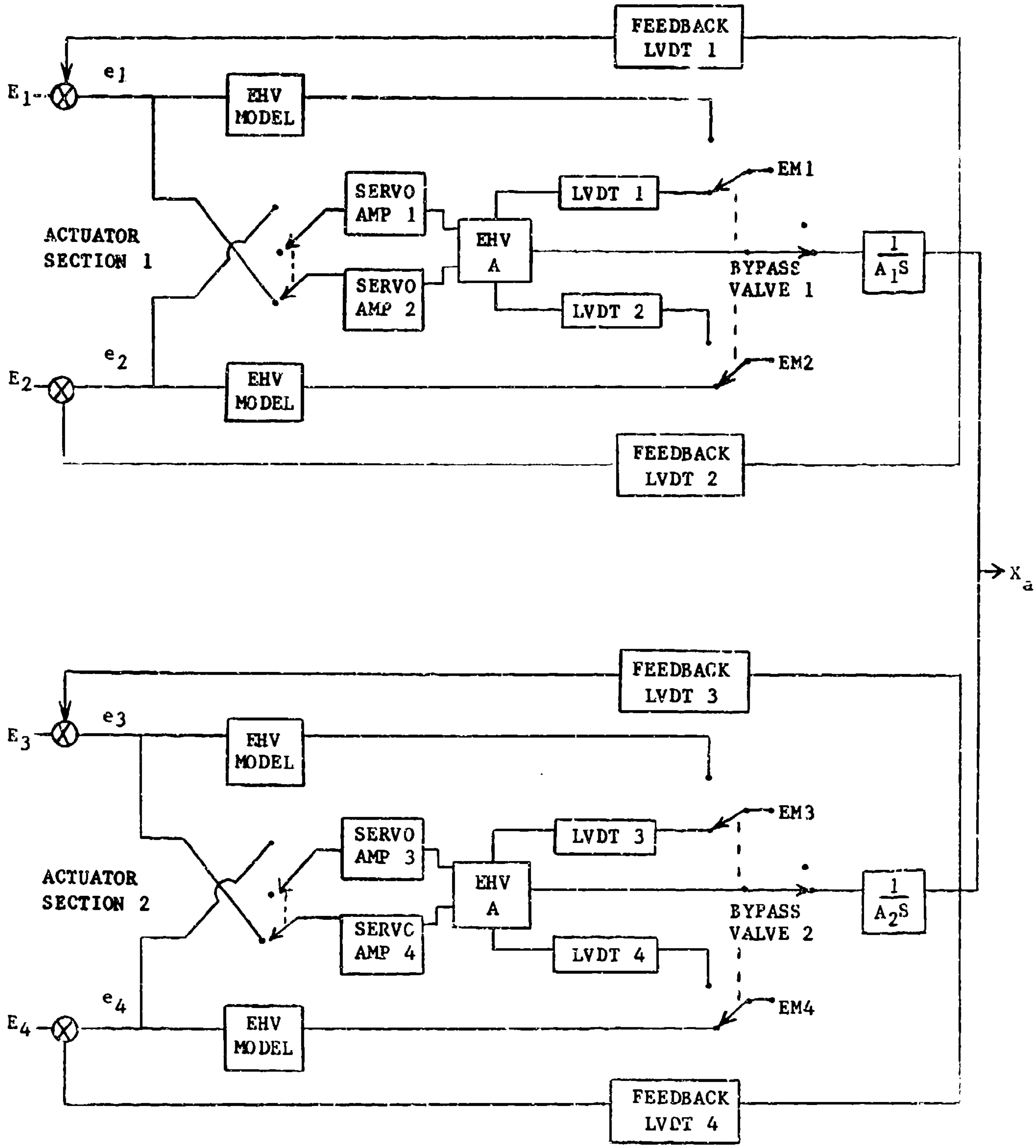


Figure 1. Actuator Control Schematic

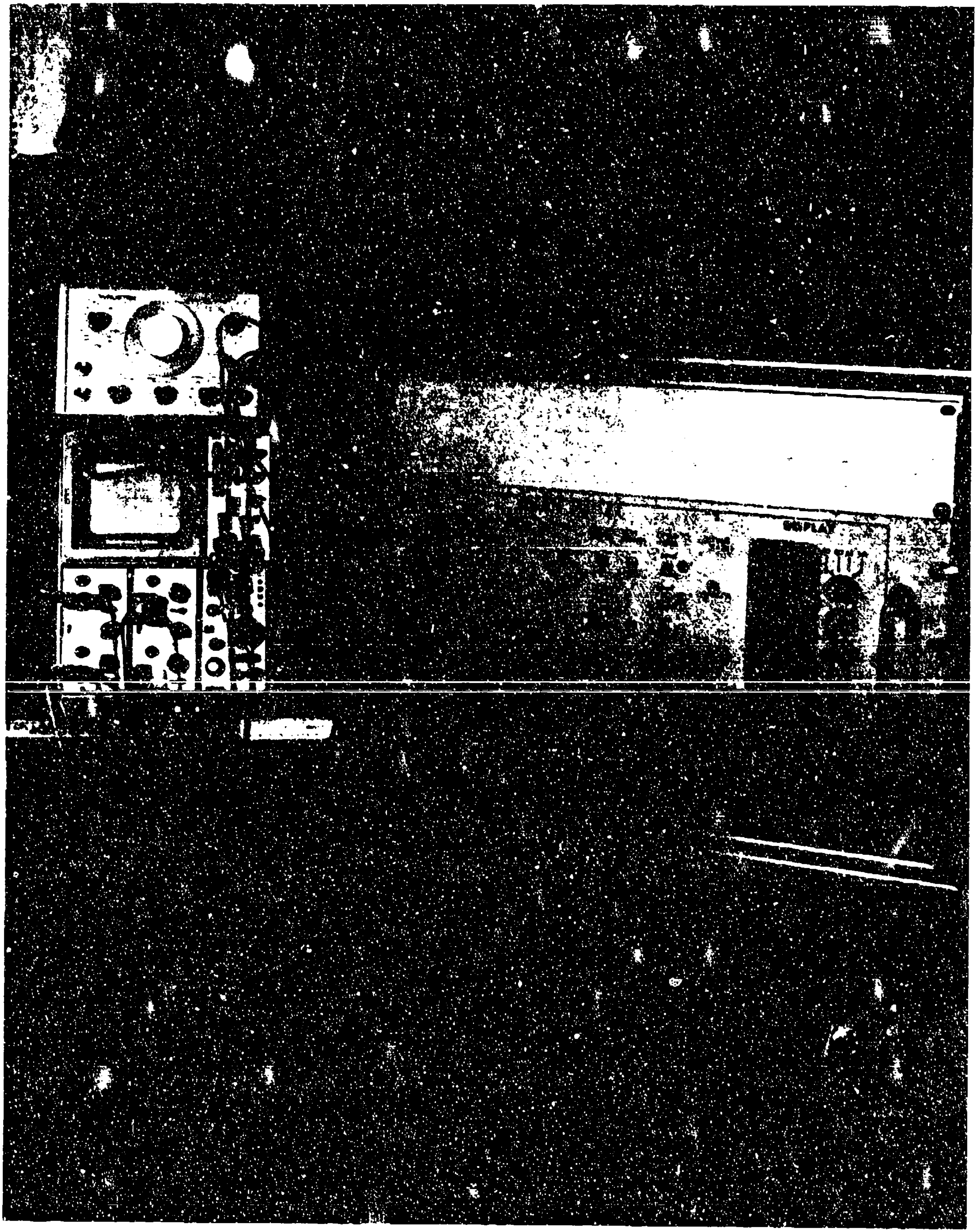


Figure 2. Microprocessor Control Box

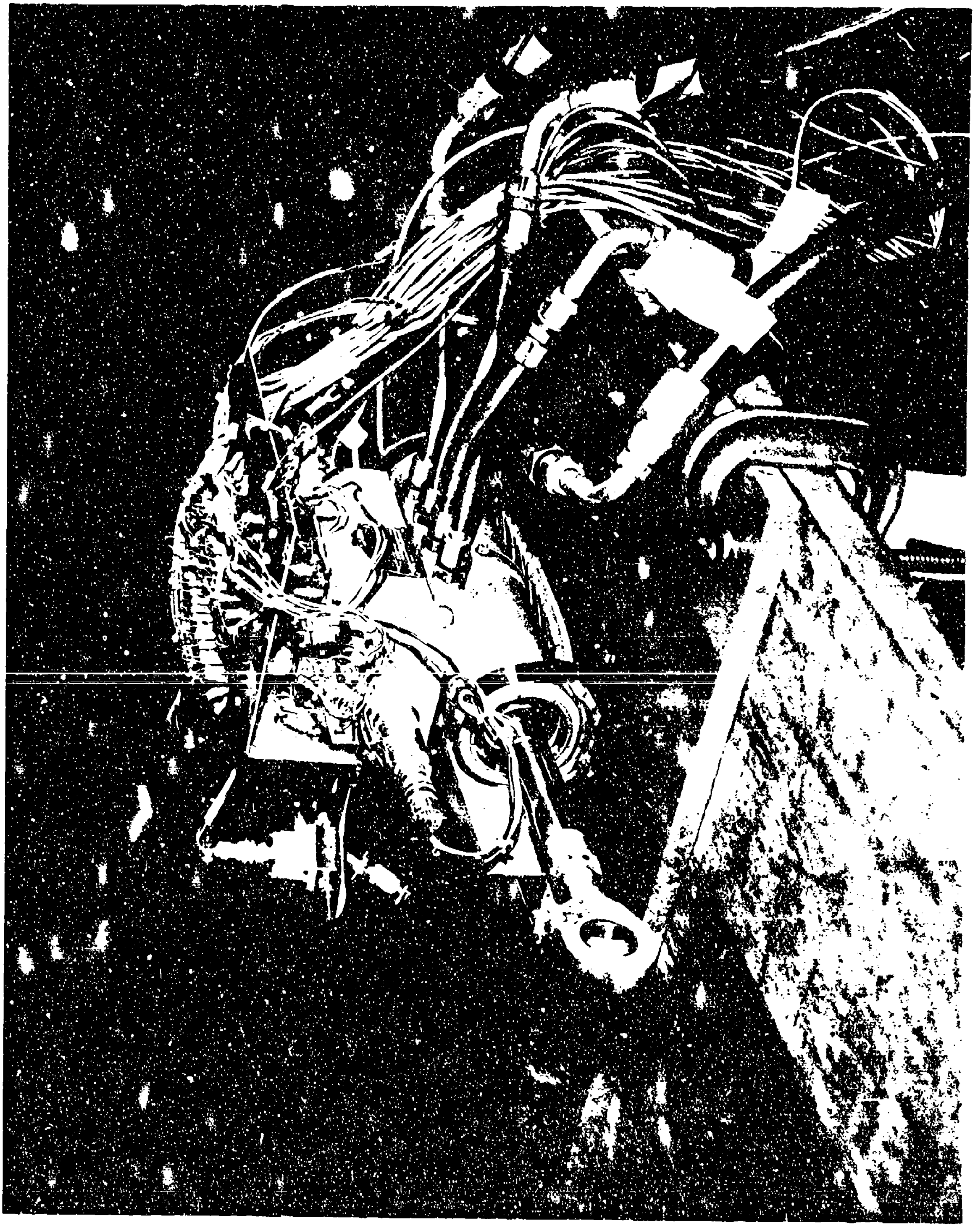


Figure 3. Actuator with Interface Panel

actuator had a drive area of 3.1 square inches per section and a stroke of ±1.69 inches. Two flapper nozzle servovalves capable of an output flow of 2.25 gpm each were mounted on the actuator. Each servovalve incorporated two LVDT's which were used to measure the servovalve spool's position. The servovalves were a conventional flapper nozzle design. The frequency response at 25% input was rated at -3 dB at 200 Hz with a phase lag of 90° at 100 Hz. The actuator incorporated two solenoid valves which were used to bypass one section of the tandem actuator upon a second electrical or a first hydraulic failure.

Although Figure 1 shows four position transducers used to measure the actuator position (as would normally be mechanized), Boeing mounted two position LVDT's and split the outputs to simulate the four transducers. This was done because of a limit of 16 A/D converters used in the microprocessor.

The failure detection circuitry design was based on sampling a failure a predetermined number of times before voting a channel out and/or reconfiguration. The failure detection was therefore a combination of an amplitude and a time window. This method was used to minimize failure declaration sensitivity to random short duration failures when the system was operated. The number of samples required for declaration of a failure could be changed from the front panel of the microprocessor. As with failure declaration, voting a sample channel "good" again and using it for failure logic required a sample amplitude window. The number of samples during which a previously declared fail channel had to operate properly before being declared "good" was variable from the front panel of the microprocessor.

TEST EVALUATION

The operation of the Boeing Microprocessor system in its operate and fail operate modes can be completely described by performance testing with selected combinations of active channels. The status of the monitor channels (failed or operational) do not affect the input/output characteristics of the system. Therefore the test conditions used for performance measurement of the system do not include all possible combinations of monitor channel status conditions, since no additional information would be obtained.

The general test evaluations conducted on the system were input/output performance measurements. These measurements defined both the linear performance and nonlinear characteristics of the mechanization. Included in the testing was evaluating the effect of failure insertion and input removal. Because the system did not use equalization to prevent force fighting in the control actuator, performance measurements with channel offsets were made. Although Boeing had previously evaluated the unloaded performance of the system, they had not tested the mechanism with the application of output loads. The following Section III is the general test procedure used for evaluating the system (This procedure has been used previously by DCI for evaluating other flight control configurations.)

SECTION III. GENERAL TEST PROCEDURE

The following general test procedure was used for evaluating the demonstration system. This procedure defines the measured parameters and states the general method used in making the measurement. The procedure is divided into the following categories:

1. Performance Measurements
2. Failure Effect on Performance
3. Input Deviations Effect
4. Failure Transients
5. Failure Logic Detection Characteristics

PERFORMANCE MEASUREMENTS

Threshold

Static Threshold "The minimum input change from zero level which causes a measurable output change."

Procedure - Apply a slowly increasing + input until a measurable output change occurs. Repeat for - input. Threshold is indicated by the minimum input change for a measurable output change.

Dynamic Threshold "The input level (at a particular frequency) required to cause a measurable output level."

Procedure - A sinusoidal input at a selected frequency of 50% of the bandpass of the actuator is applied to the actuator. The amplitude of input to create a measurable output indicates the dynamic threshold. The bandpass of the actuator is defined as the frequency at which -3 dB amplitude or 90° phase shift occurs (whichever is lower in frequency).

Frequency Response "With a sinusoidal actuator input, the frequency response of the actuator is the relationship of the output to input expressed as an amplitude ratio and phase angle as a function of frequency."

Procedure - Apply a sinusoidal input of an amplitude which is:

a. large enough to minimize the nonlinearity distortions of threshold and hysteresis

b. small enough to avoid velocity saturation in the frequency range of interest. The ratio of output amplitude to input amplitude and output phase angle relative to input is recorded.

The plot of the amplitude ratio and phase indicate the frequency response.

Linearity "The deviation of output vs input from a straight line relationship."

Procedure - Apply an input from - to + maximum input while recording the corresponding output position. Linearity is indicated by the deviation of the plotted output vs input from a straight line drawn between zero and a point which minimizes the maximum deviation of the plotted curve from the straight line. Repeat for + input to - input.

Hysteresis "The non-coincidence of loading and unloading curves."

Procedure - Apply a slowly varying input to the actuator at up to 1% of the maximum input in the following sequence while recording the actuator output position:

1. 0 to + direction input

2. + input to - direction input

3. - input to + input

Repeat for an input up to 10% of the maximum input. From the plot of output vs input, the hysteresis is indicated by the difference between + direction actuator output position and - direction output position for the same input level.

Distortion "The amount of deviation of the actuator output waveform from the input waveform."

Procedure - The harmonic distortion, at the input levels used to measure the frequency response, is recorded at sinusoidal input frequencies of 10%, 50% and 100% of the bandwidth.

Time Response

Saturation Velocity "The maximum velocity at which the actuator is capable of moving in each direction."

Procedure - With the actuator at zero position, a maximum amplitude input is applied to the actuator while the actuator motion vs time is recorded. The test is conducted for both directions of actuator motion. The slope of the position vs time record indicates the saturation velocity.

Step Response "The time response of the actuator output to an applied step input."

Procedure - Apply a step input to the actuator and record the corresponding actuator motion. The amplitude of the step should be:

a. large enough to minimize the nonlinearity distortion of threshold and hysteresis

b. small enough to avoid velocity saturation.

FAILURE EFFECTS ON PERFORMANCE

Failure Effect "The change in the performance of a redundant actuator due to input failures or internal failures of actuator components."

Procedure - Inject hydraulic or electrical input failures into the actuator under test to cause it to operate in its "failure operational" modes. For each mode, measure the performance by repeating the Performance Measurement tests. The input levels should be maintained at those used for the "no failure" performance tests, unless the performance changes dictate different levels in order to obtain reasonable test data.

INPUT DEVIATIONS EFFECT

Electrical Input Deviations "The change of electronic inputs, both power and control, with respect to the normal values and/or each other."

Procedure - Adjust the electrical inputs one at a time until either the maximum expected deviation of the input is reached or the failure trip level is reached. Section 2.1 will be measured with each electrical input deviation adjusted one at a time to the maximum deviation expected or a value of 90% of that which will cause a failure trip.

Hydraulic Input Deviations "The change of hydraulic pressure inputs with respect to the normal values."

Procedure - Adjust the hydraulic inputs one at a time until the maximum expected deviation or a failure trip level is reached. The performance parameters of Section 2.1 will be measured with each hydraulic input adjusted one at a time to the maximum deviation expected or a deviation value of 90% of that which will cause a failure trip.

FAILURE TRANSIENTS

Electrical Failure Transients "The change in actuator output during failure corrective action due to electronic input failures causing transfer from one operational mode to another."

Procedure - Apply a slowly changing input to one control channel of the actuator. Record the actuator output change during the corrective action of actuator. Repeat the test for each control channel input and failure mode condition. Repeat for a hardover step input.

Apply a sinusoidal input to all channels. Open each input while recording actuator output.

Hydraulic Failure Transients "The change in actuator output during failure removal corrective action due to hydraulic input failures causing transfer from one operational mode to another."

Procedure - Apply a slowly decreasing hydraulic input to one control channel of the actuator. Record the output change during the corrective action of the actuator. Repeat the test for all hydraulic inputs.

Repeat the preceding test with a rapid decrease of hydraulic input pressure.

FAILURE LOGIC DETECTION CHARACTERISTICS

Logic Detection Characteristics "The difference in multiple input time histories which will cause a failure logic to declare a failure."

Procedure (Static Failure Detection Level) - Apply a slowly increasing input to one channel of the system while maintaining the other channel inputs at zero level. The voltage at which the channel is declared failed, expressed as a percentage of the input for maximum position and a percentage of the input for maximum rate is the static failure detection level."

Procedure (Dynamic Failure Detection Level) - Apply a slowly increasing input to one channel of the system at frequencies from DC to a frequency at which the system response is attenuated by at least 15 dB. The other channel inputs are maintained at zero levels. The voltage at which the channel is declared failed, expressed as a percentage of the input for maximum position and a pecentage of the input for maximum rate is the dynamic failure detection level."

IV. SPECIFIC TEST PROCEDURES

SYSTEM SETUP

For all tests except the input deviation tests, the failure detection level was set at channel differences corresponding to 35% of the servovalve stroke. This value was established by performing one complete series of input testing to establish that nuisance disconnects would not occur. For the system operation the failure logic was set to declare a failure after 3 iterations of detecting the failure. Initially, the failure logic was allowed to declare "good" a previously failed channel after it tested good for 9 iterations. However during the failure detection tests, it was discovered that the failure logic would vote incorrectly because of a previously failed channel. Therefore, the failure logic was set so that it would not use a previously failed channel.

In order to allow injecting multiple inputs into the microprocessor it was necessary to change the input method. The system as delivered by Boeing allowed only a single signal input (the same signal input) for all four channels. This limitation was due to the number of A/D converters that had been installed in the microprocessor. In order to investigate the effect of input deviations, the test inputs were summed with the four feedback signals and the normal signal input connected to ground. This input connection method was electrically equivalent to driving each of the four channels with separate inputs.

To investigate the effect of hydraulic failures and deviations on the test system, the two hydraulic supplies were connected through pressure reducing valves.

DEVIATIONS AND/OR ADDITIONS TO THE GENERAL TEST PROCEDURE

Because the microprocessor was not designed as a failure tolerant electrical control device, no testing of the effect of electrical power changes to the microprocessor was conducted.

In order to make the distortion measurements on the test system, a chart recording of the output wave form was made. The harmonic distortion analyzer normally used for this test does not provide reliable distortion measurements at frequencies below 3 Hz. Since the Boeing system frequency response attentuated rapidly above 3 Hz, a chart recorder was used to record waveform fidelity. In evaluating the effect of input deviations, the system was run with D.C. bias inputs. This was done in order to evaluate the effect of null offsets of the servovalves which control the tandem actuator (since no compensation of the force fight between control channels was included in the system).

As part of the test evaluation, the system was run with the output of the actuator subjected to a load force. This test condition was added to investigate the sensitivity of the system to loads, particularly when operating with servovalve null offsets. Two load conditions were used. One condition was with the load system providing a linear spring rate of 10,000 pounds per inch around the test actuator midstroke position. The second load condition was with an applied spring gradient load of 5,500 pounds per inch and the actuator positioned 0.85 inch from midstroke. This created a bias load of 4,675 pounds with a spring rate of 5,500 pounds per inch.

SPECIFIC TEST CONDITIONS

The following list defines test conditions referenced in Table I applied to the actuator during testing. Table I is a list of the specific test conditions used in evaluating the Boeing servoactuator. On Table I the follow applies:

No suffix on test condition number	Actuator unloaded, uncoupled
"A" suffix on text condition number	Actuator connected to load system with the load commanded to zero
"B" suffix on test condition number	Actuator connected to load system with an applied linear symmetrical 10,000lbs/in. load around test actuator midstroke position
"C" suffix on test condition number	Actuator connected to load system with an applied load of 5,500 lbs/in. Actuator positioned 0.85 in. from midstroke, creating a steady bias load of 4,675 lbs.

Test conditions 1 through 20 are operating conditions for the test system. For each operating system, the entire series of performance measurements are run (including the test conditions 1 through 20 which have suffixes A, B, and C).

Test conditions 1 through 4 are baseline tests with the system operating normally.

Test conditions 5 through 8 are designed to evaluate the effect of electrical input failures on the test system. The test conditions are for a single first failure into the various four inputs. After the failure injection, the system operates in a fail operate mode.

Test conditions 9 through 12 are designed to evaluate a failure effect on performance with two channels failed.

Test condition number 13 is operation of the system with one hydraulic failure. Channels 1 and 2 are both powered by the hydraulic system section which is subjected to a failure condition.

Test conditions 14 through 19 are operational conditions of the system with both control and power input deviations. These test conditions allow evaluating the system with a range of inputs corresponding to deviations which would not be detected as failure conditions.

Test conditions 14, 15 and 16 reflect electrical null effects which are less than the null shift which would cause a failure to be declared.

Test conditions 17, 18 and 19 reflect a hydraulic supply pressure reduction from normal system pressure.

Test condition 20 is used to evaluate the system with the normal channel mismatch "nulled" out. Since the system did not use any compensation for null mismatches, this test condition corresponds to the best operating condition attainable with respect to force fight.

Test conditions 21 through 30 (including those with suffix B and C) are the failure transient tests. These test conditions define the method of testing for output changes with specific input failures.

TABLE 1
TEST CONDITIONS BOEING
RECONFIGURABLE FAIL OPERATIVE SERVOACTUATOR

Condition Number	
1	Channels 1 and 3 active - no failures
2	Channels 1 and 4 active - no failures
3	Channels 2 and 3 active - no failures
4	Channels 2 and 4 active - no failures
5	Channels 1 and 3 active, Channel 4 failed, Channel 2 model
6	Channels 1 and 4 active, Channel 2 failed, Channel 3 model
7	Channels 2 and 3 active, Channel 1 failed Channel 4 model
8	Channels 2 and 4 active, Channel 3 failed, Channel 1 model
9	Channels 1 and 2 failed, Channel 3 active, Channel 4 model
10	Channels 1 and 2 failed, Channel 4 active, Channel 3 model
11	Channels 3 and 4 failed, Channel 1 active, Channel 2 model
12	Channels 3 and 4 failed, Channel 2 active, Channel 1 model
13	One hydraulic failure (Channels 1 and 2) (zero psi)
14	Channel 3 - bias to 90% of trip level
15	Channel 3 + bias to 90% of trip level
16	Channel 1 and 3 with opposing input offsets Channel 1 + bias and Channel 3 - bias to 90% of trip level
17	Channels 1 and 2 at 2K psi

TABLE 1
TEST CONDITIONS BOEING
RECONFIGURABLE FAIL OPERATIVE SERVOACTUATOR (CONT'D)

Condition Number	
18	Channels 1,2,3 and 4 at 2K psi
19	Channels 3 and 4 at 2K psi
20	Channels 1 and 3 active - no failures - Bias on channel 3 to pressure null active channels.
21	Ground inputs to channels 2,4,1 sequentially with system initially operating 1A,2M,3A,4M and 50% extend (+4.5 volts at all inputs.)
22	Apply a ramp of zero to 1 volt at 0.4 volt/sec. (±1.0 volt at 0.1 Hz) to channels 1,2,3 sequentially with the system at null. (System initially operating 1A,2M,3A,4M.)
23	Apply a ramp of zero to 1 volt sequentially to channels 1,2,3 with system operating at 1/2 the bandpass frequency with maximum unsaturated input amplitude. (System initially operating 1A,2M,3A,4M.)
24	Ground inputs to channels 1,2,3 sequentially with output at 50% extend and initially operating at 1A,2M,3A,4M.
25	Ground inputs to channels 1,2,3 sequentially with output at 50% retract and initially operating at 1A,2M,3A,4M.
26	Ground inputs to channels 1,2,3 sequentially with system operating at 1/2 the bandpass frequency with maximum unsaturated input. (System initially operating 1A,2M,3A,4M.)
27	Apply +9 volts sequentially to channels 1,2,3 with system at null and operating at 1A,2M,3A,4M.
28	Apply -9 volts sequentially to channels 1,2,3 with system at null and operating at 1A,2M,3A,4M.
29	Apply +9 volts sequentially to channels 1,2,3 with system operating at 1A,2M,3A,4M and 1/2 the bandpass frequency with maximum unsaturated input amplitude.
30	Apply -9 volts sequentially to channels 1,2,3 with system operating at 1A,2M,3A,4M and 1/2 the bandpass frequency with maximum unsaturated input amplitude.
1A, 1B, 1C	Channels 1 and 3 active - no failures
2A, 2B, 2C	Channels 1 and 4 active - no failures

TABLE 1
TEST CONDITIONS BOEING
RECONFIGURABLE FAIL OPERATIVE SERVOACTUATOR (CONT'D)

Condition Number	
3A,3B,3C	Channels 2 and 3 active - no failures
4A,4B,4C	Channels 2 and 4 active - no failures
9B,9C	Channels 1 and 2 failed, 3 active, 4 model
11B,11C	Channels 3 and 4 failed, 1 active, 2 model
14B,14C	Channel 3 -bias to 90% of trip level
15B,15C	Channel 3 +bias to 90% of trip level
16B,16C	Channel 1 and 3 with opposing input offsets Channel 1 +bias, Channel 3 -bias
22B,22C	Apply a ramp of zero to 1 volt at 0.4 volt/sec. ($\pm$ 1.0 volts at 0.1 Hz) to channels 1,2,3 sequentially with system at null. (System initially operating 1A,2M,3A,4M.)
23B,23C	Apply a ramp of zero to 1 volt sequentially to channels 1,2,3 with system operating at 1/2 the bandpass frequency with maximum unsaturated input amplitude. (System initially operating 1A,2M,3A,4M.)
24C	Ground inputs to channels 1,2,3 sequentially with output at 50% extend and initially operating at 1A,2M,3A,4M.
26B,26C	Ground inputs to channels 1,2,3 sequentially with system operating at 1/2 the bandpass frequency with maximum unsaturated input. (System initially operating 1A,2M,3A,4M.)
27B,27C	Apply +9 volts sequentially to channels 1,2,3 with system at null and operating at 1A,2M,3A,4M.
28B,28C	Apply -9 volts sequentially to channels 1,2,3 with system at null and operating at 1A,2M,3A,4M.
29B,29C	Apply +9 volts sequentially to channels 1,2,3 with system operating initially at 1A,2M,3A,4M and 1/2 the bandpass frequency with maximum unsaturated input amplitude.
30B,30C	Apply -9 volts sequentially to channels 1,2,3 with system operating initially at 1A,2M,3A,4M and 1/2 the bandpass frequency with maximum unsaturated input amplitude.

V. TEST RESULTS

GENERAL

The test results presented in this section are arranged in the following order:

A. UNLOADED TEST RESULTS

Static Threshold	Test Conditions 1 through 20*
Dynamic Threshold	Test Conditions 1 through 20
Frequency Response	Test Conditions 1 through 20
Hysteresis	Test Conditions 1 through 20
Saturated Velocity	Test Conditions 1 through 20
Linearity	Test Conditions 1 through 20
Step Response	Test Conditions 1 through 20
Failure Transients	Test Conditions 21 through 30**

*Note that test conditions 1 through 20 include the following sub-groups:

Baseline tests	(Conditions 1 through 4)
Single Electrical Failures	(Conditions 5 through 8)
Dual Electrical Failures	(Conditions 9 through 12)
Hydraulic Failure	(Condition 13)
Input Deviation Effects	(Conditions 14 through 19)
Force Fight Nulling	(Condition 20)

**Note that test conditions 21 through 30 define the procedure used to obtain the failure transient time history.

B. LOADED TEST RESULTS

Static Threshold	Test Conditions 1A through 4A*
Dynamic Threshold	Test Conditions 1A through 4A
Frequency Response	Test Conditions 1A through 4A
Hysteresis	Test Conditions 1A through 4A
Saturated Velocity	Test Conditions 1A through 4A
Linearity	Test Conditions 1A through 4A
Step Response	Test Conditions 1A through 4A

Static Threshold Dynamic Threshold Frequency Response Hysteresis	(Test Conditions 1B,1C through 4B,4C; 9B,9C; 11B,11C; 14B,14C through 16B,16C)
Failure Transients	(Test Conditions 22B,22C 23B,23C,24C 26B,26C through 30B,30C)**

*Note that test conditions with the suffix A are tests with the load system commanded to "0" load.

**Note that these test conditions define the procedure used to obtain the failure transient time history.

C. DISTORTION (OUTPUT/INPUT FIDELITY) TEST RESULTS

The distortion test results are presented as waveform recordings of the input command signal and the output of the position transducer used to measure the actuator position. The data is presented in the following order:

1. Output Fidelity - As a Function of Input Level - Normal System @ 1/2 Bandpass Frequency
2. Output Fidelity - As a Function of Channel Offset Bias - No Load - 10% Input
3. Output Fidelity - As a Function of Channel Offset Bias - Symmetrical Load - 10% Input
4. Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 10% Input
5. Output Fidelity - As a Function of Channel Offset Bias - Symmetrical Load - 3% Input
6. Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 3% Input
7. Output Fidelity - As a Function of Channel Offset Bias - Symmetrical Load - 1% Input
8. Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 1% Input

In order to reduce the volume of test data presented in this section, the majority of the performance measurement data has been reduced to tabulated form. The principal exceptions are the results for step response and failure transients. Since time response characteristics are not well defined by listing only one or two characteristic values, the step response measurements and the failure transient measurements are presented as recorded. Also presented in graphical form is the data taken for the measurement of input/output linearity. The results are presented in tabulated form for the following tests:

1. Static Threshold
2. Dynamic Threshold
3. Frequency Response

4. Hysteresis

5. Saturation Velocity

For the test results reduced to table form, a sample of representative recorded data is included for the test.

In presenting the measurements of threshold and hysteresis, the results are given both in percent of the input for full actuator stroke and percent of the input for full valve stroke. In terms of the full actuator stroke, the percentage value for a given amount of hysteresis reduces as the maximum stroke of the actuator increases. Presenting percentage in terms of the input for maximum control valve stroke shows the threshold and hysteresis characteristics better in terms of comparing different control valve driving mechanizations, independent of the stroke sizing of the power actuator.

SPECIFIC UNLOADED TEST RESULTS

Figure 4 shows the Boeing actuator as mounted for the unloaded tests. Note the two position transducers used to measure the actuator position mounted on the outside of the actuator.

Static Threshold

Figure 5 shows the data recorded in establishing the static threshold for condition 1. Note that the 0.1 Hz ramp input is slowly increasing with increasing time. The threshold value is determined by the first input amplitude where the actuator output starts to respond to the control input. Note that the noise content of the output signal reflects an output change of 0.004 inch peak to peak. The noise is a reflection of the force fight between the servovalves and digital processing causing a small amplitude hunting. The upper edge of the noise shows the actuator responding to the 0.1 Hz input ramp. Table 2 shows the static threshold measured for test conditions 1 through 20. The change in threshold levels is generally a reflection of test conditions which change the force fight, pressure gain or seal friction force levels of the test system.

As shown in Table 2, test conditions 1, 2, 3 and 4, threshold measurements are made with different combinations of active channels and no system failures. There is no change of threshold as a function of the particular combination of active channels, indicating that the initial null conditions of the control valves are reasonably well matched. Null mismatches between the active channels will cause a force fight and a corresponding increase in threshold.

Test conditions 5 through 8 are operation of the system with single channels failed. As compared to no channel failures there is a slight increase in the static threshold. The variation in the threshold reflects the relative force fight and null conditions of the channel combination.

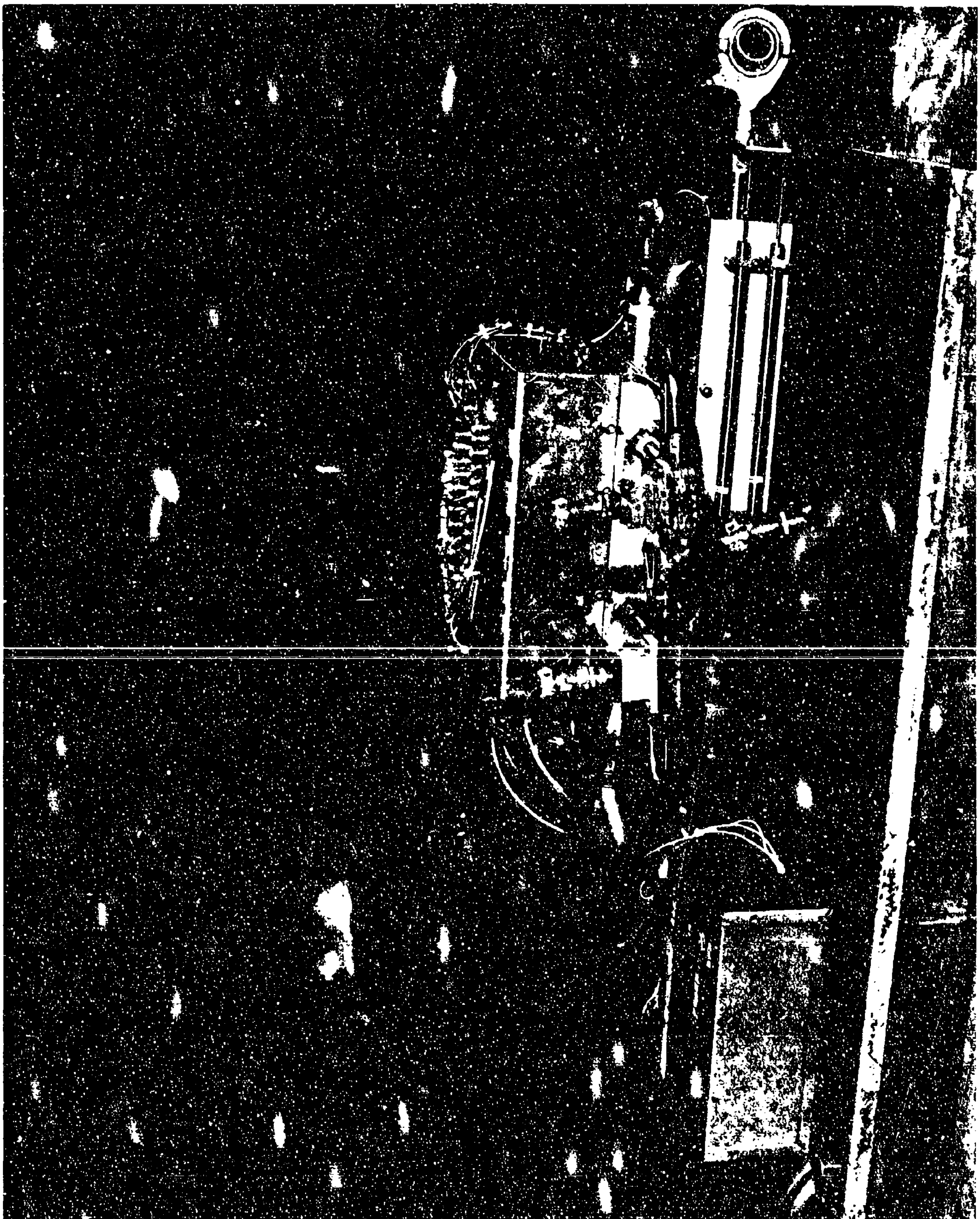
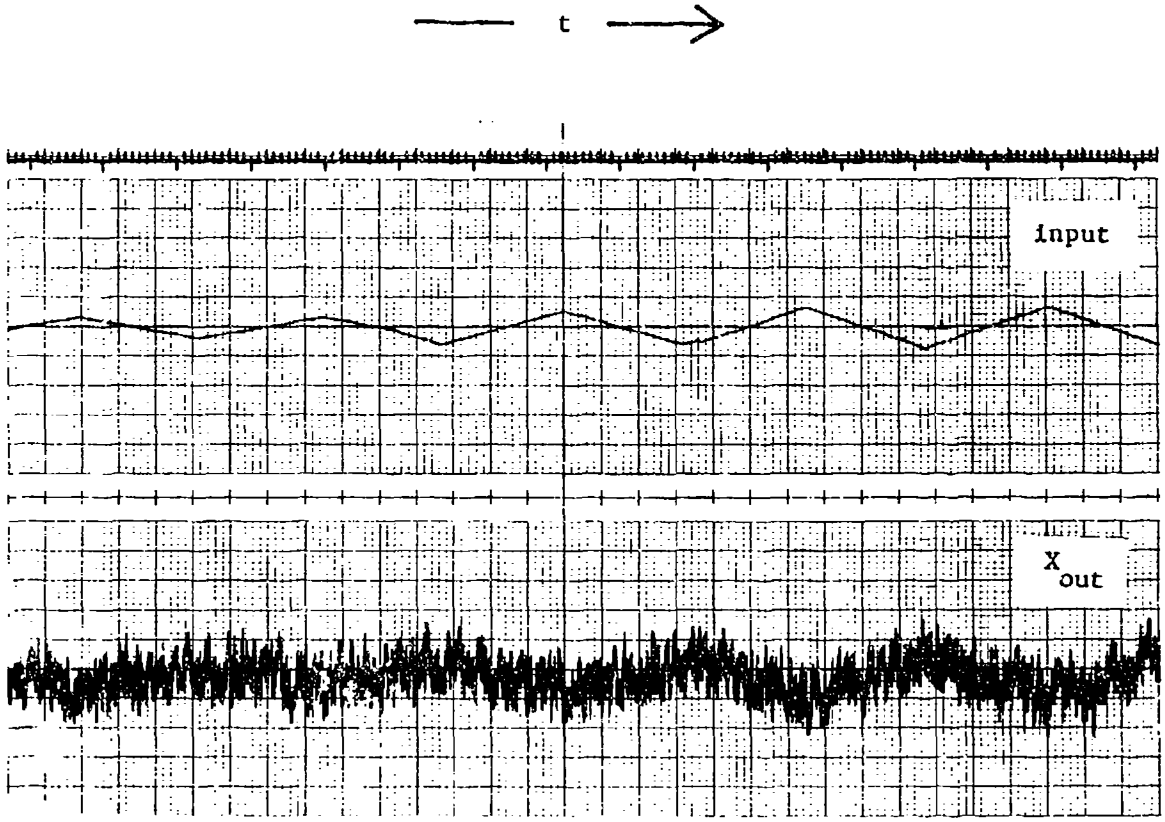


Figure 4. Boeing Actuator Mounted for Unloaded Tests

Date Prepared 9/15/83

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Static Threshold - Condition 1



Scale: Input = 0.002 v/div

X_{out} = 0.000374 in/div

t = 1.0 div/sec

Figure 5. Static Threshold - Condition 1

TABLE 2
STATIC THRESHOLD

TEST ITEM: Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator **DATE PREPARED:** 9/14/83

TEST: Static Threshold

Test Condition	Pk to Pk Input Volts	Static Threshold	
		% of Max Input	% of E_v Max
1	0.008	0.044	0.500
2	0.008	0.044	0.500
3	0.008	0.044	0.500
4	0.008	0.044	0.500
5	0.010	0.056	0.625
6	0.012	0.067	0.750
7	0.012	0.067	0.750
8	0.014	0.078	0.875
9	0.014	0.078	0.875
10	0.014	0.078	0.875
11	0.010	0.056	0.625
12	0.010	0.056	0.625
13	0.017	0.094	1.063
14	0.009	0.050	0.563
15	0.014	0.077	0.875
16	0.008	0.044	0.500
17	0.008	0.044	0.500
18	0.006	0.033	0.375
19	0.006	0.033	0.375
20	0.006	0.033	0.375

Test conditions 9 through 12 are measurements of the static threshold with two channels failed. There is no significant difference between these test conditions and the baseline and single failure measurements.

The static threshold with one hydraulic failure, test condition 13, is above 1% of the input for maximum spool position. This is considerably greater than the threshold measured for the baseline test condition 1 and reflects a reduction in the force gain in relation to the seal friction for the actuator.

The effect of the channel bias levels on static threshold as measured for test conditions 14 through 16 is not significant. For example, with test condition 14, the bias is in a direction which does not increase the threshold over the baseline null mismatch. However, with test condition 15, the bias direction does increase the static threshold.

Test conditions 17 through 19 are used to evaluate the affect of reduced hydraulic supply pressure to the test system. The effect of the supply pressure reduction is a slight reduction in the static threshold as compared to the baseline measurements. This is consistent with a reduction in seal friction with a reduction in hydraulic pressure used in the actuator.

Test condition 20 with the active channels nulled is the best operating condition for the test system and yields a static threshold of 75% of the baseline threshold. This threshold value is within to the 0.5% value typically available on electrohydraulic servovalves.

Dynamic Threshold

Figure 6 shows the data recorded in establishing the dynamic threshold for condition 1. The input is a nominal 2 Hz sinusoidal signal with the input amplitude increasing with increasing time. The amplitude is increased by a factor of 2 over a minimum time of 5 seconds. The point at which the actuator motion starts to track the amplitude increase of the input is used as the dynamic threshold point. Note that the output moves at the nominal 2 Hz frequency in phase with the input over most of Figure 6. However, the output amplitude does not increase with the input until a peak input amplitude of 0.050 volt is reached. As with the static threshold testing, changes in threshold levels generally are reflections of test conditions which change the force fight, pressure gain or seal friction force levels of the test actuator.

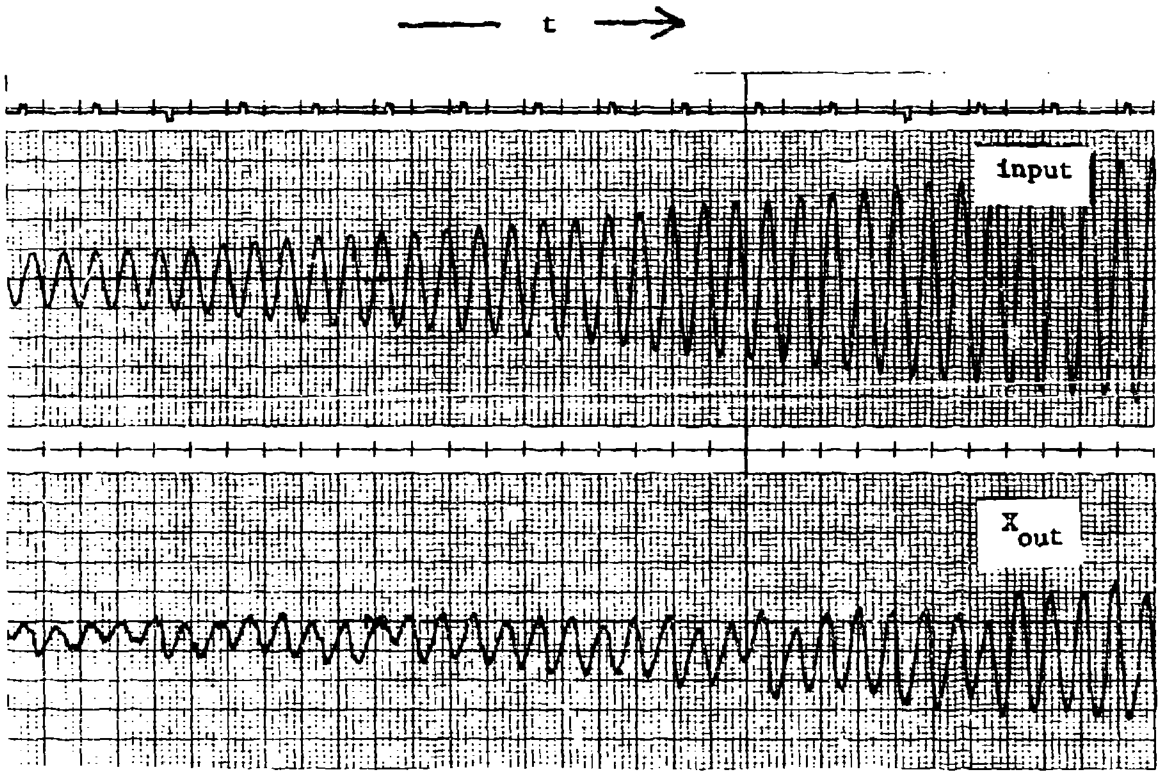
Test conditions 1, 2, 3 and 4 on Table 3 are the dynamic threshold measurements with different combinations of active channels and no system failures. There are only minor changes of dynamic threshold with the different combinations of active channels. This indicates that the dynamic response of the control channels are well matched over the frequency bandpass of the test actuator.

Test conditions 5 through 8 which evaluate the system's dynamic threshold after one electrical failure, show an increase in dynamic threshold for only test condition 8. The increase reflects the relative force fight and null condition of test condition 8's particular channel combination.

Date Prepared 9/14/83

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Dynamic Threshold - Condition 1



Scale: Input = 0.005 v/div

X_{out} = 0.000935 in/div

t = 10 div/sec

Figure 6. Dynamic Threshold - Condition 1

TABLE 3
DYNAMIC THRESHOLD

TEST ITEM: Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator   **DATE PREPARED:** 9/14/83

TEST: Dynamic Threshold

Test Condition	Pk to Pk Input Volts	Dynamic Threshold	
		% of Max Input	% of E_v Max
1	0.050	0.278	3.125
2	0.058	0.322	3.625
3	0.060	0.333	3.750
4	0.055	0.306	3.438
5	0.063	0.350	3.938
6	0.060	0.333	3.750
7	0.070	0.389	4.375
8	0.093	0.517	5.813
9	0.050	0.278	3.125
10	0.040	0.222	2.500
11	0.050	0.278	3.125
12	0.050	0.278	3.125
13	0.053	0.294	3.313
14	0.033	0.184	2.063
15	0.028	1.156	1.750
16	0.050	0.278	3.125
17	0.065	0.361	4.063
18	0.075	0.417	4.688
19	0.050	0.278	3.125
20	0.040	0.222	2.500

Test conditions 9 through 12 are dynamic threshold measurements with two electrical channels failed, leaving one servovalve bypassed and one in command of the flow to the actuator. The dynamic threshold values are the same or slightly lower than the baseline values. This indicates that there is not much mismatch between the two servovalve sections. A significant improvement in dynamic threshold with only one channel operating would indicate a significant force fight between the servovalve sections of the test system.

The hydraulic failure test (test condition 13) produces a dynamic threshold similar to the two electrical failure conditions (conditions 9 through 12). This can be expected, since in both cases the test condition is with only one half of the actuator operating.

The effect of the channel bias conditions (test conditions 14 through 16) on the dynamic threshold is not significant. Compared to the baseline dynamic threshold of test conditions 1 through 4, the input bias can either improve or degrade the dynamic threshold slightly. For example, test condition 14 with channel 3 biased with a negative input to 90% of the trip level input reduces the dynamic threshold to 2/3 that of the baseline value. The dynamic threshold measured with a positive bias input into channel three is also lower than the baseline, indicating that the dynamic threshold was not dependant on the particular bias level and polarity used. The double bias of test condition 16 yielded a nominal threshold the same as the baseline.

The effect of reducing either one or both of the hydraulic supply pressures to 2000 psi (test conditions 17, 18 and 19) increases the dynamic threshold slightly compared to the baseline conditions for conditions 17 and 18. The dynamic threshold measured for test condition 19 is similar to the baseline measurement. The increase is consistent with the reduction in the pressure/flow gain of the servovalves which results from a decrease in supply pressure to the servovalves.

Test condition 20 with the active channels nulled (a best operating condition) yields a dynamic threshold of 80% of the best baseline measurement. This result is to be expected. A nulled operating condition of the servovalves gives the highest pressure/flow gain for the servovalves operating together.

Note that the dynamic threshold values are nominally 5 times the values for the static threshold. This is due to the dynamic threshold measurement requiring flow from the valve as well as pressure. This effectively reduces the servovalve pressure gain, increasing the input level required to overcome the force fight and seal friction effects.

Frequency Response

Figure 7, the frequency response measured with test condition 1, is representative of the data obtained for all the unloaded frequency response measurements. Note that the actuator output motion (at 0 dB amplitude) is 10 percent of the full stroke of the actuator. The input level corresponding to the 10% output motion met the criteria of minimizing the effect of threshold and hysteresis on the frequency response measurement and of being below the level at which rate saturation occurs.

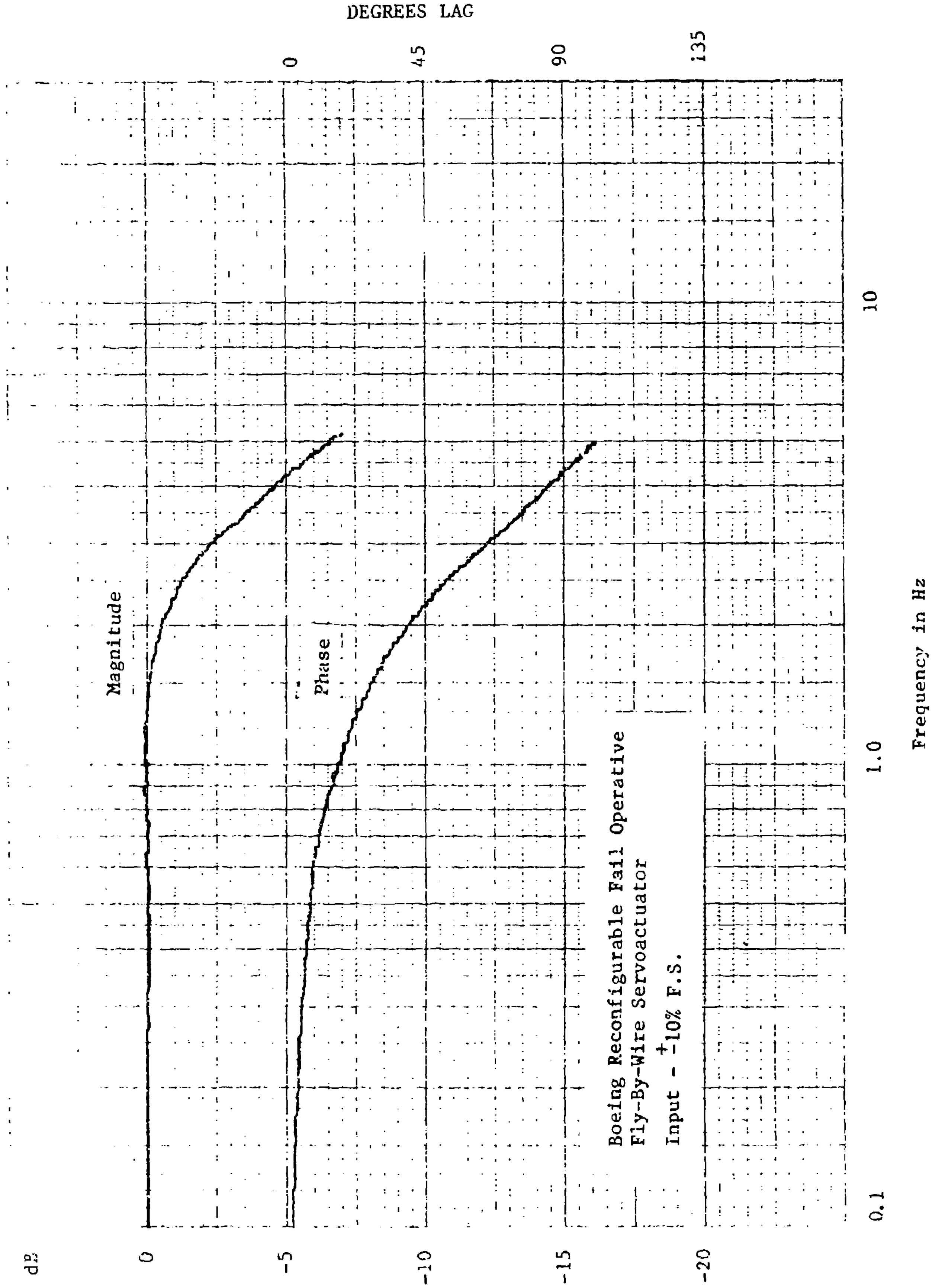


Figure 7. Frequency Response - Condition 1

Table 4 lists the frequency response for test conditions 1 through 20 in terms of the frequencies at which the -90° phase angle and the -3 dB amplitude ratio point occurred. Because the shape of the frequency response curves for all test conditions were similar (no amplitude peaking and a phase lag of less than -90° at the -3 dB frequency), the table listing provides valid indication of the response change with the different test conditions.

Note that as shown in Table 4, the change of frequency response with change of test conditions is quite small. The greatest change from the nominal baseline values occurs with the bias changes of condition 14, 15 and 16. Condition 16 is the only test condition where the -3 dB frequency occurs below 3.00 Hz. The range of variation for the -3 dB amplitude frequency is from 2.80 to 3.50 Hz for all test conditions. The range of variation in the -90° phase angle is from 3.2 to 4.2 Hz. These ranges are nominally 25% of the baseline values. Note that in comparison, the baseline test conditions generate a variation in the -3 dB frequency of from 3.00 to 3.40 Hz. (a nominal change of 13%).

Hysteresis

Figure 8 shows the test data taken for the hysteresis measurement with test condition 1. Table 5 lists the measured hysteresis for test conditions 1 through 20. The data shown on Figure 8 was obtained with an input variation of $\pm$10% of the input for maximum actuator stroke. Note that the hysteresis plot shows an output which is irregular. The plus direction and minus direction output curves separate and then coincide with small changes in input command. The hysteresis measurement as defined by "the difference between + direction actuator output position and - direction output position for the same input level" therefore refers to a local condition of input level. The irregularity where the + and - direction motions coincide is a linearity measurement, not a hysteresis. Figure 8 is representative of the hysteresis data for test conditions 1 through 8. These conditions operate with both servovalves active.

Test conditions 9 through 14 are with only one servovalve operating. For these conditions there is no force fight. The hysteresis plots for these test conditions resemble Figure 8 with smaller differences between the + and - direction motion.

Figure 9 shows the test data taken for the hysteresis measurements of Condition 15. The plot shows well separated + and - direction position lines with a slightly larger difference than with test conditions 1 through 14. The greater separation is consistent with the bias change of condition 15 increasing the null mismatch between the two controlling servovalves.

Figure 10 shows the test data for the hysteresis measurement for condition 16 with a different bias condition. The data shows an increase in the "non-linearity" of the position changes and a decrease in the separation between the + and - direction curves (compared to the baseline tests and other bias tests). This verifies that the hysteresis (and linearity) of the system are functions of the null conditions of the servovalve channels.

TABLE 4
FREQUENCY RESPONSE

TEST ITEM: Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator   **DATE PREPARED:** 9/14/83

TEST: Frequency Response

Test Condition	Output/Input -3 dB Hz	Output/Input -90° Hz
1	3.40	4.20
2	3.00	4.00
3	3.10	4.00
4	3.00	4.00
5	3.30	4.00
6	3.30	4.00
7	3.20	3.90
8	3.40	4.00
9	3.50	4.00
10	3.50	4.00
11	3.50	4.00
12	3.30	3.70
13	3.50	3.80
14	3.20	3.80
15	3.00	3.20
16	2.80	3.30
17	3.00	3.60
18	3.10	3.50
19	3.10	4.00
20	3.20	3.80

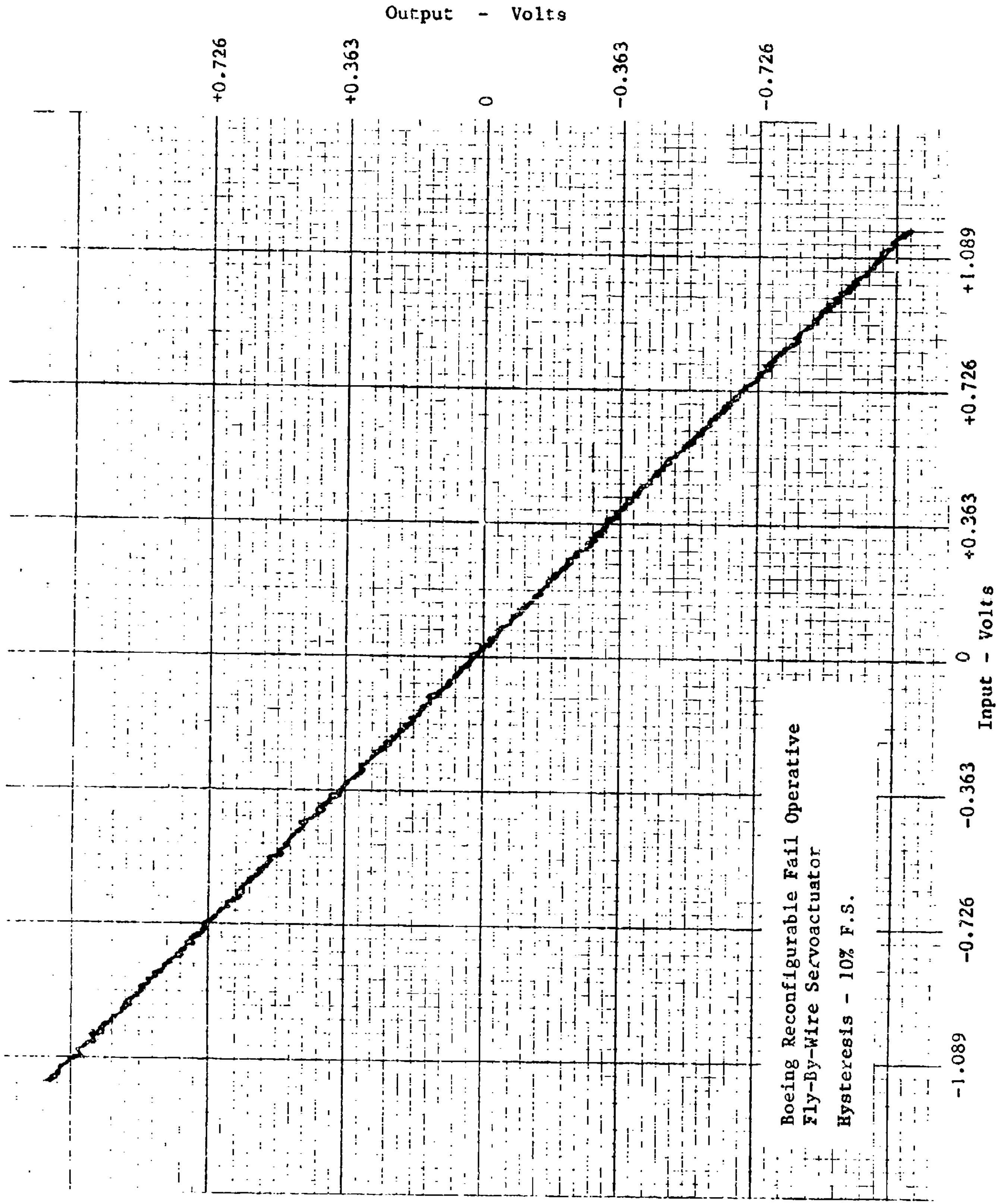


Figure 8. Hysteresis - Condition 1

TABLE 5
HYSTERESIS

TEST ITEM: Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator **DATE PREPARED:** 9/14/83

TEST: Hysteresis

Test Condition	% Full Scale	% of E_v Max
1	0.062	0.69
2	0.062	0.69
3	0.062	0.69
4	0.062	0.69
5	0.082	0.92
6	0.082	0.92
7	0.082	0.92
8	0.082	0.92
9	0.041	0.46
10	0.041	0.46
11	0.041	0.46
12	0.041	0.46
13	0.041	0.46
14	0.041	0.46
15	0.167	1.84
16	0.082	0.92
17	0.041	0.46
18	0.082	0.92
19	0.082	0.92
20	0.041	0.41

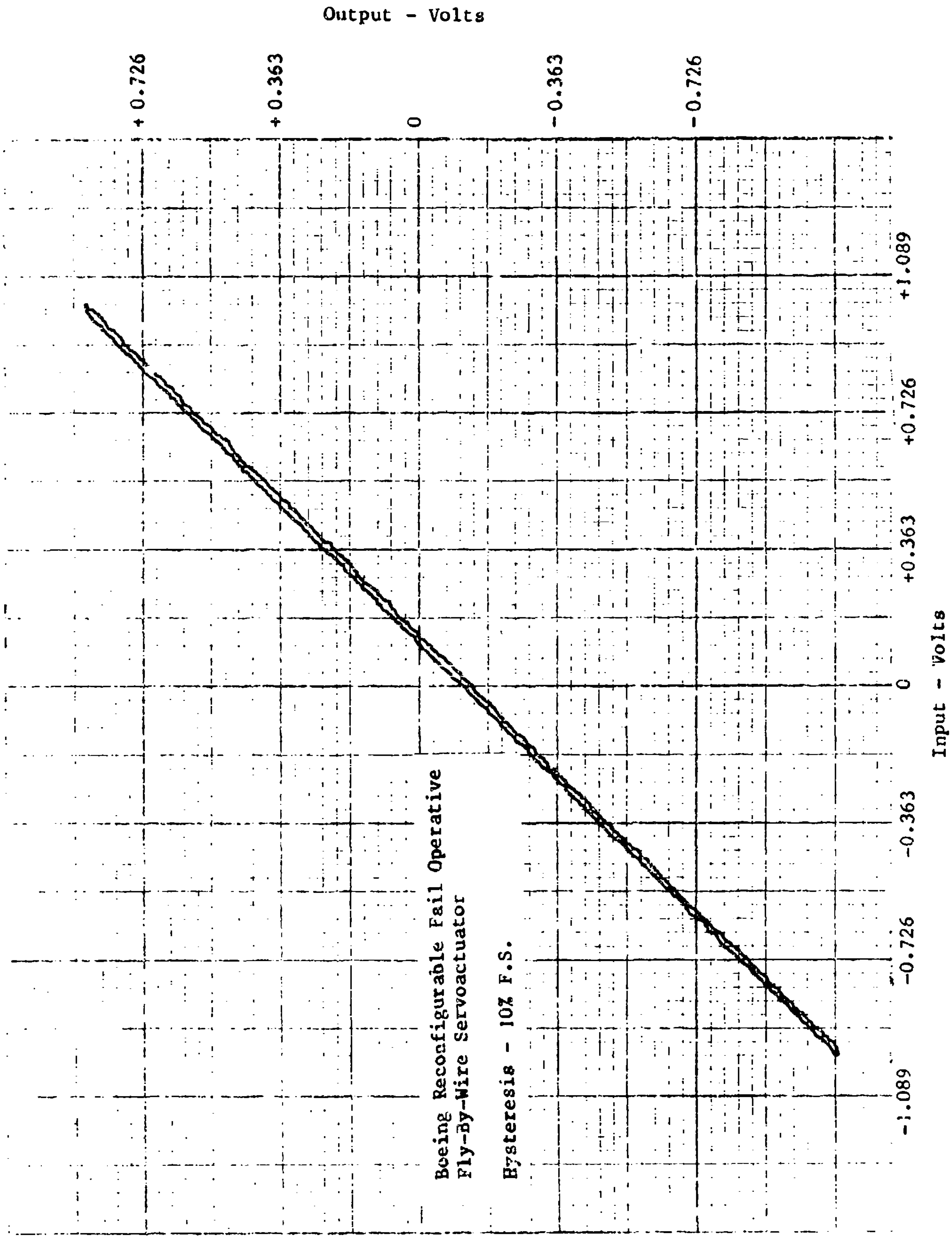


Figure 9. Hysteresis - Condition 15

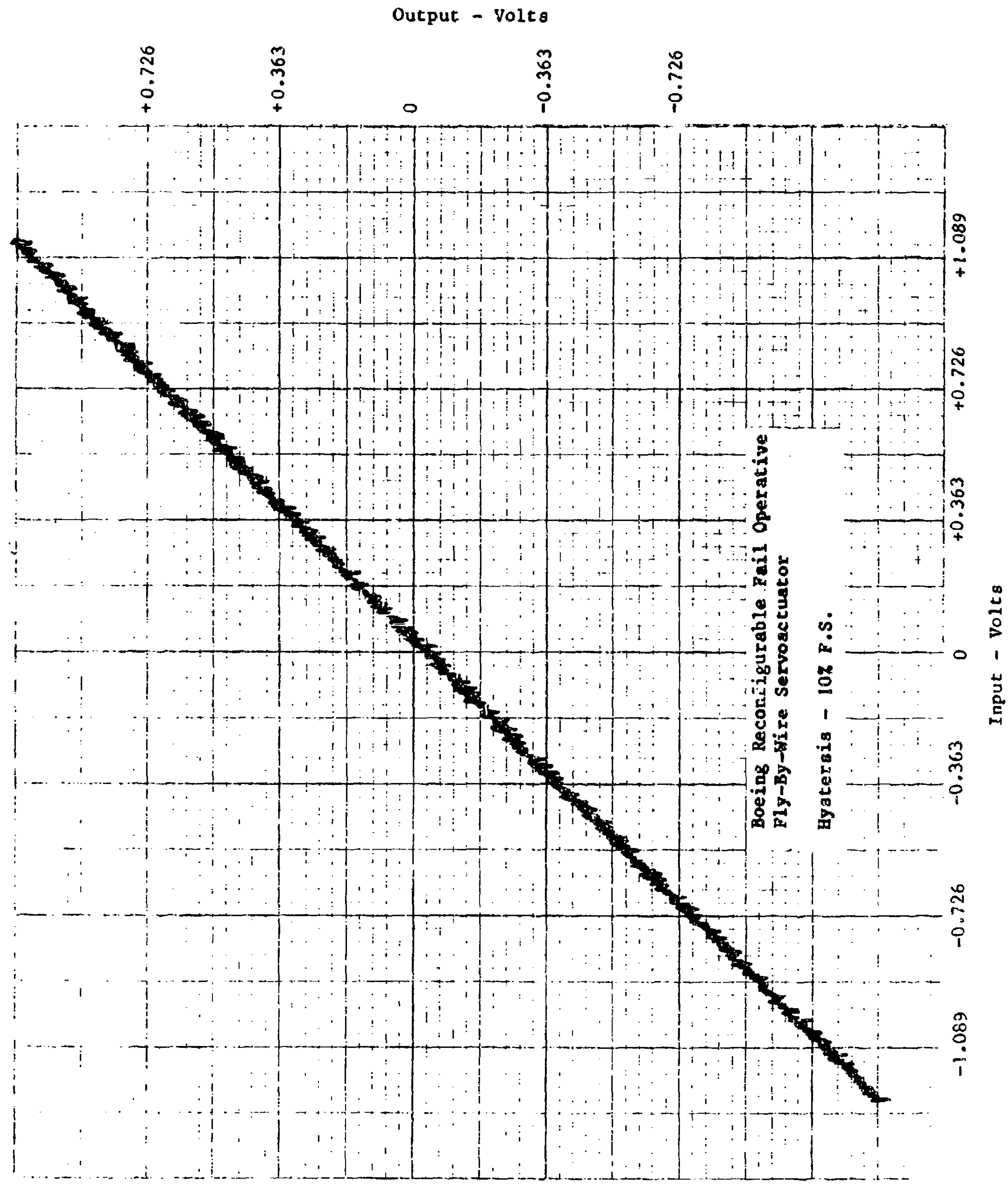


Figure 10. Hysteresis - Condition 16

The effect of the degradation of hydraulic supply pressure on the hysteresis (test conditions 17, 18 and 19) is not significant. The hysteresis measurements are similar to that obtained with the baseline and single electrical failure operating conditions. There is a slight increase with two-channel hydraulic supply pressure degradation (test condition 18) as compared with a single-channel pressure degradation (test condition 19).

Test condition 20 (with the channels operating with the best null condition) gave hysteresis similar to that obtained with single servovalve operation. This is consistent, since in both test conditions all force fight between sections has been eliminated.

As expressed in terms of the voltage for maximum servovalve position, the hysteresis is consistent with current electrohydraulic two stage valves. However, conventional hysteresis loops were not obtained for most test conditions. It appears that the calculated hysteresis values are lowered by the effect of the small irregularity in the position linearity.

Saturated Velocity

Figure 11 shows the data recorded in measuring the maximum velocity for the actuator for test condition 1. This figure is representative of the data obtained for all the test conditions. As shown on Figure 11, a step command input is applied as an input to all control channels. The actuator responds after a short time delay by moving at maximum velocity until it reaches its mechanical stroke limit. Note that the actuator starts from either full extend or full retract position and moves through the full stroke. The saturated velocity is calculated from the data as the slope of the actuator's output motion (displacement vs time). Table 6 lists the calculated values as obtained from the chart data. Small variations in the calculated rates (0.1 in/sec or less) can be considered measurement error.

As shown on Table 6, the saturated velocity remains relatively unchanged from the baseline test conditions (1, 2, 3 and 4) for all test conditions. The retract saturation velocity was slightly lower (by 15%) than the extend velocity for all test conditions. This was probably due to a slight difference in the servovalve's hardover output flows (since the actuator drive areas were all identical).

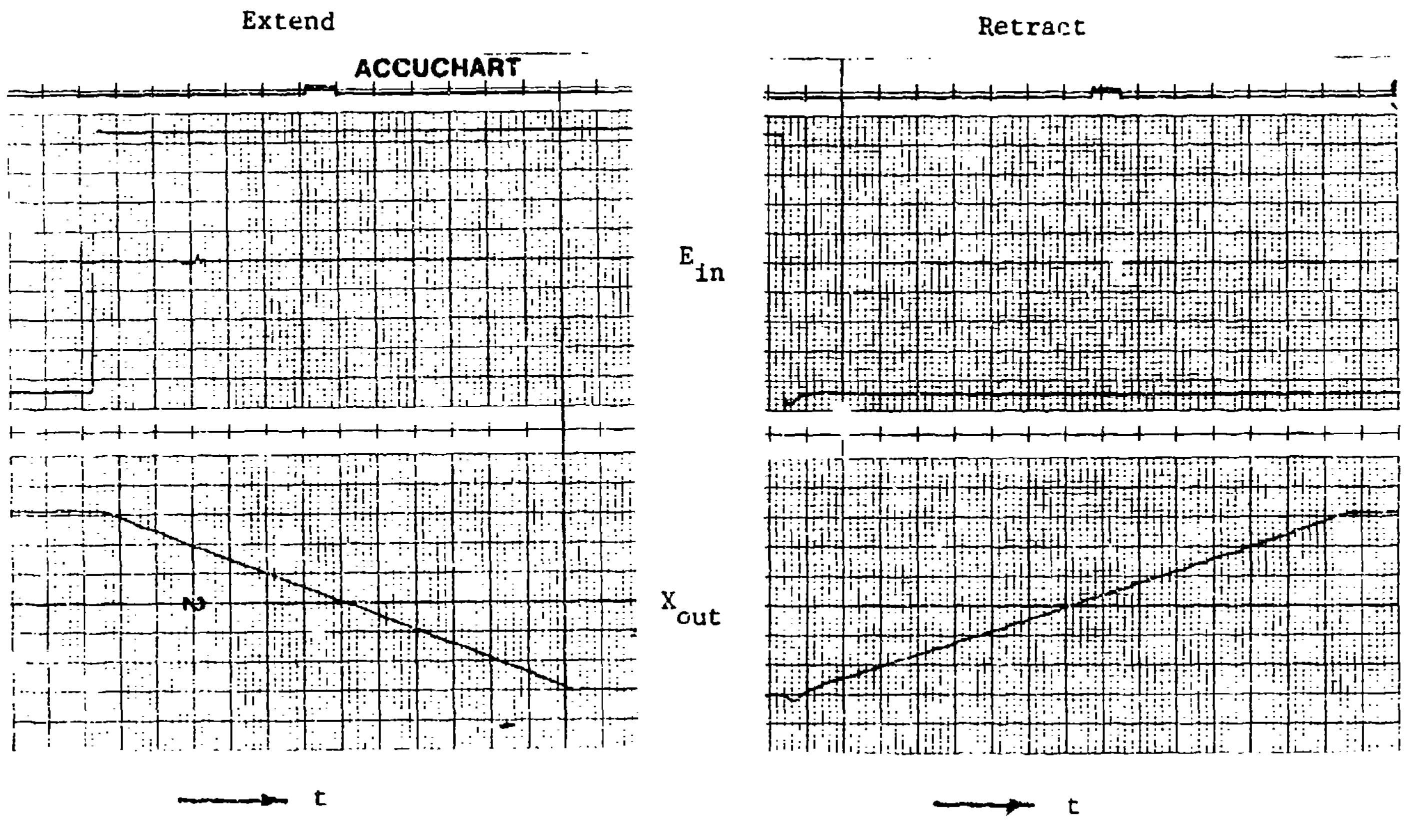
The extend velocity varied from 2.44 in/sec to 2.86 in/sec over the range of test conditions 1 to 20. The difference in the saturated rates for test conditions 1 through 4 indicate measurement error. The servovalves were hardover for all 4 conditions and no difference between different test condition rates in one direction would normally be expected. Since the actuator output for the test conditions of Table 6 is unloaded, the loss of one servovalve channel should have negligible effect on the maximum actuator rate. This is confirmed by the test results for conditions 9 through 13.

The effect of servovalve bias shifts on saturated rate would normally be negligible. The input command is large enough that the servovalves are driven hardover against their mechanical stops, eliminating any effect of input bias levels. This is confirmed by the test results from test conditions 14, 15 and 16 which are in the range of the results from the baseline test conditions 1 through 4.

Date Prepared 9/15/83

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Saturation Velocity - Condition 1



Scale:

Input	=	0.500 v/div
X_{out}	=	0.0935 in/div
t	=	50 div/sec

Figure 11. Saturation Velocity - Condition 1

TABLE 6
SATURATION VELOCITY

TEST ITEM: Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator **DATE PREPARED:** 9/15/83

TEST: Saturation Velocity

Test Condition	Extend - in/sec	Retract - in/sec
1	2.76	2.37
2	2.62	2.24
3	2.64	2.36
4	2.59	2.29
5	2.76	2.36
6	2.67	2.36
7	3.76	2.36
8	2.76	2.54
9	2.62	2.54
10	2.67	2.54
11	2.54	2.21
12	2.54	2.36
13	2.76	2.54
14	2.62	2.36
15	2.76	2.27
16	2.86	2.54
17	2.62	2.29
18	2.44	2.12
19	2.44	2.12
20	2.76	2.36

There is a slight degrading of the actuator rate with a reduction in the hydraulic supply pressure. This is shown by the results of test condition 17. The flow from the servovalves is a function of the square root of the supply pressure. A reduction in supply pressure reduces the flow from the valve and, thereby, the maximum actuator rate.

Linearity

Figure 12 shows the data recorded in measuring the output-linearity of the test system for the system operating in test condition 1. The measured results are representative of the results obtained for test conditions 2 through 20. Since the test system actuator acts as an integrator of flow, the measured linearity is primarily a measure of the actuator position feedback transducer's linearity. Threshold and hysteresis can affect the linearity curve if they have large values. However, for linearity curves reflecting 100% actuator stroke, the amount of hysteresis (in terms of the maximum actuator stroke) would have to be on the order of the rated linearity of the position transducer. For the test system, the linearity rating of the position transducer was 0.5% of the full scale output. Since the threshold (Reference Table 2) and the hysteresis (Reference Table 5) were both below 0.1% for all test conditions, the linearity would not be expected to change with a change of test conditions from 1 through 20. This was observed from the test measurements. Figure 12 accurately represents linearity for any of the test conditions 1 through 20.

Step Response

Figures 13 through 22 show the time history of the test system response to retract and extend step inputs for test conditions 1 through 22. The amplitude of the step voltage change applied as an input to the test system is nominally 1.8 volts. This input causes the actuator output to change by 10% of its stroke range.

Note that the general response of the test system to the step input is initially a straight-line ramp. The final response into the commanded position is a smooth approach at a decreasing rate. This is consistent with the amplitude of step input applied. An error voltage (the difference between the command and actuator position feedback voltage) of 0.800 volt is sufficient to move the servovalve spool to a position stop. For a step input voltage of 1.80 volts, the servovalve spool is held hardover on its stop until the actuator has moved enough to generate a -1.00 volt feedback signal. Therefore the actuator initially moves at a constant rate to the 1.8 volt input step, as demonstrated by the test results. The first 55% of the actuator step reponse is at maximum rate. For the remainder of the response to the step input, the servovalve is not saturated and the response reflects the effect of the control loop dynamics. The final response approach for all test conditions exhibits no overshoot or ringing. This is consistent with the frequency response test data, which showed no peaking.

As shown on the time response data for all test conditions, there is a minor difference in the retract direction and extend direction step response. The initial reponse of the test system for the extend direction motion is made up of a time delay of 1 millisecond with no measured change of the actuator position and a subsequent 2.5 millisecond time period where the actuator moves

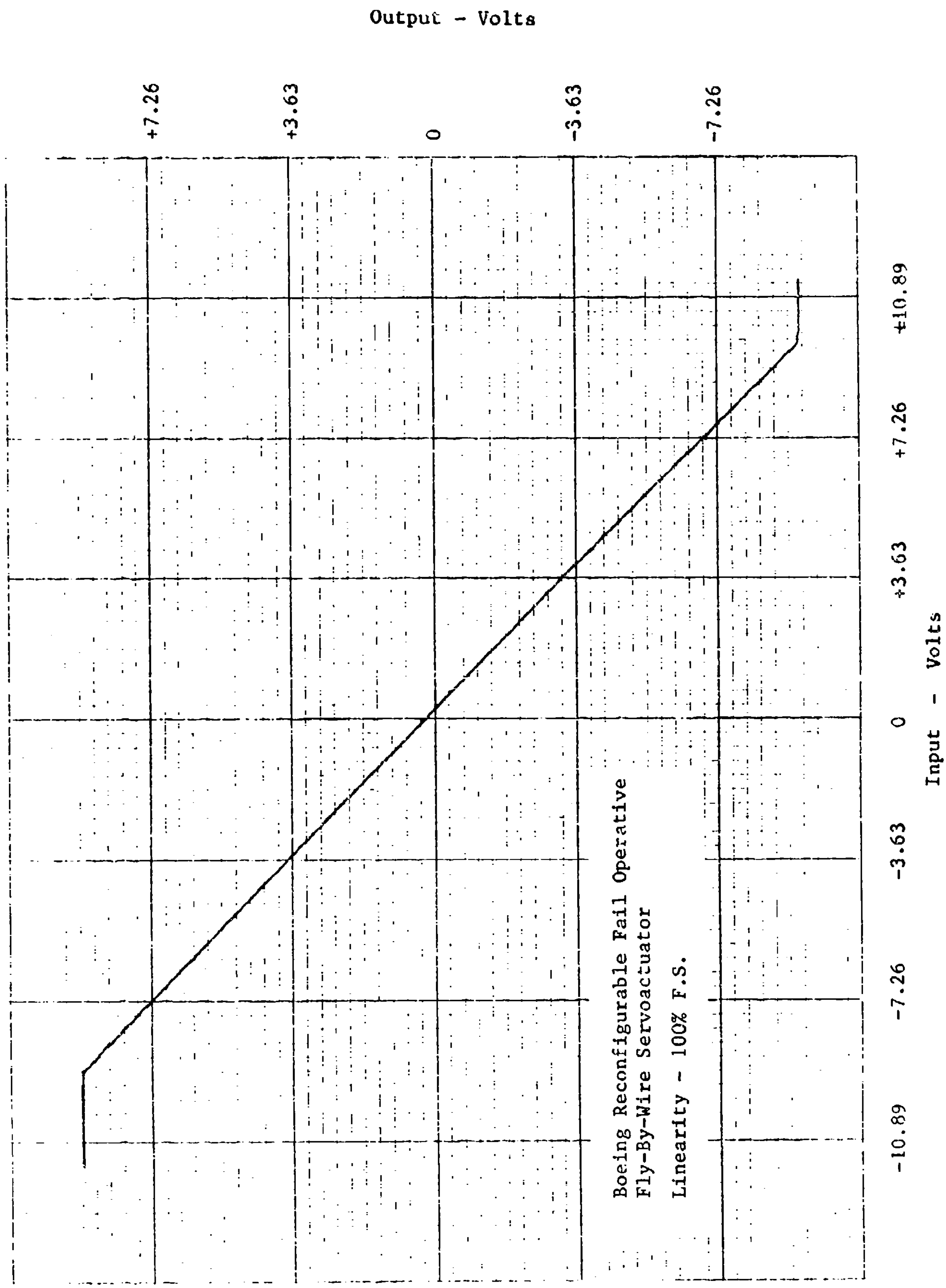


Figure 12. Linearity - Condition 1

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Step Response - Condition 1 and 2 Date Prepared 9/21/83

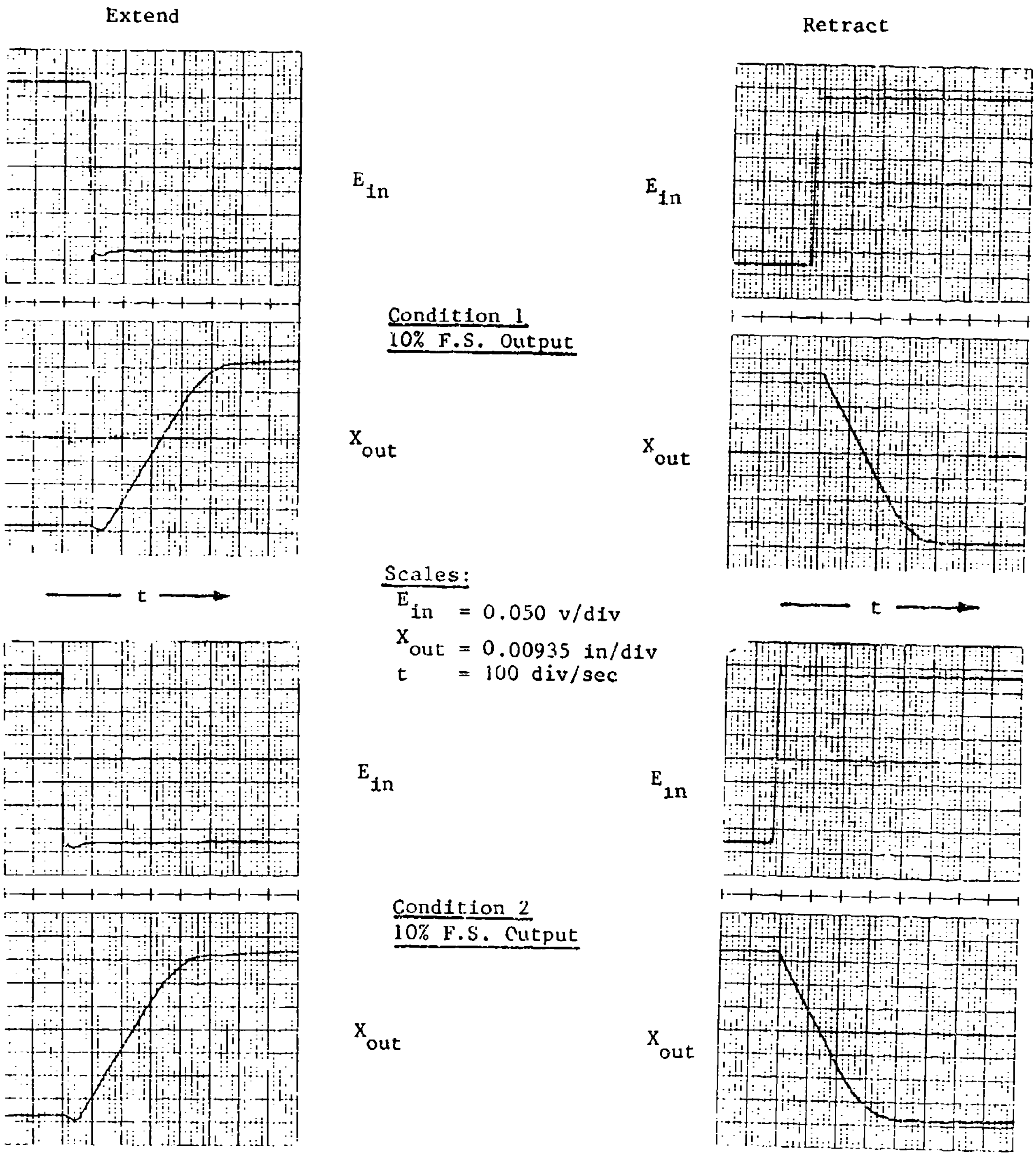


Figure 13. Step Response - Conditions 1 & 2

TEST ITEM - Boeing Reconfigurable Fail Operative
Fly-By-Wire Servoactuator

TEST - Step Response - Condition 3 and 4 Date Prepared 9/21/83

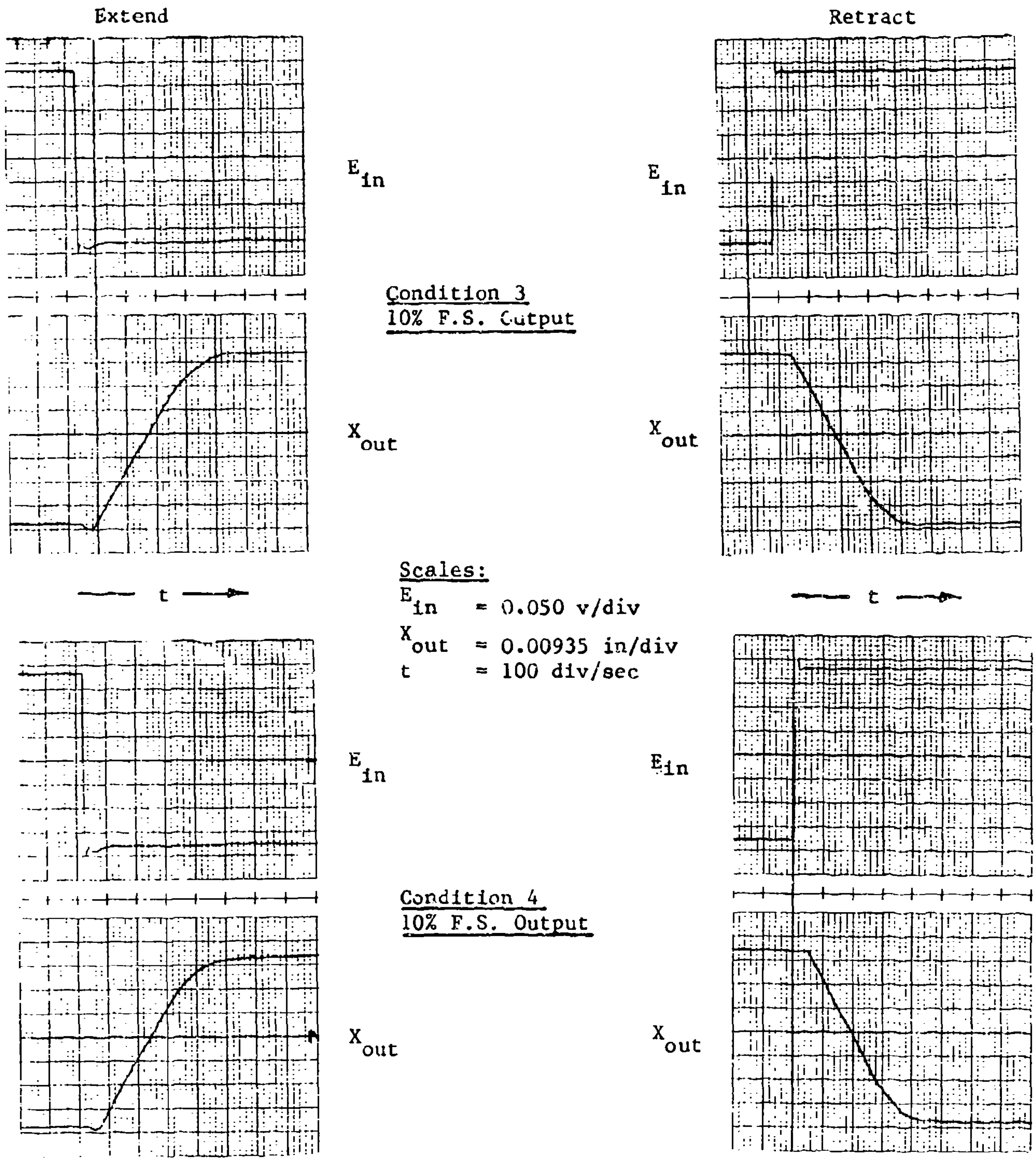


Figure 14. Step Response - Condition 3 & 4

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Step Response - Condition 5 and 6 Date Prepared 9/21/83

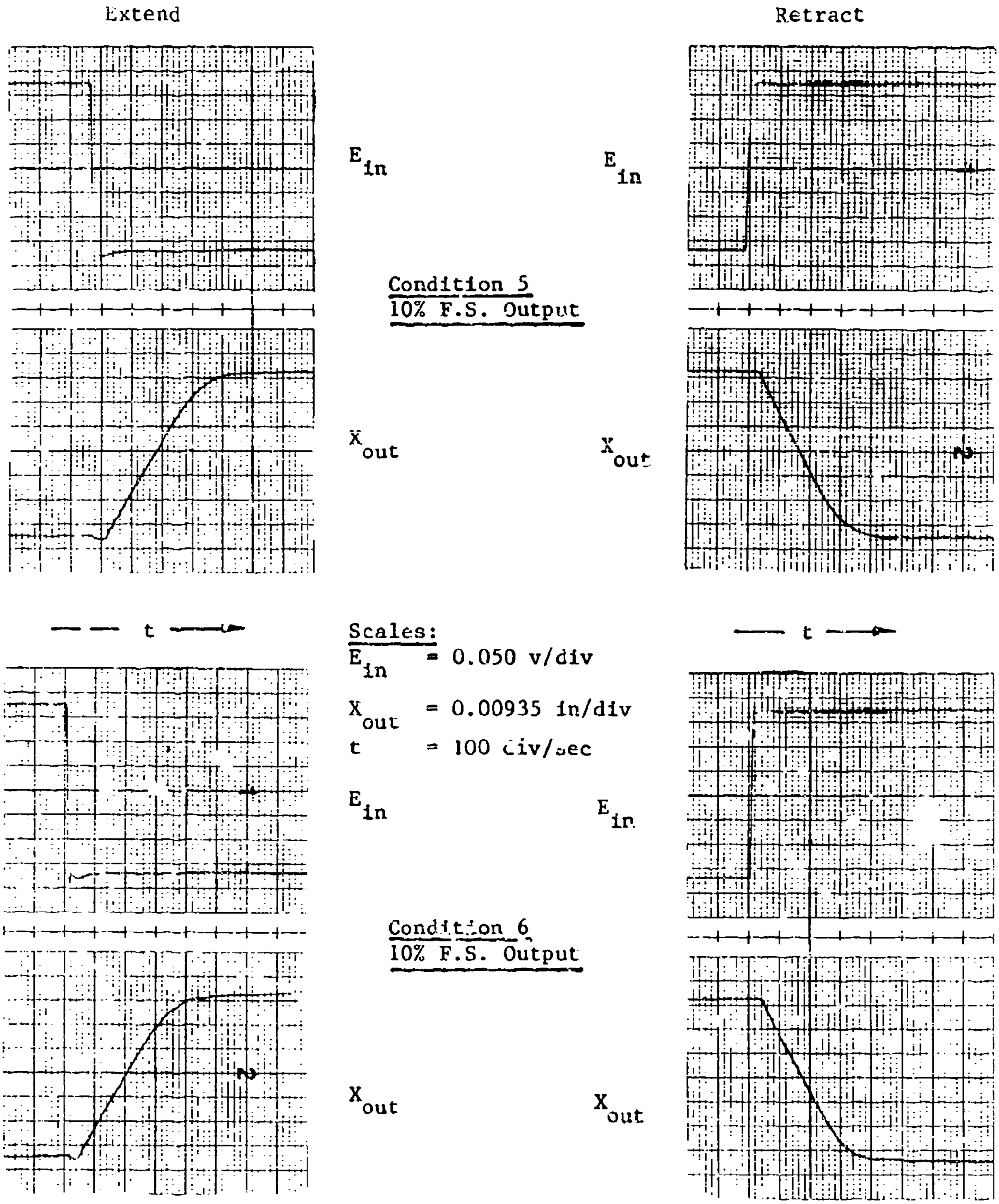


Figure 15. Step Response - Condition 5 & 6

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Step Response - Condition 7 and 8 Date Prepared 9/21/83

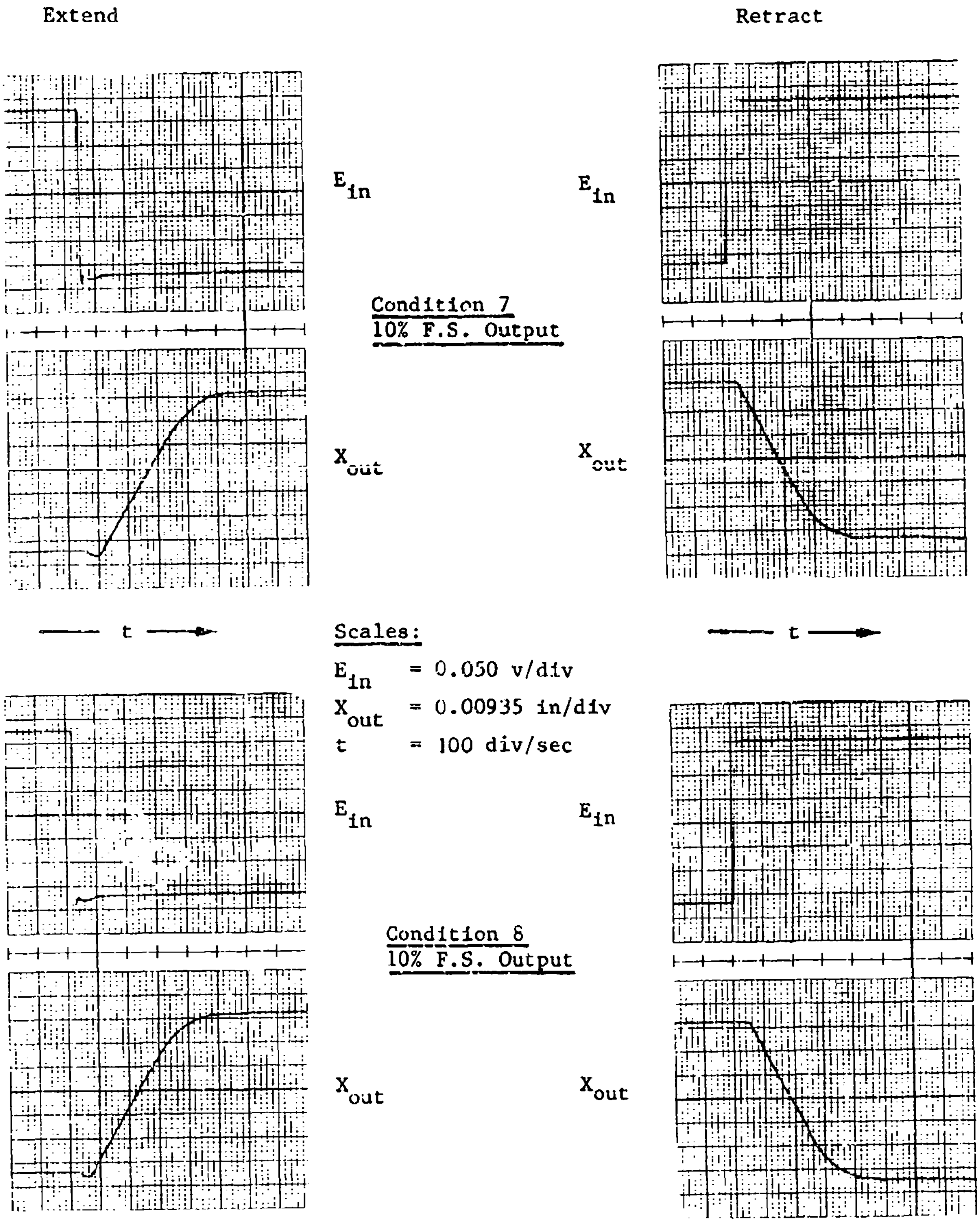


Figure 16. Step Response - Condition 7 & 8

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Step Response - Condition 9 and 10 Date Prepared 9/21/83

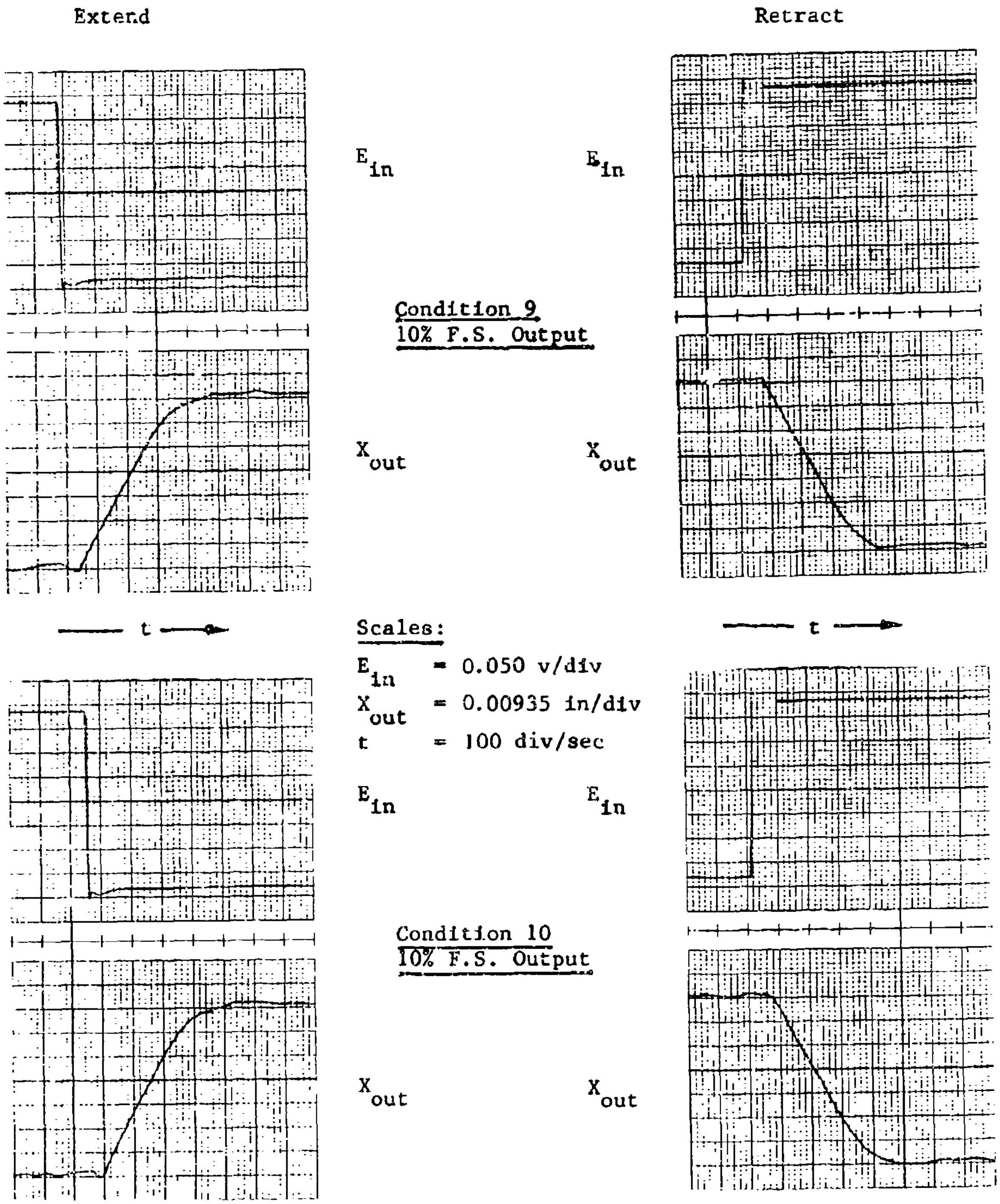


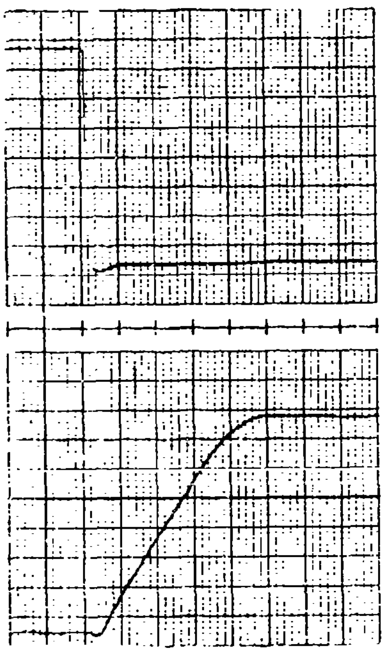
Figure 17. Step Response - Condition 9 & 10

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

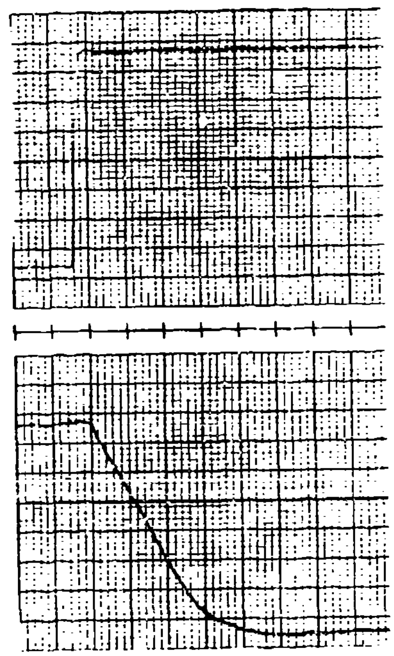
TEST - Step Response - Condition 11 and 12 Date Prepared 9/21/83

Extend

Retract



E_{in} E_{in}



Condition 11
10% F.S. Output

X_{out} X_{out}

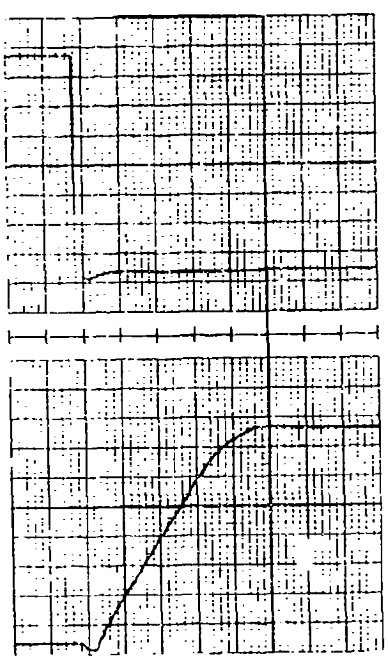
Scales:

E_{in} = 0.050 v/div

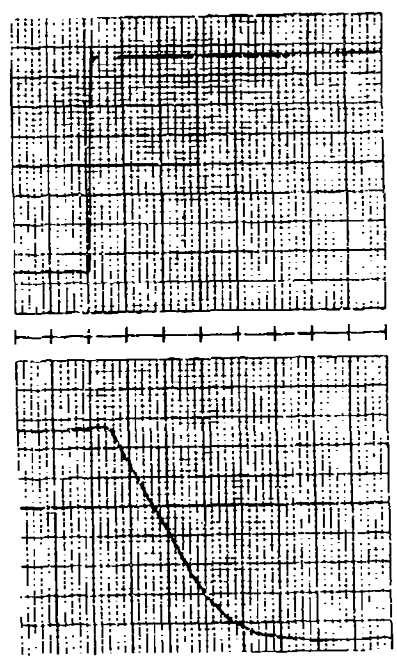
X_{out} = 0.00935 in/div

t = 100 div/sec

—— t ——→ —— t ——→



E_{in} E_{in}



Condition 12
10% F.S. Output

X_{out} X_{out}

Figure 18. Step Response - Condition 11 & 12

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Step Response - Condition 13 and 14 Date Prepared 9/21/83

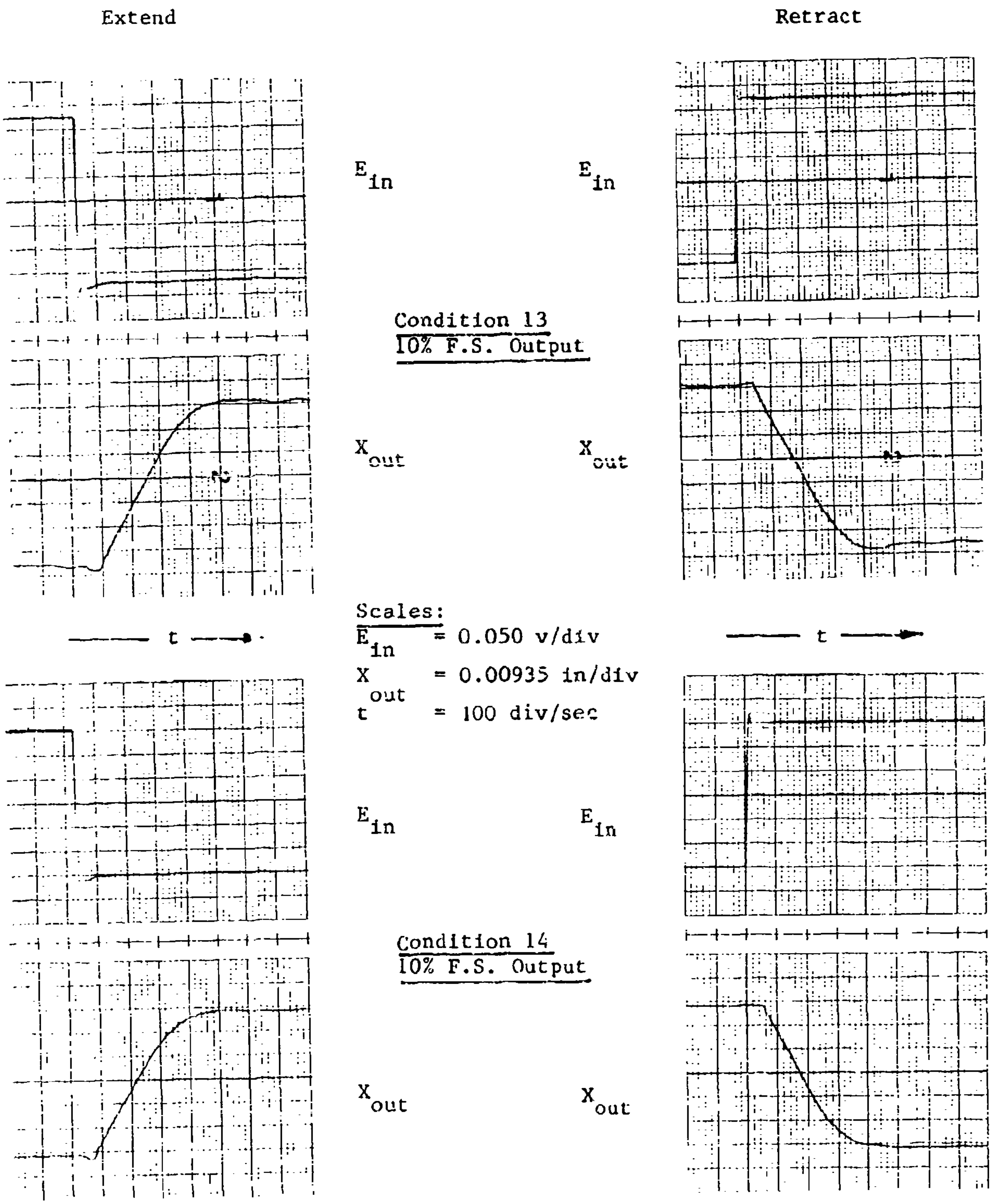


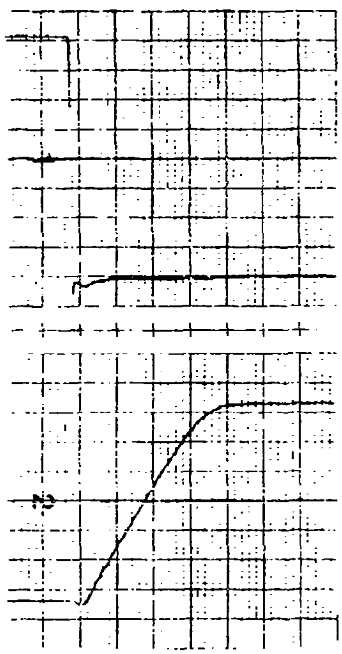
Figure 19. Step Response - Condition 13 & 14

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Step Response - Condition 15 and 16 Date Prepared 9/21/83

Extend

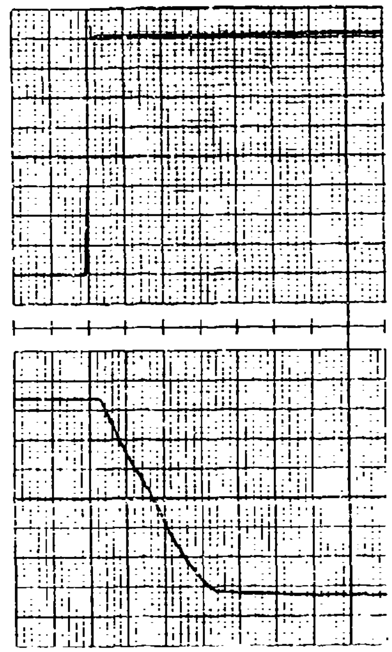
Retract



E_{in} E_{in}

Condition 15
10% F.S. Output

X_{out} X_{out}



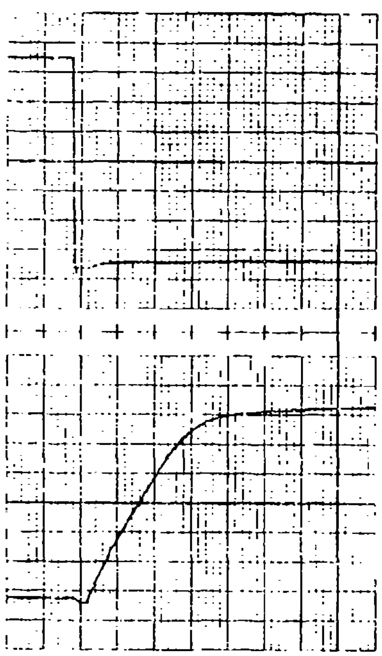
Scales:

E_{in} = 0.050 v/div

X_{out} = 0.00935 in/div

t = 100 div/sec

—— t ——> —— t ——>



E_{in} E_{in}

Condition 16
10% F.S. Output

X_{out} X_{out}

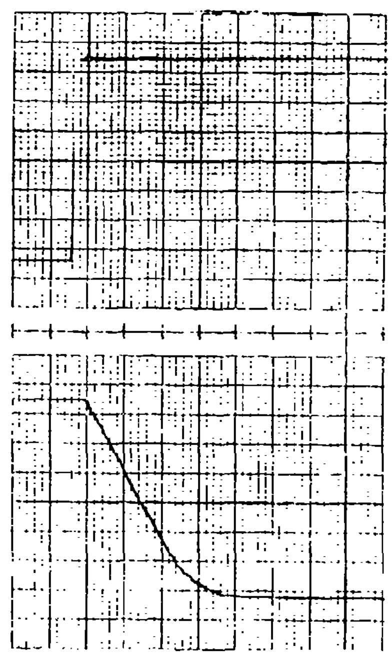


Figure 20. Step Response - Condition 15 & 16

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Step Response - Condition 17 and 18 Date Prepared 9/21/83

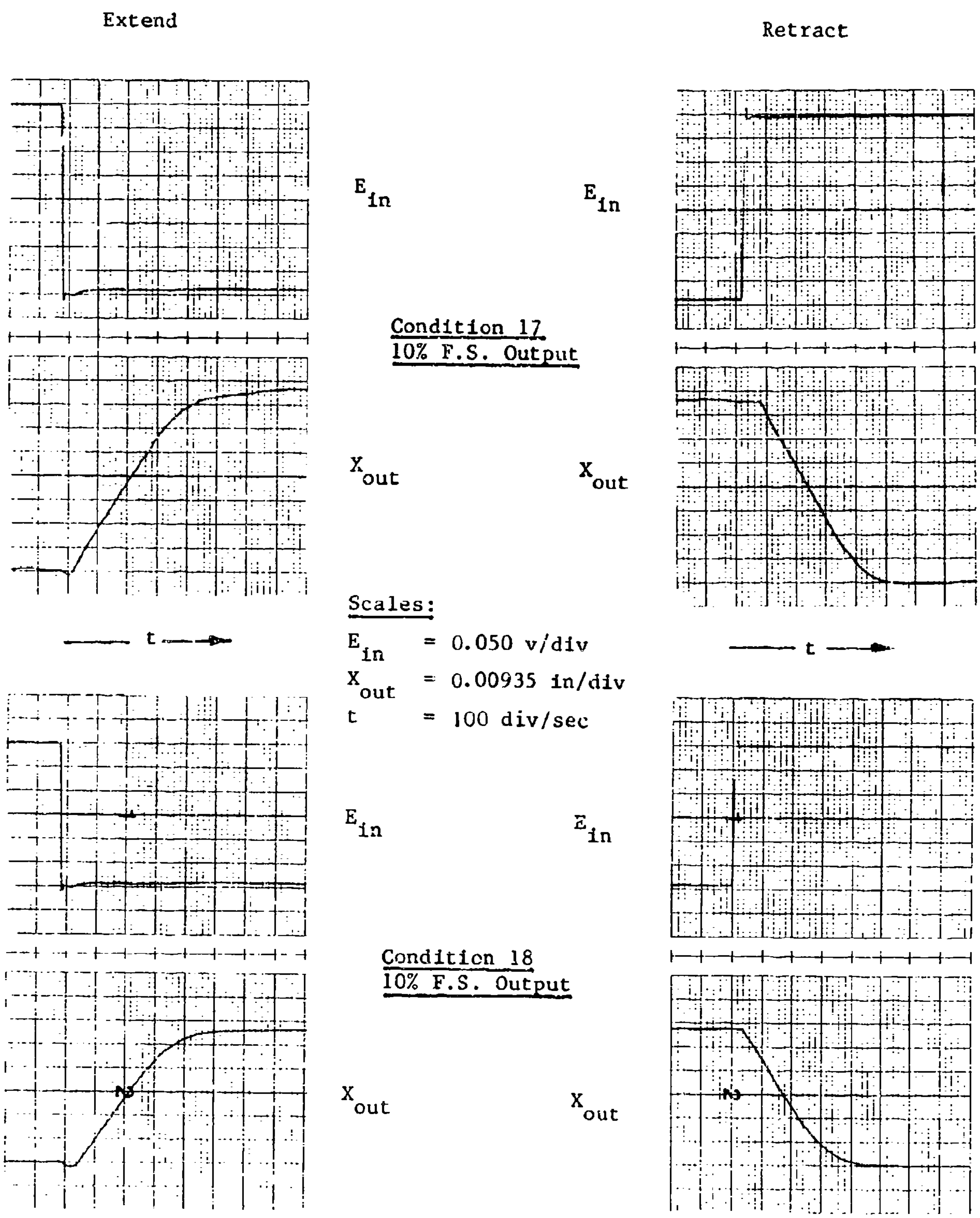


Figure 21. Step Response - Condition 17 & 18

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Step Response - Condition 19 and 20 Date Prepared 9/21/83

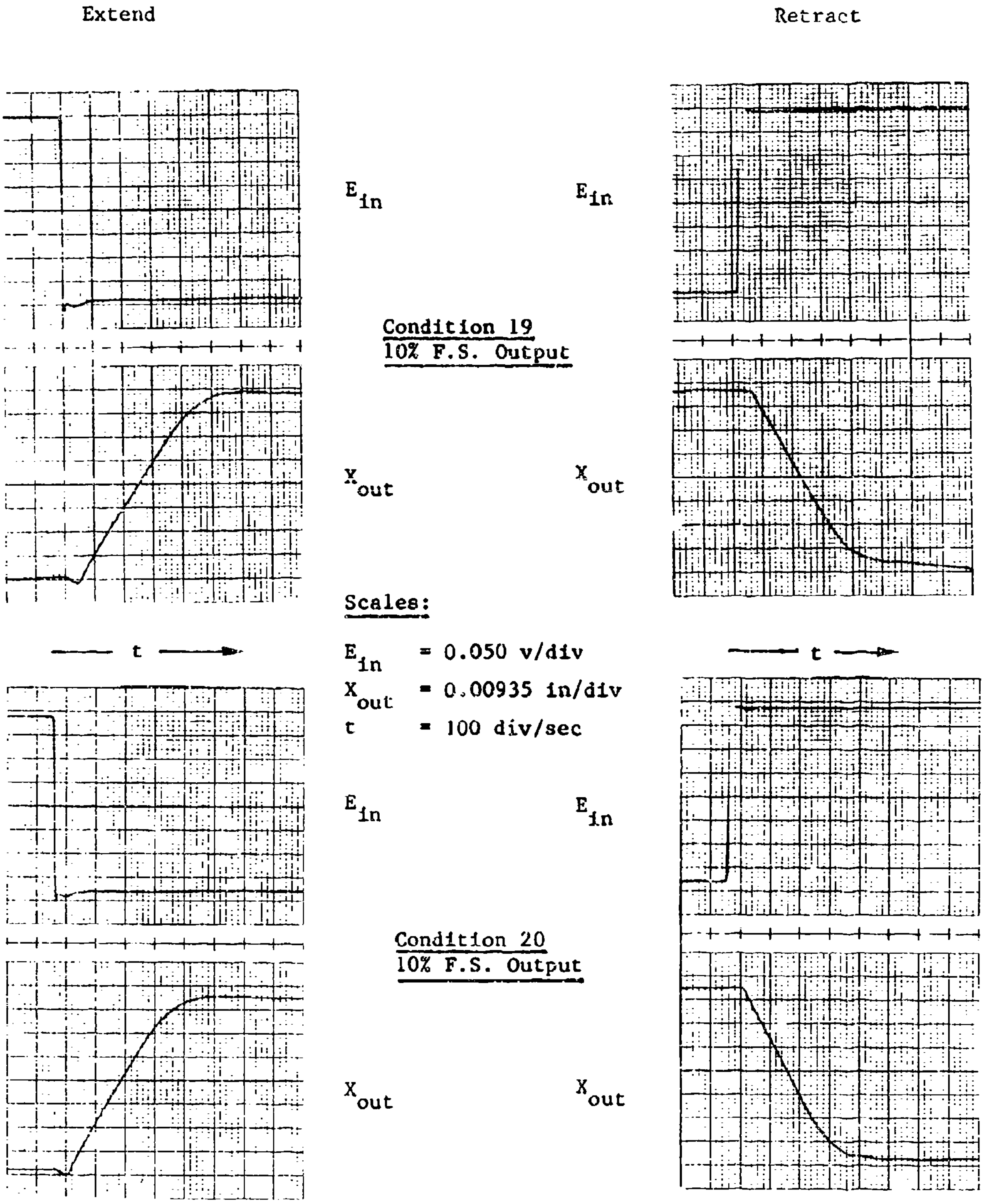


Figure 22. Step Response - Condition 19 & 20

is a direction opposite to that commanded by the step input. The 1 millisecond time delay also occurs with the retract motion. The 2.5 millisecond time period of "opposite motion" does not occur with the retract response. The "opposite motion" amplitude is small, being nominally 0.3% of the total actuator stroke.

The step response is similar for all the test conditions. For test condition 18 with the hydraulic pressure to the servovalves reduced to 2000 psi, the initial saturated rate of motion is 70% of the other test conditions with 3000 psi supply pressure. This agrees with the theoretical reduction of flow to 0.707 for a 1/3 reduction of pressure from normal.

Failure Transients

General

The failure transient data is presented in the strip chart form as recorded. For each figure, the general arrangement of the data is from the top of each figure down:

a. Channel Inputs 1 through 4 (E_{in})
b. Actuator Position (X_{out})
c. Failure Indicate for Actuator Section 1
d. Failure Indicate for Actuator Section 2

The channel inputs are used as failure injection points for the test system. The actuator position trace shows the effect of the injected failure on the system output. The failure indicate time traces show the state of the voltages used to drive the failure indicators for the two actuator sections. These voltages change when the failure logic causes the bypass solenoids to operate. The bypass solenoids drive bypass valves which disable an actuator section by bypassing the actuator drive area. Note that the failure indicate traces do not show individual control channel status. Display lights on the front panel of the microprocessor were used for that function.

Note that the test conditions 21 through 30 define both the initial operating status of the system and the input voltage changes used to cause the system to change operating status.

Specific

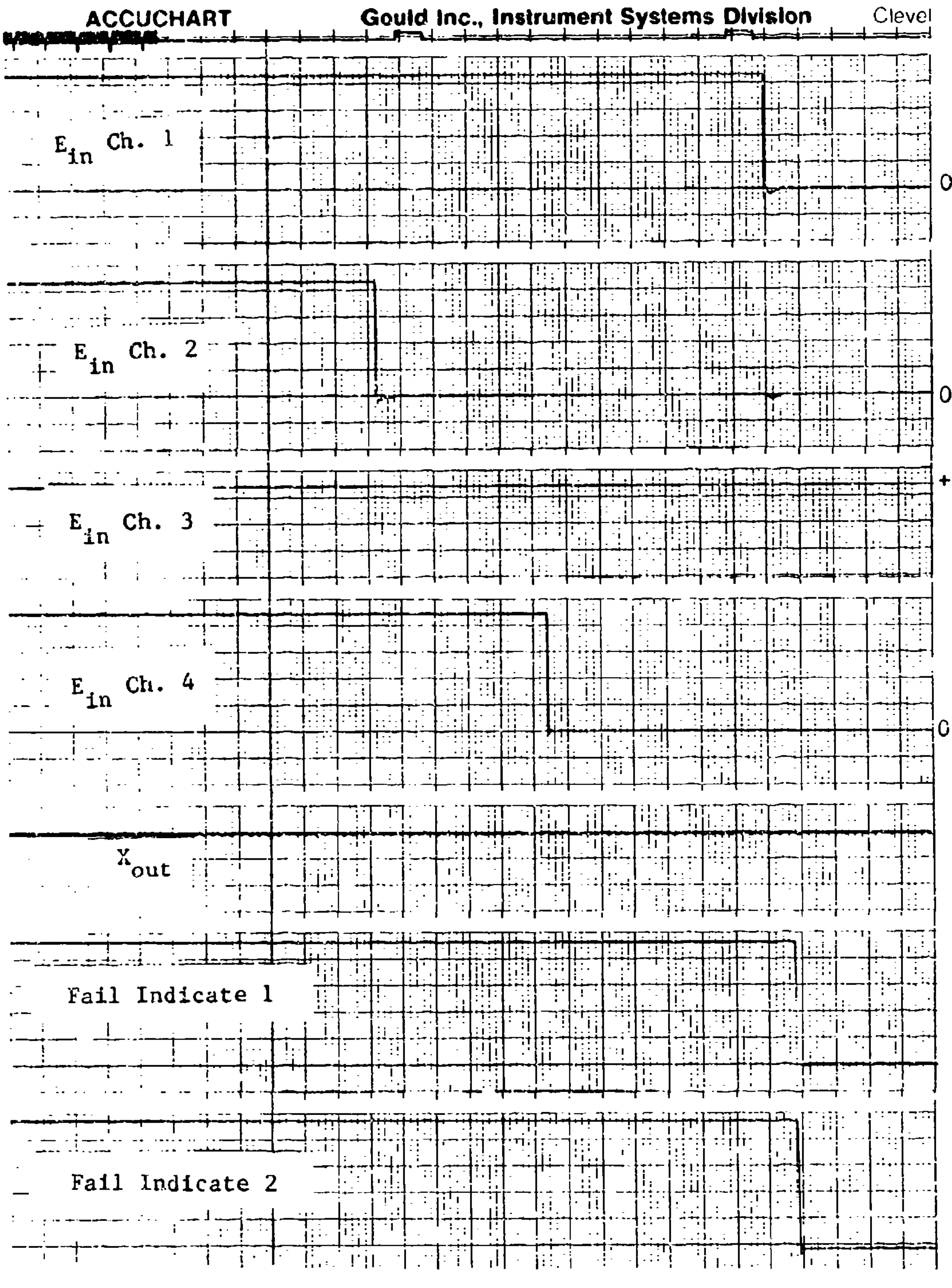
Figure 23 shows the results of sequentially grounding the input voltages to channels 2, 4 and 1. The system is operating initially with control channel 1 active, control channel 2 monitor, control channel 3 active and control channel 4 as monitor. The actuator was initially commanded to a 50% extend postion.

The significant result of this test is that there is no change in the actuator position with the three input failures. Since the first two injected failures are failures of model channels, no failure transient would be expected. The third failure (input 1) is a failure of an active channel. Since the majority vote failure logic has already detected two failures, the third failure causes the voting logic to bypass both halves of the actuator. Since no external load is applied, the actuator output remains stationary when the actuator sections are bypassed. Note that the failure logic does not bypass either actuator

TEST ITEM - Boeing Reconfigurable Fail Operative
Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 21 Date Prepared 9/21/83

← t →



Scales:

E_{in} = 0.200 v/div

X_{out} = 0.0375 in/div

t = 5 div/sec

Figure 23. Failure Transients - Condition 21

section until the third failure occurs. Before the third failure, both active channels 1 and 3 agree and are kept in control of their respective actuator sections. Since the model channels 2 and 4 have already been declared failed, only channels 1 and 3 are left for comparison. When channel 1 is then "failed", the voting logic has no information to use to determine whether 1 or 3 is the failed channel. The logic therefore declares a system failure and bypasses both halves of the actuator.

Figure 24 shows the results of sequentially applying a ramp input of 0.4 volts per second to channels 1, 2 and 3. The system is initially at null and configured with channel 1 active, channel 2 monitor, channel 3 active and channel 4 monitor. The input signal used to generate the 0.4 volt ramp is a triangle waveform input with a peak amplitude of nominally 1 volt and a frequency of 0.1 Hz. This test is designed to evaluate the effect of "slowover" failure inputs on the output of the actuator.

Note that because of space limitations, Figure 24 does not show the input to channel 4. However, because the input was maintained at 0 voltage during the test, no significant information is lost by not presenting channel 4's input recording.

Note that the Fail Indicate 1 signal shows that section 1 of the actuator is bypassed when the ramp into channel 2's input reaches 0.375 volt. Channel 1 was already voted out when the ramp applied to its input reached the failure detection voltage. Channel 2 was then changed from monitor to active status and the application of the slowover ramp to channel 2 caused the failure logic to vote the channel failure and the bypassing of section 1 of the actuator. Note that the voltage at which the failure was detected (0.375 volt) corresponds to 47% of the voltage maximum spool stroke. This is slightly greater than the 35% setting inputed for the failure detection level.

Note that upon the bypassing of section 1, the output of the actuator moves 0.075 inch or 2.2% of the actuator total stroke. This movement represents the force gain of the "good" actuator section, and the amount of force fight buildup when the second failure is detected. When detected, and upon the bypassing of section 1, the actuator moves to the null position of section 2. The third failure (the slowover into channel 3) causes the bypassing of section two. Note that the actuator moves 0.205 inch before the failure is detected. This is 6.1% of the total actuator stroke. The increase in actuator movement between the second and third failure detection is due to the lack of force fight, since Section 1 is bypoassed; and is simply the amount of the actuator movement before the failure is detected.

Figure 25 shows the effect of applying a ramp input of 0.4 volt per second sequentially to channels 1, 2 and 3 with the system operating with a sinusoidal input into all channels. The amplitude of the nominal 1.5 Hz sinusoidal signal is at the maximum input at that frequency without causing rate saturation. The system is initially operating with channel 1 active, channel 2 model, channel 3 active and channel 4 model. The input to channel 4 is not displayed on Figure 25 for reasons of room. The input was maintained at the same sinusoidal input as the other channels before the application of the ramp input. Note that the ramp input is created with the same triangular waveform input of 1 volt peak at 1 Hz that was used for evaluating the effect of slowover input failures. The purpose of this test condition was to measure the effect on the dynamic output of failure detection of slowover failures.

TEST ITEM - Boeing Reconfigurable Fail Operative
Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 22 Date Prepared 9/21/83

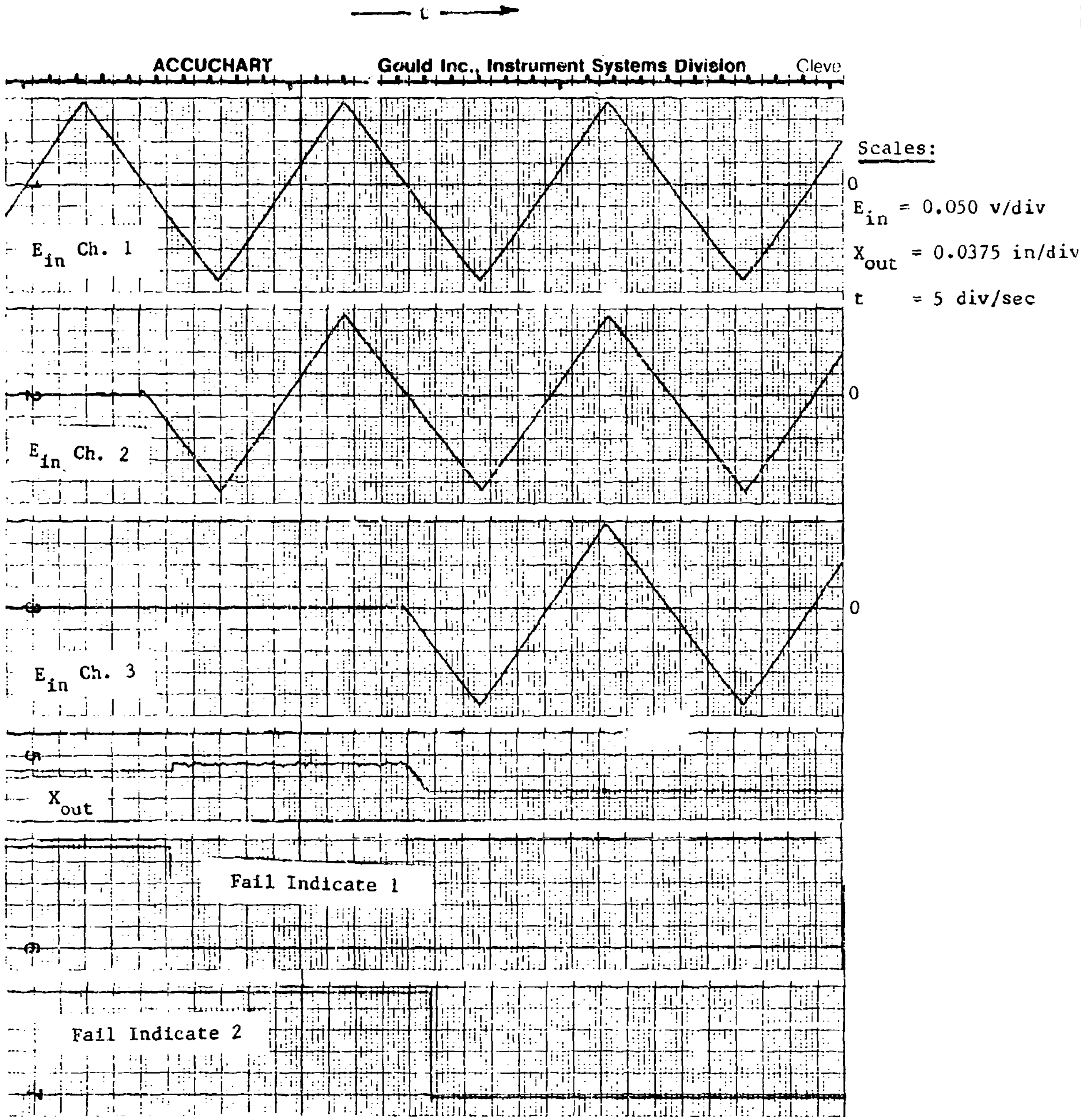


Figure 24. Failure Transients - Condition 22

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 23 Date Prepared 9/21/83

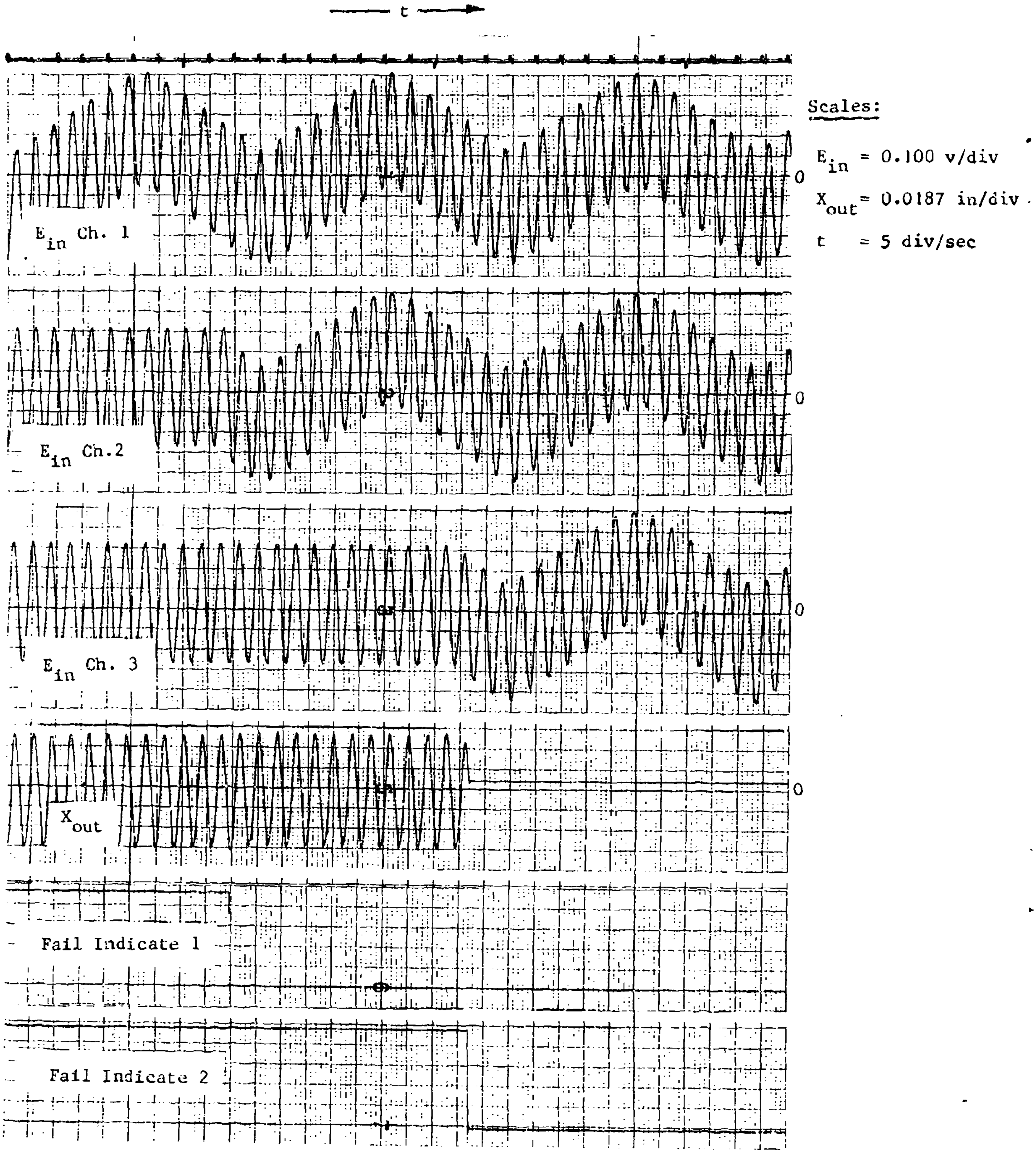


Figure 25. Failure Transients - Condition 23

As shown by the actuator output motion shown on Figure 25, there is no observable deviation of the output motion of the actuator until injection of the third failure. The dynamic response amplitude and waveform is not affected by the failure detection. There is a slight null shift of 1/2 division (0.009 in.) when section 1 is bypassed (as indicated by the Fail Indicate 1 level change).

Figure 26 shows the effect of grounding the inputs to channels 1, 2 and 3 sequentially with the system commanded to a 50% extend position. The system is initially configured with channel 1 active, channel 2 monitor, channel 3 active and channel 4 model. This test was designed to evaluate the effect of signal loss failures to the control channels while holding an "off null" actuator position. The sequence of failure injection is to inject failures into the active channels of each actuator section first (Condition 21 injected the grounding failures into the monitor channels first).

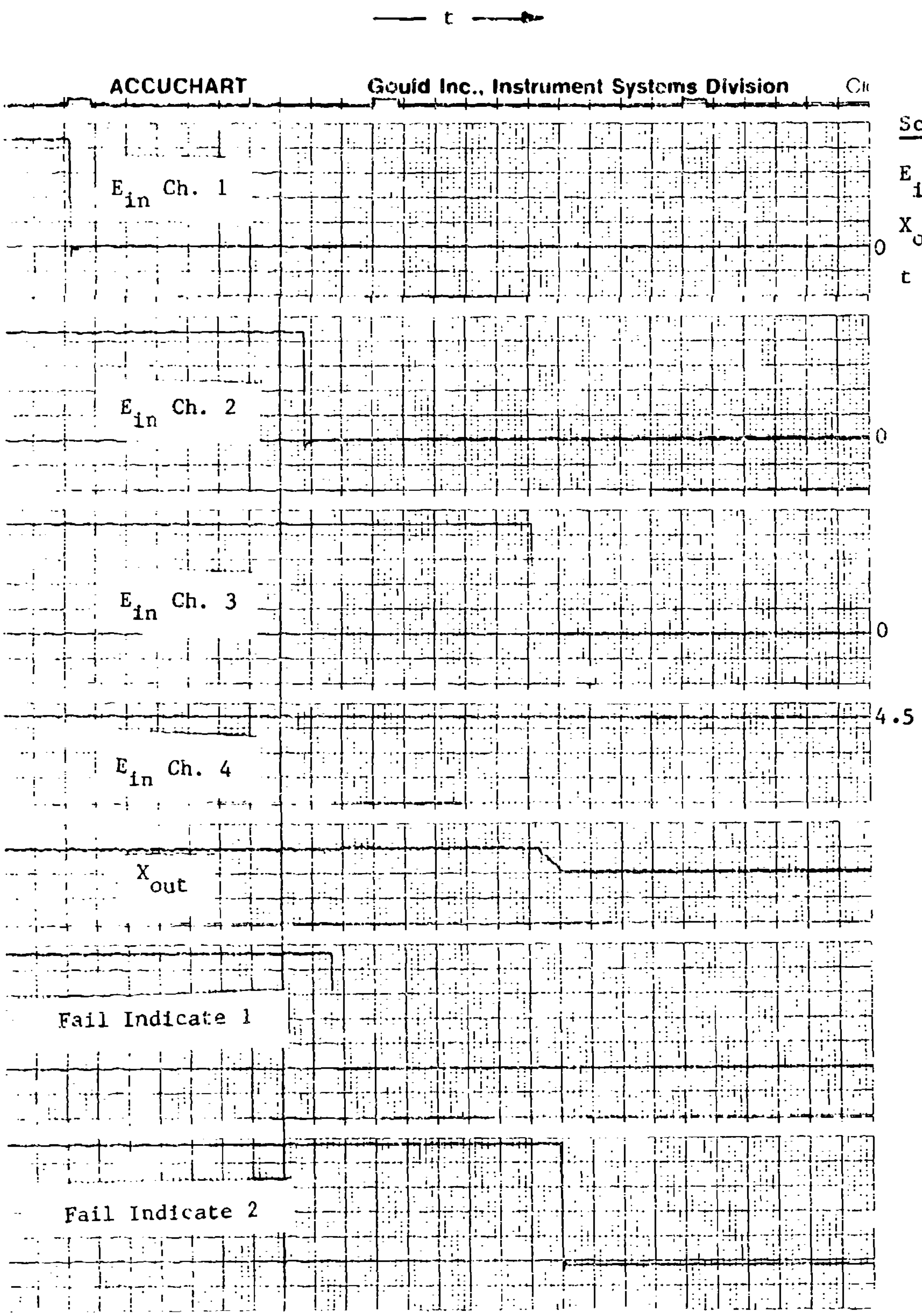
As shown on Figure 26, the failure logic detects the grounded inputs correctly and switches control from the failed channels to the model. Since the failure of channel 1 and then 2 constitutes an actuator section failure, the failure logic bypasses section one of the actuator. The actuator output shows no detectable change for the channel 1 failure and a small deviation of 0.55% of the actuator stroke upon the bypass of section 1. The bypassing of the actuator (as indicated by the actuator output change of 0.55%) occurs 0.8 second after application of the second failure input. This time length is interesting as can be observed for other failure transient tests, the time delay does not occur with hardover inputs with the actuator at null (test condition 27). With the third input failure (grounding of channel 3) the actuator does respond to the input failure. During the 0.8 second delay between the application of the failure input, the actuator moves 0.18 inch or 5.33% of the total actuator stroke. The reason for the time delay or why the delay does not occur for the similar test condition of hardover inputs applied in the same sequence to the channel inputs is not obvious. (Subsequent testing of the miroprocessor configuration by Boeing in 1986 generated similar results. The cause of the time delay was determined to be the ground return path design which could be easily modified to change the characteristic.)

Figure 27 shows the effect of grounding the inputs to channels 1, 2 and 3 sequentially with the system commanded to a 50% retract position. The system is configured initially with channel 1 active, channel 2 monitor, channel 3 active and channel 4 model. As with test condition 24, this test was designed to evaluate the effect of signal loss failures to the control channels while the actuator is at an "off null" position. Test condition 24 evaluated the extend initial condition and test condition 25 (the results of which appear on Figure 27) evaluates the retract initial position.

As shown on Figure 27, the failure logic detects the grounded inputs correctly and switches control from the failed channels to the model. Since the failure of channel 1 and then 2 is an actuator section failure, the failure logic bypasses section 1 of the actuator after the channel 2 failure. Note that the actuator output shows a 1.33% deviation after both the first and second failures. The deviation lasts about 0.8 second. This time delay is identical to the delay observed for the similar test condition 24 with the actuator positioned at a 50% extend position. On Figure 27, the 0.8 second is apparent in the time delay between the application of the second failure and the change in the level of the fail indicate 1 switch which shows the bypassing of

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 24 Date Prepared 9/21/83



Scales:

E_{in} = 0.200 v/div

X_{out} = 0.0375 in/div

t = 5 div/sec

Figure 26. Failure Transients - Condition 24

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 25 Date Prepared 9/21/83

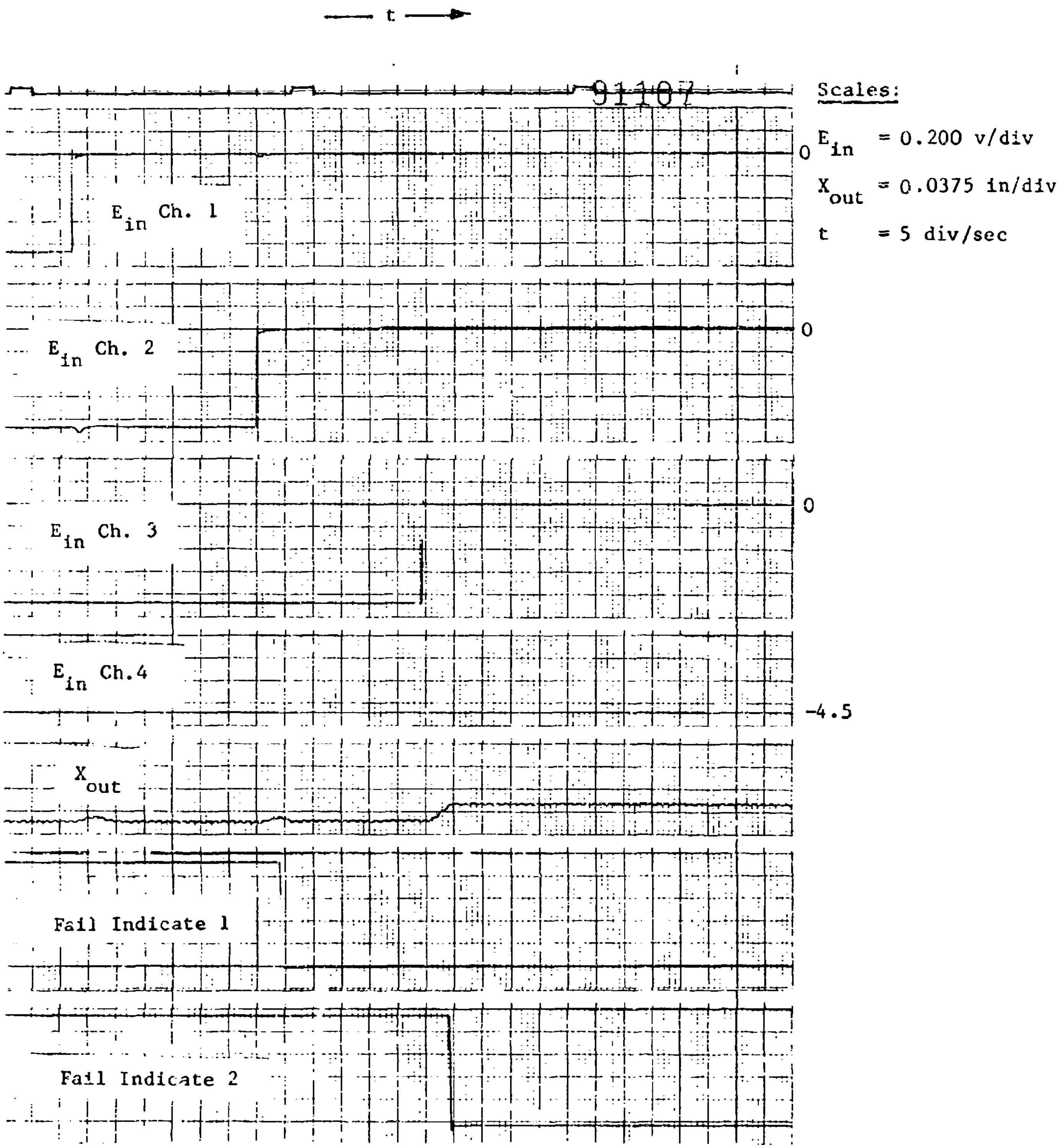


Figure 27. Failure Transients - Condition 25

actuator section 1. Upon the grounding of the input to channel 3, the actuator section 2 is also bypassed. As shown on the X_{out} trace, the actuator moves at a constant rate to a position 4.21 percent of the total actuator stroke from the initial retract position. This rate is approximately 0.18 inch/sec, considerably slower than the nominal 2.5 inches/sec maximum slew rate for the actuator. Although the actuator is moving in the correct direction in response to the grounding of channel 3's input, the rate of movement does not reflect a response to a "hardover" amplitude input (which is effectively what the grounding of channel 3's input with the actuator retracted is). There is not an apparent explanation for this minor anomoly.

Figure 28 shows the effect of grounding the inputs to channels 1, 2 and 3 sequentially with a sinusoidal input applied to all channels. The amplitude of the nominal 1.5 Hz sinusoidal input signal is at the maximum that can be applied to the system without causing rate saturation. The system is initially configured with channels 1 and 3 active, channels 2 and 4 as monitors.

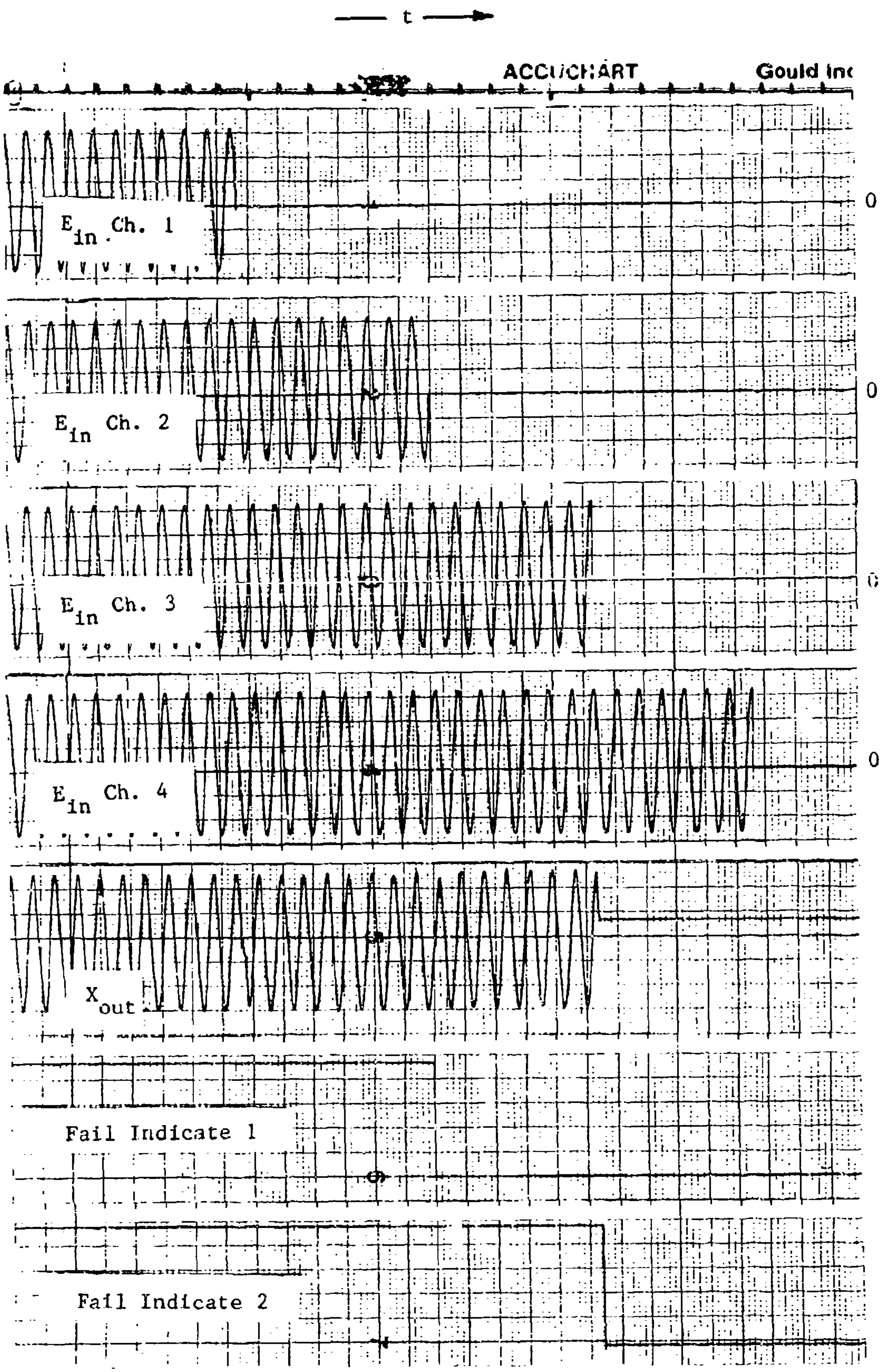
As illustrated by Figure 28, the failure logic detects the failures correctly and transfers control from and/or bypasses the failed channels correctly. For the first failure, the X_{out} recording shows that the transfer from channel 1 control to channel 2 takes 0.2 second. The transfer is a smooth amplitude deviation of 0.18 inch or about 0.5% of the maximum actuator stroke. The bypassing of actuator section 1 upon the second failure (channel 2's input) appears to take less time (0.1 second) with less output deviation than with the first input failure. Note that with both the first and second input failures, the peak amplitude of the first half cycle of output motion immediately after the failure transfer is 6.7% less than the peak amplitude of the motion before and after the failure detection. However, there is no observable long term change in the output response to the sinusoidal input after each of the first two failures. The third failure causes the system to correctly bypass actuator section 2.

Figure 29 shows the effect of applying +9 volts sequentially to channels 1, 2 and 3. The system is initially configured with channels 1 and 3 as active channels and channels 2 and 4 as monitor channels. The initial input of all channels is at zero volts. The +9 volts is a hardover extend direction input signal. This test condition is used to evaluate the hardover failure transients with the actuator operating statically around a null position.

As shown on Figure 29, the failure logic correctly detects the failed hardover inputs and transfers and/or bypasses channels. Upon the first failure input into channel 1, the actuator output responds to the failure input briefly and then returns to the null position. The output deviation amplitude for the first failure is 0.037 inch or 1.1% of the maximum actuator stroke. The duration of the transient is less than 0.2 second. The response of the system to the second input failure into channel 2 is similar to the response of the system to the first failure. The amplitude of the failure transient is 0.037 inch with a duration less than 0.2 second. The system then returns to a position offset from the initial null position by 0.014 inch. For both the first and second failures into an active channel, the failure transient is reduced by the actuator motion causing the "non-failed" active channel to fight the failed channel. The third input channel failure has no other active channel to fight the actuator output deviation and the failure transient is larger than that experienced with the first two input failures.

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 26 Date Prepared 9/21/83



Scales:

E_{in} = 0.100 v/div

X_{out} = 0.0187 in/div

t = 5 div/sec

Figure 28. Failure Transients - Condition 26

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 27 Date Prepared 9/21/83

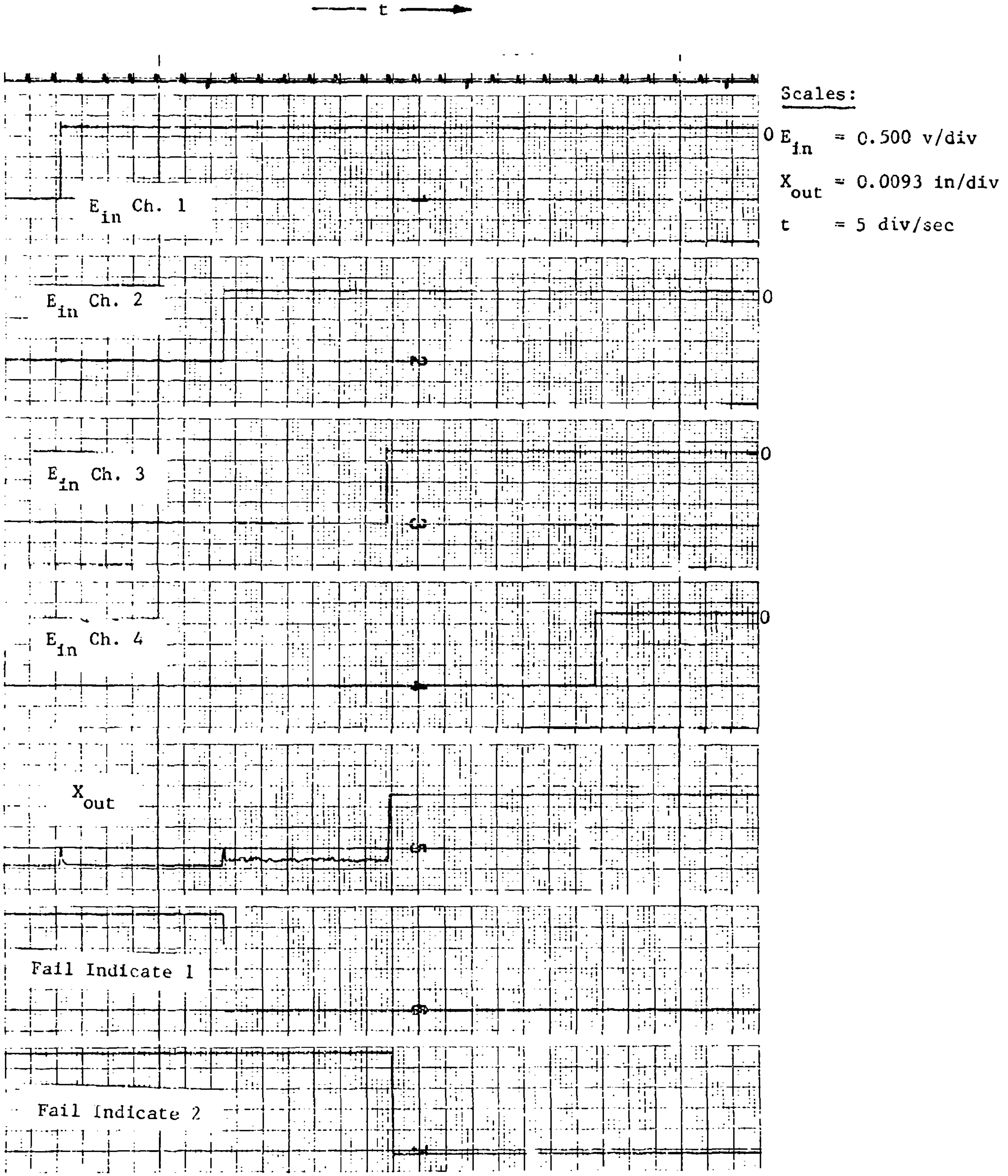


Figure 29. Failure Transients - Condition 27

The failure transient resulting from the hardover input into channel 3 results in a deviation of 0.149 inch (4.46% of the maximum actuator stroke) before the section 2 of the actuator is bypassed. The transient is simply the result of the actuator moving at maximum rate in response to the hardover input until the actuator section bypass occurs. Note that as shown on Figure 29, the actuator output exhibits some low amplitude hunting (less than 0.5% amplitude peak to peak) after section one of the actuation is bypassed. This is probably due to the particular threshold characteristics of channel 3 and the digital processing at the time of the hardover tests. The hunting does not occur on any other failure transient figures other than the hardover tests.

Figure 30 shows the effect of sequentially applying a negative hardover input signal of -9 volts to the inputs of channel 1, 2 and 3. The actuator is initially at null and configured with channel 1 and 3 active, channels 2 and 4 as monitors. This test condition is used to evaluate the hardover failure transients with the actuator operating statically around a null position. (Figure 29 showed the transients with positive hardover failures. Figure 30 shows the transients with negative hardover failures.)

As shown on Figure 30, there are only minor differences between the system response with negative (retract direction) hardovers as compared with the previous test results with positive hardover inputs. As shown by the X_{out} recording, there is no observable output transient with the first failure. The second input failure into channel two produces a null shift of the actuator output of 0.019 inch (0.5% of the maximum actuator stroke). As with the positive hardover inputs, the actuator output exhibits a low amplitude hunting after section 1 of the actuator is bypassed. Upon the injection of the third failure, the actuator moves 0.158 inch before actuator section 2 is bypassed.

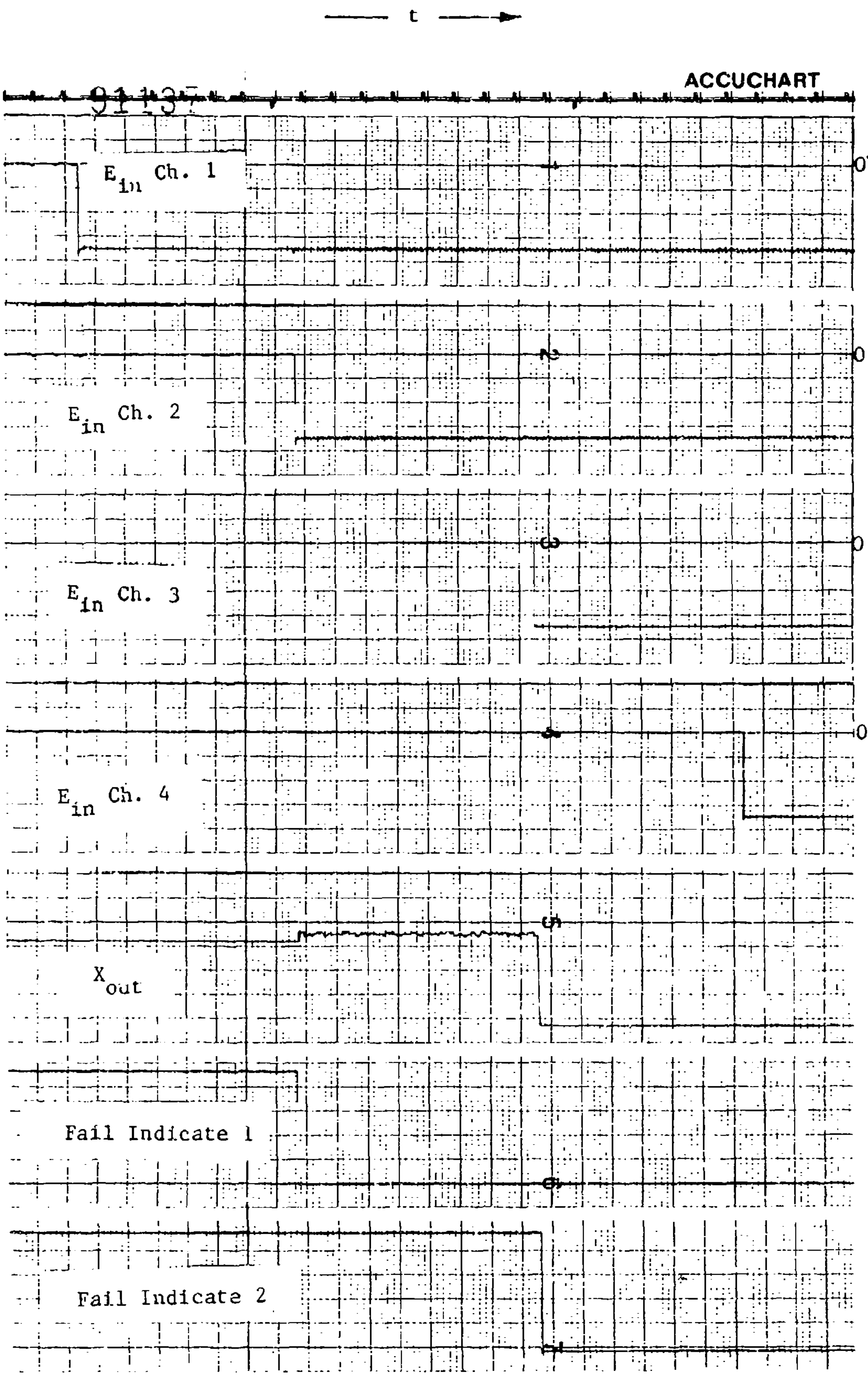
Figure 31 shows the effect of applying a hardover +9 volt step input sequentially into channels 1, 2 and 3 with the system operating with a sinusoidal input into all channels. The system is initially operating near null and the sinusoidal input is a nominal 1.5 Hz with an amplitude just below that which creates rate saturation. This test condition is used to evaluate the effect of extend direction hardover inputs on the system output with the system cycling. The system is initially configured with channels 1 and 3 active and channels 2 and 4 as model channels.

As shown on Figure 31, the failure logic correctly transfers control from the "failed" channels when the hardover inputs are applied. After the third failure, both halves of the actuator are bypassed. From the X_{out} time response, there is no failure transient that can be observed. Note that the actuator is cycling at 1.31 Hz at an amplitude of 0.469 inch peak to peak (or 13.9% of the maximum actuator stroke).

Figure 32 shows the effect of applying a hardover -9 volt step input sequentially into channels 1, 2 and 3 with the system operating with a sinusoidal input into all channels. The system is initially operating near null and the sinusoidal input is at a nominal 1.5 Hz with an amplitude just below that which creates rate saturation. This test condition is used to evaluate the effect of retract direction hardover inputs on the system output with the system cycling. The system is initially configured with channels 1 and 3 active and channels 2 and 4 as models.

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 28 Date Prepared 9/21/83



Scales:

E_{in} = 0.500 v/div

X_{out} = 0.0093 in/div

t = 5 div/sec

Figure 30. Failure Transients - Condition 28

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 29 Date Prepared 9/21/83

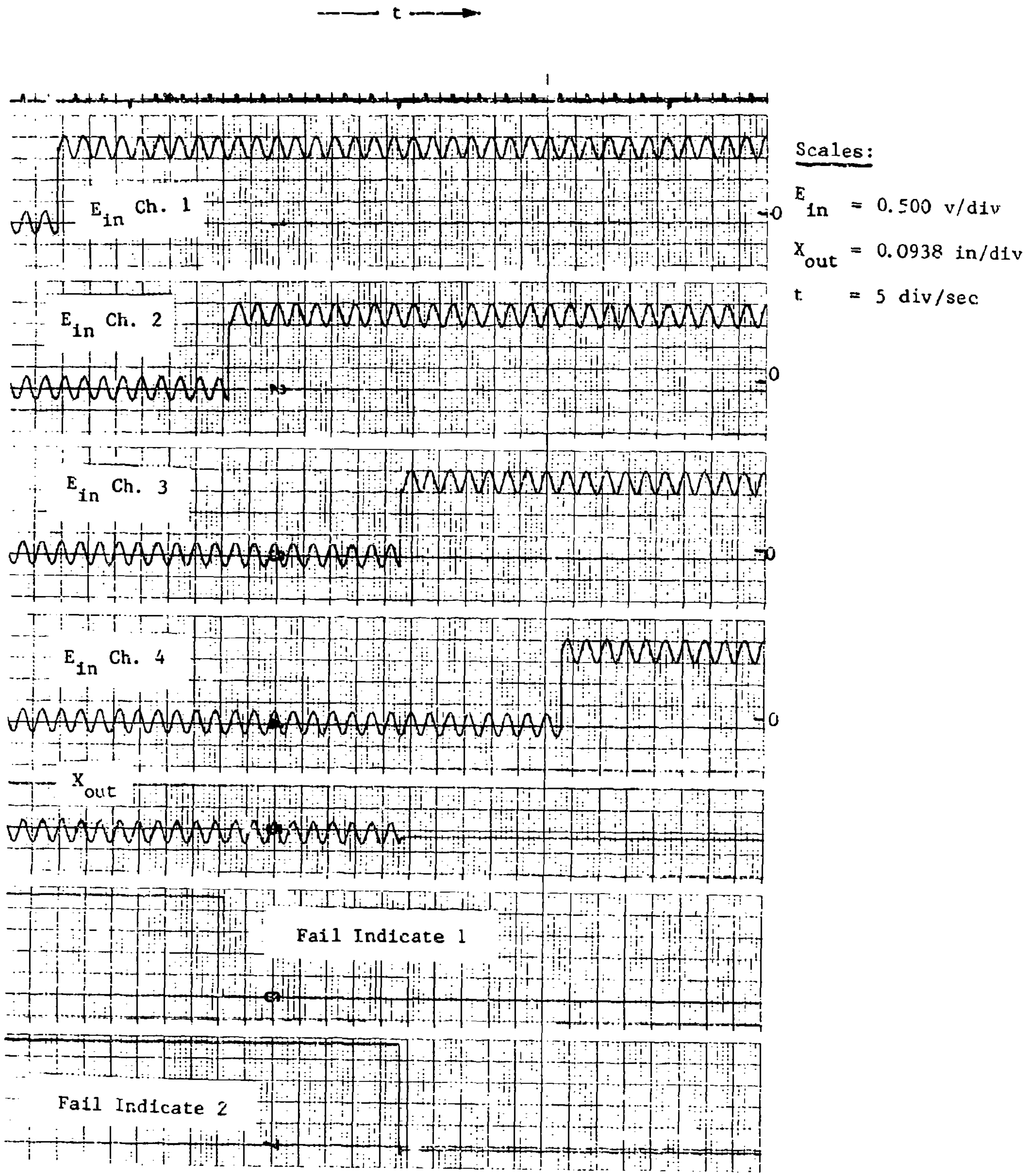
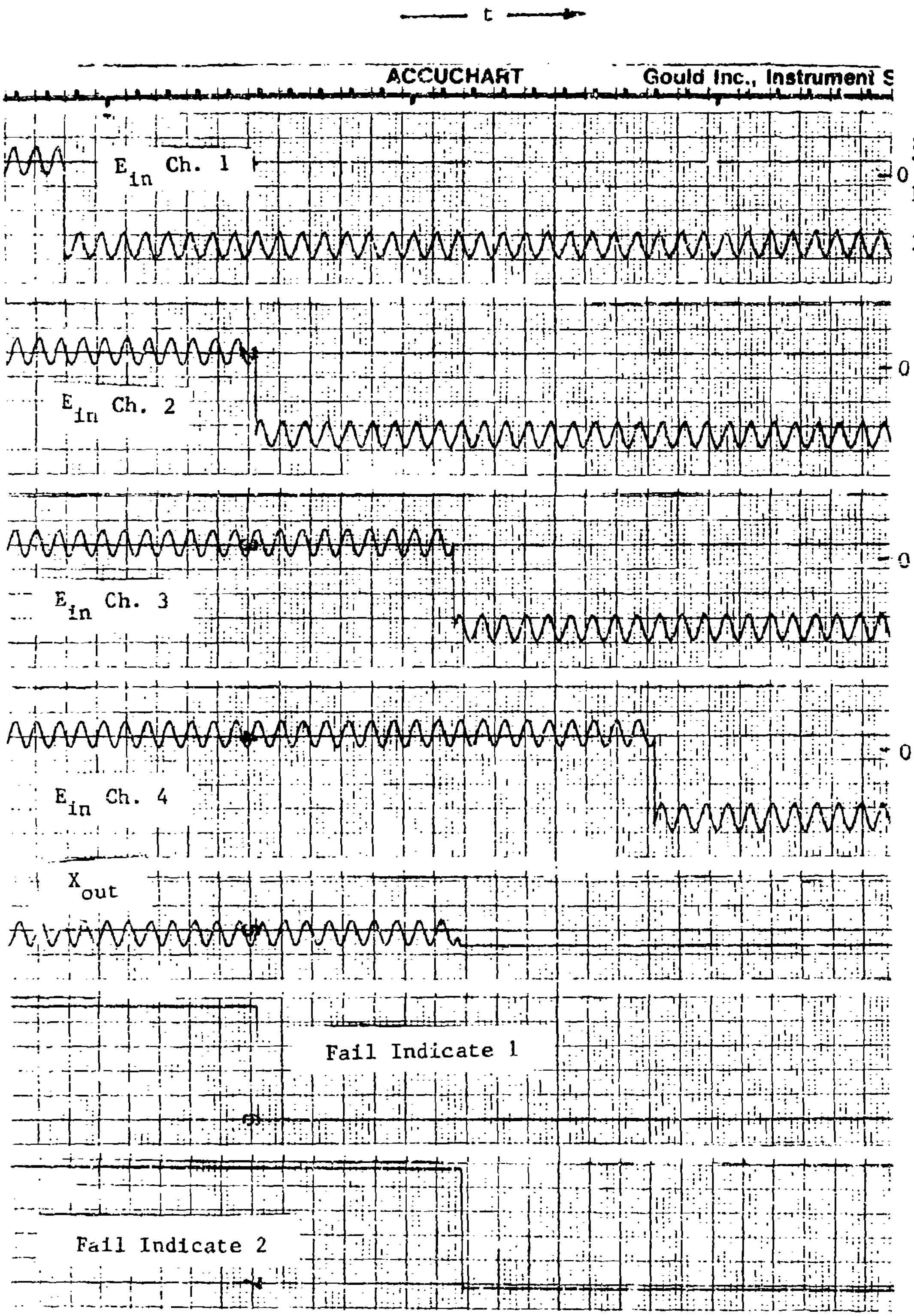


Figure 31. Failure Transients - Condition 29

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 30   Date Prepared 9/21/83



Scales:

E_{in} = 0.500 v/div

X_{out} = 0.0938 in/div

t = 5 div/sec

Figure 32. Failure Transients - Condition 30

As shown on Figure 32, the results of test condition 30 are similar to those of test condition 29 with the extend hardover failure. The failure logic again correctly identifies the failed inputs and switches control. There is no apparent change in the X_{out} trace until application of the retract hardover signal to channel 3.

SPECIFIC LOADED TEST RESULTS

Figure 33 shows the Boeing actuator mounted in the GPATR (General Purpose Actuator Test Rig) for the loaded tests. The load actuator is mounted at the left end of the GPATR frame. The center of the GPATR uses a load cell mounted in the center slide to measure the force applied to the test actuator. Figure 34 shows the attach mechanism used to mount the test actuator in the GPATR. The right end of the test actuator is mounted to a support slug. The support slug is prevented from sliding in its housing by a shear pin. The support slug and shear pin are designed to prevent damage to the test actuator. In the event that the load actuator malfunctions and applies a force to the test actuator which is greater than the proof force for which the test actuator was designed, the shear pin breaks and allows the tail stock of the test actuator to slide freely. The rod ends of the test actuator are retained by pins which are ground to be a light push fit into the rod end bearings.

In the test results presented in the following material, test conditions 1A through 4A were tests with the load system active and commanded to "0" load. These tests were used to verify that the load system static and dynamic performance characteristics would not degrade accuracy of the test actuator performance measurements. If the test system performance measurements with "0" commanded load were essentially unchanged from the measurements taken with the actuator mounted out of the load system, the load system fidelity is judged adequate.

For the loaded test condition numbers with suffix "B", the load system was commanded to provide a linear spring rate load of 10,000 lbs/inch around the actuator test actuator midstroke position. Note that the spring rate of 10,000 lbs/inch for load "B" provided a load nearly the stall load of 18,600 lbs for the test actuator.

For the loaded test condition numbers with suffix "C", the load system was commanded to provide a spring gradient load of 5,500 lbs/inch around the midstroke position. The actuator was positioned 0.85 inch from midstroke, creating a bias load of 4,675 pounds towards the midstroke position. Note that the 5,500 lbs/inch spring gradient selected for load "C" provided a load of 9,295 lbs to the test actuator at the maximum actuator stroke of $\pm$ 1.69 inches. This load is the stall load for the test actuator with one half of the tandem actuator operating.

<u>Static Threshold at "0" Load</u>

Figure 35 shows the data recorded in establishing the static threshold for condition 1A. Note that Figure 35 is similar to Figure 5 (the static threshold data representing the same measurement with the test actuator out of the GPATR). Table 7 lists the static threshold measured for conditions 1A, 2A, 3A and 4A. The thresholds measured are identical to the results of the same test

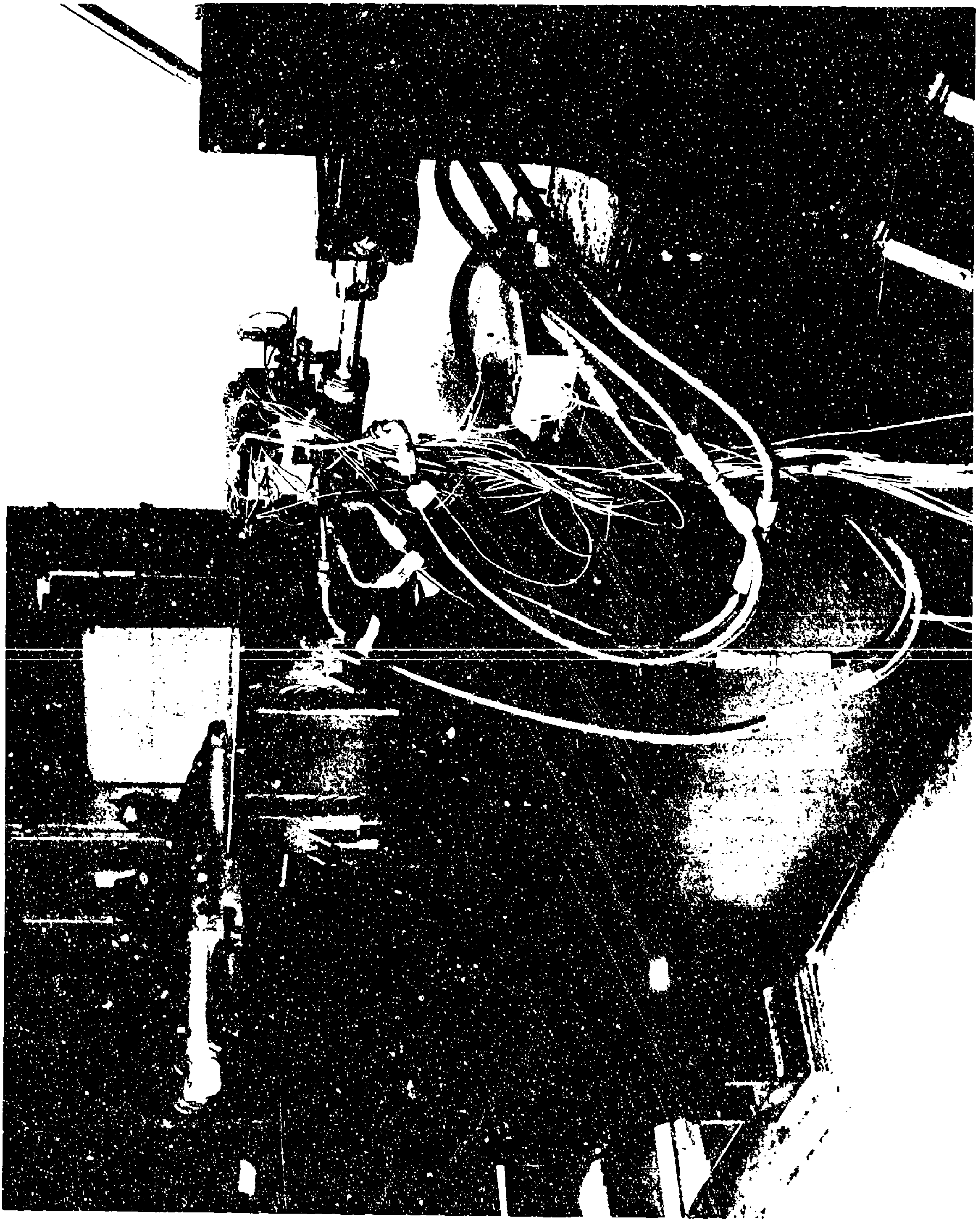


Figure 33. Test Actuator in GPATR

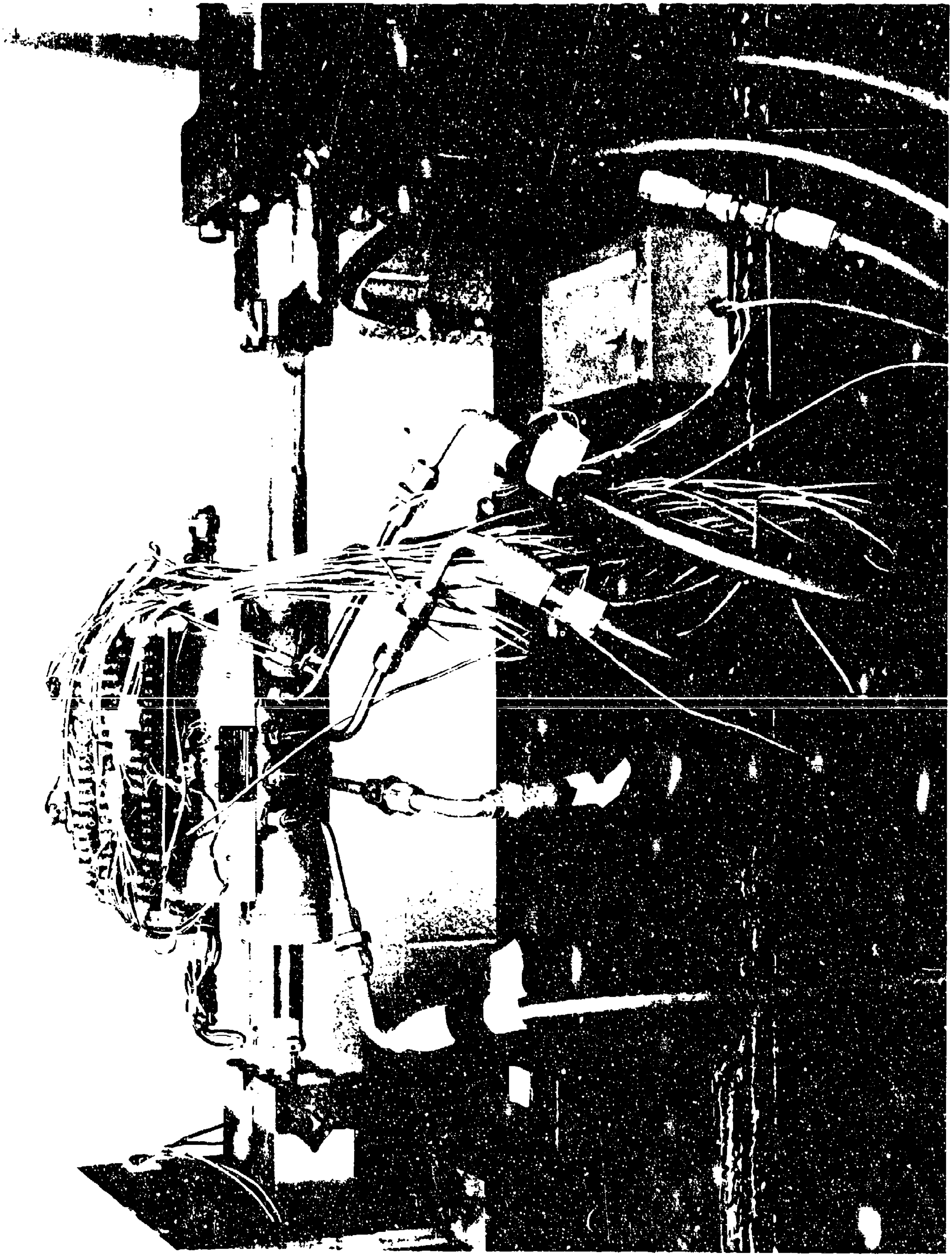
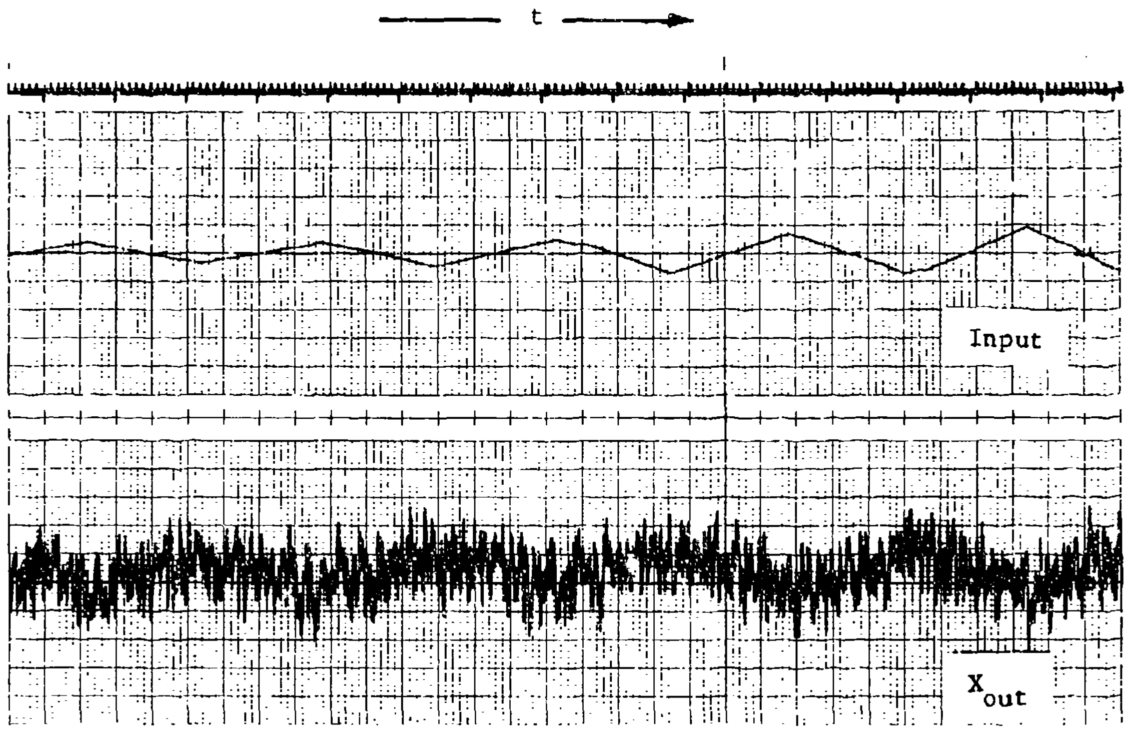


Figure 34. Test Actuator Mounting

Date Prepared 9/19/83

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Static Threshold - Condition 1A



Scale: Input = 0.002 v/div

X_{out} = 0.000374 in/div

t = 1.0 div/sec

Figure 35. Static Threshold - Condition 1A

TABLE 7
STATIC THRESHOLD - "0" LOAD

TEST ITEM: Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator **DATE PREPARED:** 9/23/83

TEST: Static Threshold in GPATR with "0" Load

Test Condition	Pk to Pk Input Volts	Static Threshold % of Max Input	% of E_v Max
1A	0.008	0.044	0.500
2A	0.008	0.044	0.500
3A	0.008	0.044	0.500
4A	0.008	0.044	0.500

with the test actuator out of the GPATR (as listed previously in Table 2). This verifies that the loading system operation does not affect the static threshold performance of the test system.

Figure 36 shows the data recorded in establishing the dynamic threshold for condition 1A. The data shown on Figure 36 is similar to the data recorded for condition 1 with the test actuator operating in air. Table 8 lists the dynamic threshold measured for conditions 1A, 2A, 3A and 4A. The values are identical with the test conditions 1 through 4 shown on Table 3, indicating that the load system does not affect the dynamic threshold performance of the the test system.

Figure 37 is representative of the frequency response test data recorded for test conditions 1A through 4A. The response is similar to the data recorded for test condition 1 through 4. As shown on Table 9, the frequency at which the amplitude is attenuated by 3 dB is nominally 3.55 Hz. This frequency is nominally 10% higher than that recorded for similar test conditions 1 through 4 (reference Table 4). The frequency at which the phase lag of 90° occurs for test conditions 1A through 4A is 3.65 Hz. This is nominally 10% lower than that measured for test conditions 1 through 4 (reference Table 4). The load system commanded to "0" load does affect the frequency response performance of the test actuator slightly. The effect is a slight improvement in the amplitude response and a slight decrease in the phase response characteristics, neither of which is judged significant enough to invalidate the frequency response measurements of the test system under loaded conditions.

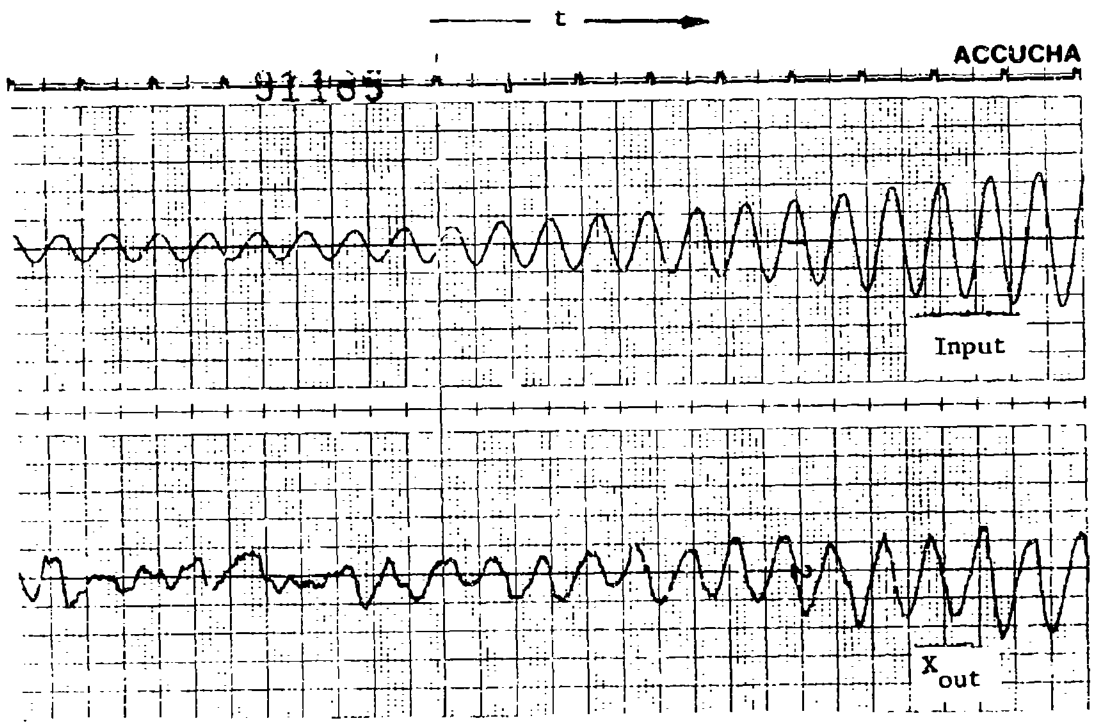
Figure 38 is representative of the hysteresis test data recorded for test conditions 1A through 4A. Note that the data recorded on Figure 38 is very similar to the data presented on Figure 8 for test condition 1. There is a non-coincidence of the output to input plot recorded for the two directions at specific input voltage levels. As shown on Table 10, the levels of hysteresis reflect the average separation at the different input levels. These values agree with those measured on the test actuator for test conditions 1 through 4 (reference Table 5). Connecting the load system with a "0" commanded load did not create a measureable change in the hysteresis performance of the test actuator.

Figure 39 is representative of the saturation velocity data recorded for test conditions 1A through 4A. The data shown on Figure 39 is similar to that shown on Figure 11 for test condition 1. Table 11 lists saturation velocities for test conditions 1A through 4A. The velocity is nominally 20% less than that listed in Table 6 for test conditions 1 through 4. The reduction in saturation velocity with a "0" commanded load reflects the affect of the load pressure applied to the test actuator by the load system tracking the test actuator. (Some minimum load is inherent with the load system in order to generate the error signal which opens the load system's control valve when tracking the maximum velocity of the test actuator.) This percent reduction in saturated rate reduction will not affect the loaded test results for the test actuator. Most of the test conditions applied to the test system maintain the actuator rate well below saturation where little error signal is required for the load system to track the velocity level of the test actuator. The principal effect would be on hardover failure transients where the reduction in saturated velocity would potentially reduce the amplitude of the failure transients.

Date Prepared 9/19/83

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Dynamic Threshold - Condition 1A



Scale: Input = 0.005 v/div

X_{out} = 0.000935 in/div

t = 10 div/sec

Figure 36. Dynamic Threshold - Condition 1A

TABLE 8
DYNAMIC THRESHOLD - "0" LOAD

TEST ITEM: Boeing Reconfigurable Fail Operative Fly-By-Wire Servoaccuator

DATE PREPARED: 9/23/83

TEST: Dynamic Threshold in GPATR with "0" Load

Test Condition	Pk to Pk Input Volts	Dynamic Threshold	
		% of Max Input	% of E_V Max
1A	0.050	0.275	3.125
2A	0.058	0.315	3.590
3A	0.060	0.330	3.750
4A	0.055	0.305	3.435

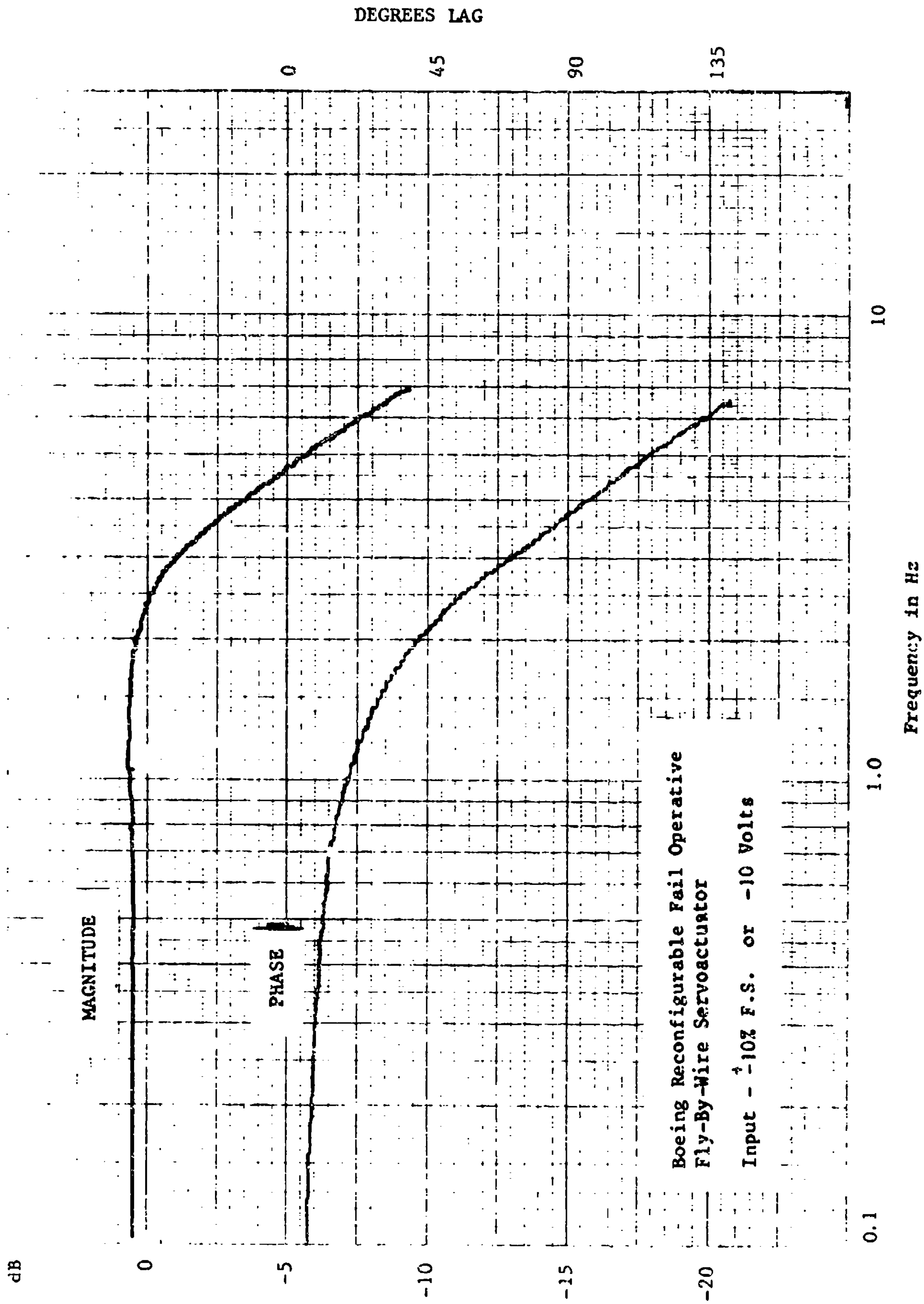


Figure 37. Frequency Response - Condition 1A

TABLE 9
FREQUENCY RESPONSE - "0" LOAD

TEST ITEM: Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator **DATE PREPARED:** 9/23/83

TEST: Frequency Response in GPATR with "0" Load

Test Condition	Output 10% Full Scale	
	-3 dB Hz	-90° Hz
1A	3.60	3.70
2A	3.50	3.60
3A	3.60	3.70
4A	3.50	3.60

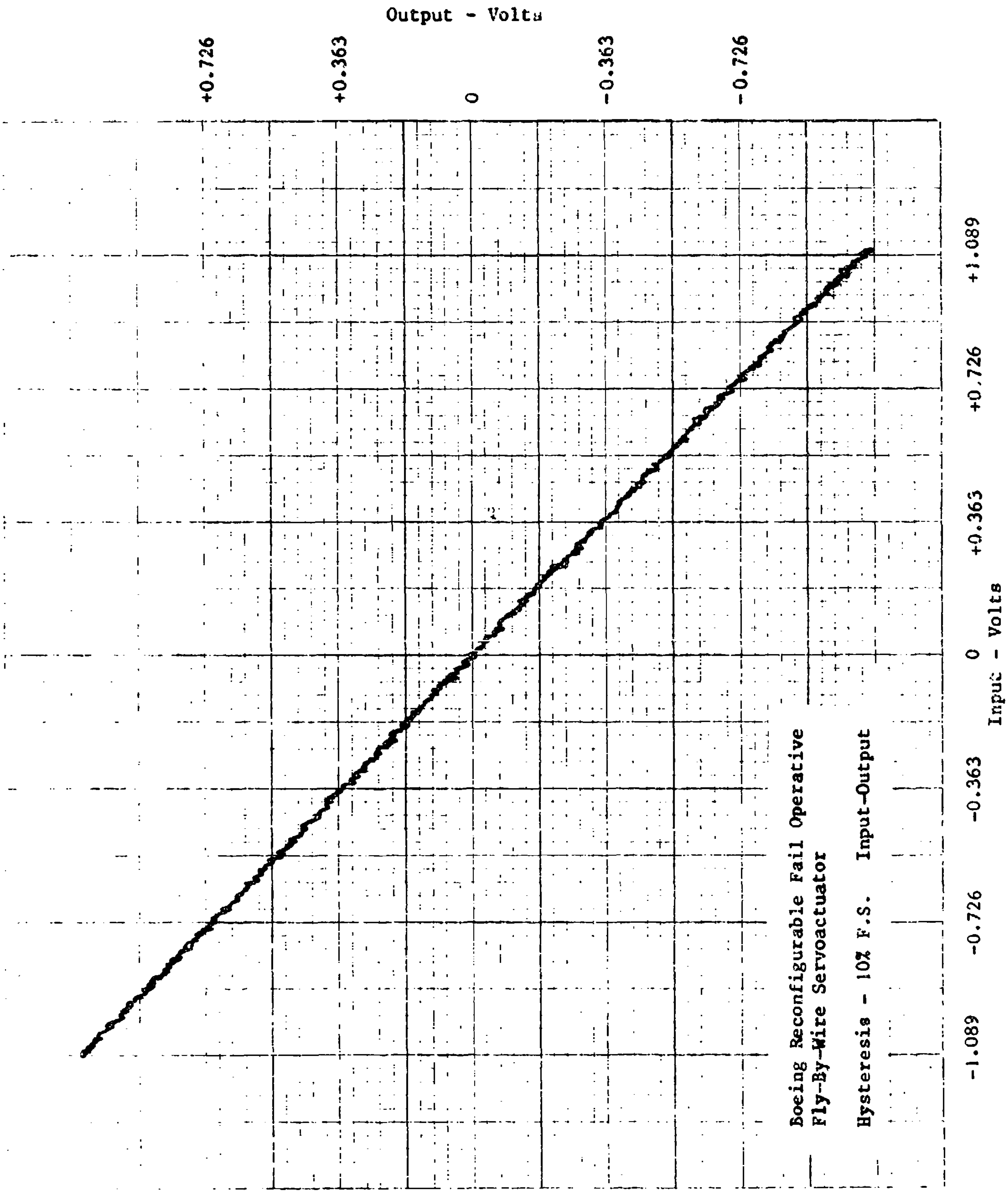


Figure 38. Hysteresis - Condition 1A

TABLE 10
HYSTERESIS - "0" LOAD

TEST ITEM: Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator **DATE PREPARED:** 9/23/83

TEST: Hysteresis in GPATR with "0" Load

Test Condition	% Full Scale	% of E_v Max
1A	0.062	0.69
2A	0.062	0.69
3A	0.062	0.69
4A	0.062	0.69

Date Prepared 9/29/83

TEST ITEM - Boeing Reconfigurable Fail Operative
Fly-By-Wire Servoactuator

TEST - Saturation Velocity - Condition 1A

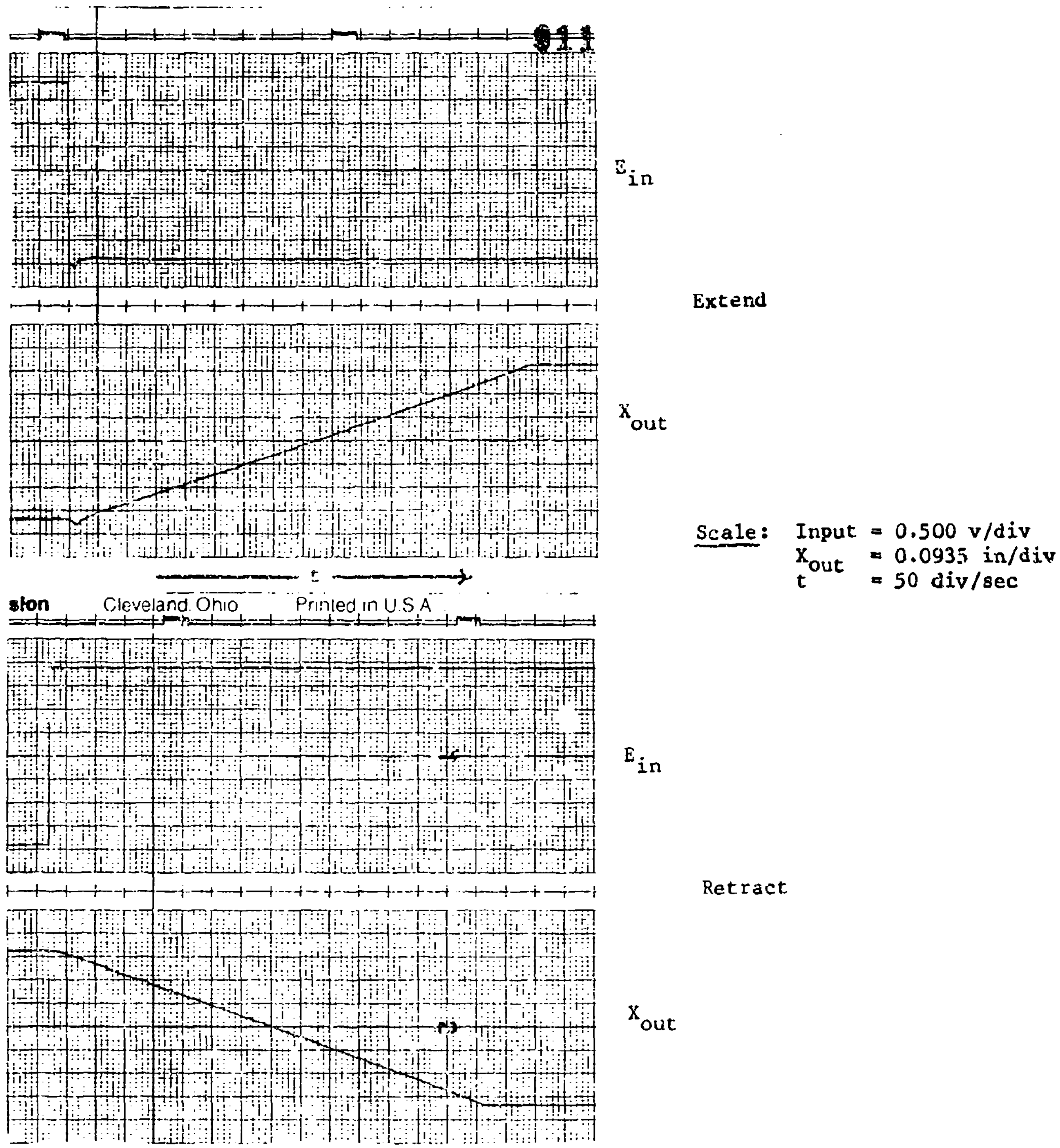


Figure 39. Saturation Velocity - Condition 1A

TABLE 11
SATURATION VELOCITY - "0" LOAD

TEST ITEM: Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator **DATE PREPARED:** 9/23/83

TEST: Saturation Velocity in GPATR with "0" Load

Test Condition	Extend - in/sec	Retract - in/sec
1A	2.16	2.01
2A	2.16	2.01
3A	2.23	2.01
4A	2.10	2.01

Figure 40 is a plot of the output to input linearity for test condition 1A. This is identical to Figure 12 for test condition 1, indicating the load system commanded to "0" load does not affect the linearity performance of the test system.

Figure 41 shows the reponse of the test actuator to a 10% of full scale input for test conditions 1A and 2A. The step response is very similar to that shown on Figure 13 for test conditions 1 and 2. The only measureable difference is a slight change in the slope of the straight line motion of the test actuator as it initially moves at maximum rate in response to the step input command. The slope is about 10% less for test conditions 1A as compared to condition 1. This change is consistent with the saturated rate change measured previously. Figure 42 showing the step response for conditions 3A and 4A is similar to Figure 14. Note that the initial response characteristic to the step input for the extend motion as shown on both Figure 42 and 14 is a small movement in the retract direction. This characteristic is unchanged by the GPATR "0" load operation.

Static Threshold at Loads B and C

Figure 43 shows the data recorded in establishing the static threshold for condition 1B. Figure 44 shows the data recorded in establishing the static threshold for condition 1C. The two figures are similar and resemble Figure 5 for the unloaded condition 1. (Note that the X_{out} scale on Figure 44 is 2.5 times the same scale as used on the other two figures.) The noise content of the output signal for both load conditions B and C is 0.004 inch peak to peak. This is the same noise amplitude as measured for the unloaded conditions. Table 12 lists the static threshold measured for the B load test conditions. Table 13 lists the static threshold measured for the C load test conditions. Note that on Table 12, the threshold measured for conditions 1B through 4B are identical. This is also true for conditions 1C through 4C listed on Table 13. This characteristic could be expected since the test conditions are just a reassignment of the model and active roles between the channels.

Test conditions 9B and 11B are with half the test actuator operational and the other half bypassed. In both cases there is an increase in the measured static threshold of about 50%. The threshold increase is similar to that measured on the unloaded actuator for the same test conditions. Note that the load condition B provides zero load at null. For small displacements around the null condition B generates only small loads compared to the force output of the test actuator. Therefore many of the measurements for load B will be similar to those for the unloaded test actuator. The increase in threshold with one half of the actuator operating is probably due to the reduction in actuator force gain compared to the actuator seal friction as compared to having both halves of the actuator operating. Test conditions 14B, 15B and 16B are a reflection of different bias conditions for the two sections of the actuator. The changes in static threshold for these test conditions as shown on Table 12 simply show the dependency of the threshold performance on channel matching. The bias changes vary the threshold from 66% to 166% of the threshold value with no bias applied.

Test conditions 1C through 4C show an increase of threshold nominally two times the unloaded static threshold. This is consistent with the test actuator holding a load. The load across the actuator drive pistons increases the

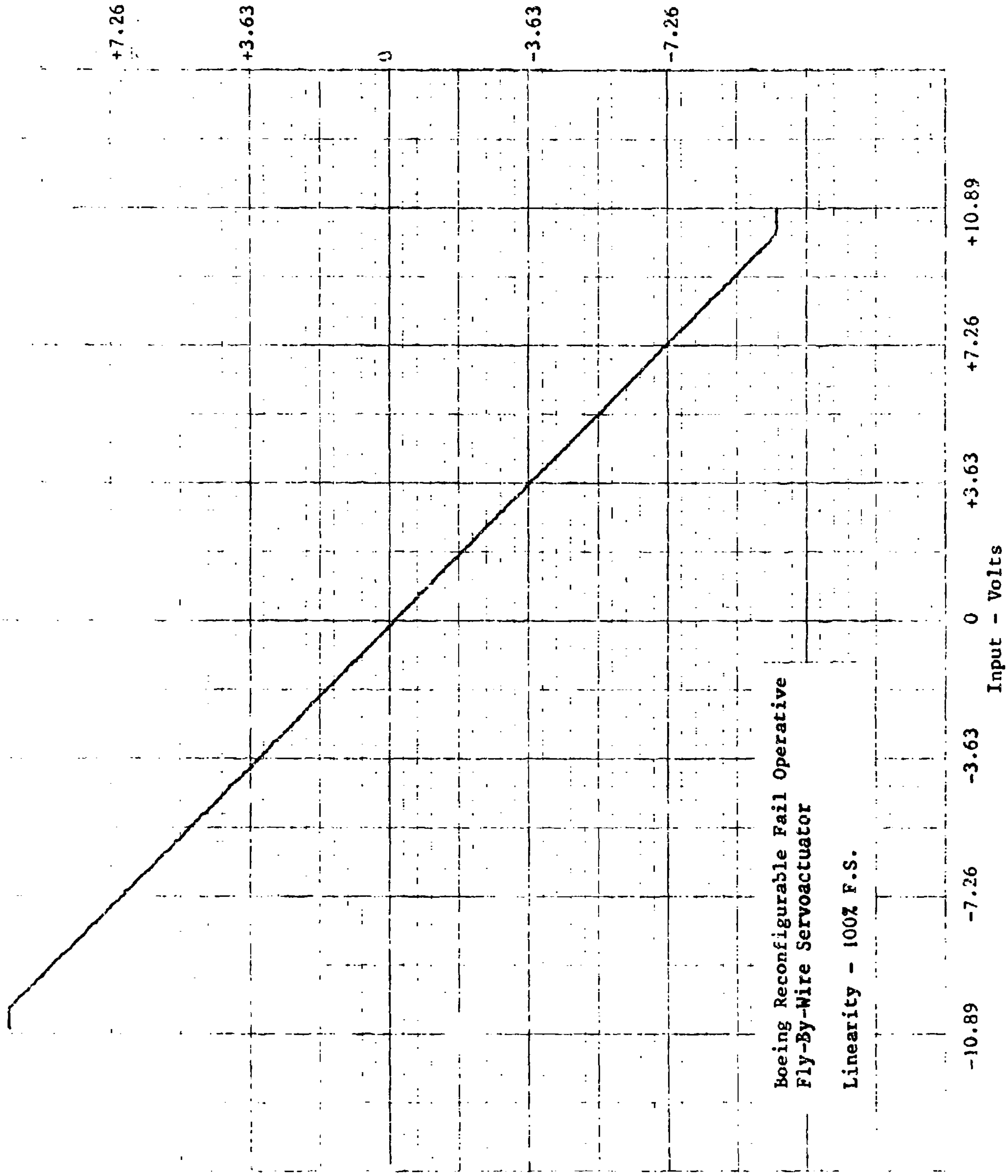


Figure 40. Linearity - Condition 1A

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

Date Prepared 9/28/83

TEST - Step Response - Conditions 1A and 2A

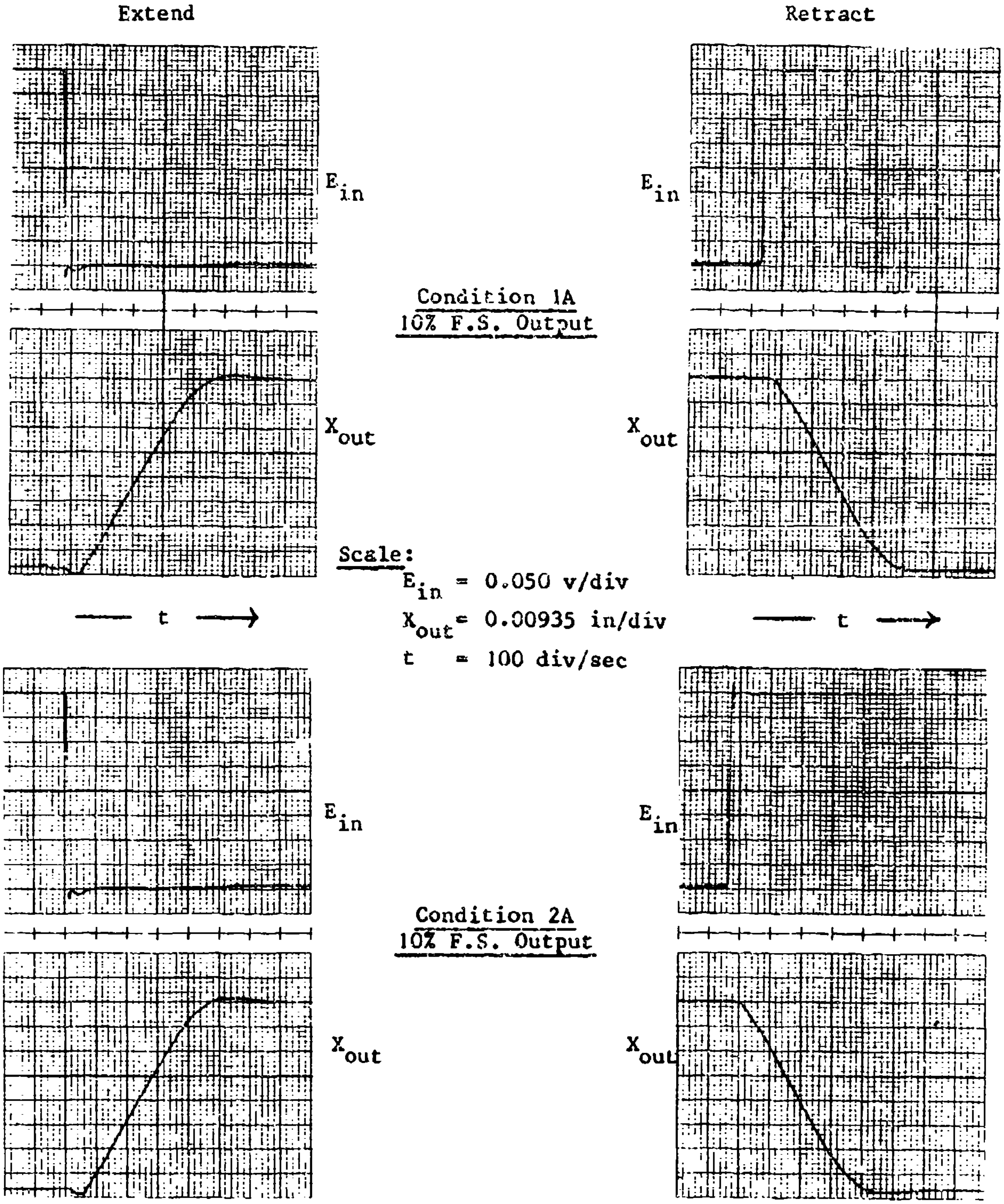


Figure 41. Step Response - Conditions 1A & 2A

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

Date Prepared: 9/28/83

TEST - Step Response - Conditions 3A and 4A

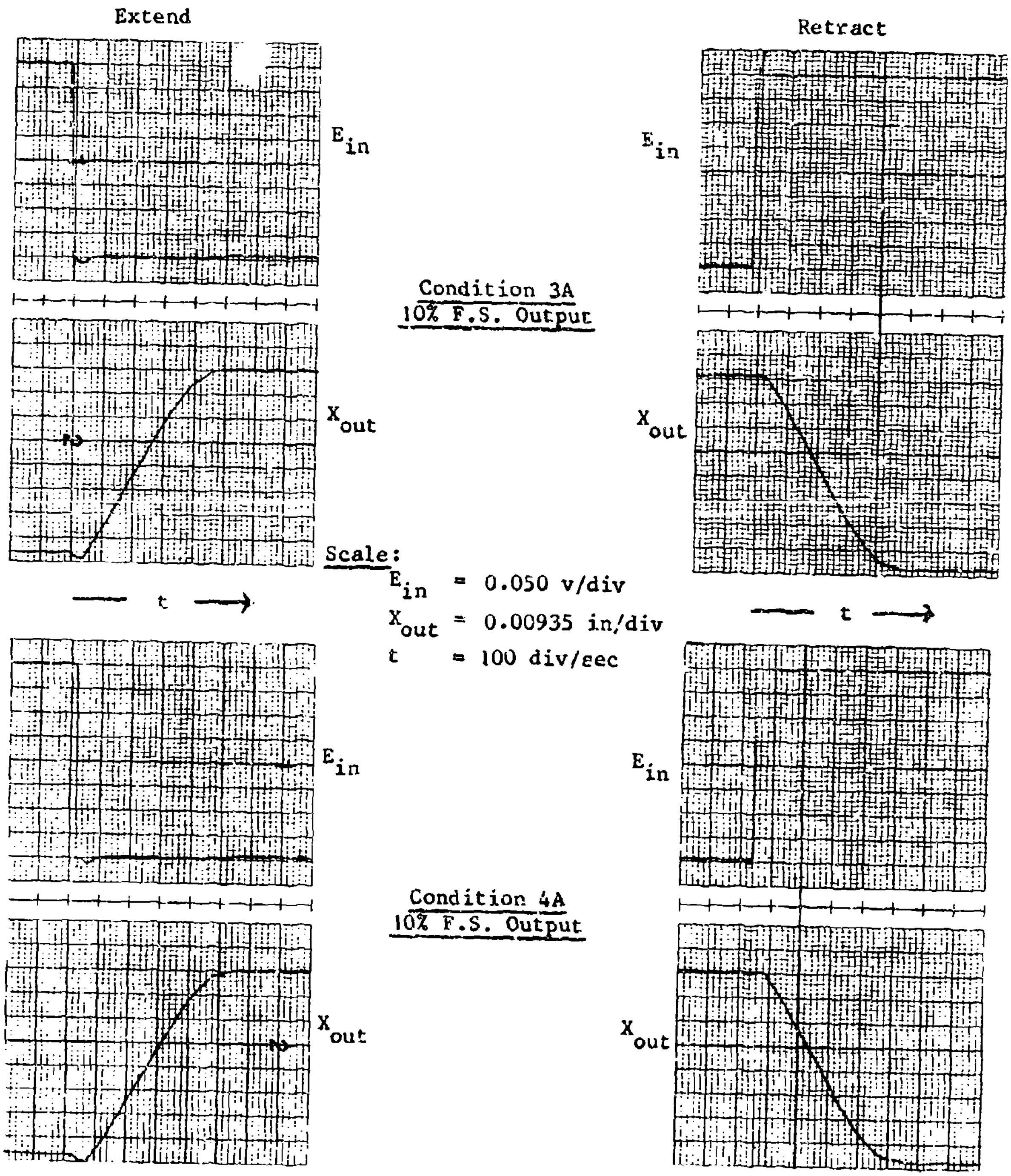
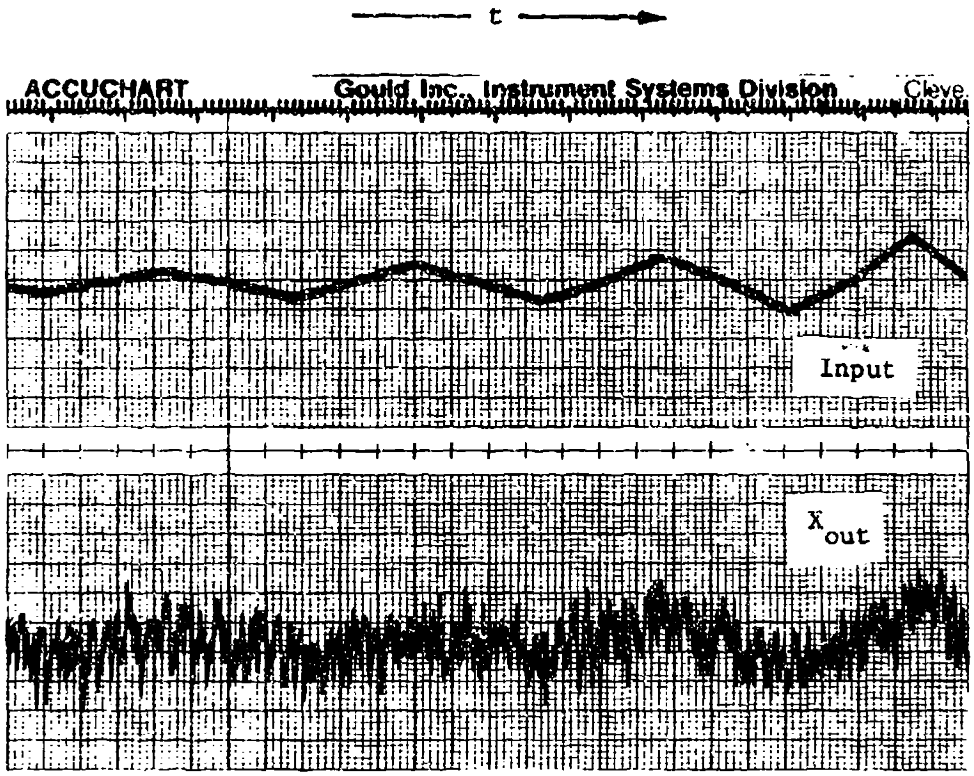


Figure 42. Step Response - Conditions 3A & 4A

Date Prepared 9/23/83

TEST ITEM -- Boeing Reconfigurable Fail Operative
Fly-By-Wire Servoactuator

TEST - Static Threshold - Condition 1B



Scale: Input = 0.002 v/div

X_{out} = 0.000374 in/div

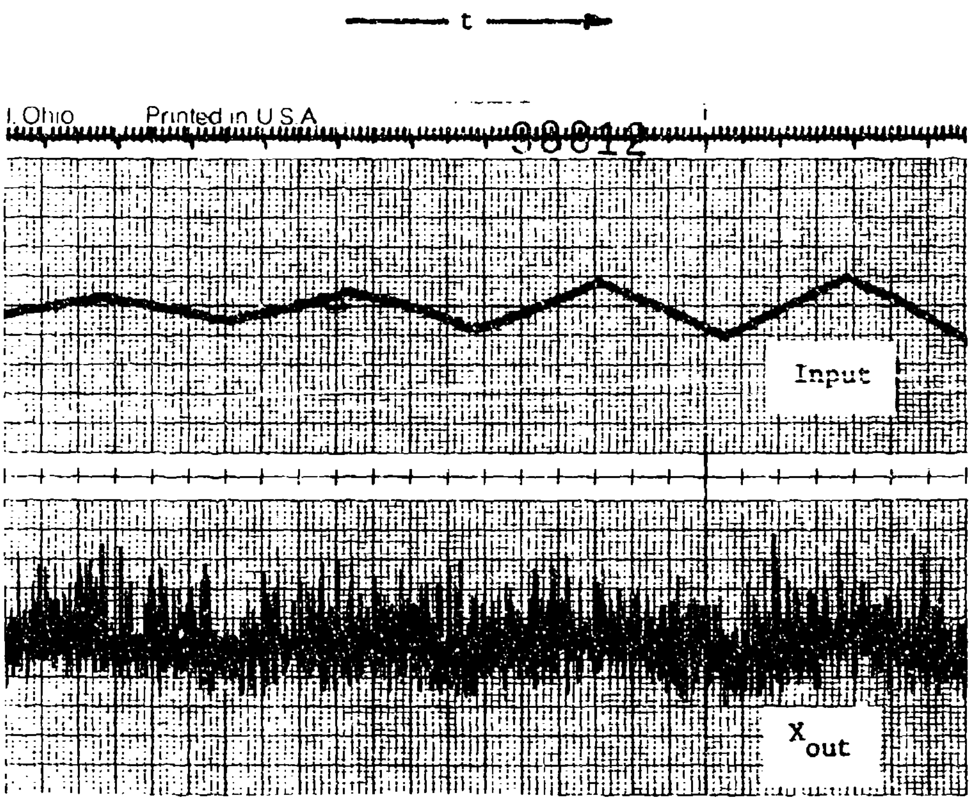
t = 1.0 div/sec

Figure 43. Static Threshold - Condition 1B

Date Prepared 9/29/83

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Static Threshold - Condition 1C



Scale:

Input = 0.002 v/div

X_{out} = 0.00093 in/div

t = 1.0 div/sec

Figure 44. Static Threshold - Condition 1C

TABLE 12
STATIC THRESHOLD - "B" LOAD

TEST ITEM: Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator **DATE PREPARED:** 10/3/83

TEST: Static Threshold in GPATR with "B" Load

Test Condition	Pk to Pk Input Volts	Static Threshold	
		% of Max Input	% of E_v Max
1B	0.012	0.066	0.75
2B	0.012	0.066	0.75
3B	0.012	0.066	0.75
4B	0.012	0.066	0.75
9B	0.020	0.111	1.125
11B	0.020	0.111	1.125
14B	0.016	0.088	1.00
15B	0.016	0.088	1.00
16B	0.016	0.088	1.00

TABLE 13
STATIC THRESHOLD - "C" LOAD

TEST ITEM: Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator **DATE PREPARED:** 10/3/83

TEST: Static Threshold in GPATR with "C" Load

Test Condition	Pk to Pk Input Volts	Static Threshold	
		% of Max Input	% of E_V Max
1C	0.050	0.277	3.125
2C	0.050	0.277	3.125
3C	0.050	0.277	3.125
4C	0.040	0.222	2.500
9C	0.070	0.388	4.375
11C	0.050	0.277	3.125
14C	0.050	0.277	3.125
15C	0.160	0.888	10.000
16C	0.180	1.000	11.250

friction of the piston seals, increasing the threshold value. The threshold measured for conditions 9C and 11C with one half of the actuator operating show changes from the threshold of test conditions 1C through 4C. The percent increases are similar to those of the unloaded test conditions (reference Table 2). The test conditions 14C through 16C as shown on Table 13 show that bias can vary the threshold measurement. For this load, the bias conditions all generate thresholds somewhat larger than that measured with no applied bias.

From these measurements of static threshold, it is apparent that the symmetrical load condition B does not greatly increase the static threshold. (All threshold values are less than 0.12% of maximum input and less than 1.25% of the input for maximum servovalve spool stroke.) The effect of the bias load C is to increase the threshold by nominally 90%. However, the static threshold values still remain less 0.200 percent of maximum input for all the load C test conditions used.

Dynamic Threshold at Loads B and C

Figure 45 shows the data recorded in establishing the dynamic threshold for condition 1B. Note that as with the unloaded tests, the test actuator output moves at the nominal 2 Hz input frequency but does not track the input amplitude change until some minimum input amplitude is reached. This figure is similar to Figure 6 for unloaded condition 1.

Figure 46 shows the data recorded for establishing the dynamic threshold for condition 1C. Note that the X_{out} scale is half as sensitive on this figure as on Figure 45. The effect of the load C is to suppress somewhat (compared to unloaded or load C conditions) the sinusoidal hunting before the output tracks the input amplitude.

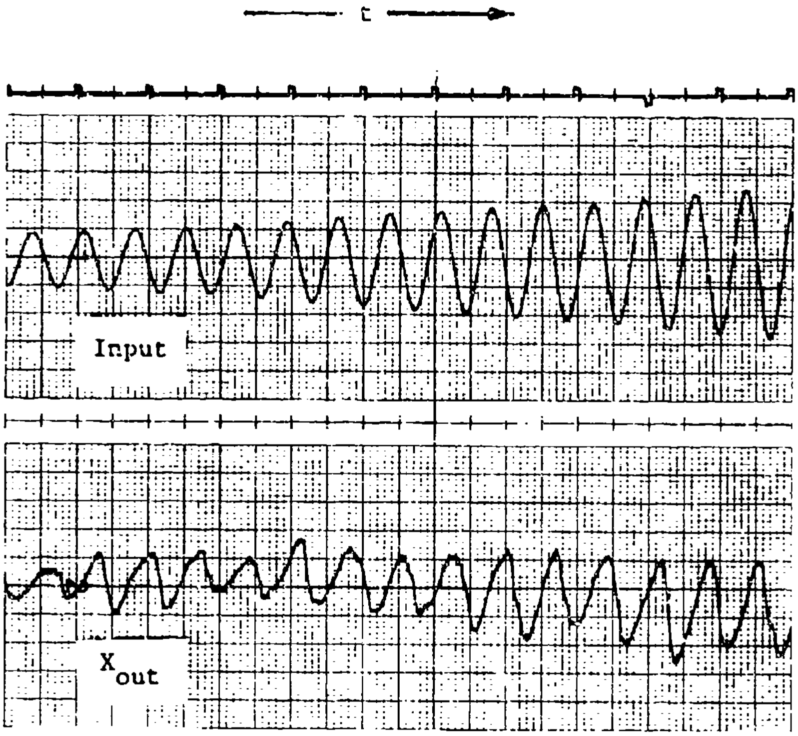
Table 14 lists the dynamic threshold measured on the test actuator for load condition B. The measurement values are very similar to those shown on Table 3 for the unloaded measurements. (The similarity is expected since load B provides only small loads with small motions around the centered actuator position.) The values for conditions 1B through 4B are almost identical with each other, as expected. The loss of one half of the actuator (test conditions 9B and 11B) results in a negligible change in the dynamic threshold. Test condition 14B and 16B show bias conditions which reduce the dynamic threshold value nominally 25% from that measured by no bias test conditions 1B through 4B. (These bias conditions reduce the servovalve force fight and improve the small signal dynamic performance.) Note that for all test conditions the dynamic threshold input does not exceed 0.50% of the maximum command input (or the corresponding 11.24% of the input to achieve maximum spool stroke).

Table 15 lists the dynamic threshold measured on the test actuator for load condition C. As stated previously, this load creates a bias load of 4.675 pounds towards the midstroke position (based upon th 5,500 lbs/inch gradient and the 0.85 inch steady state position of the actuator from the midstroke position). With the actuator holding a load, the increase in internal seal friction of the actuator due the cylinder pressaure changes would increase the actuator friction and the dynamic threshold. Comparing the Table 15's dynamic threshold for conditions 1C through 4C to corresponding test conditions 1 through 4 on Table 3 show this effect. The dynamic threshold for load C is nominally twice that for the unloaded conditions. As with load condition B,

Date Prepared 9/23/83

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Dynamic Threshold - Condition 1B



Scale: Input = 0.005 v/div

X_{out} = 0.000935 in/div

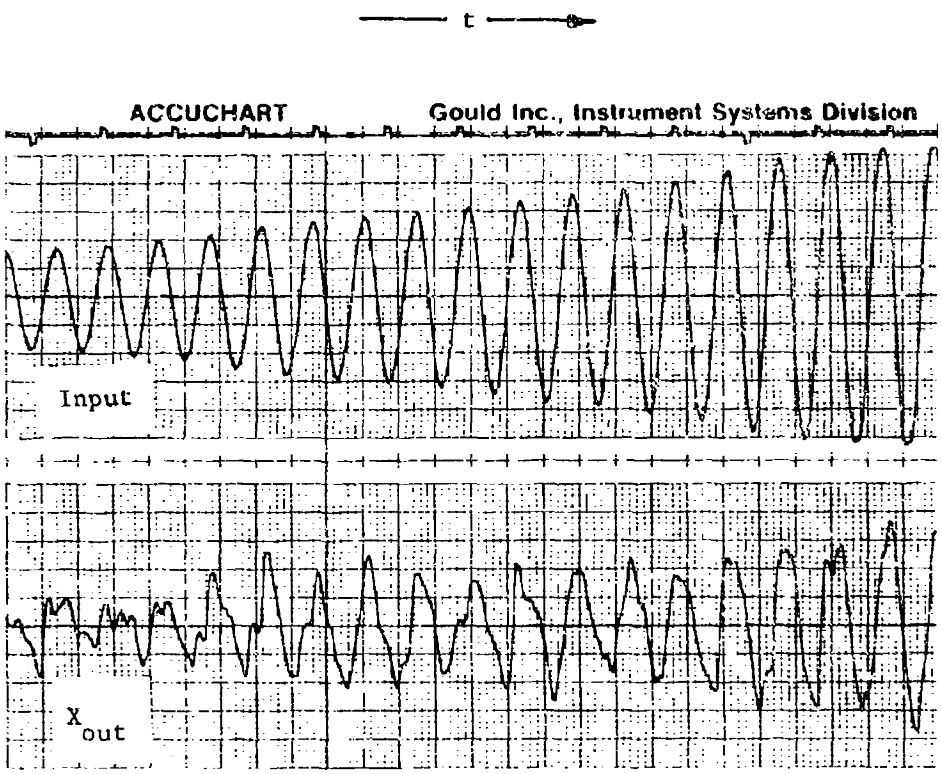
t = 10 div/sec

Figure 45. Dynamic Threshold - Condition 1B

Date Prepared 9/29/83

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Dynamic Threshold - Condition IC



Scale:

Input = 0.005 v/div

X_{out} = 0.0018 in/div

t = 10 div/sec

Figure 46. Dynamic Threshold - Condition IC

TABLE 14
DYNAMIC THRESHOLD - "B" LOAD

TEST ITEM: Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator **DATE PREPARED:** 10/3/83

TEST: Dynamic Threshold in GPATR with "B" Load

Test Condition	Pk to Pk Input Volts	Dynamic Threshold % of Max Input	Dynamic Threshold % of E_v Max
1B	0.070	0.777	8.75
2B	0.085	0.944	10.63
3B	0.075	0.833	9.37
4B	0.060	0.666	7.50
9B	0.080	0.388	10.00
11B	0.090	1.000	11.25
14B	0.058	0.638	7.18
15B	0.070	0.777	8.75
16B	0.053	0.583	6.56

TABLE 15
DYNAMIC THRESHOLD - "C" LOAD

TEST ITEM: Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator **DATE PREPARED:** 10/3/83

TEST: Dynamic Threshold in GPATR with "C" Load

Test Condition	Pk to Pk Input Volts	Dynamic Threshold % of Max Input	Dynamic Threshold % of E_V Max
1C	0.105	0.583	6.563
2C	0.075	0.417	4.688
3C	0.115	0.639	7.188
4C	0.095	0.528	5.938
9C	0.100	0.556	6.250
11C	0.120	1.667	7.500
14C	0.095	0.528	5.938
15C	0.225	1.25	14.063
16C	0.100	0.556	6.250

there is very little change with test conditions 9C and 11C with one half of the actuator bypassed. The effect of the bias variations of test conditions 14C, 15C and 16C is that test condition 15C increases the dynamic threshold to twice that when no bias is applied. The biases used for test conditions 14C and 16C show little change from the no bias conditions. This effect is similar to that experienced with load conditions B. The worst condition dynamic threshold experienced with load C yielded a value of 1.25% of the maximum input.

The effect of the bias load of load C is to increase the dynamic threshold compared to the unloaded or load condition B. Load condition B increases the dynamic threshold very little over the unloaded case for similar test conditions.

Frequency Response at Loads B and C

Figure 47 shows the frequency response data for test condition 1B. Note the slight amplitude peaking (about 1 dB) at 1.6 Hz. This is a change from the unloaded response tests which exhibited no peaking (reference Figure 7) and the "0" load tests which exhibited 0.25 dB peaking (reference Figure 37).

Figure 48 shows the frequency response data for test condition 1C. Note that the amplitude peaking is negligible (resembling the "0" load results of Figure 37.) The increased internal seal friction due to the bias load may attenuate the response peaking by increasing the damping of the actuator motion.

Table 16 lists the frequency response test results for the test conditions run with load B. The test results reflect the effect of the amplitude peaking. For the test conditions 1B through 4B, the -3 dB frequency occurs at 0.3 to 0.9 Hz higher than for the corresponding unloaded test condition (reference Table 3). The -3 dB frequency occurs 0.15 to 0.4 Hz higher than the "0" load conditions 1A through 4A. While the amplitude response "improved" with load B, the frequency at which the -90° phase angle occurs decreased for all the load B test conditions. The frequencies decrease varied from 0.7 Hz to 0.0 Hz.

Note that as listed on Table 16, test conditions 9B and 11B with one half the actuator operating reduces both the -3 dB and -90° frequencies about 10% from those of the test conditions 1B through 4B. This is consistent with the flow gain of the control valves being reduced more by a given load with one section than with the same load being shared by both halves of the actuator.

Test conditions 14B through 16B show a minor effect of control section bias changes on frequency response. There is a nominal 0.45 Hz range change in the -3 dB and -90° frequencies.

Table 17 lists the frequency response measured for the test conditions with load C. For all test conditions with load C, the -3 dB and -90° frequencies are from 0.3 to 1.15 Hz lower than that measured with the test actuator unloaded (reference Table 4). This is expected since the effect of the bias load is to reduce the flow gain of the servovalves in responding to the command inputs (in one direction of motion). The frequencies at which -3 dB and -90° for test conditions 1C through 4C are similar (as expected since the test conditions are a reassignment of the active and model roles). The effect of running only one half of the test actuator (test conditions 9C and 11C) is to reduce the frequency at which -3 dB occurs from 3.00 Hz with both halves of the

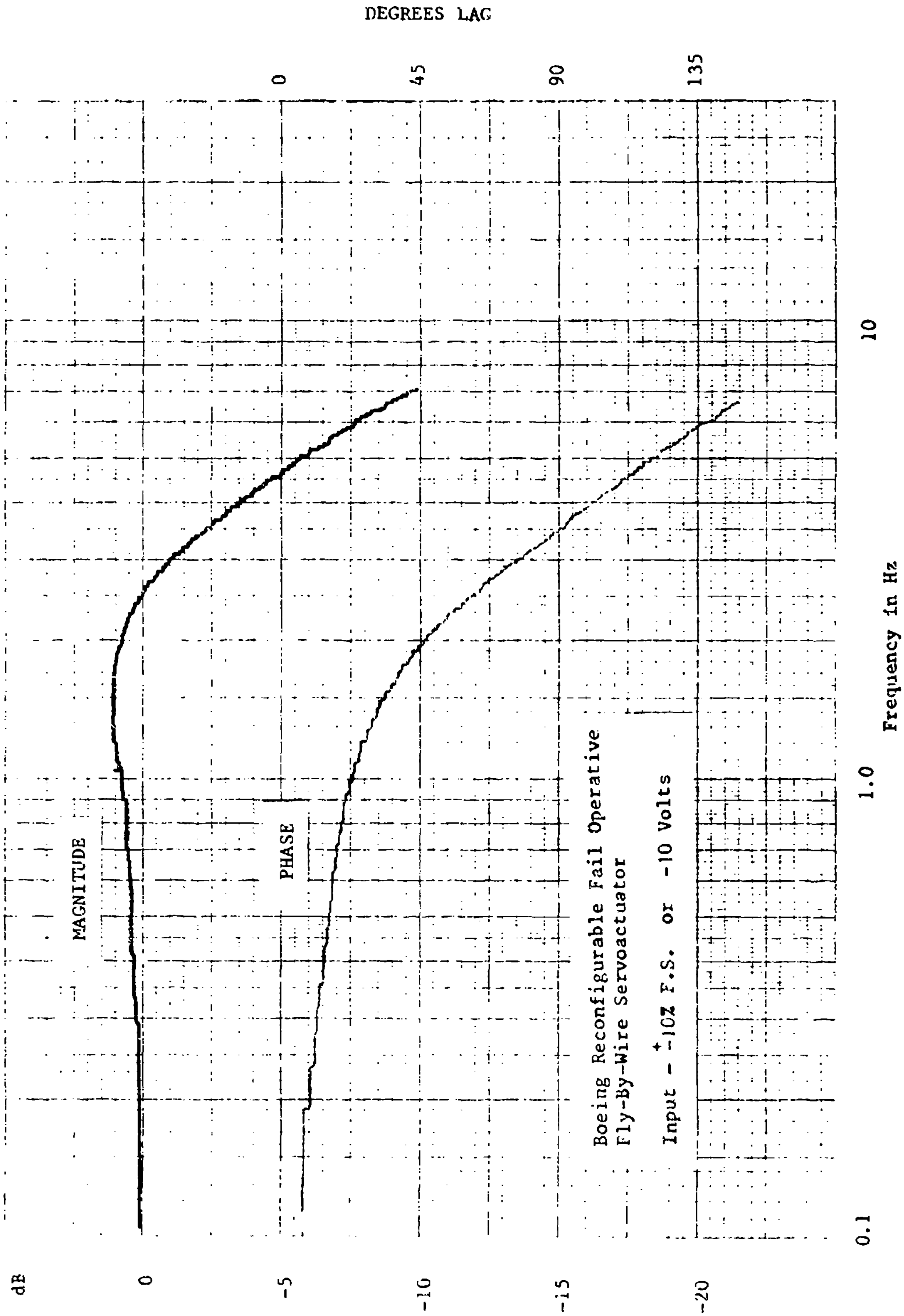


Figure 47. Frequency Response - Condition 1B

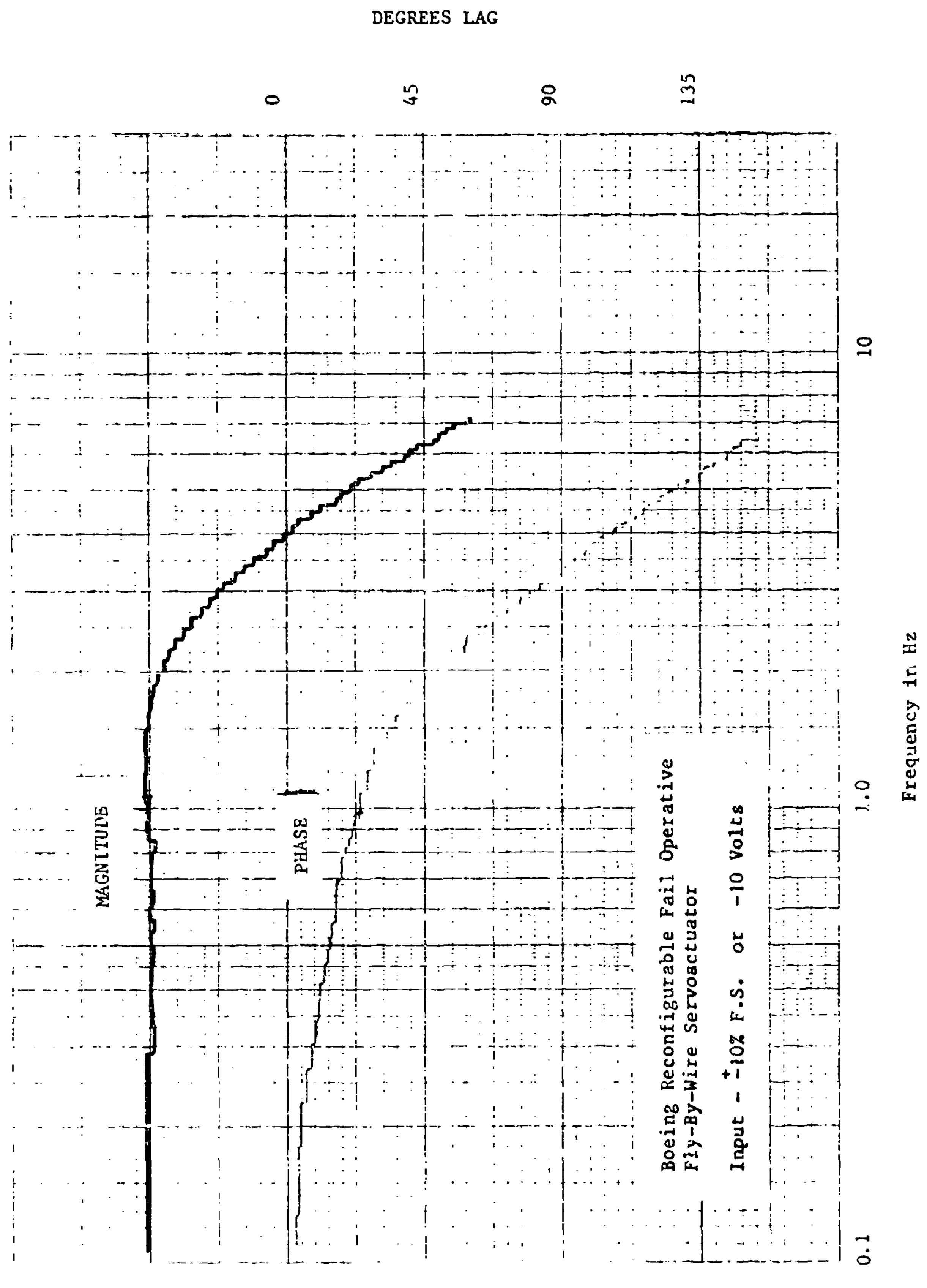


Figure 48. Frequency Response - Condition 1C

TABLE 16
FREQUENCY RESPONSE - "B" LOAD

TEST ITEM: Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator **DATE PREPARED:** 10/3/83

TEST: Frequency Response in GPATR with "B" Load

Test Condition	Output 10% Full Scale	
	-3 dB Hz	-90° Hz
1B	3.75	3.50
2B	3.70	3.40
3B	3.70	3.50
4B	3.90	3.50
9B	3.30	3.40
11B	3.00	3.20
14B	3.20	3.50
15B	3.10	3.05
16B	3.65	3.30

TABLE 17
FREQUENCY RESPONSE - "C" LOAD

TEST ITEM: Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

DATE PREPARED: 10/3/83

TEST: Frequency Response in GPATR with "C" Load

Test Condition	Output 10% Full Scale	
	-3 dB Hz	-90° Hz
1C	3.10	3.25
2C	3.00	3.25
3C	3.00	3.30
4C	3.00	3.25
9C	2.15	2.95
11C	2.00	2.85
14C	3.20	3.60
15C	2.60	2.80
16C	2.55	3.10

actuator operating to nominally 2 Hz. This is consistent with the reduced flow gain from the servovalve when the load is carried by half of the actuator. The effect of the changing bias conditions 14C, 15C and 16C show some effect on the frequency response, with the bias condition 14C giving the best frequency response of the three conditions. This response was an improvement over the "no bias" conditions 1C through 4C.

Load C degraded the frequency response more than load B. Both load conditions reduced the frequency response from that of the unloaded test conditions. The reduction in frequency response can be attributed to the reduction of flow gain with actuator load that occurs with any actuator system which uses sharp edged control valves to meter flow to the actuator drive area.

Hysteresis at Loads B and C

Figure 49 shows the test data taken for the hysteresis measurement with test condition 1B. Note the separation located near the zero input point. Load B is a symmetrical spring load on the control actuator. Therefore, as the actuator moves through null (midstroke), the load on the actuator reverses. In order to move the actuator against the load, the active channels must be commanded with an input amplitude sufficient to drive the channels out of the force fight channel mismatch. Where the load reverses, the polarity of the input amplitude which overcomes the force fight must also change. The combination of the active channels with a force fight and the reversing load on the actuator results in the type of hysteresis plot illustrated by Figure 49. Note that for input levels less than -0.363 volt and greater than +0.363 volt, the hysteresis plot resembles the unloaded plot for condition 1 (reference Figure 8) with a low amplitude hunting and no general separation.

The effect of load C on the hysteresis measurement is shown on Figure 50. Note that the hysteresis plot shows larger amplitude hunting than the unloaded plot of Figure 8. Since the load on the test actuator is primarily a bias load towards midstroke of the actuator, there is no load reversal during the data taken for Figure 50 and no corresponding opening of the hysteresis loop. The primary effect of the load of condition 1C is to increase the amplitude of the hunting compared to the unloaded condition 1.

Table 18 lists the hysteresis measured for the test conditions with load B. Note that for test conditions 1B through 4B the hysteresis measures 0.6% of the maximum input, a value 10 times that of the corresponding unloaded test conditions 1 through 4 (reference Table 5). Test conditions 9C and 11C with one half of the actuator bypassed have reduce hysteresis. This is consistent with the elimination of the force fight condition when only one half of the actuator is operating. Test condition 14C also has a low hysteresis measurement. Since this test condition is one of the varying bias conditions, and bias changes change the force fight, the results of test condition 14C are consistent. The other two bias test conditions 15B and 16B result in an increase and a decrease from the test conditions 1C through 4C with normal bias.

Table 19 lists the hysteresis measured for the test conditions with load C. The hysteresis listed for all test conditions is 0.4% of the full scale input. This value is less than that for the B load conditions. However, the values represent the nominal "non-coincidence" of the output position for the same

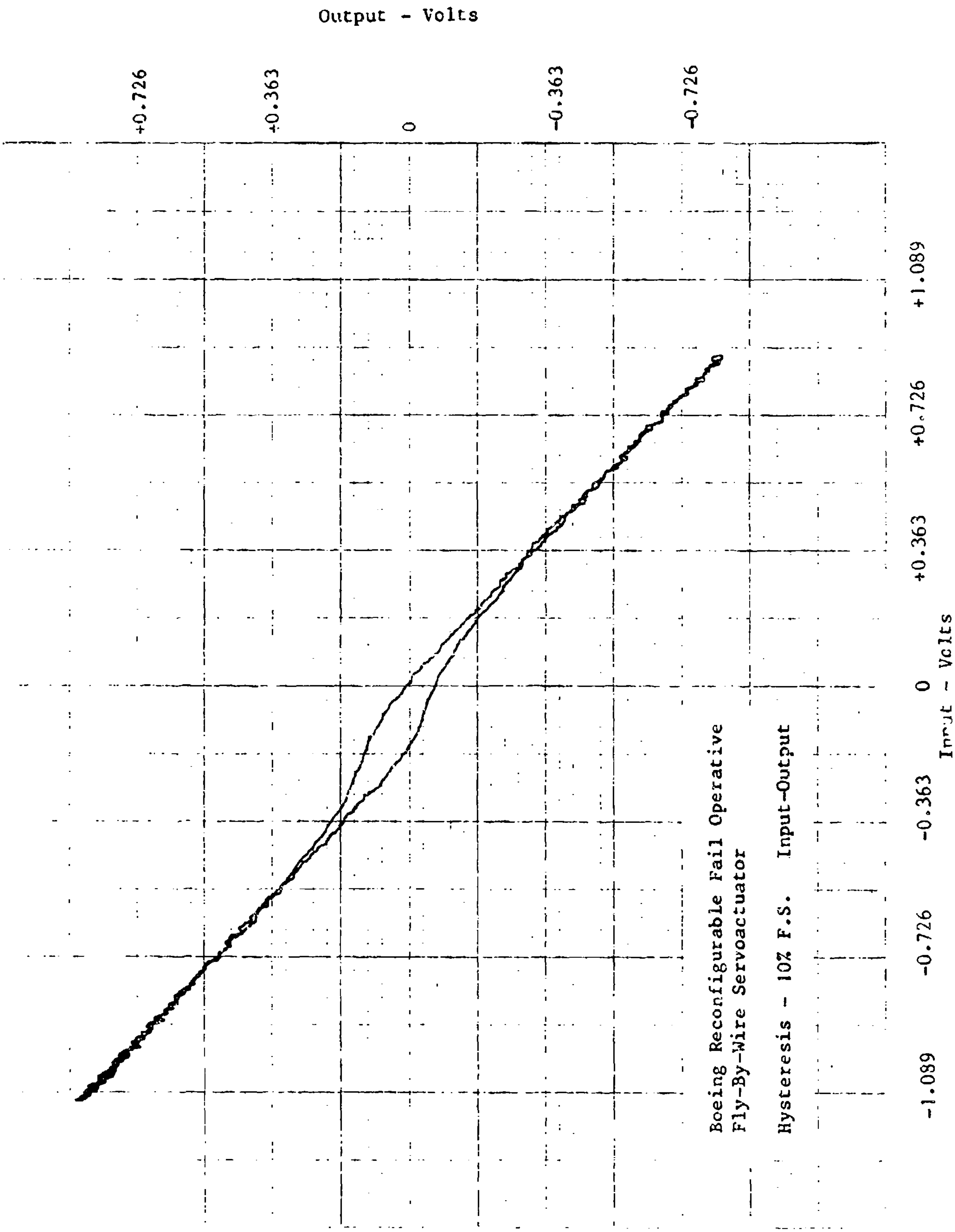


Figure 49. Hysteresis - Condition 1B

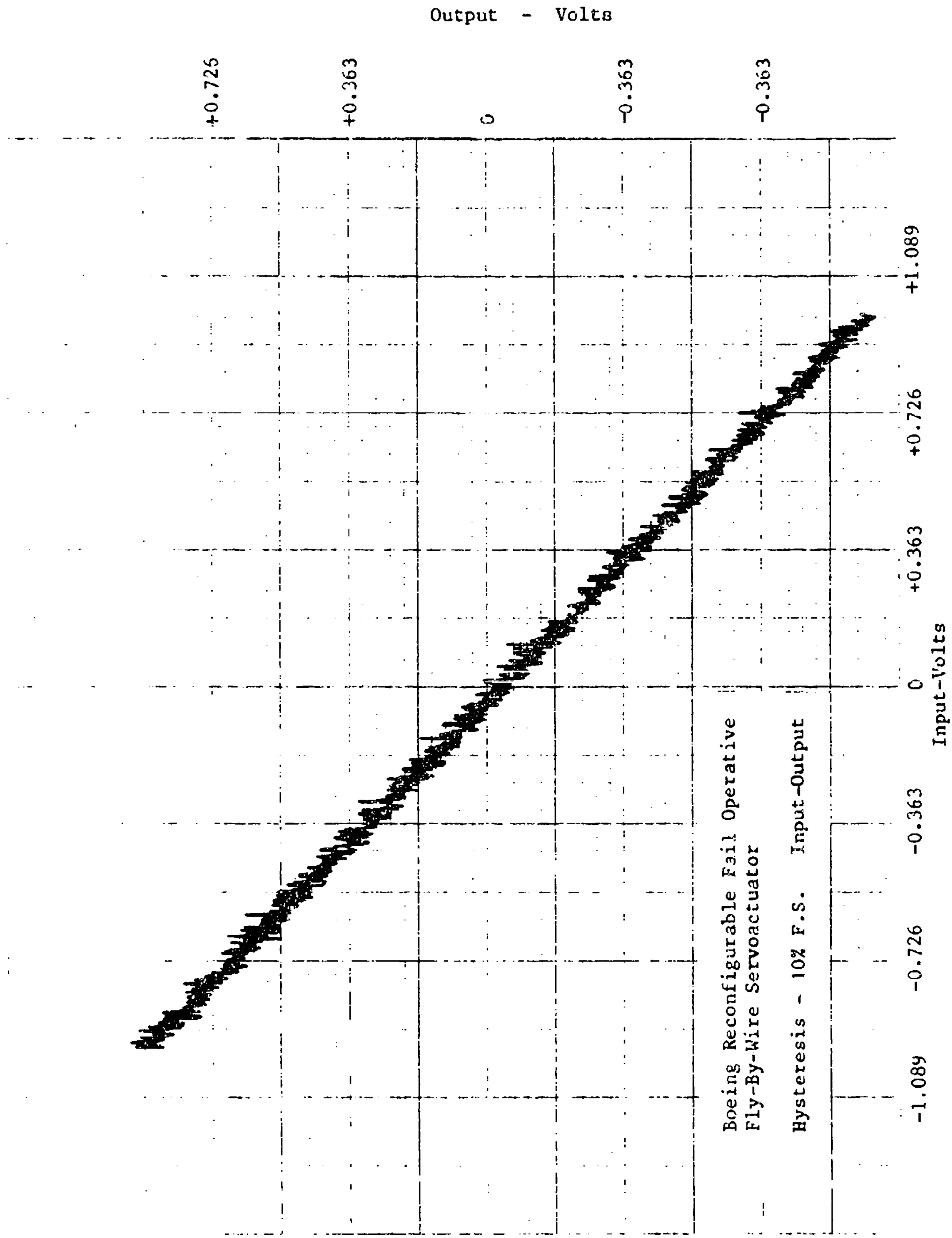


Figure 50. Hysteresis - Condition 1C

TABLE 18
HYSTERESIS - "B" LOAD

TEST ITEM: Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator **DATE PREPARED:** 10/3/83

TEST: Hysteresis in GPATR with "B" Load

Test Condition	% Full Scale	% of E_V Max
1B	0.60	6.80
2B	0.60	6.80
3B	0.60	6.80
4B	0.60	6.80
9B	0.20	2.27
11B	0.20	2.27
14B	0.10	1.13
15B	0.67	7.50
16B	0.50	5.63

TABLE 19
HYSTERESIS - "C" LOAD

TEST ITEM: Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator **DATE PREPARED:** 10/3/83

TEST: Hysteresis in GPATR with "C" Load

Test Condition	% Full Scale	% of E_V Max
1C	0.00	0.00
2C	0.00	0.00
3C	0.00	0.00
4C	0.00	0.00
9C	0.00	0.00
11C	0.00	0.00
14C	0.00	0.00
15C	0.00	0.00
16C	0.00	0.00

input level. The hunting is similar for all the load C test conditions which yields the same hysteresis value.

The general effect on hysteresis of loading the test actuator is to increase the value by 6 to 10 times that of the unloaded actuator. Bias loads increase the hysteresis less than symmetrical loads. The bias load creates a low amplitude hunting motion of the test actuator while the symmetrical loads create an "open" hysteresis loop around zero input and load reversal. The maximum hysteresis for any test condition with load B or C is less than or equal to 0.6 percent of the maximum input to the test actuator.

Failure Transients with Loads B and C

Figure 51 is the time history plot of the input and output characteristics of the test system with a slowover ramp input sequentially applied to the inputs with load B. The ramp is applied to channels 1, 2 and 3 in order. The system is configured initially with channels 1 and 3 active, and 2 and 4 as models. The number of counts before a "failed" channel which has corrected itself could be voted "good" and used again was increased enough to prevent the "good" vote.

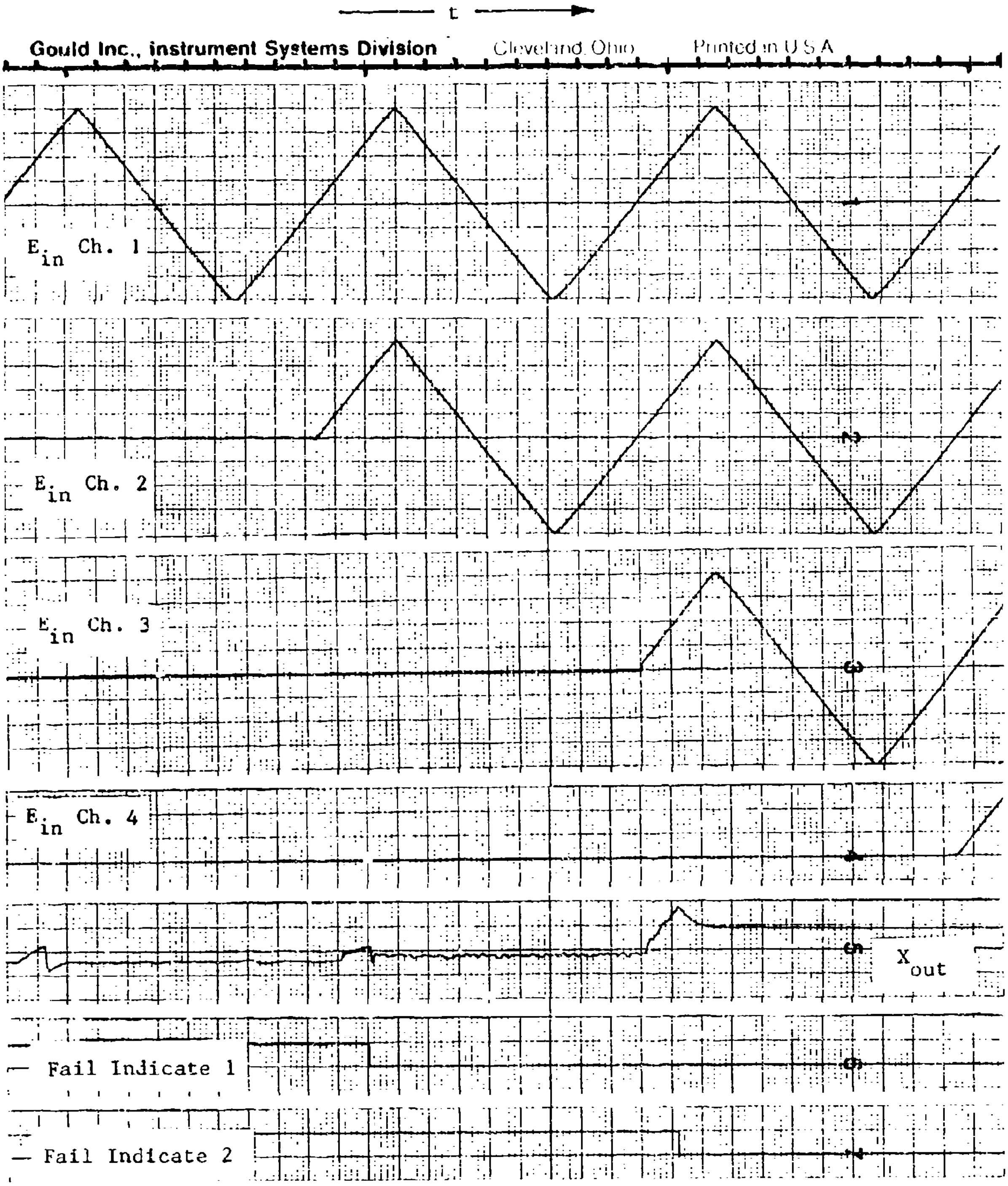
As shown on Figure 51, the ramp input into channel 1 causes the output to track the ramp input up to the failure detection level. The output change is 0.83% of the total stroke. Upon detection of the failure, control of actuator section 1 is transferred to channel 2 and the actuator moves back to the null output position. The return motion overshoots by 0.56% of the total stroke and then returns to null. Note that actuator section 2 opposes section 1's response to the ramp input, minimizing the output motion. Upon the application of the slowover ramp input to channel 2, section 1 of the actuator is declared failed and bypassed. The actuator initially moves 0.83% of the total stroke in response to the ramp input before a failure is declared and section 1 of the actuator bypassed. After the application of the ramp input to channel 3, the actuator moves 2.8% of the total actuator stroke in response to the input before a failure is declared. This movement is greater than that measured after the first and second failure inputs. The greater movement reflects that with section 1 bypassed, there is no other actuator section opposing section 2's response to the ramp input. The motion after the third input failure is a result of the load system moving the actuator output to its "zero force" position.

Figure 52 shows the same input sequence as used for Figure 51. For Figure 52, load C is used. Note that the scale used on the X_{out} trace for Figure 52 is one quarter of the scale on Figure 51. The actuator output change before detection of the first failure (with channel 1 input ramp) is 1.66% of the total actuator stroke. The actuator change before detection of the second failure is 1% of the total actuator stroke. After the third failure, the test system actuator is bypassed and driven to the "zero force" or null condition of the load actuator.

The effect of the load conditions on failure transients resulting from "slowover" failures is not significant. The amplitude of the failure transient resulting from the slowover input into channel 2 for Condition 22 is 2.2% of the maximum actuator stroke. The loaded test condition transients resulting from the same input failures are 0.83% for load condition B and 1% for load condition C. Note that the unloaded test Condition 22 transient appears as primarily a null shift with the bypassing of actuator section 1. The actuator

TEST ITEM - Boeing Reconfigurable Fail Operative
Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 22B   Date Prepared 10/3/83



SCALE: E_{in} = 0.050 v/div

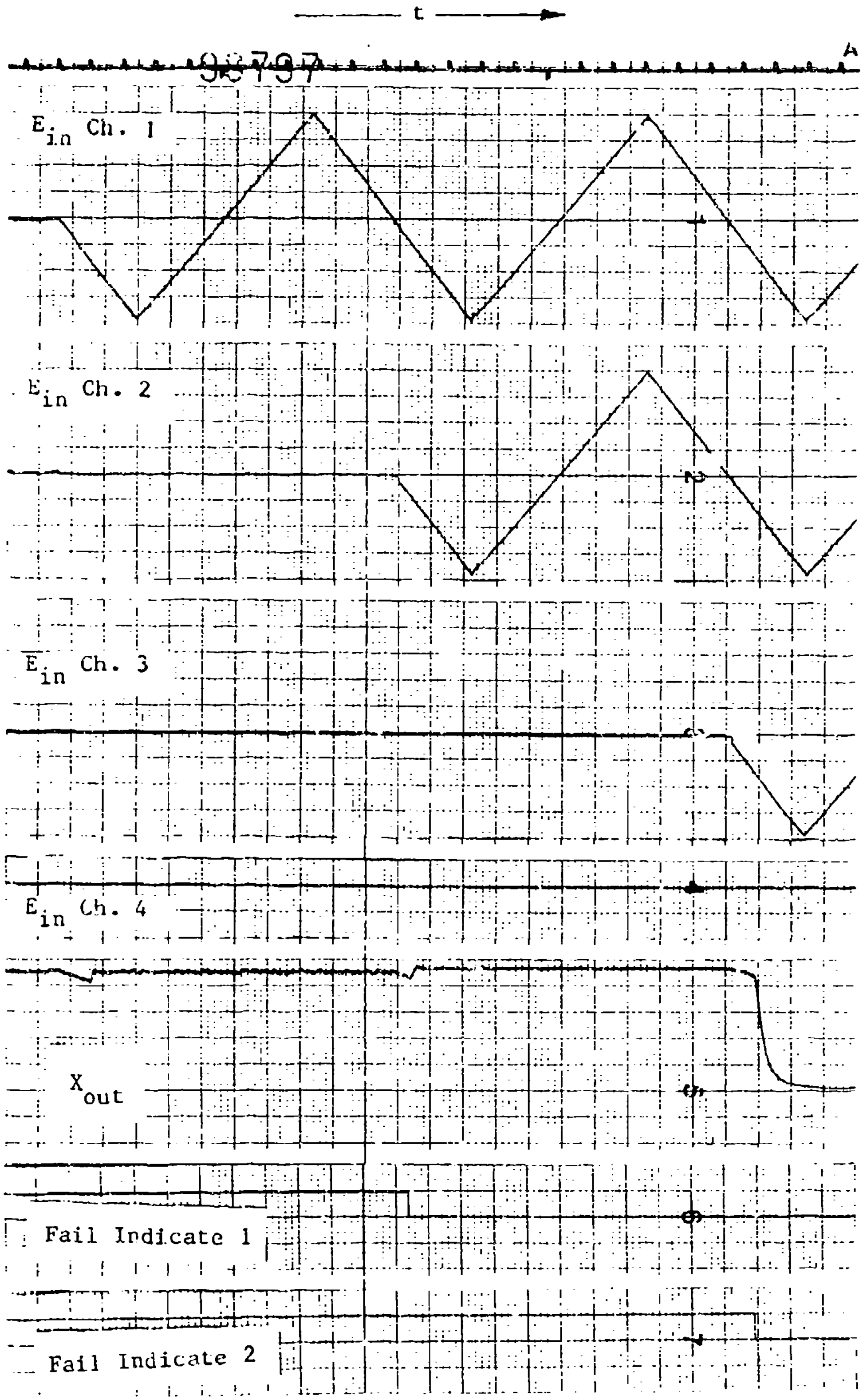
X_{out} = 0.00938 in/div

t = 5 div/sec

Figure 51. Failure Transient - Condition 22B

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 22C Date Prepared 10/3/83



SCALE:

E_{in} = 0.050 v/div

X_{out} = 0.0375 in/div

t = 5 div/sec

Figure 52. Failure Transients - Condition 22C

output does not appear to respond at all to the failure ramp input. With a load applied, the output does respond to the failure input. This difference in transient characteristic is probably due to the applied load forcing test system out of the force fight deadband (in order to hold the applied loads).

Figure 53 shows the effect of applying a ramp input of 0.4 volt/second sequentially to channels 1, 2 and 3 with the system operating with a sinusoidal input into all channels and with load B applied. The amplitude of the nominal 1.5 Hz sinusoidal signal is at the maximum input at that frequency without causing rate saturation. The principal effect of the failure is a null shift of 2.2% of the maximum actuator stroke when section 1 of the actuator is bypassed.

Figure 54 shows the effect applying a ramp input of 0.4 volt/second sequentially to channels 1, 2 and 3 with the system operating with a sinusoidal input into all channels and with load C applied. Note that the scale on X_{out} is 1/5 of that used on Figure 53 in order to include the output motion trace after application of the third slowover failure into channel 3. Note that the test actuator output moves to the load actuator "zero load" position after both test actuator sections are bypassed. The limited amplitude of the test actuator response prior to the bypassing both sections is due to the chart recorder pen operating at the edge of the brush recorder strip chart.

The slowover transient effects of Figures 53 an 54 are very similar to those shown on Figure 25 for no load conditions. The load effects of load B and C on the slowover failure transients while the test actuator is operating dynamically appear insignificant.

Figure 55 is a time response of the test system with the output initially at 50% extend (0.85 inch from null in the opposite direction from the normal "C" load) with the load C spring gradient.

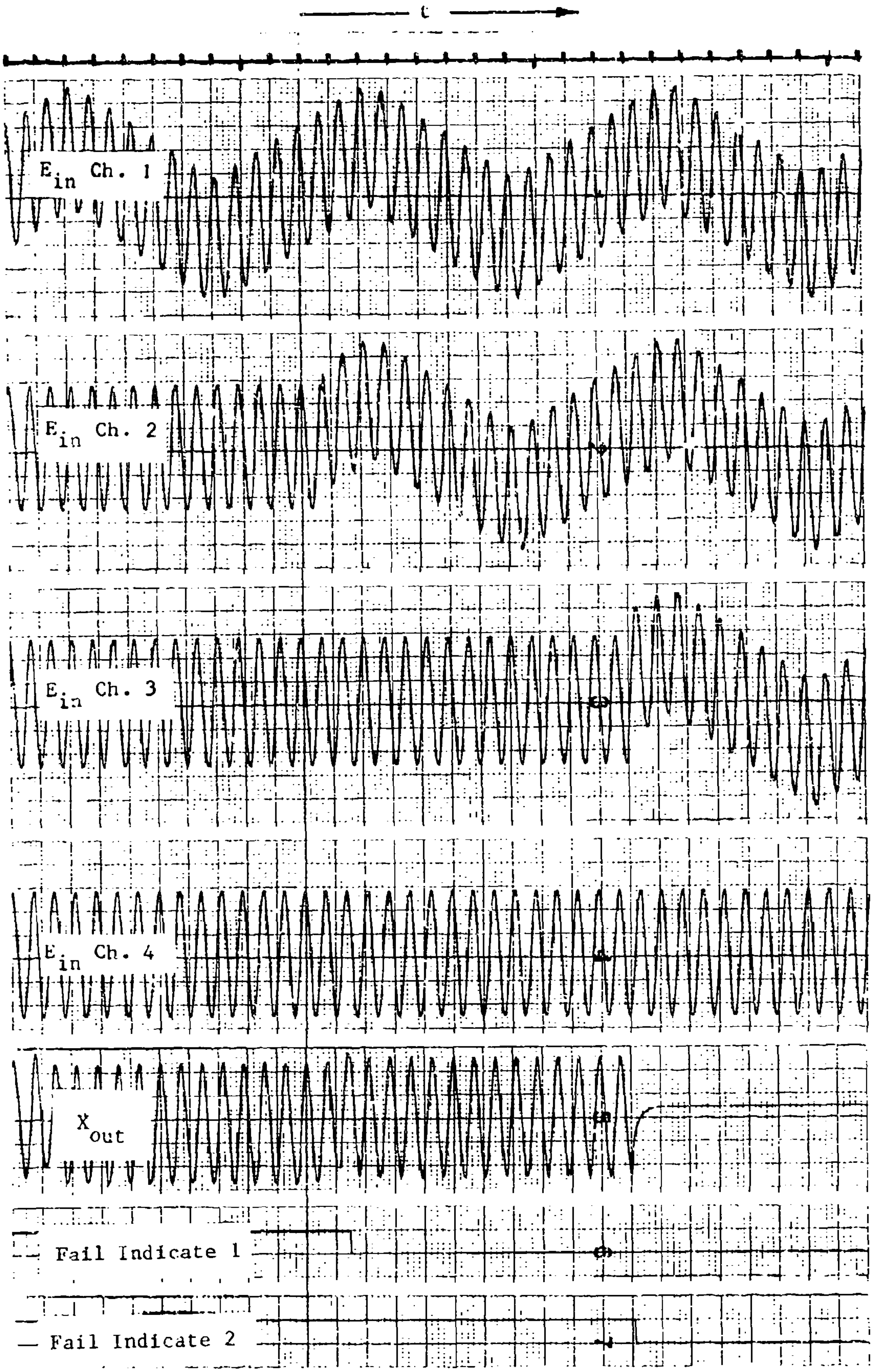
As shown on Figure 55, the first and second input failures create an output failure transient which lasts for 200 milliseconds. The output deviation amplitude is 1% of the maximum stroke for the channel 1 input failure. The output deviation amplitude for the second input failure (channel 2) which causes bypassing of section 1 is 1.5% of the maximum stroke. This second failure leaves actuator section 2 to hold the load. Upon the third input failure into channel 3, section 2 of the test actuator is also bypassed. This allows the load system to drive the test actuator to the load system's force null. Note that there is no observable null shift between the commanded position of the test actuator before and after failures. The negligible null shift reflects the effect of the bias load requiring both control channels to operate together out of the force fight region in order to hold the load.

Figure 56 shows the effect of grounding the inputs to channels 1, 2 and 3 sequentially with a sinusoidal input applied to all channels. Load B is applied to the test actuator. These results are similar to the unloaded actuator test results for condition 26 (reference Figure 28). There is a peak amplitude loss of 1% during the transition from one operating mode to another. There is a null shift of 0.8% of maximum actuator stroke between the operating modes.

Figure 57 shows the same failure test sequence for the inputs as used for Figure 56. The load condition for the test results shown on Figure 57 was load C. Note that the X_{out} scale shown on this figure is 1/5 that of Figure 56. The

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Failure Transients -- Condition 23B Date Prepared 10/3/83



SCALE:

E_{in} = 0.100 v/div

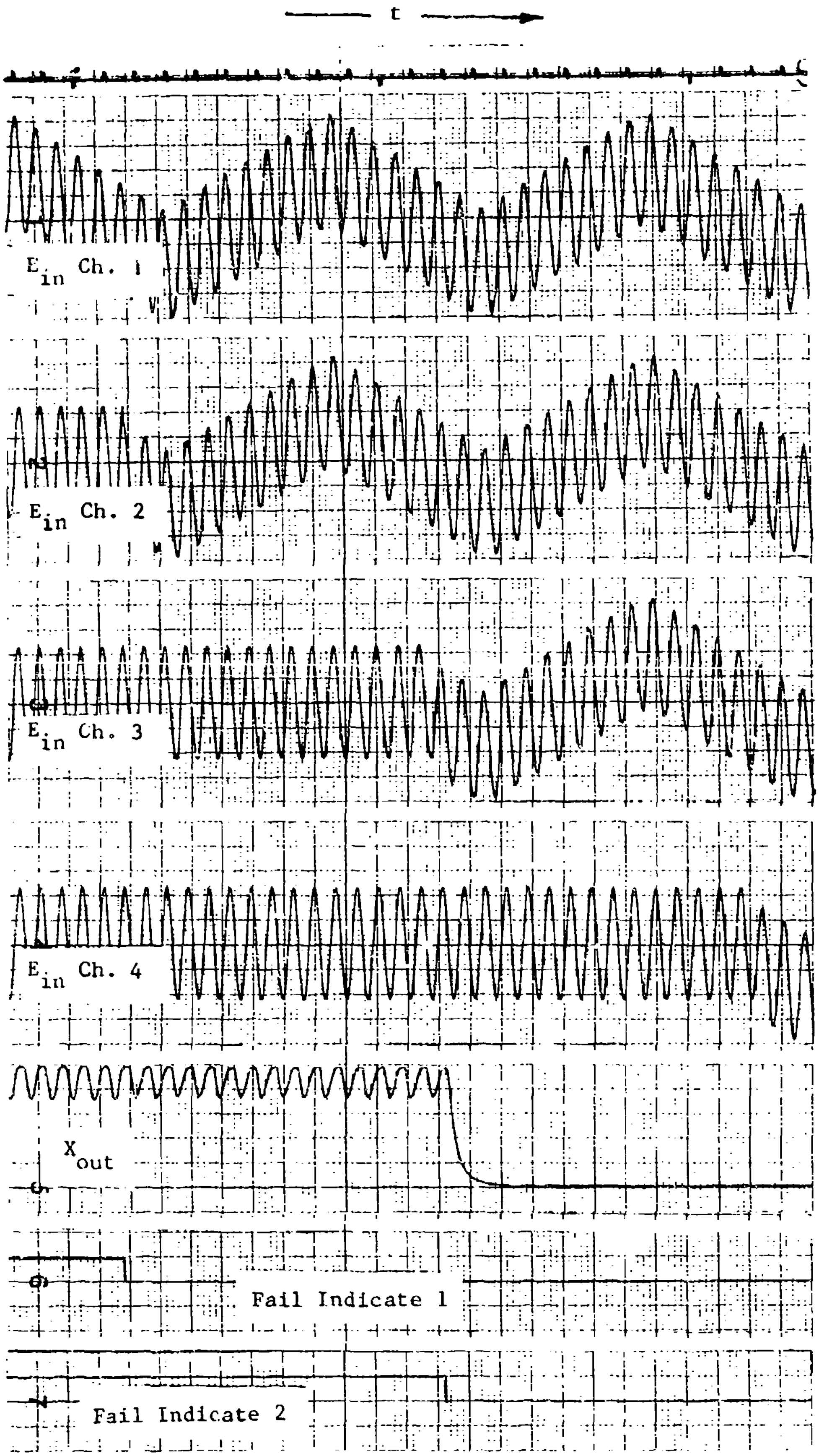
X_{out} = 0.1877 in/div

t = 5 div/sec

Figure 53. Failure Transients - Condition 23B

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 23C   Date Prepared 10/3/83



SCALE:

E_{in} = 0.100 v/div

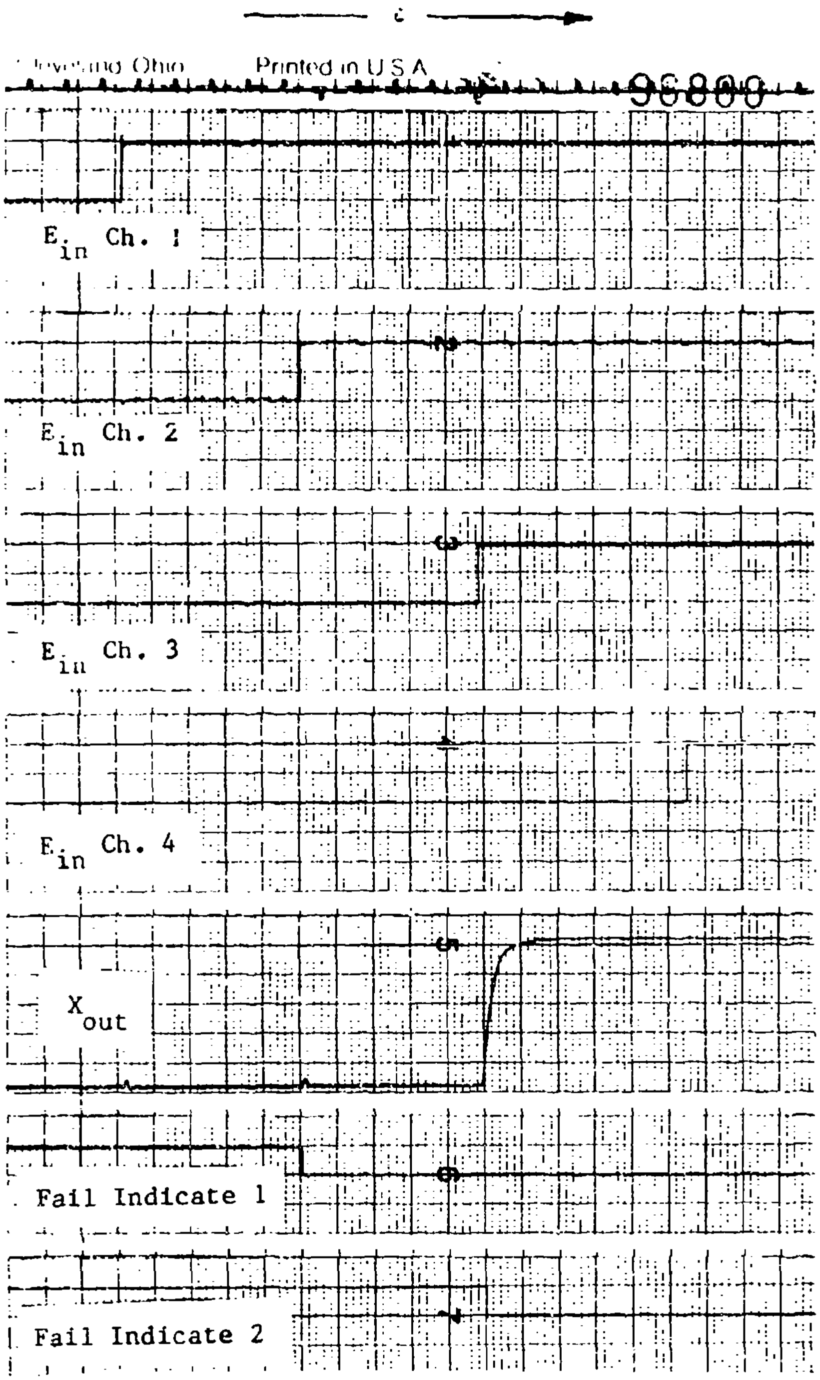
X_{out} = 0.0375 in/div

t = 5 div/sec

Figure 54. Failure Transients - Condition 23C

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 24C   Date Prepared 10/3/83



<u>SCALE</u>:

E_{in} = 0.500 v/div

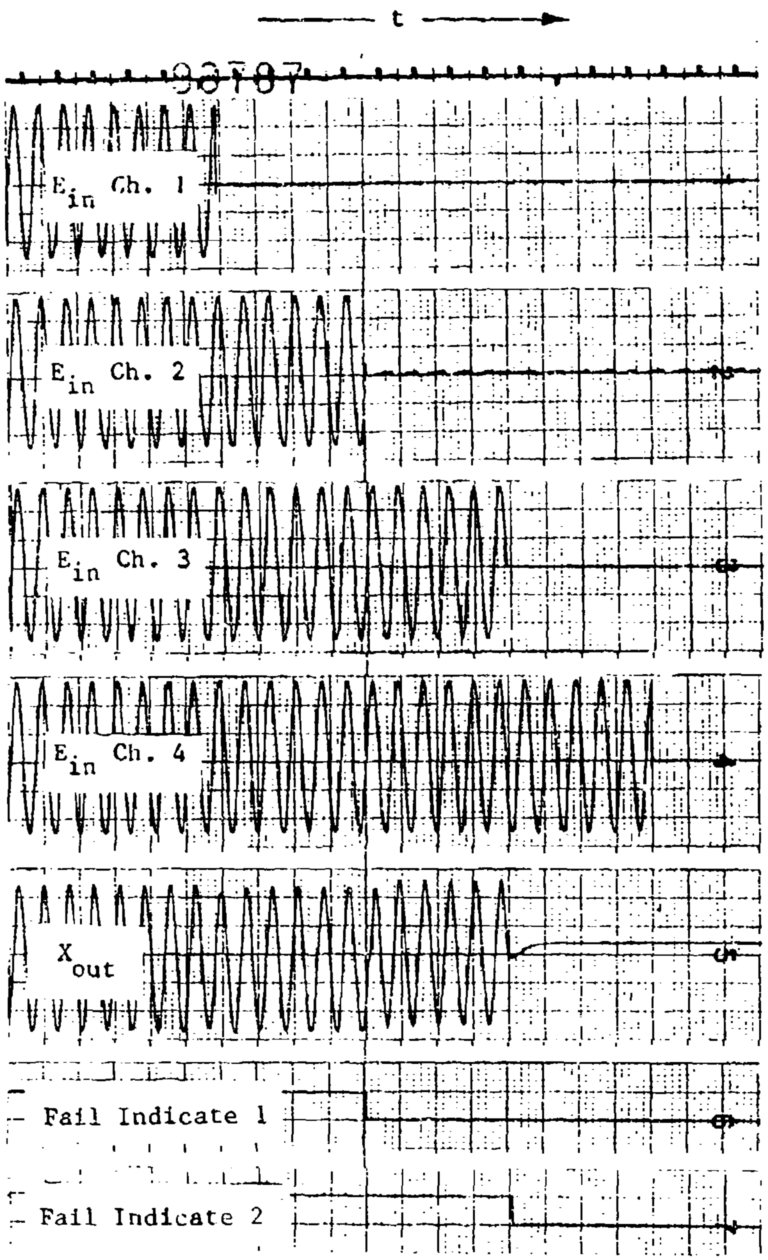
X_{out} = 0.0375 in/div

t = 5 div/sec

Figure 55. Failure Transients - Condition 24C

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 26B Date Prepared 10/3/83



SCALE:

E_{in} = 0.100 v/div

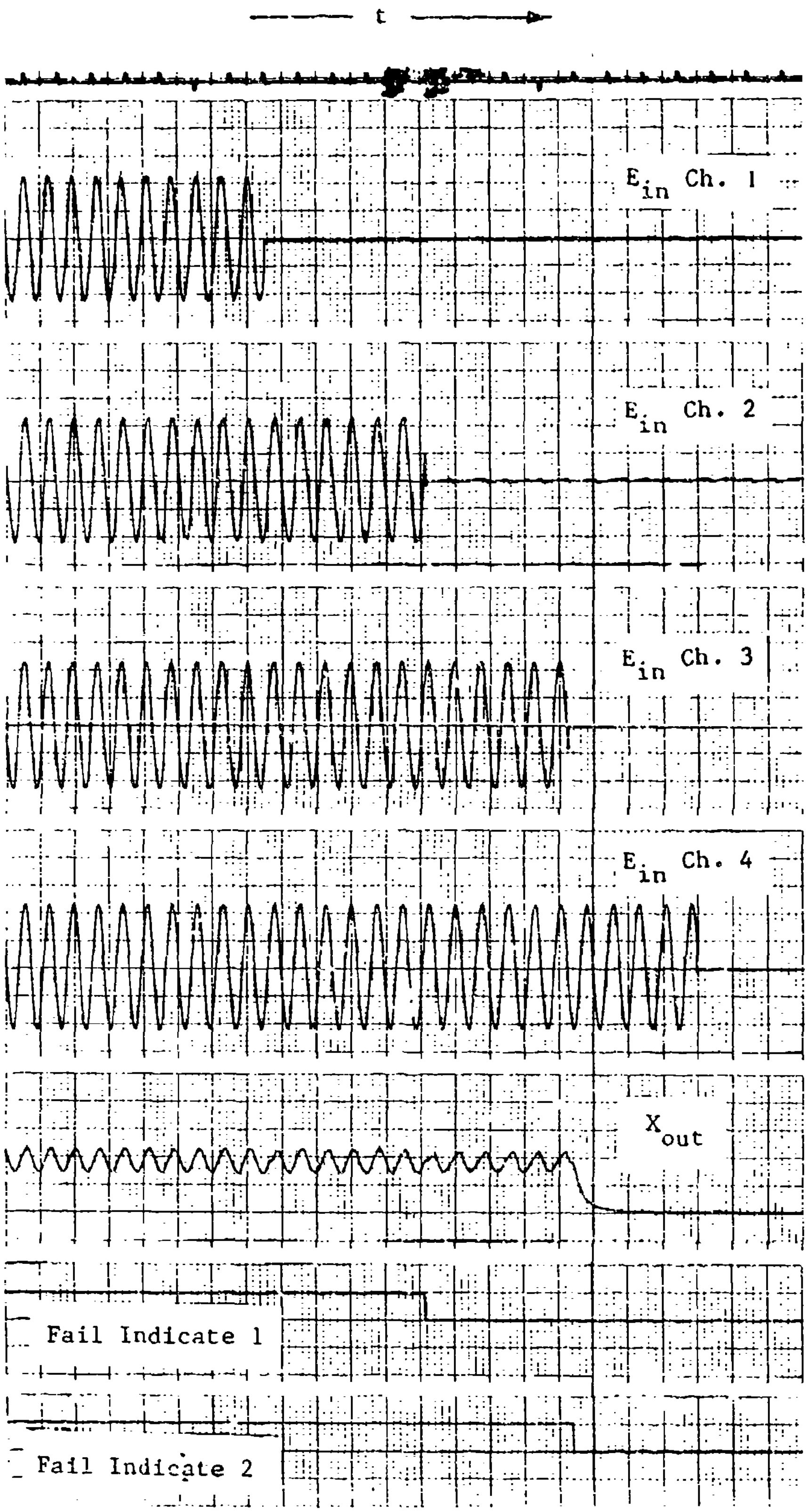
X_{out} = 0.0187 in/div

t = 5 div/sec

Figure 56. Failure Transients - Condition 26B

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 26C Date Prepared 10/3/83



SCALE:

E_{in} = 0.100 v/div

X_{out} = 0.0938 in/div

t = 5 div/sec

Figure 57. Failure Transients - Condition 26C

failure transients are waveform distortions during the 200 millisecond detection and transfer time. The amplitude of the failure transients is 2.8% of the maximum output stroke and consists of the failure of the output to track the sinusoidal input during the 200 millisecond detection and transfer period.

Figure 58 shows the effect of sequentially applying a positive hardover input signal of +9 volts to the inputs of channels 1, 2 and 3 with load B applied to the system. The failure transient amplitude for the first failure into channel 1 is 1.1% of the maximum actuator stroke. The transient duration is the 200 milliseconds. The same effect occurs for the second hardover input failure into channel 2. After the second failure and the bypassing of section 1 of the actuator, the actuator output position exhibits a small amplitude hunting. Upon application of the third input failure, the actuator output follows the input up to a position change of 4.4% of the maximum actuator stroke. At that point, the actuator is bypassed and is driven to the null force position of the load actuator. The results are similar to the test results for the unloaded actuator (reference Figure 30).

Figure 59 shows the effect of sequentially applying a positive hardover input signal of +9 volts to the inputs of channels 1, 2 and 3 with load C applied to the system. Note that the amplitude scale on this figure is 1/4 that of Figure 58 in order to show the final position of the test actuator after the third failure. The output transient resulting from the channel 1 input failure is not detectable on Figure 59. The output trans ent for the second input failure consists of a position shift of 1% of the m ximum actuator position to a new position. The third input failure causes the test actuator to bypass, and allows the load actuator to drive it to a position where no load force is applied.

Figure 60 shows the effect of sequentially applying a negative hardover input signal of -9 volts to the inputs of channels 1, 2 and 3 with load B applied to the test actuator. The amplitude of the failure transients for the first and second input failures are lower than those resulting from the +9 volts hardover input failures. This is probably a reflection of the section 1 and 2 force fight and the input necessary to have the two sections move together.

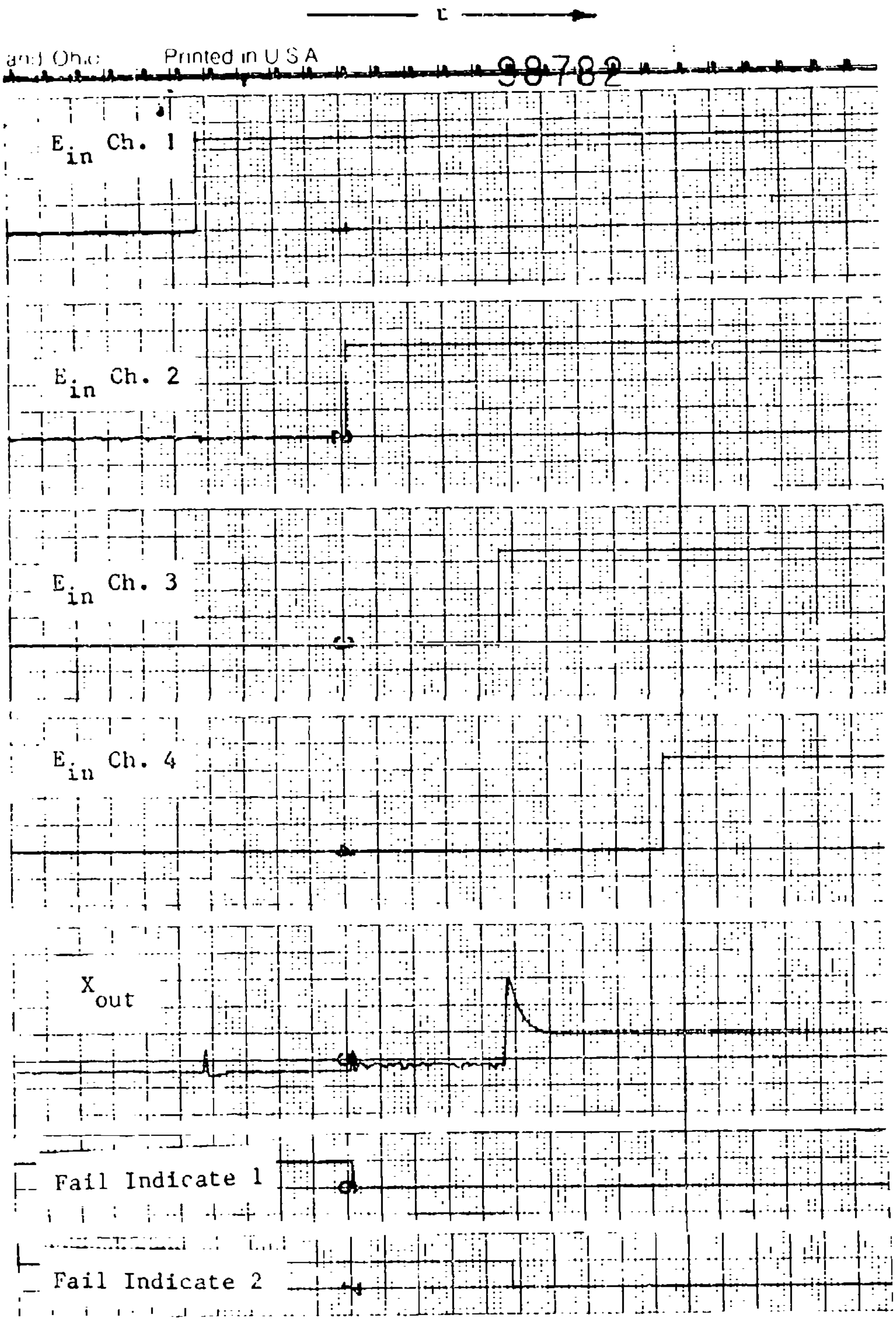
Figure 61 shows the effect of sequentially applying a negative hardover input signal of -9 volts to the inputs of channels 1, 2 and 3 with load C applied. The transient output deviations are limited to 1% of the maximum actuator stroke. These deviations are greater than that measured with positive hardover input failures under the same load conditions (reference Figure 59).

Figure 62 shows the effect of applying an extend hardover input of +9 volts sequentially to inputs 1, 2 and 3 with the system operating with a 1.5 Hz input signal and under load B. Figure 63 has the same input and failure sequence with the load C applied. The load in both figures has little effect on the output motion of the test system. The test system fails to track the 1.5 Hz input signal for about 200 milliseconds, resulting in a position error of 1% or less.

Figure 64 shows the effect of applying a retract hardover input signal of -9 volts sequentially to inputs 1, 2 and 3 with the system operating with a 1.5 Hz input at just below rate saturation. For this test, load B was applied to the test actuator. Figure 65 shows the same input test with load C applied to the

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 27B   Date Prepared 10/3/83



SCALE:

E_{in} = 0.500 v/div

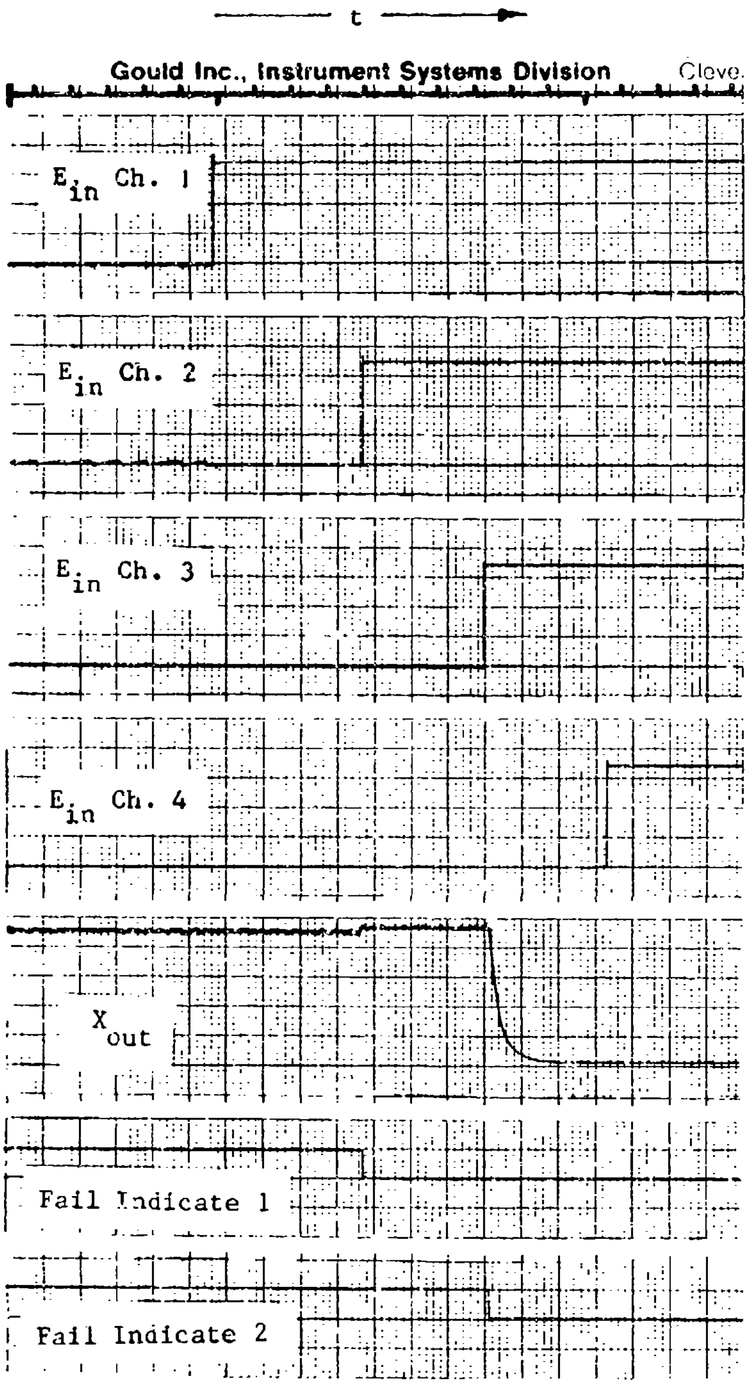
X_{out} = 0.00938 in/div

t = 5 div/sec

Figure 58. Failure Transients - Condition 27B

TEST ITEM - Boeing Reconfigurable Fail Operative
Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 27C   Date Prepared 10/3/83



SCALE:

E_{in} = 0.500 v/div

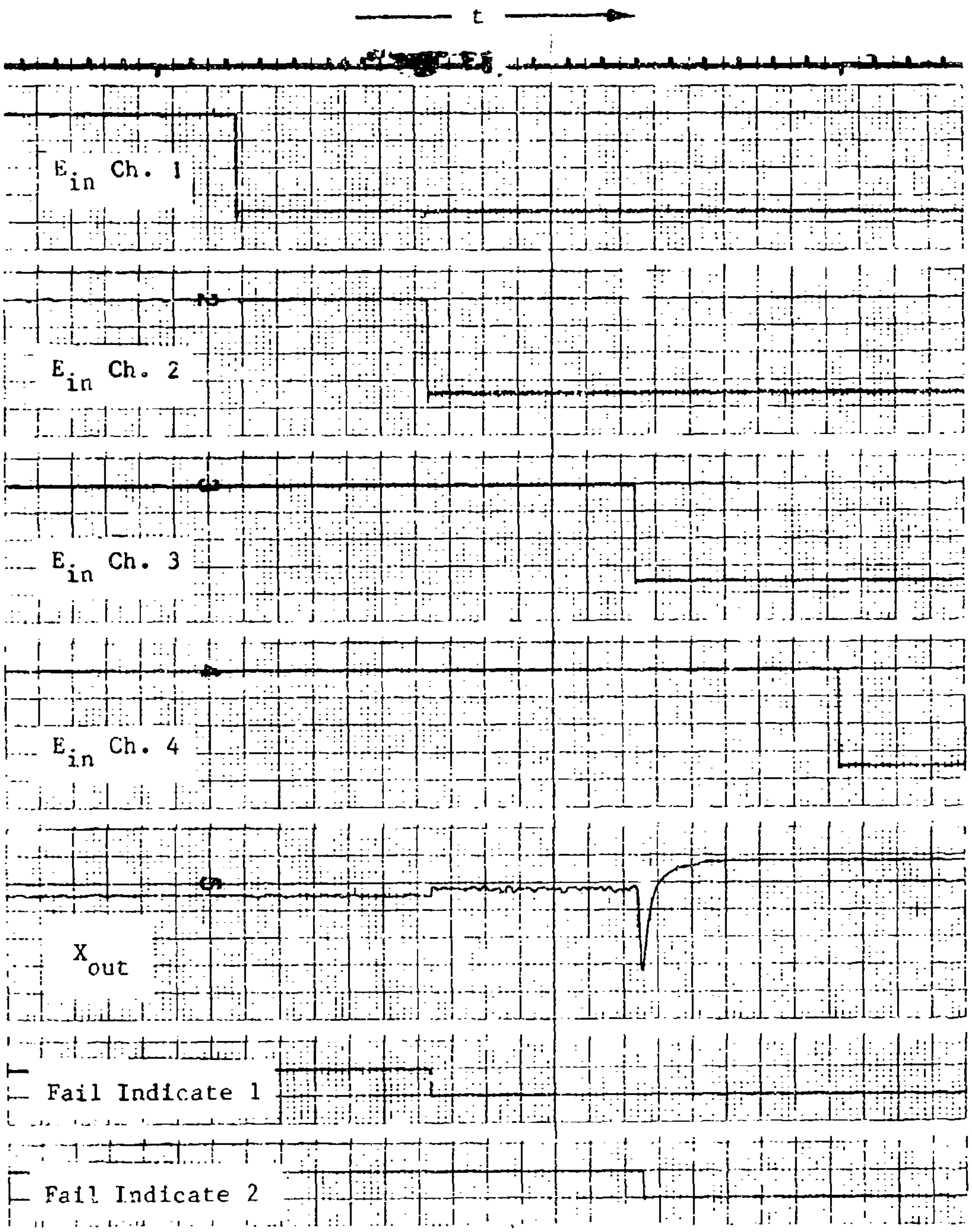
X_{out} = 0.0375 in/div

t = 5 div/sec

Figure 59. Failure Transients - Condition 27C

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 28B   Date Prepared 10/3/83



SCALE: E_{in} = 0.500 v/div

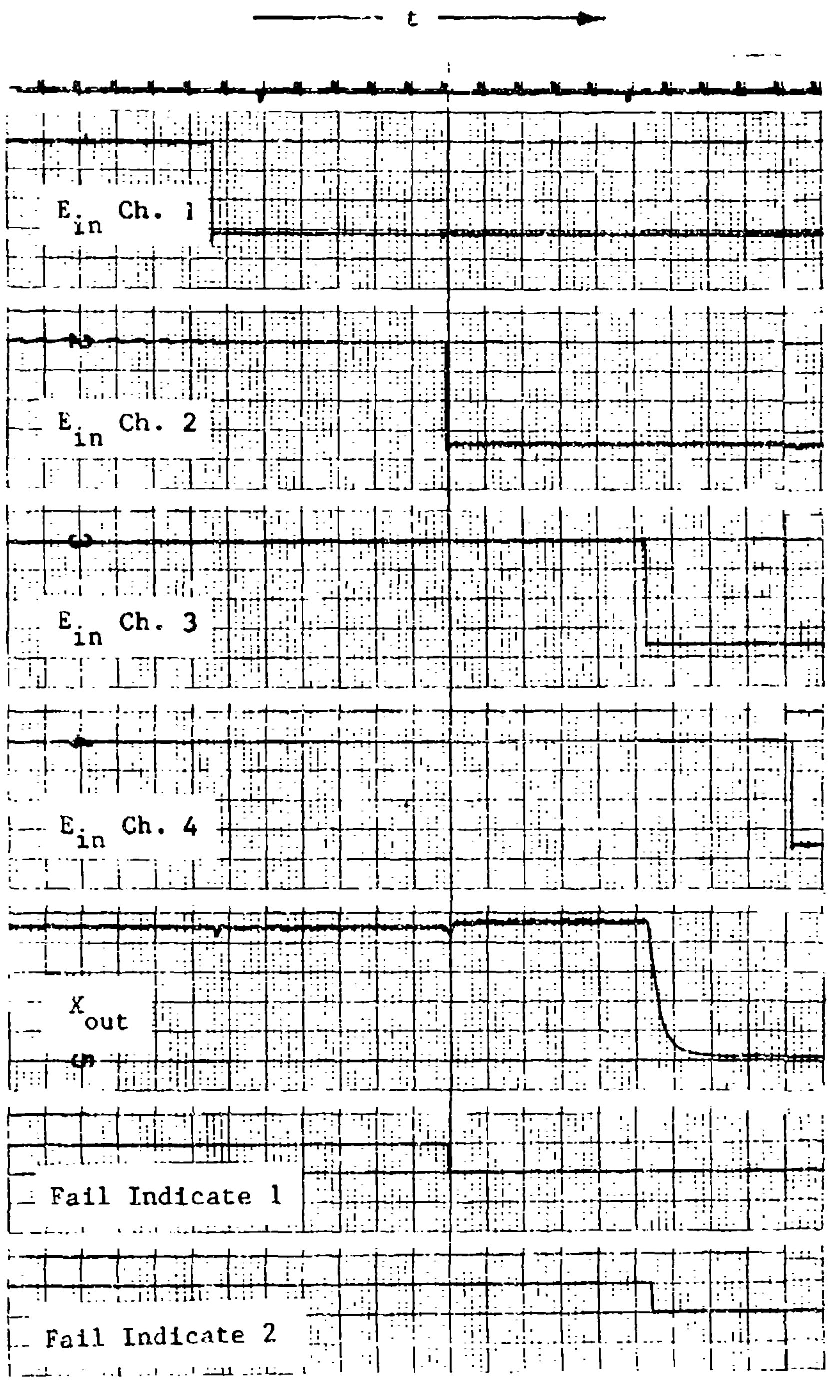
X_{out} = 0.00938 in/div

t = 5 div/sec

Figure 60. Failure Transients - Condition 28B

TEST ITEM - Boeing Reconfigurable Fail Operative
Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 28C   Date Prepared 10/3/83



SCALE:

E_{in} = 0.500 v/div

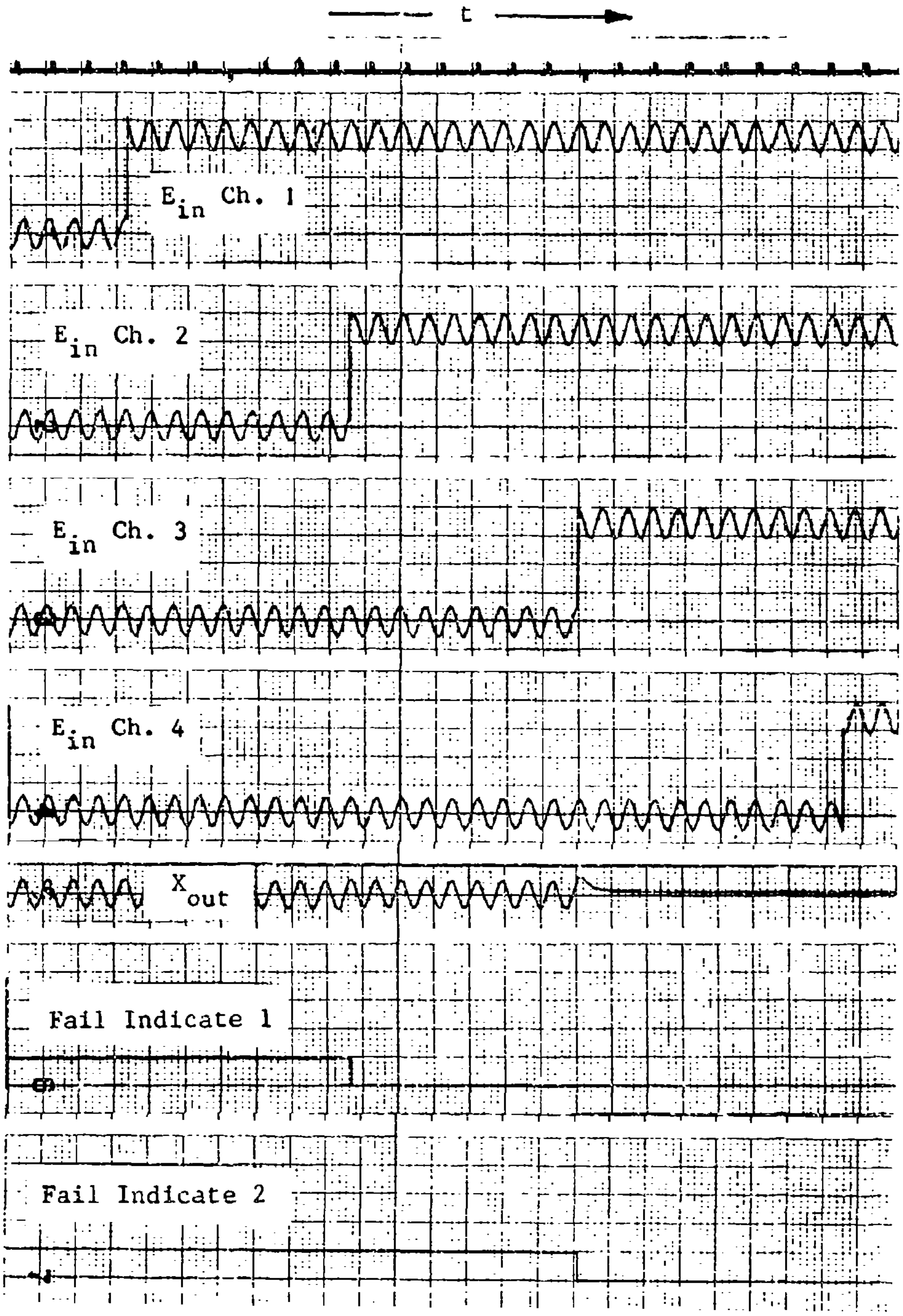
X_{out} = 0.0375 in/div

t = 5 div/sec

Figure 61. Failure Transients - Condition 28C

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 29B Date Prepared 10/3/83



SCALE:

E_{in} = 0.500 v/div

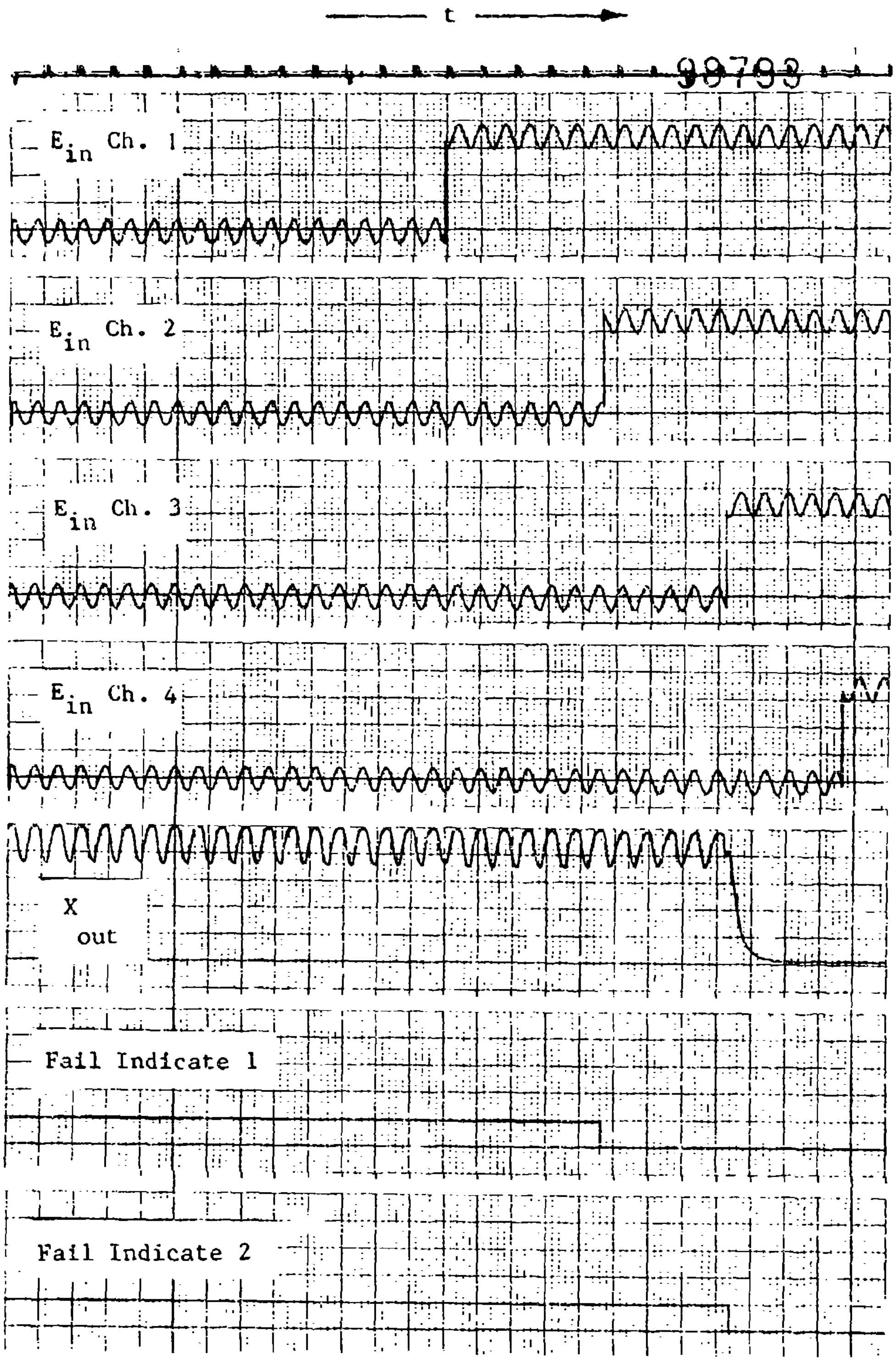
X_{out} = 0.0938 in/div

t = 5 div/sec

Figure 62. Failure Transients - Condition 29B

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 29C   Date Prepared 10/3/83



SCALE:

E_{in} = 0.500 v/div

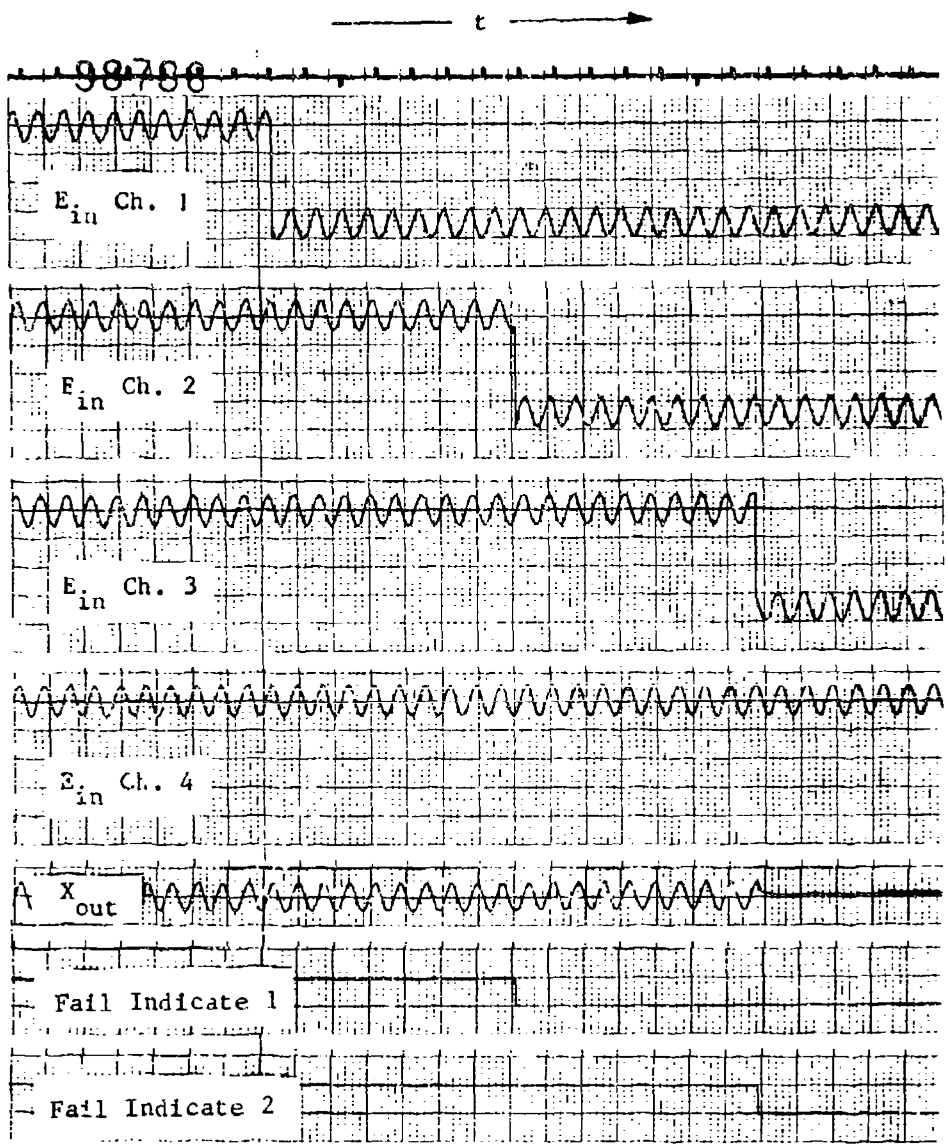
X_{out} = 0.0375 in/div

t = 5 div/sec

Figure 63. Failure Transients - Condition 29C

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 30B   Date Prepared 10/3/83



SCALE:

E_{in} = 0.500 v/div

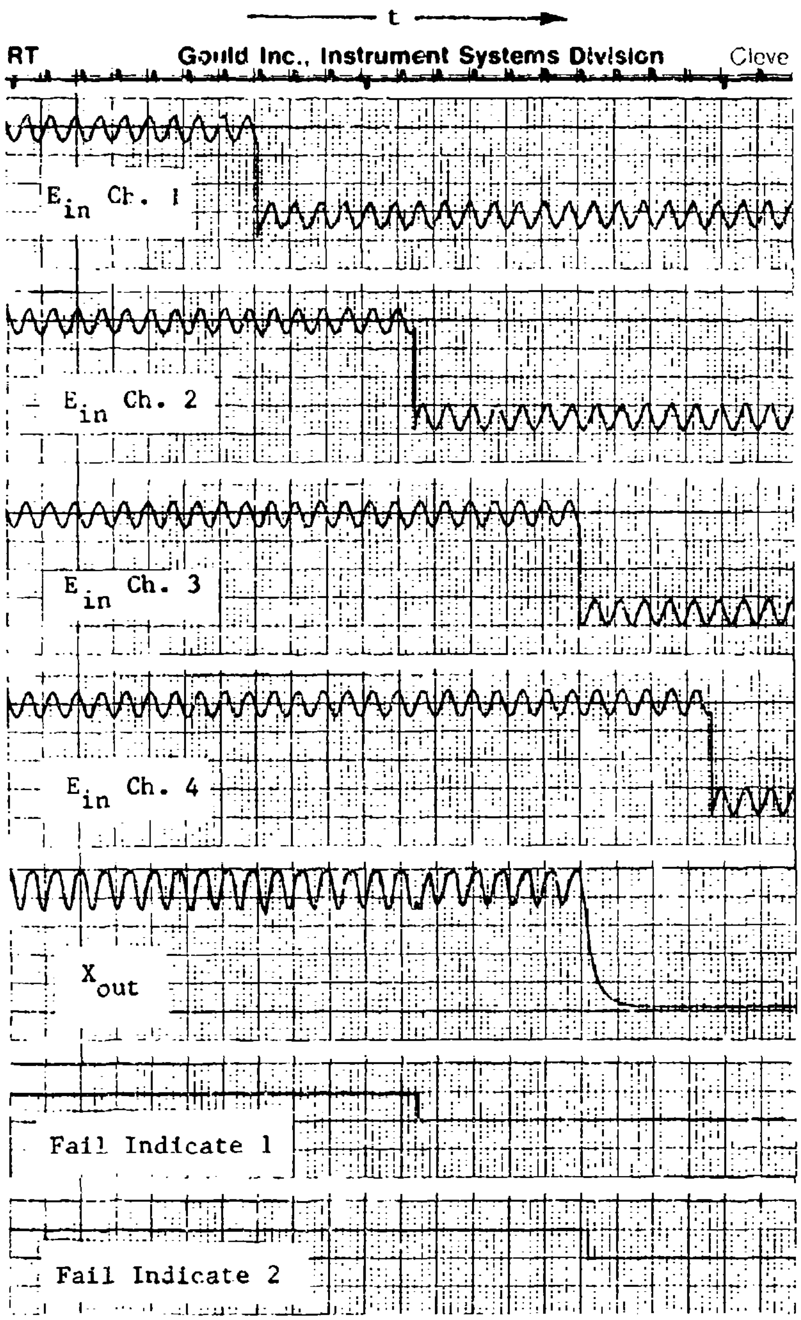
X_{out} = 0.0938 in/div

t = 5 div/sec

Figure 64. Failure Transients - Condition 30B

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Failure Transients - Condition 30C   Date Prepared 10/3/83



SCALE:

E_{in} = 0.500 v/div

X_{out} = 0.0375 in/div

t = 5 div/sec

Figure 65. Failure Transients - Condition 30C

test actuator. The results are similar to those obtained with the extend hardover input (reference Figures 62 and 63). The output tracks the sinusoidal input until the third input failure is applied. The primary effect of the failed inputs is a failure of the output to track the input for the 200 milliseconds required for failure detection and correction.

From the analysis of Figures 56 through 65, it appears that the effect of load conditions B and C on the failure transients is not significant. The deviations are greatest with load C. The greatest amplitude deviation is less than 3% of the maximum output stroke and occurs with load C.

DISTORTION (OUTPUT/INPUT FIDELITY) TEST RESULTS

General

The waveform recordings in this data section are representative of the fidelity characteristics of the test system under different operating conditions. This type of test is useful in indicating whether the output of the system has significant distortion components. In some applications, the frequency content of the distortion can create problems in stablity and structural life by exciting resonant modes of the mechanism driven by the actuator.

Output Fidelity vs Input Level

Figures 66, 67 and 68 show the output/input characteristics of the test system operating unloaded at 1.5 Hz. The test system is configured normally with channels 1 and 3 active, and 2 and 4 as models. These figures show the effect of input level changes on the output at the 1/2 bandpass frequency. Figure 66 displays the 0.5% and 1% input response. Note that at the 0.5% input level, the output of the actuator shows no response. This is an indication of dynamic threshold and is a level which is greater than the 0.28% measured earlier in the testing (reference Table 3). However, the 0.5% input is lower than the 0.77% dynamic threshold measured under load condition B (reference Table 14). Note that with the 1% input, the output is nominally sinusoidal at an amplitude which is approximately 1% of the total stroke. At the 1% output, there is both noticeable "flat topping" of the sinusoidal motion and amplitude modulation of the waveform. The noise content of the output at 1% is quite noticeable on Figure 66. On Figure 67 with 2% and 5% input data, the noise content is much less noticeable. The 5% output shows less flat topping than the 2% input data and no apparent amplitude motion. At a 10% input level, as shown on Figure 68, the output closely resembles the input with no amplitude modulation and flat topping. Note that the phase lag between the output and input remains the same for all the input levels.

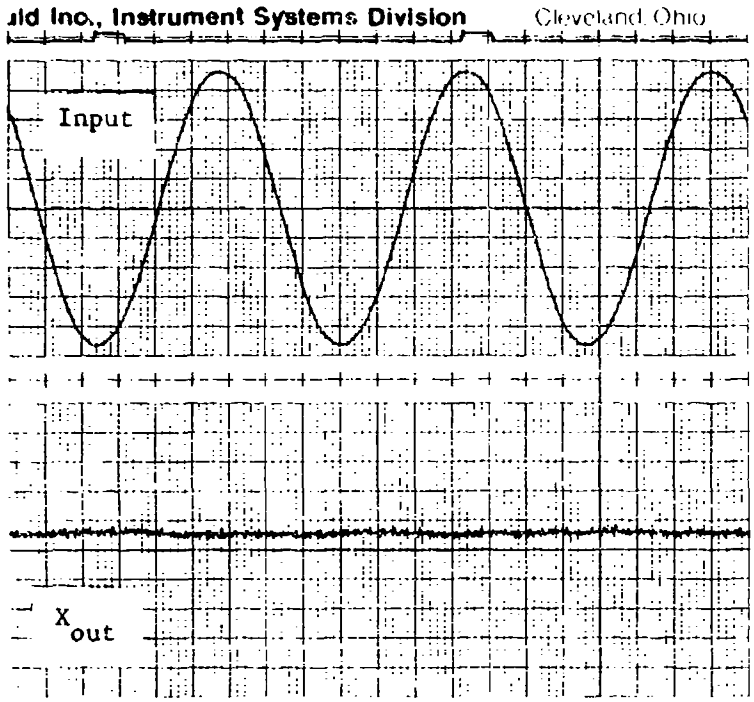
Output Fidelity vs Offset Bias - No Load - 10% Input

Figures 69 through 74 illustrate the effect of channel bias differences equivalent to 30% of the spool stroke on large amplitude (10% of the maximum stroke) output motion. Figures 69, 71 and 73 represent data with 0 bias applied. Figures 70, 72 and 74 represent data with a 30% bias applied. The input frequencies are the same for the figure pairs of 69 and 70, 71 and 72, 73

Date Prepared 10/20/83

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Input Level - Normal System @ 1/2 Bandpass Frequency



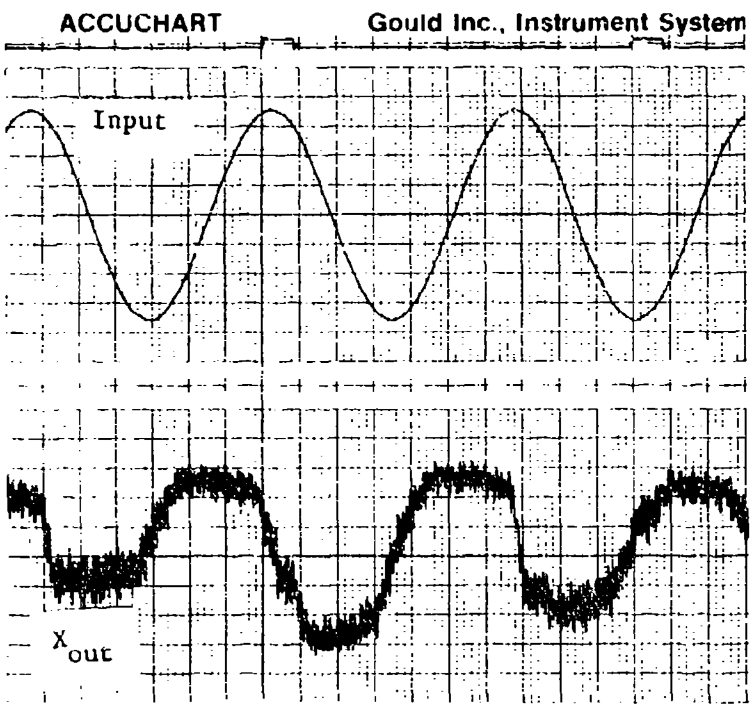
Free Air - .5%

Scale:

Input - 0.002 v/div

X_{out} - 0.00093 in/div

———— t ————→



Free Air - 1%

Scale:

Input - 0.005 v/div

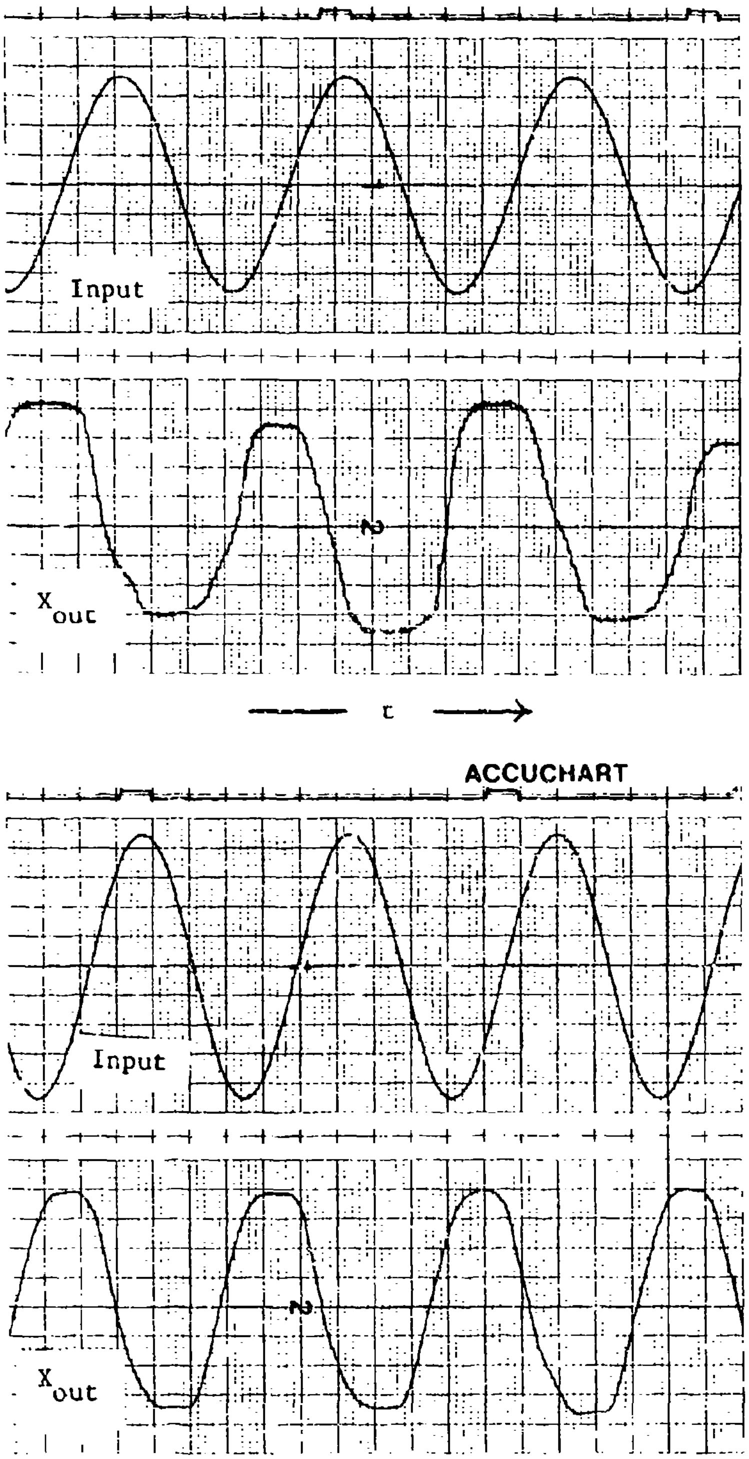
X_{out} - 0.00093 v/div

Figure 66. Output Fidelity at

Date Prepared 10/20/83

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Input Level - Normal System @ 1/2 Bandpass Frequency



Free Air - 2%

Scale:

Input - 0.010 v/div

X_{out} - 0.00186 in/div

Free Air - 5%

Scale:

Input - 0.020 v/div

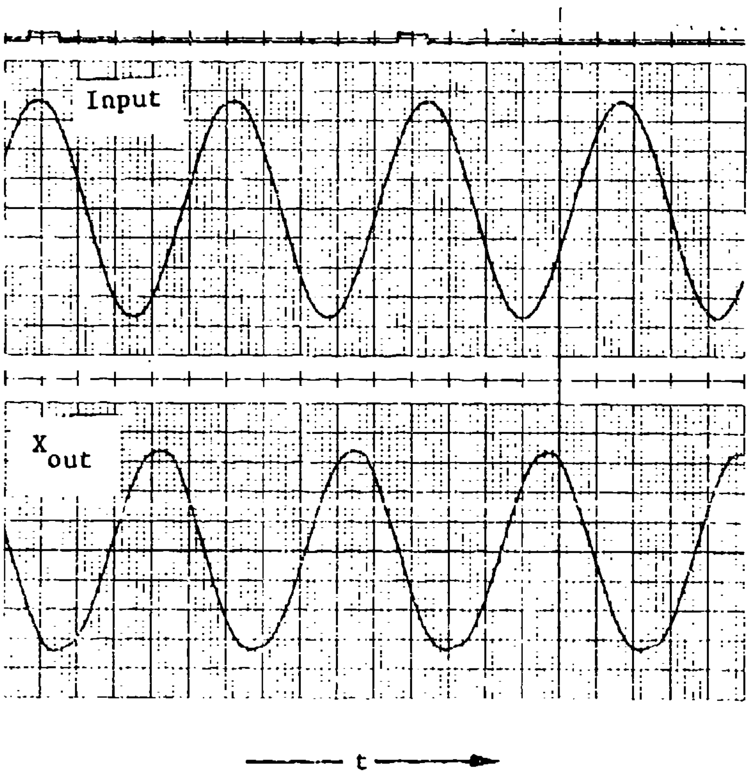
X_{out} - 0.00465 in/div

Figure 67. Output Fidelity at 2% and 5% Input

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Input Level - Normal System @ 1/2 Bandpass Frequency

Date Prepared 10/20/83



Free Air - 10%

Scale:

Input - 0.050 v/div

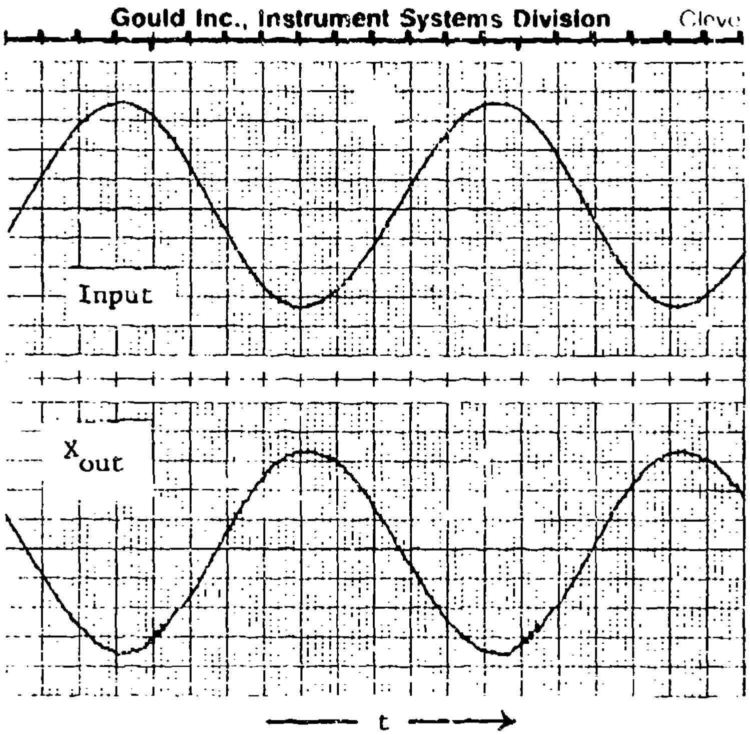
X_{out} - 0.0093 v/div

Figure 68. Output Fidelity at 10% Input

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - No Load - 10% Input

Date Prepared 10/18/83

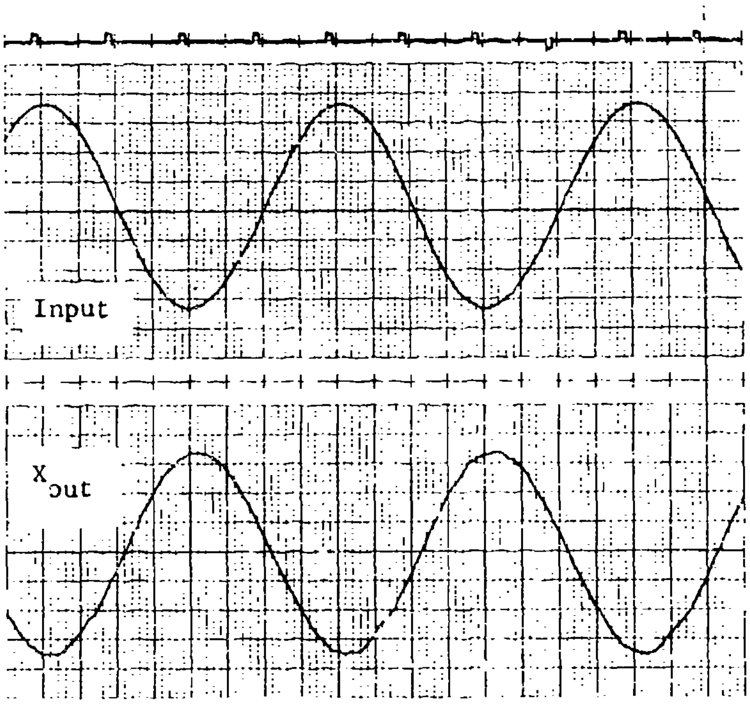


Input Level: 10%
Input Freq.: 0.1 Hz
Load Cond. : A
Bias Cond. : 0%

Scale:

Input - 0.05 v/div

X_{out} - 0.0093 in/div



Input Level: 10%
Input Freq.: 0.25 Hz
Load Cond. : A
Bias Cond. : 0%

Scale:

Input - 0.05 v/div

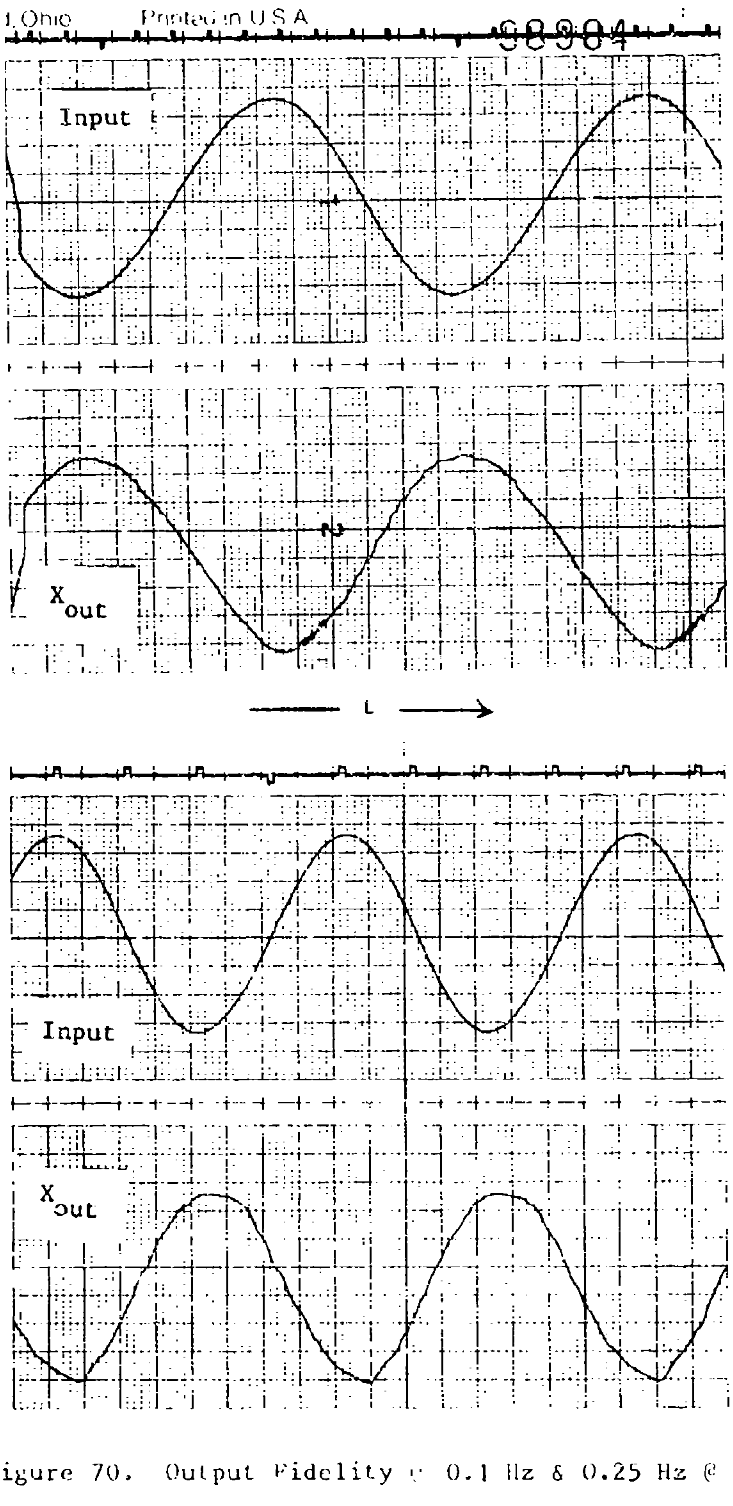
X_{out} - 0.0093 in/div

Figure 69. Output Fidelity @ 0.1 Hz & 0.25 Hz @ 10% Input - 0% Bias

TEST TIEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - No Load - 10% Input

Date Prepared 10/18/83



Input Level: 10%
Input Freq.: 0.1 Hz
Load Cond. : A
Bias Cond. : 30%

Scale:

Input - 0.05 v/div

X_{out} - 0.0093 in/div

Input Level: 10%
Input Freq.: 0.25 Hz
Load Cond. : A
Bias Cond. : 30%

Scale:

Input - 0.05 v/div

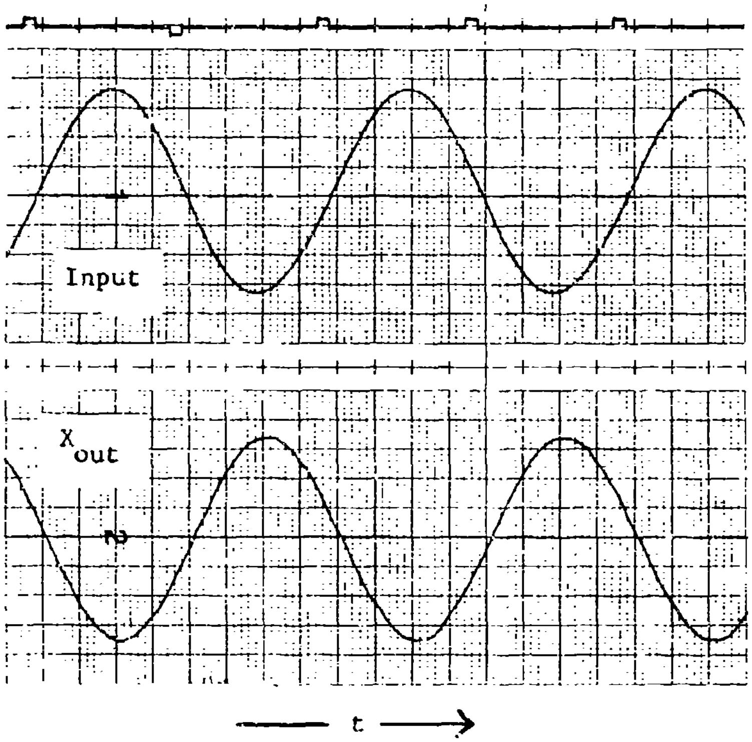
X_{out} - 0.0093 in/div

Figure 70. Output Fidelity @ 0.1 Hz & 0.25 Hz @ 10% Input - 30% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - No Load - 10% Input

Date Prepared 10/18/83

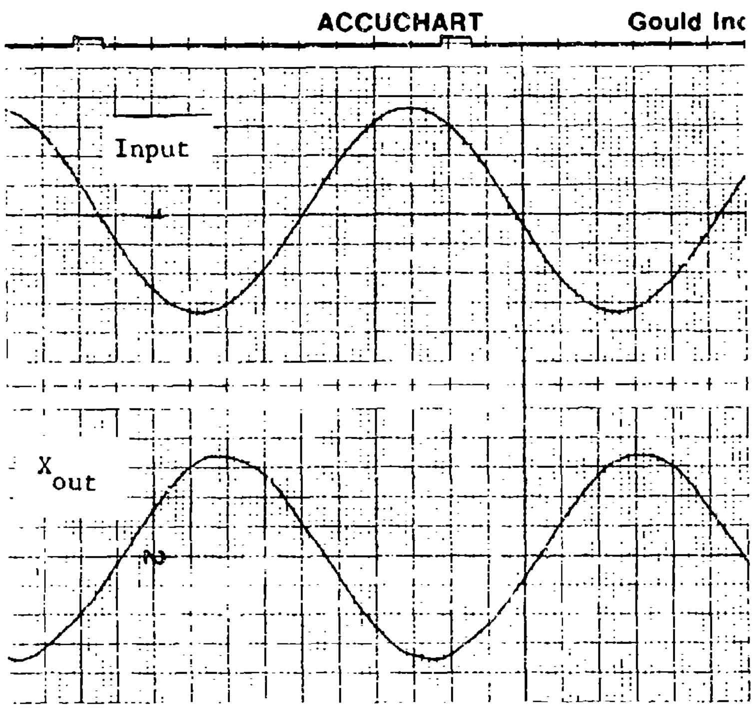


Input Level: 10%
Input Freq.: 0.5 Hz
Load Cond. : A
Bias Cond. : 0%

Scale:

Input - 0.05 v/div

X_{out} - 0.0093 in/div



Input Level: 10%
Input Freq.: 1.0 Hz
Load Cond. : A
Bias Cond. : 0%

Scale:

Input - 0.05 v/div

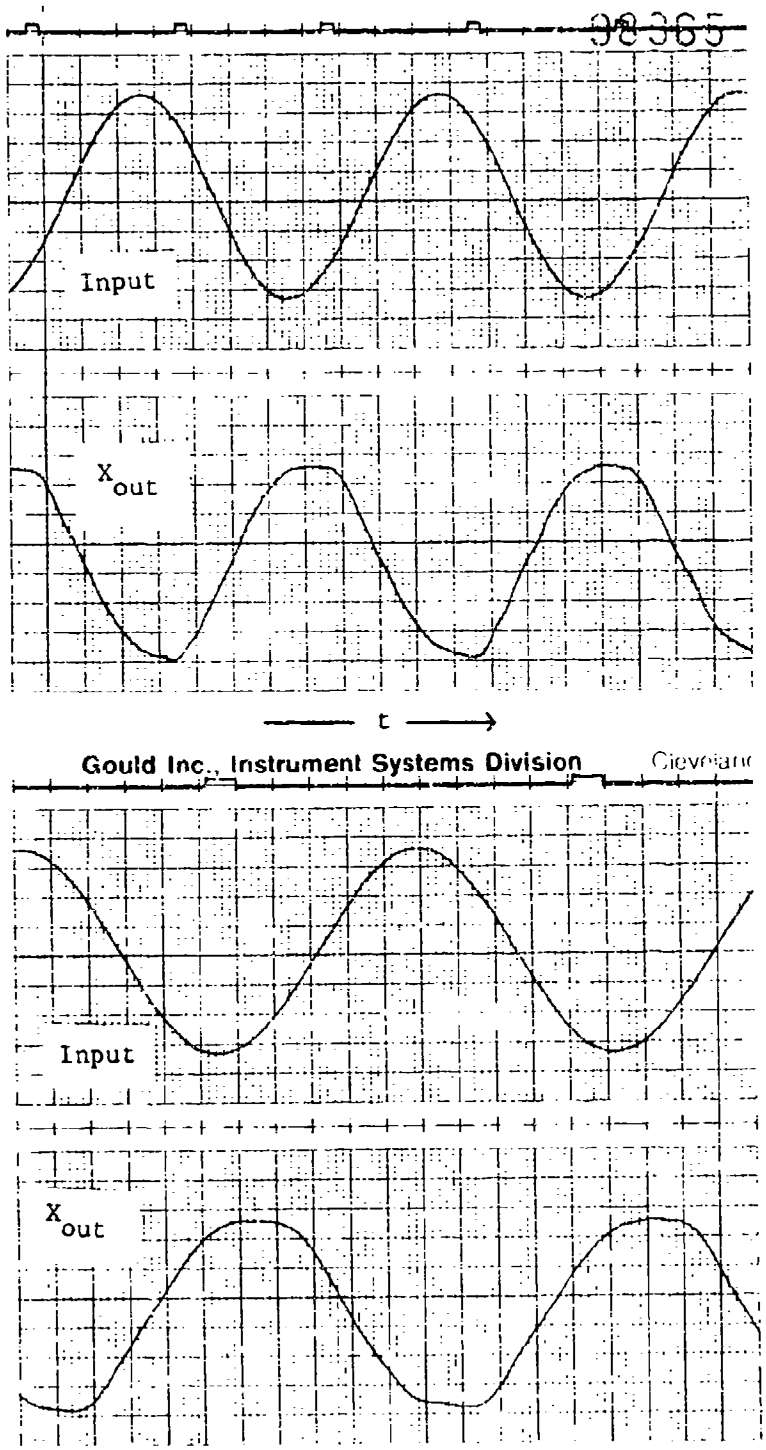
X_{out} - 0.0093 in/div

Figure 71. Output Fidelity @ 0.5 Hz & 1.0 Hz @ 10% Input - 0% Bias

TEST TIEM - Boeing Reconfigurable Fail Oper ive Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - No Load - 10% Input

Date Prepared 10/18/83



Input Level: 10%
Input Freq.: 0.5 Hz
Load Cond. : A
Bias Cond. : 30%

Scale:

Input - 0.05 v/div

X_{out} - 0.0093 in/div

Input Level: 10%
Input Freq.: 1.0 Hz
Load Cond. : A
Bias Cond. : 30%

Scale:

Input - 0.5 v/div

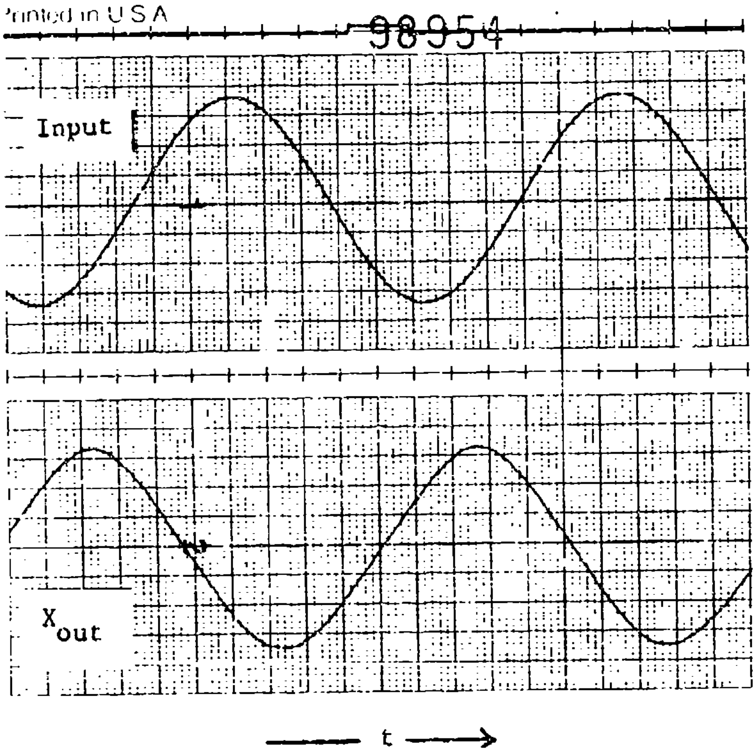
X_{out} - 0.0093 in/div

Figure 72. Output Fidelity @ 0.5 Hz & 1.0 Hz @ 10% Input - 0% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - No Load - 10% Input

Date Prepared 10/18/83

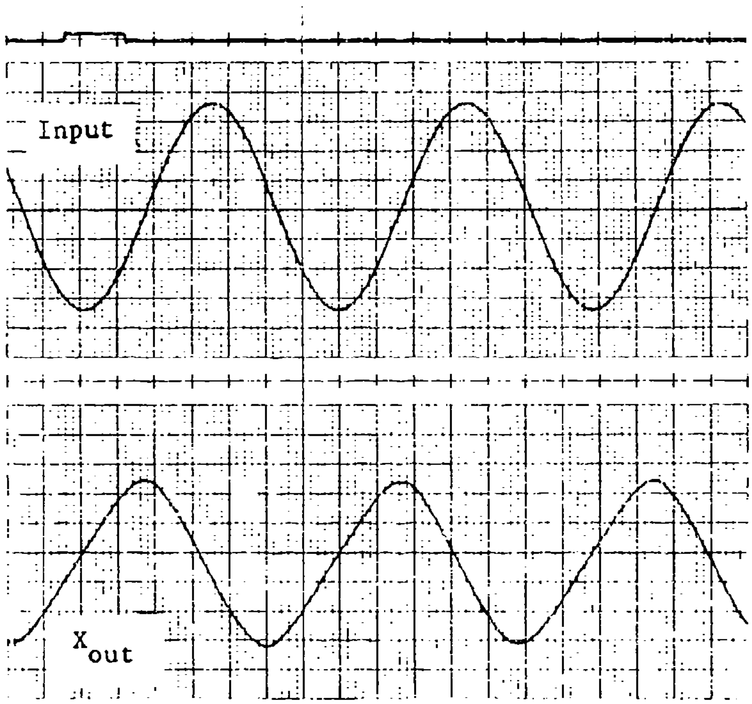


Input Level: 10%
Input Freq.: 2.0 Hz
Load Cond. : A
Bias Cond. : 0%

Scale:

Input - 0.05 v/div

X_{out} - 0.0093 in/div



Input Level: 10%
Input Freq.: 3.0 Hz
Load Cond. : A
Bias Cond. : 0%

Scale:

Input - 0.05 v/div

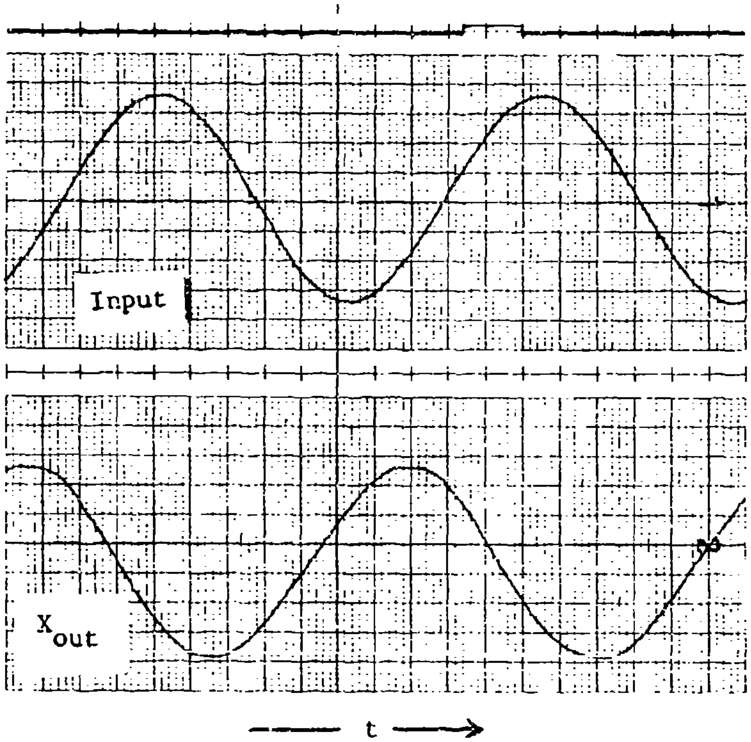
X_{out} - 0.0093 in/div

Figure 73. Output Fidelity @ 2 Hz & 3 Hz @ 10% Input - 0% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - No Load - 10% Input

Date Prepared 10/18/83

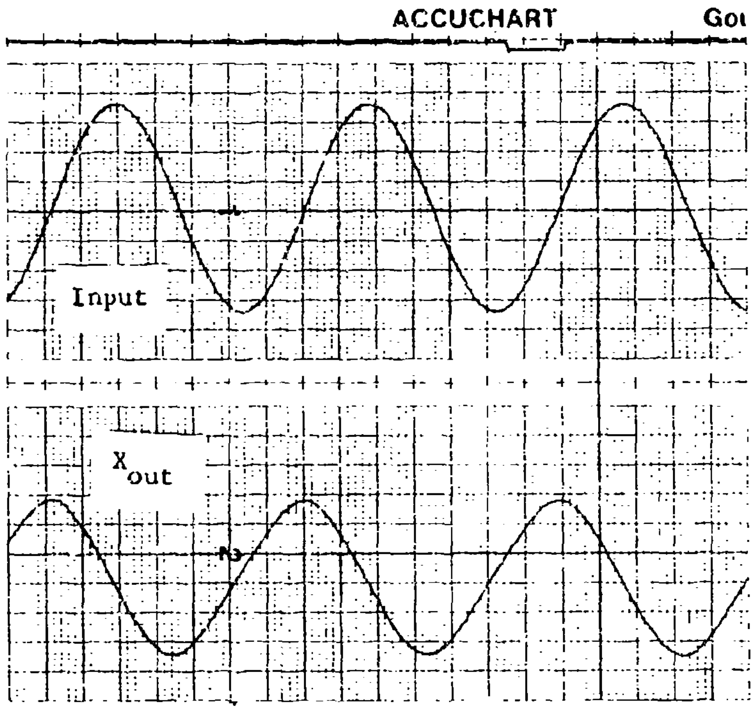


Input Level: 10%
Input Freq.: 2.0 Hz
Load Cond. : A
Bias Cond. : 30%

Scale:

Input - 0.05 v/div

X_{out} - 0.0093 in/div



Input Level: 10%
Input Freq.: 3.0 Hz
Load Cond. : A
Bias Cond. : 30%

Scale:

Input - 0.05 v/div

X_{out} - 0.0093 in/div

Figure 74. Output Fidelity @ 2 Hz & 3 Hz @ 10% Input - 30% Bias

and 74. The purpose of this data presentation format is to allow easy visual comparison of "0" bias data with "30%" bias data at the same frequencies.

As shown on Figure 69, the output closely resembles the input (with the 180° phase shift due to the polarity of the feedback transducer used to measure the output motion). Note that at .1 Hz and .25 Hz, the output tracks the input with negligibile phase shift (from a 180° phase angle). There is some minor distortion of the output visible on the output at both frequencies. The effect of the 30% bias as shown on Figure 70 is to increase the distortion of the sinusoid slightly over that shown on Figure 69. The distortion increase is in the form of a slope flattening of the motion just after the maximum amplitude peak in the retract direction.

Figures 71 and 72 illustrate the effect input bias has on the output waveform at 0.5 and 1.0 Hz. The output waveform as shown on Figure 71 with 0 bias closely resembles the input with no apparent distortion. However, the output waveform with a 30% bias at the same frequencies shows distinct "flat topping". This is caused by the force fight between channels showing up during the portion of the motion where the actuator is reversing direction.

Figures 73 and 74 illustrate the effect of input bias on the output waveform at 2.0 and 3.0 Hz. A comparison of these two figures shows no apparent difference in the output waveform. The output shows the effect of slight rate saturation at 3 Hz, indicating that the control valves are stroking full deflection.

From these test results, it appears that channel bias conditions have very little effect on the output waveform at frequencies above 1/2 the bandpass frequency. There is a minor distortion increase with channel bias mismatch for frequencies below the 1/2 bandpass frequency. This is consistent with increasing input frequency requiring increase flow and spool deflection. At low frequencies, the bias offset is a larger percentage of the spool motion and has greater effect than at higher frequencies.

<u>Output Fidelity vs Offset Bias - Symetrical Load B - 10% Input</u>

Figures 75 through 78 illustrate the output waveform of the test system at 10% inputs and frequencies of 0.1, 1.0 and 3 Hz. The load applied is the symmetrical load B. The figures are grouped so the effect of bias at the same frequencies can be easily made. The data displayed is limited to 3 frequencies since the symmetrical load B test results are not significantly different than unloaded test conditions (load A) for the 10% input level.

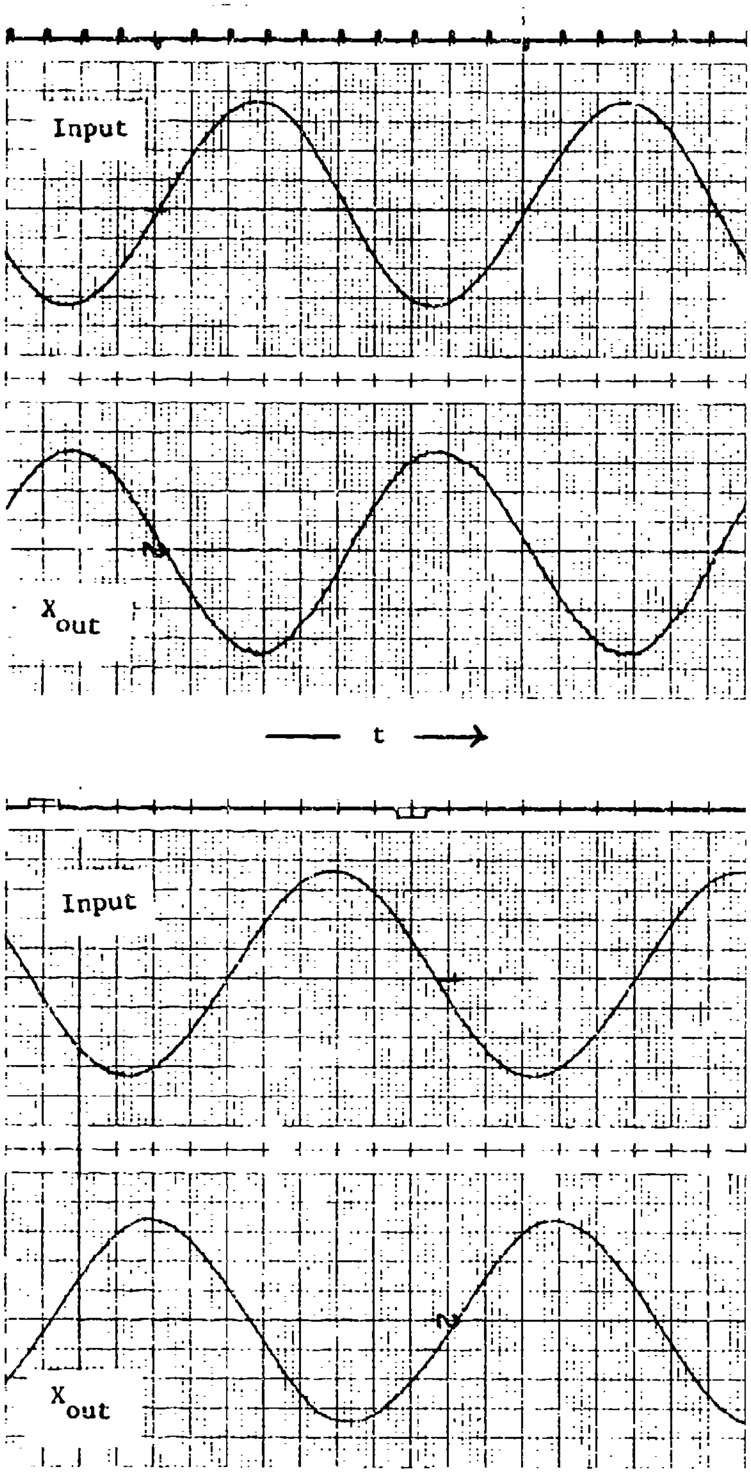
Comparing Figure 76 with 30% bias to Figure 75 with 0% bias shows that the bias creates minor output distortion in the form of flattening at or after the peak amplitude excursion. The effect is apparent at both 0.1 and 1 Hz input frequencies. The effect is only slightly more than that perviously observed with load A (unloaded) at the same frequencies (reference Figures 69 through 72).

Figures 77 and 78 at 3 Hz show no difference in the output waveform for the two bias conditions. The output exhibits a slight rate saturation for both figures. This result is similar to that of the unloaded tests at the same frequency and input level.

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Symetrical Load - 10% Input

Date Prepared 10/19/83



Input Level: 10%
Input Freq.: 0.1 Hz
Load Cond. : B
Bias Cond. : 0%

Scale:

Input - 0.05 v/div

X_{out} - 0.0093 in/div

Input Level: 10%
Input Freq.: 1.0 Hz
Load Cond. : B
Bias Cond. : 0%

Scale:

Input - 0.05 v/div

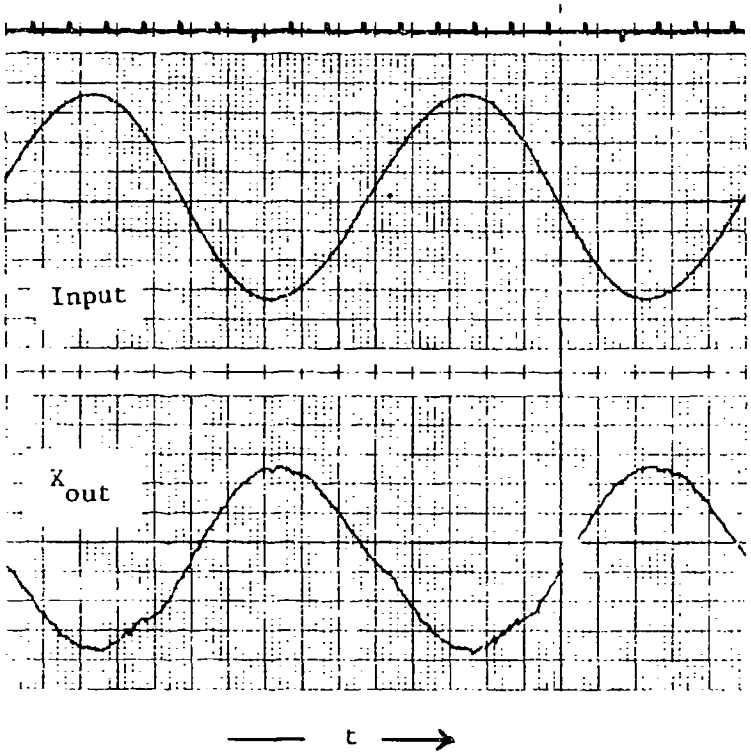
X_{out} - 0.0093 in/div

Figure 75. Output Fidelity @ 0.1 Hz & 1 Hz @ 10% Input - Load B - 0% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative
Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias -
Symetrical Load - 10% Input

Date Prepared 10/19/83

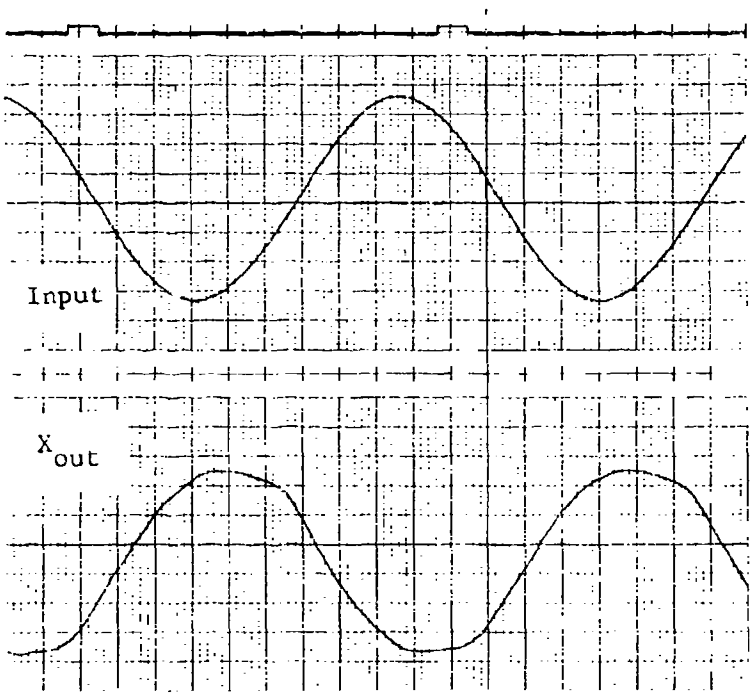


Input Level: 10%
Input Freq.: 0.1 Hz
Load Cond. : B
Bias Cond. : 30%

Scale:

Input - 0.05 v/div

X_{out} - 0.0093 in/div



Input Level: 10%
Input Freq.: 1.0 Hz
Load Cond. : B
Bias Cond. : 30%

Scale:

Input - 0.05 v/div

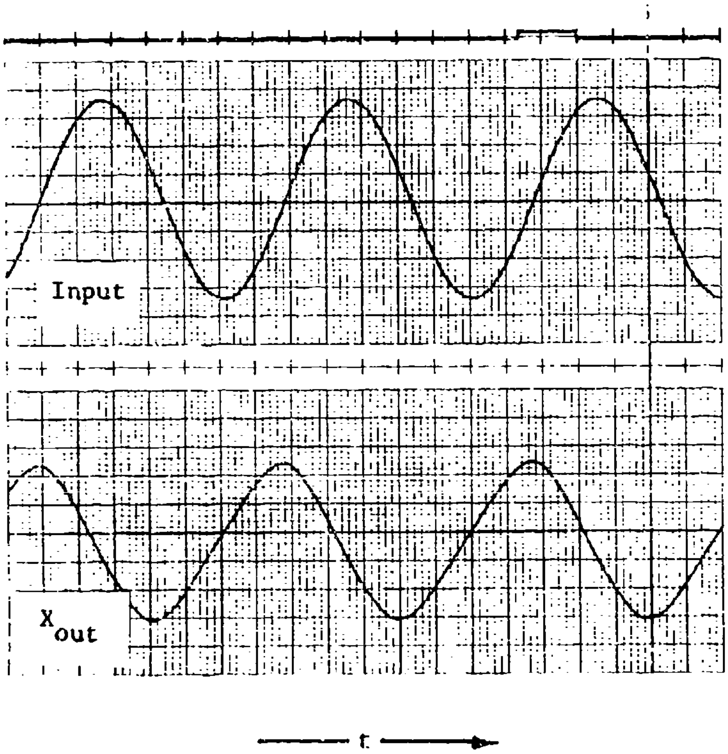
X_{out} - 0.0093 in/div

Figure 76. Output Fidelity @ 0.1 Hz & 1 Hz @ 10% Input - Load B - 30% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Symmetrical Load - 10% Input

Date Prepared 10/19/83



Input Level: 10%
Input Freq.: 3.0 Hz
Load Cond. : B
Bias Cond. : 0%

Scale:

Input - 0.05 v/div

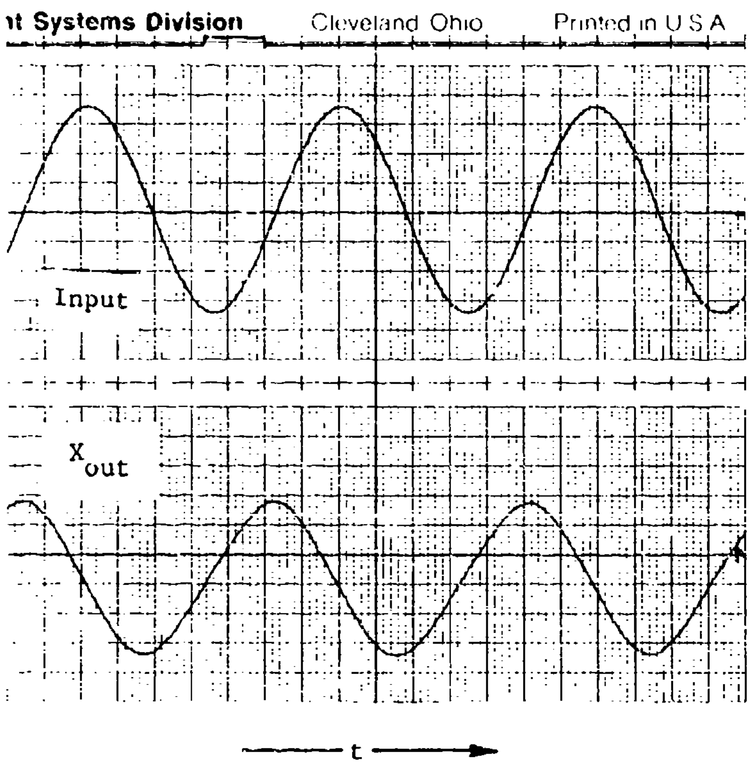
X_{out} - 0.0093 in/div

Figure 77. Output Fidelity @ 3 Hz @ 10% Input - Load B - 0% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Symetrical Load - 10% Input

Date Prepared 10/19/83



Input Level: 10%
Input Freq.: 3.0 Hz
Load Cond. : B
Bias Cond. : 30%

Scale:

Input - 0.05 v/div

X_{out} - 0.0093 in/div

Figure 78. Output Fidelity @ 3 Hz @ 10% Input - Load B - 30% Bias

Symmetrical load B does increase the distortion observed for the 10% input. The distortion is greatest for frequencies below 1/2 bandpass frequency.

Output Fidelity vs Offset Bias - Asymmetrical Load C - 10% Input

Comparing the actuator output waveform on Figures 79 and 80 for the input frequencies of 0.1 and 0.25 Hz, there is apparent minor waveform distortion at the amplitude peaks at both frequencies. The effect of the 30% bias conditions of Figure 80 is to increase the distortion compared to the 0% bias of Figure 79. The amplitude of the distortion is slightly greater than that for the unloaded or load B test data.

Figures 81 through 84 illustrate the effect of load C on the output distortion of the test system with 0% and 30% input bias conditions. Note that the 30% bias is below the failure detection level used for evaluating the test system.

Figures 81 and 82 show the output motion at frequencies of 0.5 and 1 Hz. For the 0% bias condition shown on Figure 81, the output at both frequencies shows very little amplitude distortion. However, Figure 82 with the input bias equivalent to 30% of the maximum spool stroke into one active channel, shows noticeable distortion.

Figures 83 and 84 for input frequencies of 2 and 3 Hz show little distortion for either bias condition. As with the previous 10% input command data, the 3 Hz output shows some rate saturation. With this asymmetrical load condition, the rate saturation (as shown by straightening of the sinusoid into a triangular waveform) occurs primarily in one direction of motion. This is consistent with the bias force load associated with load condition C.

From Figures 79 through 84, it is apparent that load C has greater effect on the 10% output motion than load B or unloaded operation. The effect is greater at frequencies below 1/2 bandpass (1.5 Hz).

Output Fidelity vs Offset Bias - Symmetrical Load B - 3% Input

Figures 85 through 88 show the output characteristics as a function of channel bias at two frequencies (0.25 Hz and 1 Hz) and a 3% input level and load B. This input level is a medium amplitude input test signal for an actuation system. (For example, the F-16 actuators specify a frequency response envelope at 2% of maximum input.) Four input bias levels are used for these 3% input levels: 0%, 10%, 20% and 30%. Note that the 30% bias is below the 35% equivalent spool position mismatch where a channel failure is declared.

Comparing Figures 85 through 88 show that for load B and increasing offset bias, there is significant output distortion for bias conditions of 10%, 20% and 30%. The distortion increases with the increase in offset bias. The distortion is primarily irregular slope changes which modify the shape of the sinusoid. The effect is most apparent at the peak amplitudes of the sinusoidal output.

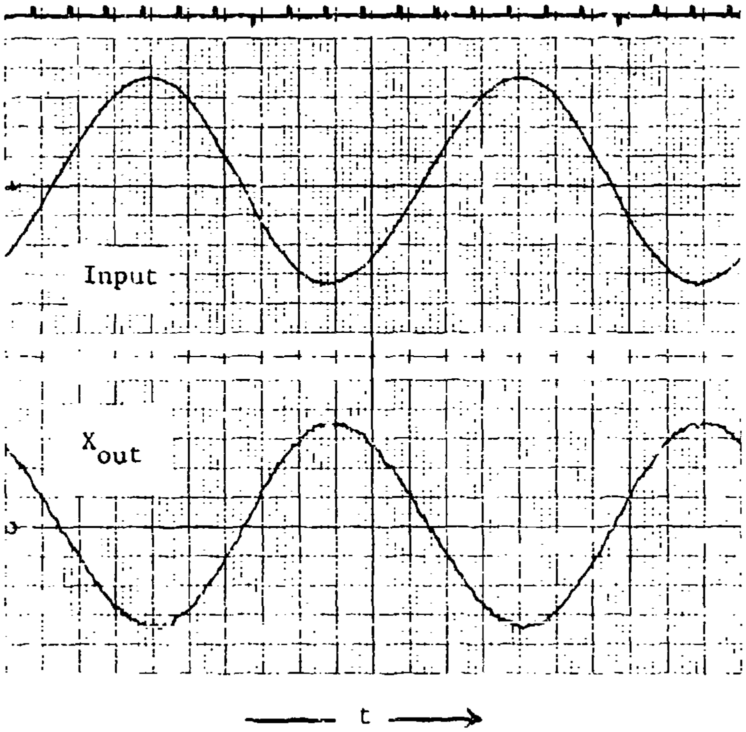
Output Fidelity vs Offset Bias - Asymmetrical Load C - 3% Input

Figures 89 through 92 show the output characteristics as a function of channel

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 10% Input

Date Prepared 10/19/83

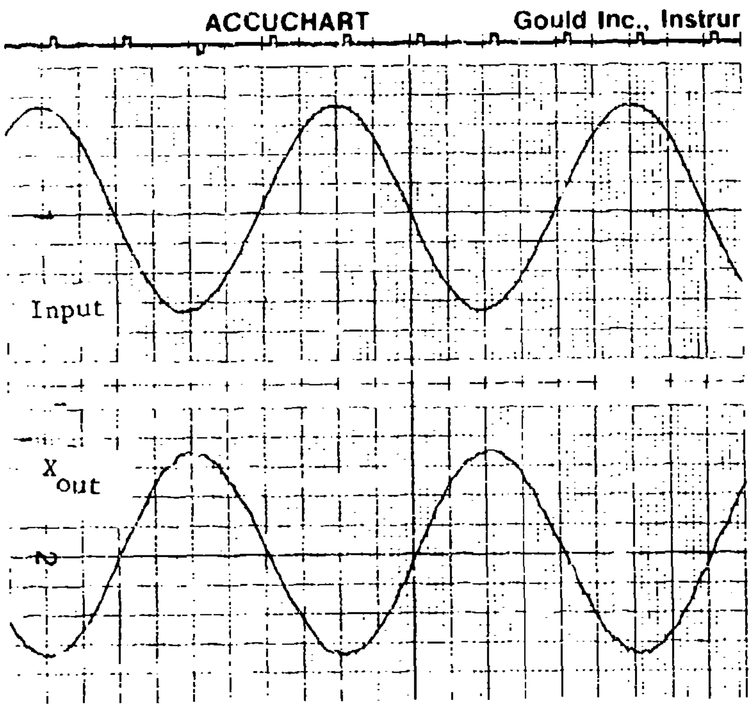


Input Level: 10%
Input Freq.: 0.1 Hz
Load Cond. : C
Bias Cond. : 0%

Scale:

Input - 0.05 v/div

X_{out} - 0.0093 in/div



Input Level: 10%
Input Freq.: 0.25 Hz
Load Cond. : C
Bias Cond. : 0%

Scale:

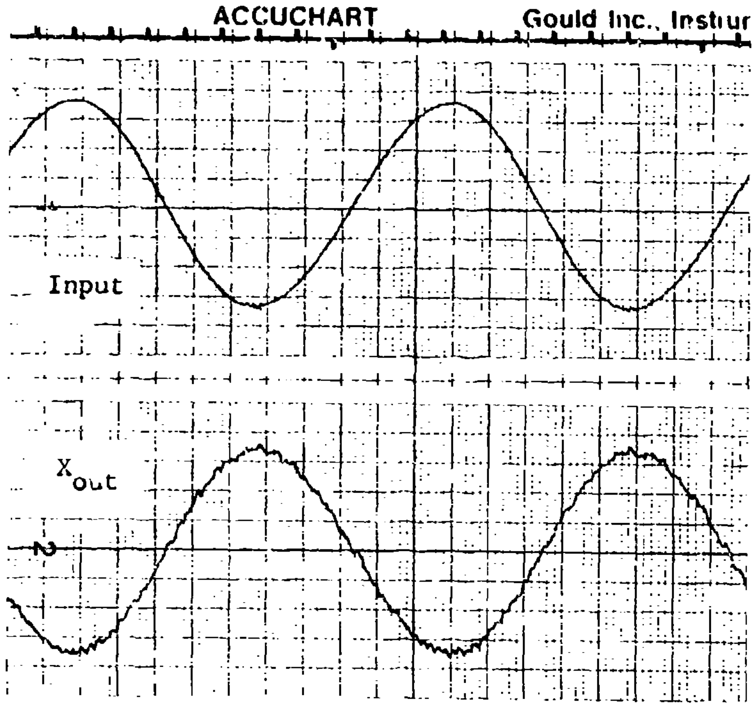
Input - 0.05 v/div

X_{out} - 0.0093 in/div

Figure 79. Output Fidelity @ 0.1 Hz & 0.25 Hz @ 10% Input - Load C - 0% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 10% Input



Date Prepared 10/19/83

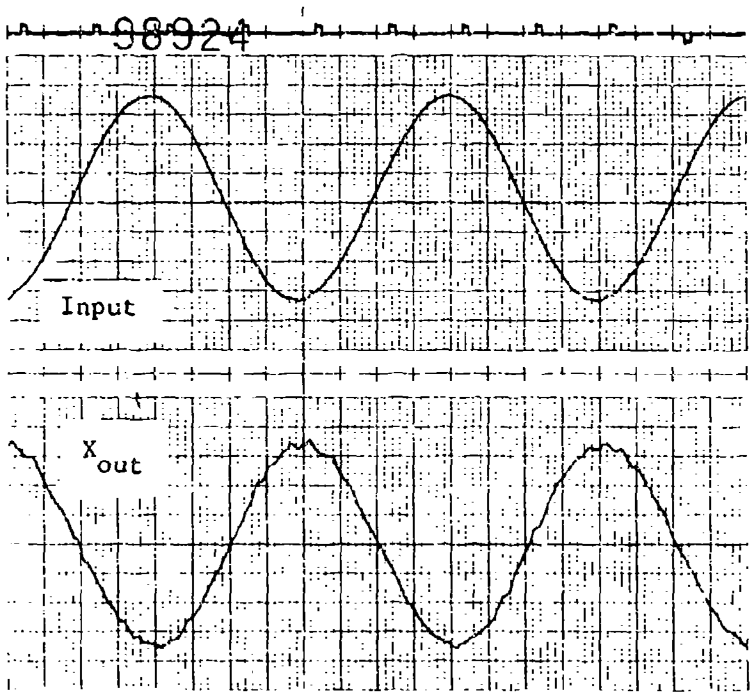
Input Level: 10%
Input Freq.: 0.1 Hz
Load Cond. : C
Bias Cond. : 30%

Scale:

Input - 0.05 v/div

X_{out} - 0.0093 in/div

— t →



Input Level: 10%
Input Freq.: 0.25 Hz
Load Cond. : C
Bias Cond. : 30%

Scale:

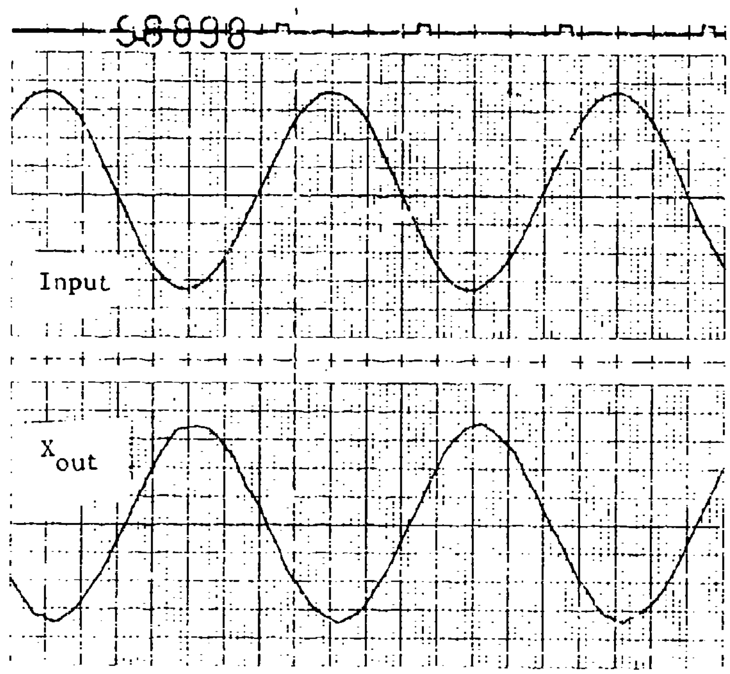
Input - 0.05 v/div

X_{out} - 0.0093 in/div

Figure 80. Output Fidelity @ 0.1 Hz & 0.25 Hz @ 10% Input - Load C - 30% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative
Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 10% Input



Date Prepared 10/19/83

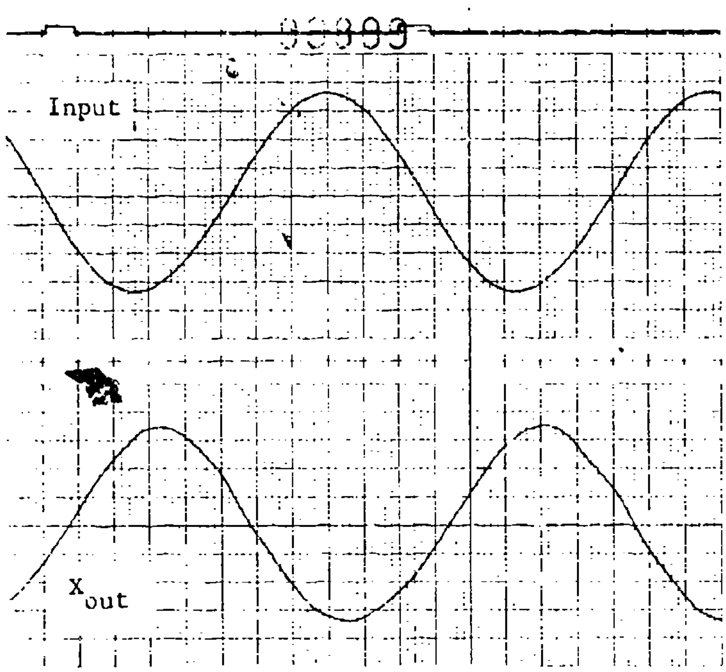
Input Level: 10%
Input Freq.: 0.5 hz
Load Cond. : C
Bias Cond. : 0%

Scale:

Input - 0.05 v/div

X_{out} - 0.0093 in/div

— t ⟶



Input Level: 10%
Input Freq.: 1.0 Hz
Load Cond. : C
Bias Cond. : 0%

Scale:

Input - 0.05 v/div

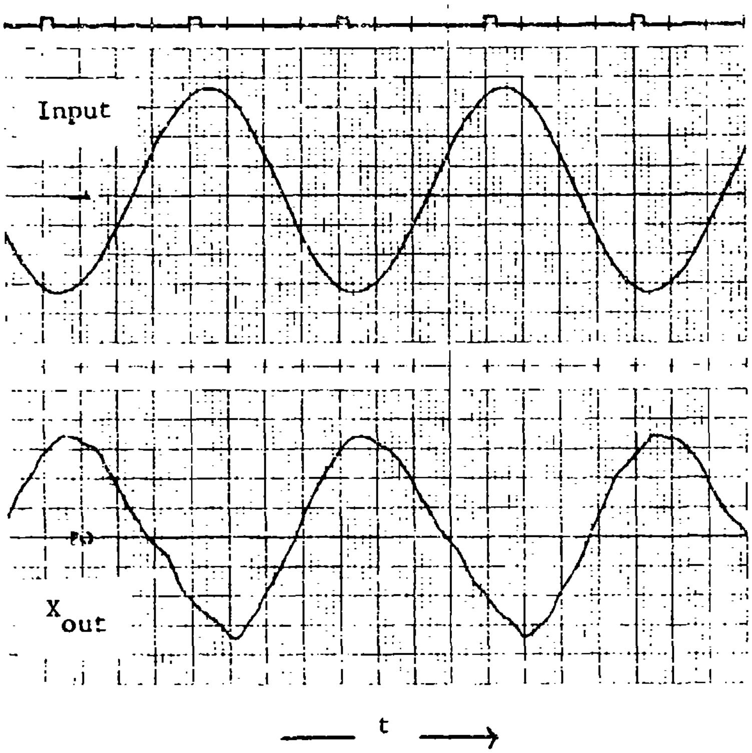
X_{out} - 0.0093 in/div

Figure 81. Output Fidelity @ 0.5 Hz & 1 Hz @ 10% Input - Load C - 0% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 10% Input

Date Prepared 10/19/83

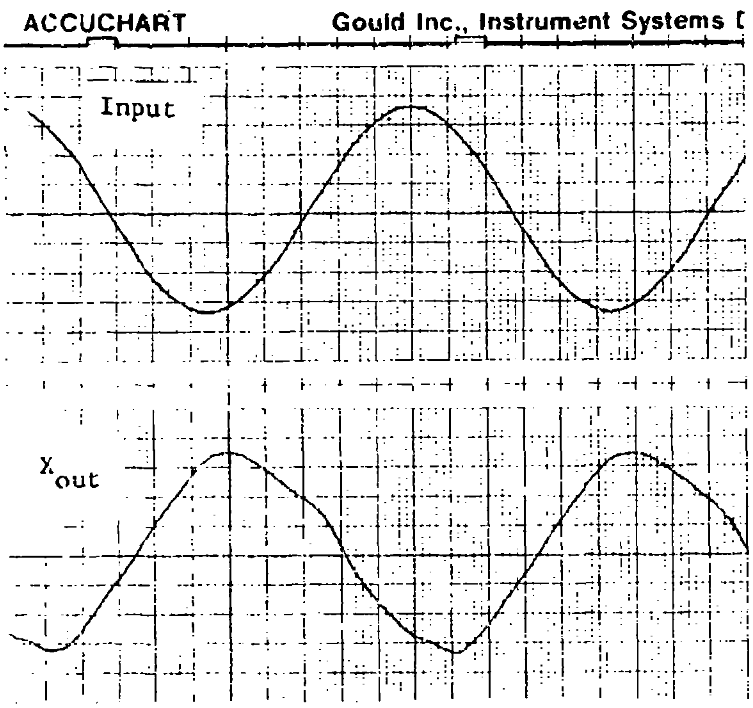


Input Level: 10%
Input Freq.: 0.5 Hz
Load Cond. : C
Bias Cond. : 30%

Scale:

Input - 0.05 v/div

X_{out} - 0.0093 in/div



Input Level: 10%
Input Freq.: 1.0 Hz
Load Cond. : C
Bias Cond. : 30%

Scale:

Input - 0.05 v/div

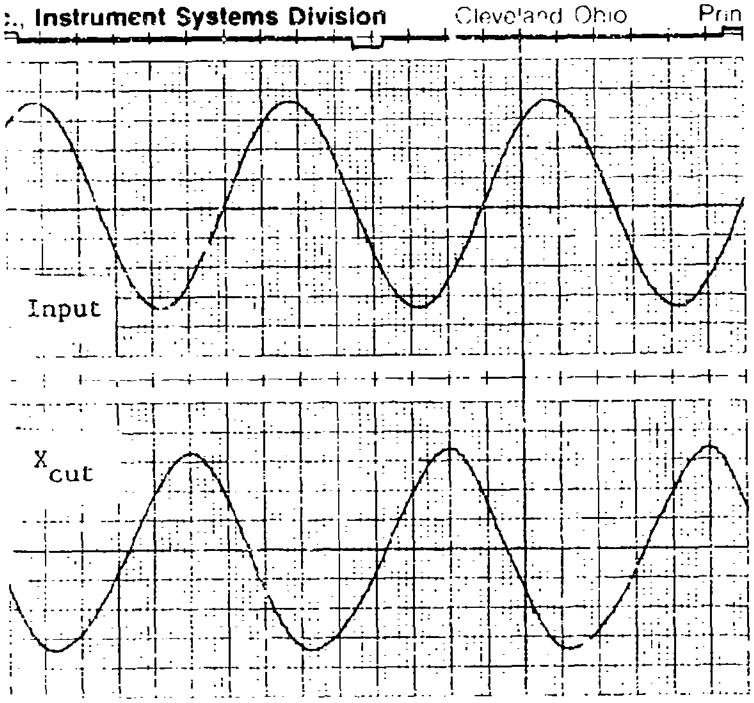
X_{out} - 0.0093 in/div

Figure 82. Output Fidelity @ 0.5 Hz & 1 Hz @ 10% Input - Load C - 30% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative
Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias -
Offset Load - 10% Input

Date Prepared 10/19/83



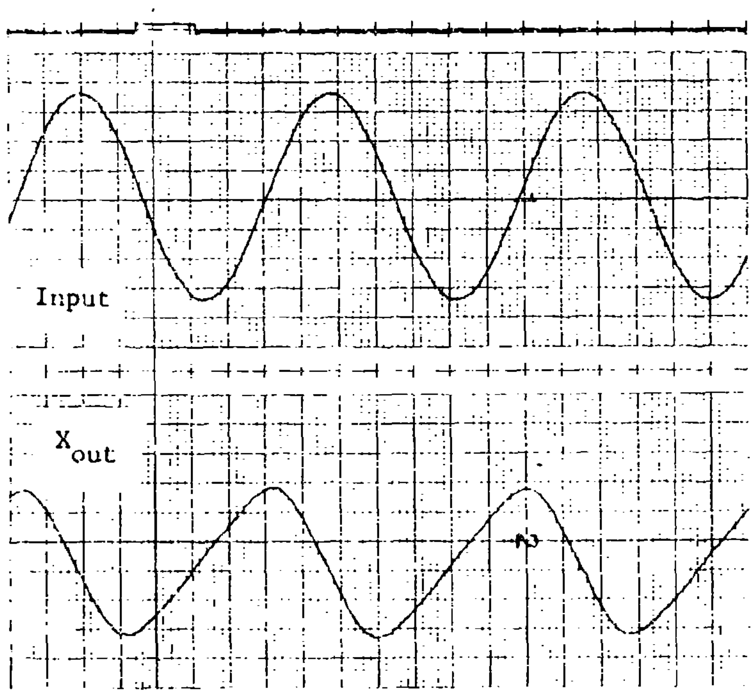
Input Level: 10%
Input Freq.: 2.0 Hz
Load Cond. : C
Bias Cond. : 0%

Scale:

Input - 0.05 v/div

X_{out} - 0.0093 in/div

— t →



Input Level: 10%
Input Freq.: 3.0 Hz
Load Cond. : C
Bias Cond. : 0%

Scale:

Input - 0.05 v/div

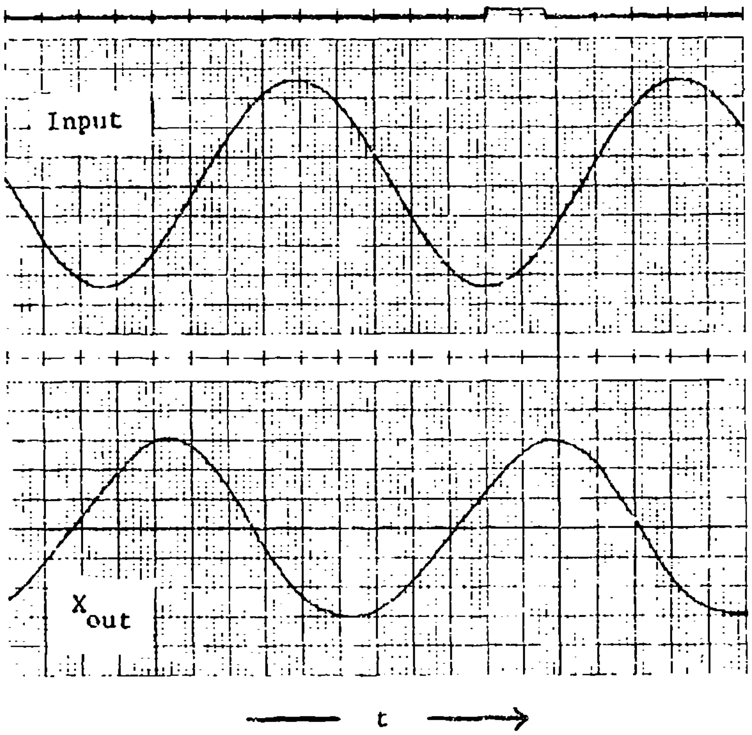
X_{out} - 0.0093 in/div

Figure 83. Output Fidelity @ 2 Hz & 3 Hz @ 10% Input - Load C - 0% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 10% Input

Date Prepared 10/19/83

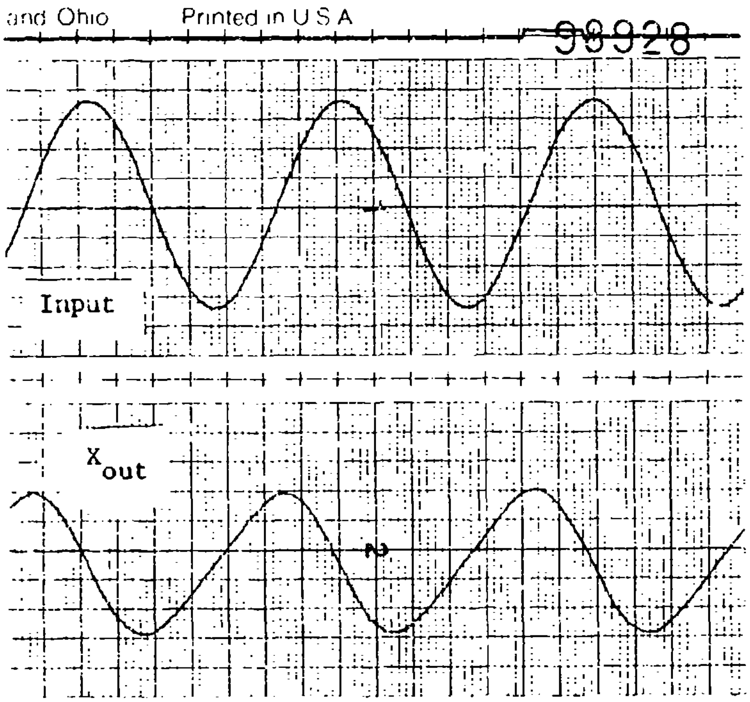


Input Level: 10%
Input Freq.: 2.0 Hz
Load Cond. : C
Bias Cond. : 30%

Scale:

Input - 0.05 v/div

X_{out} - 0.0093 in/div



Input Level: 10%
Input Freq.: 3.0 Hz
Load Cond. : C
Bias Cond. : 30%

Scale:

Input - 0.05 v/div

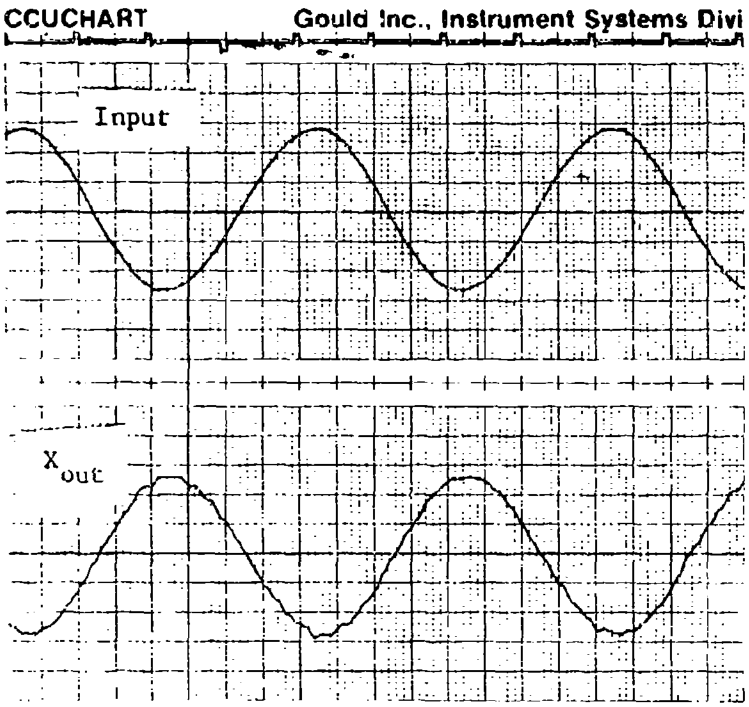
X_{out} - 0.0093 in/div

Figure 84. Output Fidelity @ 2 Hz & 3 Hz @ 10% Input - Load C - 30% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative
Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias -
Symetrical Load - 3% Input

Date Prepared 10/19/83



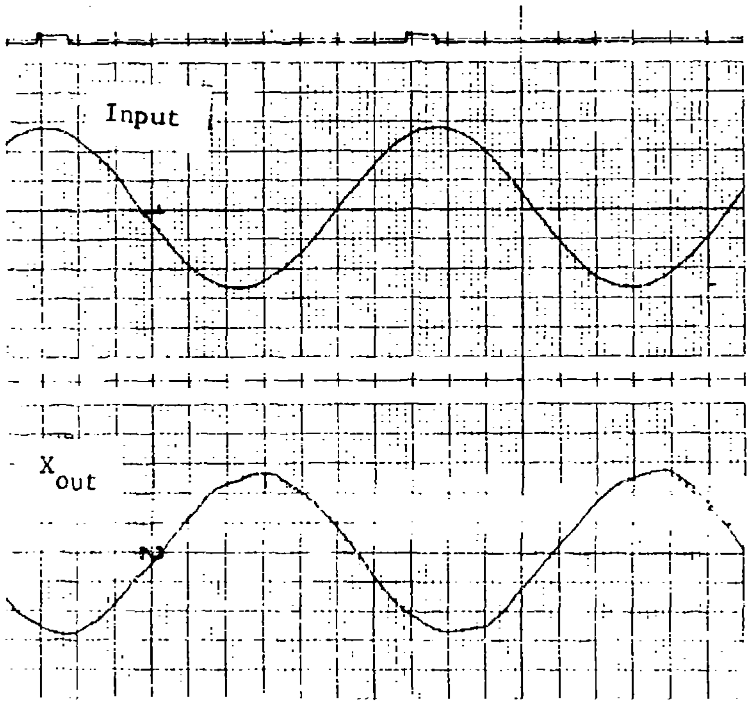
Input Level: 3%
Input Freq.: 0.25 Hz
Load Cond. : B
Bias Cond. : 0%

Scale:

Input - 0.02 v/div

X_{out} - 0.0037 in/div

t →



Input Level: 3%
Input Freq.: 1.0 Hz
Load Cond. : B
Bias Cond. : 0%

Scale:

Input - 0.02 v/div

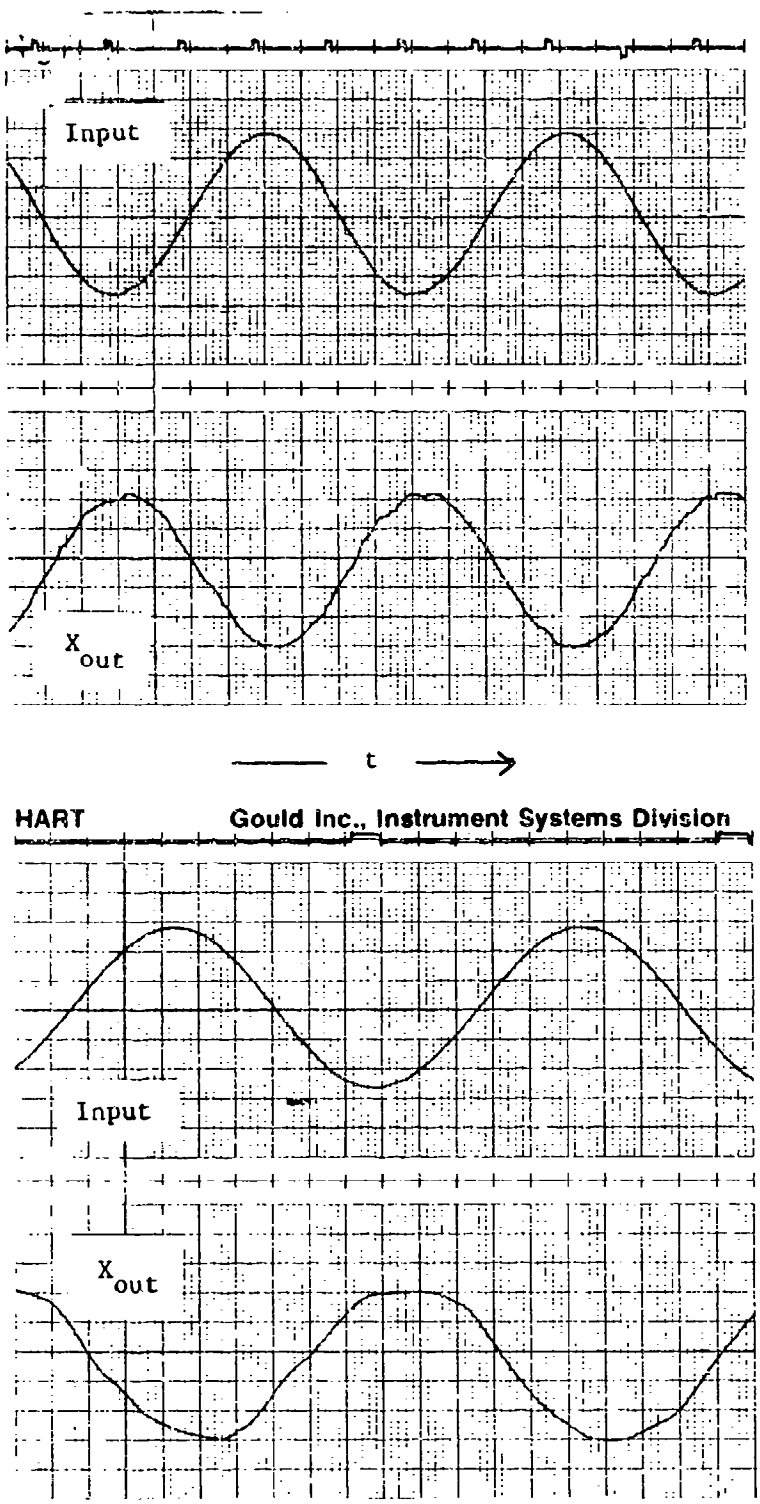
X_{out} - 0.0037 in/div

Figure 85. Output Fidelity @ 0.25 Hz & 1 Hz @ 3% Input - Load B - 0% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Symetrical Load - 3% Input

Date Prepared 10/19/83



Input Level: 3%
Input Freq.: 0.25 Hz
Load Cond. : B
Bias Cond. : 10%

Scale:

Input - 0.02 v/div

X_{out} - 0.0037 in/div

Input Level: 3%
Input Freq.: 1.0 Hz
Load Cond. : B
Bias Cond. : 10%

Scale:

Input - 0.02 v/div

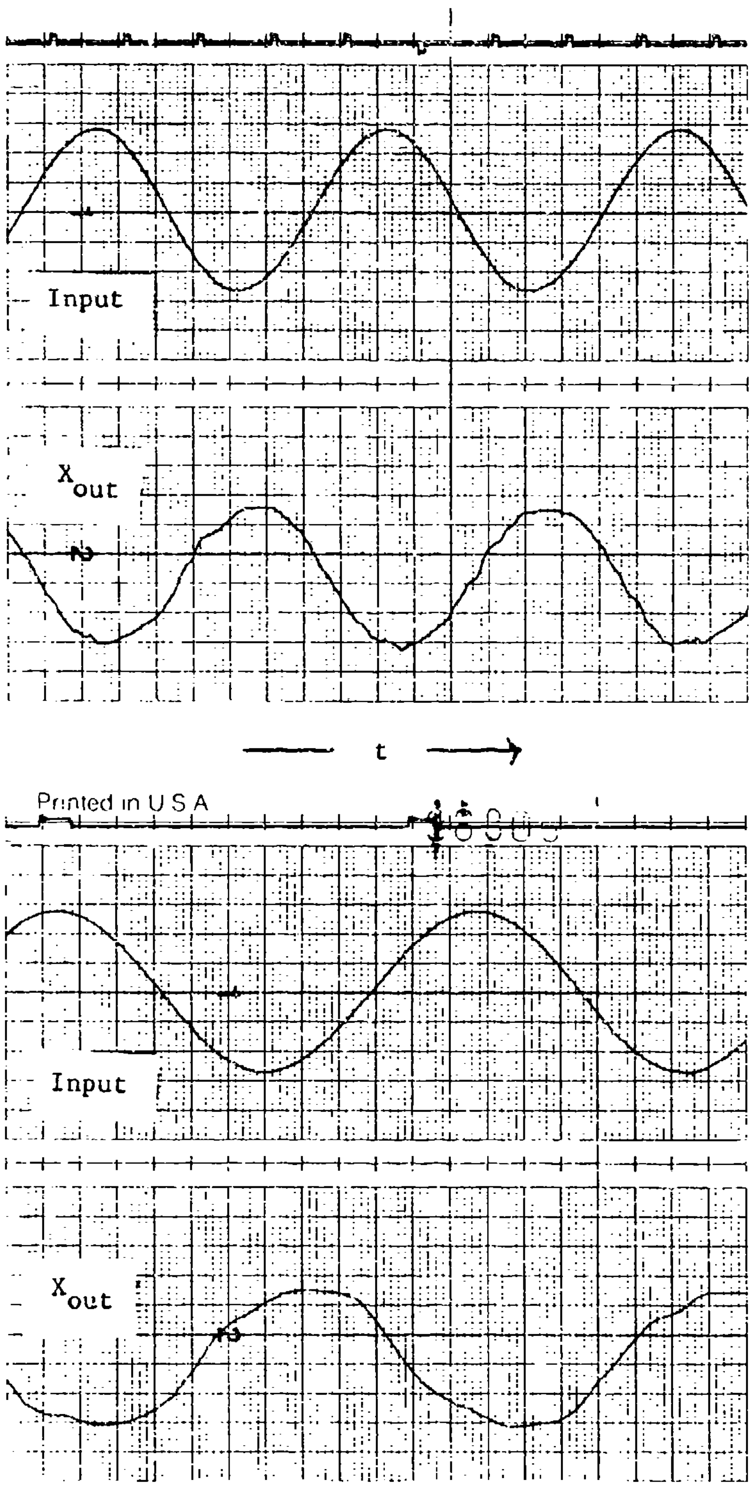
X_{out} - 0.0037 in/div

Figure 86. Output Fidelity @ 0.25 Hz & 1 Hz @ 3% Input - Load B - 10% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Symetrical Load - 3% Input

Date Prepared 10/19/83



Input Level: 3%
Input Freq.: 0.25 Hz
Load Cond. : B
Bias Cond. : 20%

Scale:

Input - 0.02 v/div

X_{out} - 0.0037 in/div

Input Level: 3%
Input Freq.: 1.0 Hz
Load Cond. : B
Bias Cond. : 20%

Scale:

Input - 0.02 v/div

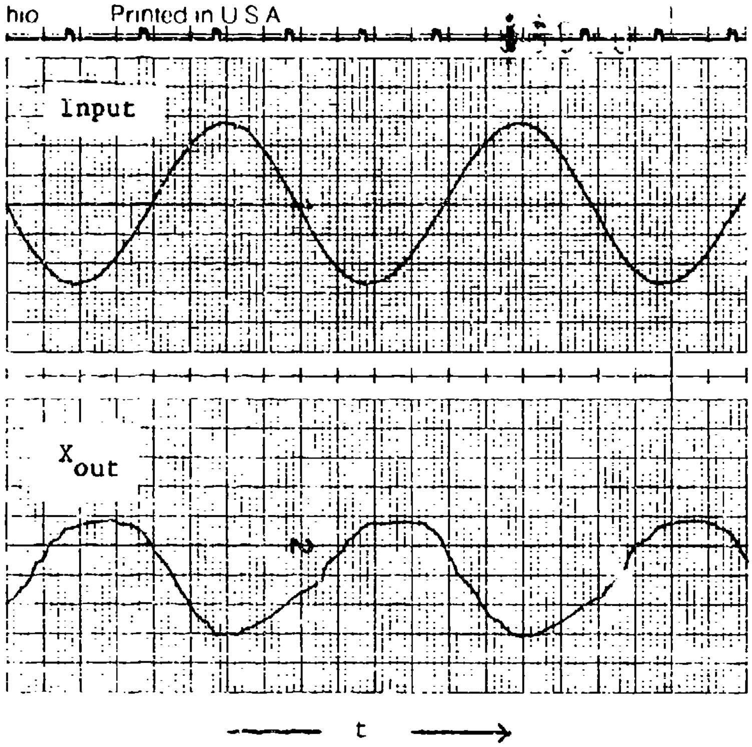
X_{out} - 0.0037 in/div

Figure 87. Output Fidelity @ 0.25 Hz & 1 Hz @ 3% Input - Load B - 20% Bias

TEST ITEM – Boeing Reconfigurable Fail Operative Fly-by-Wire Servoactuator

TEST – Output Fidelity – As a Function of Channel Offset Bias – Symetrical Load – 3% Input

Date Prepared 10/19/83

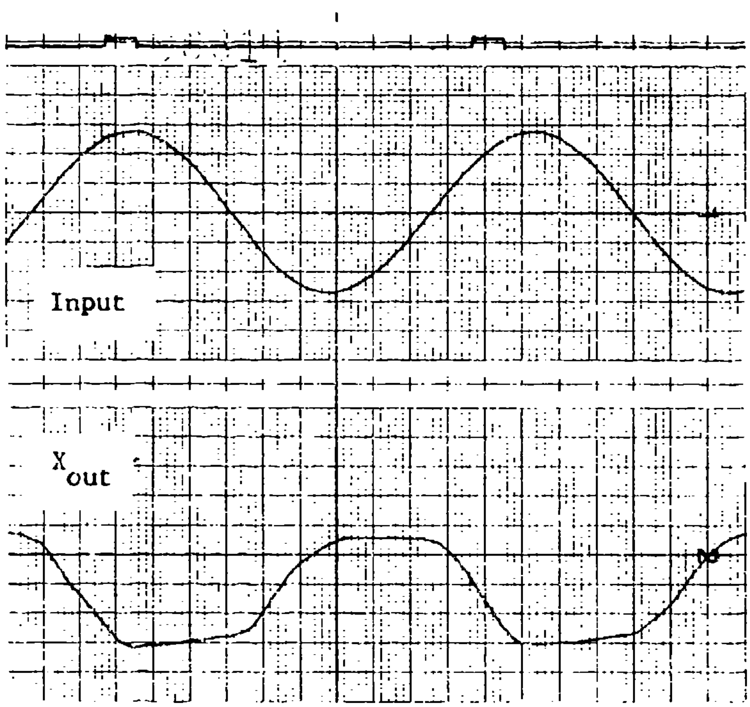


Input Level: 3%
Input Freq.: 0.25 Hz
Load Cond. : B
Bias Cond. : 30%

Scale:

Input – 0.02 v/div

X_{out} – 0.0037 in/div



Input Level: 3%
Input Freq.: 1.0 Hz
Load Cond. : B
Bias Cond. : 30%

Scale:

Input – 0.02 v/div

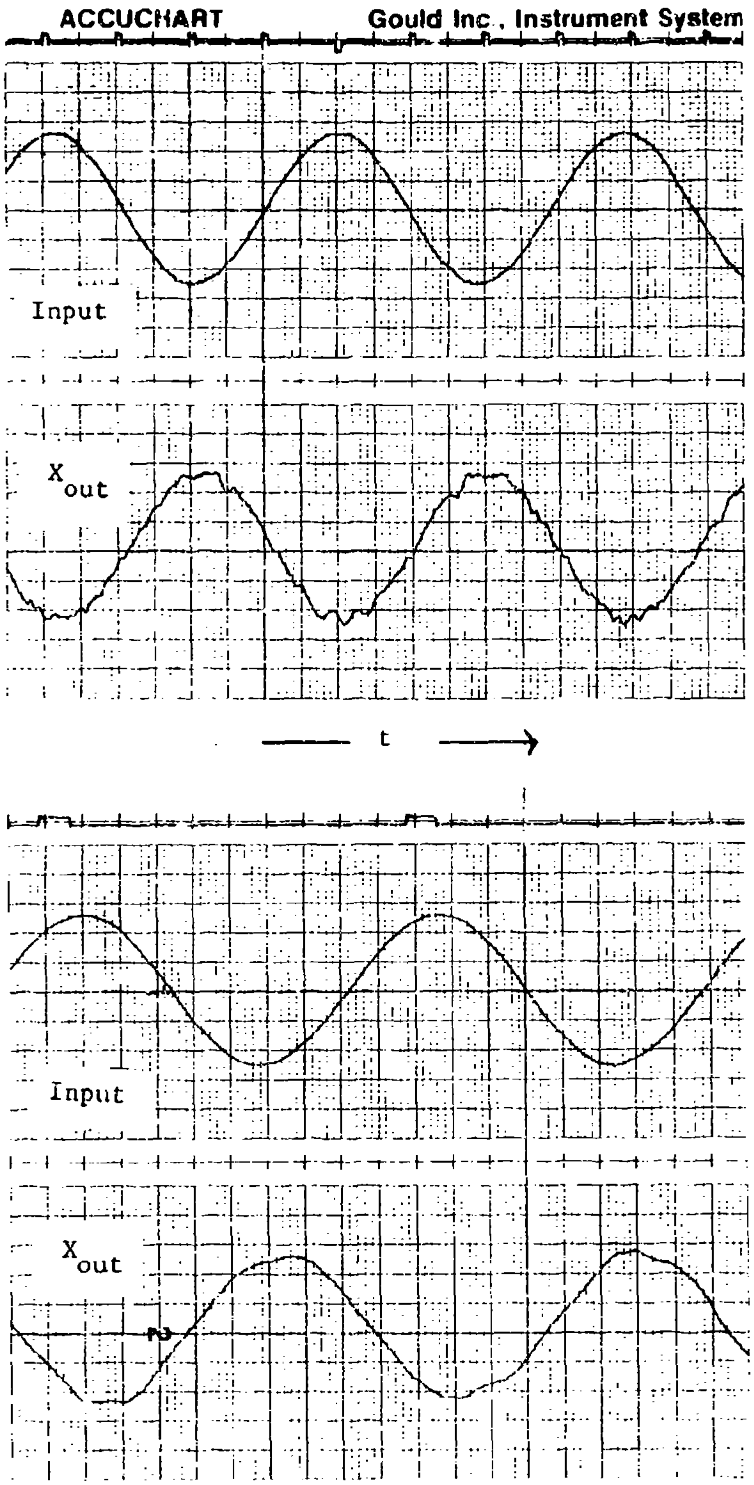
X_{out} – 0.0037 in/div

Figure 88. Output Fidelity @ 0.25 Hz & 1 Hz @ 3% Input – Load B – 30% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 3% Input

Date Prepared 10/18/83



Input Level: 3%
Input Freq.: 0.25 Hz
Load Cond. : C
Bias Cond. : 0%

Scale:

Input - 0.02 v/div

X_{out} - 0.0037 in/div

Input Level: 3%
Input Freq.: 1.0 Hz
Load Cond. : C
Bias Cond. : 0%

Scale:

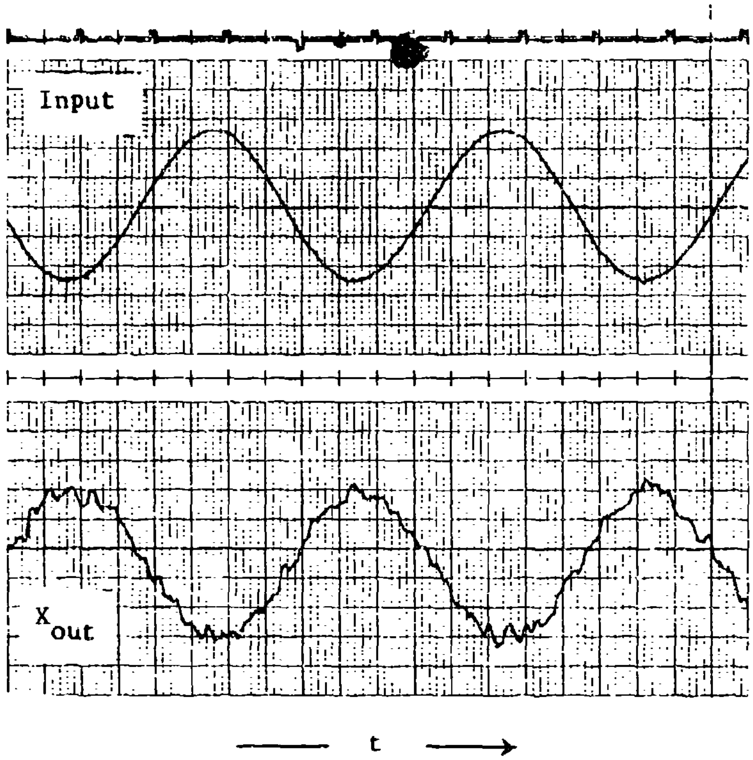
Input -0.02 v/div

X_{out} -0.0037 in/div

Figure 89. Output Fidelity @ 0.25 Hz & 1 Hz @ 3% Input - Load C - 0% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative
Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset
Load - 3% Input



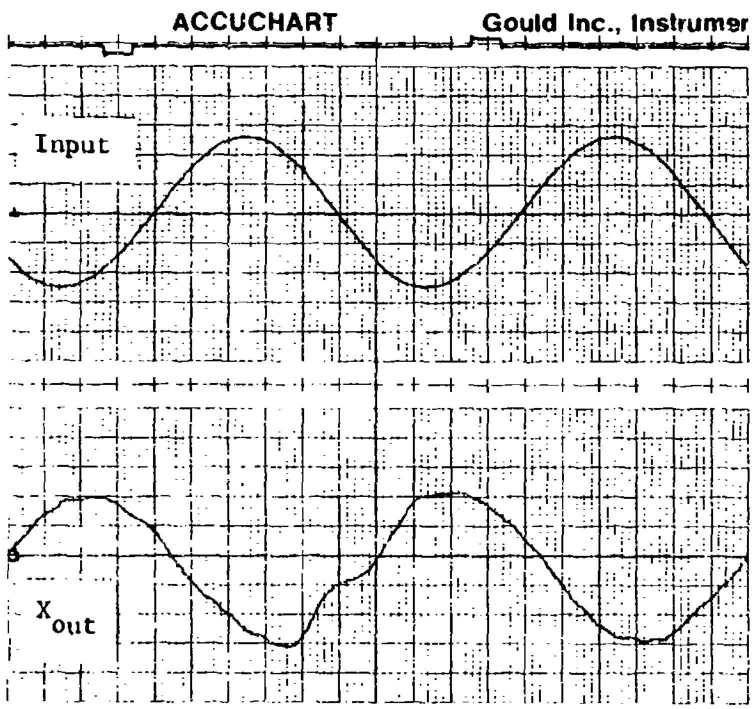
Date Prepared 10/18/83

Input Level: 3%
Input Freq.: 0.25 Hz
Load Cond. : C
Bias Cond. : 10%

Scale:

Input - 0.02 v/div

X_{out} - 0.0037 in/div



Input Level: 3%
Input Freq.: 1.0 Hz
Load Cond. : C
Bias Cond. : 10%

Scale:

Input - 0.02 v/div

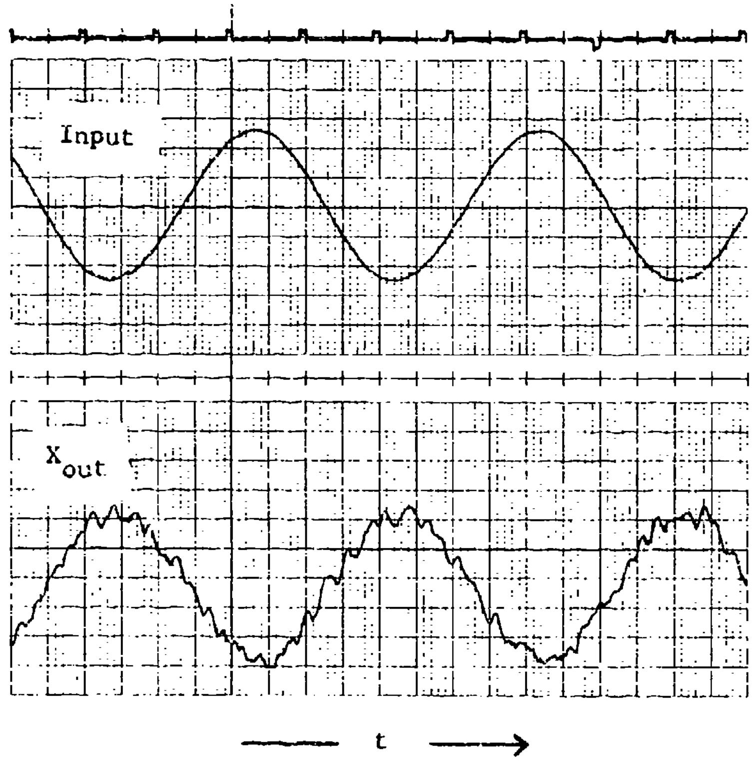
X_{out} - 0.0037 in/div

Figure 90. Output Fidelity @ 0.25 Hz & 1 Hz @ 3% Input - Load C - 10% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 3% Input

Date Prepared 10/18/83

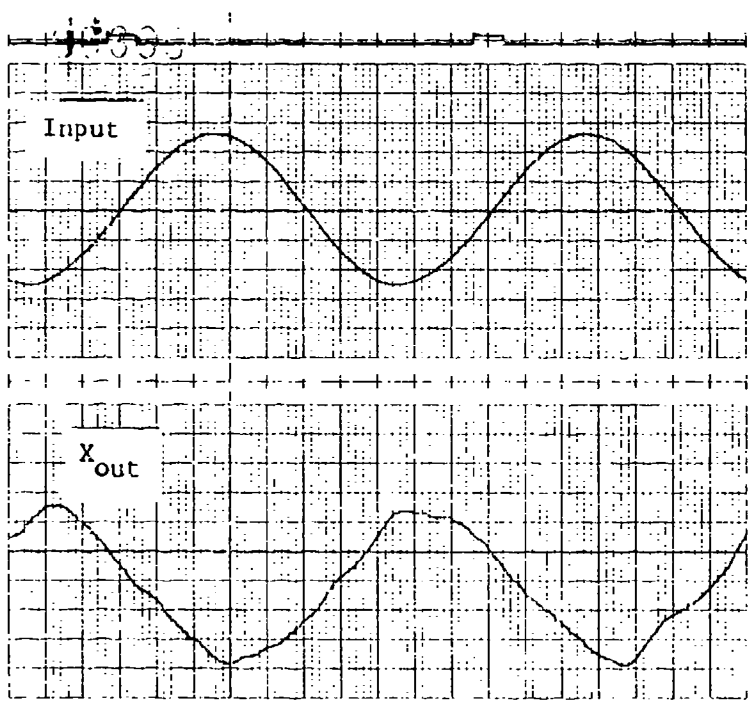


Input Level: 3%
Input Freq.: 0.25 Hz
Load Cond. : C
Bias Cond. : 20%

Scale:

Input - 0.02 v/div

X_{out} - 0.0037 in/div



Input Level: 3%
Input Freq.: 1.0 Hz
Load Cond. : C
Bias Cond. : 20%

Scale:

Input - 0.02 v/div

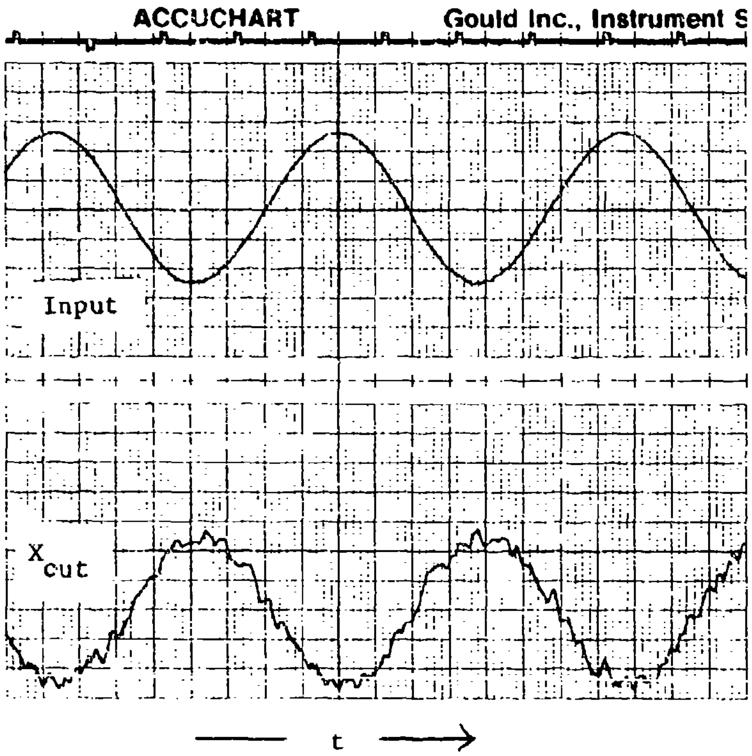
X_{out} - 0.0037 in/div

Figure 91. Output Fidelity @ 0.25 Hz & 1 Hz @ 3% Input - Load C - 20% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 3% Input

Date Prepared 10/18/83

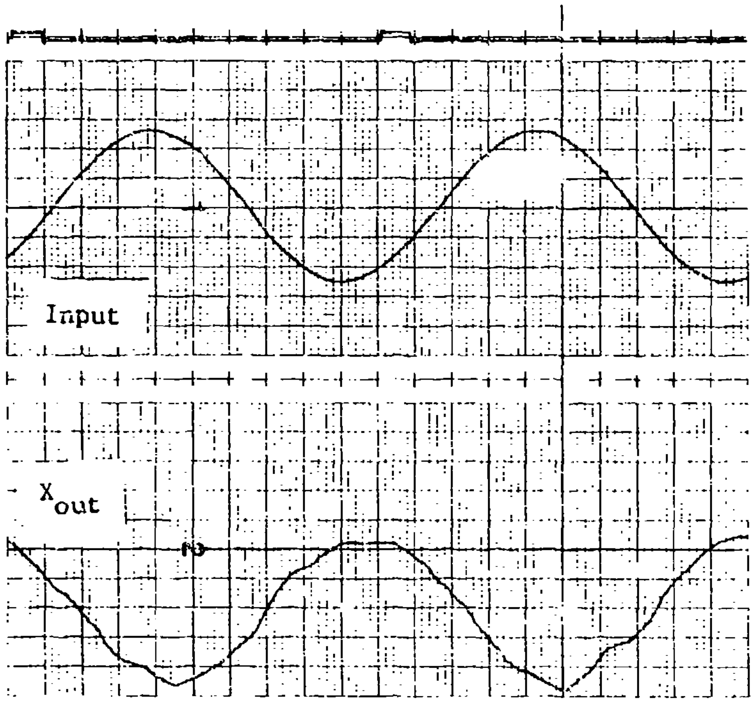


Input Level: 3%
Input Freq.: 0.25 Hz
Load Cond. : C
Bias Cond. : 30%

Scale:

Input - 0.02 v/div

X_{out} - 0.0037 in/div



Input Level: 3%
Input Freq.: 1.0 Hz
Load Cond. : C
Bias Cond. : 30%

Scale:

Input - 0.02 v/div

X_{out} - 0.0037 in/div

Figure 92. Output Fidelity @ 0.25 Hz & 1 Hz @ 3% Input - Load C - 30% Bias

bias at frequencies of 0.25 Hz. and 1 Hz at a 3% input level and load C. This load provides a significant bias load on the output of the test actuator.

Comparing Figures 89 through 92 shows that there is significant output distortion for all bias conditions at both 0.25 and 1 Hz. The distortion increases with increasing bias offset. The distortion at 0.25 Hz is in the form of a ragged modulation of the basic sinusoidal output. The amplitude of the modulation increases with bias to a value of about 12% of sinusoidal amplitude at 30% offset bias. With 0% bias, the distortion occurs at the peaks of the sinusoid. With the 30% bias, the distortion occurs over the entire sinusoidal motion.

The distortion for the 1 Hz output on Figures 89 through 92 is different from that observed at 0.25 Hz. The distortion is in the form of irregular slope changes of the sinusoidal output. The amplitude of the distortion increases with increasing offset bias.

With the 3% input level, the effect of load and bias conditions on the output distortion is more significant than that observed with the 10% input distortion testing. The distortion is least with the lowest offset bias. Load C increases the distortion compared to load B and changes the characteristic of the distortion from a slope charge to a higher frequency modulation of the sinusoidal output at 0.25 Hz.

<u>Output Fidelity vs Offset Bias - Symmetrical Load B - 1% Input</u>

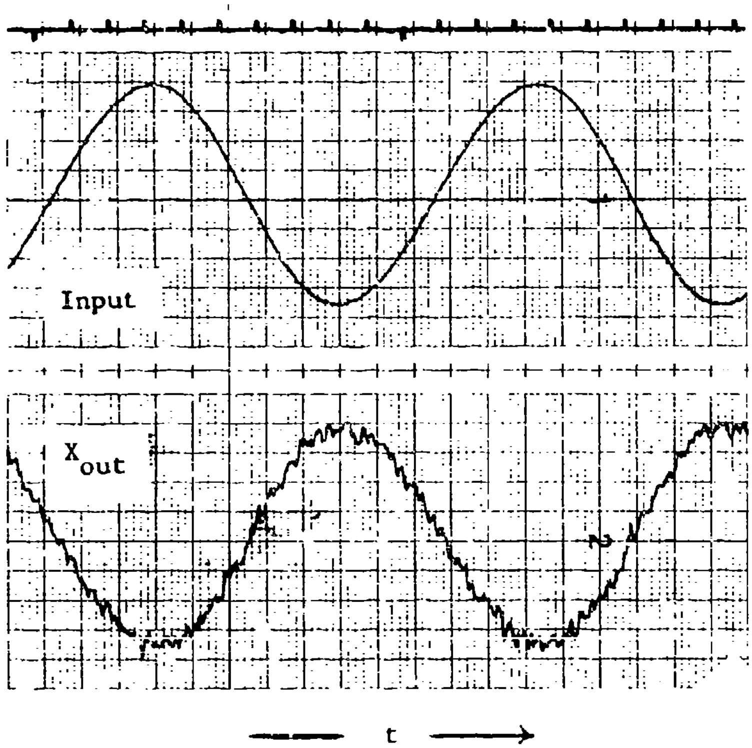
Figures 93 through 104 show the effect of channel bias mismatch on the output of the test system. Load B is used for these figures. No unloaded test results are shown. The test results for load B and unloaded are essentially identical. This could be expected since at the 1% amplitude of output motion, the maximum load applied to the test actuator at peak stroke is 169 lbs (compared to the 18,600 lbs stall output force available from the test actuator). The 1% command level is a small amplitude input signal. This amplitude is consistent with a test input amplitude for fly-by-wire actuators. (The F-16 fly-by-wire actuators are tested and qualified for frequency response at an input amplitude signal of 2% of maximum command). Because of the light loading, these figures (93 through 104) are primarily an evaluation of the effect of bias conditions between active channels on the output of the test system. Figures 93 through 95 show the system output at 1% command for 0% bias offset. Figures 96 through 98 shown the system output at 1% command for 10% bias offset. Figures 99 through 101 show the system output at 1% command for 20% offset. Figures 102 through 104 show the system output at 1% command for 30% bias offset. The 30% bias offset is just below the 35% spool position difference used for the failure detection threshold for test system failure logic.

Figures 93 through 95 with 0 input offset bias show the same distortion on the output at all test frequencies used. The distortion amplitude appears constant from 0.1 Hz to 2 Hz and at a constant frequency. Because of the input frequency change, the effect of the distortion components appears as ragged modulation of the 0.1 Hz output waveform (reference Figure 93) and a irregular slope change of the 1 Hz output waveform. The distortion amplitude appears nominally 11% of the output amplitude (or 0.11% of the maximum stroke). Figures

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Symetrical Load - 1% Input

Date Prepared 10/19/83

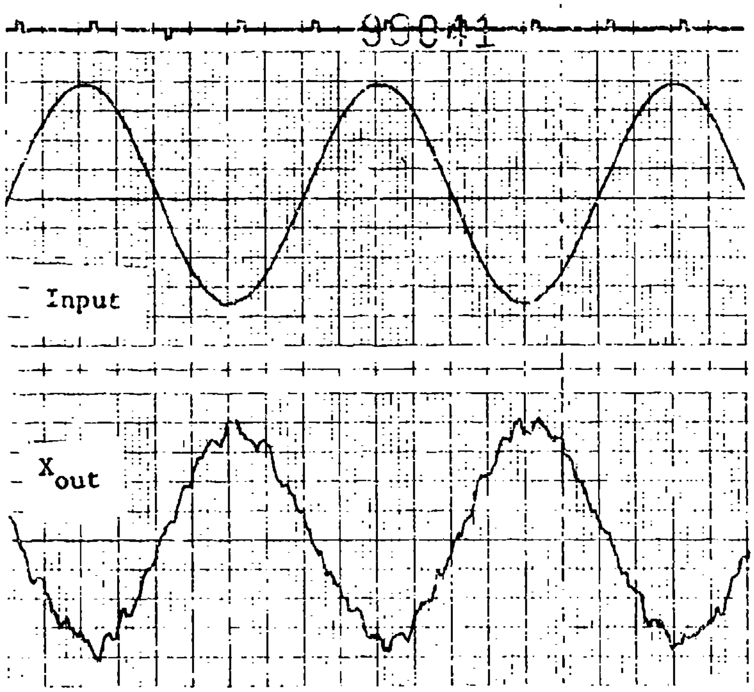


Input Level: 1%
Input Freq.: 0.1 Hz
Load Cond. : B
Bias Cond. : 0%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div



Input Level: 1%
Input Freq.: 0.25 Hz
Load Cond. : B
Bias Cond. : 0%

Scale:

Input - 0.005 v/div

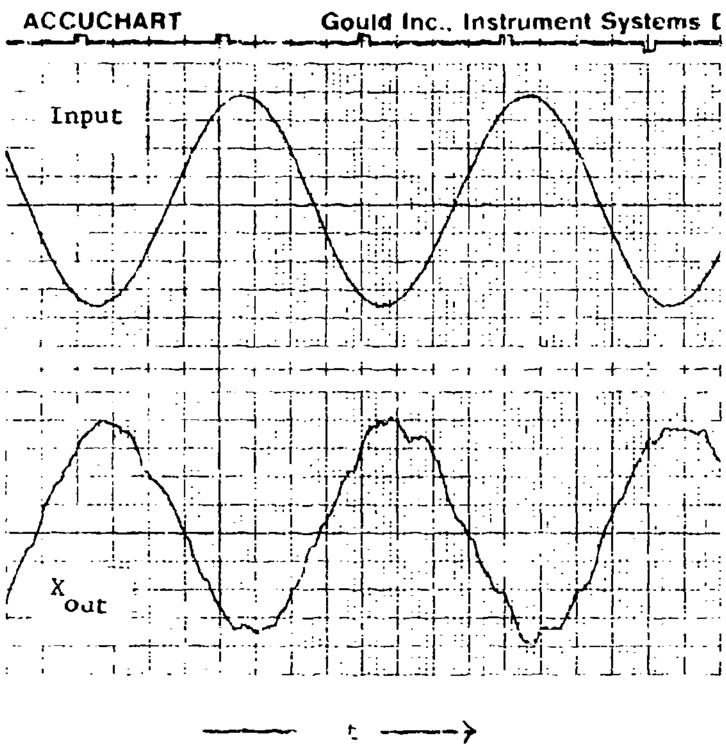
X_{out} - 0.00093 in/div

Figure 93. Output Fidelity @ 0.1 Hz & 0.25 Hz @ 1% Input - Load B - 0% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Symetrical Load - 1% Input

Date Prepared 10/19/83

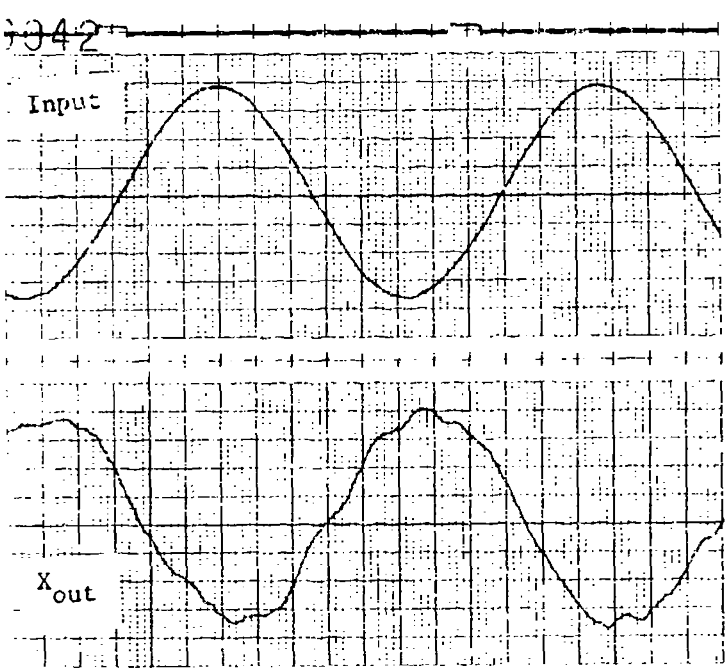


Input Level: 1%
Input Freq.: 0.5 Hz
Load Cond. : B
Bias Cond. : 0%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div



Input Level: 1%
Input Freq.: 1.0 Hz
Load Cond. : B
Bias Cond. : 0%

Scale:

Input - 0.005 v/div

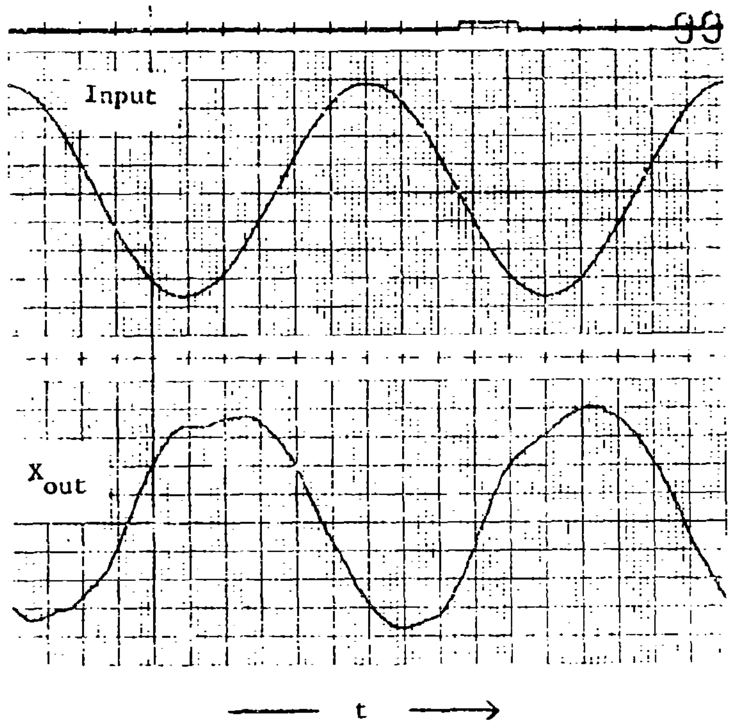
X_{out} - 0.00093 in/div

Figure 94. Output Fidelity @ 0.5 Hz & 1 Hz @ 1% Input - Load B - 0% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Symetrical Load - 1% Input

Date Prepared 10/19/83

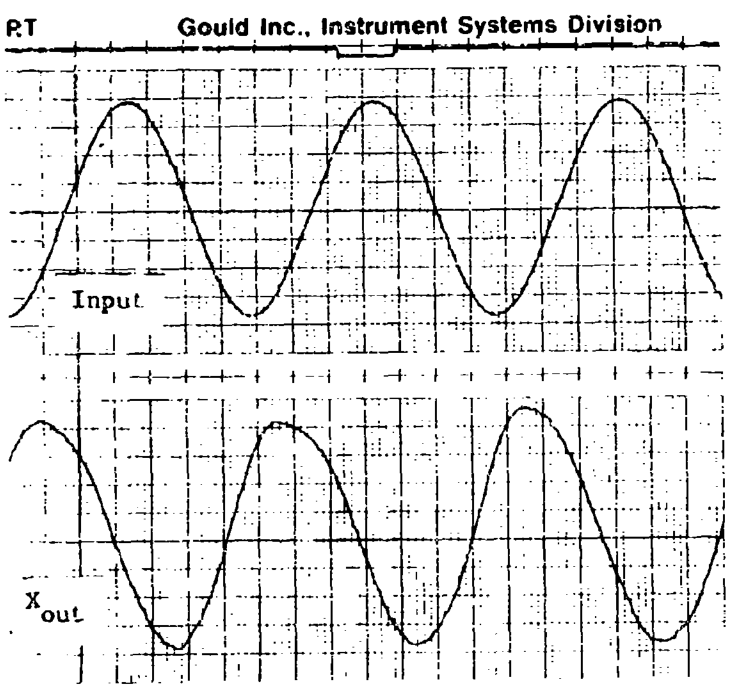


Input Level: 1%
Input Freq.: 2.0 Hz
Load Cond. : B
Bias Cond. : 0%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div



Input Level: 1%
Input Freq.: 3.0 Hz
Load Cond. : B
Bias Cond. : 0%

Scale:

Input - 0.005 v/div

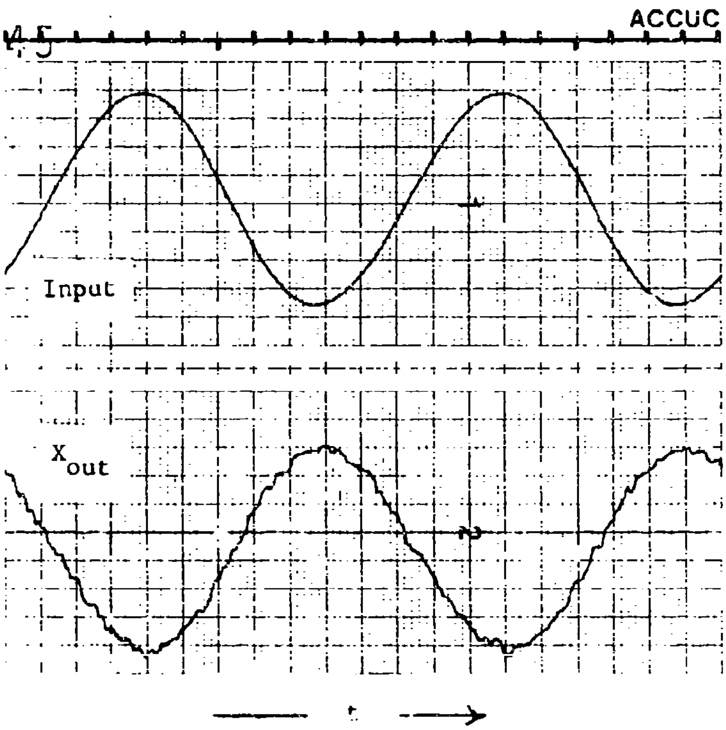
X_{out} - 0.00093 in/div

Figure 95. Output Fidelity @ 2 Hz & 3 Hz @ 1% Input - Load B - 0% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Symetrical Load - 1% Input

Date Prepared 10/19/83

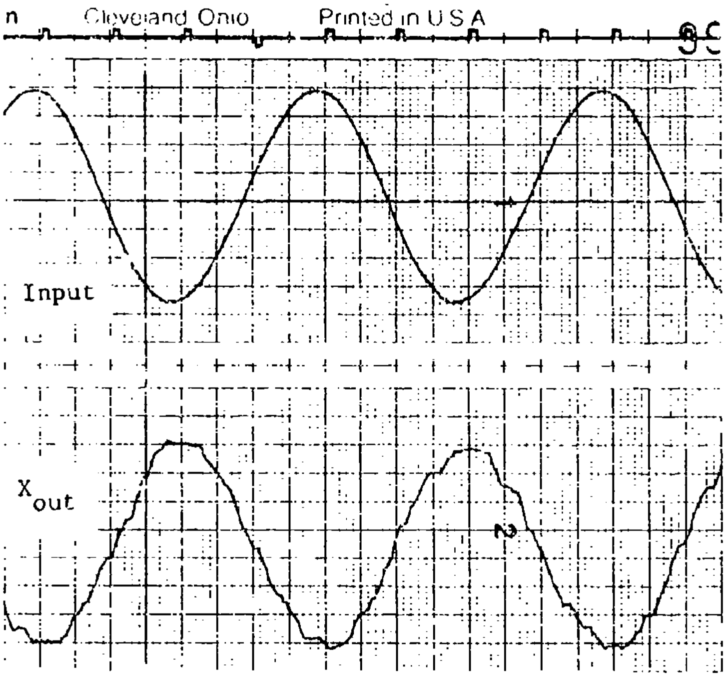


Input Level: 1%
Input Freq.: 0.1 Hz
Load Cond. : B
Bias Cond. : 10%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div



Input Level: 1%
Input Freq.: 0.25 Hz
Load Cond. : B
Bias Cond. : 10%

Scale:

Input - 0.005 v/div

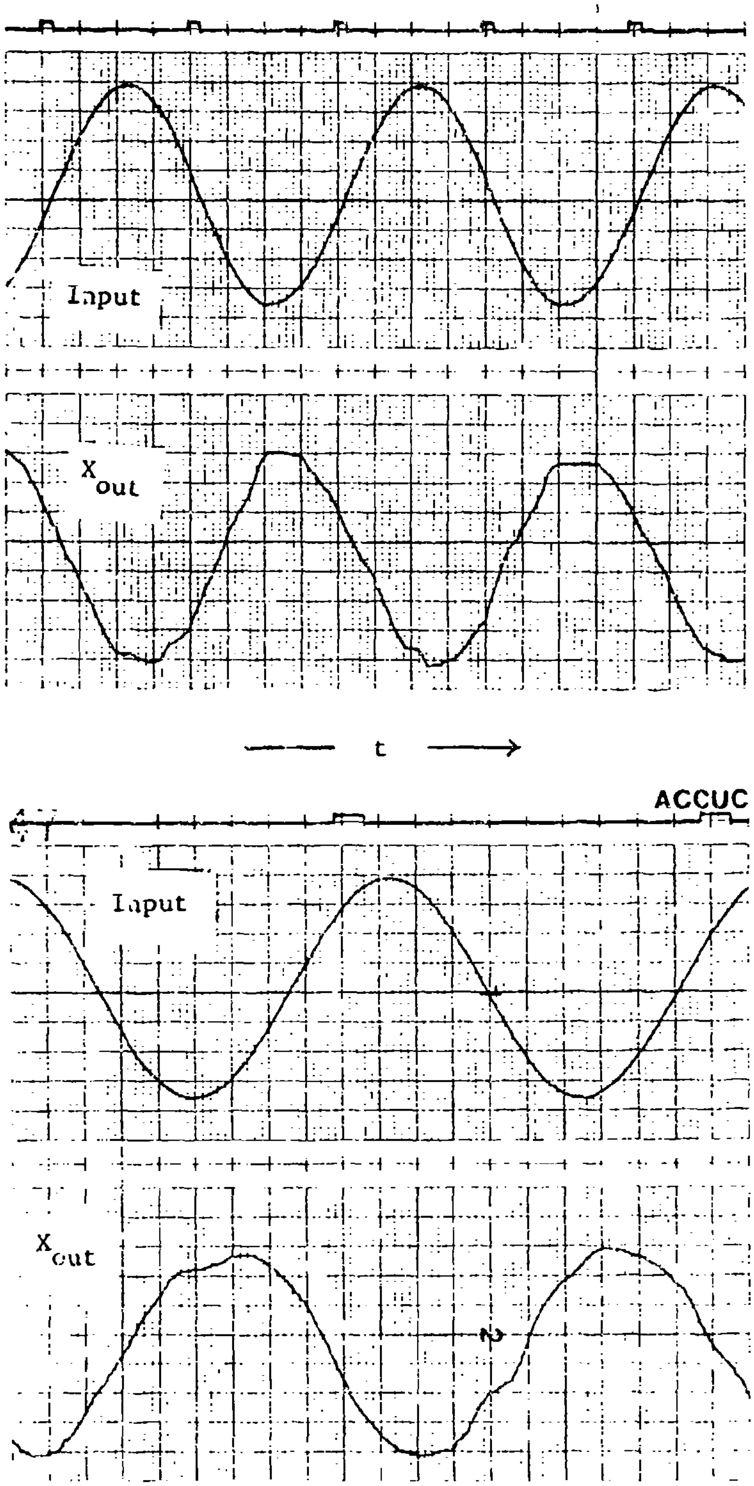
X_{out} - 0.00093 in/div

Figure 96. Output Fidelity @ 0.1 Hz & 0.25 Hz @ 1% Input - Load B - 10% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Symetrical Load - 1% Input

Date Prepared 10/19/83



Input Level: 1%
Input Freq.: 0.5 Hz
Load Cond. : B
Bias Cond. : 10%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div

Input Level: 1%
Input Freq.: 1.0 Hz
Load Cond. : B
Bias Cond. : 10%

Scale:

Input - 0.005 v/div

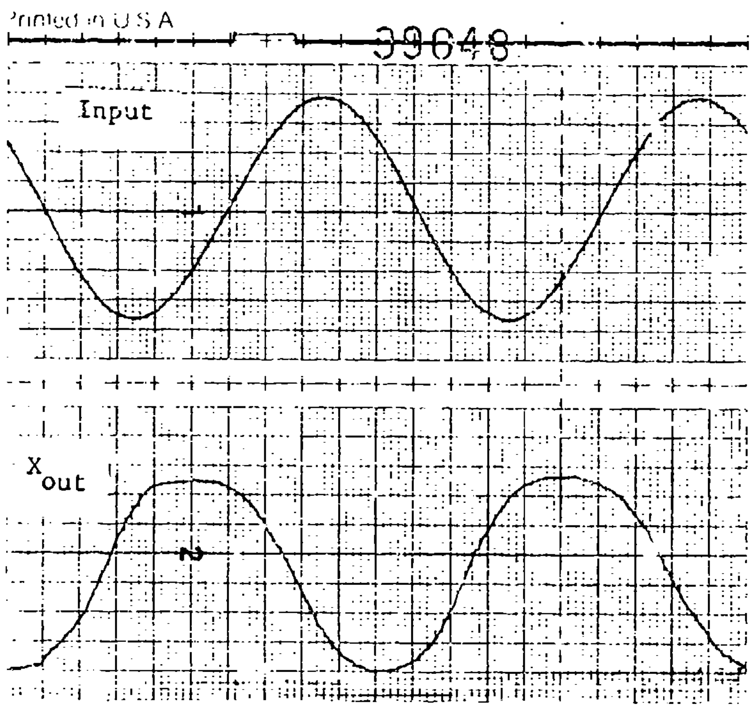
X_{out} - 0.00093 in/div

Figure 97. Output Fidelity @ 0.5 Hz & 1 Hz @ 1% Input - Load B - 10% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Symetrical Load - 1% Input

Date Prepared 10/19/83



Input Level: 1%
Input Freq.: 2.0 Hz
Load Cond. : B
Bias Cond. : 10%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div

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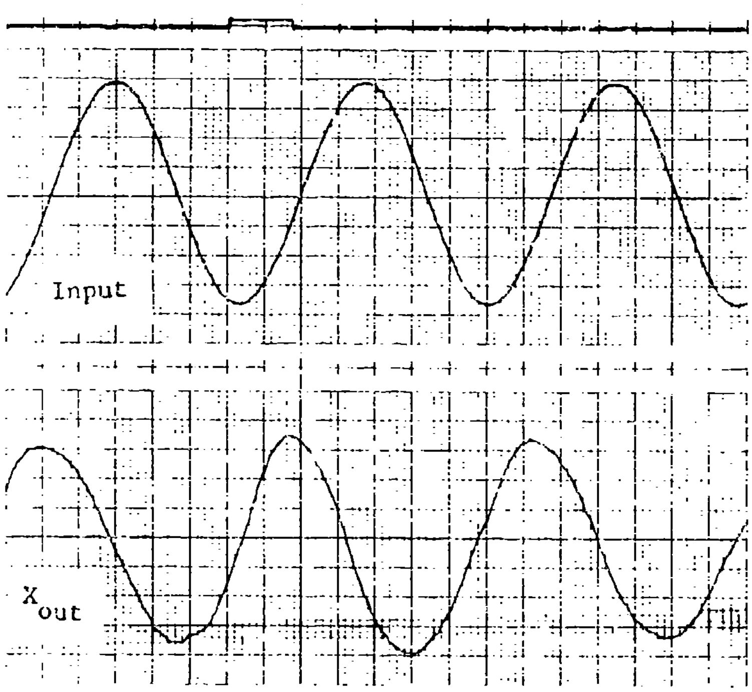


Input Level: 1%
Input Freq.: 3.0 Hz
Load Cond. : B
Bias Cond. : 10%

Scale:

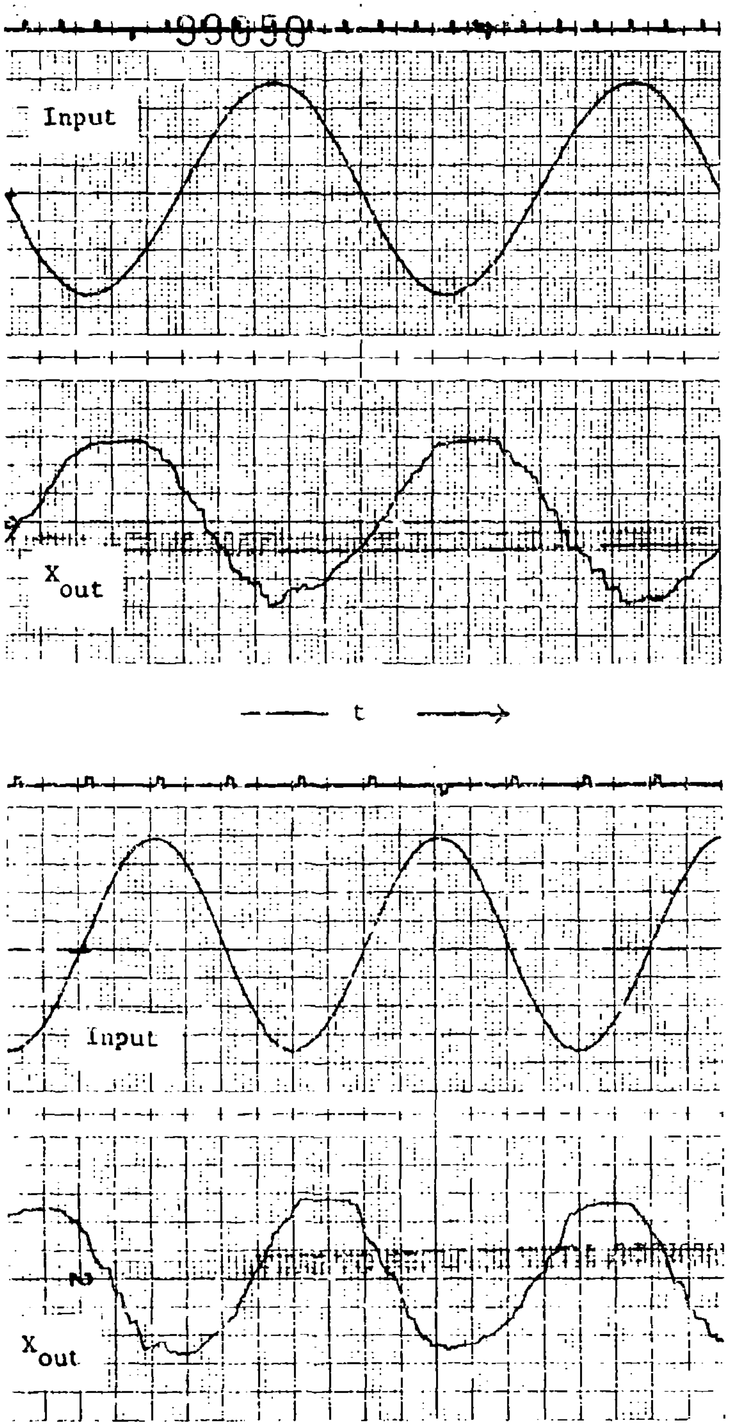
Input - 0.005 v/div

X_{out} - 0.00093 in/div

Figure 98. Output Fidelity @ 2 Hz & 3 Hz @ 1% Input - Load B - 10% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Symetrical Load - 1% Input



Date Prepared 10/19/83

Input Level: 1%
Input Freq.: 0.1 Hz
Load Cond. : B
Bias Cond. : 20%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div

Input Level: 1%
Input Freq.: 0.25 Hz
Load Cond. : B
Bias Cond. : 20%

Scale:

Input - 0.005 v/div

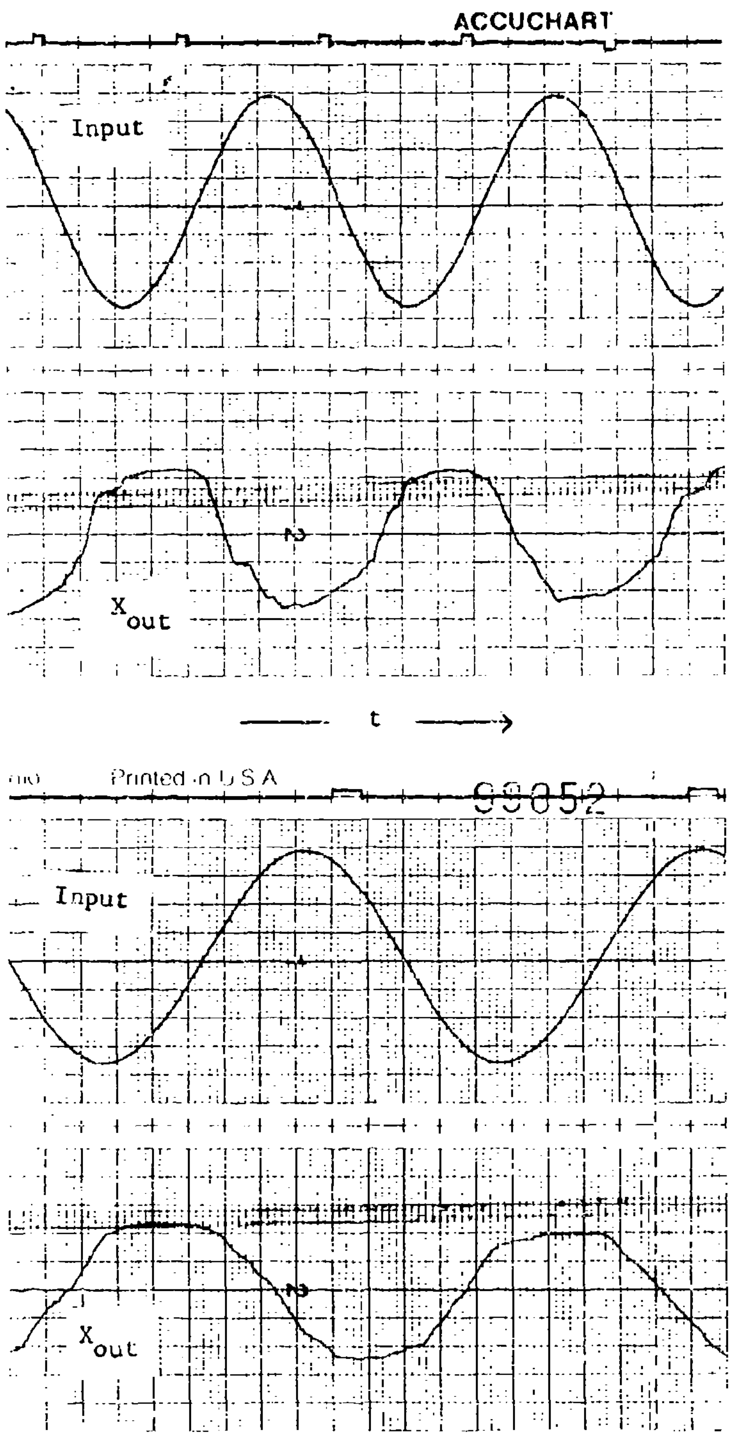
X_{out} - 0.00093 in/div

Figure 99. Output Fidelity @ 0.1 Hz & 0.25 Hz @ 1% Input - Load B - 20% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Symetrical Load - 1% Input

Date Prepared 10/19/83



Input Level: 1%
Input Freq.: 0.5 Hz
Load Cond. : B
Bias Cond. : 20%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div

Input Level: 1%
Input Freq.: 1.0 Hz
Load Cond. : B
Bias Cond.: 20%

Scale:

Input - 0.005 v/div

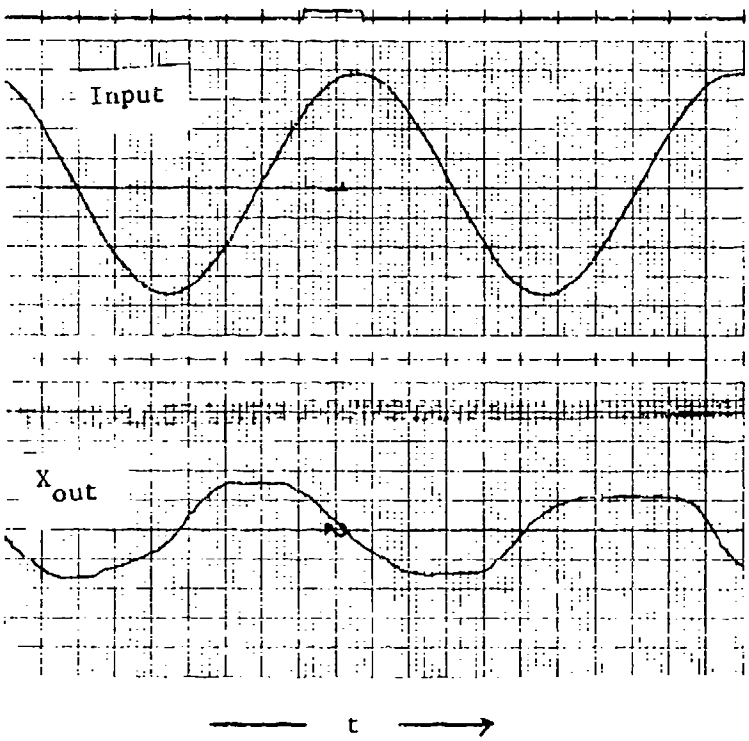
X_{out} - 0.00093 in/div

Figure 100. Output Fidelity @ 0.5 Hz & 1 Hz @ 1% Input - Load B - 20% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative
Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Symetrical Load - 1% Input

Date Prepared 10/19/83

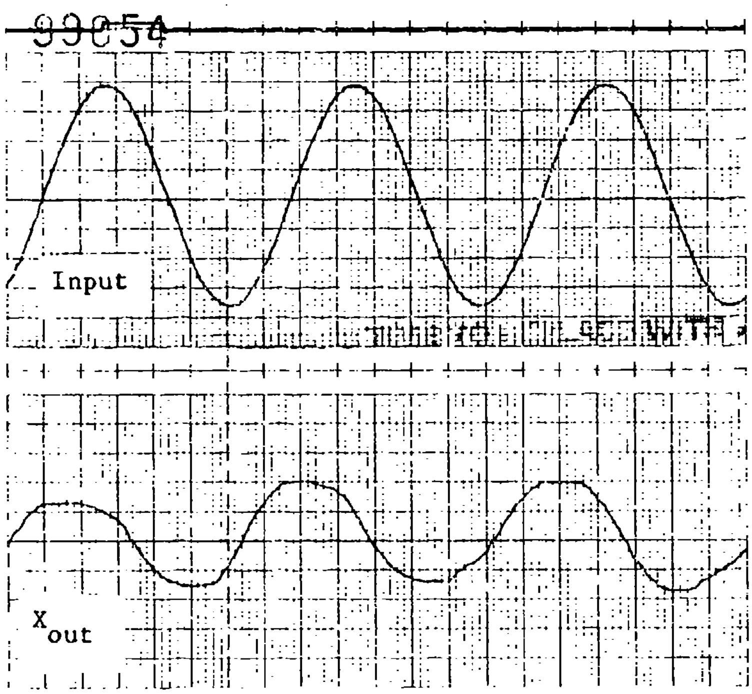


Input Level: 1%
Input Freq.: 2.0 Hz
Load Cond. : B
Bias Cond. : 20%

Scale:

Input - 0.005 v/div

X_{out} - 0.0093 in/div



Input Level: 1%
Input Freq.: 3.0 Hz
Load Cond. : B
Bias Cond. : 20%

Scale:

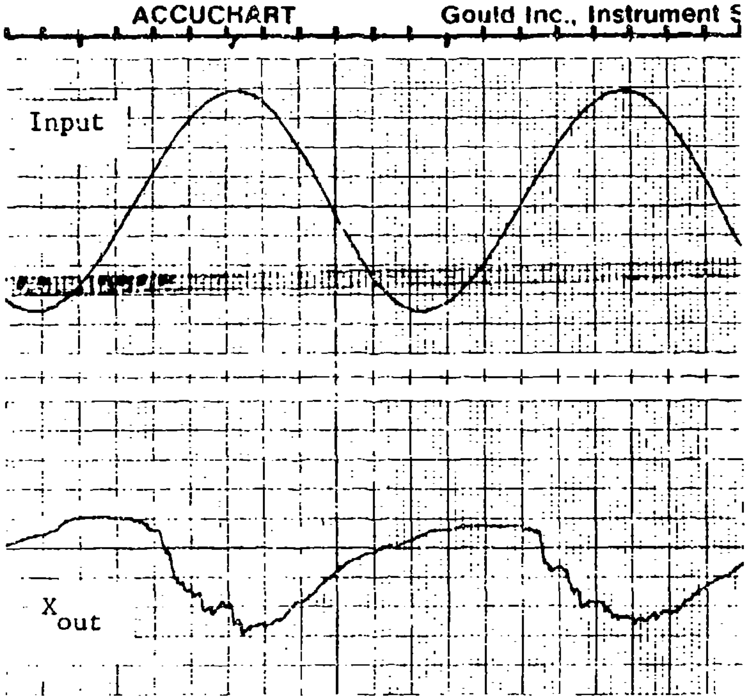
Input - 0.005 v/div

X_{out} - 0.00093 in/div

Figure 101. Output Fidelity @ 2 Hz & 3 Hz @ 1% Input - Load B - 20% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Symetrical Load - 1% Input



Date Prepared 10/19/83

Input Level: 1%
Input Freq.: 0.1 Hz
Load Cond. : B
Bias Cond. : 30%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div

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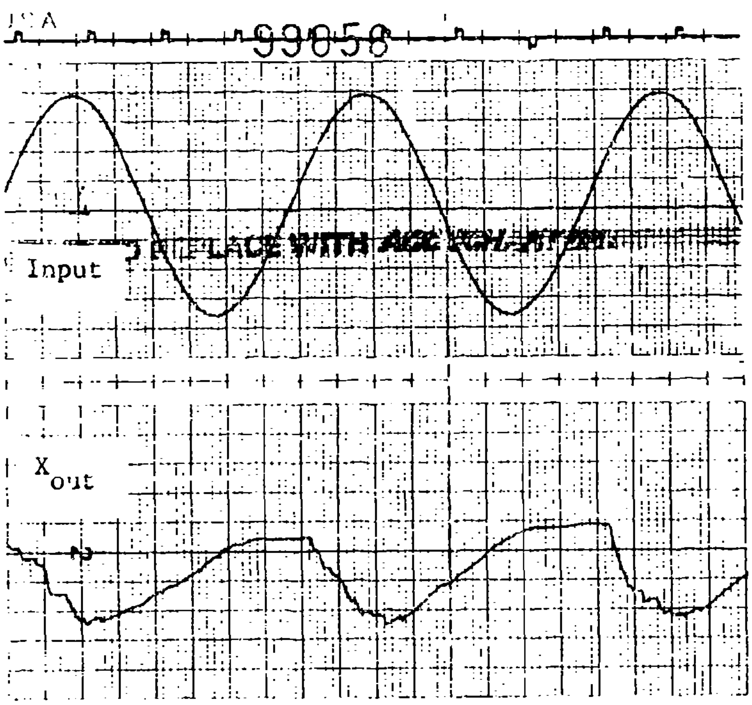


Input Level: 1%
Input Freq.: 0.25 Hz
Load Cond. : B
Bias Cond. : 30%

Scale:

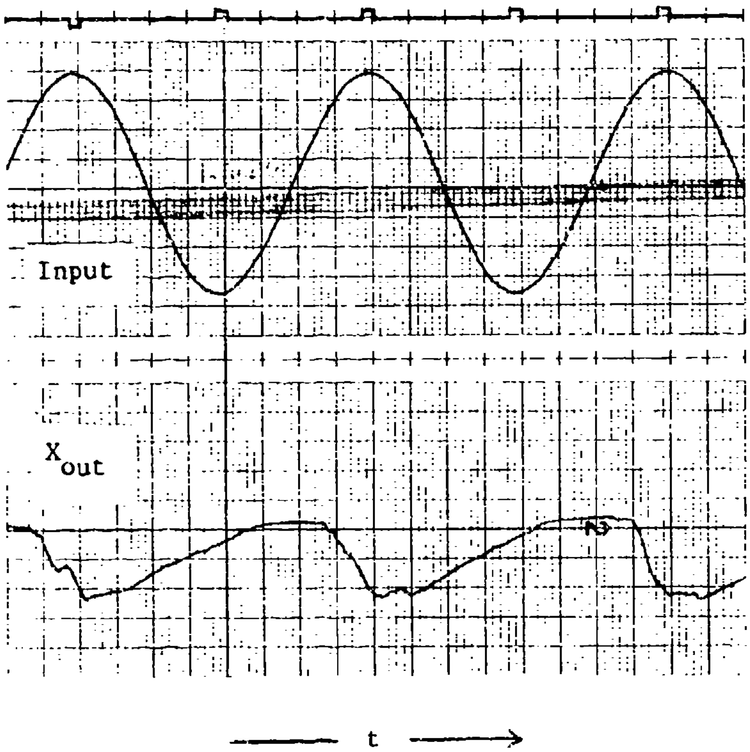
Input - 0.005 v/div

X_{out} - 0.00093 in/ iv

Figure 102. Output Fidelity @ 0.1 Hz & 0.25 Hz @ 1% Input - Load B - 30% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servaoctuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Symetrical Load - 1% Input



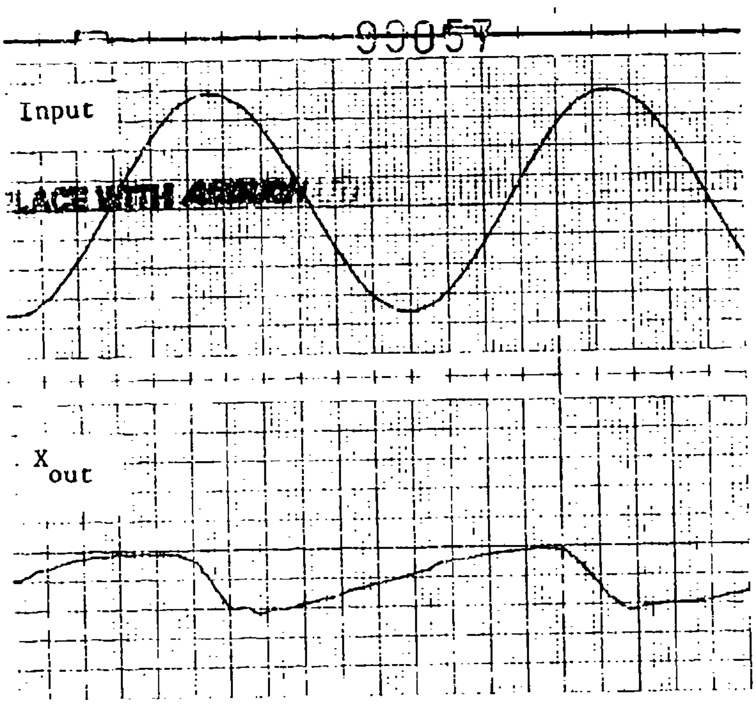
Date Prepared 10/19/83

Input Level: 1%
Input Freq.: 0.5 Hz
Load Cond. : B
Bias Cond. : 30%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div



Input Level: 1%
Input Freq.: 1.0 Hz
Load Cond. : B
Bias Cond. : 30%

Scale:

Input - 0.005 v/div

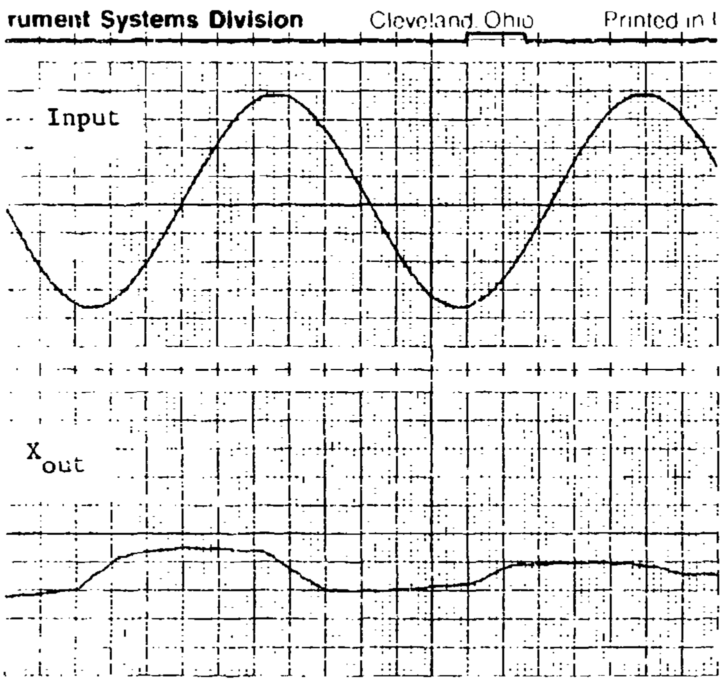
X_{out} - 0.00093 in/div

Figure 103. Output Fidelity @ 0.5 Hz & 1 Hz @ 1% Input - Load B - 30% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Symmetrical Load - 1% Input

Date Prepared 10/19/83



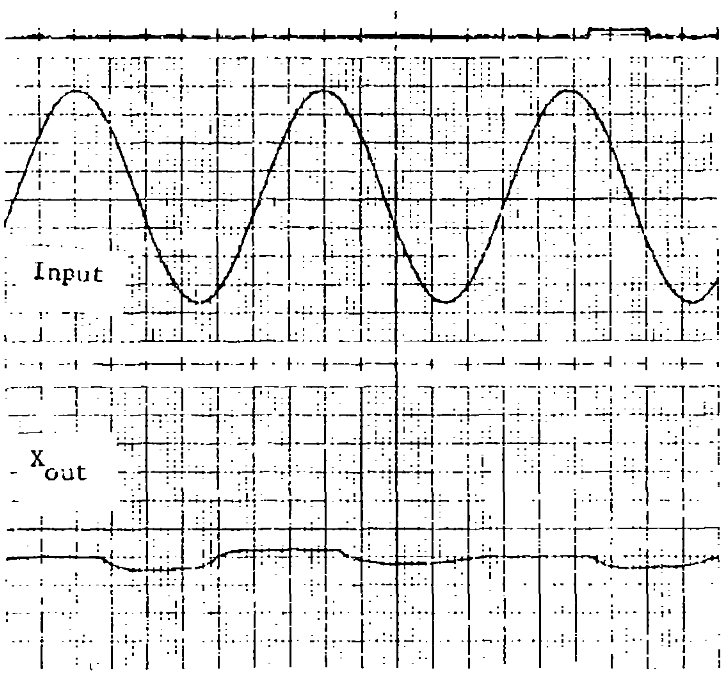
Input Level: 1%
Input Freq.: 2.0 Hz
Load Cond. : B
Bias Cond. : 30%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div

t →



Input level: 1%
Input Freq.: 3.0 Hz
Load Cond. : B
Bias Cond. : 30%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div

Figure 104. Output Fidelity @ 2 Hz & 3 Hz @ 1% Input - Load B - 30% Bias

93 through 95 are representative of the output fidelity of the system with no load and with the active channels sychnronized together. Note that at 3 Hz, there is very little distortion and the output closely resembles the input sinusoidal wave.orm.

Figures 96 through 98 with an input offset bias of 10% show output distortion similar or less than that with 0% bias offset. The distortion at frequencies from 0.1 Hz to 1 Hz appears less with a 10% bias offset than with 0 % bias offset. The distortion at 2 and 3 Hz appears slightly worse with the 10% bias offset. However, it appears that a bias offset of 10% does not greatly increase the output distortion at the 1% output level.

Figures 99 through 101 show the effect of an input offset bias of 20%. The effect (as compared to the output with 0% bias offset) of the 20% bias offset is quite noticeable. The output at all frequencies shows "flat topping". At 2 and 3 Hz frequencies, the amplitude of the output is noticeable attenuated compared to the 0% offset bias output (reference Figure 95).

Figures 102 through 104 show a severely distorted output at all frequencies from 0.1 Hz to 3 Hz. The effect of the 30% bias offset is greatest at the higher frequencies with the output motion at 3 Hz almost disappearing.

From the observed effects of the offset bias on the system output for the lightly loaded conditions shown on Figures 93 through 104, it appears that bias offsets greater than 10% have a significant negative effect on the output of the system at small sign.l input levels.

<u>Output Fidelity vs Offset Bias - Asymmetrical Load C - 1% Input</u>

Figures 105 through 116 show the effect of input offset bias on the output of the system with load C applied and the input level at 1% of the maximum commanded input. As opposed to load B at 1% output deflection which provides little load on the actuator output, load C provides a significant offset load (4,675 lbs) towards midstroke position of the test actuator. These figures illustrate the ability of the test system to respond to a dynamic input signal over the design bandpass of the system.

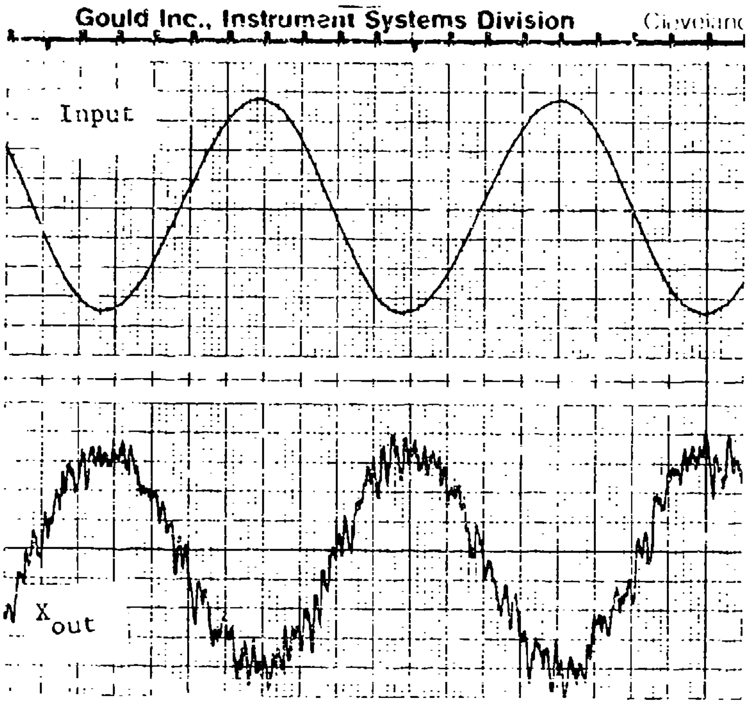
Figures 105 through 107 show the output waveform at 0% bias offset. Figure 105 and 106 show noticeable distortion of the output at frequencies of 0.1 Hz, 0.25 Hz, 0.5 Hz and 1 Hz. The distortion amplitude is nominally 13% of the output amplitude and is greater than that observed with load B at the same offset bias (reference Figures 93 and 94). At 2 Hz and 3 Hz as shown on Figure 107, the amount of distortion is reduced and the output appears only slightly distorted.

Figures 108 through 110 show the output waveform at 10% bias offset. As opposed to load B 10% bias offset results which showed little effect with a 10% offset bias, the output waveform with load C and a 10% bias offset degrades noticeably (compared to the 0% offset bias). The degradation is primarily a distortion of the sinusoid without a degradation of the output amplitude.

Figures 111 through 113 show the output waveform at 20% bias o fset. The amplitude of the distortion components is nominally 20% of the output amplitude, and increases from that observed with lower input biases. The 0.1 Hz through 1 Hz input frequencies show a similar "ragged" modulation of the output

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 1% Input



Date Prepared 10/20/83

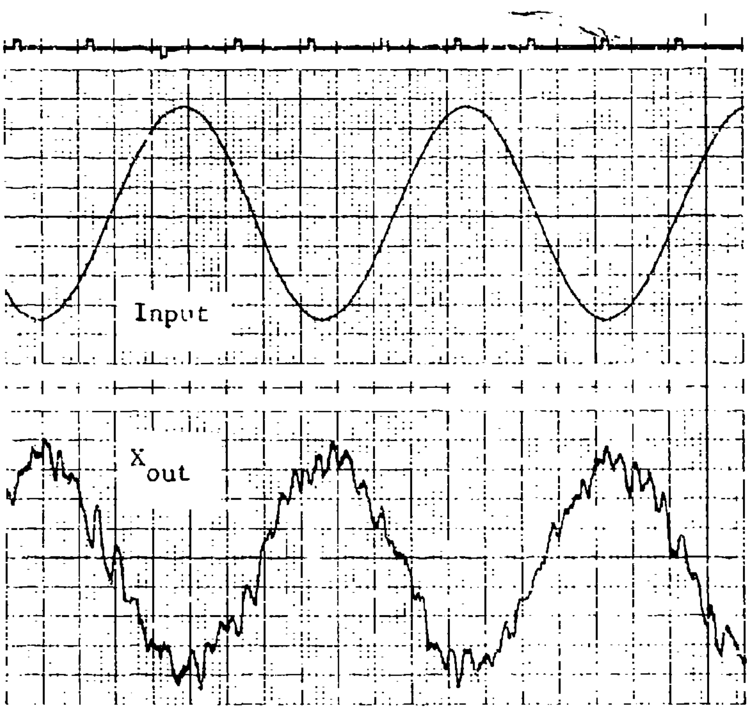
Input Level: 1%
Input Freq.: 0.1 Hz
Load Cond. : C
Bias Cond. : 0%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div

t →



Input Level: 1%
Input Freq.: 0.25 Hz
Load Cond. : C
Bias Cond. : 0%

Scale:

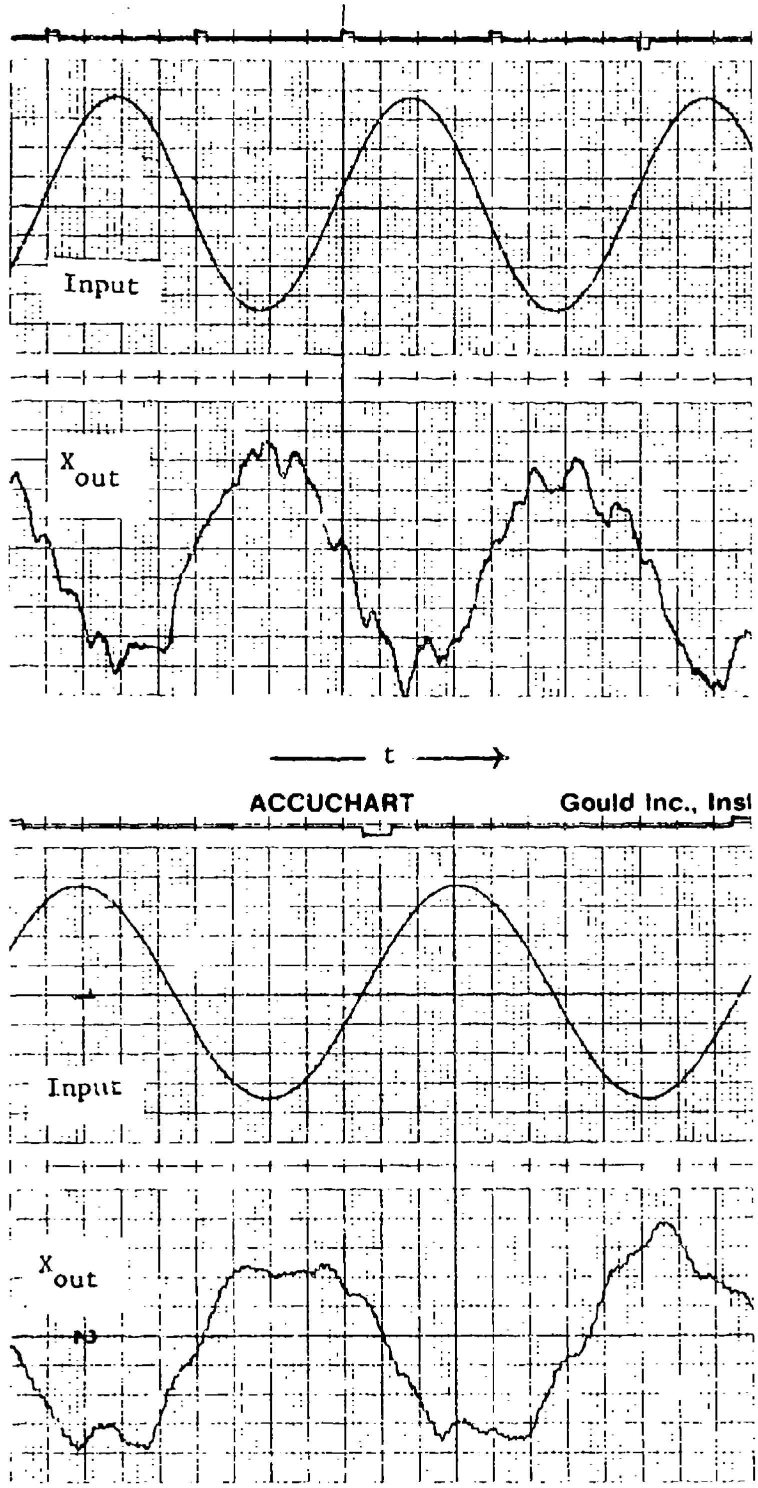
Input - 0.005 v/div

X_{out} - 0.00093 in/div

Figure 105. Output Fidelity @ 0.1 Hz & 0.25 Hz @ 1% Input - Load C - 0% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 1% Input



Date Prepared 10/18/83

Input Level: 1%
Input Freq.: 0.1 Hz
Load Cond. : C
Bias Cond. : 0%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div

Input Level: 1%
Input Freq.: 1.0 Hz
Load Cond. : C
Bias Cond. : 0%

Scale:

Input - 0.005 v/div

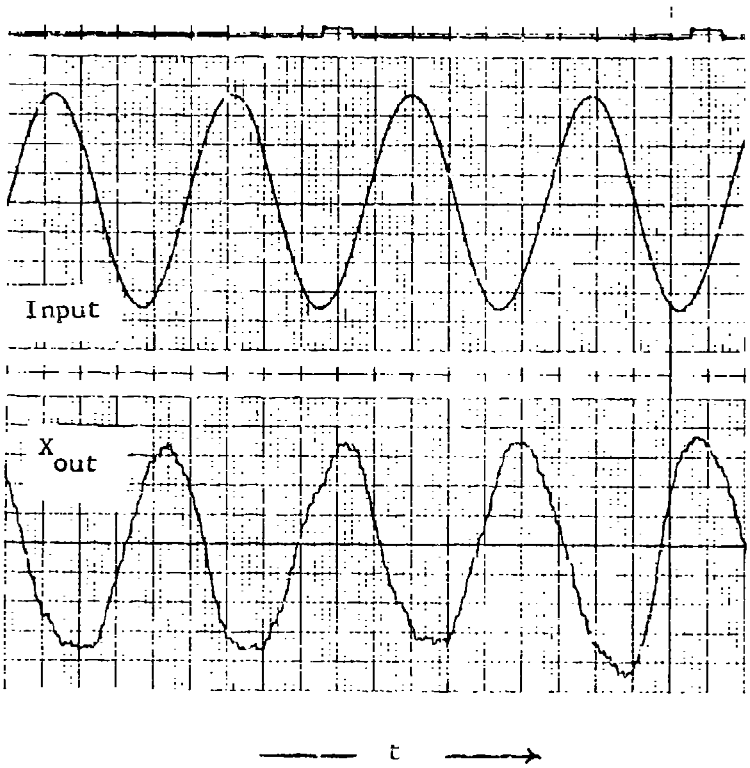
X_{out} - 0.00093 in/div

Figure 106. Output Fidelity @ 0.5 Hz & 1 Hz @ 1% Input - Load C - 0% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 1% Input

Date Prepared 10/18/83

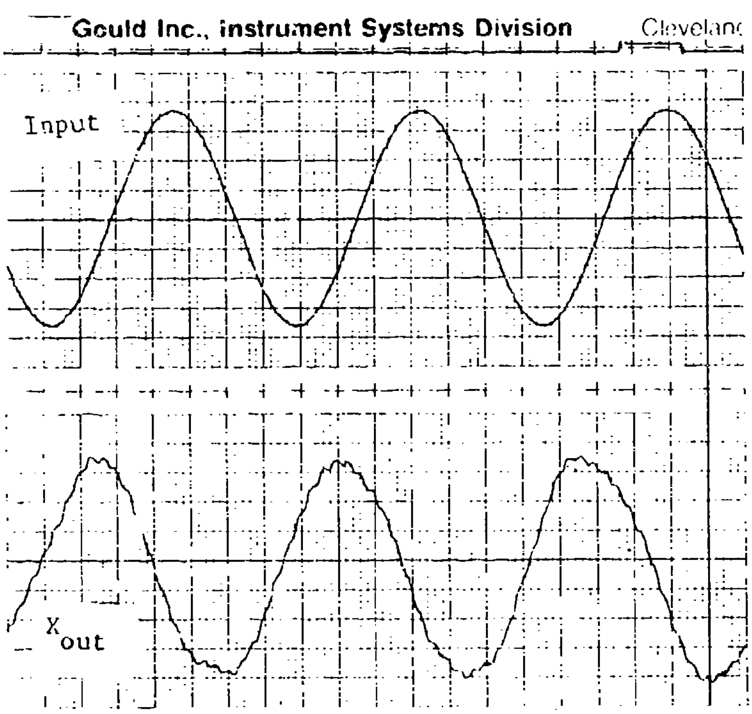


Input Level: 1%
Input Freq.: 2.0 Hz
Load Cond. : C
Bias Cond. : 0%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div



Input Level: 1%
Input Freq.: 3.0 Hz
Load Cond. : C
Bias Cond. : 0%

Scale:

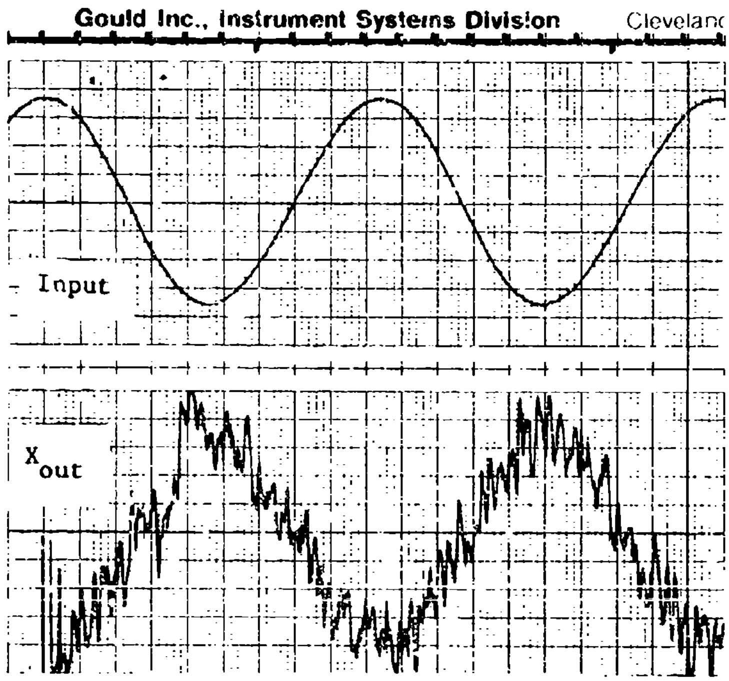
Input - 0.005 v/div

X_{out} - 0.00093 in/div

Figure 107. Output Fidelity @ 2 Hz & 3 Hz @ 1% Input - Load C - 0% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 1% Input



Date Prepared 10/18/83

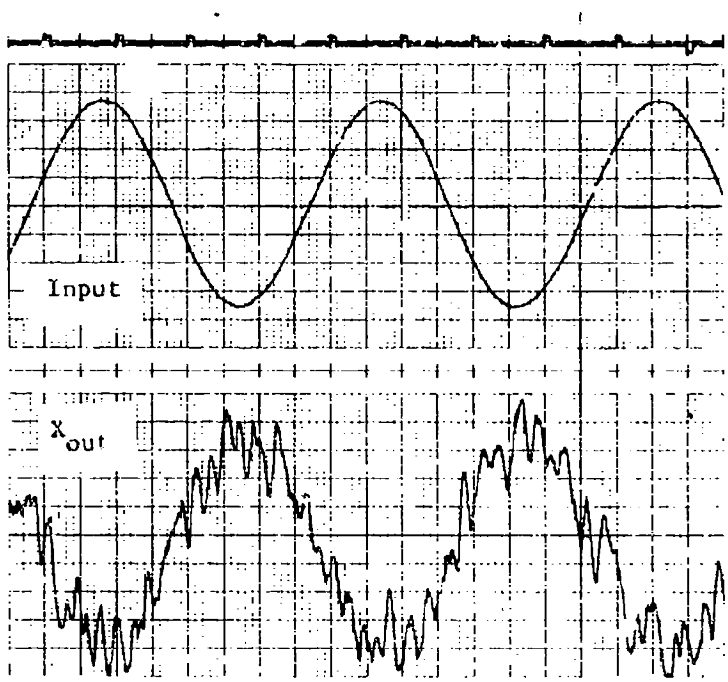
Input Level: 1%
Input Freq.: 0.1 Hz
Load Cond. : C
Bias Cond. : 10%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div

— t →



Input Level: 1%
Input Freq.: 0.25 Hz
Load Cond. : C
Bias Cond. : 10%

Scale:

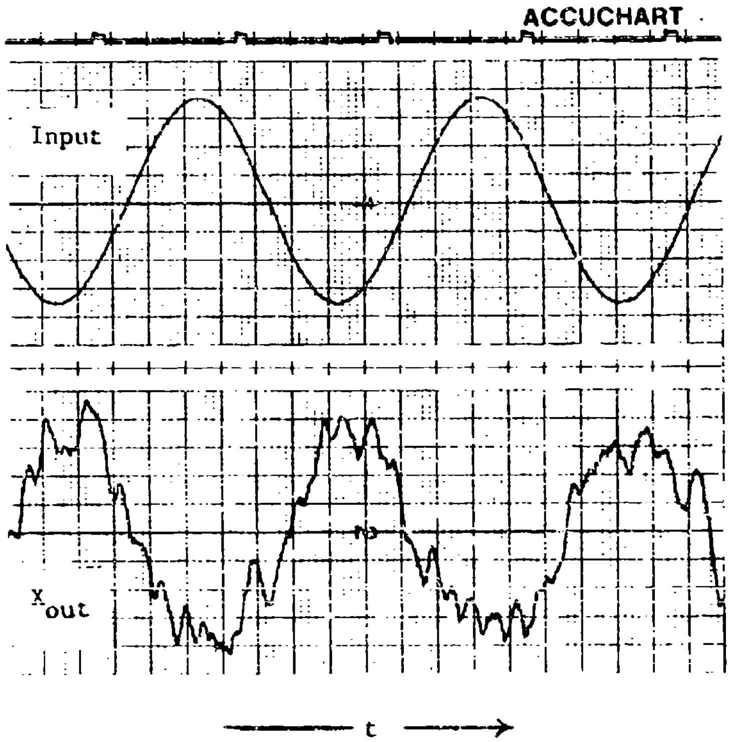
Input - 0.005 v/div

X_{out} - 0.00093 in/div

Figure 108. Output Fidelity @ 0.1 Hz & 0.25 Hz @ 1% Input - Load C - 10% Bias

TEST ITEM - Boeing, Reconfigurable Fail Operative
Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset
Load - 1% Input



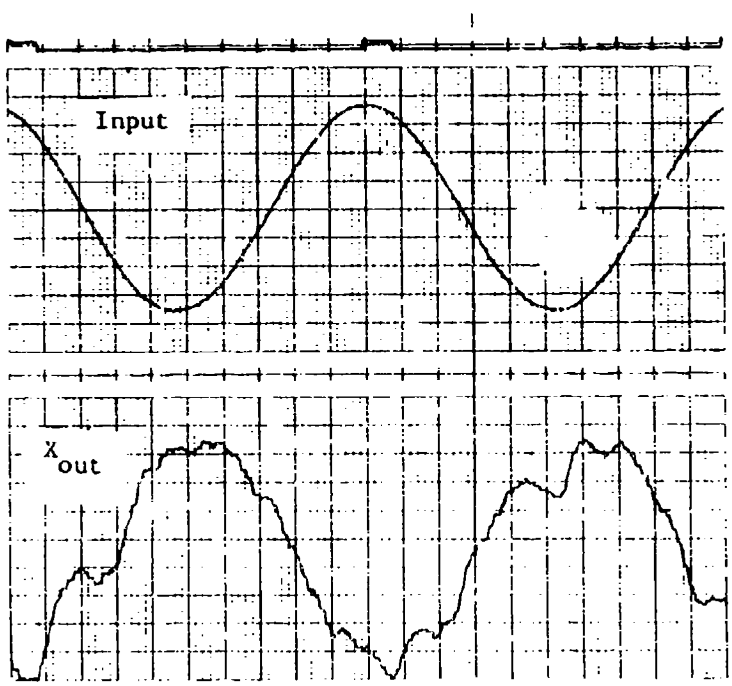
Date Prepared 10/18/83

Input Level: 1%
Input Freq.: 0.5 Hz
Load Cond. : C
Bias Cond. : 10%

Scale:

Input 0.005 v/div

X_{out} - 0.00093 in/div



Input Level: 1%
Input Freq.: 1.0 Hz
Load Cond. : C
Bias Cond. : 10%

Scale:

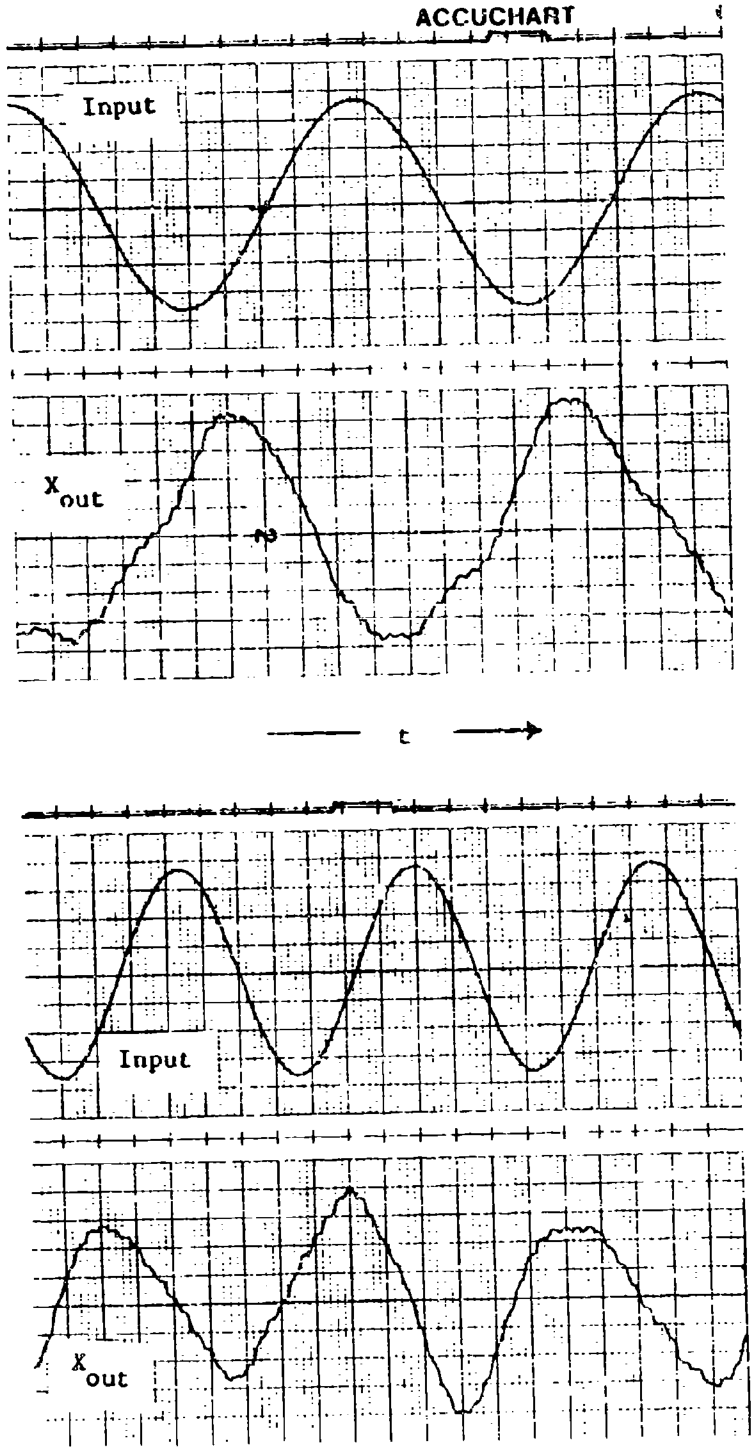
Input - 0.005 v/div

X_{out} - 0.00093 in/div

Figure 109. Output Fidelity @ 0.5 Hz & 1 Hz @ 1% Input - Load C - 10% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative
Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 1% Input



Date Prepared 10/18/83

Input Level: 1%
Input Freq.: 2.0 Hz
Load Cond. : C
Bias Cond. : 10%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div

Input Level: 1%
Input Freq.: 3.0 Hz
Load Cond. : C
Bias Cond. : 10%

Scale:

Input - 0.005 v/div

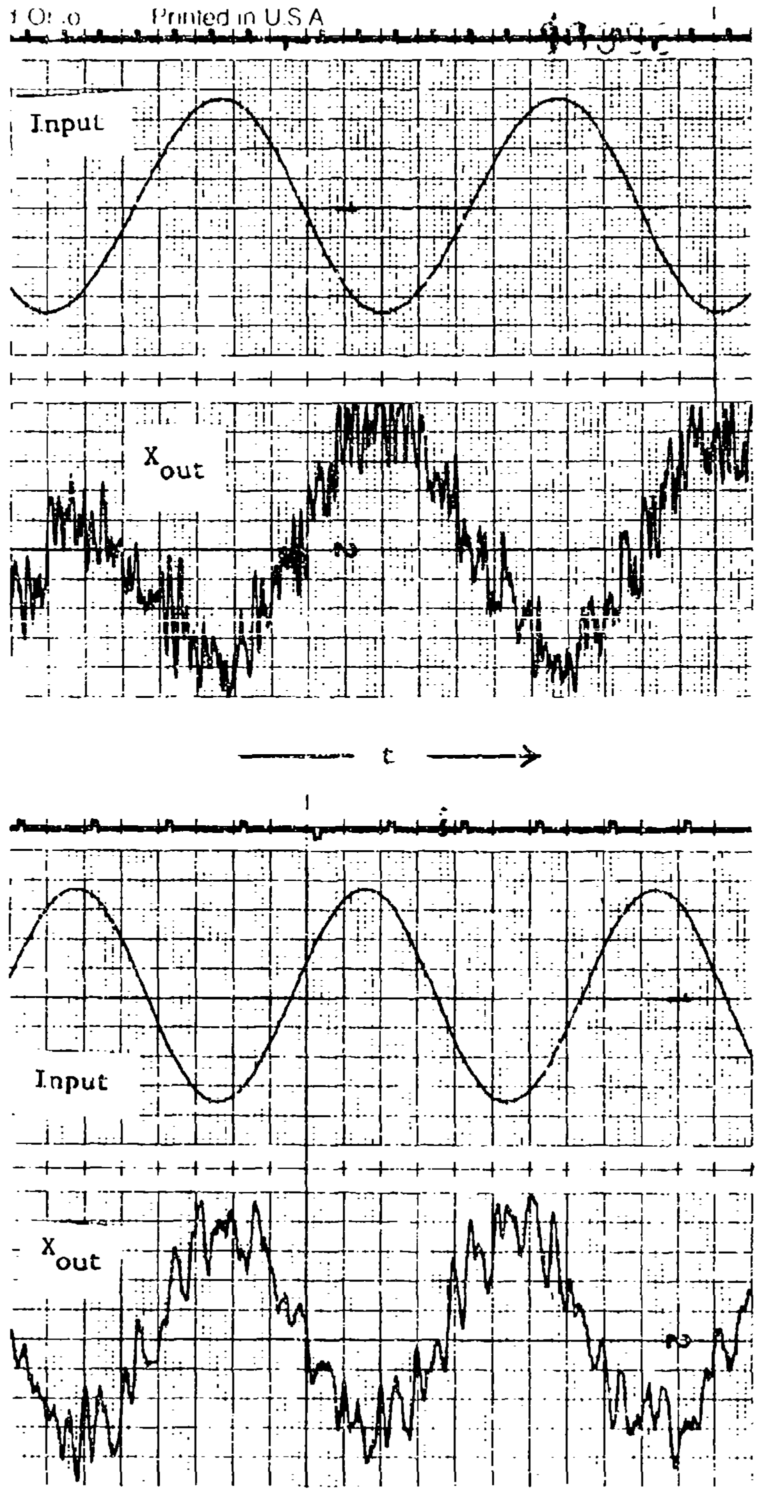
X_{out} - 0.00093 in/div

Figure 110. Output Fidelity @ 2 Hz & 3 Hz @ 1% Input - Load C - 10% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 1% Input

Date Prepared 10/18/83



Input Level: 1%
Input Freq.: 0.1 Hz
Load Cond. : C
Bias Cond. : 20%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div

Input Level: 1%
Input Freq.: 0.25 Hz
Load Cond. : C
Bias Cond. : 20%

Scale:

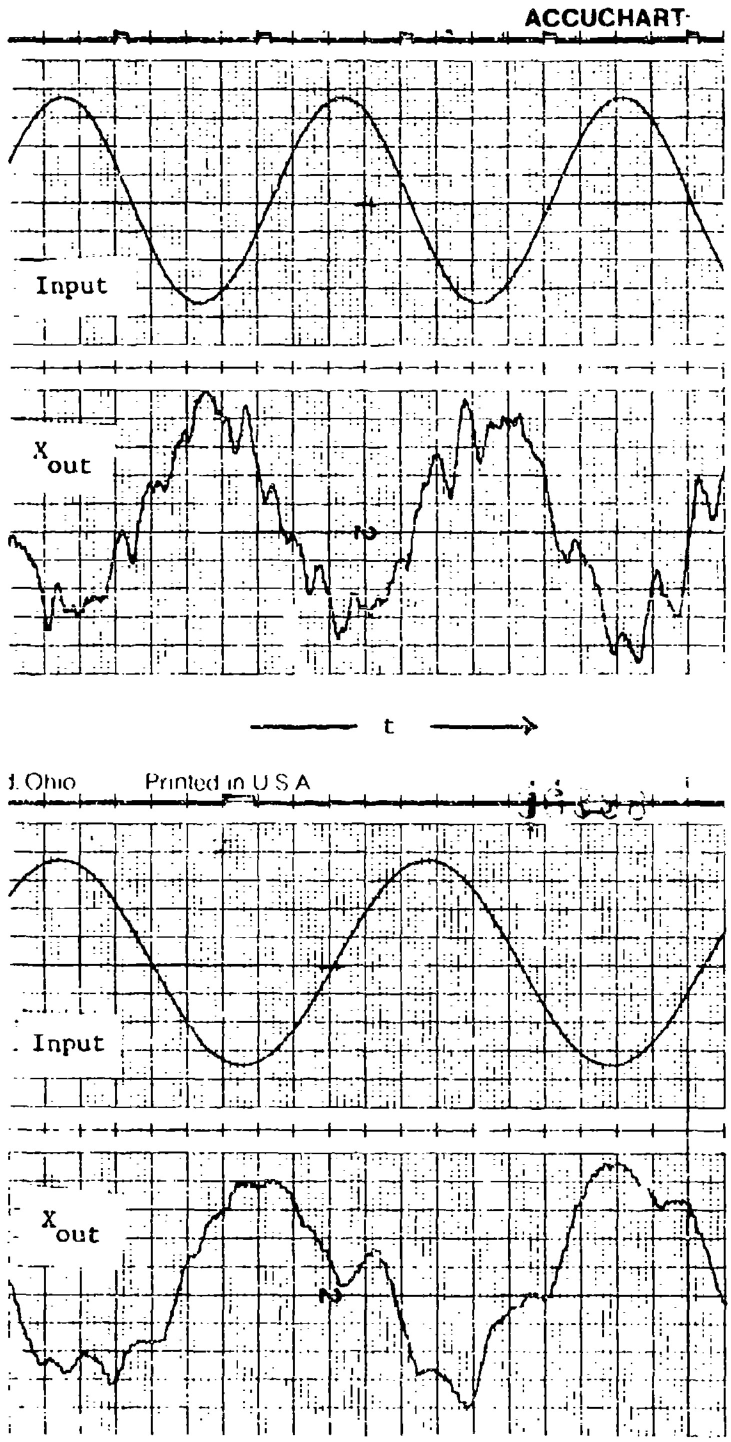
Input - 0.005 v/div

X_{out} - 0.00093 in/div

Figure 111. Output Fidelity @ 0.1 Hz & 0.25 Hz @ 1% Input - Load C - 20% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 1% Input



Date Prepared 10/18/83

Input Level: 1%
Input Freq.: 0.5 Hz
Load Cond. : C
Bias Cond. : 20%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div

Input Level: 1%
Input Freq.: 1.0 Hz
Load Cond. : C
Bias Cond. : 20%

Scale:

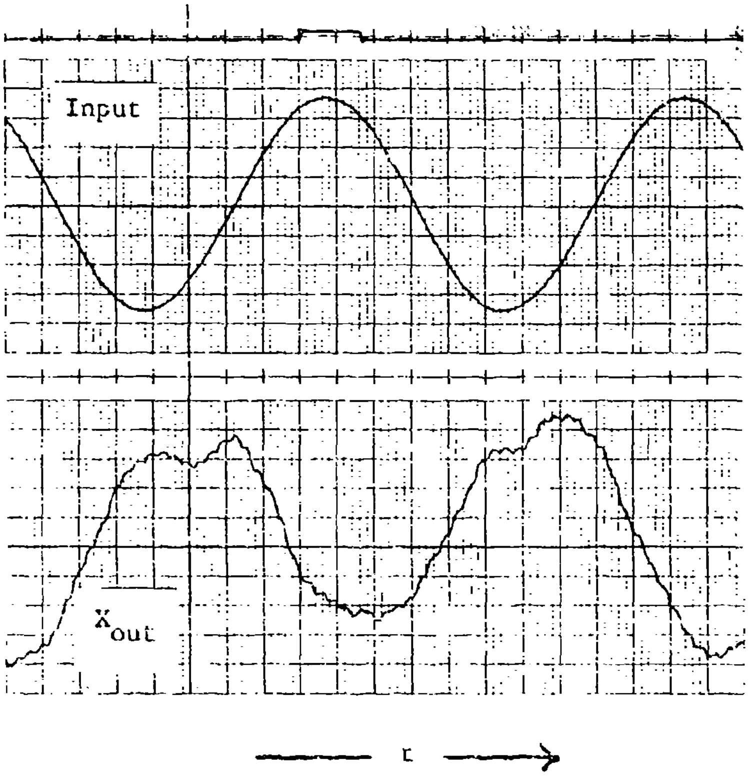
Input - 0.005 v/div

X_{out} - 0.00093 in/div

Figure 112. Output Fidelity @ 0.5 Hz & 1 Hz @ 1% Input - Load C - 20% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 1% Input



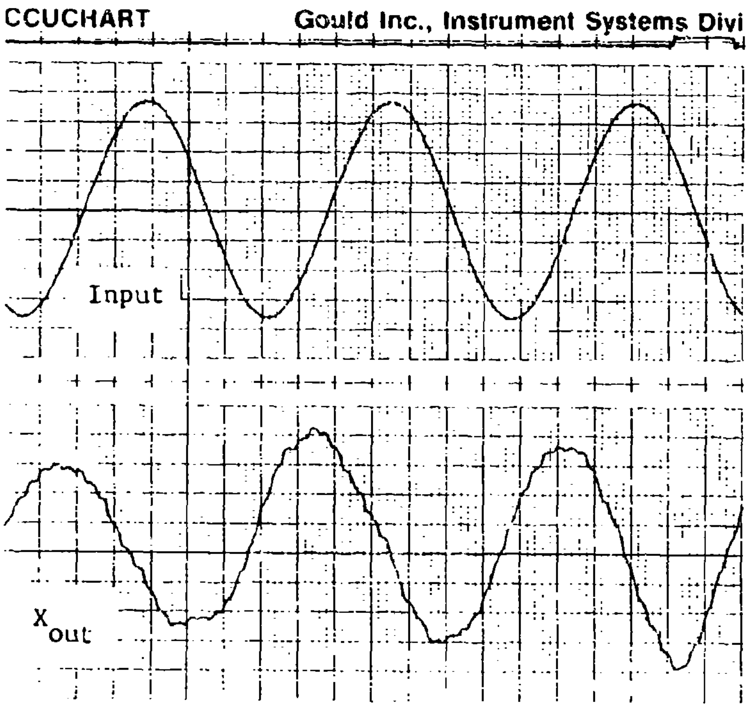
Date Prepared 10/18/83

Input Level: 1%
Input Freq.: 2.0 Hz
Load Cond. : C
Bias Cond. : 20%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div



Input Level: 1%
Input Freq.: 3.0 Hz
Load Cond. : C
Bias Cond. : 20%

Scale:

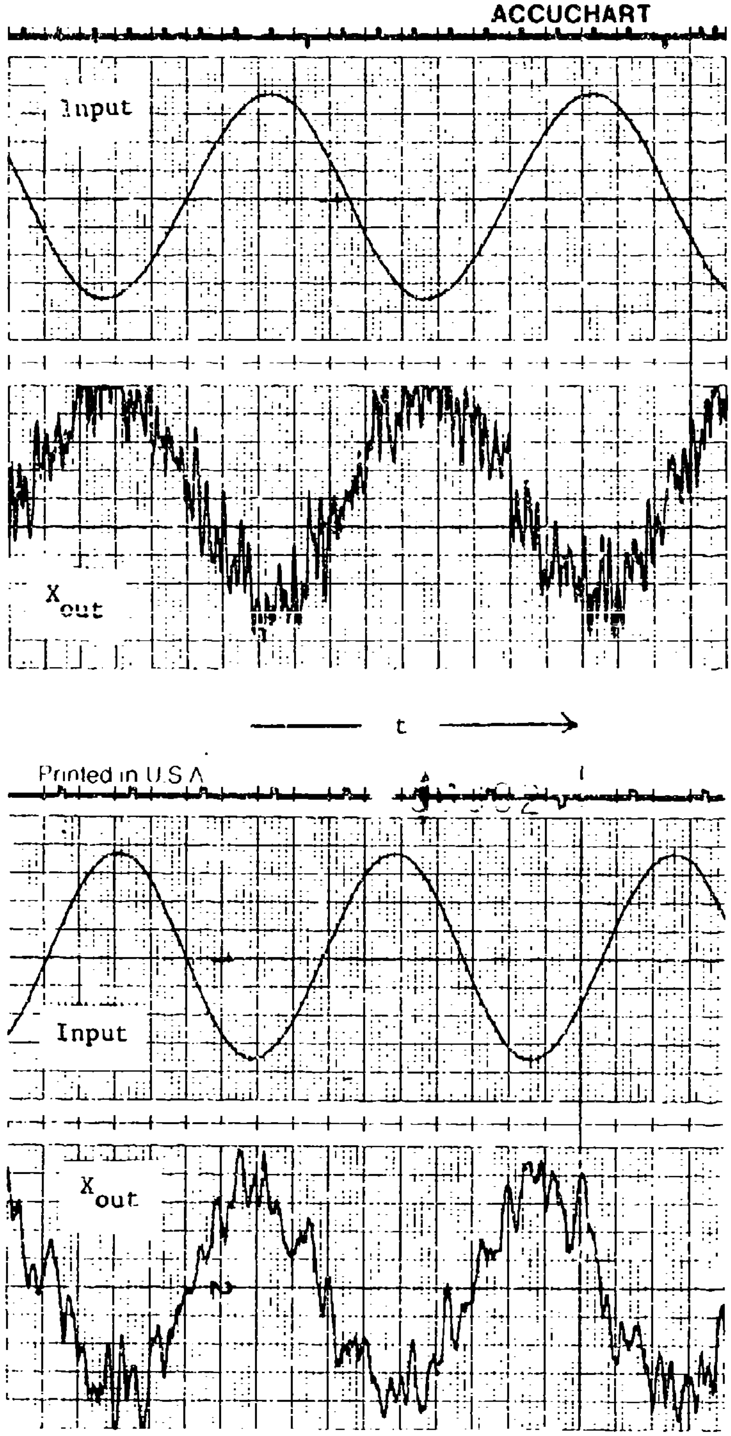
Input - 0.005 v/div

X_{out} - 0.00093 in/div

Figure 113. Output Fidelity @ 2 Hz & 3 Hz @ 1% Input - Load C - 20% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuater

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 1% Input



Date Prepared 10/18/83

Input Level: 1%
Input Freq.: 0.1 Hz
Load Cond. : C
Bias Cond. : 30%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div

Input Level: 1%
Input Freq.: 0.25 Hz
Load Cond. : C
Bias Cond. : 30%

Scale:

Input - 0.005 v/div

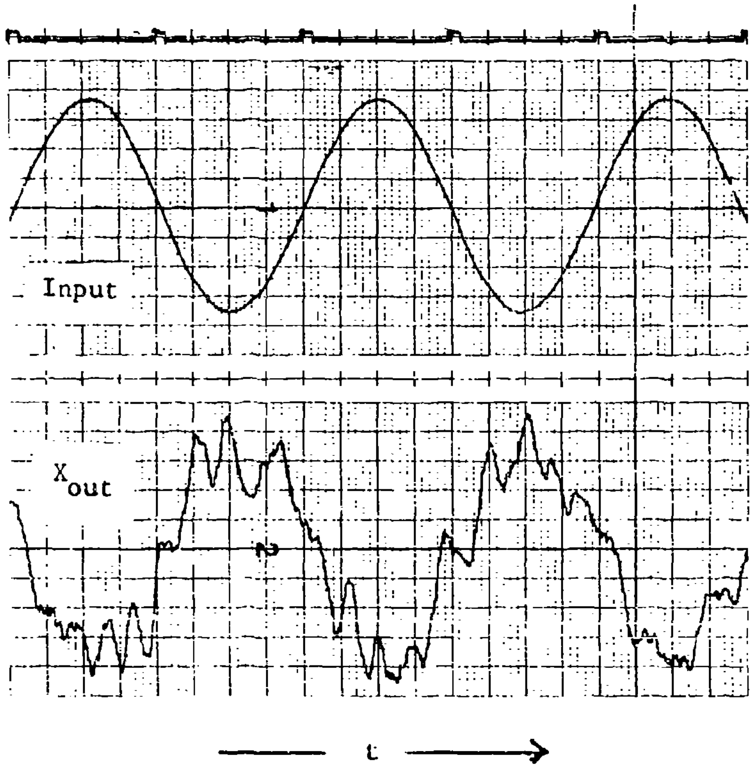
X_{out} - 0.00093 in/div

Figure 114. Output Fidelity @ 0.1 Hz & 0.25 Hz @ 1% Input - Load C - 30% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 1% Input

Date Prepared 10/18/83

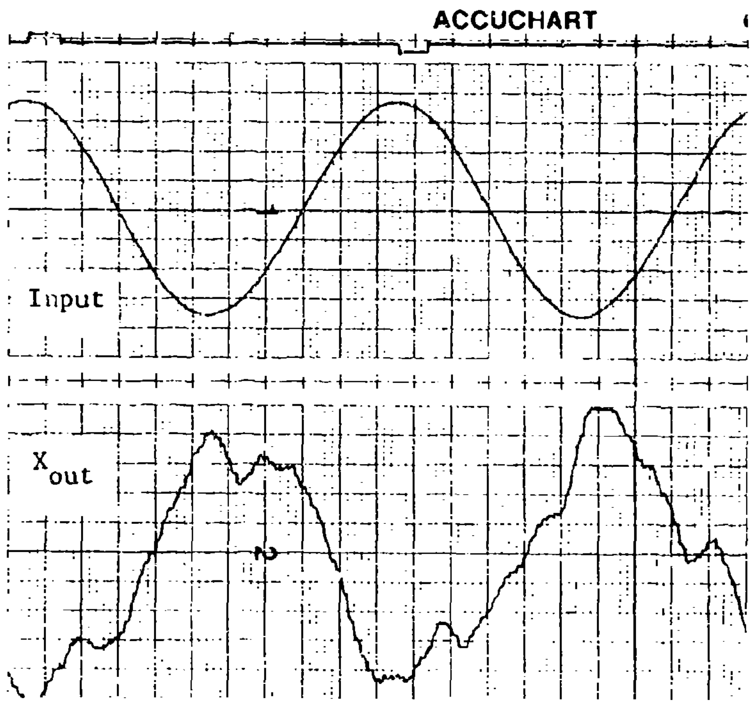


Input Level: 1%
Input Freq.: 0.5 Hz
Load Cond. : C
Bias Cond. : 30%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div



Input Level: 1%
Input Freq.: 1.0 Hz
Load Cond. : C
Bias Cond. : 30%

Scale:

Input - 0.005 v/div

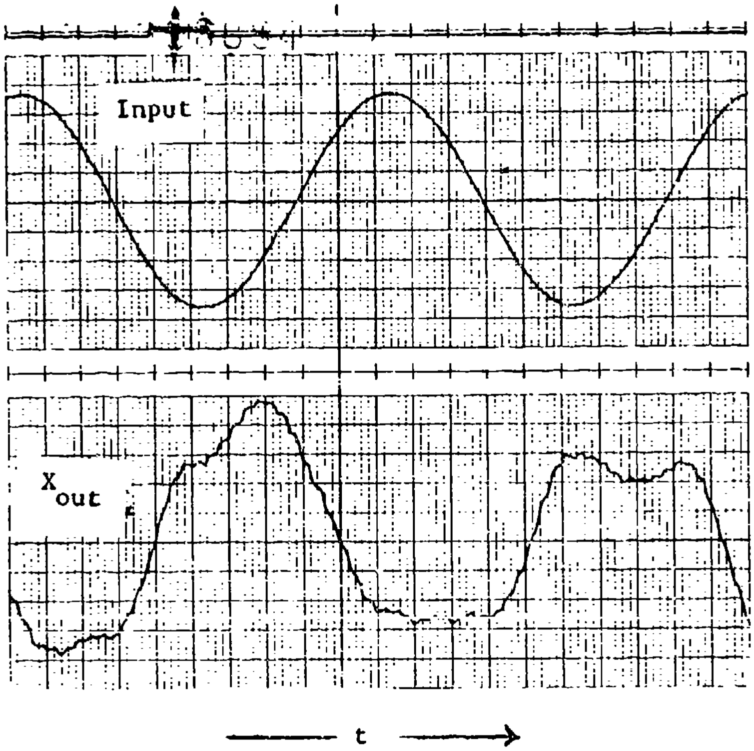
X_{out} - 0.00093 in/div

Figure 115. Output Fidelity @ 0.5 Hz & 1 Hz @ 1% Input - Load C - 30% Bias

TEST ITEM - Boeing Reconfigurable Fail Operative Fly-By-Wire Servoactuator

TEST - Output Fidelity - As a Function of Channel Offset Bias - Offset Load - 1% Input

Date Prepared 10/18/83

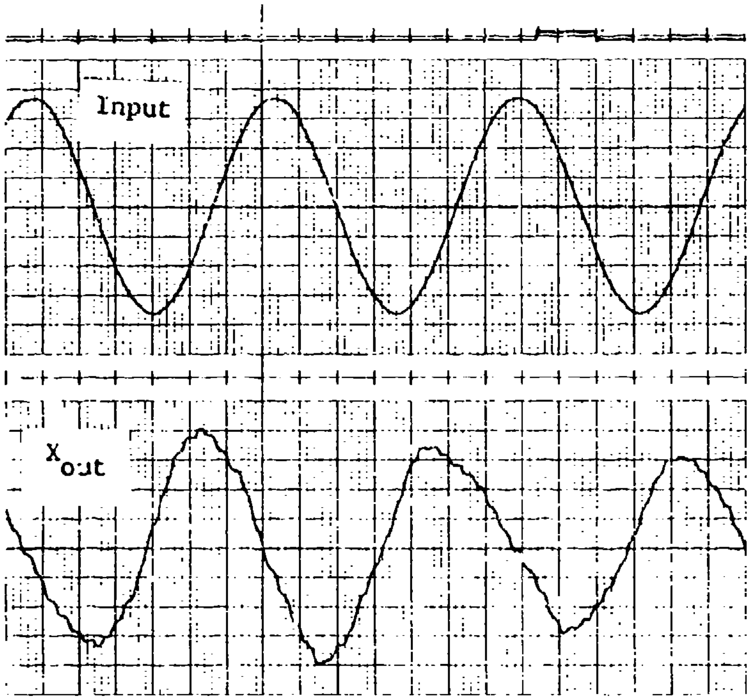


Input Level: 1%
Input Freq.: 2.0 Hz
Load Cond. : C
Bias Cond. : 30%

Scale:

Input - 0.005 v div

X_{out} - 0.00093 in/div



Input Level: 1%
Input Freq.: 3.0 Hz
Load Cond. : C
Bias Cond. : 30%

Scale:

Input - 0.005 v/div

X_{out} - 0.00093 in/div

Figure 116. Output Fidelity @ 2 Hz & 3 Hz @ 1% Input - Load C - 30% Bias

fundamental frequency sinusoid. However, at 2 Hz and 3 Hz as shown on Figure 113, the distortion is primarly a low frequency modulation of the output at a low amplitude.

Figures 114 through 116 show the output waveform at a 30% bias offset. The distortion of the output is similar to that with a 20% bias offset with the amplitude of the distortion increased. However, while with load B the output for the 30% bias offset was severely attenuated, output amplitude with load C and the same bias remains relatively constant for all the input freqencies.

From the preceding Figures 105 through 116, it appears that the bias load of load C reduces the adverse effects of bias offsets in the control channels. However, the output fidelity is best with 0% offset bias (where the channels are tracking together).

SECTION VI. CONSIDERATIONS AND ANOMOLIES

DYNAMIC FAILURE DETECTION CHARACTERISTICS AND ANALYSIS

Figure 117 shows the failure detection characteristics of the Boeing microprocessor controlled actuator for sinusoidal failure inputs. The three curves presented on Figure 117 correspond to the number of samples allowed with an error above the failure detection threshold before a failure is declared. The figure shows that the detection level is frequency dependent. For example, with the 12 iteration setting the failure logic detection level is maintained at the static failure detection only out to 1.5 Hz. Above this frequency, the amplitude required for an input failure declaration rises to more than 10 times the low frequency level. For the 6 iteration curve the failure detection amplitude remains relatively constant from 0 to 3.5 Hz. The 3 iteration failure detection curve maintains a constant amplitude up to a 10 Hz frequency.

The inability of the detection logic to maintain a uniform detection level is a direct result of the time interval before a failure is allowed to be declared. This is quite apparent for the 12 iteration case where the sample time during which a failure must be above the failure detection threshold is 0.192 second (corresponding to a 60 Hz sampling frequency). At an input frequency of 3 Hz, the input signal amplitude passes through zero every 0.165 second. This is always true at this frequency independent of amplitude. The 0.165 second is less than the 0.192 second required for a failure declaration. A corresponding point of failure detection failure occurs on the 6 iteration curve at 5 Hz. At 5 Hz the sinusoidal input passes through zero every 0.1 second. This is very close to the 0.096 second failure declaration time delay. At this input frequency, the amplitude of input signal for a failure declaration rises to more than 6 times the low frequency detection level. This same problem at 10 Hz can be predicted for the 3 iteration failure declaration time delay of 0.048 second. A 10 Hz sinusoidal signal passes through a zero amplitude every 0.05 second.

Although failure detection methods must tolerate input transients without nuisance failure declaration, the sample time delay threshold mechanization has the demonstrated weakness. For any selected time delay, there is a sinusoidal input frequency above which failure detection is not well maintained. A better approach to failure detection would be a method which looks at both the amplitude and duration (input time history) of the failure input. An actuator (or an aircraft) integrates a failure transient with the integration rate dependent on the amplitude of the transient. The output amplitude deviation is dependent on the input transient duration. A detection scheme that looks at the product of amplitude and duration and time (really a measurement of the area under the failure amplitude time history curve) would be a realistic failure detection method.

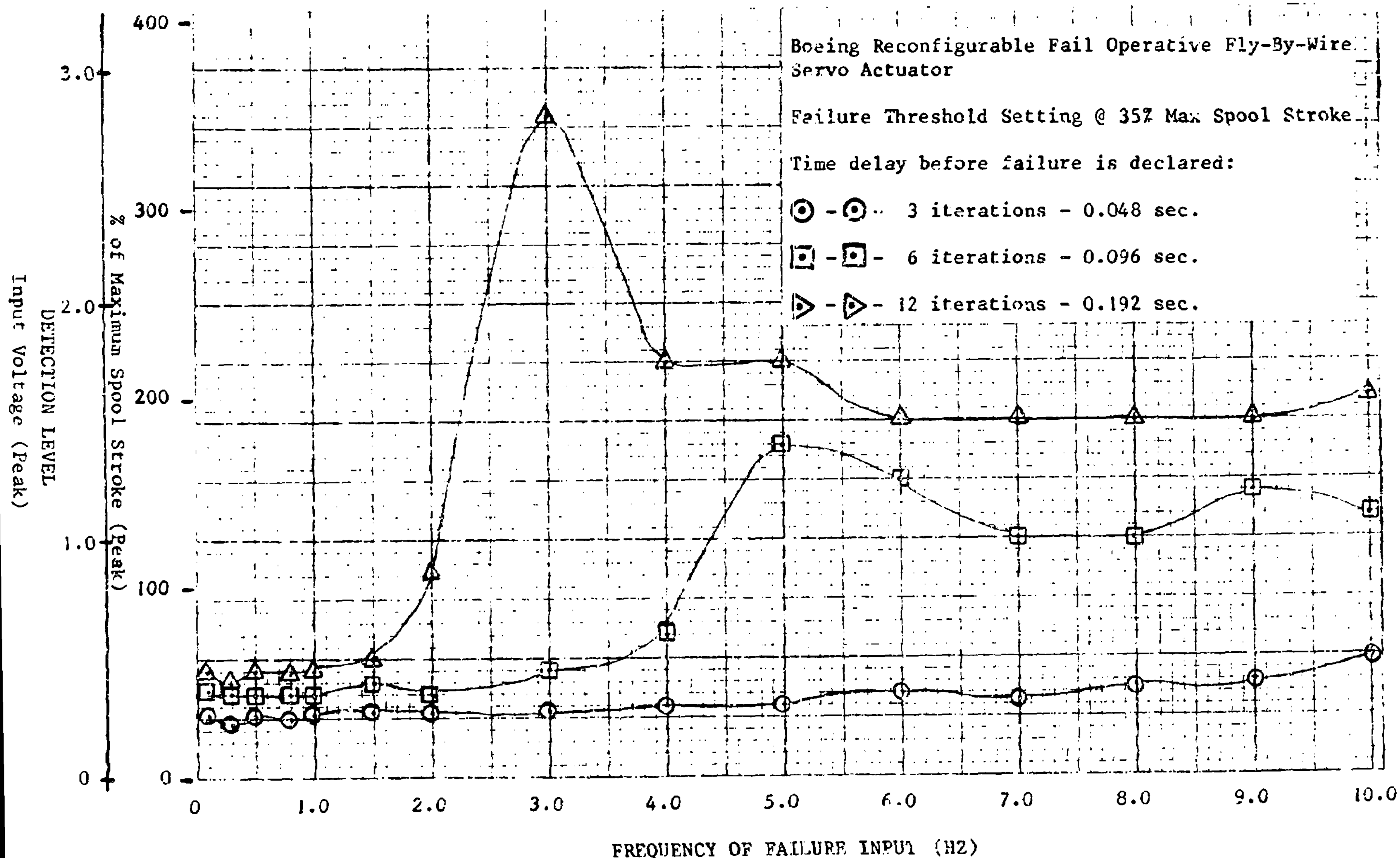


Figure 117. Dynamic Failure Detection - Frequency Effect - Condition 1

A PROBLEM WITH SAMPLING AND REUSING FAILED CHANNELS

The test system's failure detection mechanization uses a combination of amplitude threshold and persistence criteria for declaring the failure of a faulty channel (and initiating reconfiguration). The criteria require the failure detection amplitude to be exceeded a specified number of consecutive samples before a failure is declared. In a similar manner, once a channel has been declared to be failed, it will continue to be sampled and will be declared good again if no failure is detected for a given number of consecutive samples. More samples are normally specified to remove the failure declaration than to make it.

There is a problem with the approach of sampling failed channels and reusing them if they appear good. The failure logic, with a particular sequence of input or channel failures, can be "fooled" into not correctly detecting input failures. Such a sequence of failures, as illustrated on Figure 118, is not improbable in the normal operation of a flight control system. The sequence which fools the system is one in which a slowly changing input signal (varying from + to - voltage and back) is sequentially applied to the 4 inputs. Upon exceeding the failure threshold for the selected time (iterations), the channel with the failure input is declared failed. For a slowly varying failure input signal (for example a slow drift), the channel with the failure input can subsequently agree with the other channel inputs long enough for the channel to be declared "good" and used again in the voting logic. If another subsequent input failure which is similar to the first occurs with a second channel during the time where the previously failed (and uncorrected) channel appears good, there are now two channels with the same failure input and two channels with the correct input. The failure logic does not have the capability to correctly identify the failures and vote them out. The attempt to extend the failure tolerance capability of the system by monitoring failed channels for potential reuse has provided the failure logic with the same problem it would encounter with simultaneous "like" failures. The difference is that the simultaneous "like" failures require that the two failures occur within a very short time (the time for the specified failure vote iterations) while the slowly varying input failures can have an indefinitely long time (as long as the first failed input stays within failure difference amplitude window) in which they can occur and fool the system. Figure 118 illustrates the problem.

As shown in Figure 118, the same slowly varying ramp input is applied sequentially to channel 1, 2, 3 and 4 inputs. This is the same input failure test condition used in the unloaded failure transients measurements (reference Figure 24). However, for Figure 24, the failure status removal iterations were adjusted to the maximum value available (300) to prevent fooling the failure logic for the particular input ramp rate used for the test. Figure 118 shows the results of using the system's normal 9 iteration (or sample) criteria for the failure status change.

The ramp failure is first applied to channel 1. Note that the input voltage ramp slope is 0.625 volt/second. The input difference for the failure detection level equivalent to 35% the spool stroke is 0.266 volt. At the ramp slope used, it takes 0.425 second to change 0.266 volt. However, the time required to vote a failed channel good (with 9 iterations at 0.025 second/iteration) is 0.225 second. As the ramp passes through the 0 input level, the failure input amplitude is less than the failure detection threshold for 0.425 second, and is voted good after 0.225 second. A second similar

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Date Prepared 10/30/83

TEST - Failure Transients - Condition 22A Normal Setup (9 iterations to delcare a "corrected" failed channel "good")

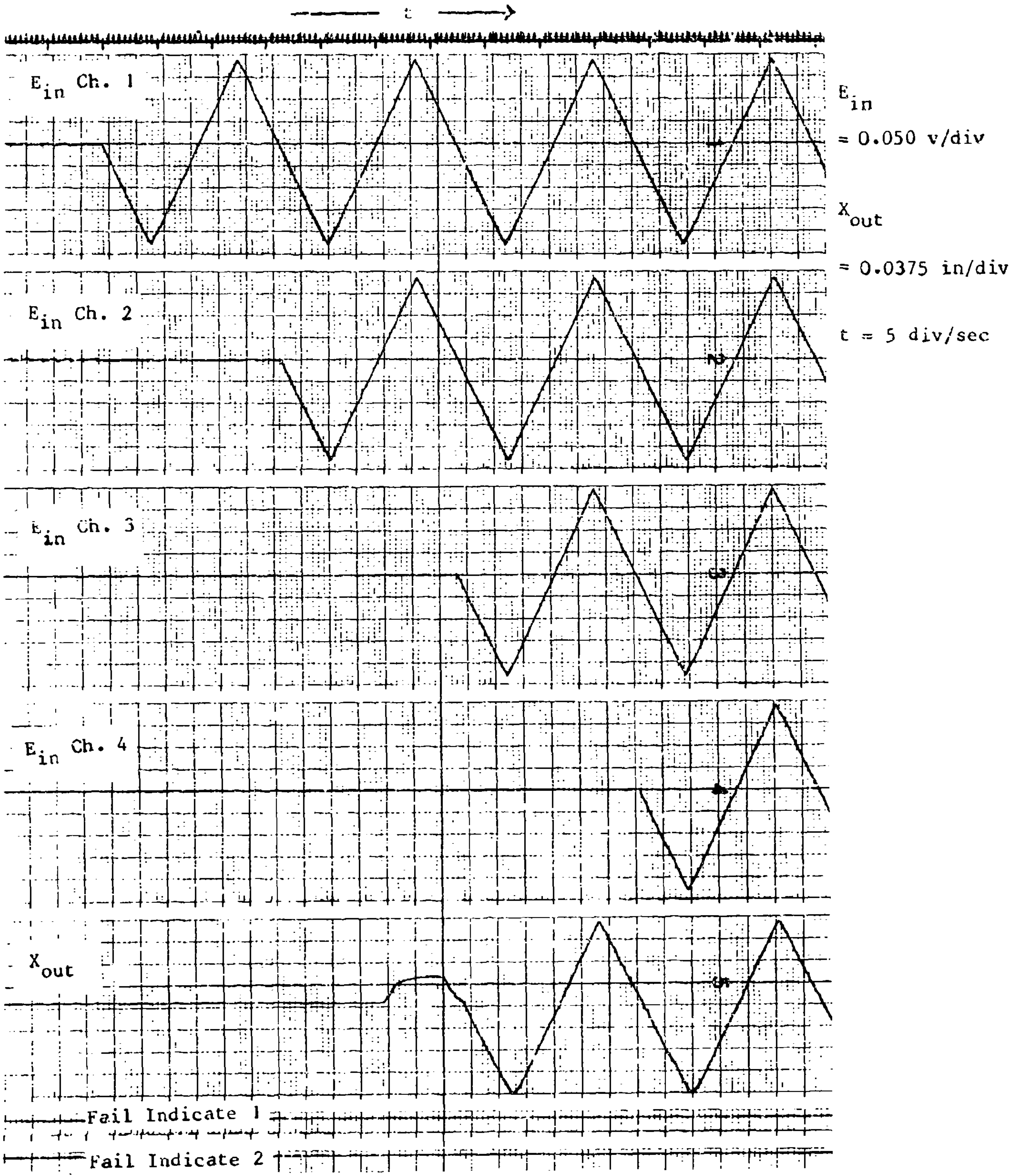

Figure 118. Failure Transients - With Normal Failure Status Removal

failure input to channel two is applied before the channel 1 input moves through the detection amplitude threshold (including the time for 3 iterations to be declared failed) and channel 1 and 2 both have the same inputs. After this second input failure, both sections of the actuator remain active, one section with a ramp input and one section with a null input. At this point, the actuator halves fight each other and there is little movement in reponse to the ramp. Note that in this condition, a normal input into channels 3 and 4 would not create a correct output motion of the test actuator because of the force fight. Upon a third failure into channel 3, the test system output tracks the failure inputs, rather than the correct null input.

The situation can be helped for a specific set of input signals by adjusting the number of iterations necessary for declaring a failed channel good. However, for any iteration number picked, another set of input signal failures can be selected which will still fool the failure logic. It is therefore recommended that the technique of reusing previously failed channels be deleted from the system.

PISTON SEAL DESIGN CONSIDERATIONS FOR THE TEST SYSTEM

During the initial unloaded testing of the test system, it was determined that the actuator used with the test system had defective piston seals. Disassembly of the actuator revealed that both halves of the actuator had pistons with seal retaining grooves whose walls had failed mechanically. The actuator rod and pistons were replaced prior to generation of the data presented in this document. Figures 119 through 122 show the failed actuator pistons after disassembly. Figure 119 shows the actuator rod assembly with both pistons. The actuator rod is made in two sections which is screwed together after insertion in th actuator body. As shown in this figure, the failure of one piston seal is quite apparent. Figures 120 and 121 are close up views of the failed piston seal, showing the mechanical failure of the metal wall retaining the Teflon seal rings. Although it is not as readily apparent, the second piston also had a similar mechanical failure. The crack at the bottom of the groove was sufficient to allow leakage, but had not resulted in a physical distortion of the retaining wall. This piston seal is shown in Figure 122. Note the smearing of the Teflon split ring seal across the face of the piston.

The actuator used with the test system had been designed and qualified for operation with a tandem flow control spool valve with the flow control edges ground to match. For this type of control valve, the force fight between the two sections is minimized and remains constant. The actuator pistons normally have small differental pressures most of the operating time.

With the test system, the flow control spools move independently. Since the valves are manufactured as high pressure gain valves (developing full system pressure at the output ports with small input currents into the valve), there is a normal force fight between actuator sections. The actuator pistons constantly being subjected to a differential pressure magnitude of full system pressure. In addition to a constant level force fight between the actuator sections due to the servovalve null conditions, the noise content of the control inputs to the servovalves cause changes in the output pressures of the servovalves. These pressure changes provide a constantly changing stress level in the seal ring retaining walls. The combination of a large magnitude and a constantly varying force fight creates both a seal life (as shown by the

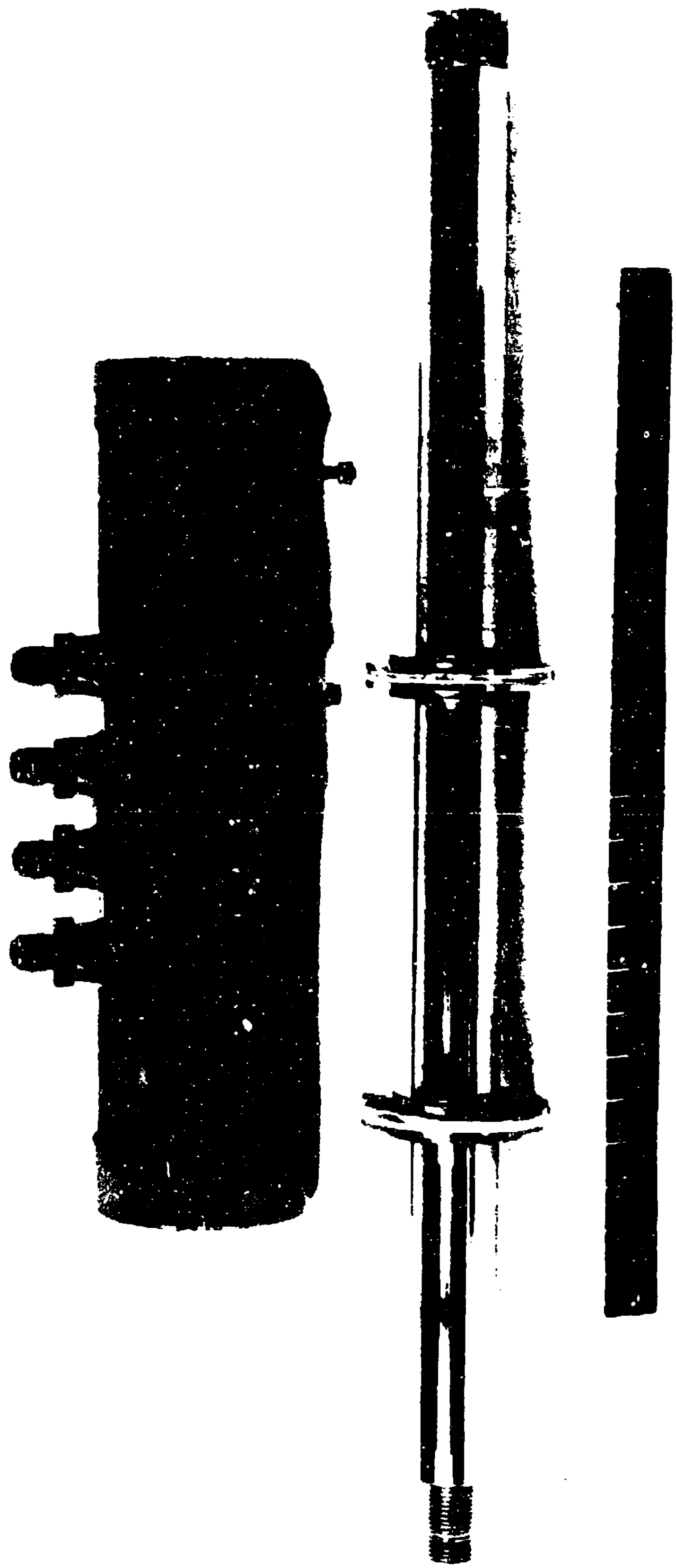

Figure 119. Actuator Rod & Piston Assembly with Body

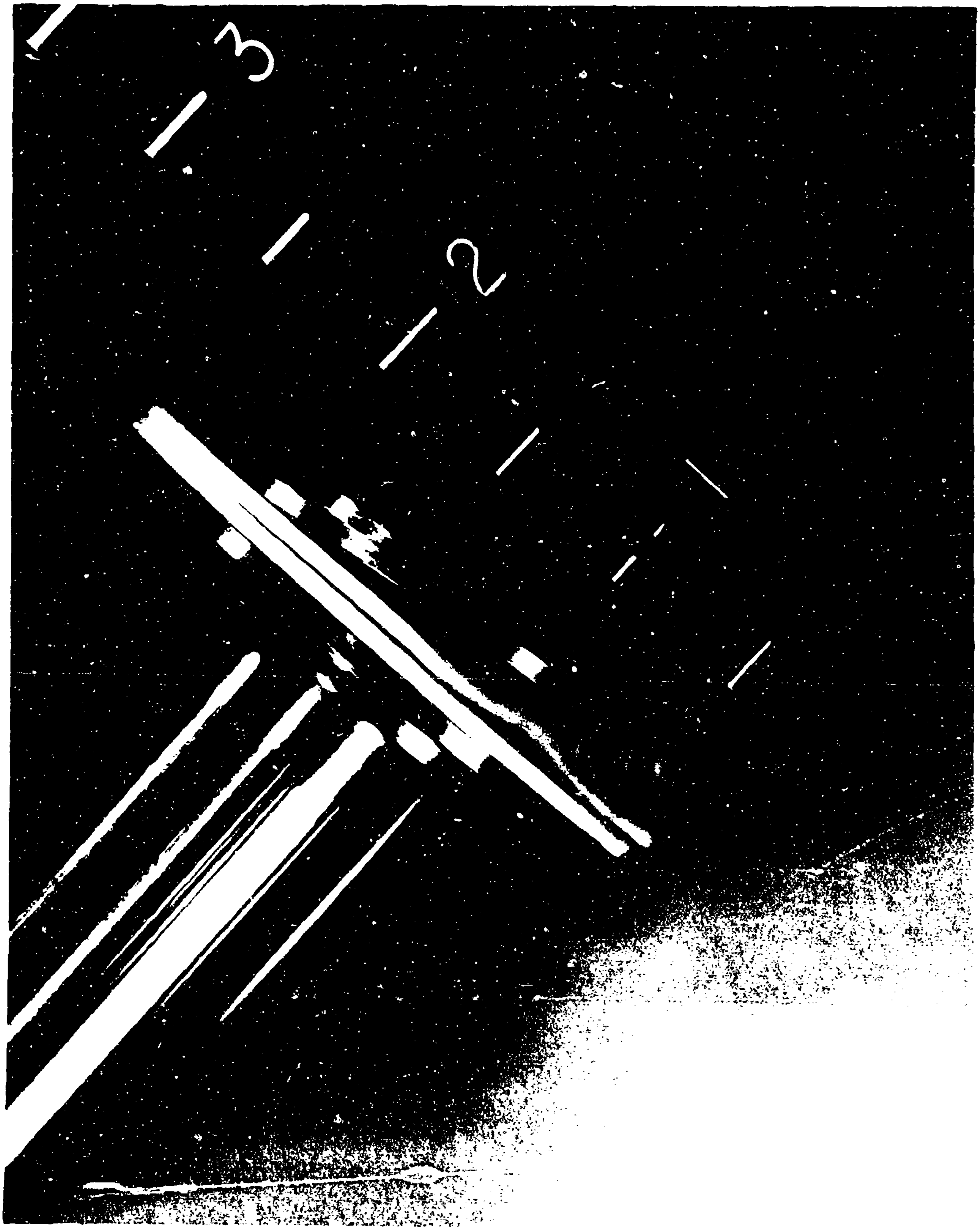

Figure 120. Failed Piston Seal 1 - Side View

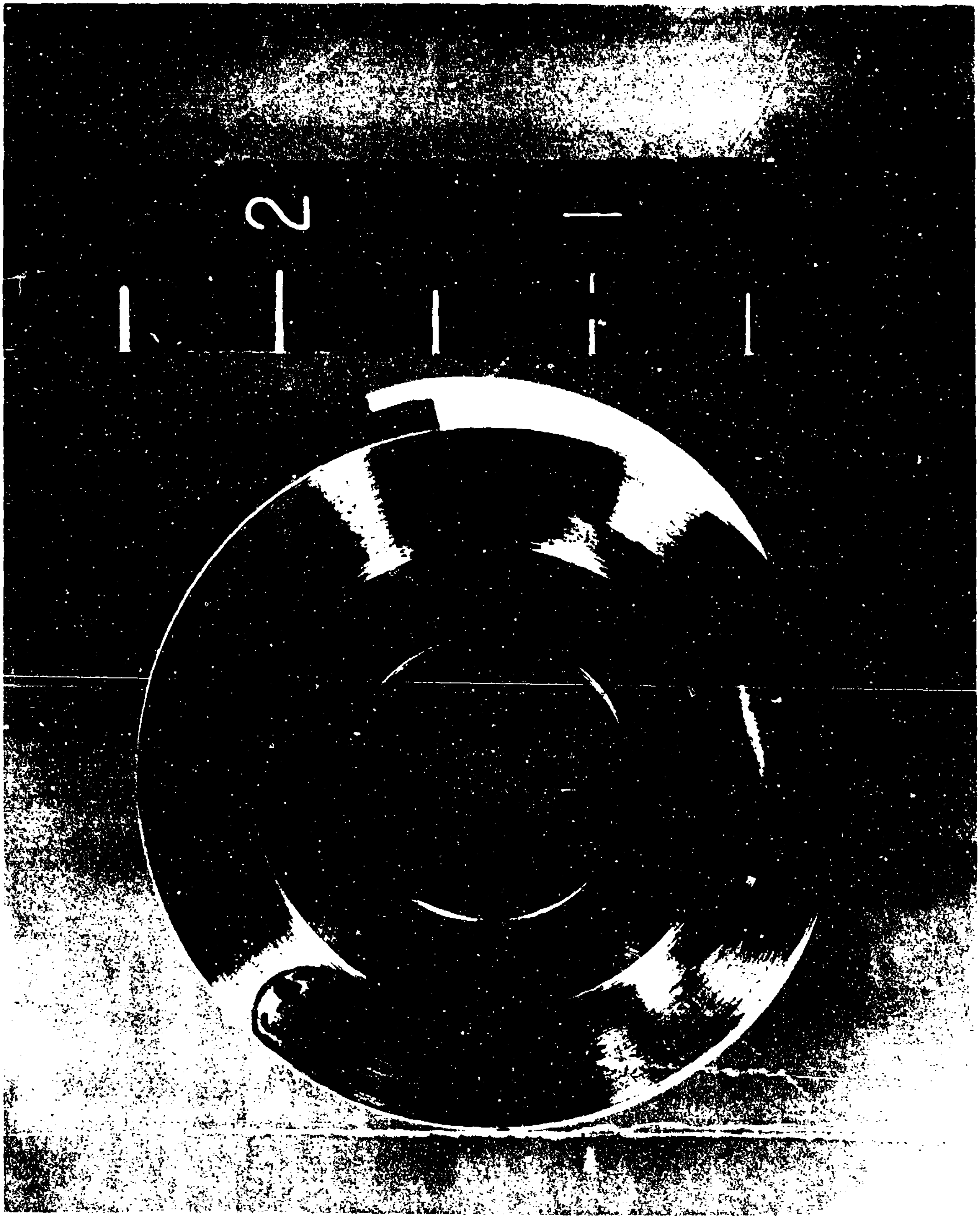

Figure 121. Failed Piston Seal 1 - End View

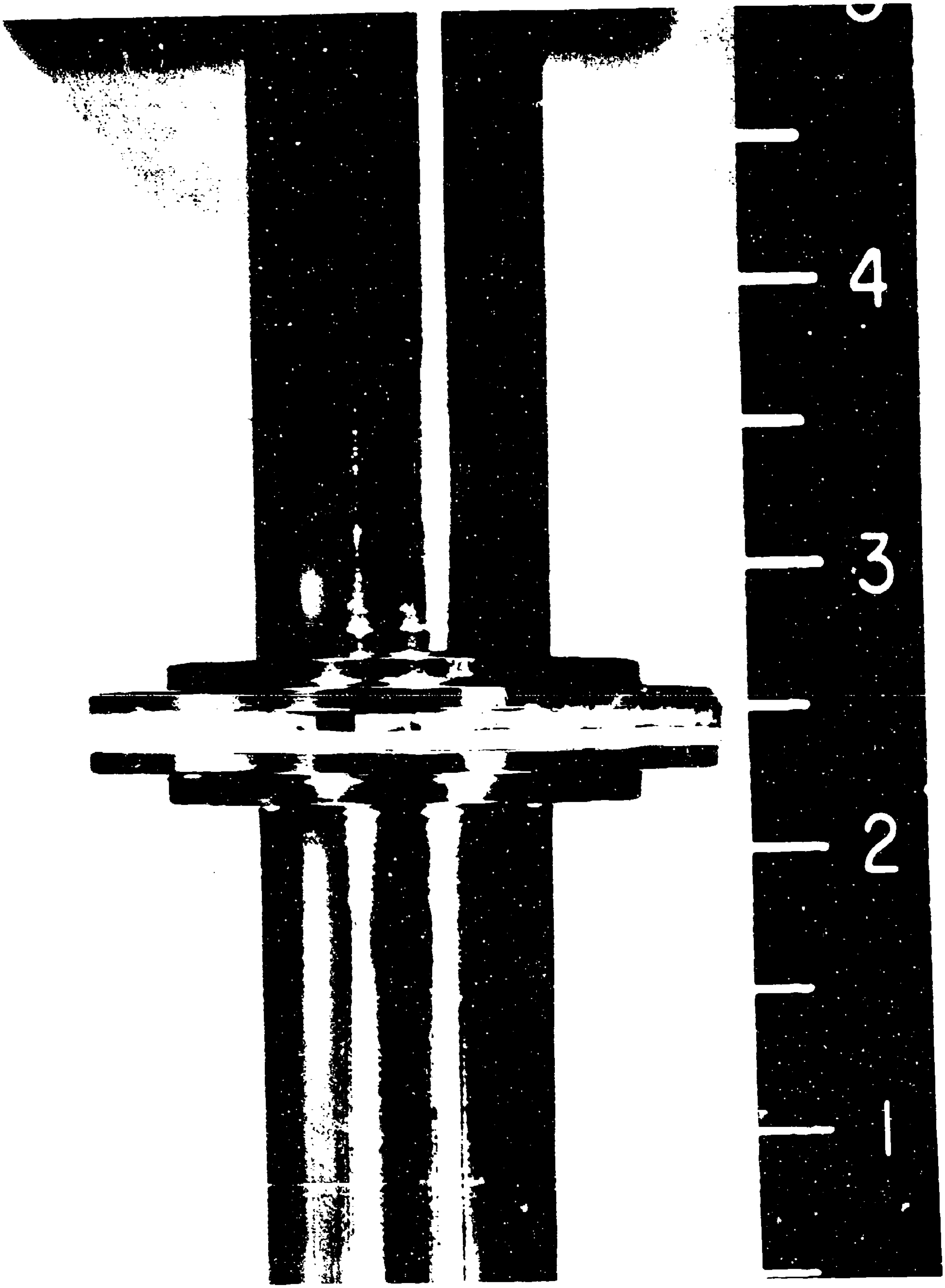

Figure 122. Failed Piston Seal 2 - Side View

smearing of the piston seal) and a fatigue stress problem for the piston seal design. The failure of the pistons on the test actuator provide a good demonstration of the problem. Pressure feedback compensation can reduce the level of force fight and provide an improvement in small signal distortion. This will reduce the amount of bias offset between active channels but will not correct for the noise content of the command channels causing constantly varying differential pressures across the actuator drive areas. The actuators used with control systems like the test system should be designed for large amplitude, constantly varying differential pressures across the drive pistons.

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