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ENGINEERING REPORT NO.

R107-722

SUBJECT

Proof and Operation Tests of Flight and Trim Controls

MODEL: XC-120

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FAIRCHILD AIRCRAFT

Division of

FAIRCHILD ENGINE & AIRPLANE CORPORATION

HAGERSTOWN, MARYLAND

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Specifications

(a) USAF Spec. R-1803-7B "Control Systems."
(b) USAF Spec. R-1815B, "Flying Qualities of Piloted Airplanes."

Reports

(c) Fairchild Eng. Report R107-720 "Control System Analysis XC-120."
(d) Fairchild Eng. Report R107-024 "Rigging Specification, XC-120."
(e) Fairchild Eng. Report R110-705 "Proof and Operation Tests of C-119B Flight and Trim Controls."

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PURPOSE:

The purposes of the tests conducted on the flight and trim controls were:

1. To determine the behavior of the flight and trim control systems under static proof loads and to observe deflections in the system for proof of conformance to the rigidity requirements of U.S.A.F. Spec. R1803-7B.

2. To perform operational tests of the flight and trim control systems to check their conformance to friction requirements as set forth in U.S.A.F. Spec. R1815-B.

3. To observe the structural behavior, alignment, clearances, etc., of the component parts of the control system while in operation under load.

4. To establish the method and forces required to engage the surface control locks, and to observe the uniformity of operation of the complete control locking system. Also to observe the rigidity and structural integrity of the system in general, when a limit design handle force of 50 lbs. was applied.

SUMMARY AND CONCLUSIONS:

a. In general, all tests were conducted using the same test setups and equipment as used on the C-119B flight and trim controls tests. Ref. (c). Operational tests of the aileron, rudder, elevator, and elevator trim tab control systems were conducted by applying dead weight to achieve the limit surface hinge moments and balancing cockpit control loads in 20% increments up to full test load. After applying each balanced increment of load, the control was operated using a spring scale to measure the additional force required to move the control through neutral in both directions. This was done to determine the system friction which was taken as the average of the added force in both directions.

b. Operational tests of the electrically actuated aileron and rudder trim tabs were conducted under load, applied to the tabs in 20% increments, up to the proof load. Data on voltage and current required by the actuator motor to operate the loaded tabs were taken to determine the power input to the systems and operating times.

c. Proof tests of the entire control systems were performed by first placing the control system under test in the most critical deflected position and then loading both the surfaces and the control units by dead weight computed from the geometry of the system to be in balance. In this manner the entire system was most effectively loaded under the full proof load. The control unit was returned to a fixed reference position at each loading position and the deflections of the surfaces determined. Proof tests of surface or control stops, brake pedals and other detail portions of the system were conducted by placing the control or surface in the required attutide and applying dead weight loads to them in 20% increments.

d. Rigging loads were maintained as closely as practicable to the values as specified in R107-024 "Rigging Specifications, XC-120." Cable tension readings were recorded before and after each proof and deflection test. Cable tensions were in general, near the high end of the range of tolerance before test, and near the middle or low end of the range after test. No cable tensions changed sufficiently during test to require re-rigging to reasonably meet minimum specification conditions.

e. The proof and operation tests of the flight and trim controls of the XC-120 airplane, as reported herein, indicate that the control systems are satisfactory from a strength, rigidity and operational standpoint. The friction forces required to operate the primary flight controls were, at the time of test, in excess of the allowances set forth in U.S.A.F. Spec. R1815-B, "Flying Qualities of Piloted Airplanes." These higher friction forces are due to the facts that:

1. The tests were conducted immediately after completion of installation and rigging of the controls and prior to any repeated operation which would "run in" the controls and reduce the friction.

2. The subject airplane has an unusual configuration, employing a dual set of controls and using a larger number of pulleys and fairleads than usually employed in a conventional airplane. The relatively large number of pulleys and fairleads is necessary because of the necessity of routing the cables through crew nacelle, wings and booms to transmit the necessary forces from the pilot's and copilot's controls to the surface.

A complete summary of the procedure for and results of each test is given in the following Table I.

Table I - Summary of Tests

[illegible] and Test [illegible]	Load Applied in Cockpit lbs.	Moment Ap. at Surface In-lbs.	Results of Tests	Tables Figures Photos
I Aileron Control System				
1. Test A-1 - Proof and Deflection Test - 70% limit hinge moment resisted by 70% double pilot max. effort - ailerons fully deflected Roll at 212 MPH - 1G load factor - pack off - left ailerons up - right ailerons down - copilot wheel maintained approx. 5° from left stop at all levels of load.	115 lbs. on pilot's wheel and 160 lbs. on copilot's wheel, applied simultaneously	2880 in. lbs. on left outbd aileron. 5990 in. lbs. on left inbd. aileron. 5980 in. lbs. on right inbd. aileron. 3360 in. lbs. on right outbd aileron. applied simultaneously.	Diagram of System. System sustained load satisfactorily. Maximum surface deflection 9.75° at 70% double pilot effort. Maximum permissable deflection (Ref. a) is 10°. Maximum deflection of pilot's wheel relative to copilot's wheel, at 70% limit load, was 2.0°.	Figs. 1 and 2 Figs. 3 and 4 Photo 18158
2. Test A-2 - Test of Wheel Stops 100% single pilot effort (160# single rim force) applied to pilot's wheel with wheel against stop.	160 lbs. on pilot's wheel only.	None	Stops sustained load satisfactorily. Negligible permanent set of 0.10° (at 20% after 100%). Wheel deflected 1.6° at 100% limit load.	Fig. 5
3. Test A-3 - Operation Test of the Entire Aileron Control System - All cables loaded up for the purpose of determining friction losses and observing the operation of the entire system under load. 50% double pilot effort - Hinge moment applied to each surface with load equally balanced at pilot's and copilot's wheels - Surfaces operated under load - Left ailerons loaded downward - Right ailerons loaded upward.	80 lbs. per wheel - applied simultaneously	1422 in. lbs. on left outbd. aileron. 2880 in. lbs. on left inbd. ailerons 1720 in. lbs. right outbd. aileron 3530 in. lbs. on right inbd. aileron.	System functioned satisfactorily. Friction at zero load was 10 lbs. measured at control wheel rim, and increased to 16.3 lbs. at 50% double pilot effort (160#) Maximum frictional force allowed by AAF Spec. R1815-B (Ref. (b)) is 6 lbs wheel rim load. System friction should decrease after control systems are further run in.	Figs 6 and 7 Photo 18160 and 18162

Item Tested and Test Condition	Load Applied in Cockpit Lbs.	Moment App. at Surface In-Lbs.	Results of Tests	Tables Figures Photos
4. Test A-4 - Proof Load Test of Partially Shot-Out System - Pilots fuselage cables disconnected and control surface locks engaged - 100% pilot effort (160#) and 50% copilot effort (80#) applied simultaneously and resisted by surface locks. This loaded locks to 150% max. single pilot effort.	160 lbs. rim force on pilots wheel. 80 lbs. rim force on co-pilot's wheel.	None	System sustained load satisfactorily. System carried 75% of double pilot load normally carried by complete system as per requirements of Par. D-3a of Spec. R1803-7B. Deflection of the entire system (pilot's fuselage cables disconnected) resulted in 86° rotation of pilot's wheel and 63° rotation of copilot's wheel. This amounted to 55% and 41% respectively of the total available angular travel (150°) of the control wheels. Permanent sets at 20% after 100% were 5° in both pilot's and copilot's wheel. These sets represent 3% of total wheel travel in either direction. Control surface locks withstood 150% max. single pilot effort (240#)	Fig. 8
II Rudder Control System			Diagram of System	Fig. 9 and 10
1. Test R-1 - Operation Test of Entire Rudder Control System. - All cables hooked up for purpose of determining friction losses and observing the operation of the system under load - 50% double pilot effort - Hinge moment applied to each rudder surface with load resisted in cockpit on pilot's and copilot's right pedals - Surfaces operated under load.	150 lbs. per right rudder pedal, applied simultaneously.	1680 in. lbs. to the left, per rudder surface, applied simultaneously.	System functioned satisfactorily. Friction varies from 19.4 lbs. at no load to 33.5 lbs. at 50% of double pilot limit load. Max. allowable friction load is 15# (Ref. (b)). Friction should decrease as systems are further run in.	Fig. 11 Photos 18051 and 18052

No., Tested and Test Condition	Load Applied in Cockpit Lbs.	Moment App. at Surface In-Lbs.	Results of Tests	Tables Figures Photos
2. Test R-2 - Proof and Deflection Test. - To determine the deflection of the complete system at a surface load producing 70% double pilot effort (420#) equally applied at pilot's and copilot's right rudder pedals - Rudder pedals in critical full fwd. length adjustment - rudders far enough to right of neutral to avoid hitting left hand stops under full load - spring tab controls not blocked - copilot's right hand pedal maintained at a constant position.	210 lbs. fwd. at each right hand rudder pedal.	2350 in. lbs. to the left on each rudder.	System sustained load satisfactorily. Max. average surface deflection due to control system stretch at 70% limit load was 9.7°. Maximum permissable deflection is 10° (Ref. (a)) Pilot's pedal deflection, relative to copilot's pedal, was not large enough to permit accurate measurement.	Figs. 12 and 13 Photos 18051 and 18052
3. Test R-3 - Test of Surface Locks Under Ground Loading - Surface locks on one rudder locked in place - upper and lower surfaces loaded in 20% increments to 100% limit hinge moment.	None	2310 in. lbs. to upper rudder 1240 in. lbs. to lower rudder	Surface and surface locks sustained load satisfactorily. At 100% proof load, upper surface deflected 1.9°, while lower surface deflected 2.8°. Permanent sets negligible.	Fig. 14
4. Test R-4 - Test of Typical Rudder Pedal Stop - 100% single pilot effort (300#) on pilot's right pedal All surface and pedal stops, except that carrying the load, backed out of contact.	300 lbs. on pilot's right - pedal	None	Stop and pedal system between pedal and stop sustained load satisfactorily at 100% single pilot effort, pilot's rudder pedal deflected 3.65°. There was no permanent set. See Fig. 15 for deflection of walking beam.	Figs. 15 and 16

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Item Tested and Test Condition	Load Applied in Cockpit Lbs.	Moment App. at Surface In-Lbs.	Results of Tests	Tables Figures Photos
5. <u>Test R-5 - Test of Mechanical Portion of Pilot's Brake System</u> - Load applied at tip of pilot's left and right pedals at right angles to pedal pivot axes - rudder pedals locked in neutral and placed in the critical full fwd. length adjustment - level of oil in one brake cylinder set so that cylinder would deflect 1 1/8", when the pedal was loaded, and come to a solid stop on the head of oil at that point - other brake cylinder replaced by a solid link. Load not carried beyond master brake cylinders.	300 lbs. at tip of both pilot's pedals, applied simultaneously.	None	System sustained load satisfactorily. See graphs for deflection data.	Fig. 17 and 18 Photo 18055
6. <u>Test R-6 - Proof Test of a Partially Shot Out System</u> - Rudder pedals set in critical fwd. length adjustment - cables in right hand boom disconnected - rudders deflected 14° right prior to loading and a corresponding fixed position noted for copilot's left pedal - position of copilot's left pedal maintained throughout test while measuring surface deflection.	300 lbs. fwd. on copilot's right pedal. 150 lbs. fwd. on pilot's right pedal.	3360 in. lbs. applied to the left on the right rudder. 1680 in. lbs. applied to the left on the left rudder.	Deflection of entire control system at 100% proof load, corrected for spring tab control deflection, was equivalent to 12° at the left rudder and 13.7° at the right rudder. Permanent sets were no more than 0.2° of rudder deflection. System sustained load satisfactorily - Defl. of rudder pedals relative to copilot's left pedal at 100% load Copilot right - 2° fwd. Pilot right - 3° aft. Pilot left - 3° fwd.	Fig. 19

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Item Tested and Test Condition	Load Applied in Cockpit Lbs.	Moment App. at Surface In-Lbs.	Results of Tests	Tables Figures Photos
III Elevator Control System				
1. Test E-1 - Proof and Deflection Test			Diagram of System.	Figs. 20 and 21
Deflection test of entire system at 70% double pilot effort applied equally at the two columns. - Copilot's control column held fixed at all levels of loading.	210 lbs. aft at each control column.	6560 in. lbs. down on elevator	System sustained load satisfactorily. Overall deflection at 70% limit load was 11.0°. Max. specified deflection is 10° (Ref. (a)) Permanent set in terms of elevator deflection, was 1.5° indicating some permanent stretching of the cables. At max. load the top of the copilot's column deflected 0.28 inches relative to the fixed base of the column.	Photo 18047
2. Test E-2 - Operation Test of Entire Elevator Control System.- 50% double pilot effort - Hinge moment applied to surface with load resisted at pilot's and copilot's columns. Surface operated under load.	150 lbss aft on each control column.	4670 in. lbs. moment applied to the elevator (down)	System functioned satisfactorily. Friction varied from 14 1/2# at no load to 19.7# at 50% double pilot limit load. Max. permissable friction is 8# (Ref. (b)) Friction should decrease after system is run in.	Figs. 25 and 26 Photo 18045
3. Test E-3 - Proof Test of a Partially Shot Out System. - Copilot's fuselage cables and left hand boom cables disconnected - hinge moment applied to surface resisted by load on both control columns - elevator deflected up 24° at no load. The copilot's column was held in fixed position while loading system.	300 lbs. aft at copilot's column. 150 lbs. aft at pilot's column applied simultaneously	7000 in. lbs. downward on the elevator.	System sustained load satisfactorily. Deflection of elevator at max. load was 20° (adjusted for deflection due to spring tab) Permanent set was 1° after loading to max. load. Torsional deflection of connecting tube between the two columns was about 2.25° at 300# column load.	Fig. 27 and 28 Photo 18047

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Item Tested and Test Condition	Load Applied in Cockpit Lbs.	Moment App. at Surface In-Lbs.	Results of Tests	Tables Figures Photos
4. Test E-4 - Test of Control Column Stop - 100% single pilot effort - fuselage cables on co-pilot's side disconnected - columns moved aft until stop at sector on the pilot's side was contacted. Load applied on copilot's column and reacted at pilot's side stop. All other stops backed off - out of contact.	300 lbs. aft on copilot's column only.	None	Stop sustained load satisfactorily. Deflection between base of pilot's column and sector, due to deflection in linkage was 0.8° column rotation at full pilot effort.	Fig. 29
IV Aileron Trim Tab Controls				
Proof and Operation Test of Tab Actuator and Connecting Linkage. Outboard tab of inboard right hand aileron loaded - aileron surface locked in neutral - tab operated to both extremes of travel.	None	480 in. lbs.	Diagram of System System functioned satisfactorily - Current to operate from neutral tp full up increased from 1.6 amps at zero load to 1.82 amps at 100% proof load. The corresponding times were 7 seconds at zero load and 8.5 seconds at 100% proof load. The structural deflection of fully deflected tab at 100% proof load was 1.36°.	Fig. 30, and 31 Fig. 32
V Rudder Trim Tab Controls				
Proof and Operation Test of Tabs Actuator and Connecting Linkage. Left hand tab loaded - right hand actuator disconnected - rudder control surface locked in neutral - tab operated to both extremes of travel.	None	527 in. lbs.	Diagram of system. System functioned satisfactorily. Current to operate surface from neutral to fully deflected varied from 1.56 amps at no load to 1.85 amps at 100% proof load. The corresponding times were 7.5 seconds to 9.9 seconds. The deflection of the rudder trim surface at 100% proof load was 0.9° in neutral and 0.7° in the fully deflected position.	Fig. 33 Fig. 34

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Item, Test and Description	Load Applied in Cockpit Lbs.	Moment App. at Surface In-Lbs.	Results of Tests	Ref. Figs.
VI Elevator Tab Controls				
1. Test #1 - Proof and Deflection Test of Elevator Spring Tab Controls - Elevator control surface locked at leading edge - elevator control horn loaded by means of elevator control system, thereby fully extending the spring cartridge and actuating spring tab - elevator surface and control horn held in fixed position with horns just contacting their stops (under zero load) to prevent any motion in the spring cartridge except that due to deflection in control tab linkage.	Sufficient to hold elevator control horn just in contact with stop on horn support bracket.	724 in. lbs.	Diagram of System System sustained load satisfactorily. Max. tab deflection, relative to elevator, was 4.9° at 100% limit load. There was no permanent set in the system.	Figs. 30, 35,36,37 Fig. 38
2. Test #2 - Proof Test of Elevator Trim Tab Control Fwd. Stop. Load applied, with spring scale, in a nose down, tab up, direction to a cotton strap wrapped around copilot's tab control wheel, and reacted at pilot's control wheel stop.	75# tangentially to rim of copilot's elevator trim tab control wheel.	None	System sustained load satisfactorily. Max. angular deflection of copilot wheel, at 100% limit load, was 8.93°. There was no permanent set in the system.	Fig. 39

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Item Tested and Test Condition	Load Applied in Cockpit Lbs.	Moment App. at Surface In-Lbs.	Results of Tests	Tables Figures Photos
3. Test #3 - Friction and Operation Test of Elevator Trim Tab Control System - Elevator trim tab moved to full up tab (nose down) position - copilot's tab control wheel operated in a nose down direction, by means of webbing and a spring scale, against the load - repeated with wheel moving in a nose up direction through neutral.	Sufficient to operate system.	822 in. lbs. at tab.	System functioned satisfactorily. Operating torque on copilot's wheel was 76 in. lb. at 100% limit load.	Fig. 40
VII Control Lock System	50# Max.	-	Description of System It was found that the control locks could be readily engaged when the control surfaces were in neutral or were gently oscillated about neutral. Under these conditions the average force needed to engage the locks varied from 42 to 43 lbs. applied at the center of the pistol grip.	Figs. 41 and 42.

DESCRIPTION OF SPECIMEN:

1. During these tests (Dec. 19, 1949 to May 2, 1950) the airplane was in the factory in a nearly completed state. Before testing each system, the system was checked and certified correct by the Fairchild Inspection Dept., as to installation, adjustment and rigging tensions, in conformance with Ref. (d).

On this aircraft, ailerons, rudders, elevator, elevator trim tab, and the control locks are cable operated while the aileron and rudder trim tabs are electrically actuated.

The primary control surfaces of this airplane are actuated by duplicate control systems with interconnects provided at several locations, between the pilot's and copilot's control systems. All primary flight controls have an interconnection between the pilot and copilot's system at the rear spar by means of cables and sectors, and the automatic pilot is connected to the control system at this point. The operation of the various control systems is as follows:

a. Ailerons - A wheel control on the control column transmits load through a chain and sprocket to cables, which in turn transmit load to sectors at the rear spar. Cables from these sectors carry the load to bellcranks and push-pull tubes that operate the inboard and outboard ailerons. The pilot's system is interconnected with that of the copilot at the rear spar and at the control columns; in both cases the interconnection is by cables (Fig. 1 and 2, Dwg. 107-720111).

b. Rudders - The rudder control system consists of conventional pedal hangers which actuate a walking beam forward of the pedals through a system of horns and links. (See Fig. 9 and 10 or Dwg. 107-720109). When rotated, the sector end of the walking beam transmits load to a cable which in turn transmits the load to a sector at the rear spar. This sector transmits the load to another cable which carries it to a sector in the tail cone where the load is transferred to the surface by a horn and torque tube.

The pilot's and copilot's systems are interconnected by cables at the rear spar and also in the stabilizer. There is no interconnection at the copilot's and pilot's station.

c. Elevator - The control column is connected by a push-pull tube to a sector whose plane is perpendicular to that of the control column. Fore and aft motion of the control column causes rotation of the sector in the plane of the cable system. (See Fig. 20 and 21 or Dwg. 107-720110) The sector transmits the load to a cable which transmits the load to another sector at the rear spar. Here another cable carries the load to a sector in the tail cone which actuates a push-pull tube thereby transmitting the force to the elevator horn.

The pilot's and copilot's systems are interconnected at the rear spar and also in the stabilizer by cables. In addition the control columns are interconnected by a torque tube.

d. Elevator Trim Tab - This tab is manually operated from a control unit on the cockpit pedestal. Load is carried by a cable system to a drum and screw type actuator mounted in the stabilizer and elevator and thence to the trim tab horn by a push-pull rod (Fig. 30 and 35).

e. Aileron Trim Tab - This is an electrically actuated tab controlled from a switch on the cockpit pedestal. The actuating motor is in the right inboard aileron and operates the one tab, which is on the right inboard aileron only, by a push pull tube connected to the tab horn. See Figs. 30 and 31.

f. Rudder Trim Tab - There is an electrically actuated tab on each rudder with the actuating motors mounted in the rudder forward of the tab hinge line. The motor operates the tabs through a push pull tube connected to the tab horn. The electrical actuators are synchronized to each other being self-centering on neutral. See Figs. 30 and 33.

g. Elevator Spring Tab - The elevator spring tab controls are a self contained system, installed and operating entirely in the elevator. When the pilot operates the control column, the spring and push pull rods of the system are actuated and cause the tab surface to move, thus automatically assisting the operation of the elevator surfaces. See Figs. 36 and 37.

h. Control Lock System - The surface lock system is operated by a handle on the pilot's pedestal. Pulling on the handle imposes a torque on a pulley connected to the handle. This pulley transmits the torque to a cable connected to a differential pulley, which in turn transmits it to a set of pulleys at the rear spar (See Figs. 41 and 42) These pulleys transmit the force to a set of take-off pulleys in each engine nacelle at the rear spar, where the force is transmitted each to a differential pulley at the stabilizer and out to the ailerons. The force transmitted to the ailerons locks the ailerons by means of locking cams located at the inboard aileron surface, while the force transmitted to the stabilizer actuates a differential pulley having two cams as intergral parts of the pulley. As these two cams rotate toward closed position, they limit the motion of elevator and rudder sectors and consequently the motion of the reppective surfaces. This permits the engagement of the "free motion spring tab lock cams" located at each end of the elevator surface and at each rudder horn spring tab mechanism. These "free motion lock cams" have retarded motion in relation to the main locking cams which actuate item, thereby permitting the entrance of the free motion cams within their mating jaws.

These jaws are an integral part of the surface so that when the "free motion" cams are mated the surface is locked and no surface loads can enter the system to strain it.

Spring retraction safety mechanisms are part of the control lock system. These mechanisms are mounted in tandem close to the main locking cams at the extremeties of each system, and each "free motion lock cam" is equipped with a torsion spring on the hub. These retraction mechanisms hold all lock cams in open position during flight, thereby, eliminating the possibility of surfaces locking in flight. In addition they serve as an aid in opening the cams during normal operation.

DISCUSSION OF TESTS:

1. Aileron Control System (See Fig. 1 and 2)

Dummy ailerons (T78-713 and T78-714) were installed on both wings in place of the regular ailerons. This was done because it would be difficult to produce the necessary hinge moment with dead weight loads without damaging the regular surfaces. The dummy ailerons, made from steel channel, were easier to load and more convenient to use. The aileron control system was tested in the following critical conditions. (Photo 18160 and 18162).

a. Test A-1 - Proof and Deflection of System.

This test was conducted to prove conformance with Spec. R1803-7B, D3 (Ref. (a)) which requires that the deflection at any aileron shall not exceed 10° when the system is loaded to 70% of limit load (70% two pilot max. effort.) It also constituted a proof test of the aileron control system under maximum flight airloads in which the ailerons were fully deflected at 212 MPH, 1G load factor, pack off, left aileron down about 12°, right aileron up about 24°. The maximum test loads applied to the ailerons (Pg. 16, Ref. (c)) were:

Surface	L. Outboard	L. Inboard	R. Inboard	R. Outbd.
Hinge Moment	2880 in. lb.	5590 in.lb.	5980 in.lb.	3360 in.lb.
Direction of Load	Down	Down	Up	Up

The above loads were reacted by the following loads on the rim of the dummy control wheels (equiv. diameter as actual diameter.)

Pilot's wheel - 115 lbs. (single rim load)
Copilot's wheel - 160 lbs. (single rim load)

The ailerons were placed in the "left bank" attitude (corresponding to the above loading) with the copilot wheel approximately 5° from the left stop to prevent the stop from taking any load. The copilot wheel was returned to this same position at each increment of load and the relative deflection of each of the four surfaces and the pilot's wheel was recorded. This data is plotted on Figs. 3 and 4 respectively.

The systems meet the requirements of Spec. R1803-7B, D3 in that the maximum deflection is less than 10 degrees at 70% of maximum double pilot effort (9.75° - See Fig. 3).

The rigging loads before and after test were as follows:

Cable	Before Test	After Test	Rigging Spec.
Pilot's fuselage	122	123	135 ± 10%
Copilot's fuselage	122	113	135 ± 10%
Left Aileron cable	127	124	135 ± 10%
Right Aileron cable	123	128	135 ± 10%

b. Test A-2 - Test of Control Wheel Stops.

The object of this test was to demonstrate the ability of the wheel stops to withstand 100% pilot effort on the control wheel. The system was rigged to permit the pilot's wheel to hit the stop before the copilot's wheel touched the stop. This condition was maintained throughout the test. 100% pilot effort (160# single rim force) was applied in increments, and angular deflection of the pilot's wheel was recorded. This data is plotted on Fig. 5. No appreciable permanent set occurred after removal of load (.10° at 20% after 100%). The test limit load was supported without noticeable distortion of any part.

c. Test A-3 - Operation Test.

This was an operation test of the entire aileron system, with all cables hooked up, for the purpose of determining friction losses and observing the operation of the system under load. The system was operated back and forth at increments of load up to 50% of double pilot effort. The cables had been rigged to the rigging spec. requirements prior to the test. The max. single rim load applied to each control wheel simultaneously was 80 lbs. (See Photo 18158) The corresponding load on the aileron surfaces was as follows:

Surface	L. Outbd.	I. Inbd.	R. Outbd.	L. Inbd.
Hinge Moment ("#)	1422	2880	1720	3530
Direction of Load	Down	Down	Up	Up

At each load increment, which was in balance throughout the system, the force (friction) required to move the controls back and forth through the neutral position was measured. These friction forces, corrected for friction losses in the test loading fixtures, are plotted on Fig. 6. (Note - the stabilizing springs in the outer wings were disconnected.)

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The friction losses in the test loading fixtures are plotted on Fig. 7. The friction of the three pulley cockpit loading fixture was used directly, the friction of the cable over a single pulley (90° wrap) was used after correcting for the number of pulleys so used and the mechanical advantage of the system. (The 90° wraps were used on the surface end of the system).

The friction was found to be from 10 to 16.3 lbs. when the surface moved through neutral with corresponding loads of zero to 50% double pilot effort. The comparable forces on the C-119B airplane were 9.5 to 15.5 lbs. It may be expected that the friction force will decrease somewhat as the system is worn in.

d. Test A-4 - Proof Test of a Partially Shot Out System.

This test was a proof load test of a partially broken system to prove conformance to the requirements of Par. D-3a (2) of Spec. R1803-7B which requires that, with one system inoperative, the remaining system be capable of carrying 75% of the load normally carried by the complete control system. This test also demonstrates the ability of the control surface locks to withstand 150% max. single pilot effort.

The pilot's fuselage cables were disconnected and the control surface locks were engaged. Since the control surface lock system was not connected at the time of the test, it was necessary to manually engage these locks and secure them with safety wire.

The max. loads applied (simultaneously) were:

Pilot's wheel 150# rim force (single force)
Copilot's wheel 80# rim force (single force)

The load was applied in increments and following each increment the angular deflections of both control wheels were recorded. This data is plotted on Fig. 8. The deflection of both inboard aileron surfaces was also checked at each increment as a means of judging lock deflection. The maximum angular deflection recorded was .3° at 100% load.

The deflection of the entire system (with pilot's fuselage cables disconnected) resulted in 82° rotation of the pilot's wheel and 61° rotation of the copilot's wheel. This amounted to 55% and 41% respectively of the total available angular travel of the control wheels. (150° available.)

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The permanent set recorded at 20% after 100% was as follows:

Pilot's wheel 5°
Copilot's wheel 5°
These sets represent only 3% of the total wheel travel in either direction.

Cable tensions before and after this test were as follows:

Cable	Before Test	After Test	Spec. Req'd
Pilot's	Disconnected		135 ± 10%
Copilot's	131	117	135 ± 10%
Left Aileron	122	126	135 ± 10%
Right Aileron	142	127	135 ± 10%

2. Rudder Control System (See Figs. 9 and 10)

Prior to the tests on the rudder control system, the relationship between rudder movement, spring tabs movement, and rudder hinge moment was determined (See Fig. 12) This was done with the rudder surface locks free and the rudder tail cone, quadrant locked. In all tests, loads were applied to the rudder and rudder pedals in the manner shown on Photos 18051, 18052 and 18055.

a. Test R-1 - Operation Test.

This was an operation test of the entire rudder control system, with all cables hooked up, for the purpose of determining friction losses and observing the operation of the system under load. The system was operated back and forth at increments of load up to 50% of double pilot effort. The cables had been rigged to the rigging spec. requirements prior to test. Maximum test loads were: (Pg. 29, Ref. (c))

1680 in. lb. moment applied to the left on each rudder.
150 lb. applied forward on each right hand pedal.

At each load increment, which was in balance throughout the system, the force (friction) required to move the controls back and forth through the neutral position was measured. These friction forces, corrected for friction losses in the test loading fixtures are plotted on Fig. 11.

Friction losses on the first C-119B airplane at the time of first tests were: 31# at zero load and 47# at 50% double pilot effort. These values were reduced somewhat by refinements made during final cleanup so that the zero load friction on this airplane was reduced to 24.5# several days after the first tests. Similar reductions can probably be made on the XC-120 rudder controls, but it is doubtful that the 15# specification limit can be met on a loaded system.

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b. Test R-2 - Proof and Deflection Test of System.

In this test the entire cable system was hooked up and rigged to specification requirements. The object of the test was to determine the deflection of the complete system at a surface load which produced 70% of double pilot effort (420#) applied equally distributed at pilot's and copilot's pedals. The max. test loads were: (Pg. 29, Ref. (c))

2350 in. lb. moment applied to the left on each rudder.
210 lb. applied forward on each right hand pedal.

The rudder pedals were placed in the full forward length adjustment, which was critical. The rudders were placed just far enough to the right of neutral under no load so that they would not hit the left hand stops under full load. The spring tab controls were not blocked, therefore, the initial deflection of the rudder resulted from the spring tab control arrangement. The copilot's right hand pedal was returned to a fixed position at each increment of load and the deflection of the rudders and the pilot's pedal were measured. The pilot's pedal deflection relative to a fixed copilot's pedal was not large enough to be accurately measured. The surface deflections, which are indicative of deflection of the entire rudder control system, are plotted on Fig. 13. Fig. 12 is a plot determining that portion of the rudder deflection which results from deflection of the spring tab controls. The rudder deflections, relative to a fixed horn were 5.2° and 6.6° for the left and right hand spring tab control systems respectively in the position of adjustment at the time of test. (Note - Specified correct setting is 7° $\pm$ 1°) These values must be subtracted from the overall deflections at 70% limit load on Fig. 13 and result in an average deflection of 9.7° for the control system exclusive of the spring tab controls.

Par. D-3 of Spec. R1803-7B requires that the deflection at 70% limit load be no more than 10°. The test value, therefore, meets the specification requirement.

The permanent set at 20% load after applying the full test load was less than could be accurately measured and is, therefore, not recorded on Fig. 13.

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The rigging loads before and after the test were as follows:

Cable	Before Test	After Test	Rigging Spec.
Pilot's fuselage	95	85	85# ± 10%
Copilot's fuselage	89	85	"
Servo. Intercon.	84	90	"
L.H. Boom	98	95	"
R.H. Boom	98	94	"
Stab. Intercon.	100	105	"

In general, the rigging loads were slightly above the upper limit of the specified values prior to the test. Loss of rigging load in the loaded cables was slight. The interconnecting cables, which were not loaded in this test, showed a slight gain in tension after the test as compared to before test. The rigging tensions in the loaded cables were sufficiently within the specification values as to justify the deflection figures quoted above.

c. Test R-3 - Ground Load Test of Surface Locks.

This was a test of the surface locks under ground loading. The surface locks on one rudder were locked in place and the upper and lower surfaces loaded in 20% increments up to the following maximum loads: (Pg. 219, Ref. (c))

2310 in. lb. moment to the upper rudder.
1240 in. lb. moment to the lower rudder.

The resulting surface deflections are plotted on Fig. 14 and show normal elastic behavior and no permanent set. The test loads were supported without noticeable distortion of any part.

d. Test R-4 - Test of Rudder Pedal Stop.

This was a test of a typical rudder pedal stop to demonstrate its ability to sustain loads due to single pilot effort. The test was conducted with the right pilot's pedal under load and all pedal and surface stops except that which carries forward load on the loaded pedal were backed out of contact. A load of 300# forward (in 20% increments) was applied to the pedal while in the forward length adjustment. Angular deflection of the rudder pedal hanger, forward deflection at the brake pedal pivot, and deflection of the walking beam and its supports were recorded. (See Photo 15052).

Fig. 15 - Shows the deflection of the forward control sector (walking beam) and its support point under a tension load applied at the rod attachment point "C" and reacted at stop point "A". The dial gages which recorded these deflections were mounted on the fixed frames of the nose section.

Fig. 16 - Shows the forward deflection at the base of the pedal hanger and the equivalent average rotation of the hanger in degrees.

The pedal system, up to and including the stop structure, supported limit load without noticeable permanent deformation.

e. Test R-5 - Brake System Test (Photo 13055)

This was a test of the mechanical portion of the pilot's brake system (typical of copilot's) from the pedal through the master control cylinders, to determine the ability of the system to sustain single pilot limit loads of 300# applied to the top tip of both pedals simultaneously and at right angles to the pedal pivot axes. The rudder pedals were locked in neutral and placed in the full forward length adjustment which was critical. One brake cylinder (a test unit) was filled with oil and the ports sealed shut. The level of the oil was so set that the cylinder would deflect 1 1/3" when the pedal was loaded, and come to a solid stop on the head of oil at that point. The other cylinder was left out of the system and replaced with a rigid steel link. (It was not desired to load the units which would subsequently be used on the airplane as the adequacy of the Warner units to sustain loads compatible with a 300# pedal force had not been established at the time of test.)

Angular deflection of the brake pedals was read at 20% increments of the 300# loads. Four dial gages, supported on fixed structure of the nose section, were made to bear at the lower ends of the rudder pedal hanger support arms, at right angles to the axis of rotation of the rudder pedal hangers. The deflection data is plotted on Figs. 17 and 18 which also show that permanent distortion after proof testing was of negligible magnitude. The mechanical portion of the brake system and its supports, sustained limit loads without distress.

f. Test R-6 - Proof Test of a Partially Shot Out System.

This test was a proof load test of a partially broken system to prove conformance to the requirements of Par. D-3a (2) of Spec. 1803-73 which requires that, with one system inoperative, the remaining system be capable of carrying 75% of the load normally carried by the complete control system. The maximum loads applied were: (Pg. 29, Ref. (c)).

3360 in. lb. moment applied to the left on the right rudder.
1680 in. lb. moment applied to the left on the left rudder.
300 lb. applied forward on the copilot's right pedal.
150 lb. applied forward on the pilot's right pedal.

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The rudder pedals were placed in the forward length adjustment and the cables in the right hand boom were disconnected. All load from the right rudder was, therefore, carried through the stabilizer interconnect where it joined with left rudder loads to be carried through the left boom cables into the crew nacelle. The servo interconnect then carried 2/3 of the load, or single pilot effort, into the copilot's fuselage cables.

The rudders were deflected 14° to the right prior to loading and a corresponding fixed position noted for the copilot's left pedal. This position of the pedal was maintained throughout the test, while the angular deflections of the rudders and the other pedals were read at 20% increments of load. Deflections of the rudders are plotted on Fig. 19. Deflections of the three pedals (other than the fixed copilot's left pedal) at full test load were:

Copilot right - 2° forward
Pilot right - 3° aft
Pilot left - 3° forward.

The deflection of the entire control system at full proof load, exclusive of the deflection resulting from the spring tab controls, was equivalent to 12° at the left rudder and 13.7° at the right rudder. The entire system sustained the proof load without noticeable distress. Permanent sets averaged only 0.2° of rudder deflection and are, therefore, not noted on Fig. 19.

3. Elevator Control System (See Fig. 20 and 21)

Prior to conducting these tests, the relationship between surface movement, spring tab movement, and elevator hinge moment was established (See Fig. 22) This was done with the elevator locks free and the elevator boom cone quadrant locked. In all tests, loads were applied to the elevator and the control columns in the manner shown on Photos 13045 and 13047.

a. Test E-1 - Proof and Deflection Test of System.

In this test, the entire cable system was hooked up and rigged to rigging spec. requirements. The purpose of the test was to determine the deflection characteristics of the complete system at a load on the surface which produced 70% of double pilot effort (420#) applied equally to the two wheels. Loads applied were:

6560 in. lb. moment on elevator surface applied downward with the elevator in neutral trail prior to loading (to balance control column load.

210# aft to each control column.

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The copilot's column was held in fixed position throughout the test and the angular rotation of the elevator and pilot's column read at 20% load increments. The pilot's column deflection relative to the fixed copilot's column was not large enough to be measurable. The surface deflection, which is indicative of deflection throughout the control system is plotted on Fig. 23. Fig. 22 evaluates that portion of the surface deflection which results from deflection of the spring tab controls. The maximum elevator deflection relative to a fixed horn proved to be 10°. This value is subtracted from the overall deflection at 6560 in. lb. load on Fig. 23 and results in a deflection of 11.0° in the system exclusive of the spring tab.

Par. D-3 of Spec. R1803-7B requires that the deflection at 70% limit load be not more than 10°. The test value, therefore, slightly exceeds the specification limit. Permanent set in terms of elevator deflection was 1.5°, indicating some permanent elongation in the cable system. The rigging loads before and after the test were as follows:

Cable	Before Test	After Test	Rigging Spec.
Pilot fuselage	111	90	100# ± 10%
Copilot fuselage	110	85	100# ± 10%
Servo Intercon	98	96	85# ± 10%
L.H. Boom	96	96	85# ± 10%
R.H. Boom	82	85	85# ± 10%
Stab. Intercon.	104	99	85# ± 10%

All cables were rigged on the high side at the time of test but the rigging tensions dropped, particularly in the fuselage cables, as a result of loading.

Fig. 24 is a plot of the bending deflection at the top of the control column relative to a fixed base at the pivot.

b. Test E-2 - Operation Test.

In this test, the control system was operated back and forth through neutral at zero load and in increments of load up to 50% of double pilot effort applied in balance to the control column (aft) and the elevator (down). Max. loads applied were:

4670 in. lb. moment applied to the elevator (to balance control column load.)
150# applied aft on each control column.

At each increment of load, which was in balance throughout the system, the force (friction) required to move the controls back and forth through the neutral position was determined. These friction forces are plotted on Fig. 26 and are corrected for friction losses in the test loading fixtures. The latter are plotted on Fig. 25. The friction varies from 14.5# at no load on the system, to 19.7# at 50% of double pilot limit load.

Friction losses on the first C-119B production airplane were 12.3 lbs. at zero load and 16.5# at 50% double pilot effort when first tested. These values were somewhat reduced by final cleanup of interferences after completion of the airplane and it is likely that reductions can be made on the XC-120 also. However, it is doubtful that the 8# limit can be met.

c. Test E-3 - Proof Test of a Partially Shot Out System

This was a proof test of a partially broken system to prove conformance to the requirements of Par. D-3a (2) of Spec. R1803-7B which requires that, with one system inoperative, the remaining system be capable of carrying 75% of the load normally carried by the complete control system. The maximum loads applied in this test were: (Pg. 27, Ref. (c))

7000 in. lb. moment applied downward on the elevator (to balance load of control column.
300# aft applied at the copilot's column.
150# aft applied at the pilot's column.

The copilot's fuselage cables and left hand boom cables were disconnected so that all load was routed through the right hand boom cables, fuselage, crossover and left hand fuselage cables to the pilot's column. The torque tube between the two columns was made to carry torque due to single pilot effort.

Deflection in the entire control system in terms of elevator deflection relative to a fixed copilot's column is plotted on Fig. 27. The elevator was placed in the 24° up position at no load. The deflection of the entire control system at the maximum test load was 20°, exclusive of the deflection due to the spring tab. The permanent set was only 1°. The rigging loads before and after the test were:

Cable	Before Test	After Test	Rigging Spec.
Pilot's fuselage	100	93	100# ∓ 10%
Servo Intercon	91	69	85# ∓ 10%
R.H. Boom	95	88	85# ∓ 10%
Stab. Intercon.	90	93	85# ∓ 10%

The load in the servo interconnect, which was loaded for the first time in this test, was the only one which dropped markedly.

Torsional deflection of the connecting tube between the two columns is plotted on Fig. 28, and equalled about 2 1/4° at 300# load.

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d. Test E-4 - Test of Control Column Stop.

This was a test of a typical control column stop for ability to sustain single pilot effort loads. The fuselage cables on the copilot's side were left disconnected and the columns moved aft until the stop at the sector on the pilot's side was contacted. All other stops, including surface stops, were backed off out of contact. A load of 300#, in increments, was applied aft on the copilot's column and reacted on the stop. The resulting deflections are plotted on Fig. 29 which shows the angular rotation at the base of each column and the linear motion of the cable immediately aft of the sector at the stop. The deflection between the base of the pilot's column and the sector, which includes the push-pull rod, was only .3° of column motion at full pilot effort.

4. Aileron Trim Tab (See Figs. 30 and 31)

This tab is electrically actuated, and during this test it was operated under 20% incremental loadings up to 100% limit hinge moment. These loads were applied by means of dead weight.

The purpose of this test was to demonstrate the structural integrity of the trim tab operating mechanism and to determine the operational characteristics of its actuating unit. The tab in question is the outboard tab of the inboard right hand aileron. The design limit hinge moment is 480"# (Ref. (c), Pg. 310) which was applied on an 8 5/8" arm, measured when the tab was in neutral.

The aileron surface was locked in neutral by means of the control locking system. Without any hinge moment, the trim tab surface was operated to both extremes of travel. While so doing, the current and time required was recorded. The recording ammeter was used throughout the test. Starting with the trim tab in neutral, the hinge moment was applied in 20% increments up to 100% proof load. This load was acting in a down direction simulating load on an upward deflected trim tab. The deflection of the surface under the incremental load was recorded. Allowances were made for the deflection of the aileron surface itself. The trim tab was then actuated in an upward direction until motion was halted by the cutoff switches. The current and time for this operation was recorded as was the deflection caused by the load in the fully deflected upward position.

The maximum current required to operate from neutral to full up increased from 1.6 amps at 0 load to 1.82 at 100% proof load. The time varied from 7 seconds at 0 load to 8.5 seconds at 100% proof load. The deflection of the fully deflected flap was 1.36° at 100% proof load. (See Fig. 32). The system operated in a satisfactory manner throughout the test and no permanent set was recorded.

5. Rudder Trim Tab (See Figs. 30 and 33)

Like the aileron trim tabs, this trim tab is electrically actuated. During this test it was loaded, in 20% increments, to full limit load by means of dead weight. The tab was operated against the load at each increment.

The purpose of this test was to demonstrate the structural integrity of the rudder trim tab operating mechanism and to determine the operational characteristics of the actuating unit at loads from 0 to 100% proof load. The rudder trim tab is the upper tab located on the rudder surface. There is one trim tab on each rudder; however, only one (left hand) was loaded and operated during this test. The right hand actuator was disconnected from the electric circuit. The limit design hinge moment as specified by the Structures Section (Pg. 308, Ref. (c)) is 527"#, which was applied in this test on a 7 3/4" arm, measured when the surface was neutral. With the rudder control surface locked by use of the locking system, and with the trim surface in neutral, the tab was deflected to both extremes of travel. Starting at neutral and with the hinge moment applied in 20% increments from 0 to 100% proof load (the load operated to the left with respect to the airplane) the trim tab was operated to the right, to the full extreme of travel. The deflection of the trim tab with respect to the rudder was recorded both in neutral position and the fully deflected position. The current and time to operate from neutral to fully deflected was recorded.

The current required to operate the surface from neutral to fully deflected varied from 1.56 amps at 0 load to 1.85 at 100% proof load. The time varied from 7.5 seconds to 9.9 seconds respectively. The deflection of the rudder trim surface at 100% proof load was .9° and .7° in the neutral and fully deflected positions respectively. (See Fig. 34) The rudder trim tab actuating system operated in a satisfactory manner throughout the test. No permanent set of the trim tab surface was produced.

6. Elevator Tab Controls (See Figs. 30, 35, 36, and 37)

a. Test T-1 - Proof and Deflection Test of Elevator Spring Tab Controls.

The purpose of this test was to demonstrate the structural integrity of the spring tab operating mechanism under limit load of 724"# (Pg. 256, Ref. (c)) about the tab hinge line and to measure the deflection of the tab control mechanism in terms of deflection of the tab.

By locking the elevator control surface at the leading edge and pulling on the elevator control horn by means of the elevator control system, the spring cartridge was fully extended and the spring tab actuated to its full up position. The elevator surface and the elevator control horn were held in this fixed position so that there would be no motion of the spring cartridge except by deflection in the tab control linkage.

Dead weight load was applied to the tab and its position away from the no load position recorded. The deflection so recorded is the accumulated deflection in the tab control system between the elevator horns and the tab.

The tab deflection relative to the elevator is plotted in Fig. 38 Maximum deflection at 100% limit load was 4.9°. After several operations of the tab at zero load after 100% load, a second reading at 20% load showed that no permanent set had occurred in the system. No signs of distress were noted during the test.

b. Test T-2 - Proof Test of Elevator Trim Tab Control Forward Stop.

The purpose of this test was to demonstrate the structural integrity of the trim tab control wheel, the wheel interconnecting torque tube, and the wheel stop mechanism under a limit load of 75 lb. (Pg. 235, Ref. (c)) applied tangentially to the rim of the copilot's elevator trim tab control wheel.

Load was applied with a spring scale in a nose down, tab up, direction to a cotton strap wrapped around the copilot's tab control wheel. This load was carried through the interconnecting shaft to the pilot's tab control wheel and was reacted at the stop on the pilot's control wheel rim. The angular deflection of the copilot wheel was recorded and is plotted in Fig. 39. Maximum deflection at 100% limit load was 8.93° with no permanent set recorded. Inspection of the system revealed no failures or other signs of distress.

c. Test T-3 - Friction and Operation Test of Elevator Trim Tab Control System.

The purpose of this test was to measure the operating forces of and the efficiency of the elevator trim tab control system. It was also desired to observe the entire tab control system when operating under load for evidence of binding, yielding, sagging cables, or other malfunction due to deflection. Limit load applied to the tab was 822 in. lbs. (Pg. 235, Ref. (c)).

The elevator trim tab was moved to the full up tab (nose down) position and the tab control wheel then returned 1/2 turn in the nose up position. Dead weight load was applied in increments to the tab in a down direction to give the required hinge moment. The copilot's tab control wheel was then moved, in a nose down direction, by means of webbing and a spring scale. This test was repeated with the tab being moved through the tab neutral position, moving up against the same loads used in the first test.

Since the operating forces were slightly higher with the tab neutral, this data is presented in Fig. 40, which also shows a graph of theoretical frictionless operating force. The ratio between the two curves is the total efficiency of the system.

An inspection of the system after test disclosed no points of failure or signs of distress.

7. Control Lock System

The purpose of this test was to determine the method and forces required to engage the surface control locks. It was also the purpose of this test to observe the structural integrity of the system when a limit design handle force of 50 lbs. was applied.

Prior to the test, the control lock system had been thoroughly checked out and certified correct by the Inspection Dept. in accordance with Ref. (d). The control rigging was checked by the Test Laboratory and found to be within rigging specifications except as noted.

System	Cable Tension Rigging Spec. Pg. B-6, Ref. (d).	Cable Tension as Measured at Start of Test (Lbs.)
Pedestal	65# ± 10#	65
Fuselage	65# ± 10#	68
Fuse. Interconnect	85# ± 10#	88
L. Fuse. to Boom	"	98*
R. Fuse. to Boom	"	82
Left Wing	"	85
Right Wing	"	85
Left Boom	"	87
Right Boom	"	87
Left Rudder	70# ± 10#	79
Right Rudder	"	75
Left Elevator	"	80
Right Elevator	"	87*

*While this tension was slightly above specification value the tension was not altered.

The force required to engage the control locks was found to vary from 42 to 43 lbs. The handle travels a total of 57° from the unlocked to the locked position. The surface locks are fully locked within the final degree of travel. Unlocking motion of the cam commences within 5.5 to 6.25° unlocking travel of the operating handle.

The system sustained the 50# proof handle load without permanent distortion of any part.

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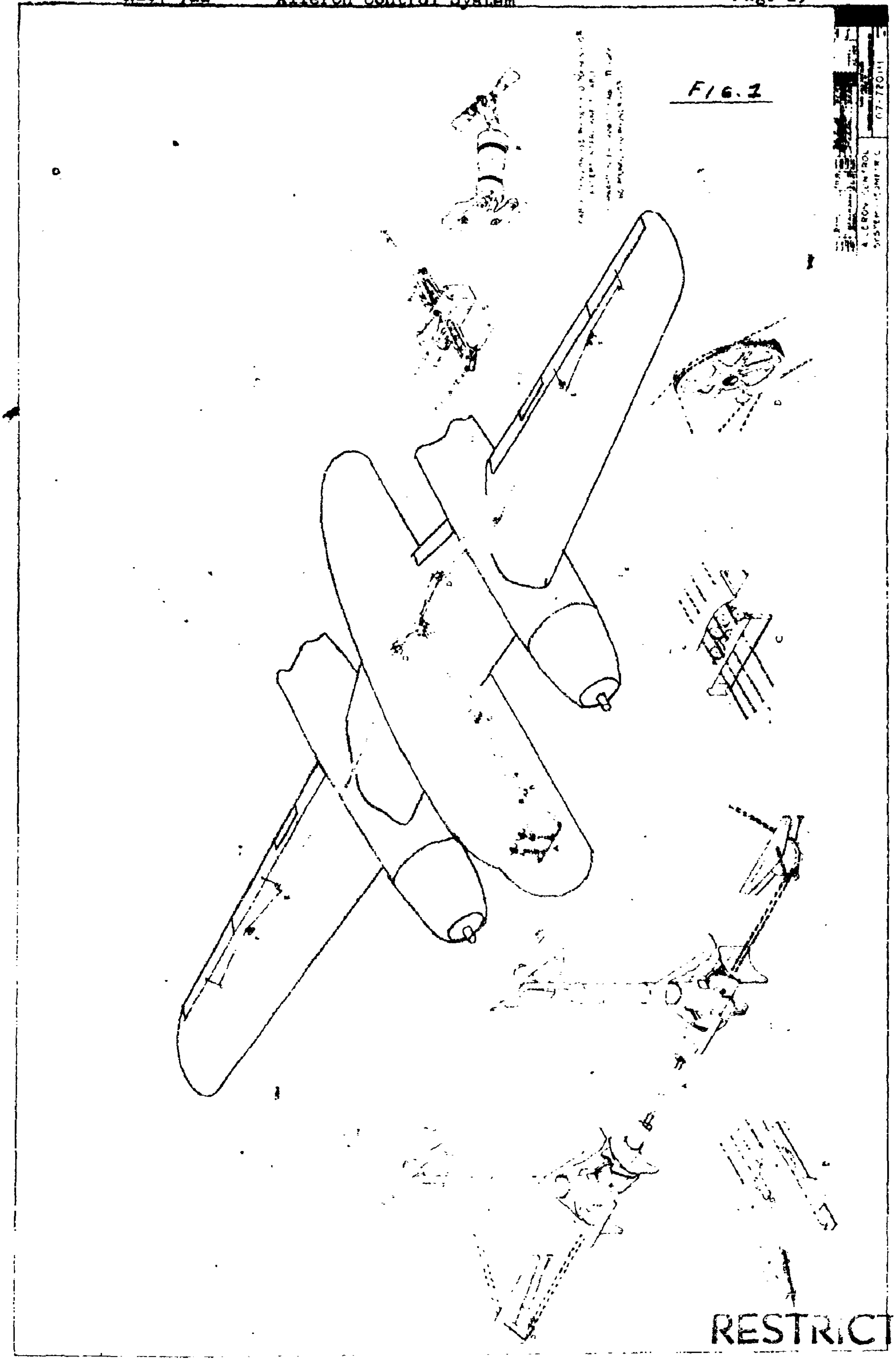
FIG. 1



REPORT NO. R107-722	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION		70 PAGES	PAGE 30
MODEL XC-120	PREPARED BY Bonk	CHECKED BY Bonk	APPROVED BY	

Subject:— AILERON CONTROL SYSTEM - BASIC DATA

DATE 9-2-49
REVISED * 1-10-50

FIG. 2

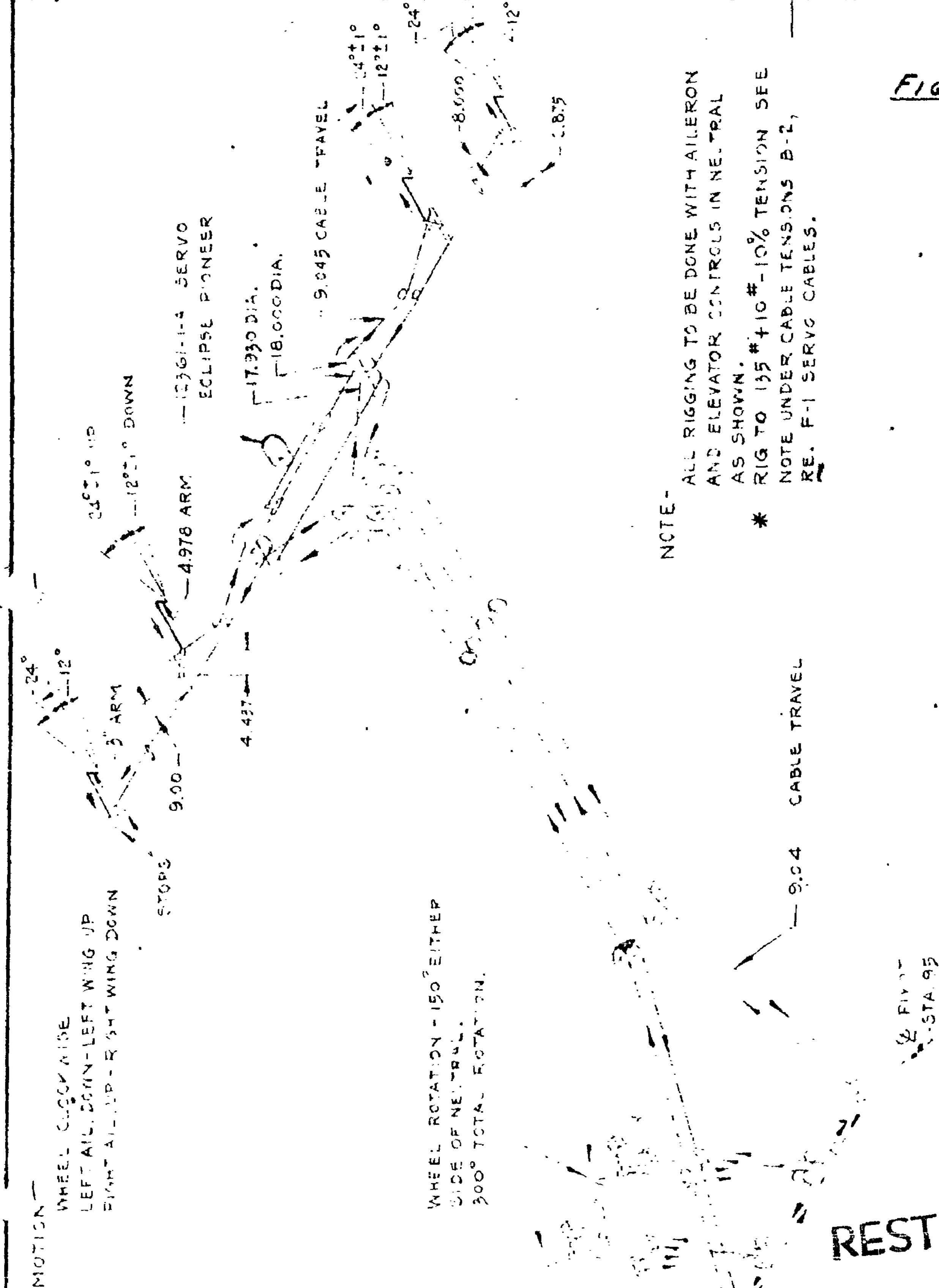


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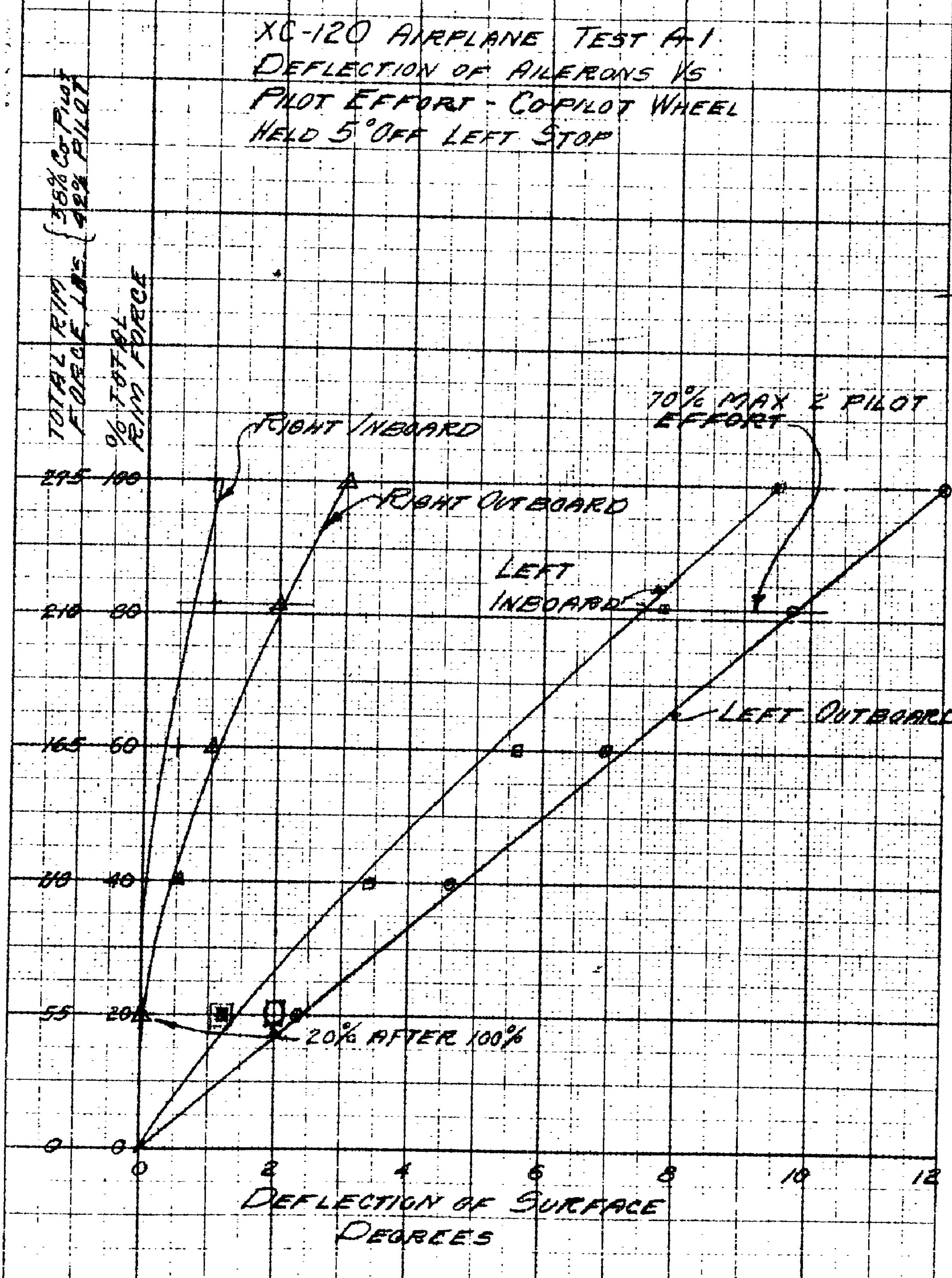
FIG 3


XC-120 AIRPLANE TEST A-1
DEFLECTION OF AILERONS VS
PILOT EFFORT - COPILOT WHEEL
HELD 5° OFF LEFT STOP
TOTAL RIM FORCE LBS {38% COPILOT 62% PILOT
% TOTAL RIM FORCE
RIGHT INBOARD
RIGHT OUTBOARD
70% MAX 2 PILOT EFFORT
LEFT INBOARD
LEFT OUTBOARD
20% AFTER 100%
295 100
210 80
165 60
110 40
55 20
0 0
0 2 4 6 8 10 12
DEFLECTION OF SURFACE
DEGREES

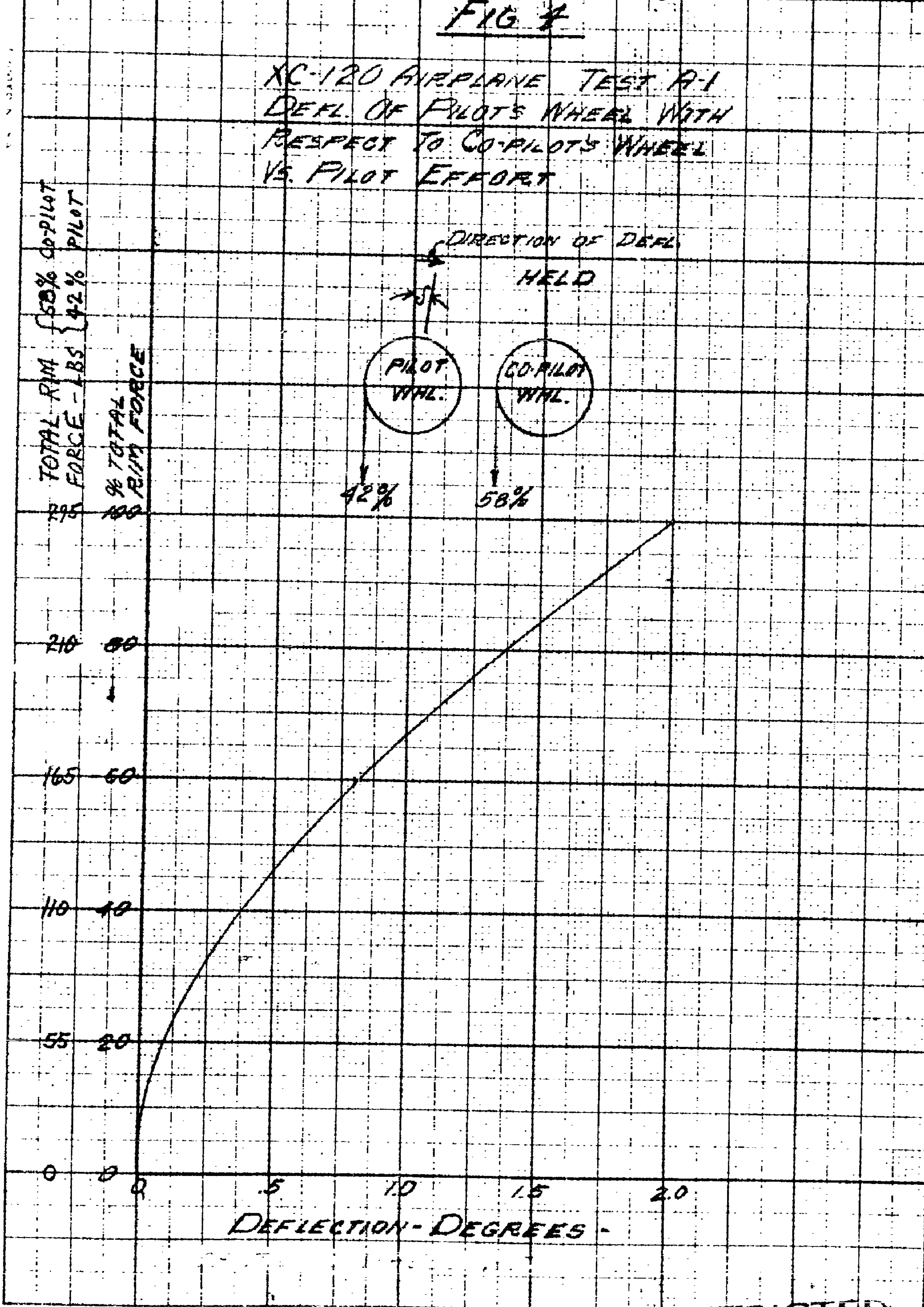
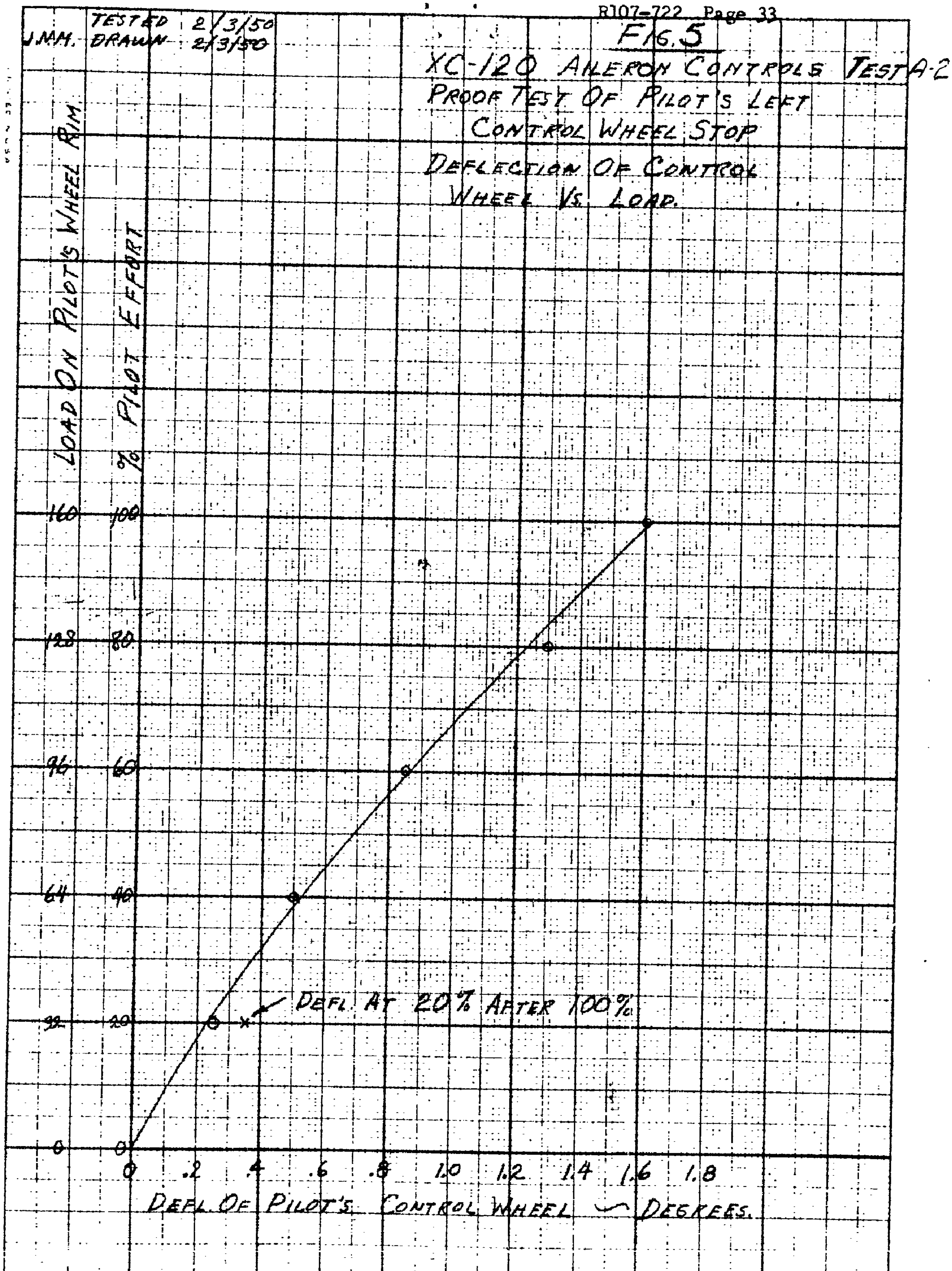


FIG 7
XC-120 AIRPLANE TEST A-1
DEFL. OF PILOTS WHEEL WITH
RESPECT TO CO-PILOTS WHEEL
VS. PILOT EFFORT
DIRECTION OF DEFL.
HELD
δ
PILOT WHL.
CO-PILOT WHL.
42%
58%
TOTAL RIM FORCE - LBS { 58% CO-PILOT 42% PILOT
% TOTAL RIM FORCE
275
210
165
110
55
0
100
80
60
40
20
0
0
.5
1.0
1.5
2.0
DEFLECTION - DEGREES -

RESTRICTED



TESTED 2/3/50
J.M.M. DRAWN 2/3/50
FIG. 5
XC-120 AILERON CONTROLS TEST A-2
PROOF TEST OF PILOT'S LEFT CONTROL WHEEL STOP
DEFLECTION OF CONTROL WHEEL VS. LOAD.
LOAD ON PILOT'S WHEEL RIM
% PILOT EFFORT
160 100
128 80
96 60
64 40
32 20
0 0
DEFL. AT 20% AFTER 100%
0 .2 .4 .6 .8 1.0 1.2 1.4 1.6 1.8
DEFL. OF PILOT'S CONTROL WHEEL ~ DEGREES.

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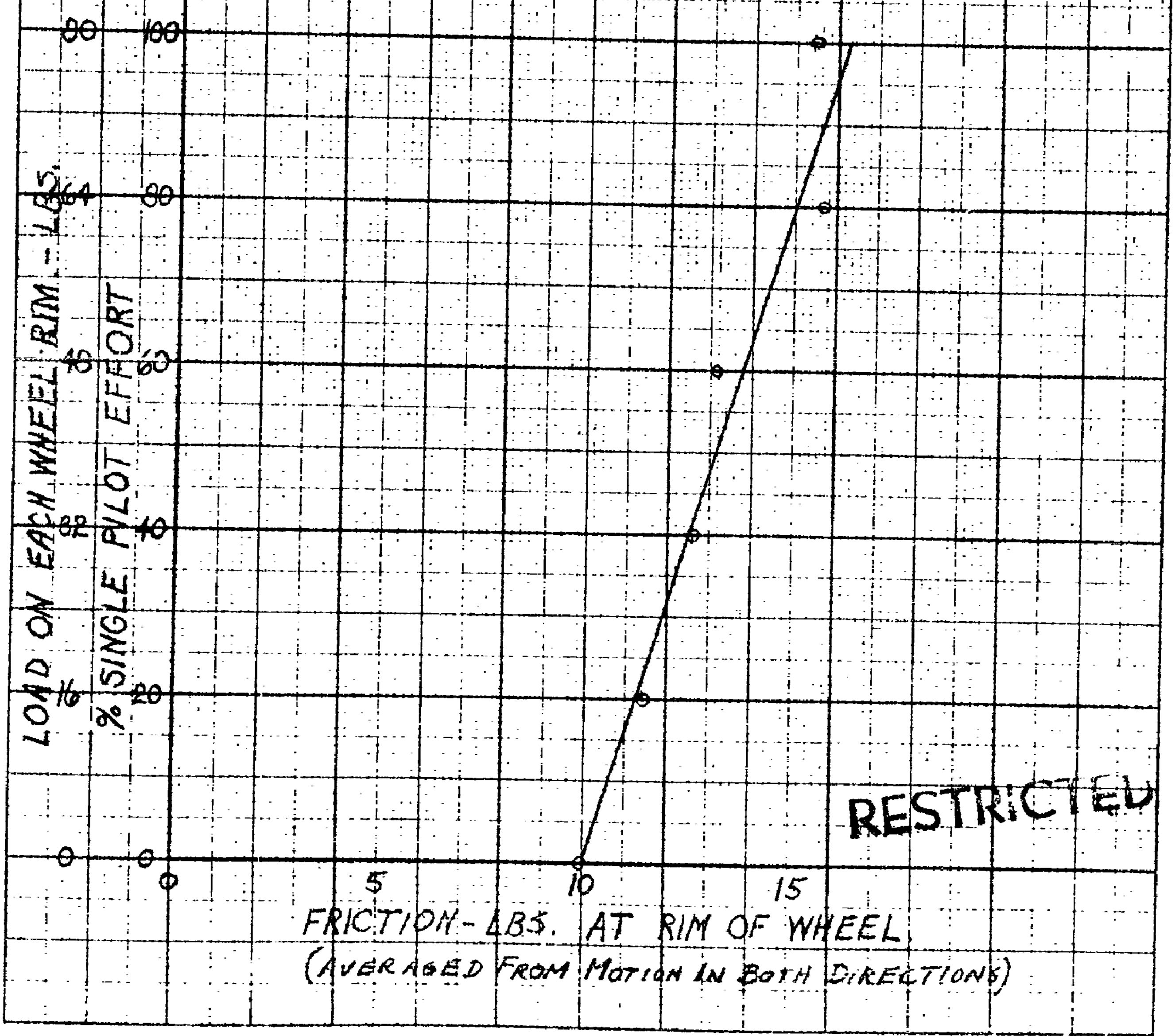
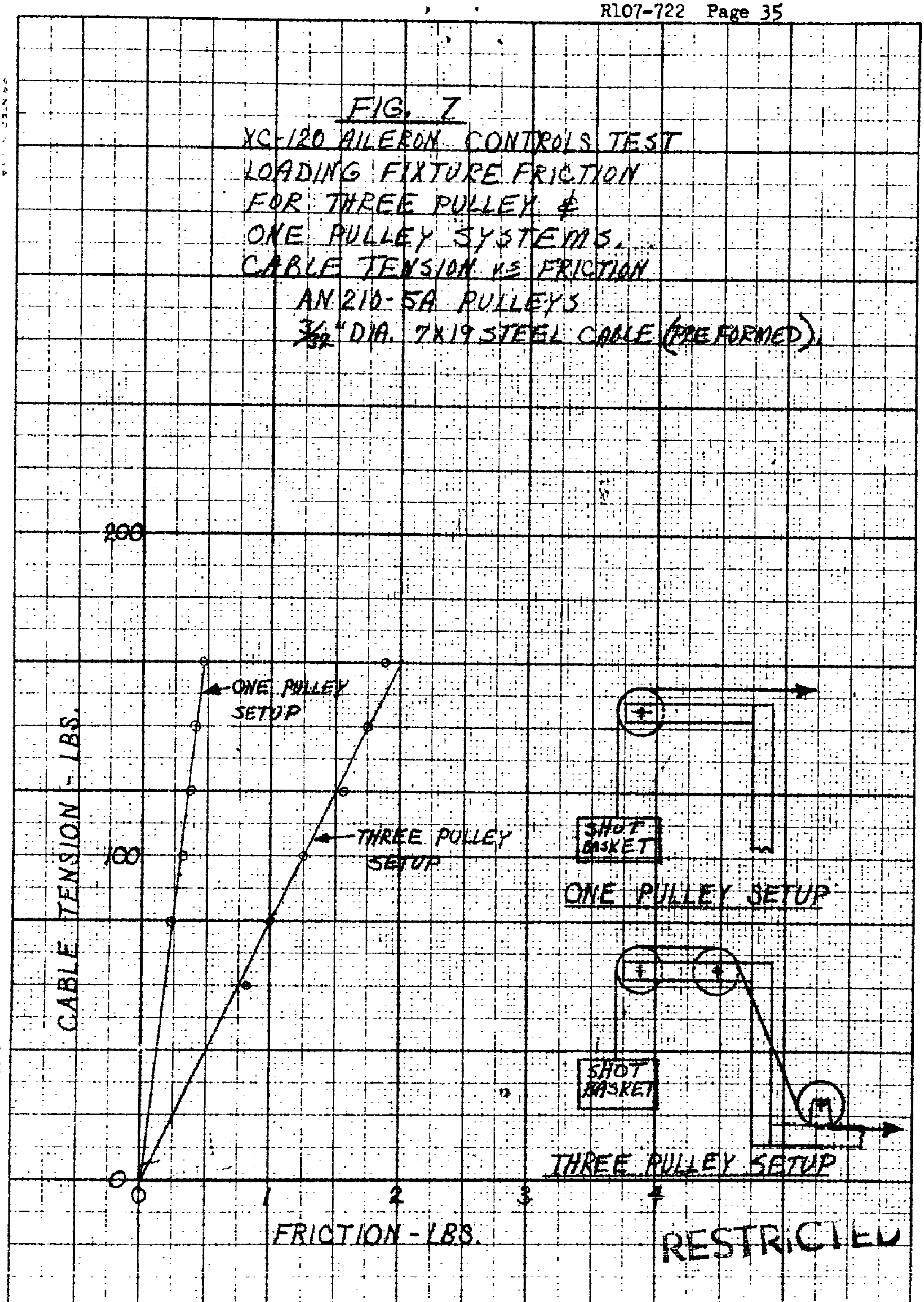
FIG. 6

XC-120 AILERON CONTROLS TEST (A-3)
FRICTION IN AILERON CONTROL SYSTEM
VS. PILOT EFFORT.
FRICTION MEASURED AS CONTROLS
MOVED THROUGH NEUTRAL AND
CORRECTED FOR FRICTION IN THE
LOADING DEVICES.

SURFACE HINGE MOMENTS AT 100%
R.O.B. = 1720 IN.-LB.
R.I.B. = 3530 IN.-LB.
~~L.I.B. = 2880 IN.-LB.~~
L.O.B. = 1422 IN.-LB.
BALANCED WITH 80 LB. WHEEL RIM LOAD
PER PILOT & CO-PILOT WHEEL.



EFK 2-7-50



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GFK 4-7-50

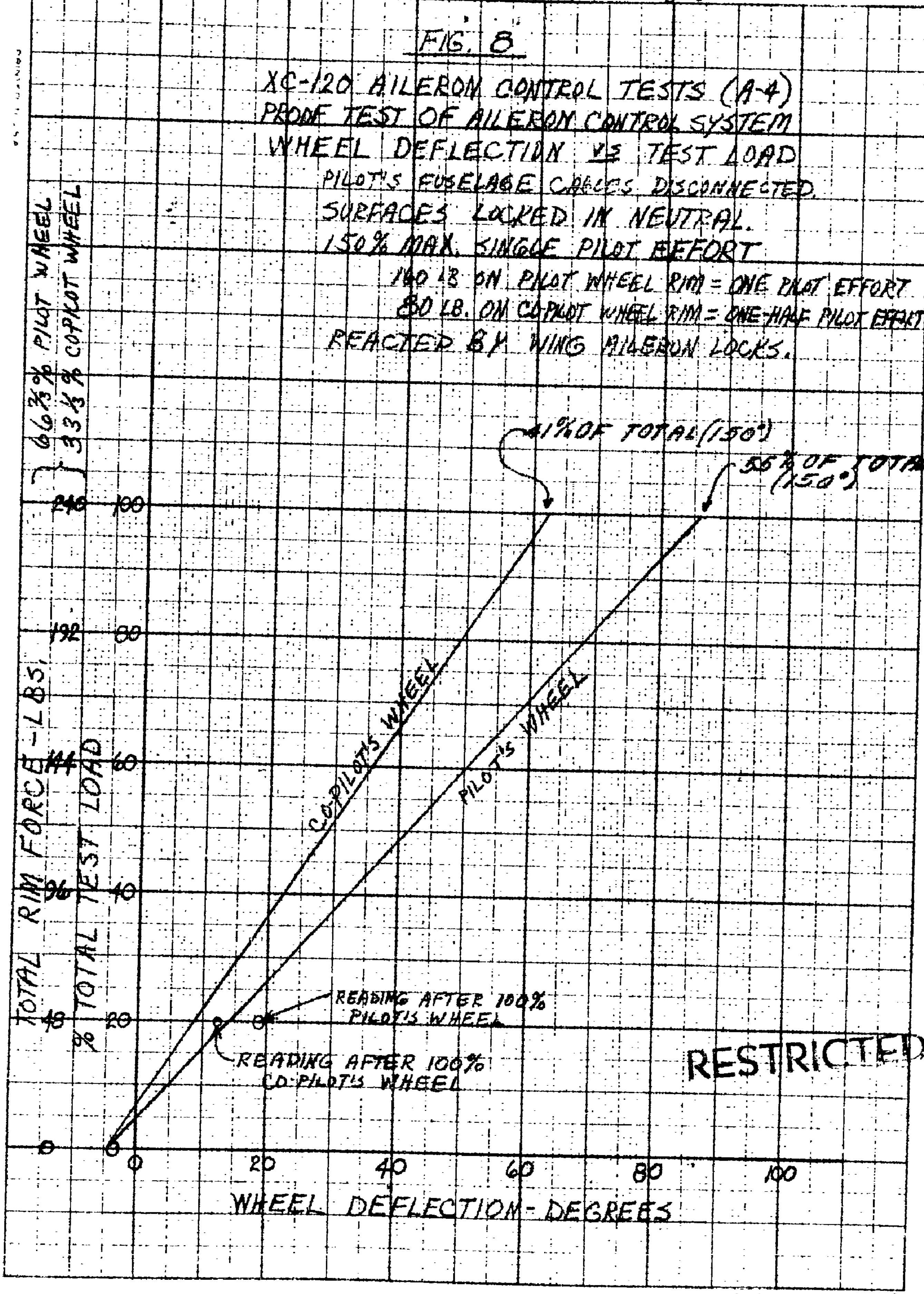
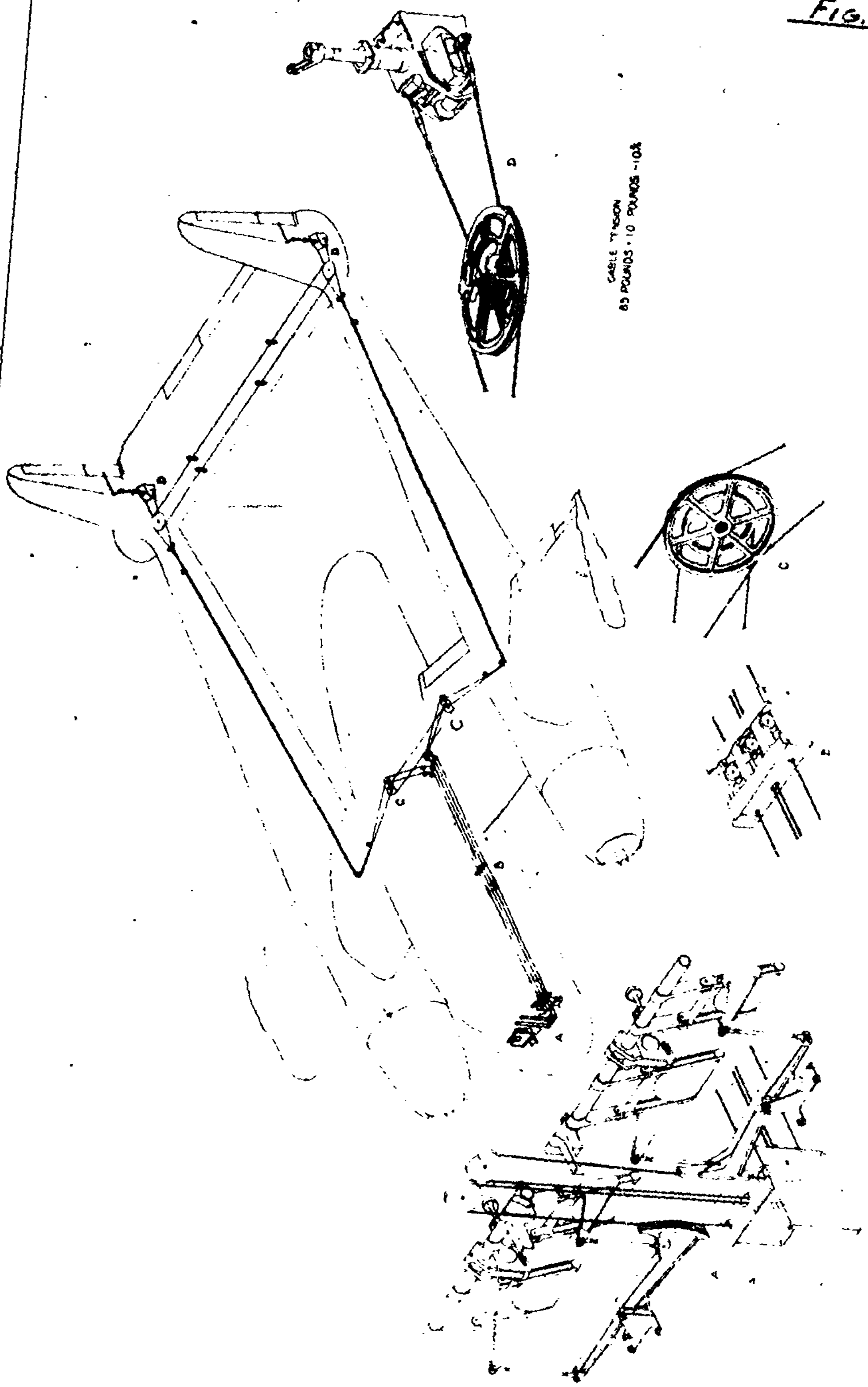


FIG. 8
XC-120 AILERON CONTROL TESTS (A-4)
PROOF TEST OF AILERON CONTROL SYSTEM
WHEEL DEFLECTION VS TEST LOAD
PILOT'S FUSELAGE CABLES DISCONNECTED.
SURFACES LOCKED IN NEUTRAL.
150% MAX. SINGLE PILOT EFFORT
140 LB ON PILOT WHEEL RIM = ONE PILOT EFFORT
80 LB. ON CO-PILOT WHEEL RIM = ONE-HALF PILOT EFFORT
REACTED BY WING AILERON LOCKS.
41% OF TOTAL (150°)
56% OF TOTAL (150°)
CO-PILOT'S WHEEL
PILOT'S WHEEL
READING AFTER 100% PILOT'S WHEEL
READING AFTER 100% CO-PILOT'S WHEEL
RESTRICTED
TOTAL RIM FORCE - LBS.
66 2/3% PILOT WHEEL
33 1/3% COPILOT WHEEL
% TOTAL TEST LOAD
WHEEL DEFLECTION - DEGREES
0
48
96
144
192
240
20
40
60
80
100

GFK 2-3-50

Fig. 9



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Subject: RUDDER CONTROL SYSTEM - BASIC DATA

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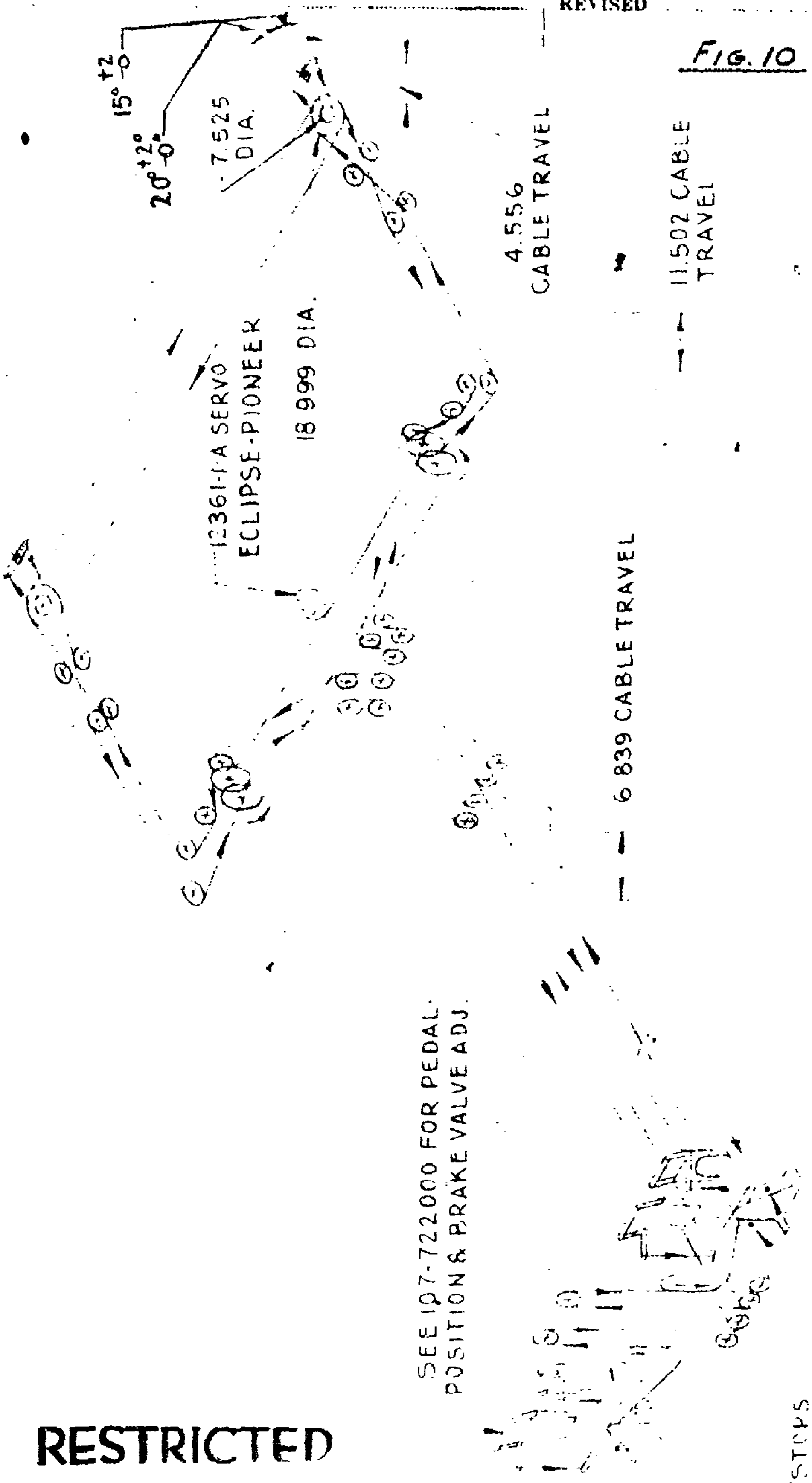
FIG. 10



NOTE: ALL RIGGING TO BE DONE IN NEUTRAL AS SHOWN - RIG TO 85# + 10# - 10# TENSION

OPERATION

LEFT PEDAL FWD - SURFACE LEFT - NOSE LEFT.

RIGHT PEDAL FWD - SURFACE RT. - NOSE RIGHT.

MOTION

RUDDER MOTION $15^{\circ}\,{}^{+2}_{-0}$ LEFT & $20^{\circ}\,{}^{+2^{\circ}}_{-0}$ RIGHT

(SEE SPRING TAB DIAGRAM 107-728575 REF.

NOTES.

2.00 FORE & AFT ADJ. OF PEDALS POSSIBLE FROM STA 80.75

LEFT PEDAL TRAVELS 3.601 FWD & 4.347 AFT.

RIGHT PEDAL TRAVELS 4.347 FWD & 3.601 AFT TO

PRODUCE RUDDER MOTION.

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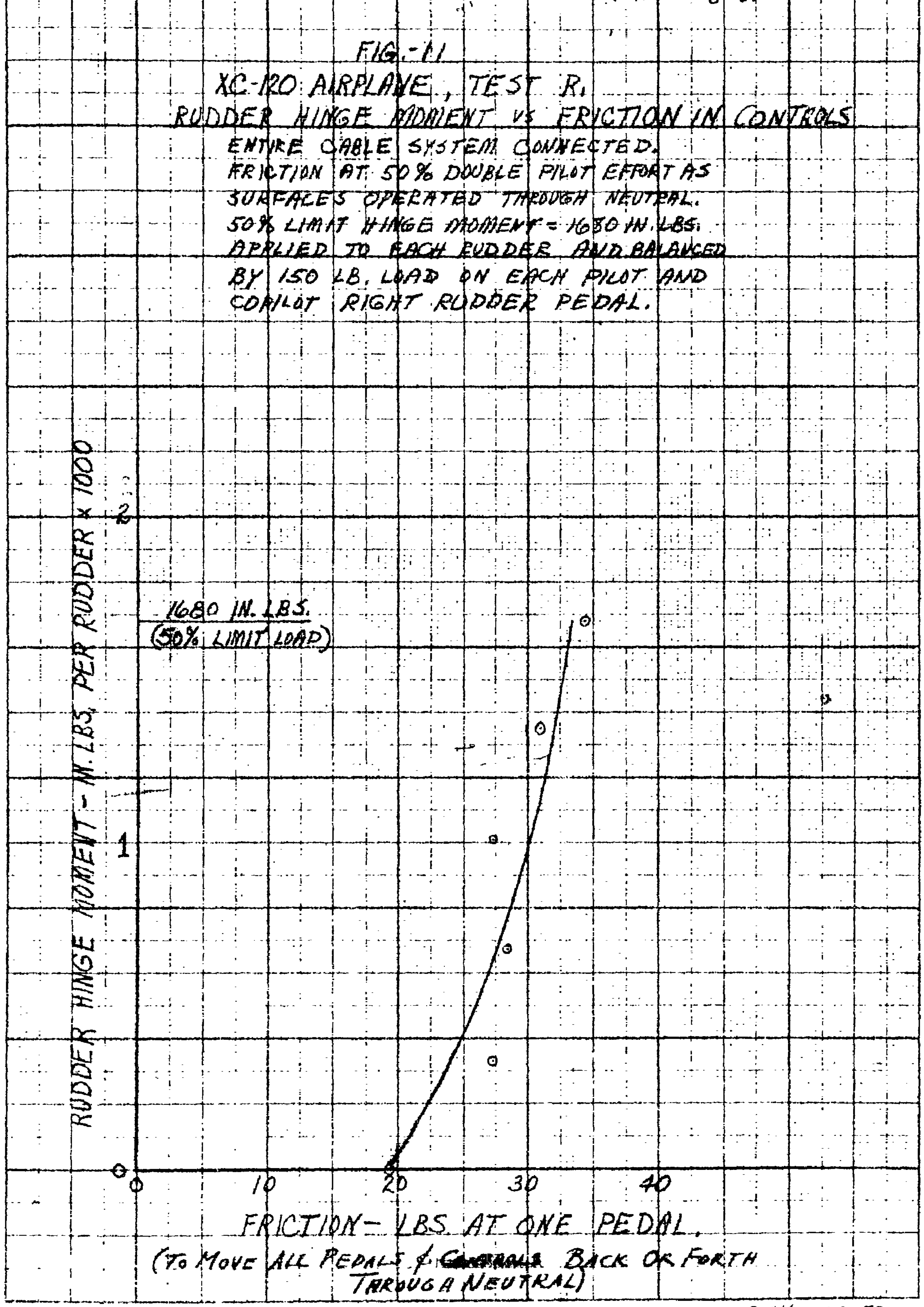


FIG.-11
XC-120 AIRPLANE, TEST R.
RUDDER HINGE MOMENT vs FRICTION IN CONTROLS
ENTIRE CABLE SYSTEM CONNECTED.
FRICTION AT 50% DOUBLE PILOT EFFORT AS
SURFACES OPERATED THROUGH NEUTRAL.
50% LIMIT HINGE MOMENT = 1680 IN. LBS.
APPLIED TO EACH RUDDER AND BALANCED
BY 150 LB. LOAD ON EACH PILOT AND
COPILOT RIGHT RUDDER PEDAL.
1680 IN. LBS.
(50% LIMIT LOAD)
RUDDER HINGE MOMENT - IN. LBS. PER RUDDER × 1000
2
1
0
10
20
30
40
FRICTION - LBS AT ONE PEDAL.
(TO MOVE ALL PEDALS & ~~CONTROLS~~ BACK OR FORTH
THROUGH NEUTRAL)

EFK 1-3-50

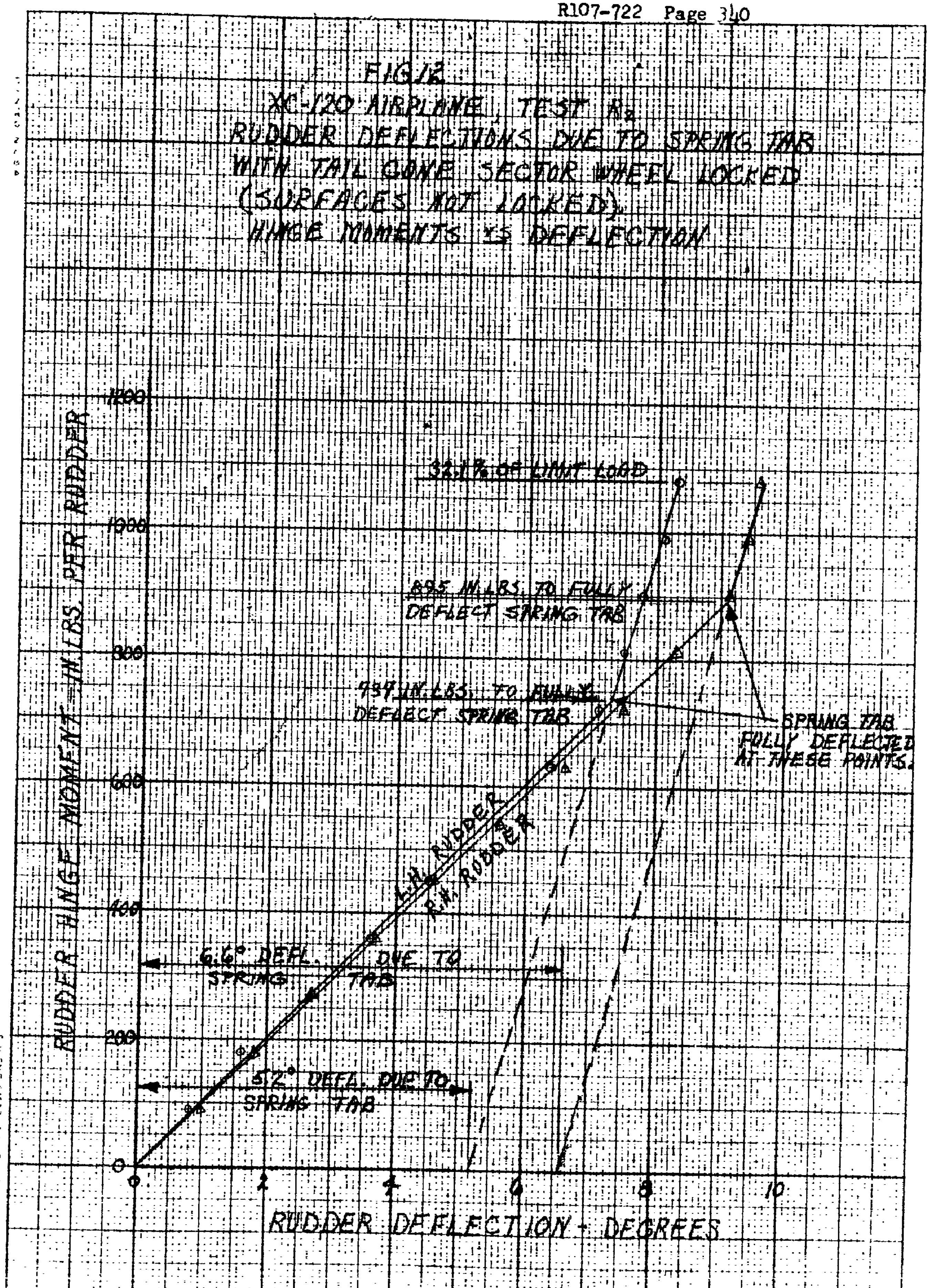
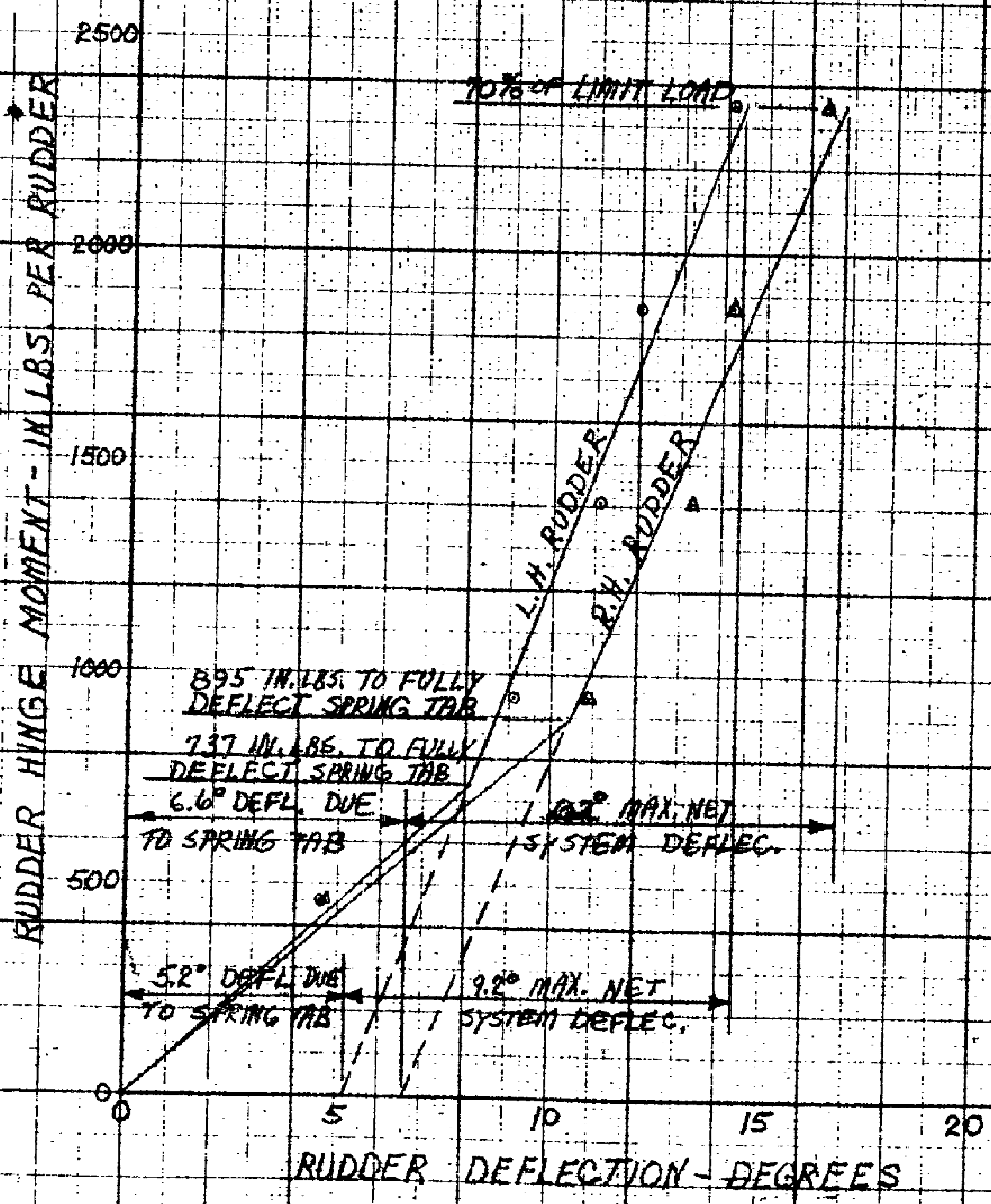


FIG 12
XC-120 AIRPLANE, TEST K2
RUDDER DEFLECTIONS DUE TO SPRING TAB
WITH TAIL CONE SECTOR WHEEL LOCKED
(SURFACES NOT LOCKED)
HINGE MOMENTS VS DEFLECTION
32.1% OF LIMIT LOAD
895 IN.LBS. TO FULLY DEFLECT SPRING TAB
759 IN.LBS. TO FULLY DEFLECT SPRING TAB
SPRING TAB FULLY DEFLECTED AT THESE POINTS.
L.H. RUDDER
R.H. RUDDER
6.6° DEFL. DUE TO SPRING TAB
5.2° DEFL. DUE TO SPRING TAB
RUDDER HINGE MOMENT - IN.LBS. PER RUDDER
RUDDER DEFLECTION - DEGREES
1200
1000
800
600
400
200
0
0
2
4
6
8
10

GFK 12-30-49

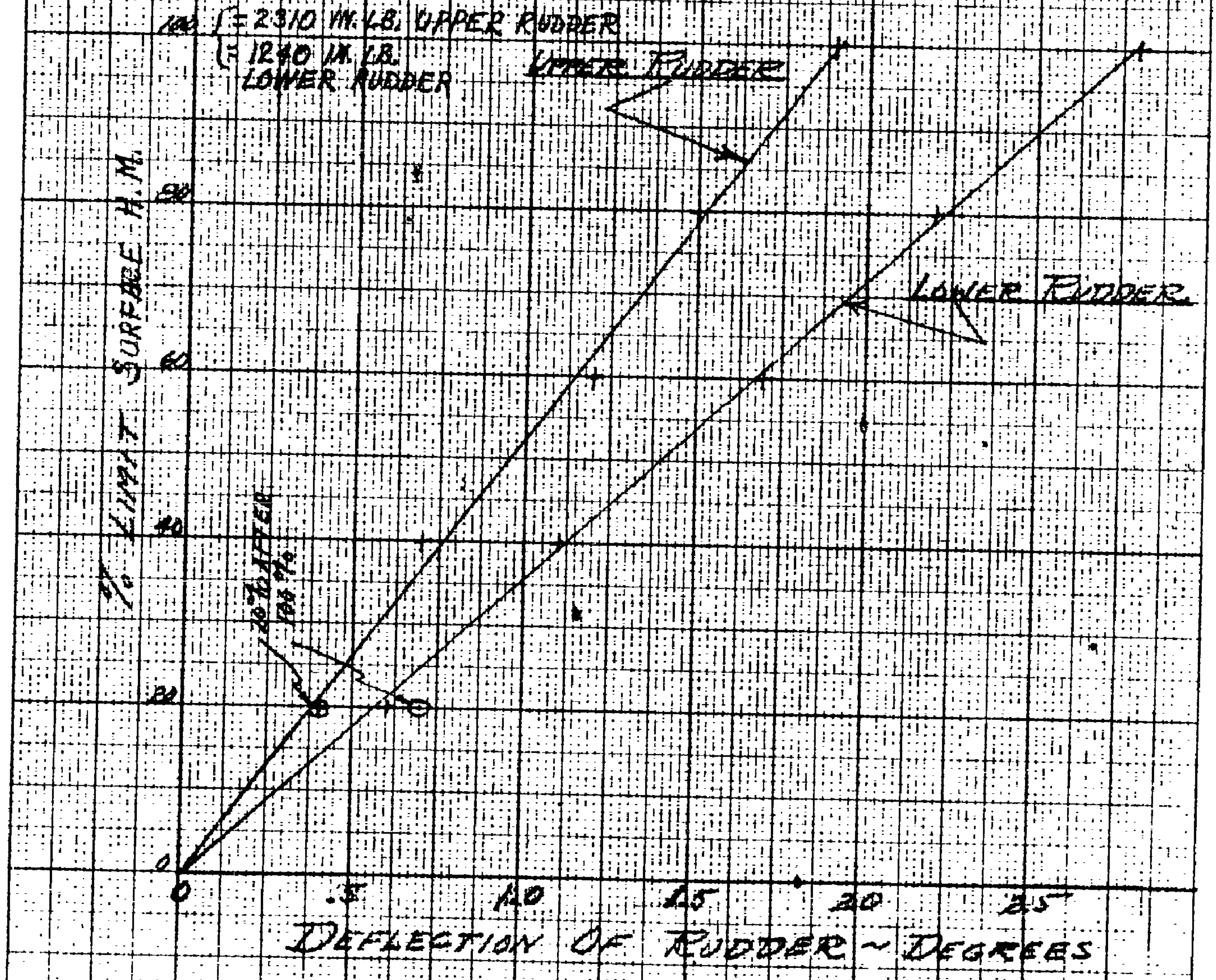
FIG 13
XC-120 AIRPLANE, TEST R_2
PLOT OF GROSS DEFLECTIONS
(SPRING TAB & CONTROL SYSTEM)
FOR 70% LIMIT LOAD (70% OF
TWO PILOTS MAX. EFFORT)
2350 IN. LB. HINGE MOMENT
PER RUDDER SURFACE



EFK 12-30-49

FIG. -14 -

XC-120 RUDDER CONTROL SYSTEM
TEST-#R3 - PROOF TEST OF RUDDER
SURFACE LOCKS
% LIMIT SURFACE H.M. vs SURFACE DEFLECTION.



Long
12-22-4

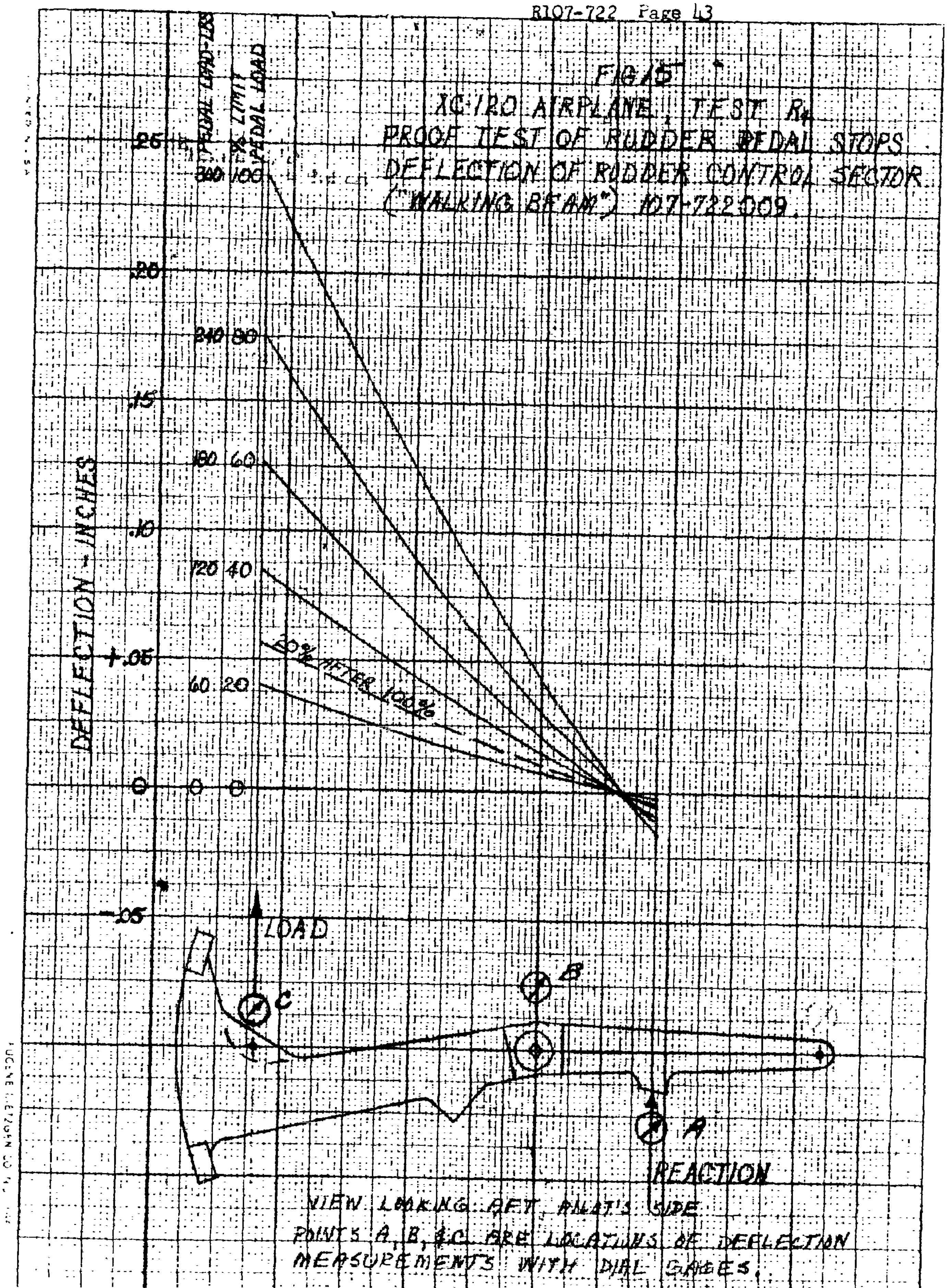
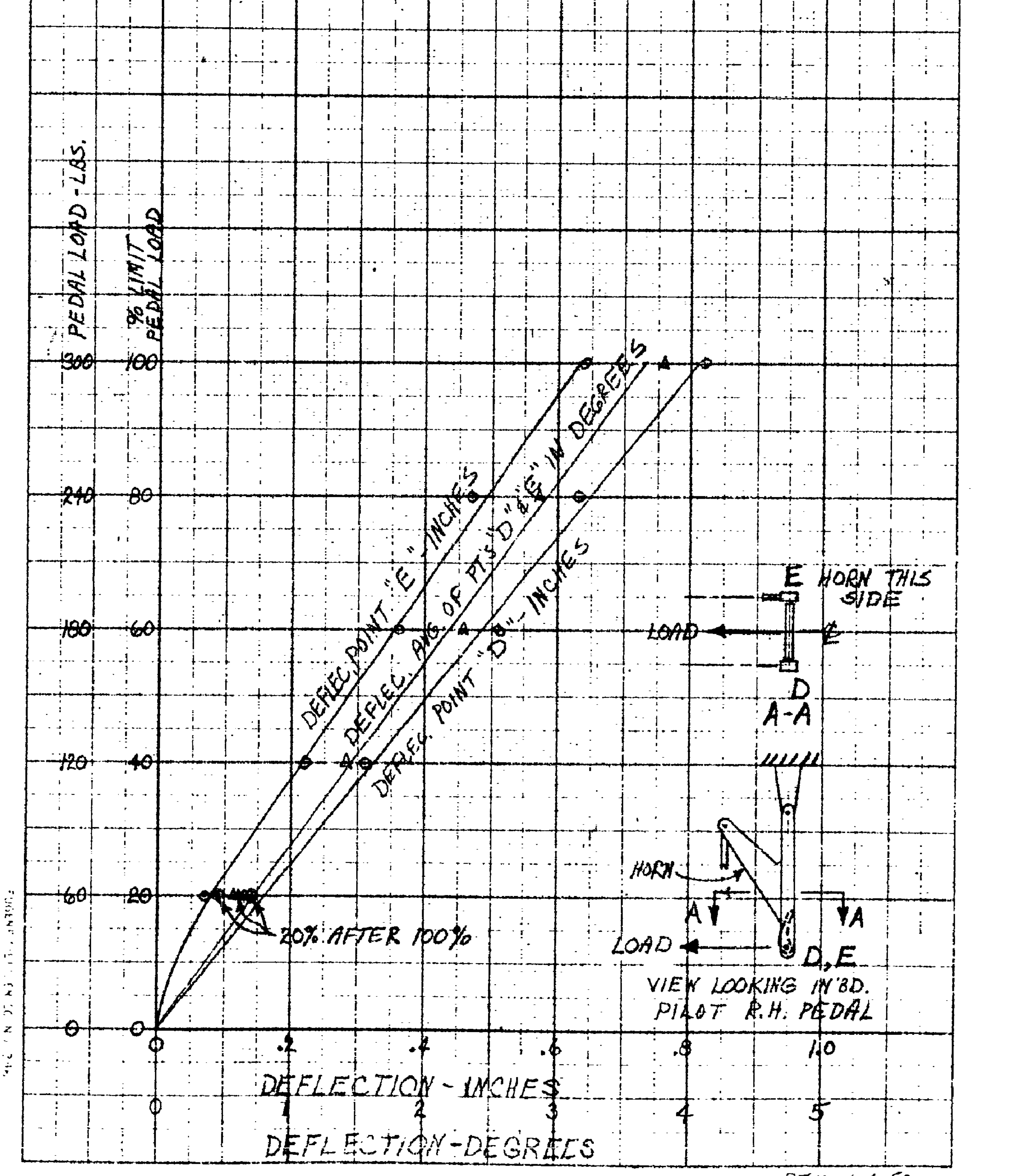


FIG 15
XC-120 AIRPLANE, TEST R4
PROOF TEST OF RUDDER PEDAL STOPS.
DEFLECTION OF RUDDER CONTROL SECTOR
("WALKING BEAM") 107-722009.
DEFLECTION - INCHES
PEDAL LOAD-LBS
% LIMIT PEDAL LOAD
.25
.20
.15
.10
+.05
0
-.05
300 100
240 80
180 60
120 40
60 20
20% AFTER 100%
LOAD
C
B
A
REACTION
VIEW LOOKING AFT, PILOT'S SIDE
POINTS A, B, & C ARE LOCATIONS OF DEFLECTION
MEASUREMENTS WITH DIAL GAGES.

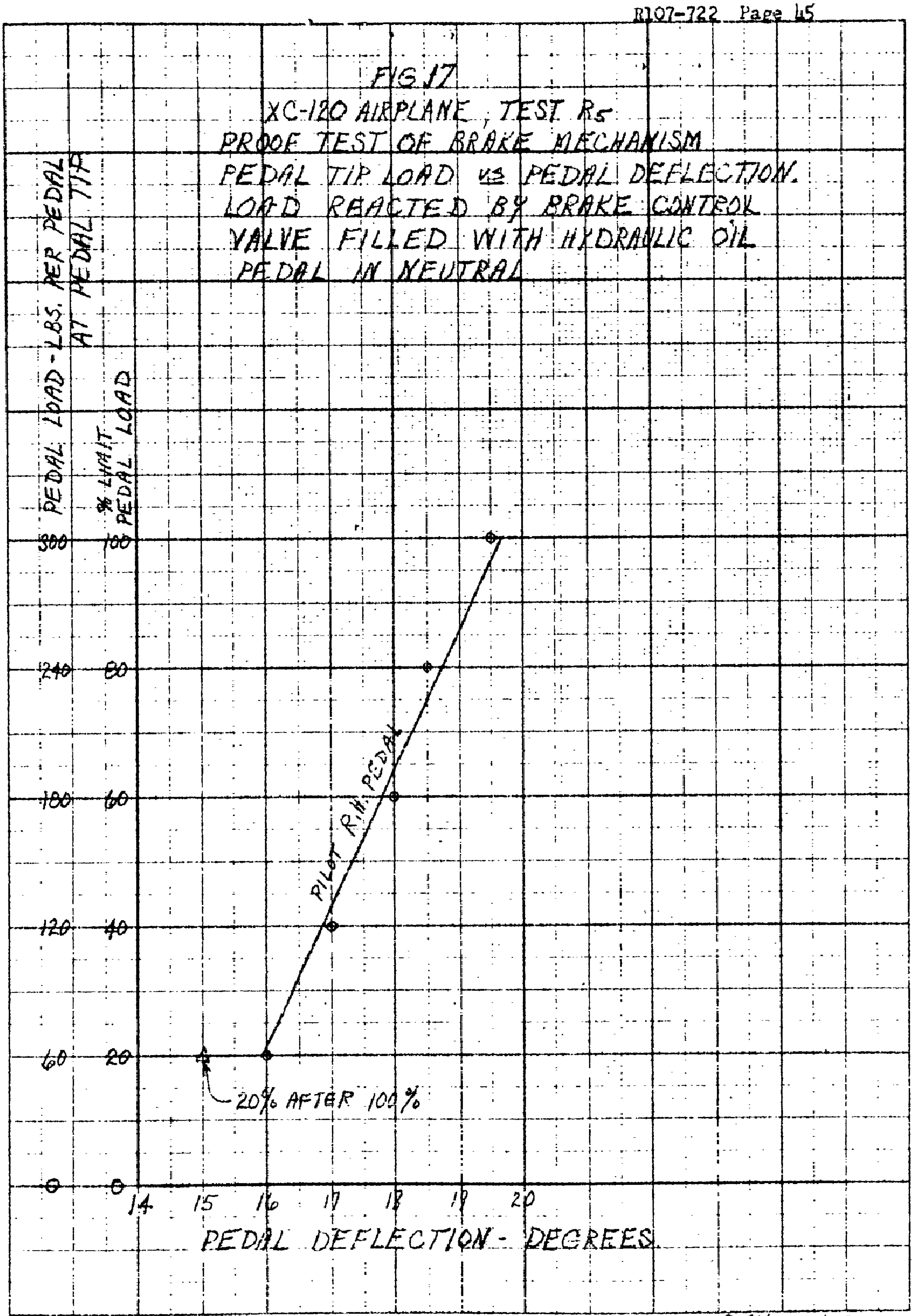
6FK 1-4-50

FIG. 16
XC-120 AIRPLANE, TEST R4
PROOF TEST OF RUDDER PEDAL STOPS.
GROSS DEFLECTION OF RUDDER PEDAL HANGER ASS'Y.



EFK 1-4-50

FIG 17
XC-120 AIRPLANE, TEST R5
PROOF TEST OF BRAKE MECHANISM
PEDAL TIP LOAD vs PEDAL DEFLECTION.
LOAD REACTED BY BRAKE CONTROL
VALVE FILLED WITH HYDRAULIC OIL
PEDAL IN NEUTRAL



EFK 1-4-50

FIG. 18
XC-120 AIRPLANE, TEST R_5
PROOF TEST OF BRAKE MECHANISM
PEDAL TIP LOAD vs HANGER SUPPORT DEFLECTION
RIGHT HAND PEDAL LOAD REACTED BY BRAKE CONTROL VALVE FILLED WITH OIL. LEFT HAND PEDAL LOAD REACTED BY STEEL LINK IN PLACE OF BRAKE CONTROL VALVE. PEDALS IN NEUTRAL.

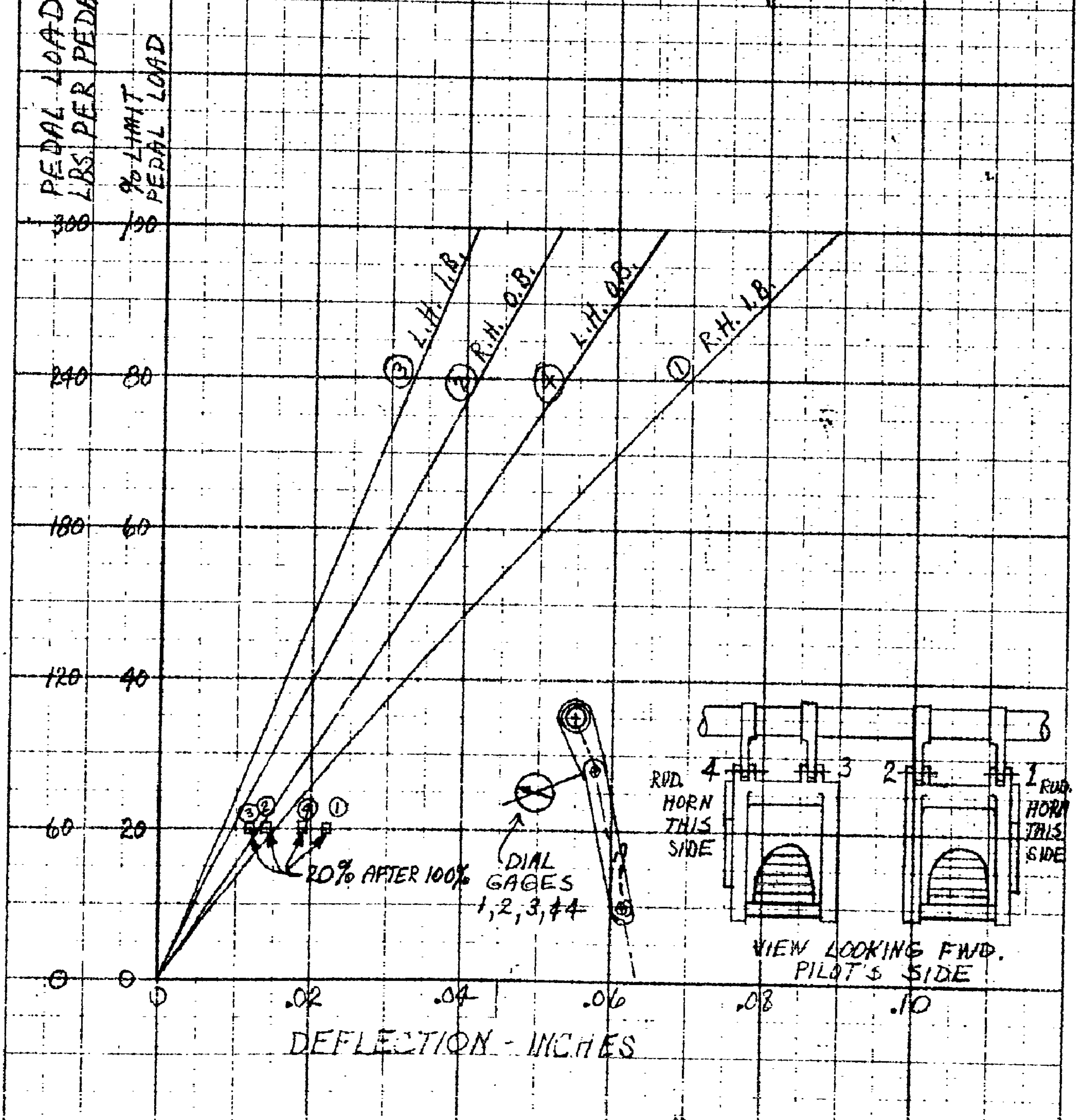


FIG. 19

XC-120 AIRPLANE, TEST R6

PROOF TEST OF ENTIRE RUDDER CONTROL SYSTEM

HINGE MOMENT vs RUDDER DEFLECTION

RIGHT BOOM CABLES DISCONNECTED

150% PILOT EFFORT APPLIED TO SYSTEM.

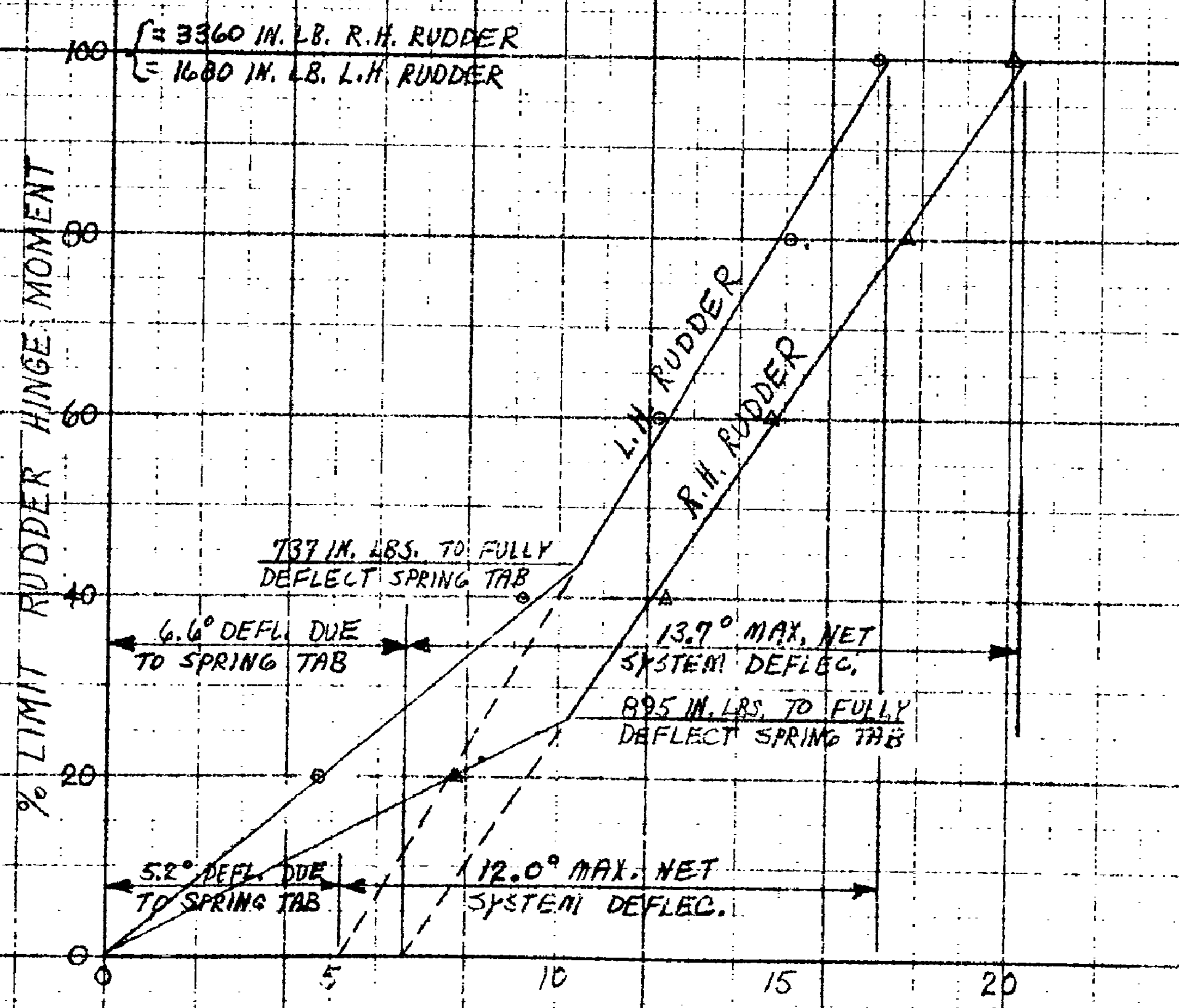
100% LIMIT H.M. ON RIGHT RUDDER = 3360 IN. LB.

~~50% LIMIT H.M. ON LEFT RUDDER = 1680 IN. LB.~~

RESISTED IN COCKPIT WITH

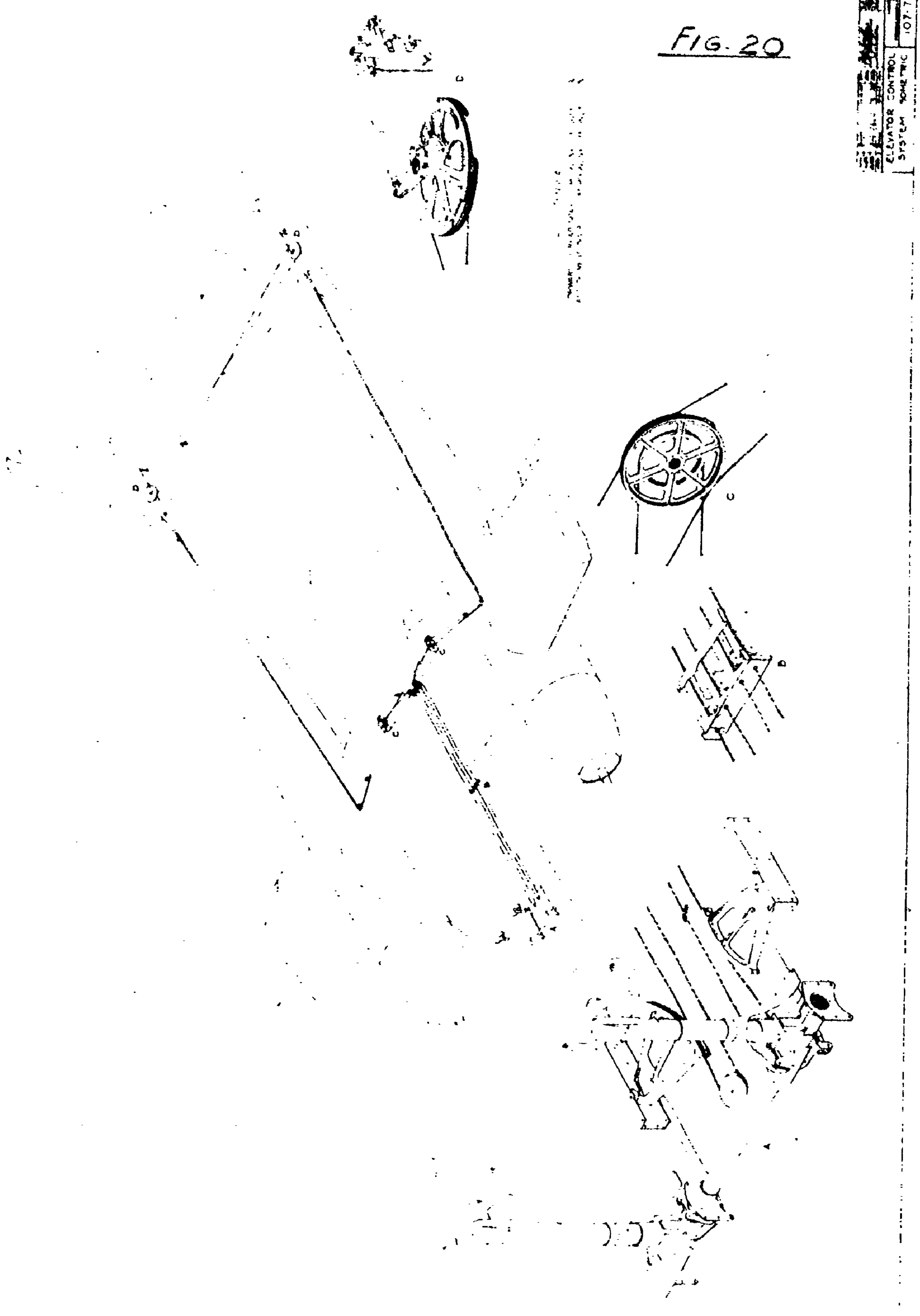
100% LIMIT LOAD ON CO-PILOT'S LEFT PEDAL = 300 LB.

50% LIMIT LOAD ON PILOT'S LEFT PEDAL = 150 LB.



EFK 1-5-50

FIG. 20



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MODEL XC-120	PREPARED BY BONK	CHECKED BY BONK	APPROVED BY
Subject: ELEVATOR CONTROL SYSTEM - BASIC DATA			DATE 9-2-49 · REVISED * 1-10-50

FIG 21

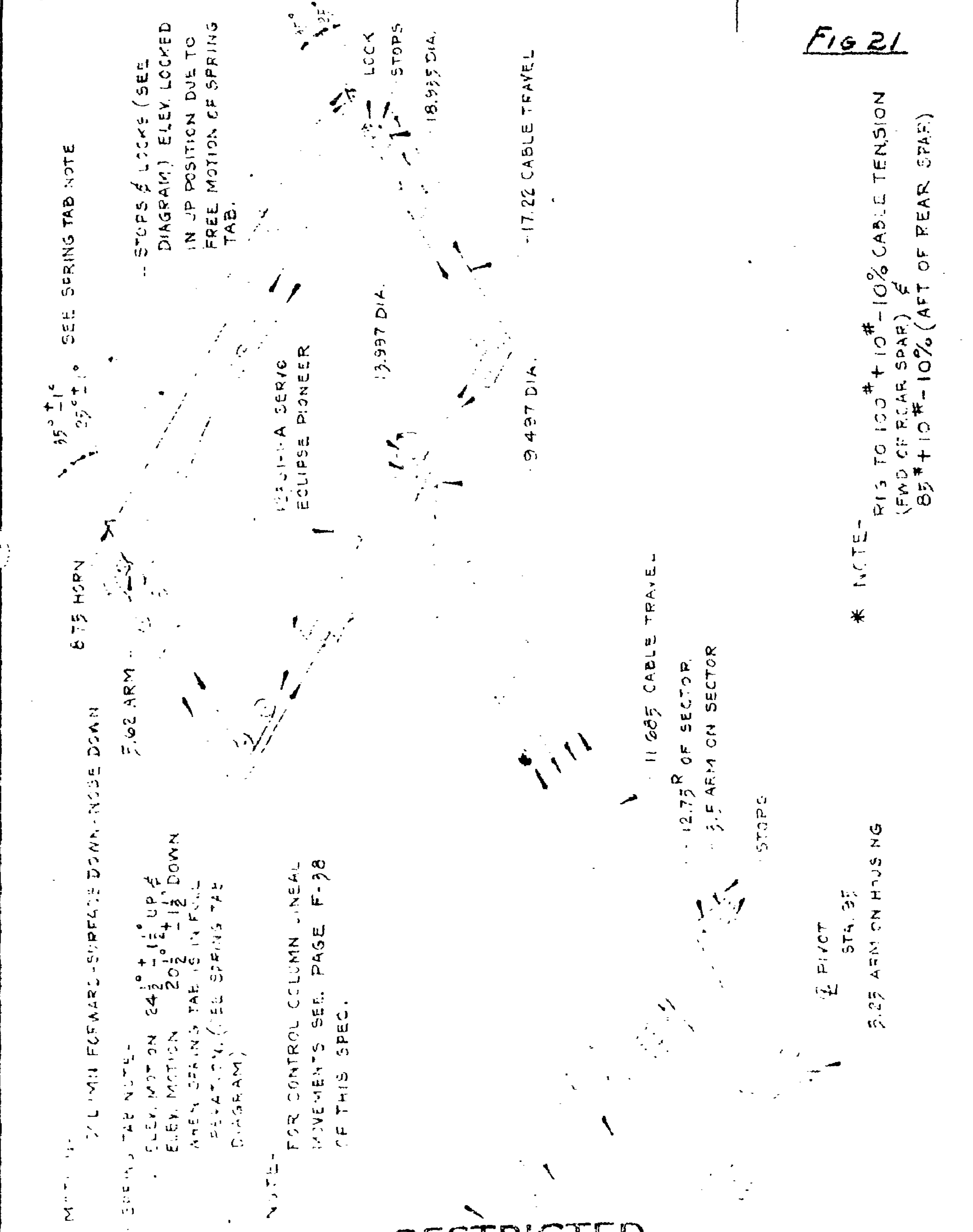
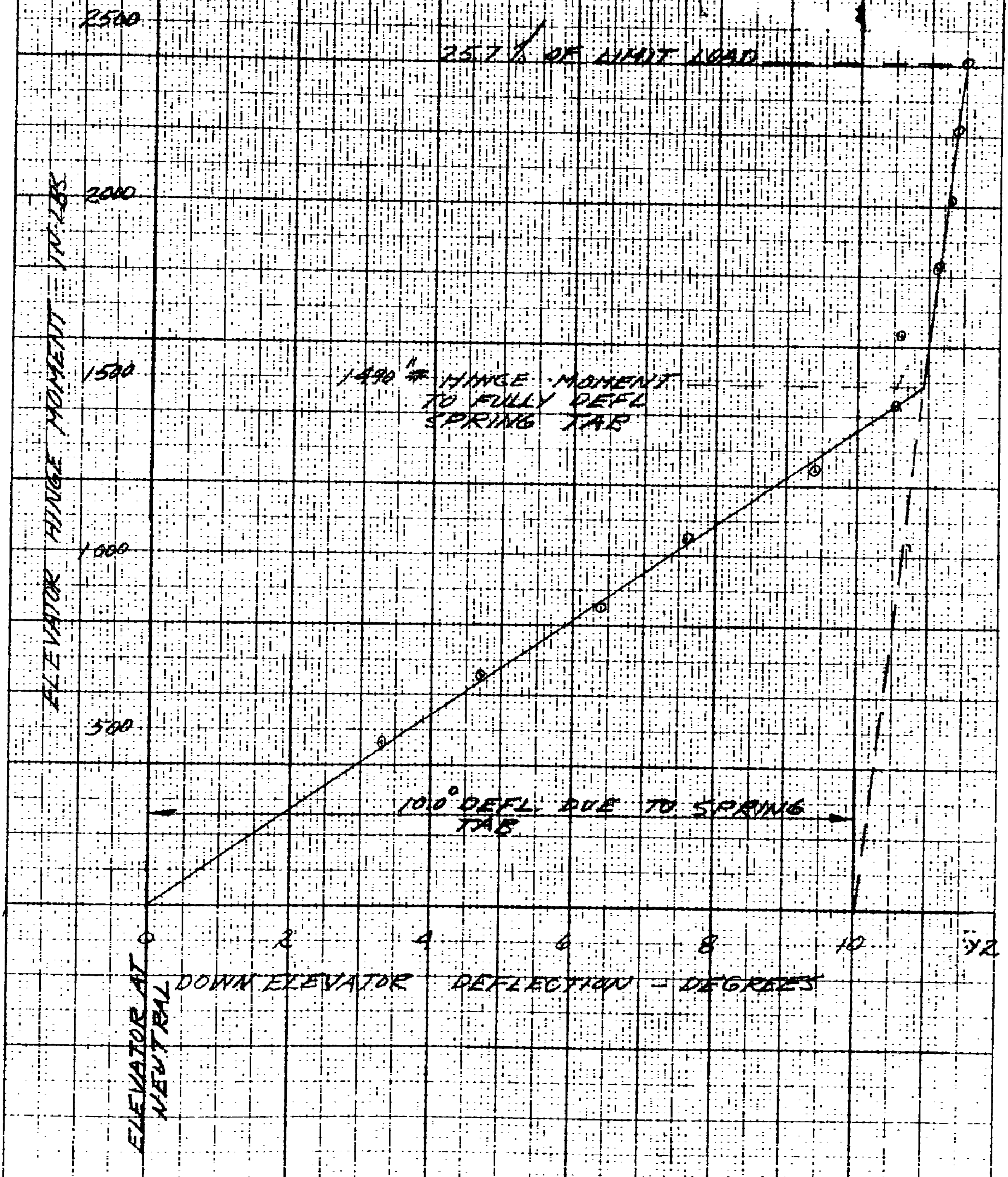


FIG 22

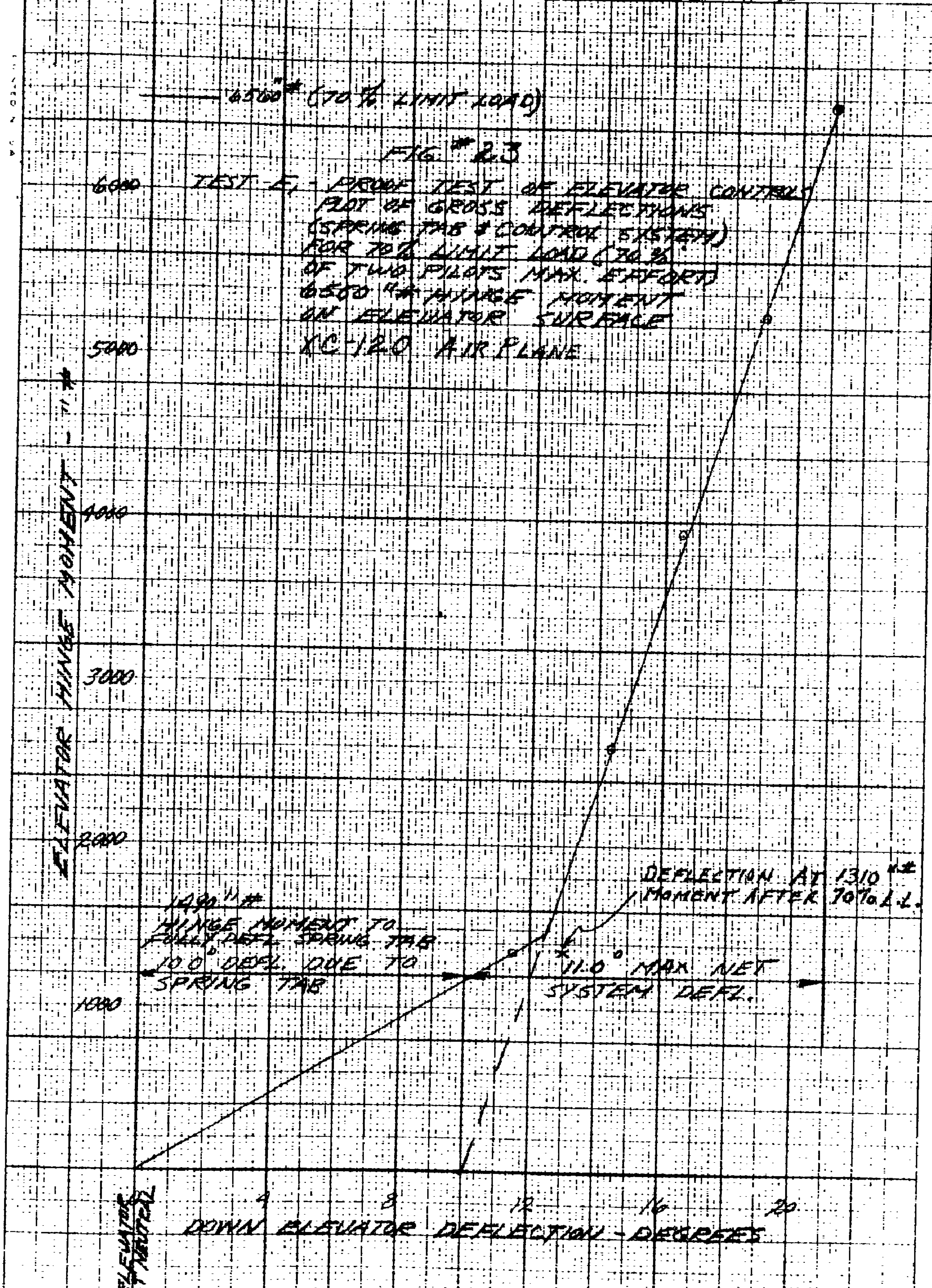
TEST E, - PROOF TEST OF ELEVATOR CONTROLS - PLOT OF ELEVATOR DEFLECTIONS DUE TO SPRING TAB WITH TAIL CONE SECTION LOCKED

HINGE MOMENT VS DEFLECTION

XC-120 AIRPLANE



12-28-47



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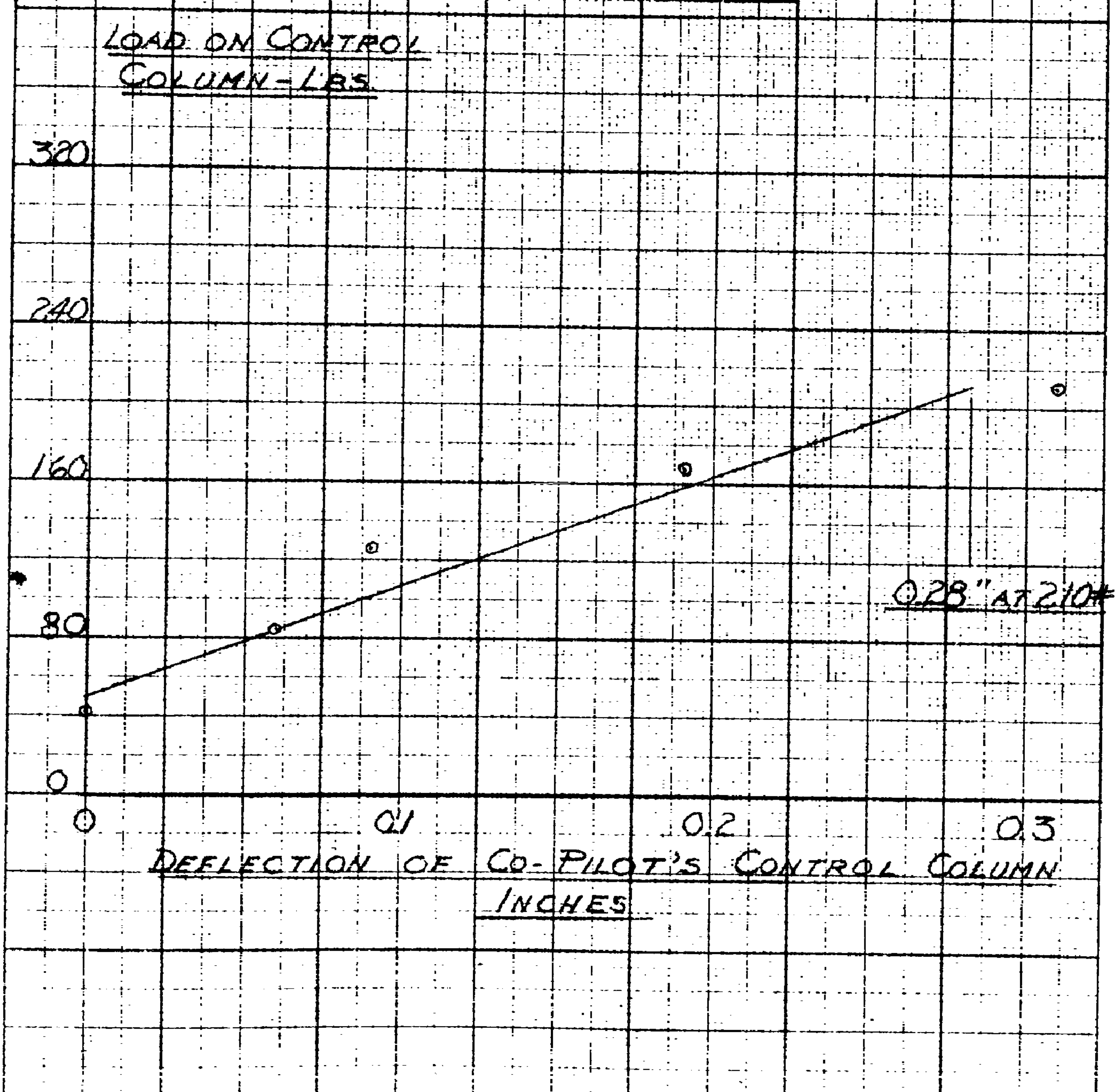
12-21-49

FIG. 24

TEST E1 - PROOF TEST OF ELEVATOR CONTROLS XC-120

DEFLECTION OF THE TOP
OF THE CO-PILOT'S CONTROL
COLUMN RELATIVE TO BASE
1. BASE PIVOT HELD FIXED
2. LOAD APPLIED AFT FROM
CENTER OF CONTROL
WHEEL



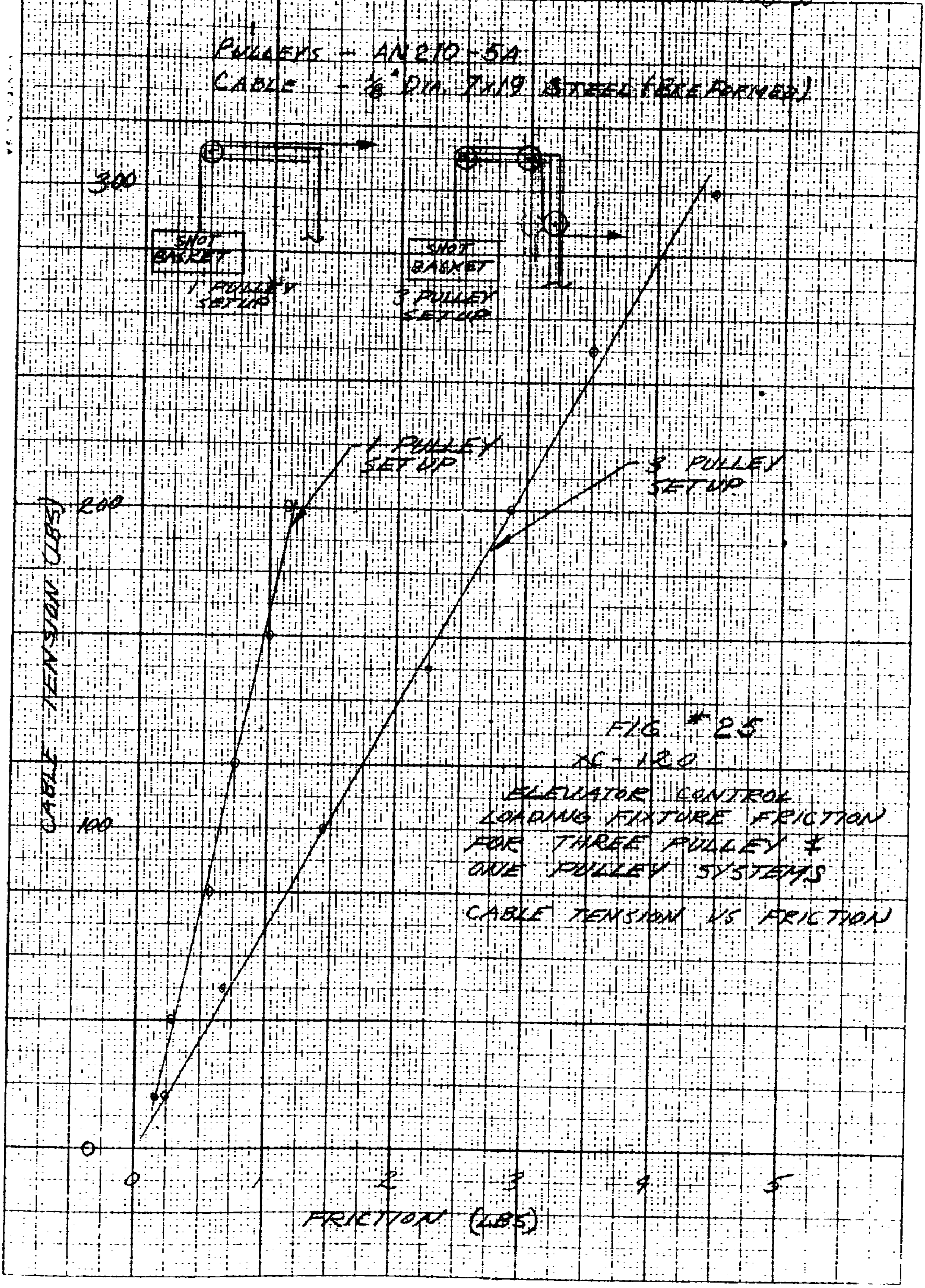


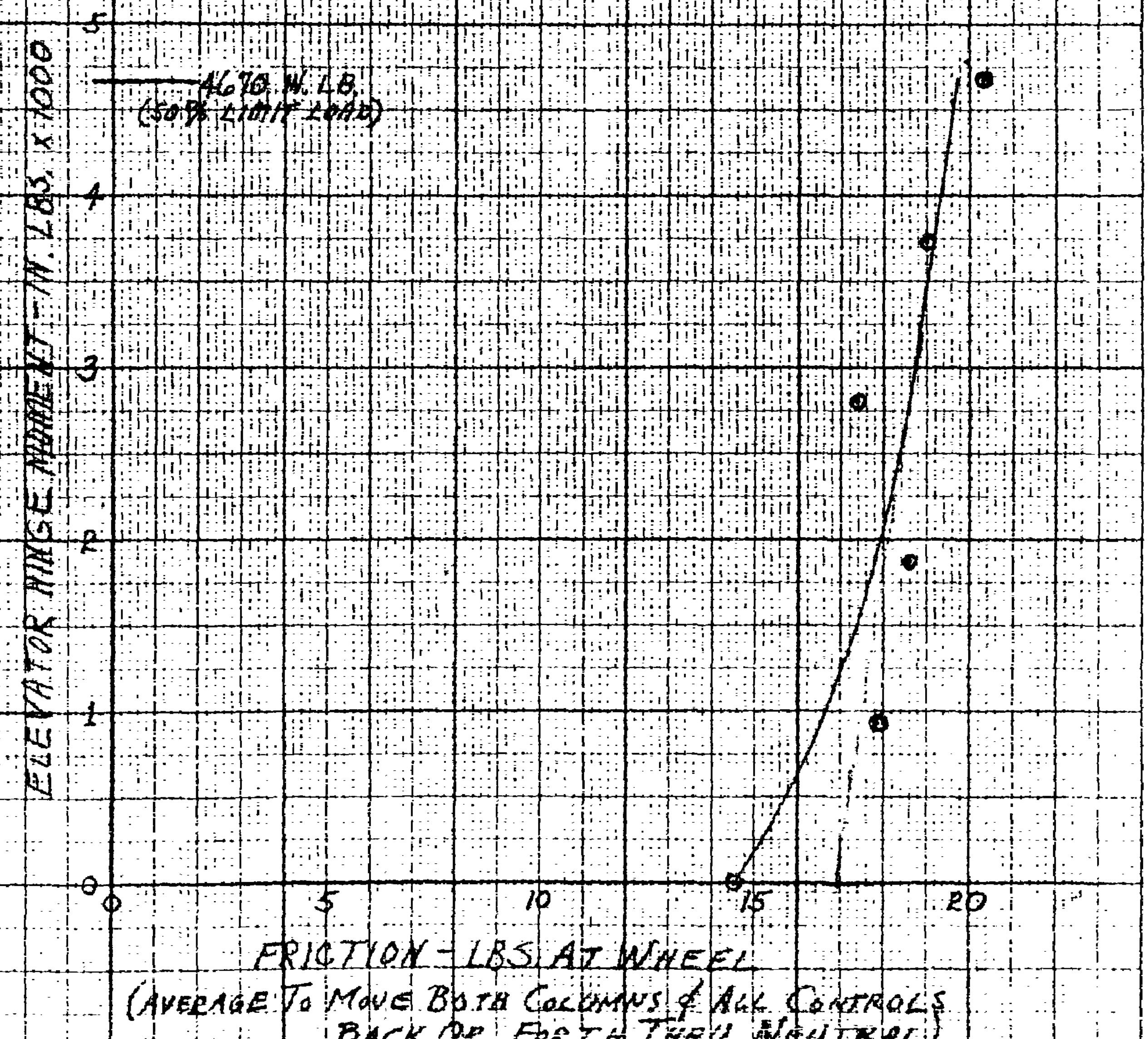
FIG. #25
XC-120
ELEVATOR CONTROL
LOADING FIXTURE FRICTION
FOR THREE PULLEY &
ONE PULLEY SYSTEMS

CABLE TENSION VS FRICTION

FIG. 26

TEST E-2, XC-120 ELEVATOR
HINGE MOMENT VS FRICTION

ENTIRE CABLE SYSTEM CONNECTED.
FRICTION AT 50% DOUBLE PILOT EFFORT AS SURFACES OPERATED THROUGH NEUTRAL.
50% LIMIT HINGE MOMENT = 4670 IN.LBS. APPLIED TO SURFACE, BALANCED BY 150 LB LOAD ON EACH, PILOT AND COPILOT CONTROL COLUMN.



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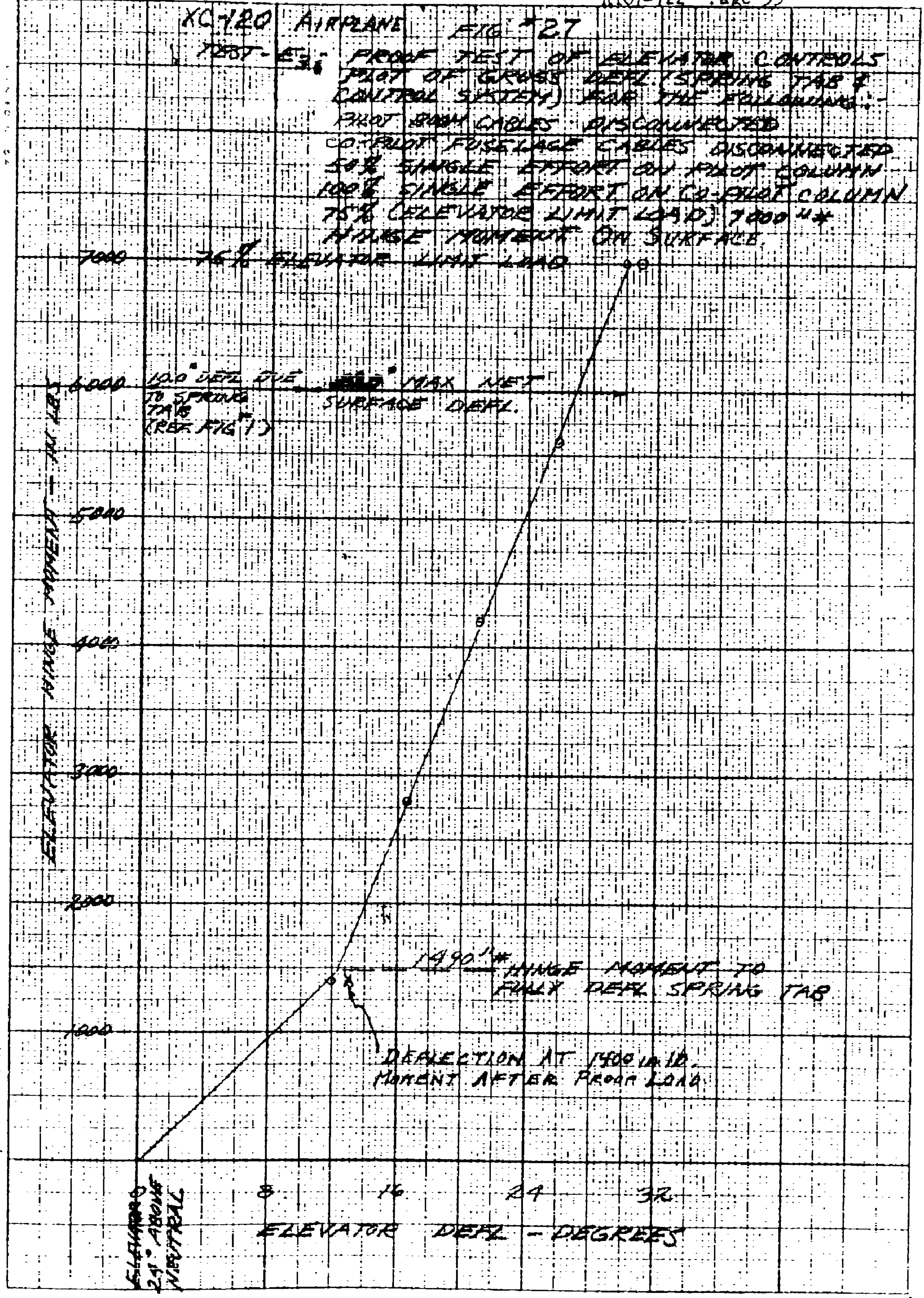
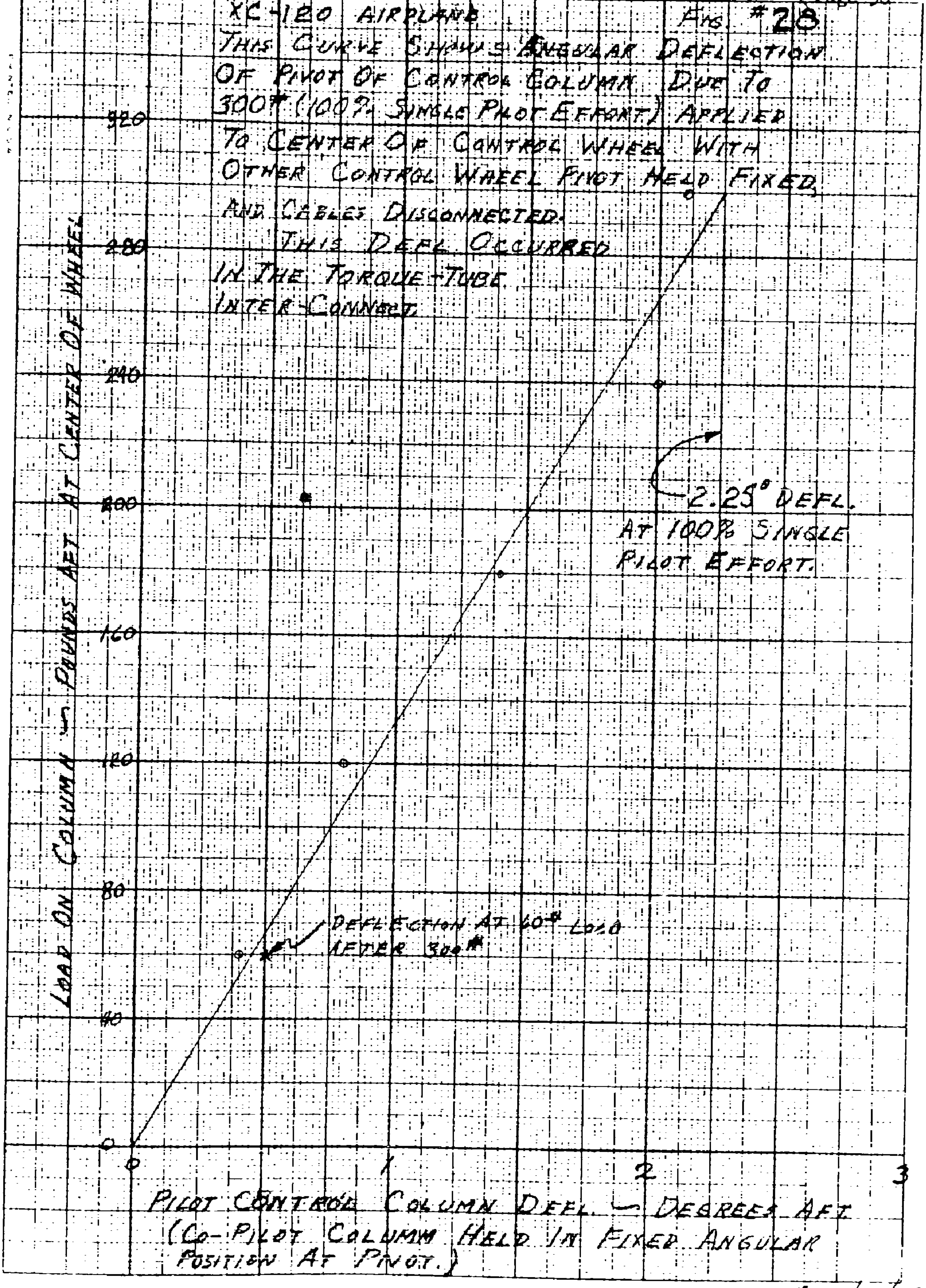

XC-120 AIRPLANE FIG #27
TEST - E3: PROOF TEST OF ELEVATOR CONTROLS
PLOT OF GROSS DEFL (SPRING TAB & CONTROL SYSTEM) FOR THE FOLLOWING:-
PILOT BOOM CABLES DISCONNECTED
CO-PILOT FUSELAGE CABLES DISCONNECTED
50% SINGLE EFFORT ON PILOT COLUMN
100% SINGLE EFFORT ON CO-PILOT COLUMN
75% (ELEVATOR LIMIT LOAD) 7000 "# HINGE MOMENT ON SURFACE
76% ELEVATOR LIMIT LOAD
7000
6000
10.0° DEFL DUE TO SPRING TAB (REF FIG #1)
MAX NET SURFACE DEFL.
5000
4000
3000
2000
1000
1490 "# HINGE MOMENT TO FULLY DEFL SPRING TAB
DEFLECTION AT 1400 in lb. MOMENT AFTER PROOF LOAD
ELEVATOR HINGE MOMENT - IN LB
ELEVATOR 2.1° ABOVE NEUTRAL
8
16
24
32
ELEVATOR DEFL - DEGREES

12-28-97



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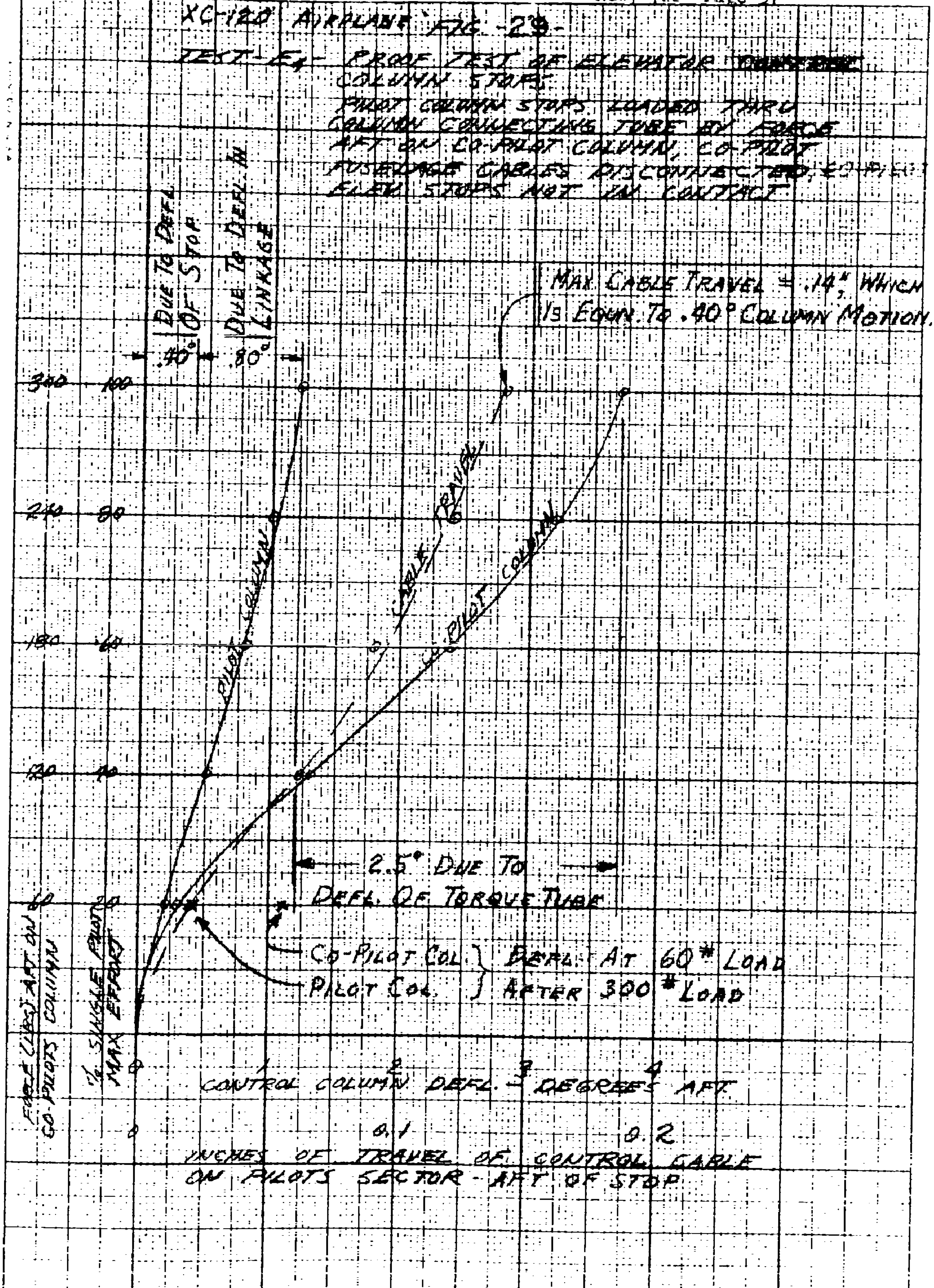
XC-120 AIRPLANE FIG -29-
TEST E4 - PROOF TEST OF ELEVATOR COLUMN STOPS
PILOT COLUMN STOPS LOADED THRU COLUMN CONNECTING TUBE BY FORCE AFT ON CO-PILOT COLUMN, CO-PILOT FUSELAGE CABLES DISCONNECTED, CO-PILOT ELEV. STOPS NOT IN CONTACT
DUE TO DEFL OF STOP
DUE TO DEFL IN LINKAGE
.40°
.80°
MAX CABLE TRAVEL = .14" WHICH IS EQUIV. TO .40° COLUMN MOTION.
300
100
240
80
180
60
120
40
60
20
PILOT COLUMN
CABLE TRAVEL
CO-PILOT COLUMN
2.5° DUE TO DEFL. OF TORQUE TUBE
CO-PILOT COL
PILOT COL.
DEFL AT 60# LOAD AFTER 300# LOAD
FORCE (LBS) AFT ON CO-PILOTS COLUMN
% SINGLE PILOT MAX EFFORT
0
1
2
3
4
CONTROL COLUMN DEFL. - DEGREES AFT
0
0.1
0.2
INCHES OF TRAVEL OF CONTROL CABLE ON PILOTS SECTOR - AFT OF STOP

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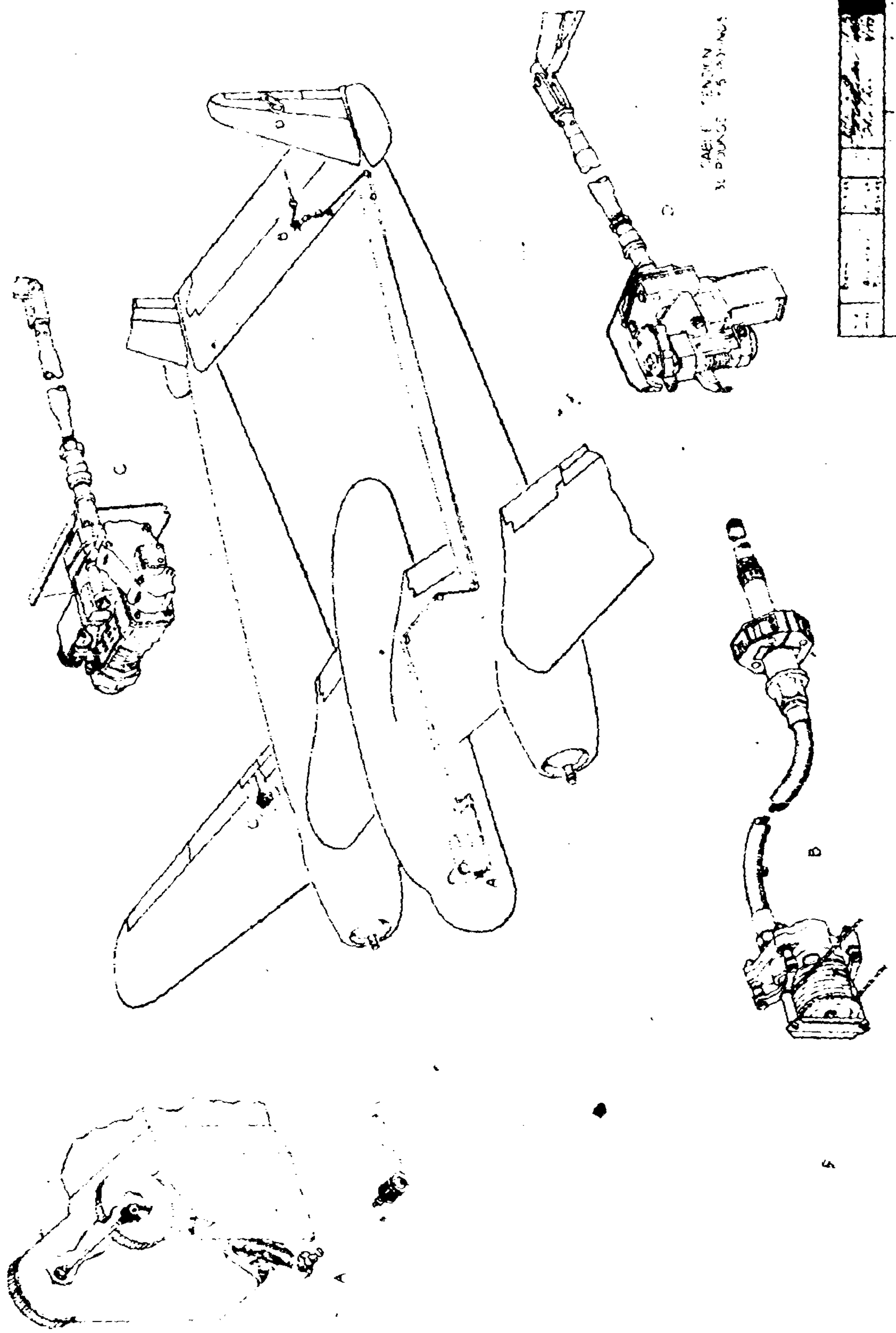
12-28-49

FIG. 3∅



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DATE 9-2-49

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BASIC DATA

Subject:— AILERON TAB CONTROL SYSTEM

FIG 31

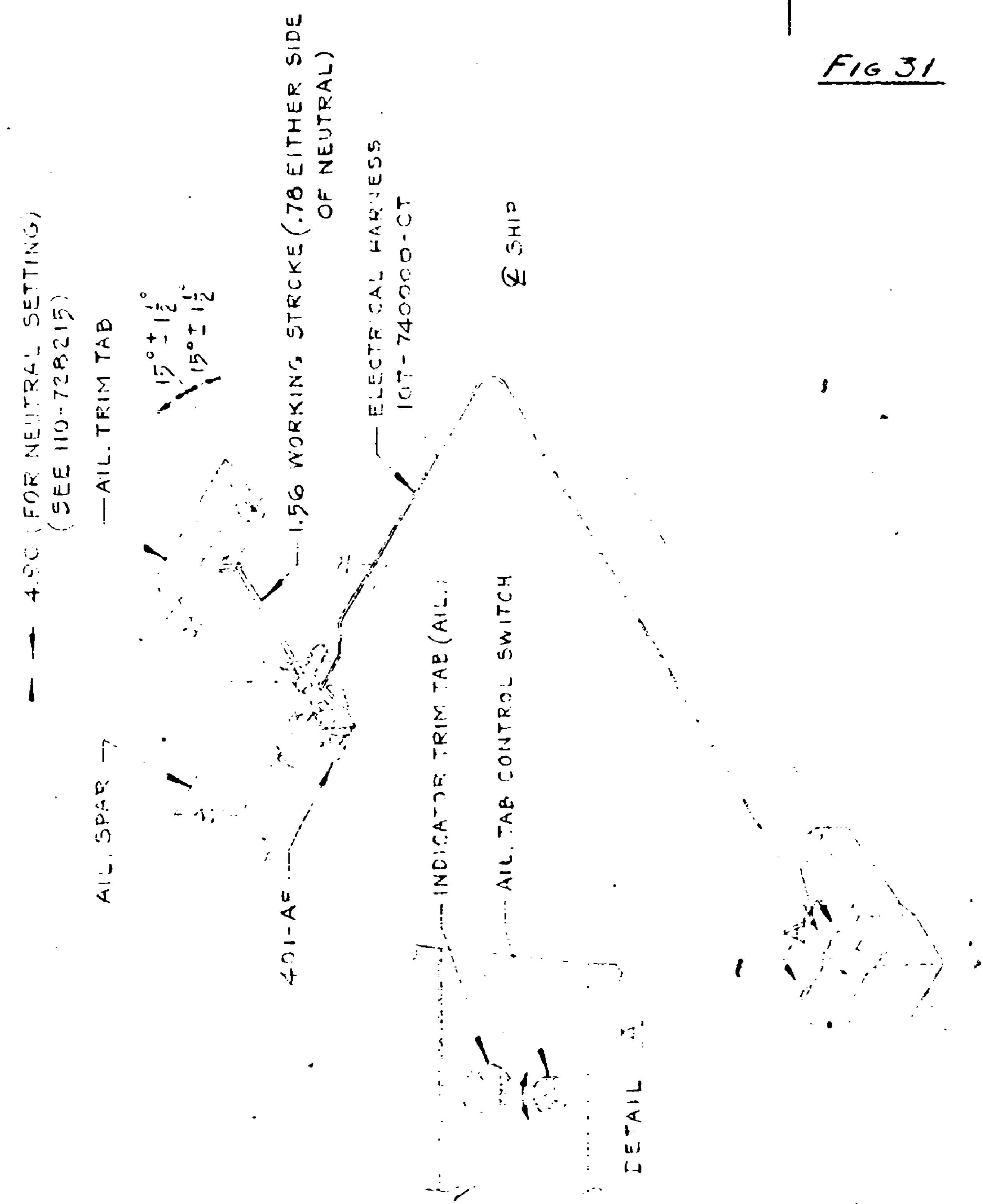
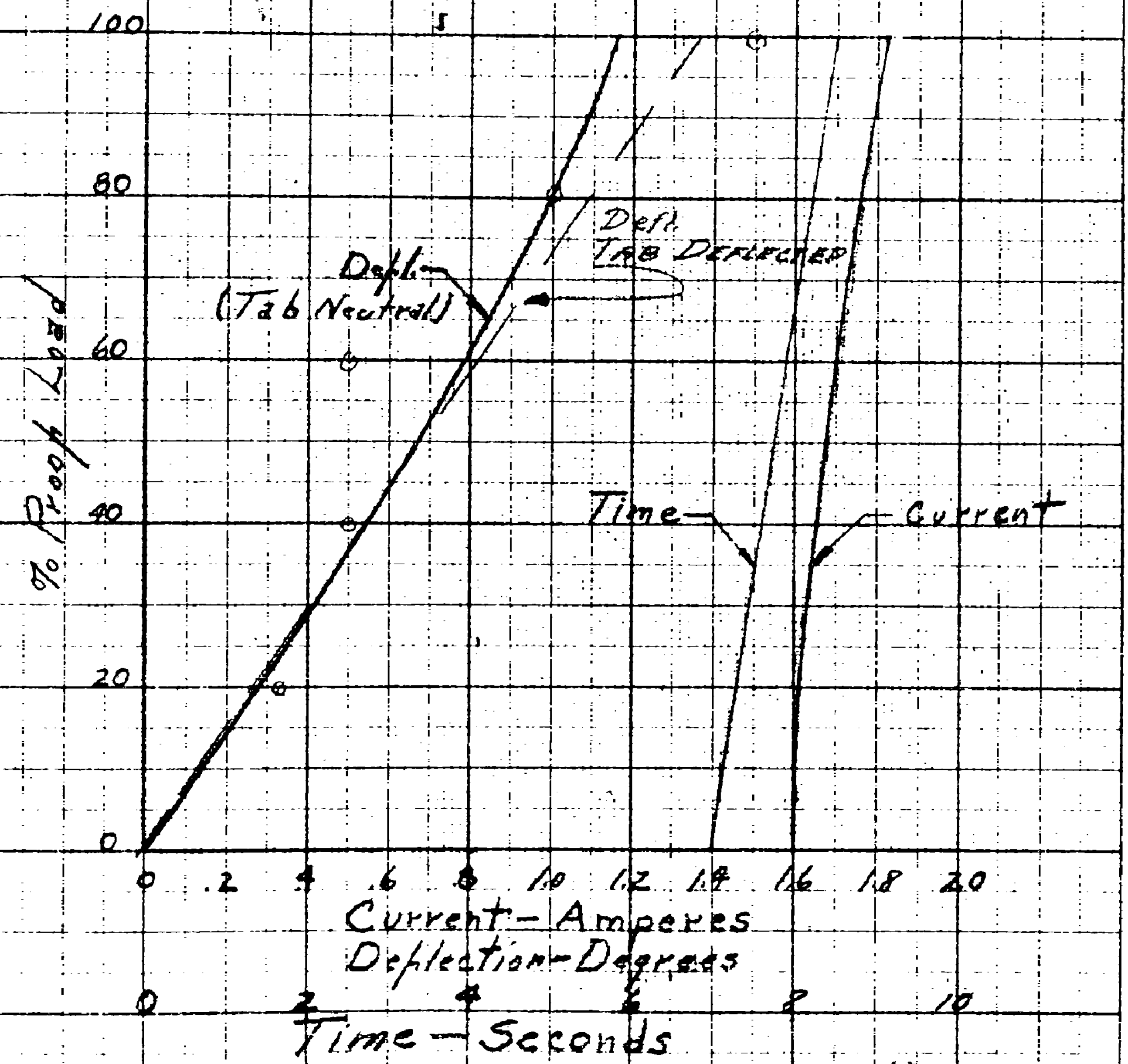


Fig-32

Aileron Trim Tab XC-120
Proof & Operation Tests

These curves show deflections due to load, Actuator current & operation time from Neutral to 14.5° Up. Working against loads up to limit hinge moment.

100% Proof Load = 480 in. lbs.
Operating Voltage was 28.0 to 28.8 volts.



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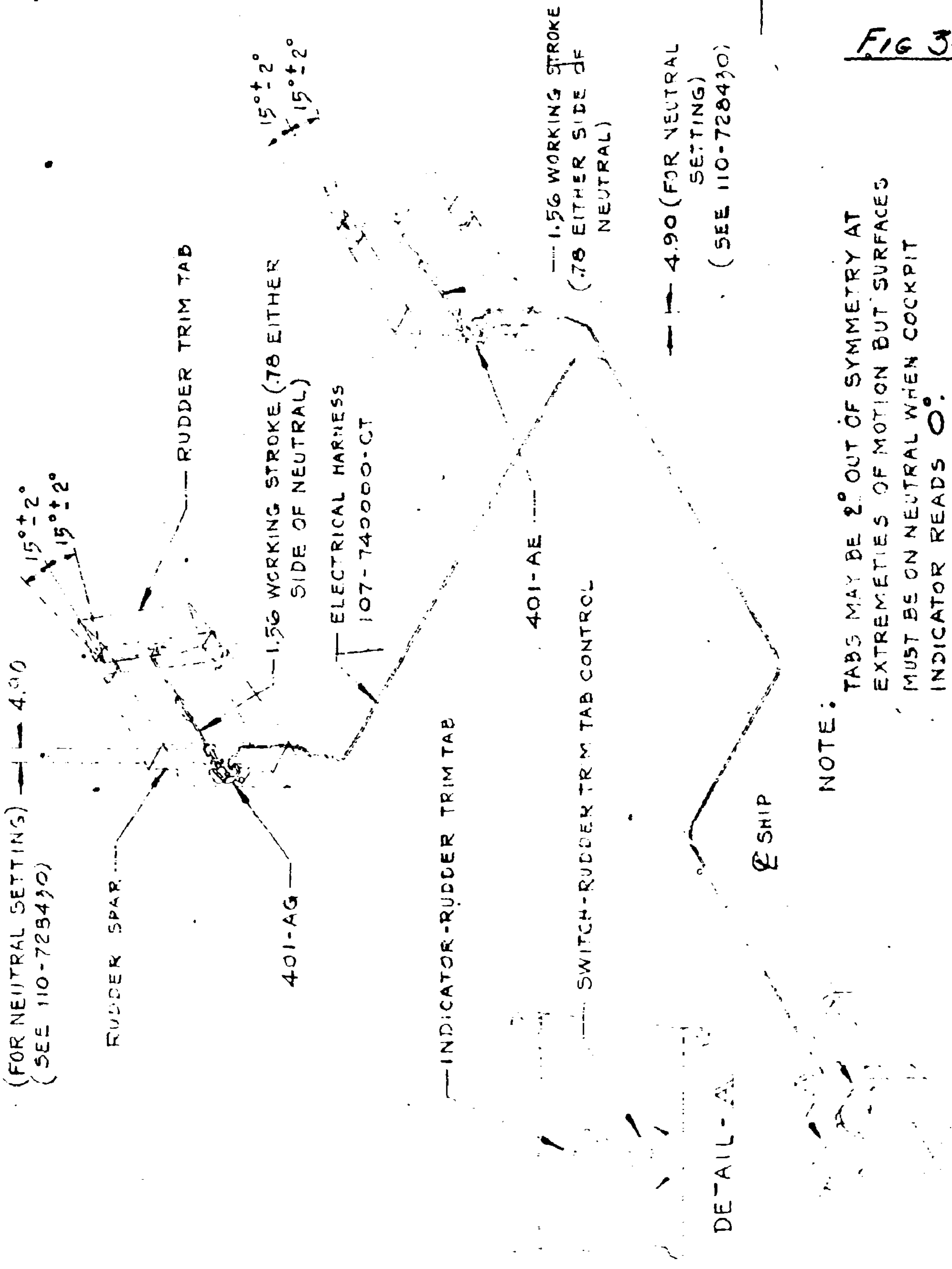
Subject:— BASIC DATA RUDDER TRIM TAB CONTROL SYSTEM			DATE 9-2-49 REVISED

FIG 38



NOTE: TABS MAY BE 2° OUT OF SYMMETRY AT EXTREMETIES OF MOTION BUT SURFACES MUST BE ON NEUTRAL WHEN COCKPIT INDICATOR READS 0°.

FA-800-23

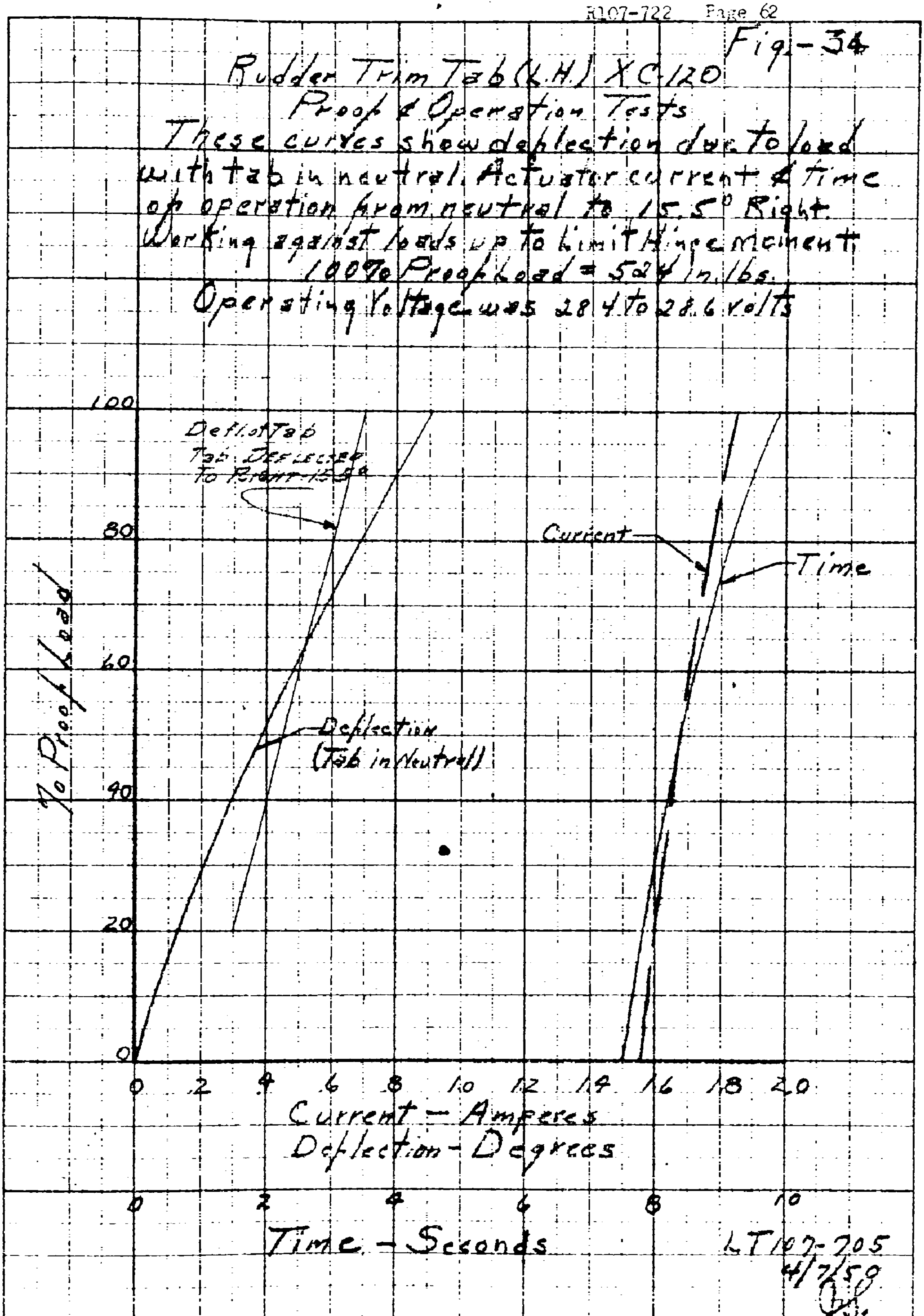


Fig-34
Rudder Trim Tab (L.H.) XC-120
Proof & Operation Tests
These curves show deflection due to load
with tab in neutral. Actuator current & time
of operation from neutral to 15.5° Right.
Working against loads up to limit Hinge moment.
100% Proof Load = 524 in. lbs.
Operating Voltage was 28.4 to 28.6 volts
% Proof Load
100
80
60
40
20
0
Current
Time
Deflection
(Tab in Neutral)
0 .2 .4 .6 .8 1.0 1.2 1.4 1.6 1.8 2.0
Current - Amperes
Deflection - Degrees
0 2 4 6 8 10
Time - Seconds
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FAIRCHILD AIRCRAFT DIVISION
OF FAIRCHILD ENGINE & AIRPLANE CORPORATION

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MODEL XC-120 PREPARED BY BONK CHECKED BY BONK APPROVED BY

DATE 9-2-49
REVISED 11-18-49

Subject: BASIC DATA - ELEV. TRIM TAB CONT. SYSTEM

NOTE - RIG IN NEUTRAL POSITION AS SHOWN - RIG TO 30# ± 5# TENSION.

A = 2.374 ℄ CABLE DIA.

B = 2.60 ACTUATOR TRAVEL (WITH OVERTRAVEL)

2.34 ACTUATOR TRAVEL (WITHOUT OVERTRAVEL)

OPERATION -

WHEEL FWD - TAB UP - ELEV. DOWN - NOSE DOWN

WHEEL AFT - TAB DOWN - ELEV. UP - NOSE UP

MOTION -

TAB 12° UP ± 2°37' OVERTRAVEL

TAB 22° DOWN ± 2°37' OVERTRAVEL

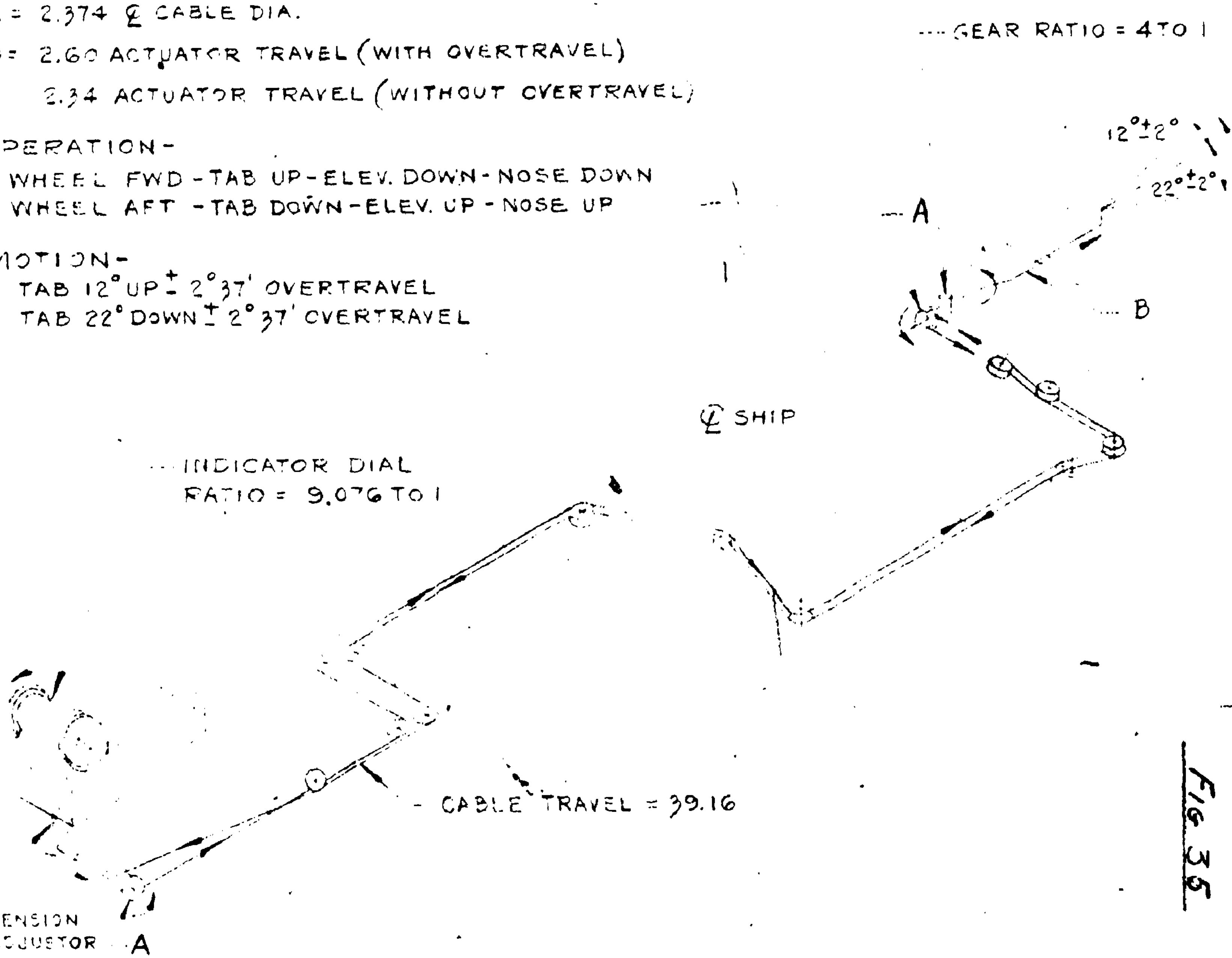

FIG 35

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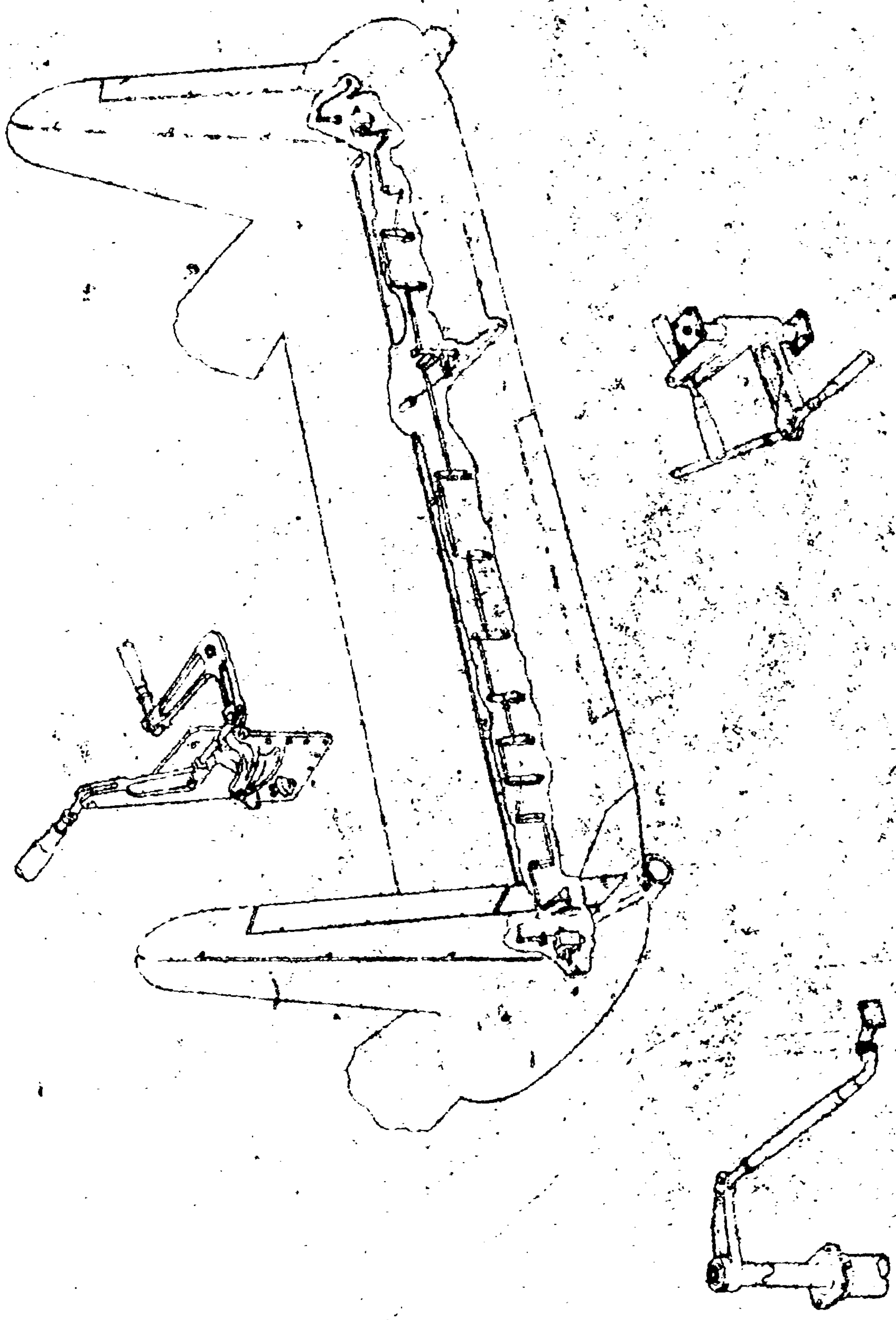
FIG 36

ELEVATOR SPRING TAB



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BASIC DATA

DATE 9-2-49

Subject — ELEVATOR SPRING TAB CONTROL SYSTEM

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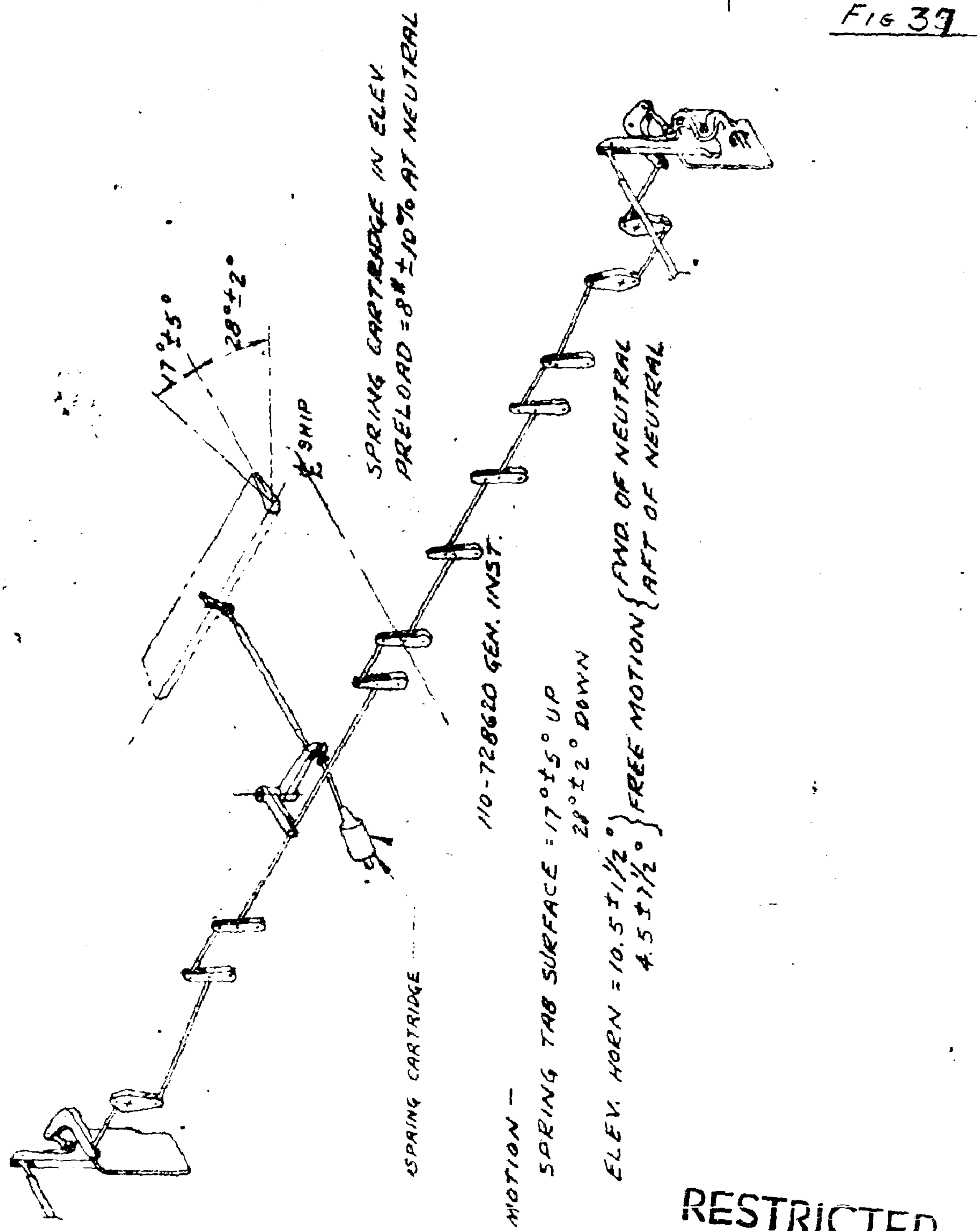
FIG 39



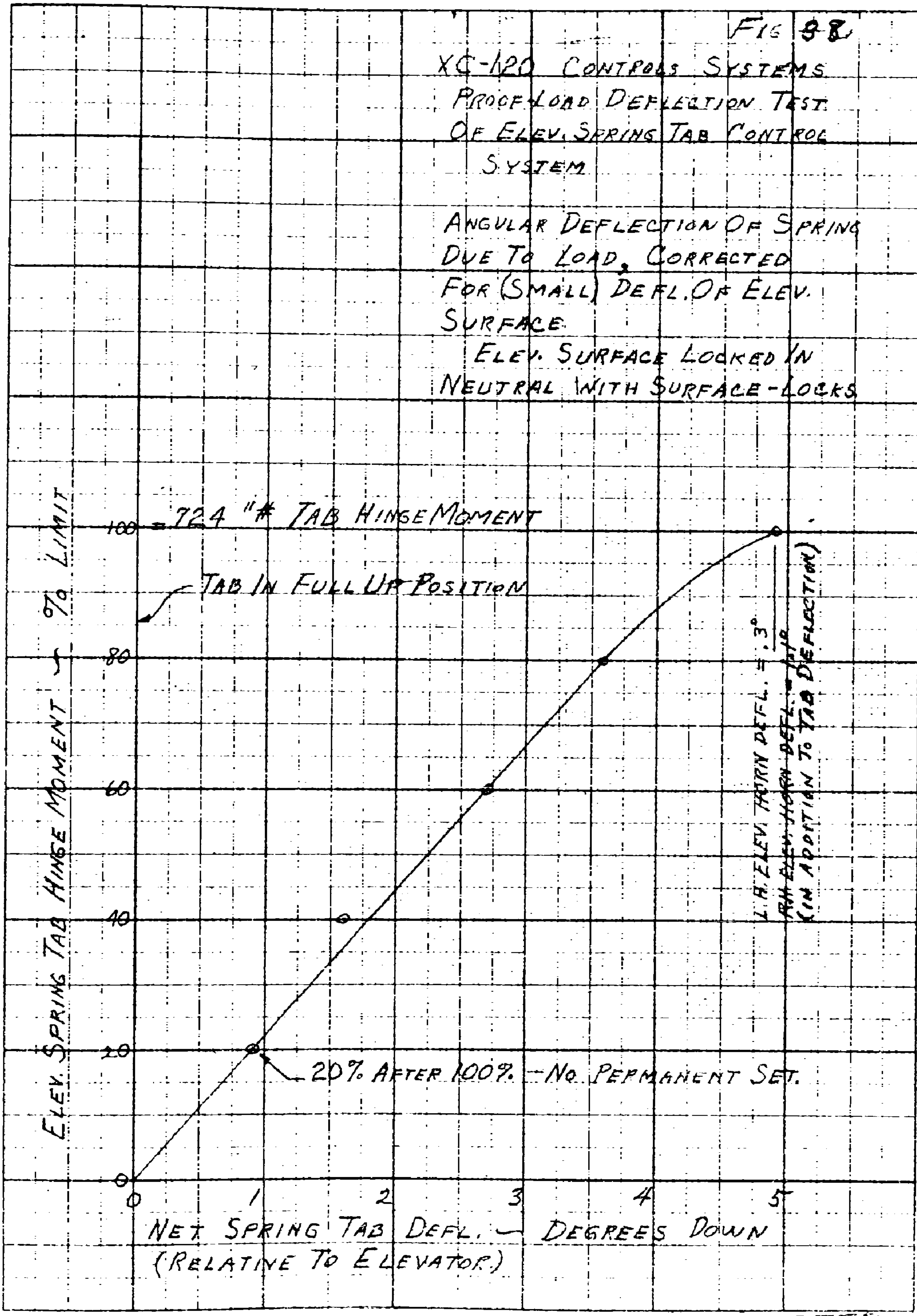


FIG 38
XC-120 CONTROLS SYSTEMS
PROOF-LOAD DEFLECTION TEST OF ELEV. SPRING TAB CONTROL SYSTEM
ANGULAR DEFLECTION OF SPRING DUE TO LOAD, CORRECTED FOR (SMALL) DEFL. OF ELEV. SURFACE
ELEV. SURFACE LOCKED IN NEUTRAL WITH SURFACE-LOCKS
100 = 724 "# TAB HINGE MOMENT
TAB IN FULL UP POSITION
20% AFTER 100% - NO PERMANENT SET.
L.H. ELEV. HORN DEFL. = .3°
(IN ADDITION TO TAB DEFLECTION)
0
20
40
60
80
100
0
1
2
3
4
5
ELEV. SPRING TAB HINGE MOMENT ~ % LIMIT
NET SPRING TAB DEFL. ~ DEGREES DOWN
(RELATIVE TO ELEVATOR)

RESTRICTED

FIG. 39

XC-120 AIRPLANE. TEST T_2

PROOF TEST OF ELEV. TRIM TAB CONTROL FWD. STOP — LOAD APPLIED TANGENTIALLY AT RIM OF CO-PILOT'S WHEEL THRU INTERCONNECTING TUBE TO STOP AT PILOT'S WHEEL.

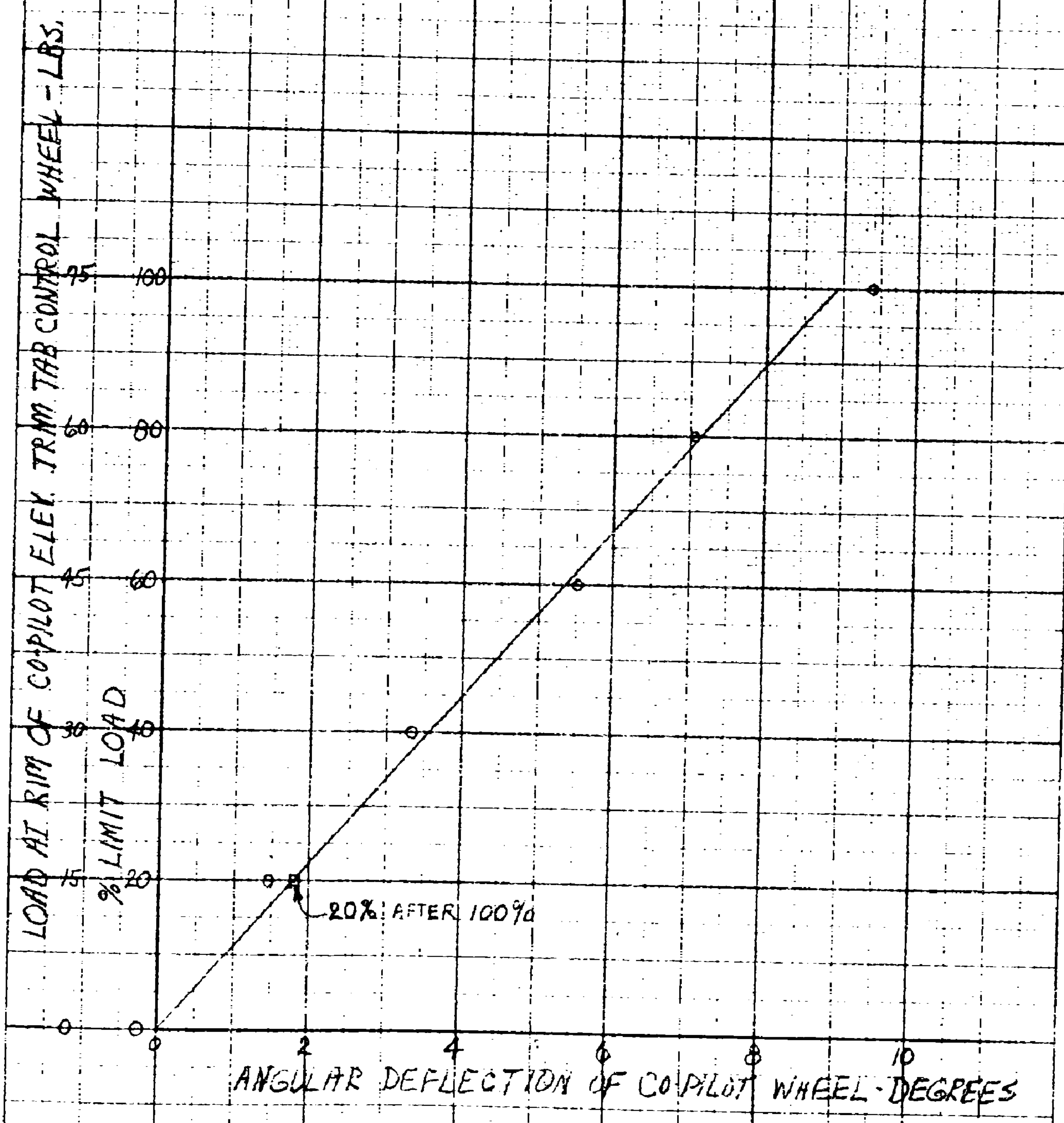


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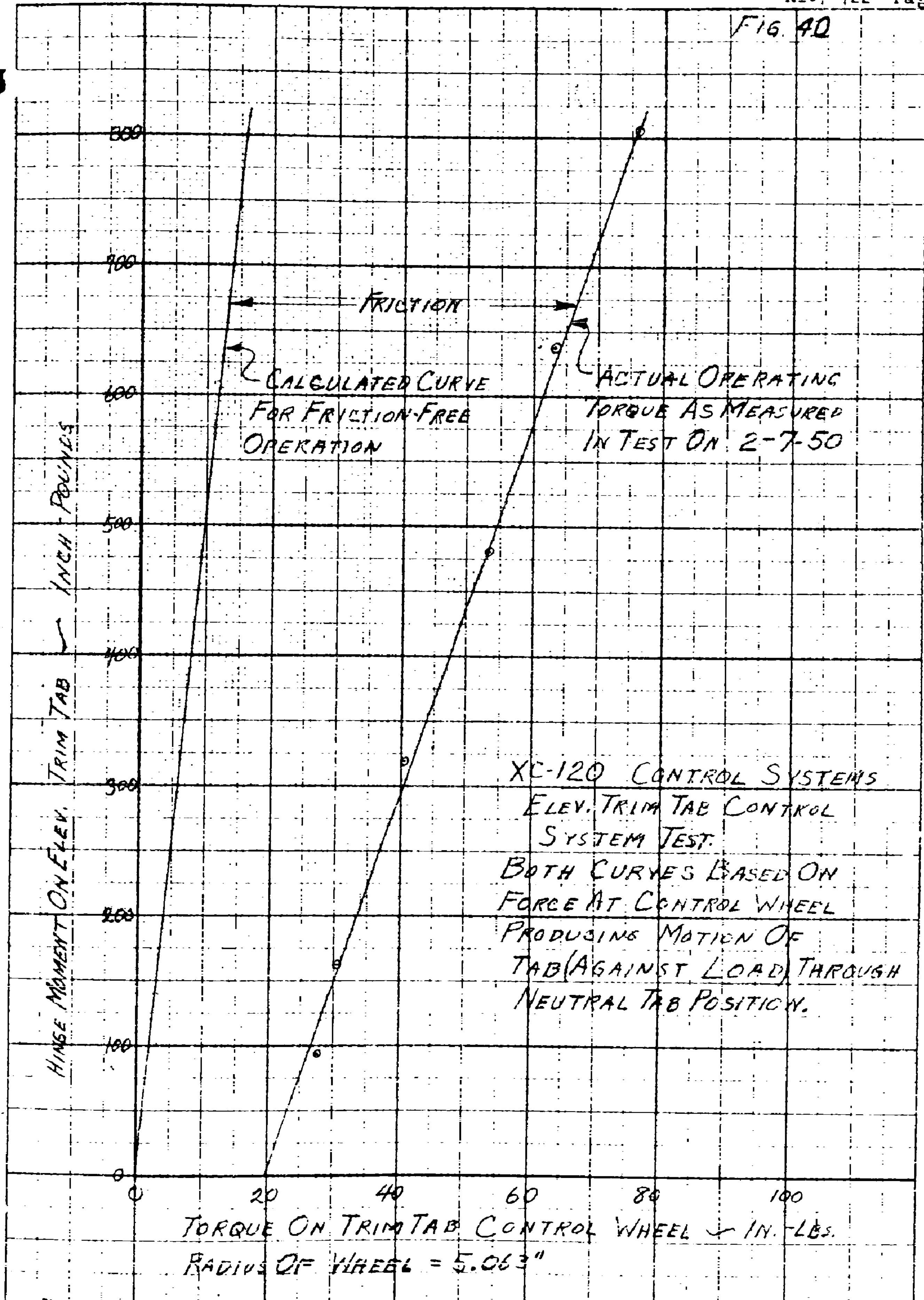
FIG. 40



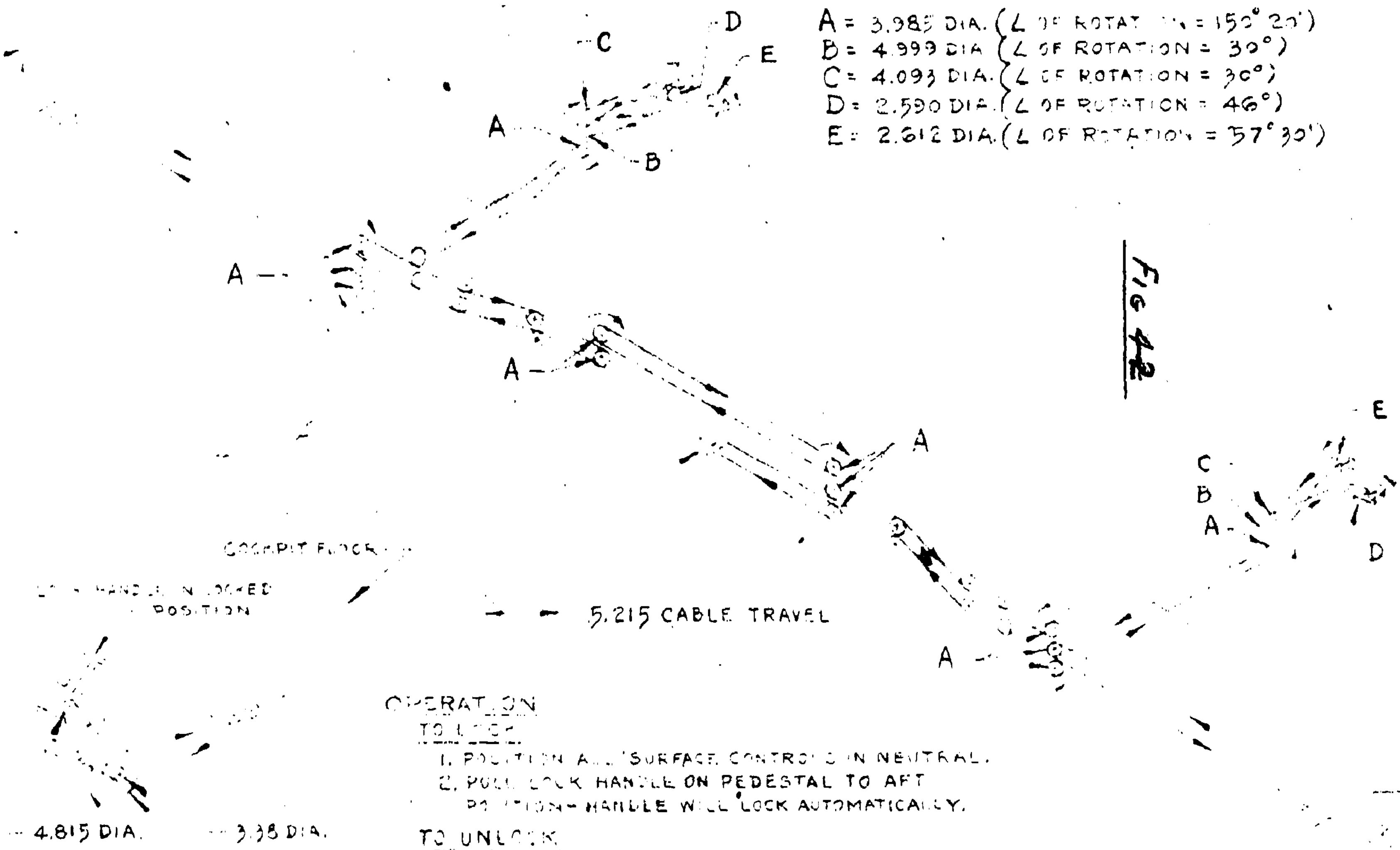
800
700
600
500
400
300
200
100
0
0
20
40
60
80
100
FRICTION
CALCULATED CURVE FOR FRICTION-FREE OPERATION
ACTUAL OPERATING TORQUE AS MEASURED IN TEST ON 2-7-50
XC-120 CONTROL SYSTEMS
ELEV. TRIM TAB CONTROL SYSTEM TEST.
BOTH CURVES BASED ON FORCE AT CONTROL WHEEL PRODUSING MOTION OF TAB (AGAINST LOAD) THROUGH NEUTRAL TAB POSITION.
HINGE MOMENT ON ELEV. TRIM TAB ~ INCH POUNDS
TORQUE ON TRIM TAB CONTROL WHEEL ~ IN.-LBS.
RADIUS OF WHEEL = 5.063"

FIG 4I

Subject: SURFACE LOCK CONTROL SYSTEM - BASIC DATA

A = 3.985 DIA. (∠ OF ROTATION = 150° 20')
B = 4.999 DIA. (∠ OF ROTATION = 30°)
C = 4.093 DIA. (∠ OF ROTATION = 30°)
D = 2.590 DIA. (∠ OF ROTATION = 46°)
E = 2.612 DIA. (∠ OF ROTATION = 57° 30')

FIG 42



OPERATION

TO LOCK

1. POSITION ALL SURFACE CONTROLS IN NEUTRAL.
2. PULL LOCK HANDLE ON PEDESTAL TO AFT POSITION - HANDLE WILL LOCK AUTOMATICALLY.

TO UNLOCK

1. PULL HAND GRIP ON LOCK HANDLE UP TO RELEASE.
2. PUSH LOCK HANDLE ON PEDESTAL FORWARD - HANDLE WILL LOCK AUTOMATICALLY AFTER RELEASING CONTROL LOCKS.

NOTE - ALL RIGGING TO BE DONE IN LOCKED POSITION AS SHOWN.

* RIGGING TENSIONS GIVEN ON PAGE B-6

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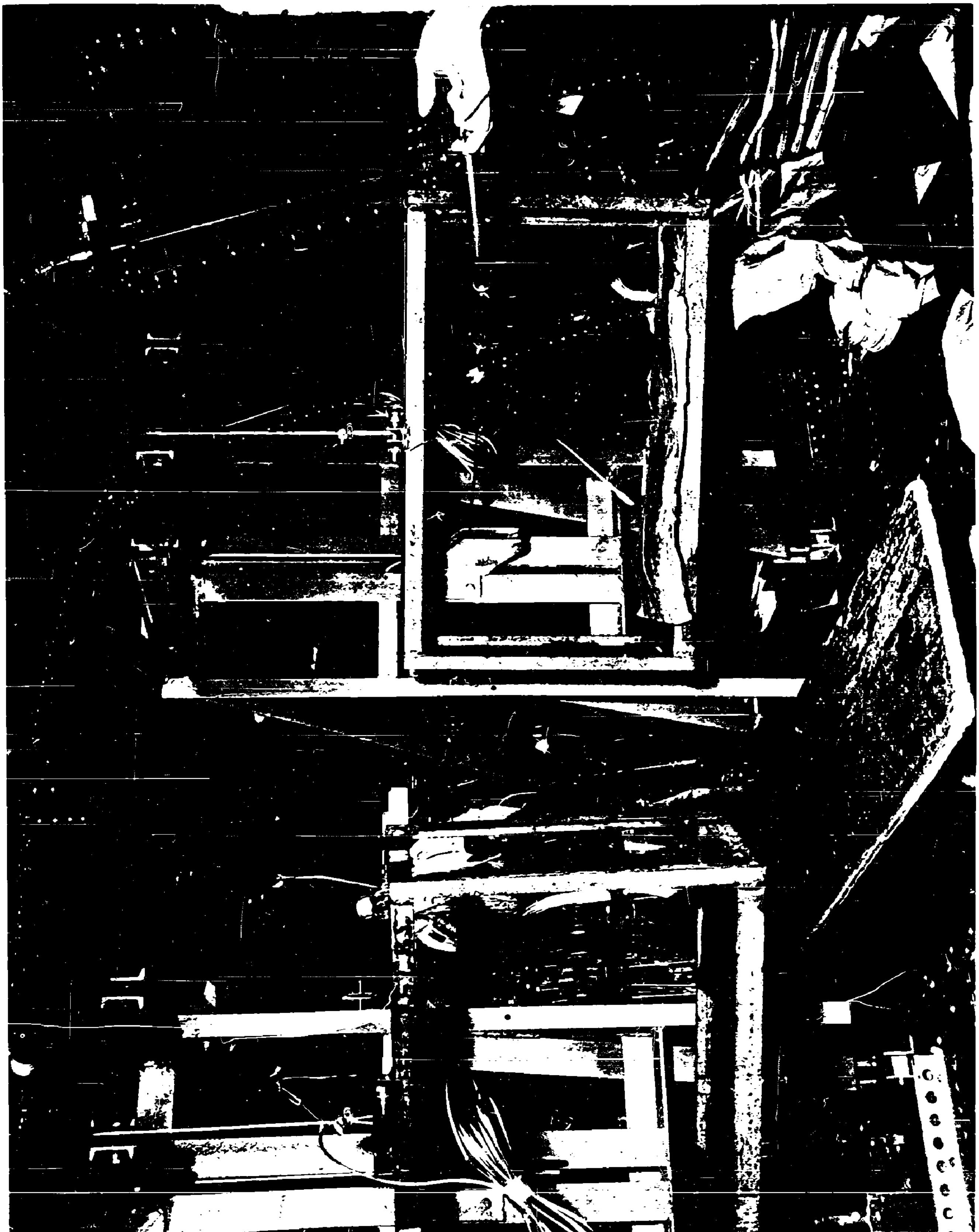
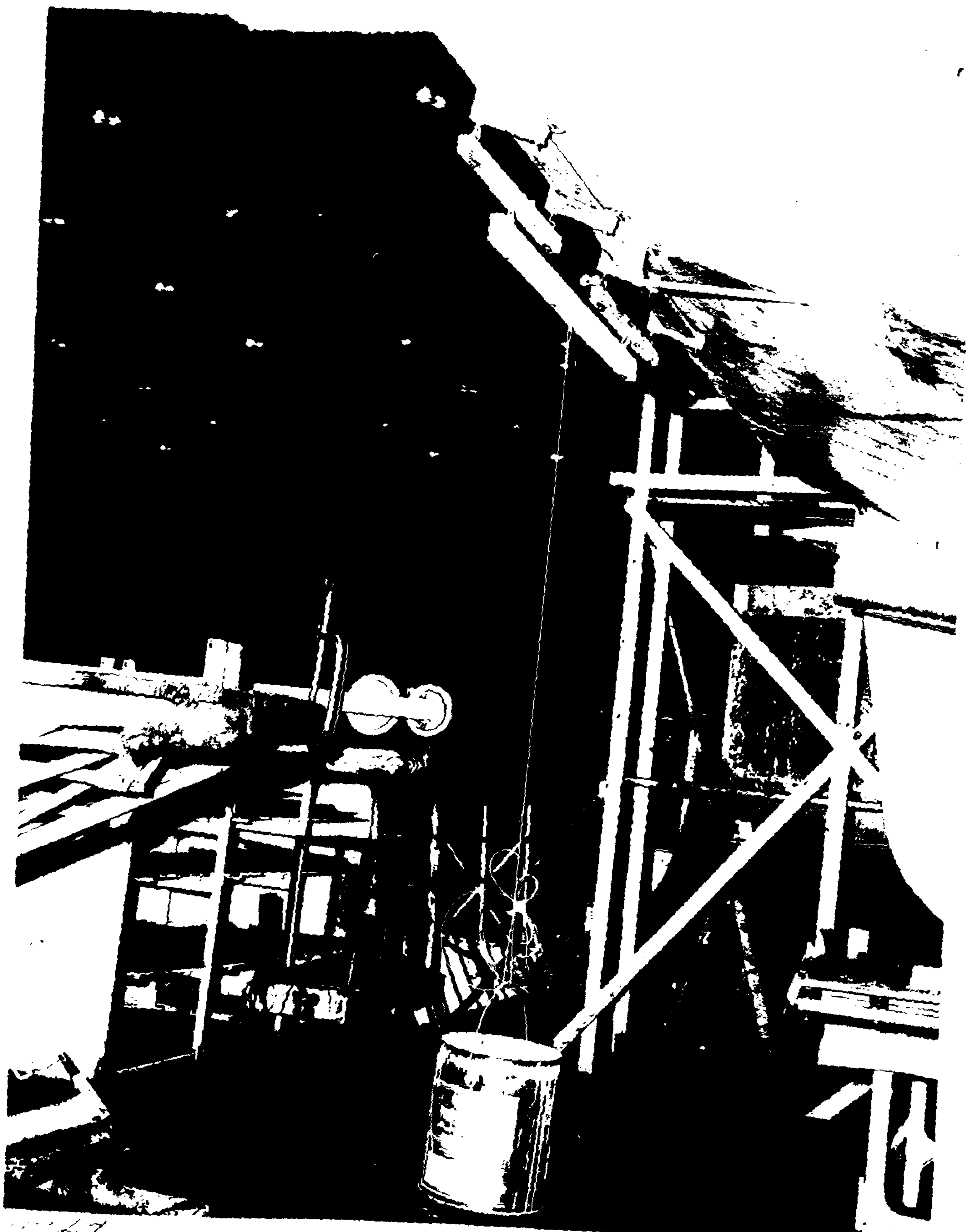
18045 - View of Cockpit Loading Fixture Showing Method of Determining Elevator Control System Friction under Load.

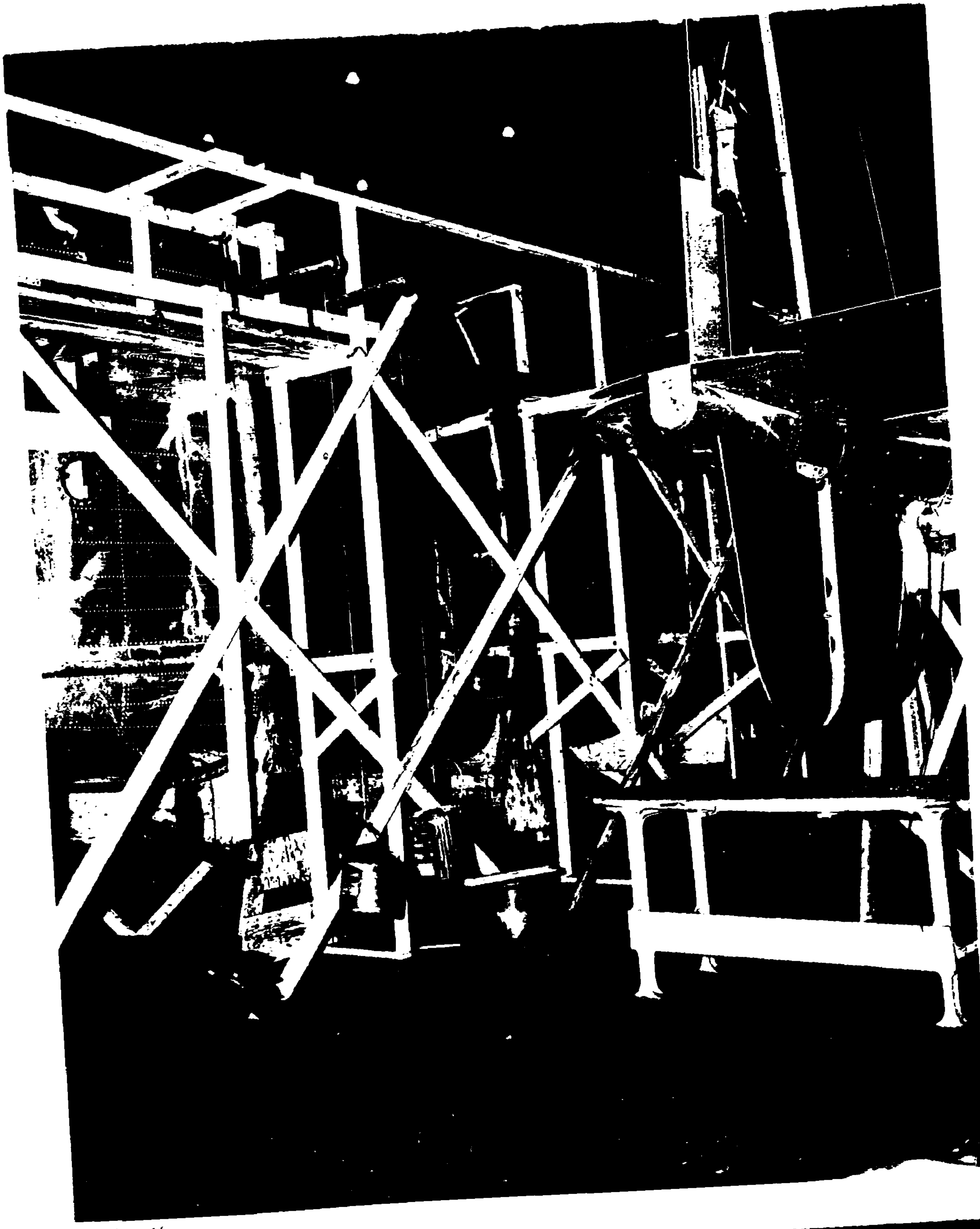


18045

18047 - View showing Method Used to apply a Hinge Moment on the Elevator or Elevator Trim Tab Through a Whiffle Tree Attached to the Trim Tab.



18051 - Setup View of Method Used to Apply a Side Load on Both Rudders Simultaneously

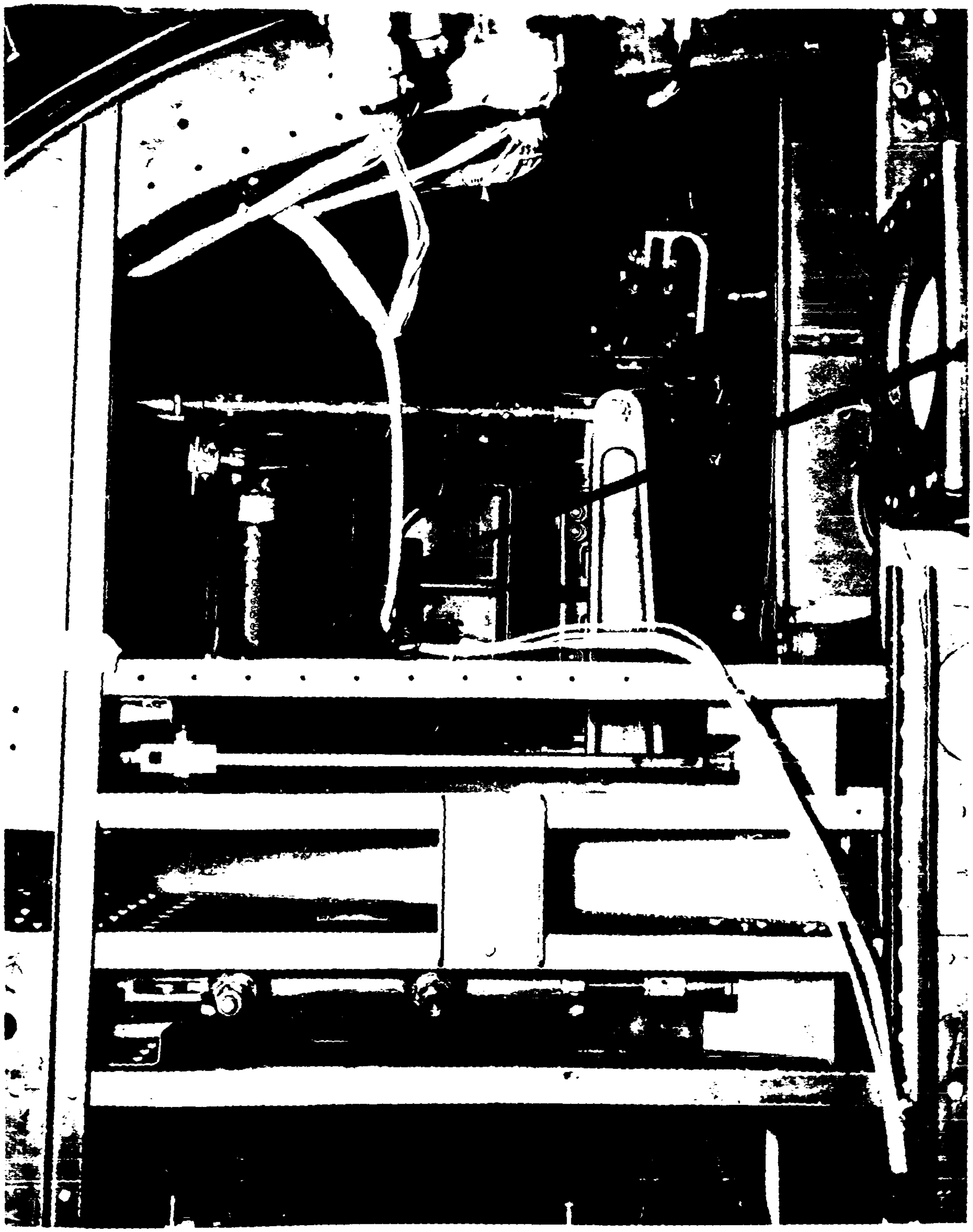


18051

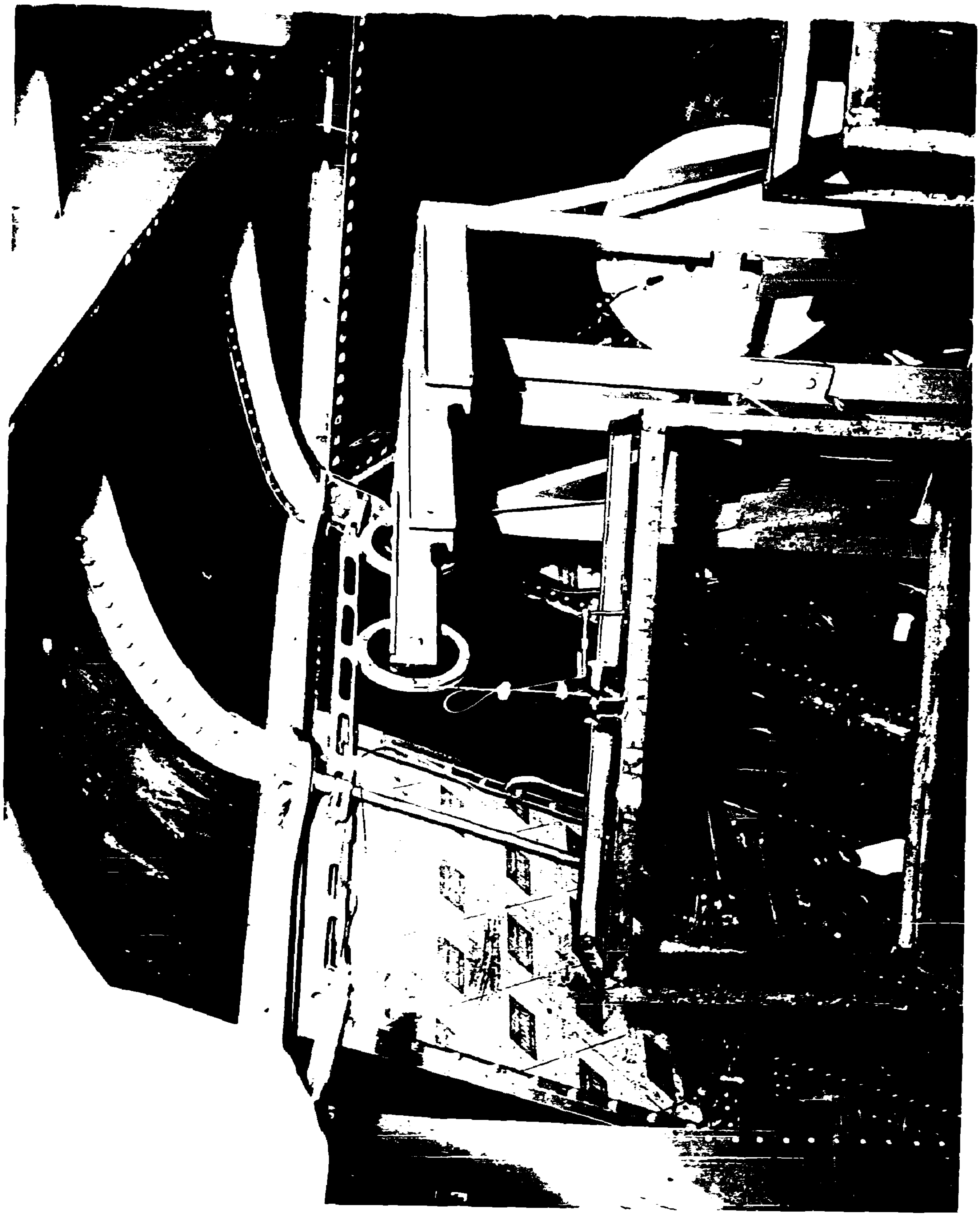
18052 - View of Device Used to Apply a Forward Load on the Rudder Pedal. Nose Section of Airplane was removed to permit this type of loading.

18052

18055 - View of Loading Device Attached to the Rudder Pedal for Application of Simulated Brakin Forces at the Tip of the Brake Pedal, Note Dial Gages Used to Determine Structural Delfections.

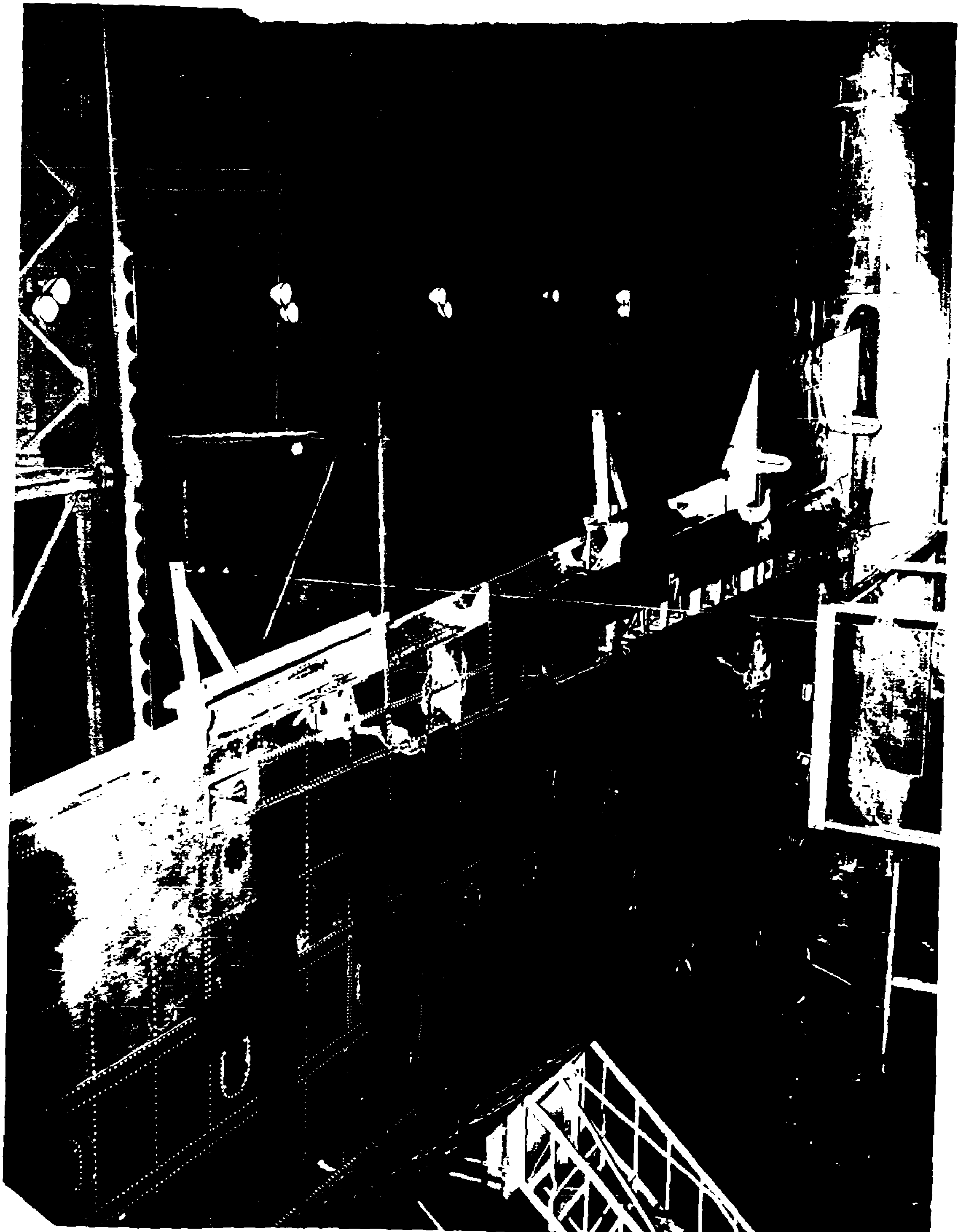


18158 - View of Cockpit Loading Fixture Used to Apply Simulated Aileron Control Wheel Forces. Note Dummy Wheel and Quadrant Which Replaced Pilot's Wheel.



18158

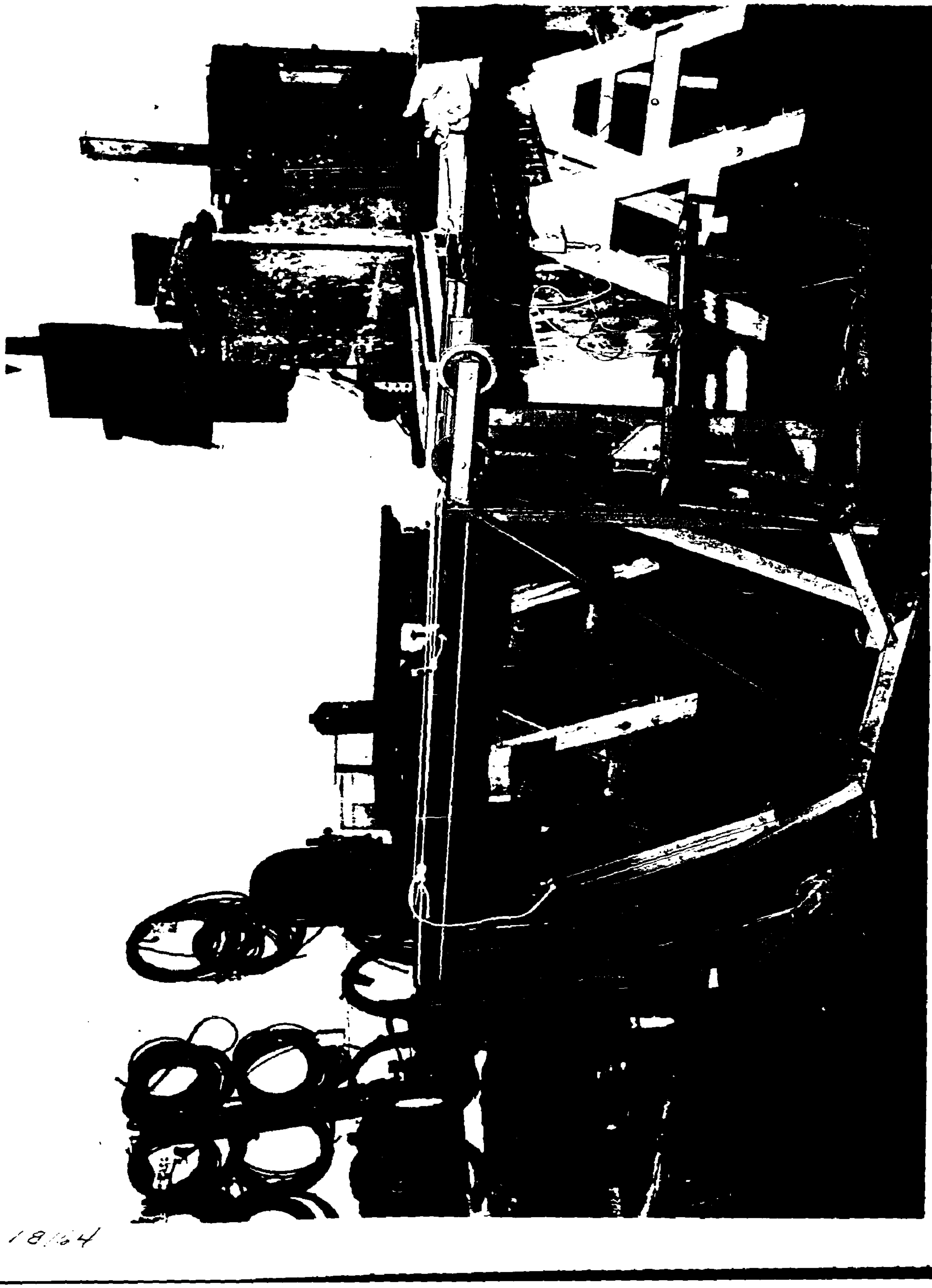
18160 - View showing Installation of Dummy Ailerons on Left Hand Wing and method of applying a Down Load on them.



18160

18162 - View showing Dummy Ailerons Installed on Right Hand Wing and Method of Applying an Up Load on Them.

18164 - View of Setup Used to Determine the Friction of the Pulleys in the Loading System Used During Some of the Control Systems Tests.



18164

18563 - View of Engine Control Pedestal Showing Control System Locking Handle and Method Used to Apply Load to it.

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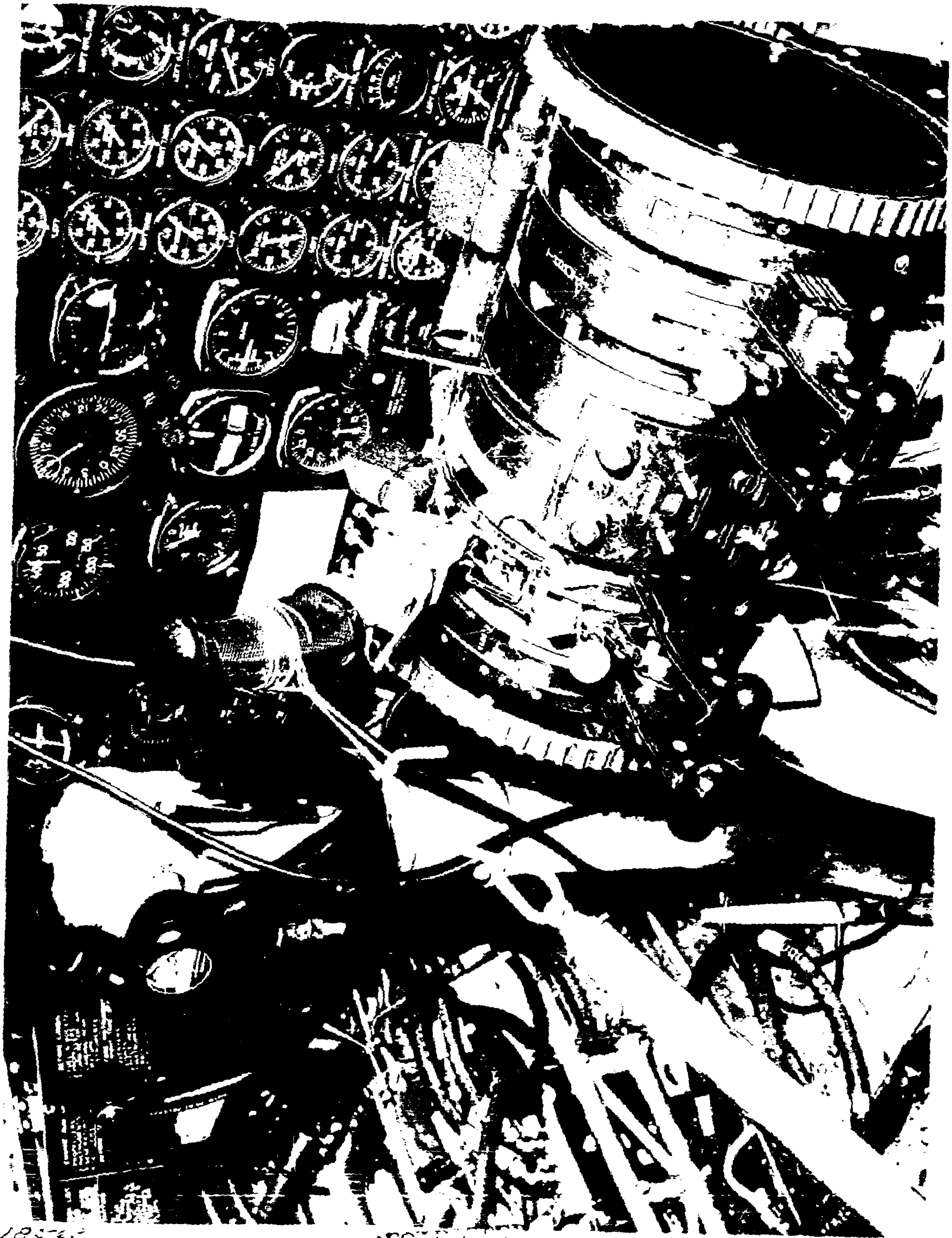

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FAIRCHILD ENGINE AND AIRPLANE CORP., FAIRCHILD AIRCRAFT DIV., HAGERSTOWN, MD. (ENGINEERING REPORT NO. R107-722)

PROOF AND OPERATION TESTS OF FLIGHT AND TRIM CONTROLS - MODEL XC-120

M.M. CUTLER; C.S. HUBER 24 JULY'50 80PP. PHOTOS, DIAGRS, GRAPHS, DRWGS

STRUCTURES (7) DESIGN AND DETAILS (3)	CONTROL SYSTEMS - STRUCTURAL TESTS C-120 - STRUCTURAL TESTS C-120 Aircraft

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