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ARROW 2
FLYING CONTROLS SYSTEM

REPORT NO. 72/SYSTEMS 15/28

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This brochure is intended to provide an accurate description of the system(s) or service(s) for purposes of the Arrow 2 Mock-up Conference, and is not to be considered binding with respect to changes which may occur subsequent to the date of publication.

COMPILED BY [signature] APPROVED BY Alan R. Buley

ENGINEERING DIVISION

AVRO AIRCRAFT LIMITED

MALTON — ONTARIO

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1. Introduction

1.1 A fully powered flying control system has been selected for the Arrow 2 because:

1.1.1 It appears to be the only practical solution to the operation of such large chord, highly loaded control surfaces as are employed on the airplane.

1.1.2 Any reversible type of system would require that the control surfaces be mass balanced, and in view of the large chord surfaces used this would involve a prohibitive weight penalty.

1.1.3 Any type of aerodynamically assisted control system employing geared tabs, etc., is ruled out because of the unpredictable and variable nature of supersonic airflow as far aft on the wing chord.

1.2 Having chosen a fully powered control system, it is necessary that a standard of reliability comparable to a mechanical system be achieved. To this end, a system employing a minimum number of actuators having max. reliability has been selected, and complete duplication of power systems has been provided for.

1.3 An hydraulic power system has been selected because of its reliability, actuator compactness, system response, and performance under extreme environmental pressure and temperature conditions. (Ref P/Systems/25, Description of Flying Control Hydraulic System).

1.4 Because of the large power requirements for control of the aircraft the hydraulic power is derived from engine driven pumps, one half of each flying control hydraulic system being driven from each engine. This arrangement provides for full control forces at reduced max. rate of movement during single engine operation, and thereby permits complete control during the most extreme asymmetric thrust conditions.

1.5 The airplane has two fundamental modes for control, Emergency Mode and Electrical Mode.

1.5.1 The Electrical Mode utilizes various control inputs to the damper circuits to operate the command (parallel) servos in the case of elevator and aileron. These servos operate the surface actuator control valves through mechanical linkages. The rudder is controlled by turn co-ordination signals produced within the damper circuits and fed to the differential servo mounted on the rudder actuator.

1.5.1.1 The control inputs available are divided into three groups; pilot inputs, automatic flight inputs and pilot assist inputs. For pilot stick inputs, stick force per "G" feel is provided except for special cases such as landing and take-off where positional feel is provided.

1.5.2 The mechancial method of control for elevator and aileron is an emergency mode primarily for use in the event of a failure of the electrical mode. This mode utilizes a conventional cable and a mechanical linkage system to control the surface actuator control valves. In this mode, feel and trim are provided by positional feel springs. Rudder turn co-ordination is provided in emergency mode through the emergency damper.

1.5.2.1 A mechanical rudder system is incorporated to permit the pilot to make unco-ordinated turns. This system is hinge moment limited by a feel spring having a spring rate variable with q_c in order to protect structure. This system also provides a method of control in the event of a double yaw damper failure.

1.5.3 A normal damping system is provided to give stability augmentation about all three axes. In addition, an emergency damping system is provided which gives stability autmentation in yaw only. This emergency system is made necessary by the marginal stability of the aircraft in yaw. Turn co-ordination is incorporated into both normal and emergency damping systems. A differential servo is inserted into the linkage for each control surface. These servos are capable of moving the control surfaces independently of command inputs. In the aileron and elevator systems feed-back from the differential servo to the stick is prevented by the parallel servos. In the rudder system, since no parallel servo exists, the spring feel unit is utilized to react differential servo feed-back.

1.5.4 Safety devices are incorporated which will cause the roll or pitch axis damper to disengage independently if roll rate normal acceleration or acceleration in pitch threaten to produce excessive loading on the aircraft. If the safety margin in yaw is approached, the complete normal damping system is disengaged leaving the aircraft in mechanical control for pitch and roll with emergency turn co-ordination and damping in the yaw axis. This arrangement is used since the yaw case is the most critical and yaw limits may possibly be exceeded by pitch or roll damper malfunction due to cross coupling effects.

1.5.5 Speed Brakes are provided on the aircraft and are operated manually by a selector switch which controls two hydraulic jacks. In the interest of reliability of the Flying Control System, the brakes are powered by the Utility Hydraulic System. More detailed information on the Speed Brakes is given in 72/ Systems 19/26.

1.5.6 <u>Brochure Layout</u>

The flying controls are subsequently described in greater detail under the following headings:-

2. Design Objectives
3. Modes of Control and Auto Stabilizer
4. Detailed Description of System
5. Appendix 1 Quotes from Military Specifications
6. Appendix 2 Deviations from Military Specifications
7. Appendix 3 Drawing Numbers of Significant Components

2. Design Objectives:

2.1 General

To provide a fully power operated system for the operation of the primary control surfaces, consisting of elevator, aileron and rudder.

2.1.1 To provide a primary electrical system of controlling the surface actuators which will give suitable response and electric stick feel. Such a system is to be capable of accepting command inputs from an electronic fire control system or other electronic Auto Flight Control Systems.

2.1.2 To provide an additional system of controlling the surface actuators by utilizing a conventional mechanical input system in all axes and simple artificial feel installation, which is to provide adequate stick force per unit stick displacement and stick force per "G" relationship.

2.1.3 To provide that switchover to the emergency system be automatic in the event of any failure of the electrical system which causes a dangerous build-up of load factors. Switching from one mode to another should not result in sudden changes in control forces.

2.1.4 To provide, in the case of the mechanical rudder input system, a feel system which will prevent the pilot from inadvertently applying large rudder deflections at high equivalent air speeds.

2.1.5 To achieve a very high degree of reliability of the primary flying control system by utilizing a completely independent, duplicated hydraulic power system, and by duplicating the input system as outlined in 2.1.1 and 2.1.2 and also 2.1.3.

2.1.6 To provide a control system which will give full rate operation at a soaked temperature of 0°F and limited performance at -20°F. The system is to be capable of being rapidly warmed up from -65°F to operating temperature by pilot's input to the control system with the pumps running. See Fig. 28 for graph of time required to warm-up.

2.1.7 To restrict nominal operating temperature to a maximum of 250°F, and local hotspots to 275°F, by the use of heat exchangers in the hydraulic circuits (ref: 72/Systems 32/25 to permit the use of proven sealing methods and materials in the construction of the actuators.

2.1.8 To provide cable system tension regulation, by the use of mechanical tension regulators, to compensate for the effects of temperature changes and structural deflections.

2.1.9 To provide a normal damping system capable of fulfilling the stability augmentation requirements of the aircraft.

2.1.10 To provide an emergency damping system capable of producing satisfactory stability augmentation in yaw which will engage automatically in the event of normal damping system failure.

2.1.11 To provide turn co-ordination circuits within both the emergency and normal damping systems to minimize sideslip in manoeuvres.

2.1.12 To provide for unco-ordinated manoeuvres at option of pilot.

2.1.13 To provide a primary flying control system capable of superimposing inputs from the damping system upon command inputs to the control system.

2.1.14 To provide a system which will produce adequate control during an asymmetric flying case, resulting from failure of an engine at any speed or altitude condition within the flight envelope.

2.1.15 To provide a mechanical system of adequate stiffness, free from structural feed-back, and effectively mass balanced, to preclude the possibility of spurious signals being transmitted to the control valves.

2.1.16 To provide a system which will meet the requirements of:

(a) AIR 7-4 Specification for Prototype Supersonic All-weather Interceptor Aircraft, Type CF-105.

(b) CAP 479 Aircraft Design Requirements for the Royal Canadian Air Force.

(c) ARDCM 80-1 Handbook of Instructions for Aircraft Designers.

(d) Publications and specifications referred to by the above publications (Ref. Appendix 1).

2.2 Elevator System

2.2.1 To provide an elevator movement of 30° up and 20° down at a max. rate of 40°/sec. and to provide for a max. hinge moment of 60,000 ft.lbs. per surface.

2.2.2 To provide the elevator pitch dampers movement authority of ±6° for any position of the elevator within the limits allowed by full travel.

2.3 Aileron System

2.3.1 To provide an aileron movement of 19° up and 19° down at a max. rate of 35°/sec at a max. hinge moment of 25,000 ft. lbs. per surface.

2.3.2 To provide the aileron roll dampers movement authority of ±7.5° for any position of the aileron within the limits allowed by full travel.

2.4 Rudder System

2.4.1 To provide rudder movement of ±30° at a max. rate of 40°/sec at a max. hinge moment of 15,000 ft. lbs.

2.4.2 To provide the rudder movement yaw damper authority of ±13°, for any position of the rudder, within the limits allowed by full travel.

2.4.3 To achieve the yaw damping requirement of para. 2.4.2 and 2.1.11 by the use of a duplicated signalling and power system, with automatic and immediate "switch over" from the normal to the emergency system in the event of a system failure.

3. Modes of Control

3.1 Electrical Mode

This mode for controlling the aircraft uses electrical inputs from a stick force transducer or an automatic flight control system to the damper system for controlling the aircraft. A constant stick force per "G" is provided for control stick steering in high speed flight. The selection of undercarriage down gives stick feel proportional to surface position for low speed flight such as landing and take-off.

The output from the damper system to the aircraft control system consists of movement of the main surface actuator jack control valves which in turn cause these actuators to move the surfaces. In the roll and pitch axes the pilot command is transmitted electrically to parallel servos, one for each axis, which servos move the relevant control valve by direct mechanical linkage. A differential servo inserted in the linkage acts effectively as an extensible link and can move the control valve (and hence the surfaces) in response to aircraft motion sensing devices, independently of the pilot commands. Any force talk back from this differential servo is reacted by the parallel servo and hence is prevented from feeding back to the stick. Under normal operation the pilot does not command rudder movement, the necessary co-ordination with aileron being effected by a signal produced by aileron deflection which is fed to a differential servo in the rudder system: thus no parallel servo is used and rudder differential servo talk back is reacted by using a fairly high value of break out force on the rudder bar. The rudder differential servo also receives damping signals from the aircraft motion sensors for stability augmentation purposes.

Since an independant emergency yaw damper system is incorporated the rudder bar is only used to introduce intentional side slip or when the complete damper system is disengaged at the pilots discretion.

Safety devices are incorporated which will cause the roll or pitch axes to disengage automatically and independently if roll rate, normal acceleration or acceleration in pitch threaten to produce flight conditions which would prejudice the structural integrity of the aircraft. (If the safety margin in yaw is approached the complete normal damping system is automatically disengaged and the aircraft is in mechanical control for pitch and roll, with emergency electrical co-ordination and damping on the yaw axes. This arrangement is used since the yaw case is the most critical and yaw limits may possibly be exceeded by pitch or roll damper malfunction due to cross coupling effects). The pilot may disengage the complete normal damper and engage emergency yaw damper by depressing a button mounted on the control column grip. The emergency yaw damper may not be engaged by depressing this button unless the normal damper is engaged.

3.1.1 Normal Flight Configurations

3.1.1.1 Pitch Axes (Fig 2)

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For high speed flight the inputs to the pitch axes system consist of pilots' stick force, normal acceleration and pitch rate. These inputs are sensed by a force sensor, an accelerometer and a gyro respectively which transform the physical quantities they measure into voltages. The stick force signal and the normal acceleration signal are added algebraically such that while the stick force signal exceeds the latter signal the parallel servo will move in a direction to give the desired control surface deflection. When an acceleration is established on the aircraft sufficient to provide a signal of equal magnitude and opposite sign to the stick force signal, the parallel servo will remain stationary at that position. Hence, the stick force at a prescribed "G" will be constant and also the stick force per "G" can be arranged to be constant. The error between the stick force signal and the normal acceleration signal is also passed to the differential servos, which will move to assist the pilot command thereby producing a more rapid initial response. Extraneous short period oscillations are also damped out by differential servo movement.

The pitch rate signal has its short period (greater than 4 cps) components passed to the differential servos. This provides a signal which anticipates normal acceleration and hence causes a correction which is correspondingly more advanced in time to be applied to dampen short period oscillations.

The low frequency components are lagged in time and fed to the parallel servo to damp out long period oscillations such as phugoid motion. The aircraft can be trimmed into manoeuvres of from -1 to +3 incremental G by feeding an electrical signal in addition to the stick force signal thereby effectively cancelling the desired amount of normal acceleration signal. This trim signal is introduced by a motorized potentiometer controlled by the pilots' stick trim button. As an adjunct to the damper system to prevent sudden large changes in stick force when the damper and electrical control system is disengaged, a spring unit (Ref 4. 10. 1) with a motor driven earthing point is attached to the mechanical linkage and the motor is driven in such a direction as to balance the applied steady stick force by loading the spring: thus on disengagement of the parallel servo the pilot will feel approximately the same force as previously at the stick position at which disengagement occurred. This same spring unit becomes the feel unit in emergency mechanical mode. Further devices are incorporated in the system which ensure that no stored commands can exist in the damper system prior to engagement since such stored commands would give sudden control surface and stick movements at the moment of engagement resulting in stick force or aircraft "bumps".

For landing and low speed flight with gear down, instead of the parallel servo moving when a signal is fed to it and stopping when no signal exists, one circuit element is by-passed, which results in a given signal producing a given parallel servo (and hence control surface) position: thus on application of stick force the control surfaces will move to a certain position and remain there just as long as the stick force is maintained, the result being a positional feel.

The normal acceleration signal is cut out to avoid inputs due to ground bumps on landing. The only alteration to the damping part of the system is to pass all pitch rate frequencies to the differential servos since it is no longer possible to lag the low frequency signals and feed them to the parallel servo without adding unwarranted complication to the system.

The pitch damper will be disengaged automatically upon the build-up of conditions about the pitch or the yaw axis which would tend to take the airplane beyond it's structural limits.

3.1.1.2 Roll Axis: (Fig 3)

For high speed flight the inputs to the roll damper comprise stick force and roll rate. These inputs are measured by a force sensor and a gyro respectively. The long period of the error between the stick force signal and the roll rate signal is passed to the parallel servo causing it to react such that the control surface moves to the commanded position. The whole content of the above mentioned error signal is passed to the differential servos which move to damp out short period oscillations causing roll. The effect of feeding the long period portion of this signal to the differential servos is to improve the response of the system to command inputs. Stick force per rate of roll is produced similarly to the stick force per "G" in the elevator system. Again by this system the control surface motion required for damping is not transmitted to the stick. There is no automatic equalization of stick force in electrical and mechanical mode as in the pitch axis. A similar auxiliary system as is used in pitch to prevent command storage and consequent engagement bumps is designed into the roll axis system as is also a disengage switch to disengage the roll axis damper if the rate of roll becomes excessive. Disengagement also occurs if the yaw axis limiter parameters are exceeded. Trim is achieved by the same means as in pitch. Positional feel in roll for low speed is produced by the same means as in pitch. A reduction of trim drag at high altitudes is achieved by a symmetrical up deflection of the ailerons (Ref. para. 4. 9. 2). This deflection is controlled through a pressure sensitive switch.

3.1.1.3 Yaw Axis (Fig 3)

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For high speed flight the inputs to the yaw damper are yaw rate for damping, lateral acceleration to prevent unintentional side-slip, aileron position which is required during rolling pull out conditions due to cross coupling effects which tend to produce a high angle of sideslip.

Yaw rate and aileron position are bi-passed to the differential servo which can, therefore damp oscillations and coordinate the rudder for temporary or fluctuating aileron movements, but a steady aileron displacement for trim will be filtered out and cause no corresponding rudder deflection. Similarly a steady yaw rate during a turn will be filtered out and no rudder deflection will occur. Lateral acceleration and the combined pitch rate times aileron position signals are passed directly to the differential servo since damping for these is required at all frequencies. A limiter is incorporated which disengages normal damping on the yaw axis if the maximum permissible lateral acceleration is exceeded or if a sideslip greater than 10^{o} exists. Sideslip is sensed by a vane on the aircraft nose boom.

The yaw axis is still co-ordinated with the roll axis in low speed flight but it is necessary to allow the pilot to produce intentional side-slip conditions. To this end the lateral acceleration signal is partially bi-passed which means that the rudder deflection due to low frequency signals, and in particular steady signals, is reduced and therefore the pilot can command such steady state side-slip conditions more easily by using his rudder bar and direct linkage to the actuator control valve.

3.1.1.4 Scheduling and Command Limiting

Since the required rudder co-ordination angle and also elevator angle per "G", etc. will vary with mach number and altitude the effect of the various inputs must be varied for different flight conditions. This is done in most cases by varying the gains on the inputs as a function of the compressible dynamic pressure. Also to assist in the rolling pull out case, the pilot command signal to the ailerons is reduced if a lateral acceleration exists on the aircraft. As an added precaution the pilots' force transducers are fitted with stops so that they cannot produce a signal which will command a value of "G" which exceeds the structural integrity of the aircraft no matter how much force is applied to the stick.

3.1.2 An automatic flight control system (AFCS), is provided which produces various signals which can be substituted for stick force signals to the damper. This system consists of Mach

Hold, Altitude Hold, or Attitude and Heading Hold as pilot assist functions as well as several pilot replacement functions such as attack Navigation, such functions can be used to automatically guide the aircraft on an electronically computed attack course.

Providing no further selection is made, heading and attitude hold take effect at time the stick is released when AFCS is engaged. Two switches are provided for the pilot aside the AFCS engage switch. With one, either altitude or mach hold can be selected to replace attitude hold and with the other either navigation or attack can be selected (replacing attitude and heading hold). Heading may be adjusted by control stick steering during either altitude or mach hold.

In order that the pilot may override such automatic control, a switch is incorporated in the stick grip which switches out the AFCS signal when operated by the pilot. Also a stick force operated switch is incorporated which when open (i.e. no force on) stick will cause the stick force balancing spring to be left in such a position as to return the stick to a position, corresponding to 1/2 "G" incremental acceleration in the event of automatic damper disengagement. This ensures that the aircraft will not be left trimmed into a high "G" maneouvre with the pilots' hands off the stick. This switch also transfers control to the stick when the AFCS attitude hold functions are in operation.

3.2 The Emergency Mode

3.2.1 In the emergency mode the elevator and aileron control valves are operated by a cable and mechanical linkage system which is terminated by the control column. Artificial feel and trim are provided by means of positional feel springs and by a turn-co-ordination circuit in the emergency damper. The mechanical connection between the rudder pedals and the rudder control valve is used only to produce intentional side-slip or as a "Last-Ditch" method of controlling the rudder in the event of a complete failure of both the normal and the emergency damper. A system for deflecting the aileron as in 3.1.1.2 is also provided.

Emergency damping is provided in yaw only and is a complete duplicate of the normal system with the exception that the rolling pull-out signal is not included since this circuit is not fully effective when the roll damper is not functioning. This duplication of components is taken as far as the differential servo electro hydraulic valve and electrical signal feed back generator. The differential servo cylinder and piston are duplicated as well and a duplicate hydraulic supply system is used. The emergency yaw system electrical supply is taken from the electrical supply system emergency bus, whereas the normal damper is supplied from the normal bus. The

Mechanical System is Described in 3.2.2.

3.2.2 The Mechanical System

Identical tension regulator quadrants for each control run, situated under the cockpit floor, are driven by control column and rudder pedal movement and convey this movement by cable to an elevator quadrant and aileron quadrant situated in the aft fuselage below the wing, and to a rudder quadrant situated in the fin.

3.2.2.1 The position for the introduction of the elevator and aileron command (parallel) servos into the mechanical system is at the aft fuselage quadrants as mentioned in para. 4.9.

3.2.2.2 The rear elevator quadrant is connected by push rods to the bell cranks at the elevator jack pivot points. The other arm of each bell crank, which is inside the wing, is connected to the centre of the main lever of the elevator jack follow-up mechanism which operates the valve controlling the jack itself as described in section 3.2.2.5.

3.2.2.3 The rear aileron quadrant is connected, by push rods to the inner wing tension regulator quadrant levers. Movement is then conveyed from this quadrant by cables, along the rear face of the rear spar and close to the neutral axis, to the quadrant at the aileron jack pivot point. This quadrant is connected by push rod to the centre of the main lever of the aileron jack follow-up mechanism which operates the valve controlling the jack itself.

3.2.2.4 The rudder quadrant in the fin is connected by push rod to a bell crank at the rudder jack pivot point. The other arm of the bell crank is connected by push rod to the centre of the main lever of the rudder jack follow-up mechanism which operates the valve controlling the jack itself.

3.2.2.5 The follow-up mechanism is similar for elevator, aileron and rudder jacks. Primary movement of the main lever is effected by movement of the push rods connected at its centre as described in previous paragraphs. The main lever pivots about an inboard extension of the jack piston rod so that actuation of the jack induces the follow-up motion. The other end of the main lever is connected by push rod to the valve operating lever which pivots about an end fitting on the differential servo piston rod and is connected at its other end to the end of the valve spool. Primary movement of the main lever therefore effects movement of the valve operating lever and so displaces the valve spool. The jack is displaced accordingly until the follow-up process induced by secondary movement of the main follow-up lever closes the valve again at the required control surface deflection. By pivoting the valve-operating lever on the end of the differential servo piston rod, the valve position may be modified by differential movement of the valve operating lever resulting from damping servo movements during the electrical modes to achieve the required artificial damping.

3.2.2.6 Displacement of the jack is conveyed from the outboard end of the piston rod by a bell crank and push rod system described in section 4.17 to the control surfaces.

3.2.2.7 Artificial feel for the emergency mode is built into each system by the use of spring-loaded feel units. The aileron unit is situated at the quadrant under the cockpit floor, the elevator unit is at the rear quadrant, and the rudder unit is at the quadrant in the fin. The rudder quadrant feel unit also supplies hinge moment limitation. The elevator unit also performs an auto trim function during electric stick operation. The artificial feel units are described in section 4.10.

4. Detailed Description

4.1 Hydraulics (See 72/Systems 32/25 "Description of Flying Control Hydraulic System")

4.1.1 Pressures and Temperatures of the System

Normal Operating Pressure	=	4,000 p.s.i.
Proof Pressure	=	6,000 p.s.i.
Bursting Pressure	=	10,000 p.s.i.
Normal System Return Pressure	=	90 p.s.i.
Max. Operating Temperature	=	275°F
Min. Operating Temperature	=	-65°F

4.1.2 The system consists of two pumps per engine, with one pump from each, providing fluid under pressure for the "A" system and the other pump supplying the "B" system.

4.1.3 The hydraulic control surface actuators are of the tandem (dual piston and cylinder) type, each piston being driven by a completely independent system.

4.1.4 In the event of a single engine failure the pressure of each system is maintained at half flow capacity which is adequate for a limited capability not involving maximum rate of movement of control surfaces.

4.1.5 In the event of the loss of a single hydraulic system maximum rates of control surface movement are obtainable but with reduced hinge moment.

Under this condition the aircraft is limited by reduced available "G" throughout most of the flight envelope.

4.1.6 The double differential servo unit which actuates the rudder jack valve consists of two integral units, one of which is supplied by the "B" hydraulic system for normal operation, and the other by the "A" system for emergency operation.

Similarly the electrical and electronic signalling network is duplicated because of the vital nature of the yaw axis damping.

4.1.7 The elevator and aileron differential servo units, which actuate the jack valves for damping purposes, are supplied by the "B" hydraulic system only.

4.1.8 The command (parallel) servos, which actuate the jack valves on receipt of signals from the electrical mode circuits are supplied by the "B" hydraulic system only.

4.1.9 The system is designed for full rate operation at a soaked temperature of 0°F, and will give limited capability at -20°F. The system may be warmed up to these operating conditions by manual movements of the cockpit flying controls during engine running within the times shown on figure 28.

4.2 <u>Control Column (Fig 7)</u>

4.2.1 The control column, which is located at station 155.26, is of the conventional stick type, and operates the tension regulator quadrants under the cockpit floor through a lever system. A stick force transducer is mounted in the grip.

4.2.2 Max. stick travel at grip for elevator movement = 10.99 ins.

Max. stick travel at grip for aileron movement = 9.96 ins.

Pilot limit load at grip for elevator movement = 200 lbs.

Pilot limit load at grip for aileron movement = 100 lbs.

4.2.3 Emergency mode trim control for aileron and elevator is by means of a thumb button control on the grip, which changes the earthing point of the feel spring by means of an electrical actuator.

4.3 Rudder Pedals (Fig 8)

4.3.1 The rudder pedals are suspended from an overhead hinge point at station 134.25 and are connected by push rods to the rudder bar which pivots about a torque tube passing through the cockpit floor. The torque tube is connected at its other end to the rudder tension regulator quadrant.

4.3.2 Trim control for rudder is by a switch mounted in the cockpit.

4.3.3 Total Rudder Pedal Travel = 6.31 ins for neutral adjustment.

Pilot Limit Load = 300 lbs.

4.4 Tension Regulators Under Cockpit Floor (Figs 19 and 9)

4.4.1 The cable tension regulators used for each control axis cable system in their run aft to the rear fuselage quadrants and fin quadrant, are identical, and are located under the cockpit floor as shown in Figs 1 and 9. For information on the cable run aft from the tension regulators see section 4.8.

4.4.2 Tension Regulator Diameter (to cable centres) = 12.0 ins.

Range of compensation	=	$2\frac{1}{2}$ ins per cable
Rig load variation for full range of compensation	=	35 lbs. min. 60 lbs. max.

4.4.3 The Elevator Tension Regulator Quadrant (Fig 19)

This is supported on a horizontal axis at station 168.25 and displaced 16 ins. to the R.H. side of the aircraft centre line.

The quadrant is supported by mounting brackets secured by bolts to two short fore and aft beams under the cockpit floor and is mounted on a torque tube 14 3/4" below the fuselage datum. The quadrant is driven by a 3.50 inch lever mounted on the inboard end of the torque tube and connected by push rod to an 8.36" crank on the control column.

A Bob-Weight which is also supported from this torque tube produces 43.5 lb. in/G about it.

4.4.3.1 Angular movement of elevator quadrant. = 60° total

Total cable movement = 6.283 ins.

4.4.4 The Aileron Tension Regulator Quadrant (Fig 19)

This is supported on a vertical axis, situated at station 158.77, 4.25 inch to the left of the A/C centre-line and 10 3/4 ins. below the aircraft datum, by a mounting built integrally with the control column mounting structure.

The quadrant is supported at the lower end of a torque shaft which passes through the cockpit floor. The upper end of the torque shaft carries a 3.67 inch lever which is connected by a short push rod to the control column.

A 3.67 inch radius lever attached to the same shaft carries the aileron spring feel unit and is located below the aircraft datum. The aileron spring feel unit is described in section 4.10.2.

4.4.4.1 Angular movement of aileron quadrant = ± 23°-39'

Total cable movement = ± 4.953 ins.

4.4.5 The Rudder Tension Regulator Quadrant (Fig 19)

This is a horizontal quadrant situated at station 126.6 and supported on the rudder pedestal 9.75 ins. below the fuselage datum.

4.4.6 Angular movement of quadrant = ± 20°

Total cable movement = ± 4.189 ins.

4.5 Aileron Tension Regulator - Inner Wing (Fig 20)

4.5.1 The tension regulator in the aileron wing cable run (see section 4.8.7) is situated at station 692.79 and 14.50 ins. outboard of aircraft centre line. The quadrant transfers a total cable movement of 3.26 ins. from the rear fuselage system to the wing cables by means of a shaft passing through the bottom surface of the wing. The lower end of the shaft is connected by a lever and push rod to the rear fuselage quadrant.

4.5.2 Tension regulator diameter = 5.65 ins.

Range of compensation = 1.28 ins.

Rig load variation for full range = 35 lbs. min.
60 lbs. max.

4.6 Bob Weight (Fig.9)

The purpose of the bob-weight mentioned in 4.4.3 is to supplement the feel springs to provide some natural feel proportional to "G" in the pitching plane. The effect, of aircraft normal acceleration on it, is to produce an opposing force at the stick grip which is added to the positional feel of the spring system. The force produced at the stick for neutral stick position is 4.25 lb/G and is the sum of the bob-weight force and the force due to unbalance of the stick. A balancing spring is provided, such that the bob-weight will not be felt by the pilot in a level flight condition.

4.7 Stick Force Transducer (Fig.29)

The elevator and aileron stick force transducer consists of a housing which contains the differential transformer type transducers and switches, a stick grip which is attached to the housing by means of a stiff spring, and a transistor amplifier. The housing is rigidly attached to the top end of the control column. The transistor amplifier is mounted in the cockpit.

Pilot input forces to the stick grip for both axes are translated into electrical signals for transmission via the transistor amplifier to the damper system.

The transducer unit includes switches which permit the passing of AFCS signals to the damper system at times when the force applied to the stick grip is less than approx. 1 lb. for either axis.

Adjustable Mechanical stops are also provided in the unit in order to limit its output and therefore its command authority.

4.8 Cable System (Figs 1, 4, 9, 5, and 6)

4.8.1 The main cable run connects the tension regulator quadrants under the cockpit floor (see section 4.4) with the elevator and aileron quadrants in the rear fuselage (see section 4.9) and with the rudder quadrant and "feel" system in the fin. (see section 4.10.3). There is also a cable run in the wing, transferring movement from the rear fuselage aileron quadrant by way of a tension regulator in the wing (see section 4.5) out to the aileron jack quadrant.

4.8.2 All cables are 1/8 in. dia. to spec. MIL-C-1511. They are aluminum clad to .201" dia. over as great a length as possible in order to reduce elastic stretch, and to increase their overall coefficient of expansion to a value more nearly that of the airframe structure.

4.8.3 The following table gives the approximate percentage of bare and clad cables to total length together with approximate cable movements.

Cable	Total Length Ins.	% Total Length		Total Cable Movement Ins.	Total Tension Regulator Compensation Ins.
		Bare	Aluminum Clad		
Elevator	1047.6	24.0	76.0	6.283	2.5
Aileron (Fuselage)	1081.9	19.0	81.0	4.95	2.5
Aileron (Wing)	828.24 (2)	23.0	77.0	3.26	1.28
Rudder	1301.2	16.5	83.5	4.189	2.5
Overall	4258.94	20	80		

4.8.4 All cable end fittings are swaged ball ends which fit into sockets at the end of the cable groove on the quadrants. The ball ends are locked in the sockets by means of cotter pins fitted to the sockets.

4.8.5 Adjustment of all cables is provided for by the use of turnbuckles which, in the case of the main cable runs, are located in the armament bay area between station 285 and station 485. A minimum of 1/4 inch longitudinal clearance is provided between turnbuckle ends.

4.8.6 The main cable run is carried between quadrants by a total of 32 pulleys and these are distributed as follows:

Station	No. of Pulleys & Locations From C.L. Aircraft					
	Elevator		Aileron		Rudder	
	L.H.	R.H.	L.H.	R.H.	L.H.	R.H.
187.						1
187.3					1	
188.8			1			
189.			1			
194.2		1				
228.		1	2		1	1
283.		2	2		1	1
487		2	2		1	1
507.5					1	1
512.9		2	2			
583.8					1	1
620.5		1				
629.3			1			

The main cable run aft from the front fuselage tension regulator quadrants is as follows:-

From the tension regulator quadrants to stations 228 the cables run close under the cockpit floor. Rising between station 228 and 283 they then run close under the armament bay roof to station 487. Rising again between station 485 and 512 they run aft close under the lower wing surface. The rudder cables change direction at station 583 and run up into the fin through $\frac{1}{2}$ in. dia tubes to the rudder feel unit and hinge moment limitation quadrant. The elevator and aileron cables continue on under the lower wing surface to attach to the rear fuselage quadrants.

4.8.7 The Wing Cable Run

The wing cable run between the wing tension regulator and aileron jack quadrant is carried by 4 pulleys positioned along the aft face of the rear spar of each wing. They are distributed spanwise as follows:

1 pulley centred 1.35" inboard of T.E. Rib #1 intersection with rear spar datum

1 pulley centred 1.50" inboard of T.E. Rib #5 intersection with rear spar datum

1 pulley centred 1.40" inboard of T.E. Rib #5A intersection with rear spar datum

1 pulley centred 2.20" inboard of T.E. Rib #6 intersection with rear spar datum

4.9 Rear Fuselage Quadrants (Figs 4 and 5)

The rear fuselage quadrants transfer elevator and aileron cable movement from the main run in the fuselage (see section 4.8) out to the control surface actuator systems in the wing.

4.9.1 The Elevator Rear Fuselage Quadrant: (Fig 10)

is supported on a horizontal torque tube at station 675.75,19.58 ins. above aircraft datum.

A push rod connects the quadrant with the R.H. elevator jack pivot point bell-crank.

A similar push rod connects the L. H. elevator jack pivot point bell-crank with a lever on the opposite end of the torque shaft. This quadrant is the position at which both the elevator command (parallel) servo (see 4.11.2) and the elevator feel unit (see 4.10.1) are introduced to the elevator system.

The total angular movement of the quadrant is 60° and with a diameter of 10.0 ins. (cable centre). This caters for a total cable movement of 6.283 ins. less cable stretch.

4.9.2 The Aileron Rear Fuselage Quadrant (Fig 10)

This is situated on the centre line at station 683.14 close under the bottom surface of the wing and mounted in a horizontal position.

This quadrant is the position at which the aileron command (parallel servo) is attached (see 4.11.3).

Push rods between levers connect the quadrant with the aileron tension regulators in the wing.

An electrically activated adjustable link, controlled by an altitude sensitive switch, automatically varies the angle between the quadrant levers. At high altitudes, this results in a 4° up movement of the ailerons (both in the same direction). This feature was incorporated to provide an improvement in high altitude manoeuverability of the aircraft.

The total angular movement of the quadrant is 60^{o} and with a diameter of 8.9 ins. (cable centres) this caters for a total cable movement of 4.95 ins. less cable stretch.

4.10 Feel and Trim Units (Figs 4 and 5)

The feel units of the elevator, aileron and rudder systems are essentially positional springs of a suitable spring constant earthed to the fixed structure through an electrical trimming device. In addition, the rudder system employs a mechanical system to vary the bungee effective spring constant with q_c in order to restrict the pilot's input to the rudder actuator as a function of the compressible dynamic pressure.

For elevator and aileron the trim section is an electric actuator and brake operating through a gear train.

In the emergency mode the pilot controls the aileron and elevator trim actuators by means of the thumb button on the control column.

The rudder unit is controlled by a switch in the cockpit.

4.10.1 The Elevator Feel and Trim Unit (Fig 21)

This unit is connected to the rear fuselage elevator quadrant. The unit is earthed to structure by having its fixed end attached to the bottom end of a lever suspended from the bottom skin of the wing.

The lever is rigidly attached to structure through an electrically operated emergency release mechanism which may be released by a manual switch in the pilots compartment. Release of the mechanism allows the lever to pivot freely about the top end, and will allow the system to operate through its full range if the feel/trim unit is jammed or seized or runs away.

A ball detent clutch is provided integral with the feel trim unit so that a force of approx. 65 lb. at the control column will support it against a runaway trim motor but which will support a 150 lb. load if the motor is inoperative. The unit has been positioned as near to the elevator actuator control valves as is practical in order to minimise the effect of cable elasticity and backlash on system response when operated on the mechanical mode. Experience on fully powered elevator systems has indicated that this type of arrangement is desirable. The elevator feel spring is the positive break-out stick force type adjustable from 0 lb to 7.5 lb.

The trim range forward is 33% of full forward stick movement and the trim range aft is 60% of full aft stick movement.

For trim forces under non accelerated flight see the graph in the system loads column of Fig 4. 1.

In emergency mode the unit is purely a feel and trim device, however with electric stick it performs an auto-trim function. To accomplish this function it is controlled by a differential pressure switch in the parallel servo in such a way that it assumes the loads in the system initially reacted by the parallel servo.

This is done to prevent a sudden increasing "bump" force at the stick if the parallel servo disengages. A curve is shown on Fig 12 indicating the characteristics of this auto-trim function. The trim range forward is 33% of full forward stick movement and the trim range aft is 60% of full aft stick movement.

For trim forces under non accelerated flight see the graph in the system loads column on Fig 4A.

4.10.2 <u>The Aileron Feel and Trim Unit (Fig 22)</u>

This is connected to the aileron tension regulator quadrant under the cockpit floor and is earthed to the bulkhead at station 176 (see Fig 9).

The aileron feel spring unit is of the positive break-out type, with a break-out stick force adjustable from 0 lb to 5 lb. The trim range is 40% of stick movement, symmetrical about the centre.

For trim forces see the graph in the system loads column on Fig 5A.

4.10.3 Rudder Feel and Trim Unit and Hinge Moment Limitation System (Fig 14)

4.10.3.1 The rudder feel and trim unit and hinge moment limitation system is installed in the rudder control system close to the rudder jack between fin ribs #4 and #5 and spars #7 and #10. At this position, the main rudder cable run (see section 4.8.1) transfers movement to the rudder jack valve linkage by means of a push rod connection between a cable quadrant and the bell crank lever at the jack pivot point. The rudder system feel and hinge moment limitation is achieved by means of the mechanism shown in Fig 14, the power for driving the mechanism being supplied by an electric actuator acting on signals proportional to q_c. Trim is obtained by means of an electrical actuator mounted on the cable quadrant. The actuator displaces the earthing point of the lower feel unit, up to 20° in either direction, as required changing the geometry of the feel system and thus unloading the controls.

4.10.3.2 This diameter of the quadrant to cable centres is 5.5 ins., and this caters for a total cable movement of 4.189 ins. less cable stretch at an angular movement of $\pm 30^\circ$.

4.10.3.3 The reason for placing the feel and hinge moment limitation system as close to the rudder jack valve as is practicable, is to ensure full movement of the rudder over the very small range available when the system is subjected to max. hinge moment limitation conditions.

Under these conditions displacement of the control system is limited, but deflection may be a maximum.

Deflection in the part of the system between the feel unit and rudder jack valve is reduced to a minimum by positioning these items close together, while the high deflection between the feel unit and rudder pedals is catered for by unrestricted rudder pedal movement.

Curves of rudder deflection vs. pedal force with relation to q_c are shown in Figs 26 and 27. A curve showing maximum available trim vs. q_c is presented in Fig 27.1.

4.11 Parallel Servos (Figs 4 and 5)

4.11.1 General

The parallel servo is an electro-hydraulic actuator which performs the function of moving the aircraft controls upon command from the stick force system or the A.F.C.S.

It consists of an hydraulic ram with force limiting over-power valves, and electric torque motor type control valve, a solenoid operated two stage engage valve which automatically dis-engages servo in event of electrical or hydraulic supply failure or shut down, a small differential pressure sensing ram which actuates an incorporated sensing device and operates whether servo is engaged or disengaged, and a differential transformer type feed back which senses servo ram position.

The parallel servos are powered by the "B" hydraulic system and control the main elevator and aileron control valves through the mechanical control linkage in parallel with the pilots' mechanical system. They may be overpowered by the application of a predetermined force on the control stick. They can also be disengaged automatically or manually by electrically de-energizing the solenoids.

4.11.2 The Elevator Parallel Servo (Fig 36)

This is attached by its piston rod to the bottom part of the rear fuselage elevator quadrant. The pivot point is mounted forward of the quadrant on a bracket supported from the lower wing surface.

The servo is force limited to 175 lbs, and at an arm of 3 ins. exerts a max. moment of 525 lbs. ins. on the quadrant which may be overpowered by 53.5 lbs. at the stick grip. The servo is unloaded by the elevator feel spring.

4.11.3 The Aileron Parallel Servo (Fig 10)

This is attached by its piston rod to a lever on the rear fuselage aileron quadrant. The pivot point is mounted aft of the quadrant on the fuselage support yokes at station 679.39. The servo is force limited to 90 lbs. and at an arm of 3 ins. exerts a max. hinge moment of 270 lbs. ins. on the quadrant, which may be overpowered by 30 lbs. applied at the stick grip.

4.12 Differential Servos (Figs 4 and 5)

The differential servo is a high response electro-hydraulic actuator which performs the function of differentially actuating the main control valve to apply artificial damping control. It consists of an hydraulic ram with device to self centre servo in event of failure or shut down of electrical or hydraulic supply, an electric torque motor type control valve, a solenoid operated two stage engage valve which automatically dis-engages in event of failure or shut down of electrical or hydraulic supply and a differential transformer type feed back which senses servo ram position.

In the case of the elevator and aileron installations the units receive hydraulic power from the "B" hydraulic system through internal connections to the main control valve on which they are mounted directly.

Because of the critical nature of the rudder system its servo valve is a duplicated unit receiving hydraulic power from both the "A" and "B" hydraulic systems by similar internal connections to the main rudder control valve. In the non-duplicated elevator and aileron units provision is made for spring self centering on shut-down of the servo electrical or hydraulic supply. The rudder unit self centers similarly if both normal and emergency systems fail. These units (for elevator and aileron) are identical, having an output force of 300 lbs. and a stroke of ±0.375 ins.

The standby electrical system is continuously in operation and on failure of either the normal electrical or hydraulic power source the solenoid by-pass valve admits fluid to the stand-by side and by-passes the normal side.

The rudder servo has a 300 lb. minimum output force and a stroke of ± .05 ins.

4.13 <u>Actuator Valves (Figs 11, 13 and 15)</u>

The actuator valves are of the double tandem type employing a common spool and two separate sets of pressure and return ports, one set each for the "A" and "B" hydraulic systems. They are mounted directly on the actuators with internally connected hydraulic systems between valve and actuator and provision is made for internal hydraulic connection between them, and the differential servos which are mounted directly to them as outlined in 4.12.

Interchangeability is provided between valves and actuators and valves and differential servos of similar type.

Control of the actuator valves is effected through the follow up mechanisms as described in 3.2.2.5.

Valve travels are as follows:

Elevator spool travel	± 0.125 ins.
Aileron spool travel	± .07 ins.
Rudder spool travel	± .05 ins.

4.14 Control Surface Actuators (Figs 4, 5 and 6)

4.14.1 General

The hydraulic control surface actuators are of the tandem dual piston and cylinder type with each piston supplied by a separate hydraulic system.

The use of high temperature flexible hose is avoided by the use of semi-coiled steel line, which is adequate for the small amount of jack movement involved.

Flexible hose is only used on the return lines of the elevator and aileron units and on one return line for the rudder unit.

4.14.2 Elevator Jack (Fig 16 and 11)

The root end of the jack pivots about a pre-loaded self-aligning bearing which is located 12.323 ins. outboard of the aircraft centre-line at station 698.66. In order to facilitate removal of the jack through the access panel in the wing top skin this pivot point bearing is an interchangeable assembly bolted to the jack body. It is mounted on a support projecting from the fin box structure at the centre of the wing.

4.14.2.1 Data

For curve of available max. hinge moment see Fig 23.

Static jack load at 4,000 p.s.i.	71,000 lbs.
Cylinder bore (max.)	4.19 ins.
Piston rod O.D. (max.)	2,50 ins.
Working area (2 x 8.88)	17.76 ins.^{2}
Normal operating stroke	10.0 ins.
Total stroke	10.20 ins.

4.14.3 Aileron Jack (Figs 13 and 17)

The root end of the jack pivots about a universal joint which is located 5.5 ins. outboard of Rib 18 and 4.60 ins. aft of the centre spar. This universal joint utilizes four needle bearings and is mounted on a support which is attached to the wing skin and centre spar.

In order to facilitate removal of the jack through the access panel in the bottom skin, the universal joint is an interchangeable sub-assembly bolted to the jack body.

4.14.3.1 Data:

For curve of max. static hinge moment see Fig. 24.

Static jack load	42,100 lbs.
Cylinder bore (max.)	3.125 ins.
Piston rod O.D. (max.)	1.75 ins.
Working area (2 x 5.26)	10.5 $ins.^2$
Normal operating stroke	5.56 ins.
Total stroke	5.76 ins.

4.14.4 Rudder Jack (Figs 15 and 18)

The root end of the jack pivots about a pre-loaded self-aligning bearing which is located 10 ins. outboard of rudder station 42 and 2.75 ins. aft of spar #3. The jack is mounted on a support projecting from Spar #3 and Rib #4 and is removable through an access panel on the port side of the fin.

4.14.4.1 Data:

For curve of max. static hinge moment see Fig 25.

Static jack load	30,900 lbs.
Cylinder bore (max.)	2.75 ins.
Piston rod O.D. (max.)	1.625 ins.
Working area (2 x 3.865)	7.73 $ins.^2$
Normal operating stroke	5.9 ins.
Total stroke	6.10 ins.

4.15 Mechanical Connections Between Jacks and Control Surfaces

4.15.1 General

The linkage from the end of the jack piston to the actual control surface is similar for all three systems. It consists of a long push rod, joined by a master bell crank to the fork end of the jack piston, and connecting a row of bell cranks which convey motion to the control surfaces by way of a link attachment.

The bell crank system is housed in the fin and wing trailing edge structure which extends from the rear spars to the hinge spars in the vicinity of the jacks.

4.15.2 Elevator Bell-Crank System (Figs 1 and 4)

The elevator output system consists of the lateral push rod and six bell cranks per side which are located aft of the wing rear spar between ribs #4 and #17. The bell cranks are steel, each housing a needle bearing at the pivot point and a self-aligning bearing at the push rod attachment. The master bell-crank contains an additional self aligning bearing at the actuator pick up.

The assembly is supported at the pivot points by stiff forged aluminum alloy ribs which are attached to the rear spar and to the thick wing skins.

The lateral push rod is a one piece machining with a fork end at the master bell-crank pick-up and slots for the other bell crank attachments.

The connecting links between the bell cranks and elevator fittings are machined steel forgings and house a self-aligning bearing at each end.

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4.15.3 Aileron Bell-Crank System (Figs 1 and 5)

The aileron output system consists of seven bell-cranks and a push rod per side, all located aft of the wing spar between ribs #17 and #25 and supported at the pivot points by aluminum alloy ribs.

The bell-cranks are steel each housing a canted roller bearing at the pivot point and self-aligning bearings at the pick-up points.

Each push rod consists of short steel rods flash welded together at the ends, each having a slotted fitting at the bell-crank pick-up points.

The connecting links between the bell-cranks and the aileron are machined steel forgings consisting of a fork end at the bell crank pick-up point and a self-aligning bearing at the aileron attachment.

4.15.4 Rudder Bell-Crank System (Figs 1 and 6)

The rudder linkage system is of similar detail design to that of the aileron, except that it consists of a single push rod and only five bell-cranks all located aft of the fin spar #3 and outboard of rib #4.

The bell-cranks are supported at the pivot points by the integrally machined skin of the fin trailing edge structure.

4.16 Control Surfaces

4.16.1 The Elevator:

This surface is constructed of thin light alloy skin attached to formed light alloy ribs and an extruded light alloy front spar.

Extruded integrally with the spar is one half of the elevator piano hinge, which runs along the leading edge on the upper surface.

The ribs at the control points are stiffened by the extrusions which attach to the connecting links from the bell-cranks.

4.16.1.1 Data

Elevator area (2)	106.90 ft.^{2}
Elevator span (each)	10 ft. - 2 ins.
Elevator chord (constant)	5 ft. - 3 ins.
Elevator movement	30° up
	20° down

4.16.2 The Aileron

This is constructed similarly to the elevator, but at each control point the complete rib is formed by the machined extrusion which picks up with connecting links from the bell cranks.

4.16.2.1 Data

Aileron area (2)	66.5 ft.^{2}
Aileron span (each)	10 ft.
Aileron chord	5 ft. - 3 ins. at root
	1 ft. - 4.8 ins. at tip
Aileron movement	19° up
	19° down

4.16.3 The Rudder:

This consists of light alloy skins attached to a formed light alloy spar and rib. It contains seven hinge points, all off set to the starboard side; the middle five, which form part of the ribs, support the bearings which pick up with the connecting links from the bell cranks. The bottom hinge point, which takes the vertical loads, and the top hinge point, are separate fittings which pick up with oilite bearings in the fin.

4.16.3.1 Data

Rudder area	38.17 ft.^{2}
Mean Rudder Height	119 ins.
Rudder Chord	5 ft. - 4.67" (mean)
	5 ft. - 5.14 ins. at root
	2 ft. - 3.7 ins. at tip (mean)
Rudder movement	± 30°

APPENDIX 1

The flying control system is designed to meet the relevant sections of the following requirements as noted hereunder:

1. In Document AIR 7-4

2.4 Design

2.4.1 The aircraft shall be designed in accordance with the Canadian Air Publication 479, "Manual of Aircraft Design Requirement for the Royal Canadian Air Force", the technical requirements of the United States Air Research and Development Command Manual No. 80-1, "Handbook of Instructions to Aircraft Designers", publications and specifications referred to therein and such additional requirements as may be specified by the Department and such deviations from these specifications and requirements as may be agreed between the Department and the Contractor. Where there is any conflict between the requirements of CAP 479 and those of USAF ARDCM 80-1, the former shall govern.

3.4 Scramble Time

3.4.1 With the aircraft at normal gross weight and positioned at the end of the runway, the elapsed time from the pushing of the first button to start the first engine until the aircraft becomes airborne shall not be more than one minute.

4.1 Design Requirements - General

4.1.4 The aircraft shall be designed such that the pilot can perform all the normal and emergency functions required to fly the aircraft without the assistance or presence of the second crew member.

4.4 Speed Brakes

4.4.1 Speed brakes shall be fitted to produce additional drag when opened. The controls shall be arranged such that the pilot can set the brakes at the fully opened, intermediate, or fully closed position. Actuation of the speed brakes shall have a minimum effect on the trim or attitude of the aircraft throughout the speed range of the aircraft.

4.6 Protection From Enemy Fire

4.6.1 During the design of the aircraft special consideration shall be given to the building-in of invulnerability to enemy fire by the maximum use of such inherent types of protection as positioning of components, fuel and hydraulic lines and like items, providing such positioning does not result in unreasonable complication or weight penalty.

4.7 Servicing and Stand-by

4.7.1 Servicing

4.7.1.1 Turn-Around

The time required for "turn-around" shall not be more than 5 minutes. The turn-around operation shall include replenishment of all consumable stores and liquids and the required spot checking of equipment to bring the aircraft to a state of readiness such that it shall be capable of fulfilling its primary role as detailed in this specification.

4.7.1.2 Special servicing tools and ground handling equipment, subject to the prior approval of the Department shall be delivered concurrently with each aircraft.

4.7.2 Stand-by

4.7.2.1 The aircraft shall be designed such that, once having been certified serviceable, it shall be capable of maintaining a state of immediate readiness for a period of at least 24 hours in the open.

4.7.2.2 Complete stand-by equipment required to meet the provisions of paragraph 4.7.2.1, subject to the approval of the Department shall be delivered concurrently with each aircraft.

7.4 Automatic Control System

7.4.1 Components of the type E10 autopilot suitably coupled to accept signals from the fire control system and navigation system shall be installed. The system shall provide three axis stabilization and shall automatically control the aircraft during normal flight, attack and breakaway.

7.4.2 Manual control shall be possible at the option of the pilot during any phase of the flight.

10.2 Fire Control System

The aircraft shall be equipped with a two man position Fire Control System.

2. In CAP 479

Chapter 21 Controls

Section 1 - General

21.02 Clearance and Ease of Operation

Controls shall be designed and located so that the operator can readily move each control throughout its entire range of travel, without moving any other control, while wearing heavy gloves and flying equipment, and with shoulder harness in place, but not necessarily locked.

21.03 Direction of Motion

(1) Except for three-position switches, where the centre position shall be "OFF", all controls shall be so designed that their movement in a predominantly forward, upward, or clockwise direction shall result in increased performance of the component or the aircraft, and, conversely, their movement in a backward, downward, or counter-clockwise direction shall result in decreased performance of the component or the aircraft.

(2) All variable controls operated by a rotary motion shall move clockwise from the "OFF" position through "LOW" or "DIM" to "HIGH" or "BRIGHT".

21.04 Shape and Location of Control Knobs

To assist identification without visual reference, control knobs shall be of distinctive shape. All controls of a like function should be grouped together, with normal operating and emergency controls having a preferred position.

Section 2 - Flight Controls

21.20 **General**

The control column and the rudder pedals shall be symmetrical about the plane of symmetry of the pilot's seat. All parts of the aircraft structure, auxiliary controls, furnishings, instruments etc., shall clear by at least one inch the control wheel or column and rudder pedals in all positions as well as the limbs of the pilot while he is wearing heavy gloves and flying equipment.

21.22 **Rudder Controls**

Rudder pedals shall be at least six inches wide to accommodate heavy flying boots. The rudder pedals shall be adjustable horizontally through a range of six inches. The rudder pedal adjustment mechanism shall be conveniently located and may be arranged for operation either by hand or foot. Depending upon the type of mechanism used, the motion of the control to move the rudder pedals forward shall be clockwise, forward, or upward. If the pedals are locked in position by a catch, the motion to release the catch should be arranged in the most convenient manner.

21.25 **Speed Brake Control**

The speed brake control shall be located on the power control or adjacent to the power quadrant in stick - controlled air-craft. The control motion shall be aft for the speed brake operative, and the control shall be marked "OUT" and "IN" respectively.

21.26 Automatic Pilot

In single or tandem pilot aircraft all automatic pilot controls, except the emergency electrical release switch, shall be located on the right hand side of the cockpit adjacent to the radio controls. The emergency electrical automatic pilot release switch shall be on the control column grip.

24.01 Temperature Range

(1) All aircraft shall be designed so as to ensure satisfactory operation up to their service ceiling under the ICAO Standard Atmosphere Conditions and at ground lever under atmospheric conditions associated with temperatures ranging from -65°F to +100°F. All accessories and items of equipment however shall be designed for satisfactory operation over a temperature range of -65°F to +160°F, and when not in operation to withstand without damage temperatures up to 180°F.

(2) The aircraft, its accessories and equipment shall not be adversely affected by other climatic conditions incident to the temperature ranges specified in paragraph (1) and shall be capable of being transferred from one climate to

another without the necessity of extensive modification or adjustment.

24.30 Allowance for Winter Clothing

Aircraft operating in the Arctic normally have to be maintained in the open by personnel clad in heavy and bulky Arctic clothing. Provision shall be made in the design for ease of maintenance under such conditions ...... All controls and switches shall be large enough and so placed that they can be operated by personnel wearing heavy mitts and flying boots.

24.31 Control Systems

Suitable precautions shall be taken to prevent slackening of control cables caused by differential contraction of cables and airframe in low temperatures. Control cables shall be fitted with automatic tensioning devices where necessary, to prevent excessive slackening.

24.33 Snow Guards

The use of snow guards is recommended to exclude blowing snow from airframe components such as wings, stabilizers, control surfaces, etc. It is particularly important that openings near control surface hinges be effectively sealed against the entry of blowing snow and rain, since the formation of ice on control surface hinges might lead to loss of control in flight.

24.34 Seals

Particular attention should be given to the selection of materials and design of seals for shock absorbers, hydraulic jacks and valves which must be able to withstand operation and to remain leakproof after prolonged exposure to extremes of temperature.

24.35 Pipe Joints

Joints in pipe lines must remain leakproof throughout the temperature range specified in article 24.01.

Chapter 50 General Reqts.

Standards and Specifications

See CAP 479

3. In ARDCM 80-1

4.11 Control Surface Lift Reversal

4.111 Measures Generally used for Prevention

The parameter most important for avoiding control surface lift reversal is a high torsional rigidity of the fixed surface supporting the control surface -----

4.112 Detail Requirements for Prevention

The requirements for the prevention of control surface lift reversal are contained in Specification 1817.

5.40 Primary Controls

The size, proportions, location and motions of the primary control surfaces shall be such as to give performance in accordance with AF Specification R. 1815 ---------- .

6.A.11 Flight Controls

6.A.110 General

The primary flight controls shall be located and have the motions as indicated in drawings AD. 1 and AD. 2. The primary flight controls shall clear all aircraft structures, auxiliary controls, furnishings, instruments, instrument panel, pilot's actuating members, etc., by at least $1\frac{1}{2}$ inches in all control positions ---------.

6.A.116 Speed Brake Control

The speed brake control shall be located on the throttle in stick controlled aircraft. The control motion shall be forward for "in" (speed brake inoperative) and aft for "out" (speed brake operative).

8.3 Bearings, Universal Joints, Pulleys and Hinges

MIL. JAN and AN Standard bearings, pulleys and universal joints shall be used wherever possible ....

see Chapter 8. Detail Design ARDCM 80-1.

9.01 Design for Maintenance

Control surfaces shall be capable of replacement without disturbing rigging.

9.1 Flight Controls

Flight controls include primary flight controls - ailerons, rudder, elevators ...... and flight path angle and speed controls .... These devices shall be operable from the cockpit either through direct or servo means.

9.10 Extreme Temperature Operation

Flight control systems shall be designed for operation at temperatures between +160°F (+70°C) and -65°F (-54C). The systems shall be designed to permit take-off of the aircraft within 5 minutes from the starting of the engine after the aircraft has remained idle for 72 hours at either the highest or the lowest temperature indicated above. After the initial breakaway, the increase in force required to operate the control system at -65°F (-54°C) shall not exceed 150% of the force required at +70°F (+21°C).

9.11 Cable Control Systems

The reliability, strength and simplicity of the system shall be the paramount considerations. Routing shall be as straight as possible and the number of bends kept to a minimum. Cable control systems shall conform to para. 10.41 Cable Actuated System.

9.12 Push-Pull Tube Control Systems

Wherever push-pull tube systems are used in flight controls, they shall be so arranged that all the tubes are in tension for the greater load for which the system is designed. Push-pull tube control systems shall conform to para 10.42., Tube Actuated System.

9.13 Shielding

Wherever possible, advantage shall be taken of the shielding afforded by heavy structural members or existing plate installations for cable protection, such armour plate to be extended where necessary to protect vital places in the systems such as points of cable convergence, horns, bellcranks, main sheaves and walking beams..

9.2 Primary Flight Control Systems

9.20 General Requirements

Control mechanisms and systems shall be as simple and direct as possible and shall be adequate for their loads and functions. Early and careful consideration shall be given in new designs to the arrangement of cables and other connecting elements that extend from the cockpit to the control surfaces so as to effect the most direct routing possible.

The number of bends or changes in direction shall be held to a minimum.

All practical compromises in the installation of equipment shall be made to favour the most direct control system possible..... A power control system shall be used ...... where the magnitudes of the hinge moments are too high for the pilot to handle through a mechanical control system. In order to reduce pilot effort and fatigue, the degree of mechanical efficiency shall be such that requirements of Specification 1815 will be met for the frictional forces developed in the control system. For this purpose, antifriction bearings shall be used throughout the control system. Reference para. 8.31, Antifriction Bearings. The mechanism and controls shall be located so as to permit convenient, rapid and frequent inspection.

9.200 Control System Devices

Control system devices such as bob weights ----- spring bungees for trim ----- tension regulators etc., shall be submitted in drawing form with other data to the procuring agency for necessary approval. Consideration shall be given in the detail design of the control system and component parts so that their failure will not cause discontinuity of the control system, large changes in trim, or any other momentary flight hazard.

9.201 Differential Controls

Differential controls may be used either to secure better aerodynamic efficiency or to provide higher sensitivity in a portion of the range of movement of a control.

Control systems in which differential motion is obtained by an arrangement of cranks shall incorporate stops to prevent the cranks from reaching the dead-centre position -------.

Toggle motions, which may go by dead-centre when the system is sprung, forced, or maladjusted, shall not be used.

Stops shall be incorporated to prevent approach to critical positions.

9.202 Fool-Proofness

Control systems shall be so designed that incorrect assembly and reversed operation of controls will be impossible. This can be accomplished by so arranging turnbuckle ends, cable lengths, and joints that the assembly will not function in case of error.

9.203 Fouling Prevention

All control elements, mechanisms and cables shall be suitably protected or covered in cockpit ----- to prevent their being fouled by the dropping of articles, such as microphone, gloves etc. ----- .

9.204 Irreversible & Damping Features

Irreversible and damping features will greatly inhibit the development of control surface flutter, but such damping should never be attained by the increase of friction which would reduce the efficiency of control. These features should be such as to oppose rapid motions energetically, but not interfere with slow motions of the controls. Fluid damping is suitable for the purpose.

In the design of inreversible or damping control systems, particular attention should be paid to avoiding the use of any device in which play or backlash might develop and eliminate the irreversible or damping properties of the system, thus encouraging flutter of the control surfaces. In designs where play or backlash might develop, the irreversible or damping features of the system will not be accepted as valid reasons for waving the usual dynamic balance requirements for the control surfaces involved.

9.205 Power Control Systems

Where power control systems are employed, an emergency manual or power means shall be provided and the system shall be designed to meet the requirements of Spec. 1815.

9.206 Rigidity of Controls

The design of the control system and its structural supports shall be based on rigidity and the elimination of deflection, as well as upon the ultimate strength requirement of Specification C-1803, superseded by MIL-S-5707.

Provisions shall be made to compensate for slack and lost motion due to the effect of temperature changes. Only approved types of mechanical tension regulators are permissible and if used, shall maintain rigging tensions over the full range of temperatures specified in para. 9.10, Extreme Temperature Operation, and shall compensate for any change in tension that may develop in the control system due to loads applied thereto. Consideration shall be given to the effect of heat from local areas such as engine nacelles, cabins, heat de-icers, etc., which may cause temperature rises in an adjacent portion of a control system while the aircraft structure proper remains at the ambient air temperature. The designer shall determine proper rigging tensions in control cables and shall specify them in erection and maintenance instruction. It may be found desirable or necessary to revise the originally specified tensions after flight and other tests have been completed. In the interest of reducing control system friction, initial tensions should be held to the lowest practical values that provide safe and satisfactory operation considering probable application of limit loads to the system and the effect of temperature changes. As a general rule, the minimum initial tension in a main or counter cable is either 2% of the breaking strength of the cable, or high enough to keep the system snug, prevent cables from jumping off pulleys and avoid slapping of adjacent portions of the aircraft by the cables, whichever is the greater. As a general rule, initial tension for main cables in normal fighter types should be not less than 35 lbs. to 50 lbs. When power control systems are used, the rigidity and balance of the control surfaces involved shall be such as to preclude flutter or undesirable oscillations if the actuator or any one of the actuators used is disconnected for any reason including battle damage.

9.207 Spacing

Cables to any one control surface shall be separated by at least 3 inches, preferably more. Adjacent control-horns, or bellcranks, shall be separated by the maximum distance possible consistent with design.

9.21 Duplicate Controls

9.210 Aircraft Other Than Bombardment

(a) Elevator: Elevators shall be rigidly interconnected or consist of a continuous structure. ------- in aircraft of more than 7,500 lbs. gross weight the direct operation system shall be duplicated from the base of the stick or control column to the elevator spars ......

(b) Rudder: Where cables are used for the rudder control on aircraft equipped with a single rudder duplicate cables shall be provided from each pedal to the rudder mast ----.

(c) Aileron: Aileron systems shall be so designed that, from the cockpit outboard, any or all elements leading to one aileron may be destroyed without interference with the operation of the other aileron.

9.212 Vulnerability

Power Control Systems - Power Control Systems shall be equal to, or better than mechanical systems from the standpoint of resistance to battle damage.

9.213 Stops

Adjustable stops shall be provided to limit the cockpit control items to the desired motion ranges. The stops shall be located as near these items as possible. Stops shall maintain the clearances specified in para. 6.A.110, Flight Control, General.

10.4 Mechancial System

10.40 General

Clearances in all the primary parts of mechanisms shall be kept to a minimum so that lost motion or play will be practically eliminated.

10.400 Bearings

All moving joints between push-pull rod, levers and brackets shall have anti-friction bearings in accordance with para. 8.31.

10.401 Horns and Brackets

Horns and brackets used in air control systems shall be forgings, or high grade castings of aluminum alloy, magnesium alloy, bronze or steel. Welding shall not be used -----. Horns may be fastened to torque tubes with rivets or taper pins. In the latter case, taper pins shall be safetied by a nut and cotter pin at the small end.

Attachment of control brackets shall be made with aircraft bolts and castellated or self-locking nuts of approved type. Rivets may be used provided that none are in tension.

10.41 Cable Actuated System

The kinematic relation of the cables shall be such as to prevent an objectionable amount of change in cable tension throughout their flight and ground operational range.

10.410 Cables

Control cables shall conform to Specification MIL-C-1511 or MIL-C-5424. ----- Continuous or intermittent loaded cables shall not be subjected to critical bends at the junction with cable terminals or other attaching points such as drums, horns, etc.

10.411 Pulleys

Pulleys shall be of adequate capacity and diameter for cables used. They shall conform to and meet the requirements of paragraph 8.315, Aircraft Control Pulleys.

10.412 Fairleads

Fairleads shall be used wherever necessary to keep cables from chafing and slapping against parts of the aircraft in flight or in taxiing. Fairleads may be used to deflect cables through angles of not more than 2 degrees if the initial tension in the cable is less than 50 lbs, and through angles of not more than one degree, if the initial tension is between

51 lbs. and 150 lbs. ----- They shall be of non-hydroscopic, non-abrasive material, and split to permit easy removal.

10.413 Terminals

Terminals shall be of the swaged, wrapped, or spliced type. Swaged terminals shall be used wherever possible. ----- Swaged terminals shall conform to Specification JAN-T-781. Type II (i.e. AN 663 and AN 664) swaged terminals shall not be used in primary control systems except for attaching primary control cables to bellcranks and quadrants where standard fork and eye fittings AN 667 and AN 668 are not adaptable.

10.414 Turnbuckles

Turnbuckle terminals shall have not more than three threads exposed at either end. All turnbuckle assemblies shall be safetied as specified in para. 8.25, Safety Wiring.

10.415 Maintenance

Adequate provision shall be made for rapid inspection of all components of cable actuated systems, and for convenient use of a tensiometer adjacent to the point of cable tension adjustment.

10.42 Tube Actuated System

10.420 Universal Joint

Universal joints shall conform to and meet the requirements of paragraph 8.314 Universal Joints.

4. IN 1815 - B

3. Mechanical Characteristics of Control Systems

3.1 Control System Friction

For all airplanes, the control system friction for all three controls shall be as low as possible, and shall not exceed the values given below throughout the deflection range. The friction, measured on the ground at a temperature between 40° and 80°F, shall be that required to start the movement of the controls.

Elevator Friction Limit = 4 lbs.

Aileron Friction Limit = 3 lbs.

Rudder Friction Limit = 10½ lbs.

5.6 Directional Trimming Devices

5.6.1 All prototype airplanes shall have provisions for maintaining directional trim in flight.

5.6.2 The rudder trimming device shall maintain a given setting indefinitely unless changed intentionally.

5.6.3 The directional trimming device shall be capable of reducing the pedal force to zero -----

6.6 Lateral Trimming Devices

6.6.3 The lateral trimming device shall be capable of reducing the aileron control force to zero -----

8. IN 1815 - B

8.1 Mechanical Design

8.1.1 The control system should be capable of moving the control surfaces at rates up to 50°/sec. under any hinge moments encountered.

8.1.2 When the cockpit control is moved at its maximum rate to a given deflection, the control surface shall reach its corresponding deflection with a time by not greater than 0.05 secs.

8.1.3 The dead spots in the controls should not exceed plus or minus 0.1 inches motion of the pilot's cockpit controls. The dead spots are defined as the motions of the pilot's cockpit controls which produce no motions of the control surfaces.

APPENDIX II

The following is a list of the points in which detail design varies from the requirements laid down in the terms of reference in Appendix I.

1. In AIR 7-4

All requirements satisfied.

2. In CAP 479

All requirements satisfied.

3. In ARDCM 80-1

3.1 Paragraph 9.206 states that rigidity and balance of the control surfaces shall be sufficient to prevent flutter if the actuator is disconnected. This requirement is not compatible with the design aims of a fully powered, irreversible flying control system and, if met, would involve prohibitive weight penalties. There is no more chance of the actuators being disconnected than the pilot's input control system, and the consequences of the latter would be equally, if not more, catastrophic.

3.2 Paragraph 9.207 states that cables to any one control surface shall be separated by at least 3 inches. This requirement has not been met because of space restrictions. The worst situation for each pair of cables is as follows:

2 1/4 inch spacing for elevator cables

1 7/8 inch spacing for aileron cables

5/8 inch spacing for rudder cables

The elevator and aileron cable minimum spacing occurs at a point just forward of the rear fuselage quadrants and reasonable assurance of non-fouling of these calbes ir provided by properly spaced fairleads. The rudder cable minimum spacing occurs at the pulleys which change the direction of the cable run along the fuselage up into the fin. The clearance of the cables is assured by running them through $\frac{1}{2}$ inch dia. tubes.

3.3 Paragraph 9.210a. states that elevators shall be rigidly interconnected or consist of a continuous structure.

This requirement is not met because of the difficulty of achieving complete synchronization between two jacks on a single surface, and also because of the severe weight penalty which would be paid for such an installation on an aircraft of the CF-105 plan form.

There do not appear to be any real advantages to the interconnection of the surfaces on this design.

3.4 In paragraph 9.21a, b & c, it is required that the control surface operating systems be duplicated.

This requirement is not met because of complexity and space reasons, and it is believed that this deviation is very common on many acceptable new designs, including the CF-100.

3.5 Paragraph 10.411 refers to paragraph 8.3 in which it is stated that all pulleys and quadrants shall be provided with stationary guards fitting close to the point of tangency of the control cables.

In most this requirement has been met, but the front fuselage tension regulator quadrants and the rear fuselage elevator quadrant are equipped with cable guards attached to the quadrants themselves. These guards, which move with the quadrants, are much lighter and very simple compared to fixed guards, and provide ample protection against cables jumping the quadrant grooves.

3.6 Paragraph 10.413 states that Type 11 (ball end) swaged terminals shall not be used in primary control systems except for attachment to bell cranks and quadrants where standard fork and eye fittings are not adaptable.

In the CF-105 control system, although it is possible to adapt fork and eye fittings for attachment to the quadrants, the use of tension regulator quadrants will take up any slack produced by the untwisting of loaded cable strands allowed by ball end type terminals.

The use of ball end type terminals results in a weight saving and a neat installation when compared with the result produced by the use of standard fork and eye fittings.

4. In 1815-B

4.1 Paragraph 8.1.1 states that the control system should be capable of moving the control surfaces at rates up to 50°/sec.

This requirement is not met. Dynamic analysis of the aircraft indicates that such high rates are not required, and the system has therefore not been penalized by an oversized hydraulic system.

APPENDIX III

Avro Part No.	Quantity	Description	Manufacturer & Part No. (Where Applicable)
7-3262-15 7-3262-18	2	Elevator Jack	Jarry Hydraulics 1020-3 & 1020-5
7-3262-33	2	Elevator Jack Valve	Minn. Honeywell D.V.G17B-1
7-3264-23 7-3264-24	2	Aileron Jack	Jarry Hydraulics 1420-3 & 1420-5
7-3264-12	2	Aileron Jack Valve	Minn. Honeywell D.V.G19B-1
7-3283-5	1	Rudder Jack	Jarry Hydraulics 1915
7-3283-8	1	Rudder Jack Valve	Minn. Honeywell D.V.G20A-1
7-3262-165 7-3262-167	1 1	Parallel Servo - Elev. Parallel Servo - Aileron	M-H DMG55N-1 M-H DMG55M-1
7-3260-11	2	Differential Servo (Elev. & Aileron)	Minn. Honeywell DMG51B-2
7-3283-7	1	Double Differential Servo - Rudder	Minn. Honeywell DMG62B-1
7-1552-165	3	Tension Regulator Quadrant - Front Fuselage	Pacific Scientific R.86-5001-50-00
7-1562-51	2	Aileron Tension Regulator Quadrant - Inner Wing	Pacific Scientific R.75-9006-50-00
7-1562-247	1	Spring Feel Unit - Elevator	AiResearch 3972-1
7-1552-341	1	Spring Feel Unit - Aileron	AiResearch 3972-2
7-1583-145	1	Spring Feel Unit - Rudder	
7-1552-3	1	Control Column	
7-1552-2	1	Rudder Pedals	

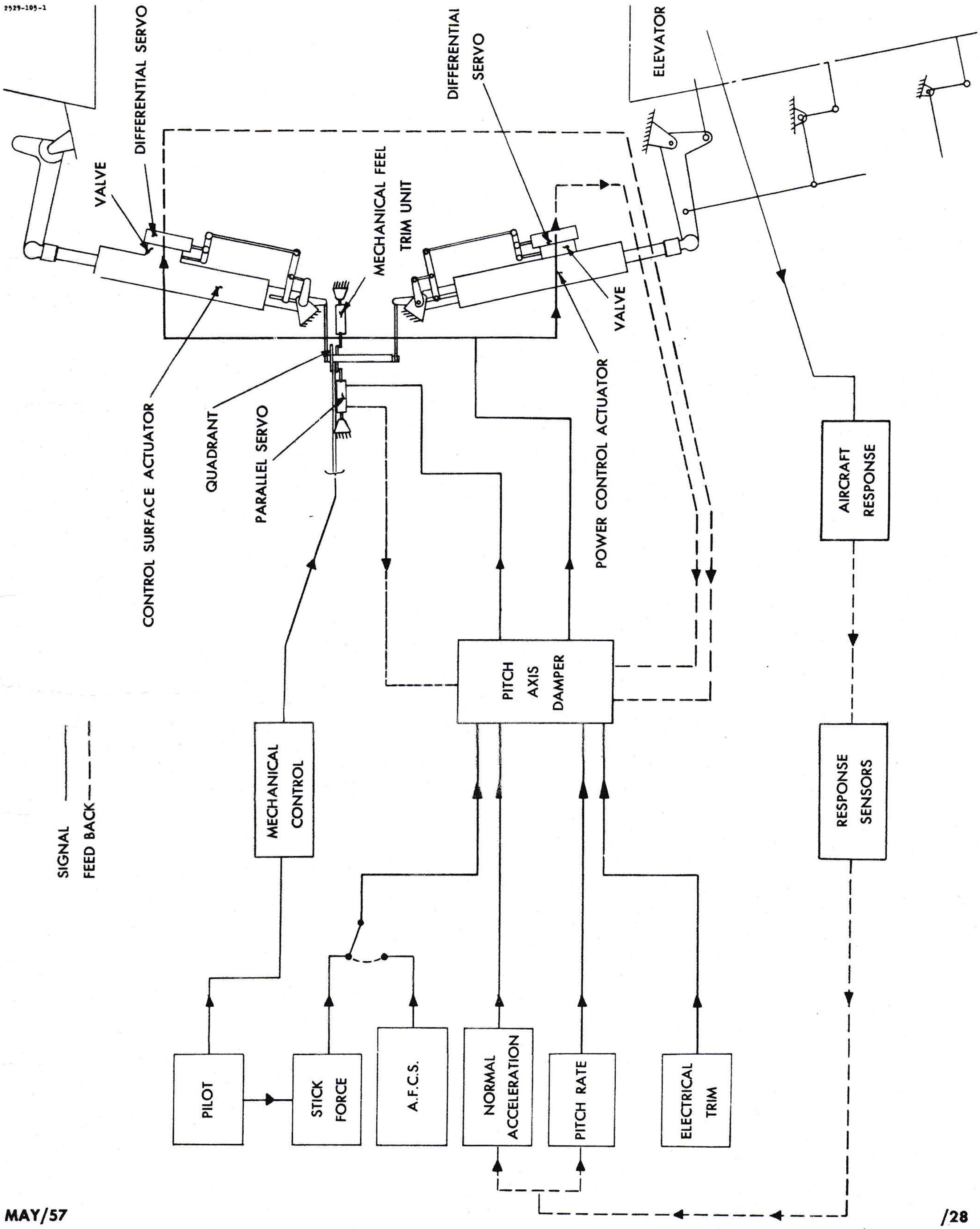


FIG. 2 ELEVATOR BLOCK DIAGRAM

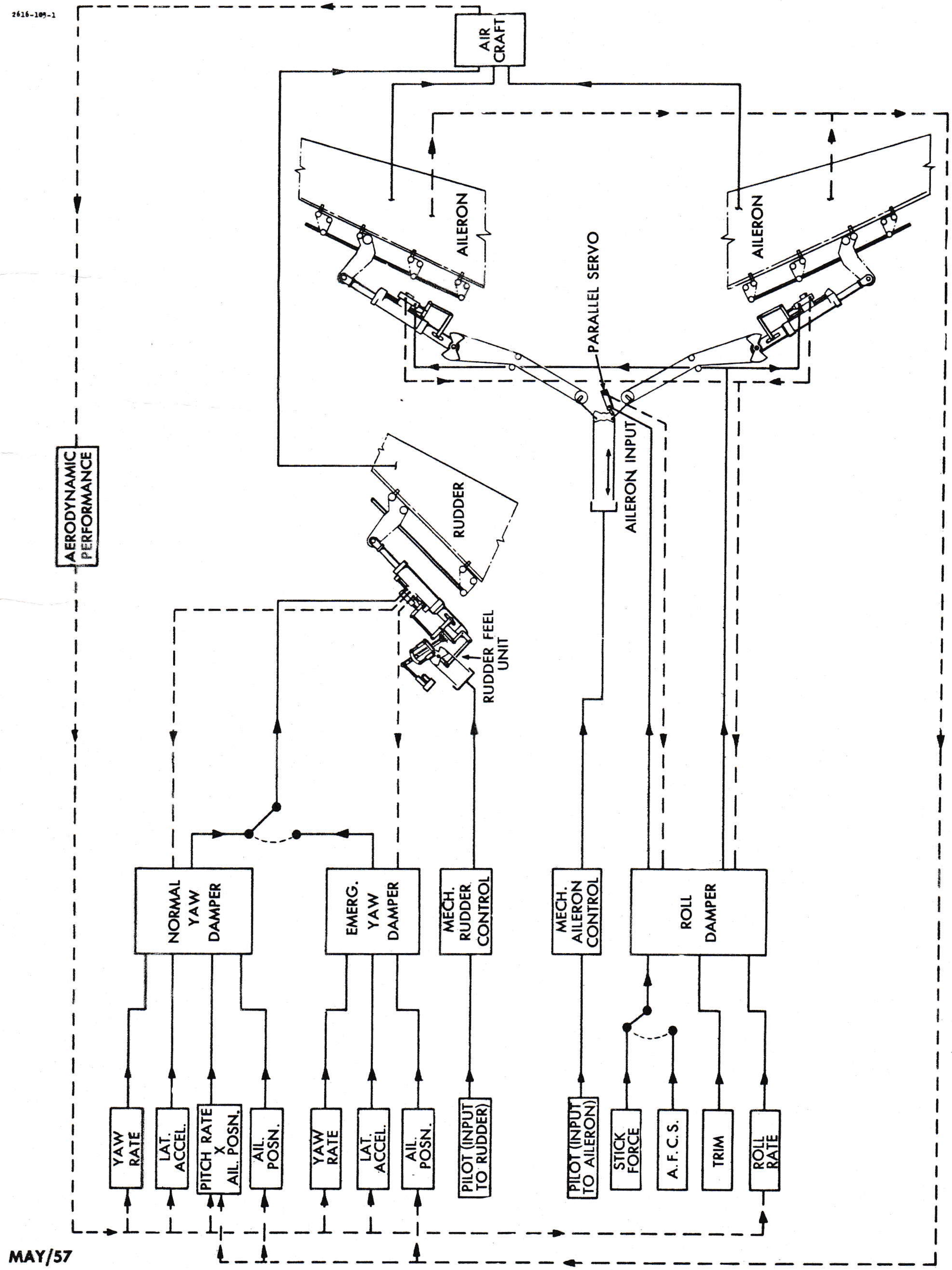


FIG. 3 RUDDER AND AILERON BLOCK DIAGRAM

2993-109-1

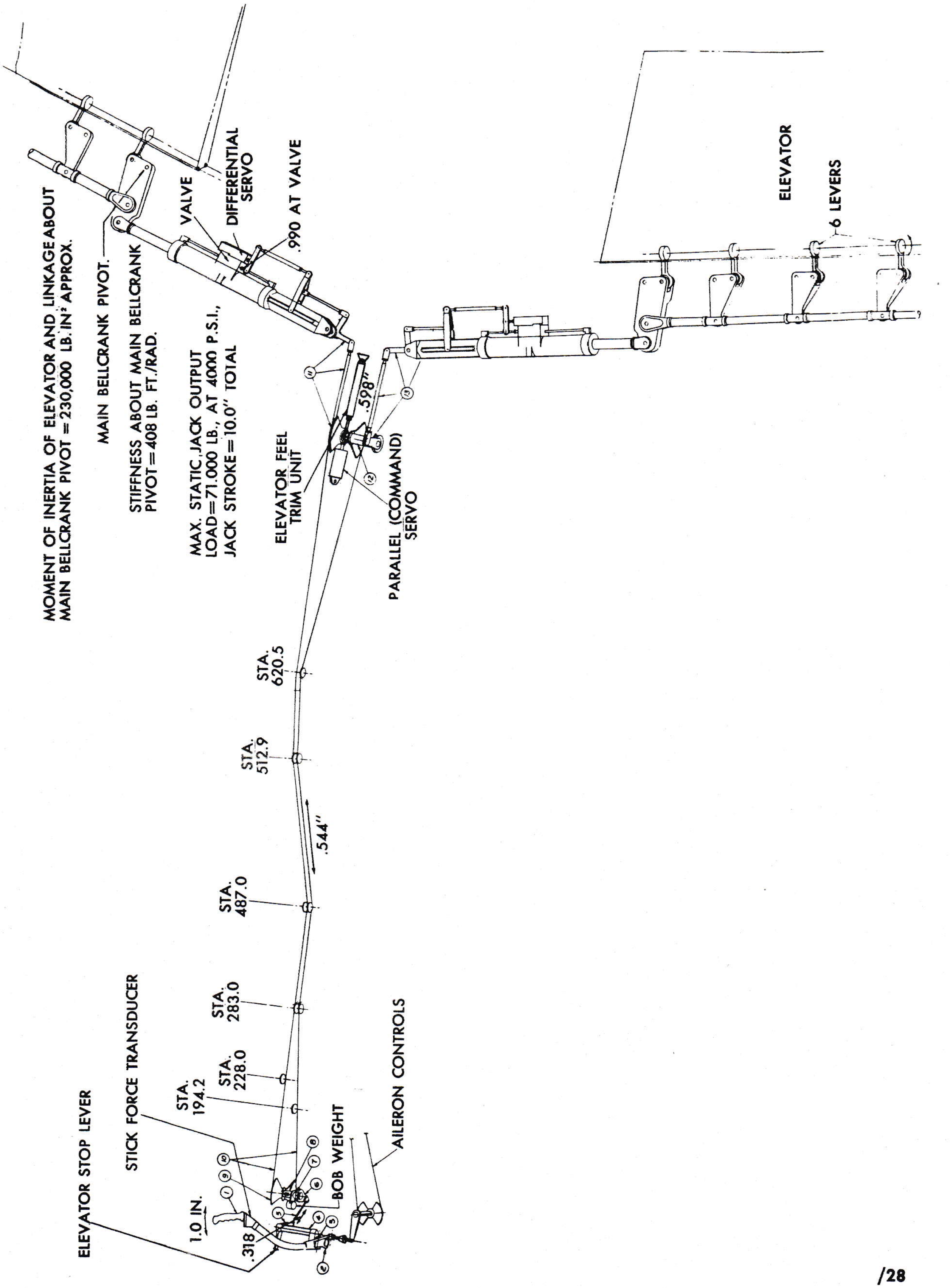
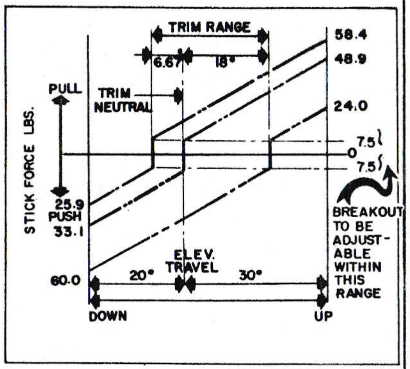


FIG. 4 ELEVATOR SYSTEM DATA

LOADS APPLIED TO SYSTEM

A-PILOT'S LIMIT LOAD 200 LB. AT STICK GRIP.

B-FEEL FORCES:



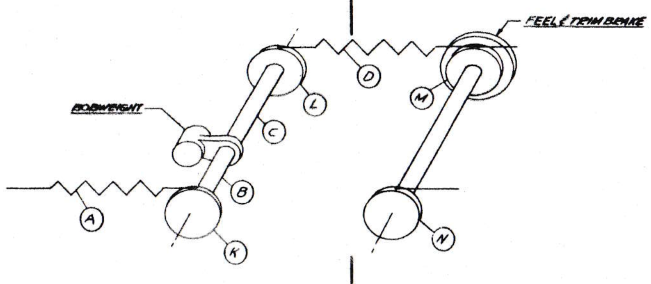
C-BOB WEIGHT
FORCE REQ'D AT STICK 4.25 LBS. PER 'G'
MOMENT REQ'D AT STICK-LIMIT NORMAL ACCELERATION 7.33 G= 817.7 LB.IN.-MADE UP OF-
STATIC MOMENT OF STICK 58.6 LB.IN.
STATIC MOMENT OF BOB WEIGHT 759.1 LB.IN.

D-PARALLEL SERVO
175 LBS. OUTPUT 3.0 STROKE ON 3.0 ARM AT UNIT.
53.5 LB. AT STICK.

E-DIFFERENTIAL SERVO
300 LB. OUTPUT ±.375 IN. STROKE.

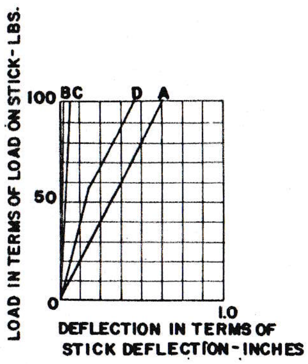
STIFFNESS

MOMENT OF INERTIA



See Fig. 4 for numbers of members.

STIFFNESS OF SECTIONS A, B, C, & D.



A-MEMBERS 1-6
B-FRONT TORQUE TUBE F'W'D OF BOBWEIGHT-MEMBER 7.
C-FRONT TORQUE TUBE AFT OF BOBWEIGHT-MEMBER 8.
D-MEMBER 10-CABLE.

MOMENTS OF INERTIA OF SECTIONS	EFFECTIVE INERTIA ABOUT STICK PIVOT LB. IN.^{2}
K CONSISTS OF MEMBERS 1-6.	2253.6
L CONSISTS OF MEMBERS 8,9,& 1/2 OF 10.	701.3
M CONSISTS OF MEMBERS 11 & 1/2 OF 10. PISTON & ROD OF PARALLEL SERVO	738.1
N CONSISTS OF MEMBER 13 WHICH INCLUDES LEVER ARM, PUSH ROD, & FOLLOW UP LINKAGE.	157.93
BOBWEIGHT	177.2
FEEL & TRIM	26.0

RELATIVE LOADS & DISPLACEMENTS OF MEMBERS OF SYSTEM.

See field of drawing for values of system movement. Assuming 1.0 in movement of stick, with zero load in system. Values are taken at mean position of stick.

FIG. 4A ELEVATOR SYSTEM DATA

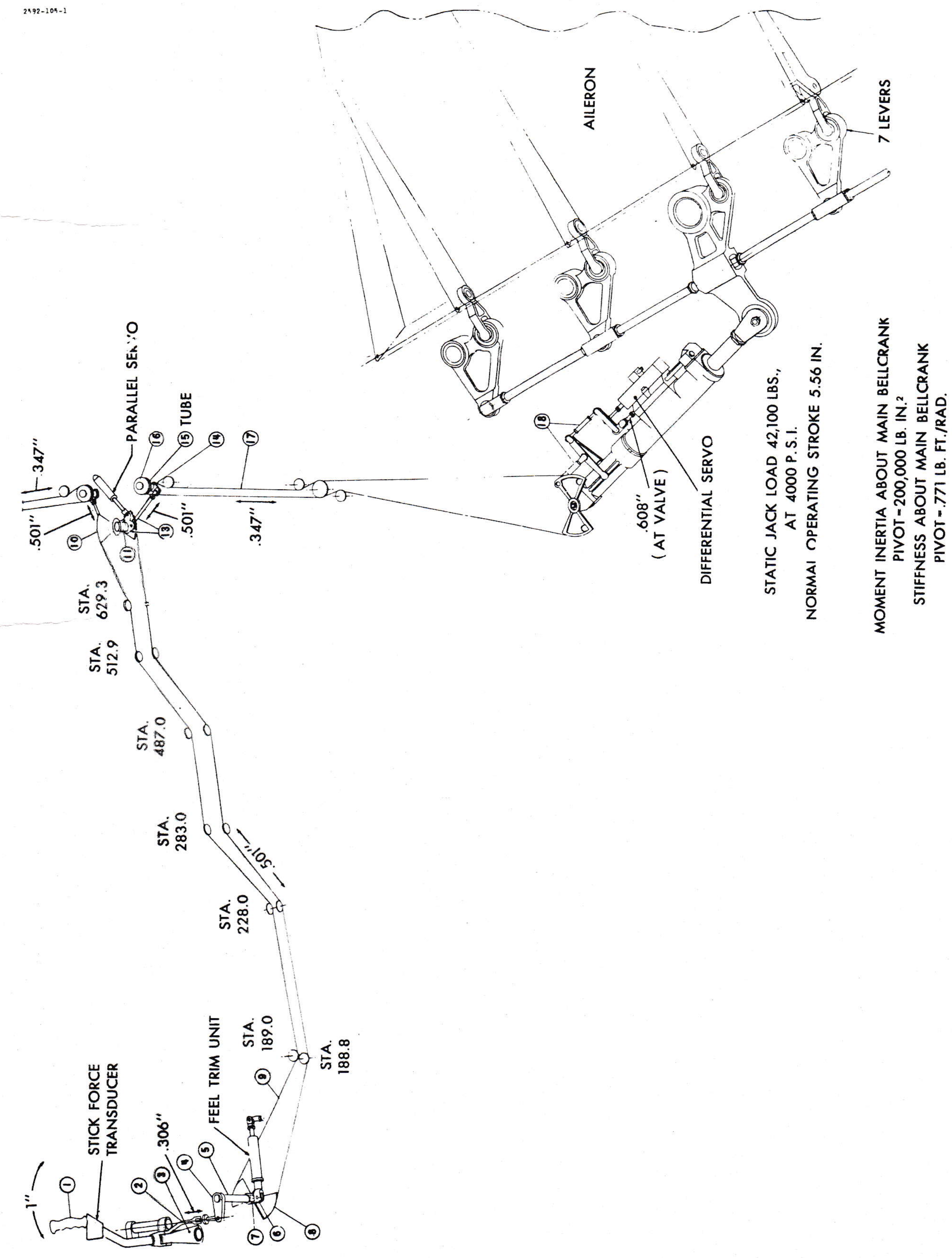
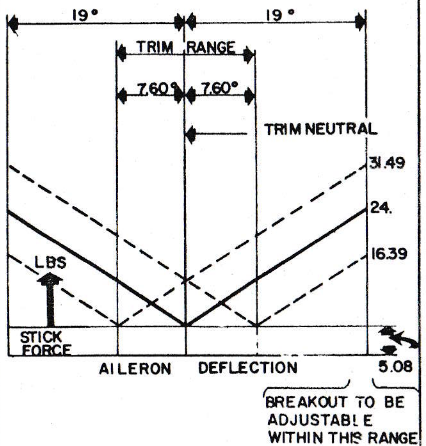
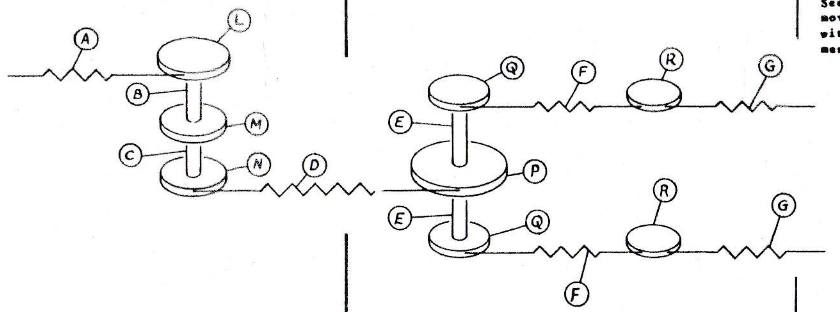
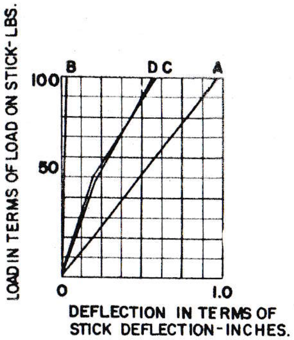


FIG. 5 AILERON SYSTEM DATA

LOADS APPLIED TO SYSTEM	STIFFNESS	MOMENT OF INERTIA	RELATIVE LOADS & DISPLACEMENTS OF MEMBERS OF SYSTEM.
A - PILOT'S LIMIT LOAD 100 LB. AT STICK GRIP. B - FEEL FORCES. C - PARALLEL SERVO 90 LB. AT UNIT 3.0 IN. STROKE ON 3.0 IN. ARM - 30.0 LB. AT STICK. D - DIFFERENTIAL SERVO 300 LB. OUTPUT ± .375 IN. STROKE.	STIFFNESS OF SECTIONS A, B, C, & D.	SEE FIG.5 FOR NUMBERS OF MEMBERS.	See field of drawing for values of system movement. Assuming 1.0 in movement of stick with zero load in system. Values are taken at mean position of stick.







A - MEMBERS 1 TO 5
B - MEMBERS 5 TO 8
C - MEMBER 9
D - MEMBER 17

MOMENT OF INERTIA OF SECTIONS	EFFECTIVE INERTIA ABOUT STICK PIVOT LB. IN.^{2}
L CONSISTS OF MEMBERS 1 TO 5	1982.8
M CONSISTS OF MEMBERS-6-FEEL & TRIM UNIT.	55.1
N " " " 7+8+1/2 OF 9	331.3
P " " " 1/2 OF 9+10+11 +12+13+14	287.2
Q " " " 15+16+1/2 OF 17	224.0
R " " " 1/2 LF 17 + QUAD.	64.
GROUP 18	94.

FIG. 5A AILERON SYSTEM DATA

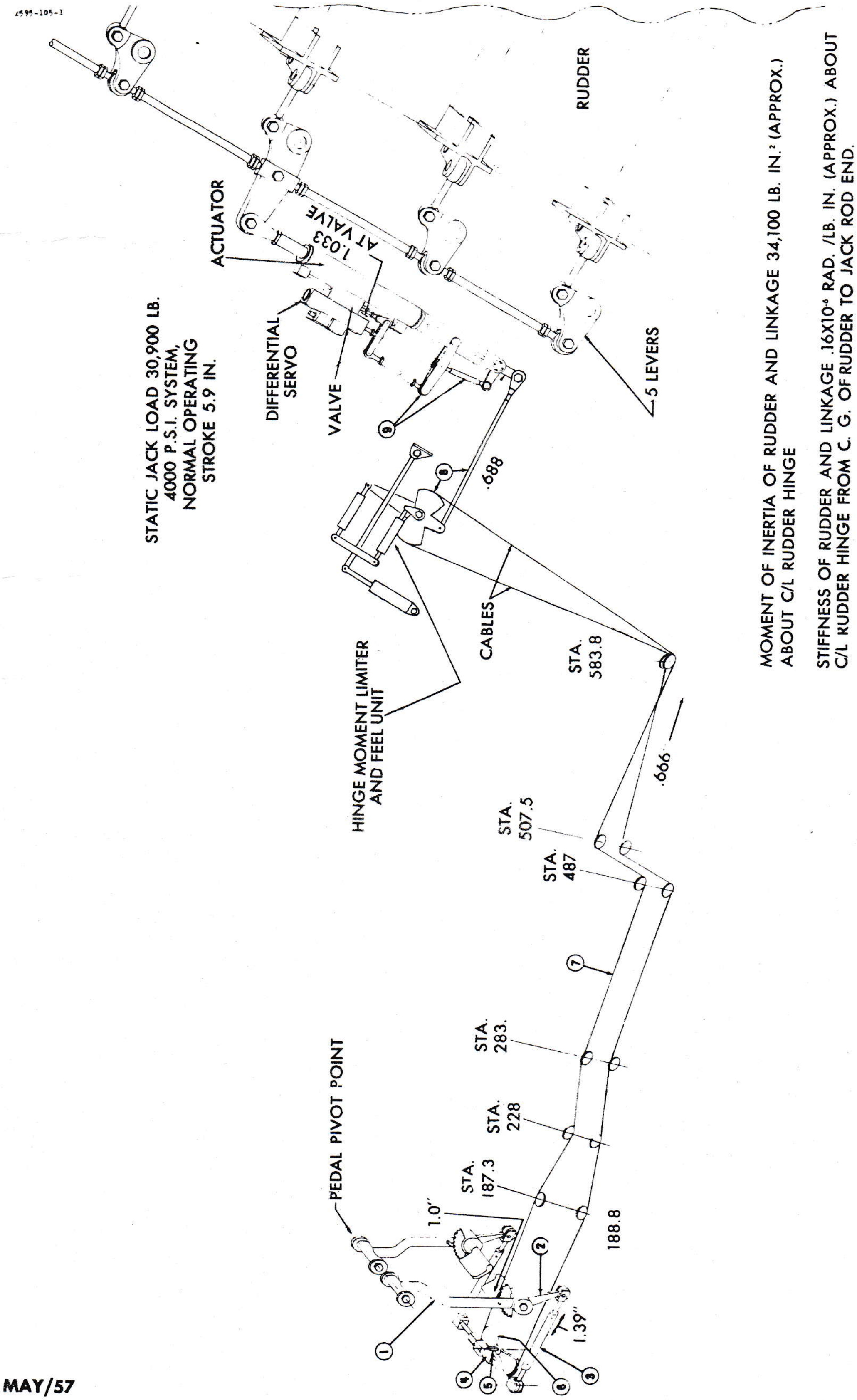


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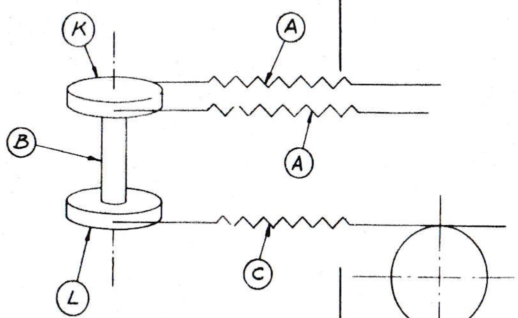
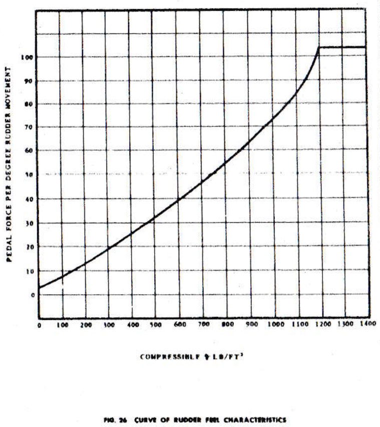
FIG. 6 RUDDER SYSTEM DATA

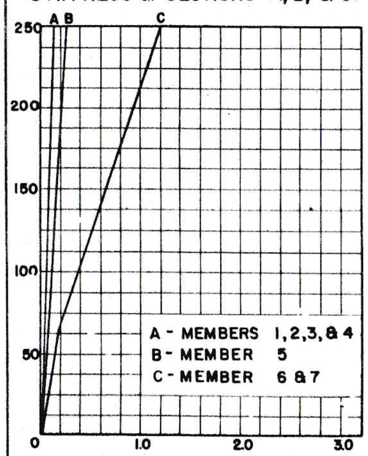
LOADS APPLIED TO SYSTEM	STIFFNESS	MOMENT OF INERTIA	RELATIVE LOADS & DISPLACEMENTS OF MEMBERS OF SYSTEM.
A - PILOT'S LIMIT LOAD 300 LBS. AT PEDAL. B - DIFFERENTIAL SERVO - OUTPUT FORCE 300 TO 400 LBS. STROKE ± 0.5 IN.		SEE FIG. 6 FOR NUMBERS OF MEMBERS.	See field of drawing for values of system movement. Assuming 1.0 in movement of pedal with zero load in system. Values are taken at mean position of pedal.



RUDDER - FEEL CHARACTERISTICS



STIFFNESS OF SECTIONS A, B, & C.



MOMENT OF INERTIA OF SECTIONS	EFFECTIVE INERTIA ABOUT PEDAL PIVOT POINT LB. IN.²
(K) CONSISTS OF MEMBERS: 2×(1)+2×(2)+2×(3)+2×(4) + ½ OF (5)	3356.1
(L) CONSISTS OF MEMBERS: ½ OF (5) + (6) + ½ OF (7)	252.8
(M) CONSISTS OF MEMBERS: ½ OF (7) + (8) + (9)	370.

FIG. 6A RUDDER SYSTEM DATA

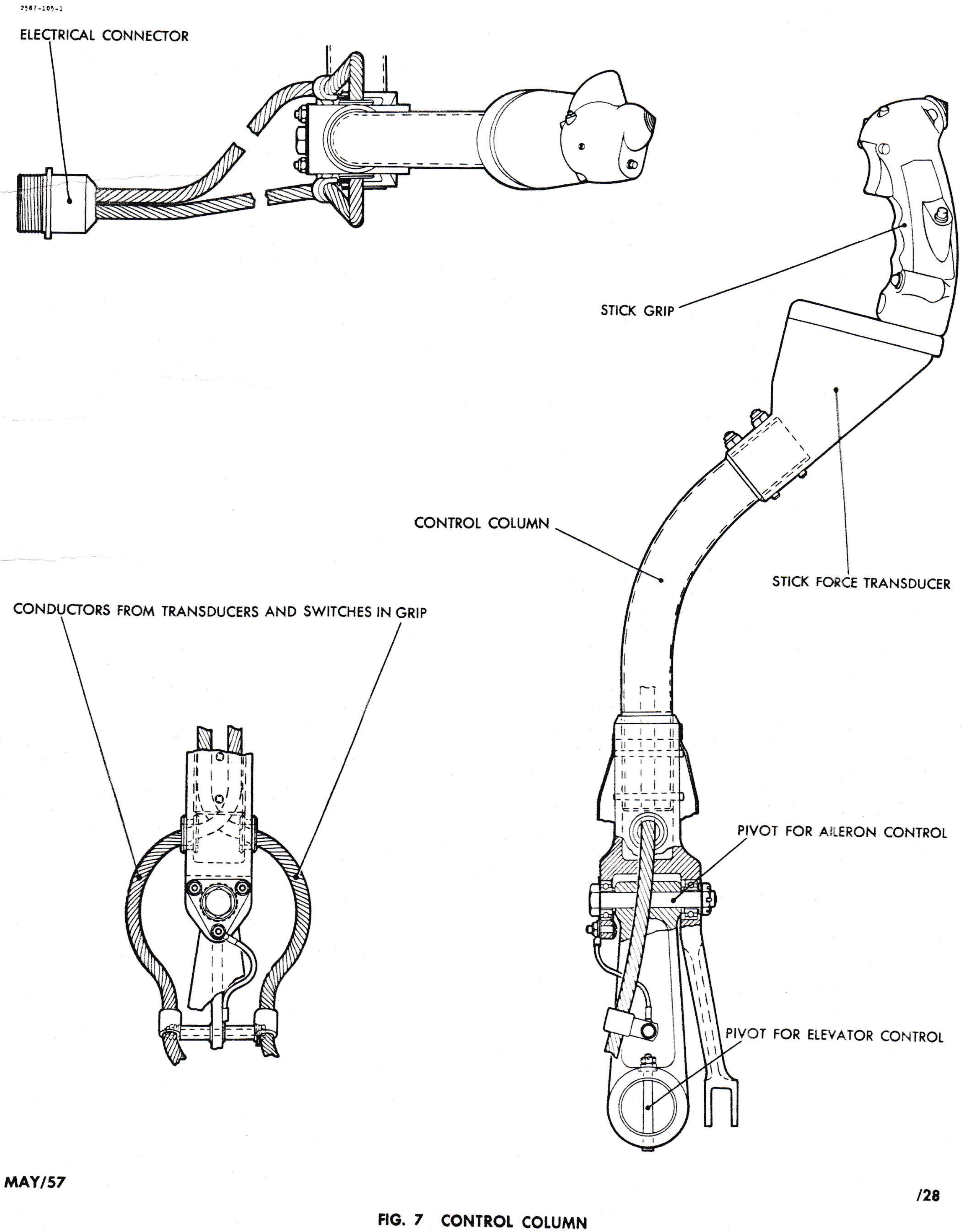


FIG. 7 CONTROL COLUMN

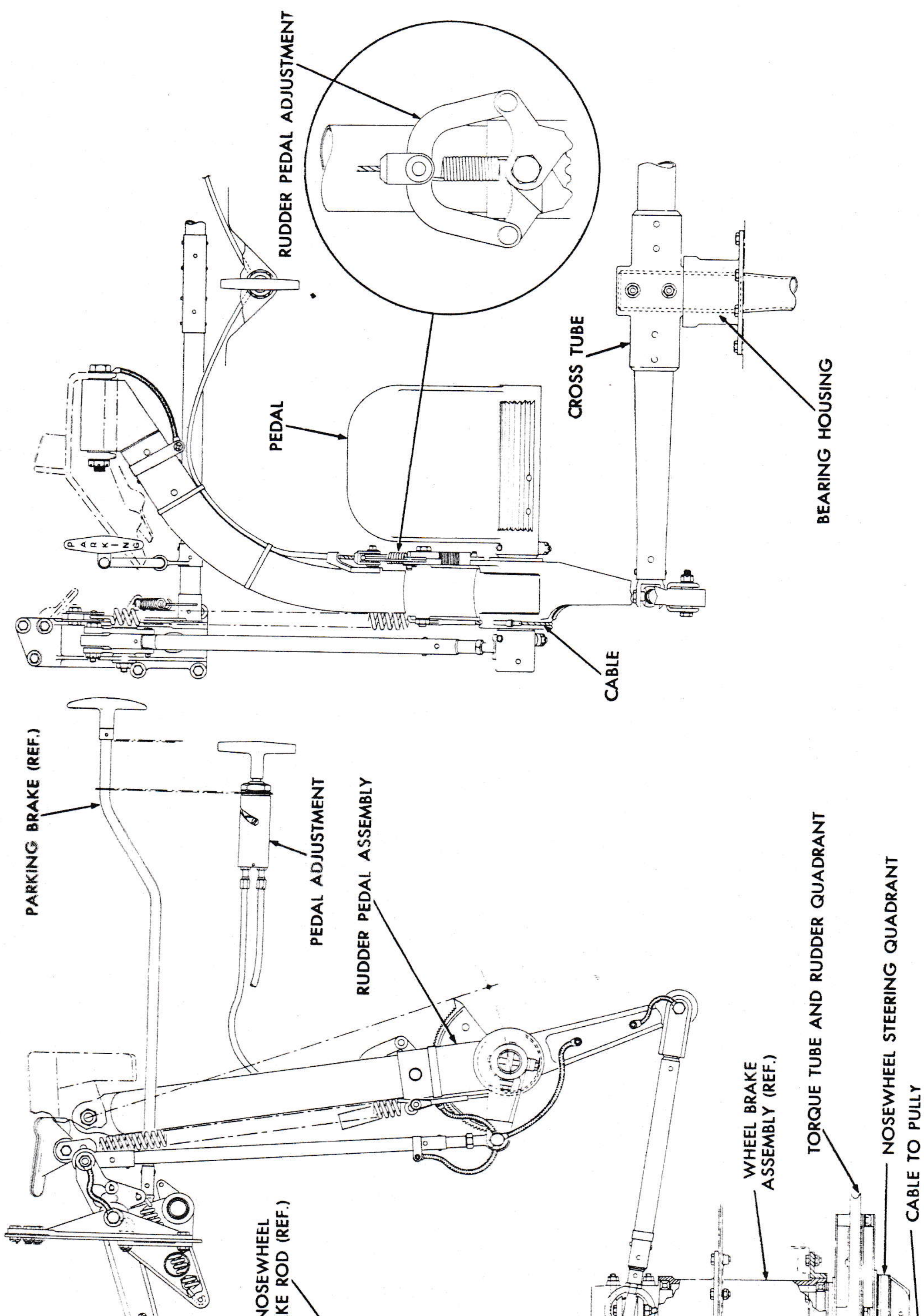


FIG. 8 RUDDER PEDALS

2546-105-1

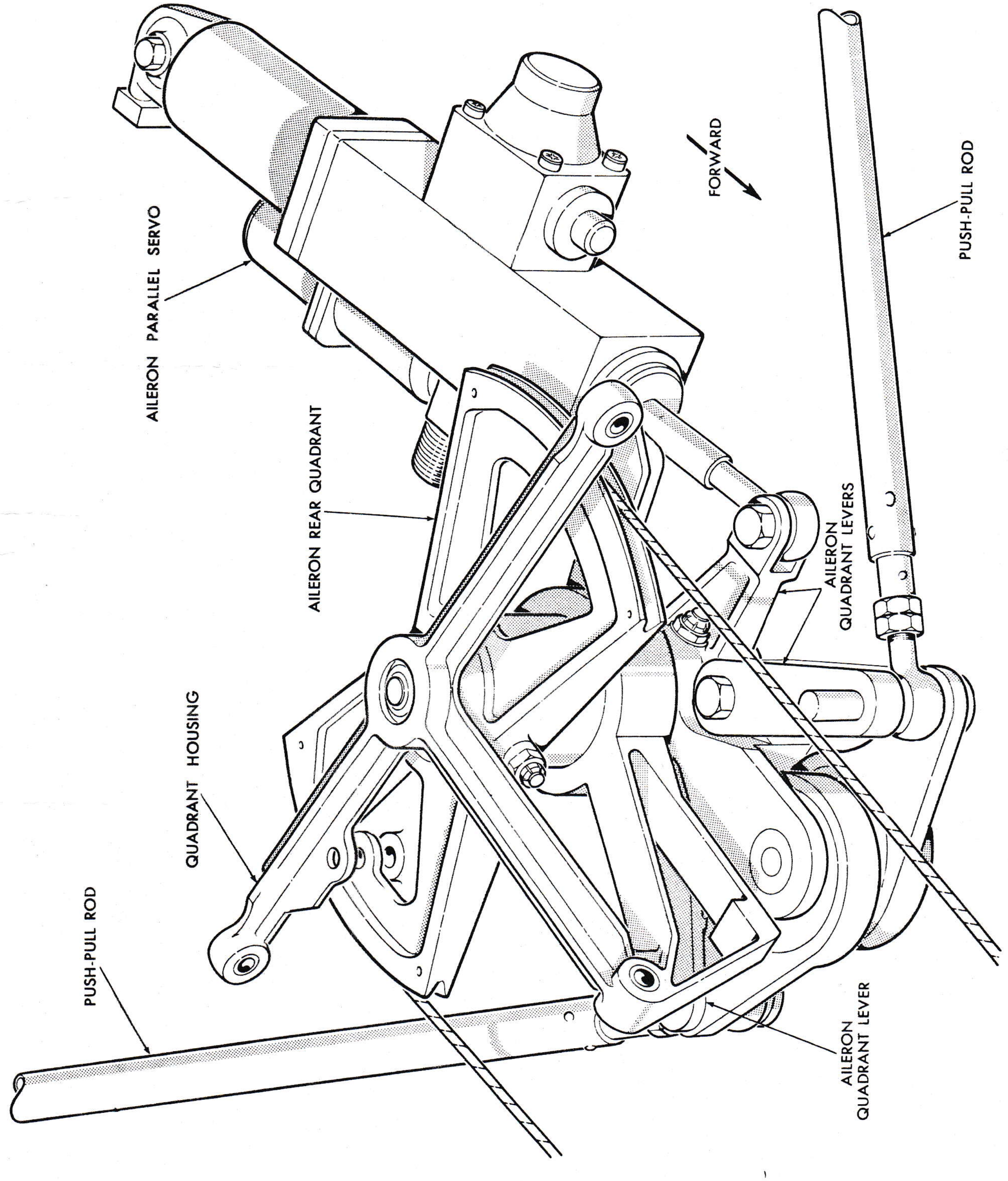


FIG. 10-1 AILERON - PARALLEL SERVO INSTALLATION

2547-105-1

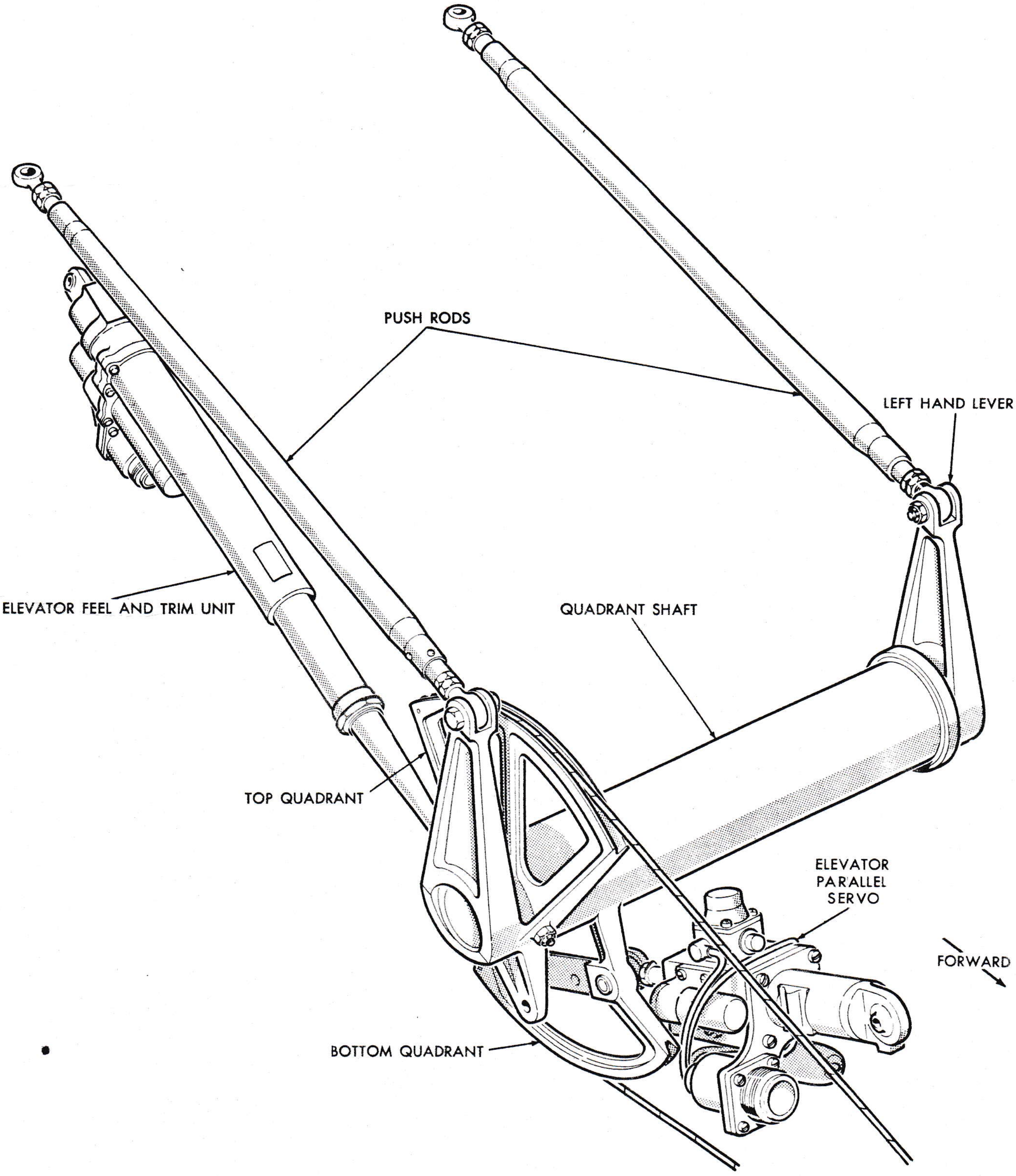


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FIG. 10-2 ELEVATOR - PARALLEL SERVO INSTALLATION

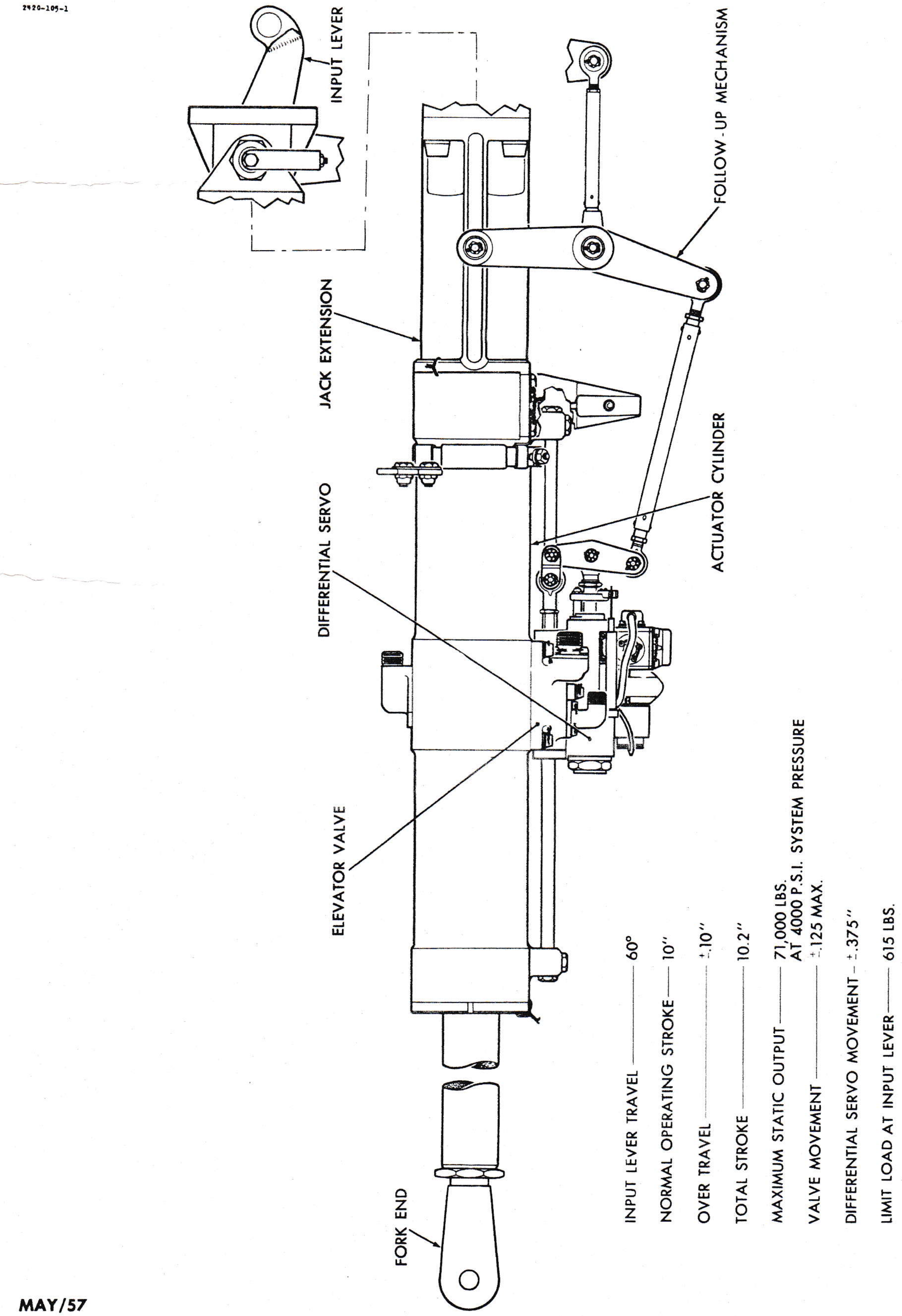


FIG. 11 ASSEMBLY OF POWER CONTROL ACTUATOR - ELEVATOR VALVE AND DIFFERENTIAL SERVO

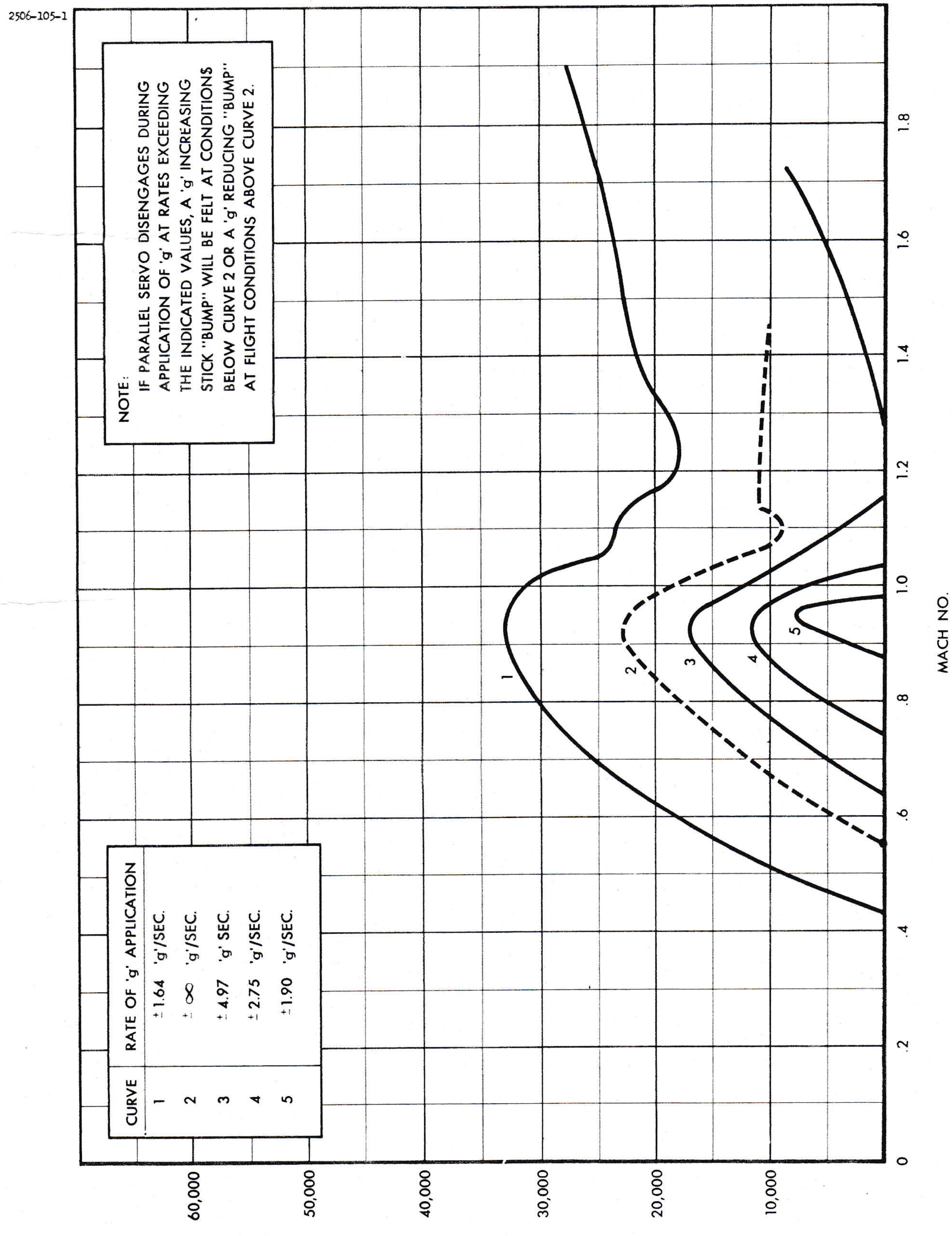


FIG. 12 ELEVATOR AUTO-TRIM CURVE

2421-105-1

INPUT QUADRANT TRAVEL ______ 46.75°
NORMAL OPERATING STROKE ______ 5.56"
OVER TRAVEL ______ ±.10"
TOTAL-STROKE ______ 5.76"
CABLE TRAVEL ______ 3.26"
MAXIMUM STATIC OUTPUT ______ 42,000 LBS.
AT 4000 P.S.I. SYSTEM PRESSURE

VALVE MOVEMENT ______ ±.07"
DIFFERENTIAL SERVO MOVEMENT ___ ±.375"
LIMIT LOAD AT INPUT QUADRANT _500 LBS. (ON EITHER CABLE)

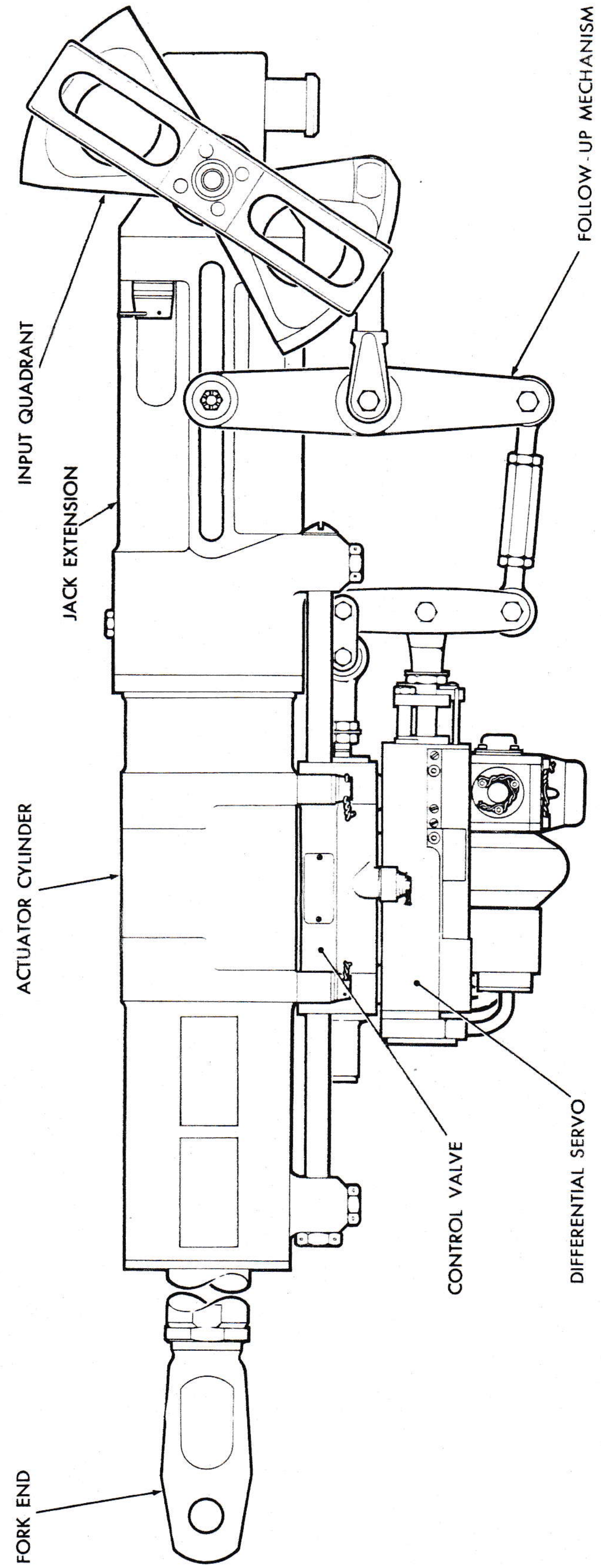
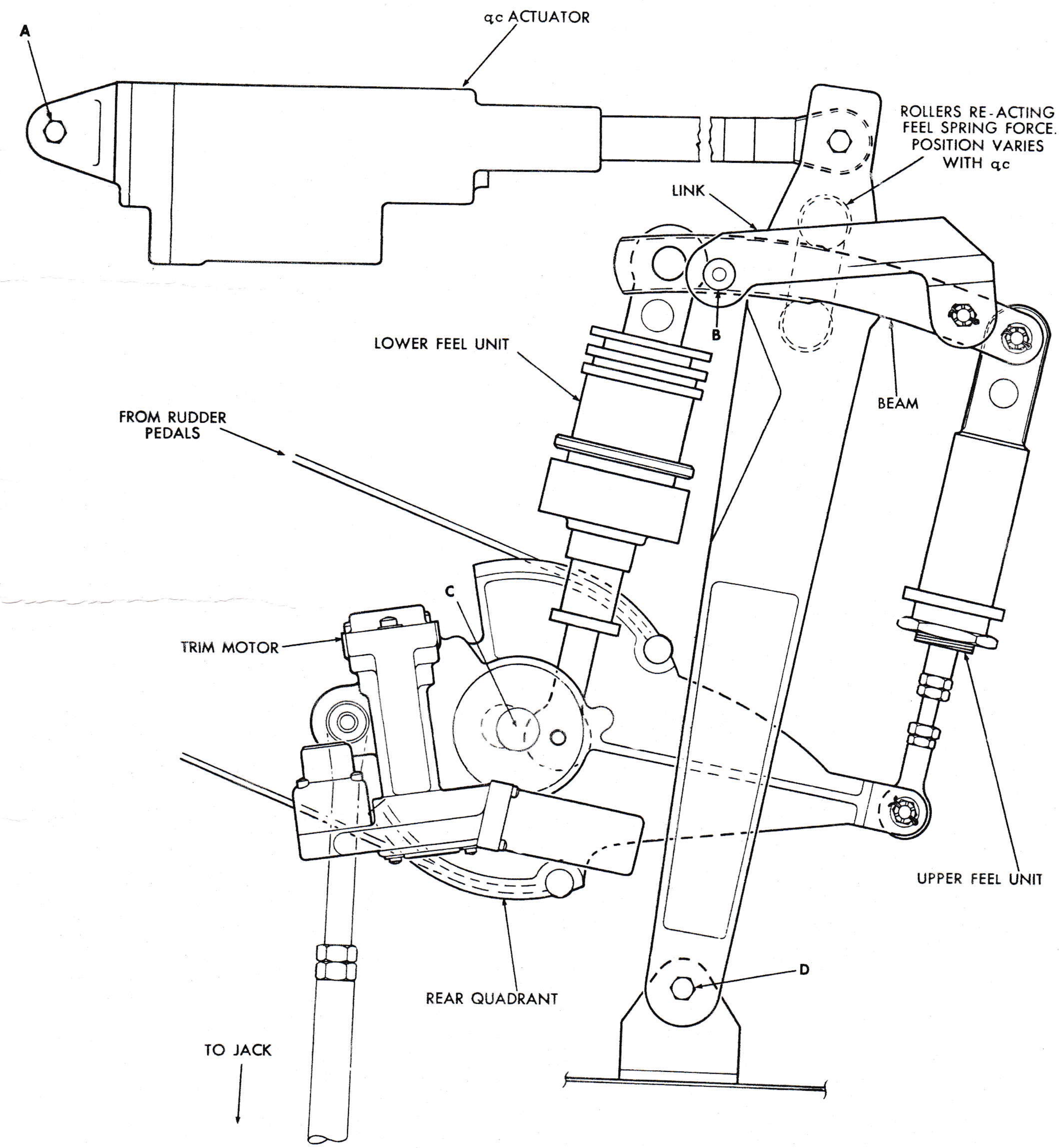


FIG. 13 ASSEMBLY OF POWER CONTROL ACTUATOR - AILERON VALVE AND DIFFERENTIAL SERVO

2513-105-1



POINTS A, B, C & D ARE ATTACHED TO FIN STRUCTURE

FIG. 14 RUDDER FEEL SYSTEM & QUADRANT ASSEMBLY-FIN

2425-105-1

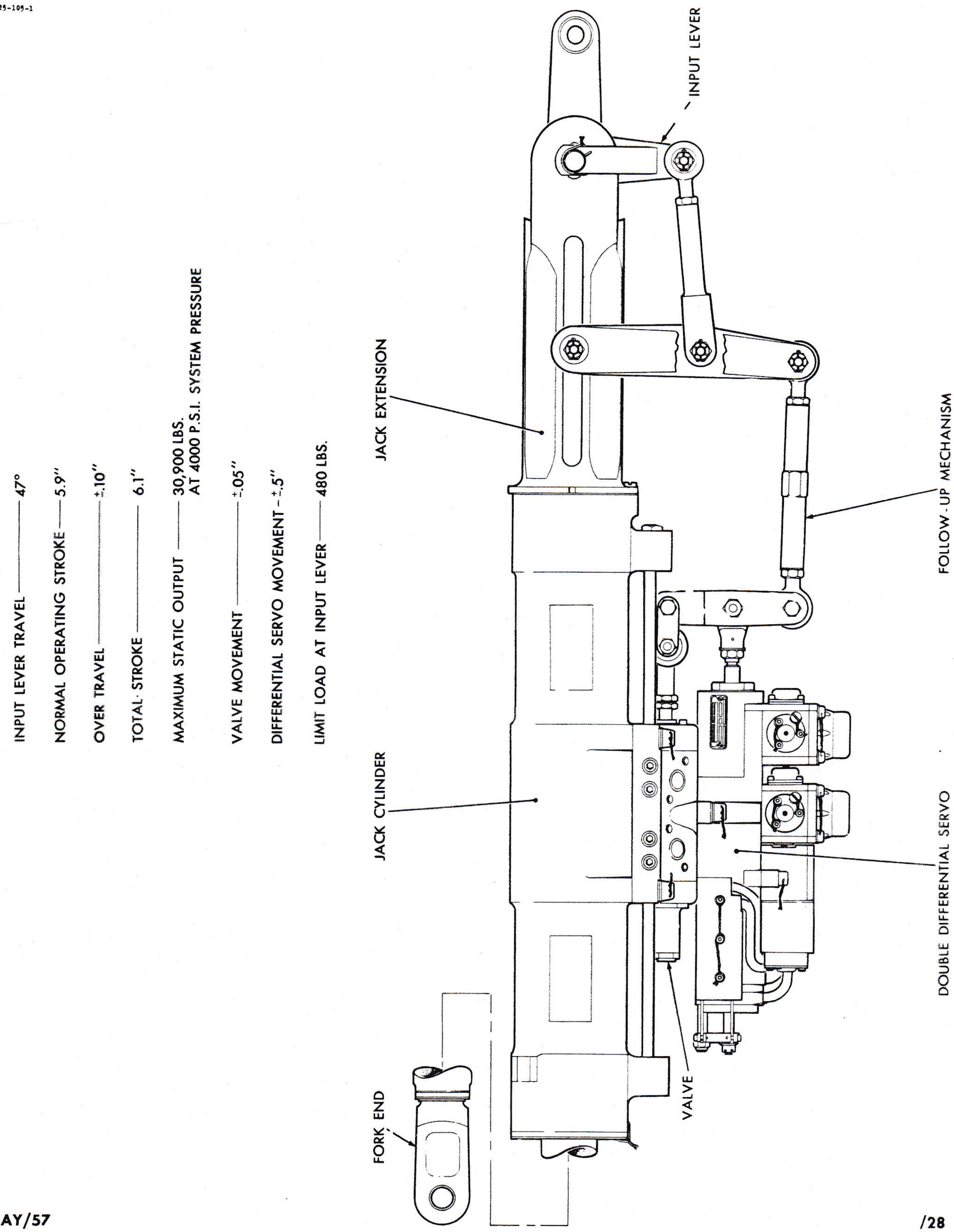


FIG. 15 ASSEMBLY OF POWER CONTROL ACTUATOR - RUDDER VALVE AND DIFFERENTIAL SERVO

2423-105-1

INPUT LEVER TRAVEL — 60°

NORMAL OPERATING STROKE — 10"

OVER TRAVEL — ±.10"

TOTAL STROKE — 10.2"

MAXIMUM STATIC OUTPUT — 71,000 LBS. AT 4000 P.S.I. SYSTEM PRESSURE

LIMIT LOAD AT INPUT LEVER — 615 LBS.

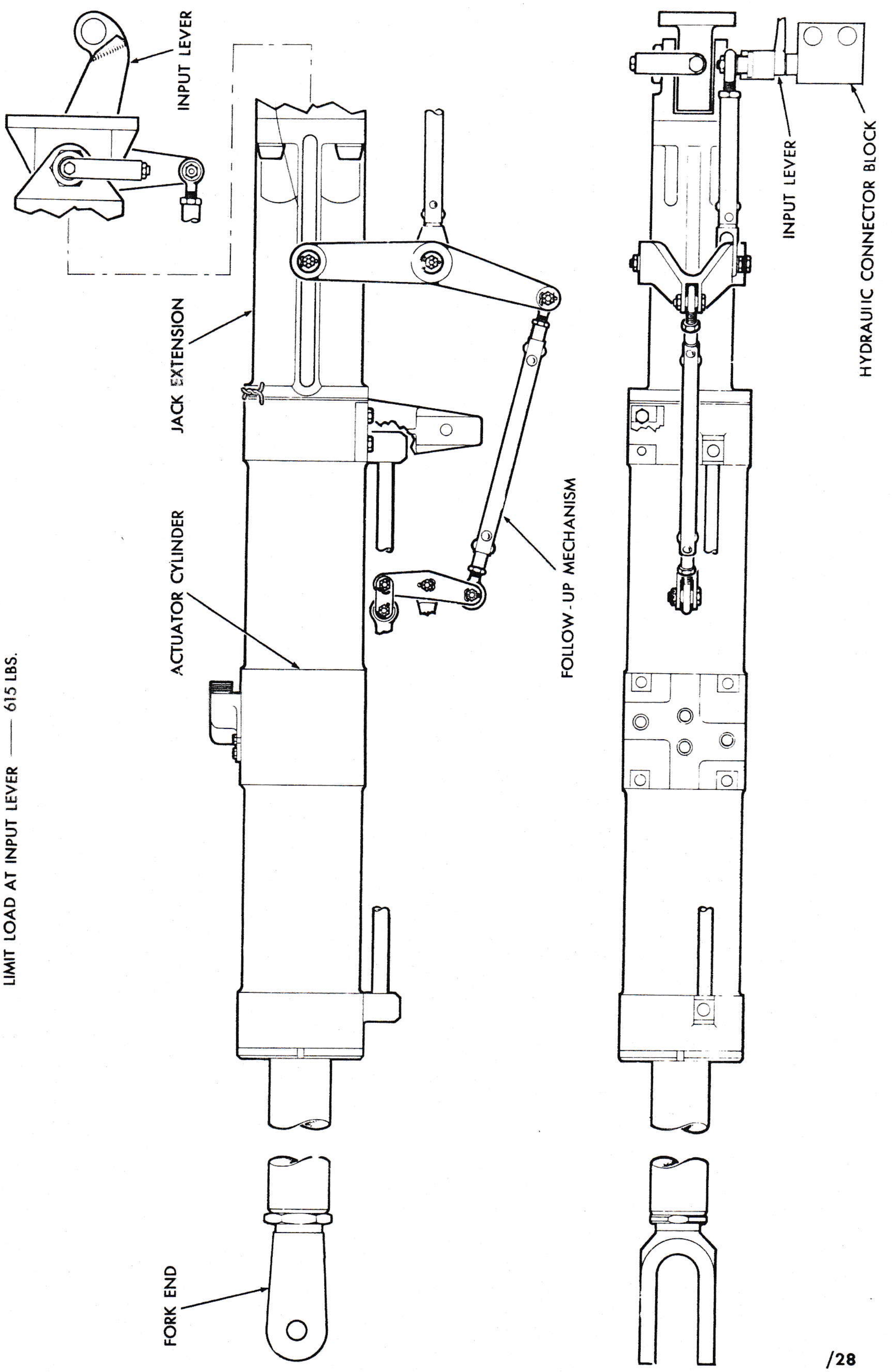


FIG. 16 POWER CONTROL ACTUATOR - ELEVATOR

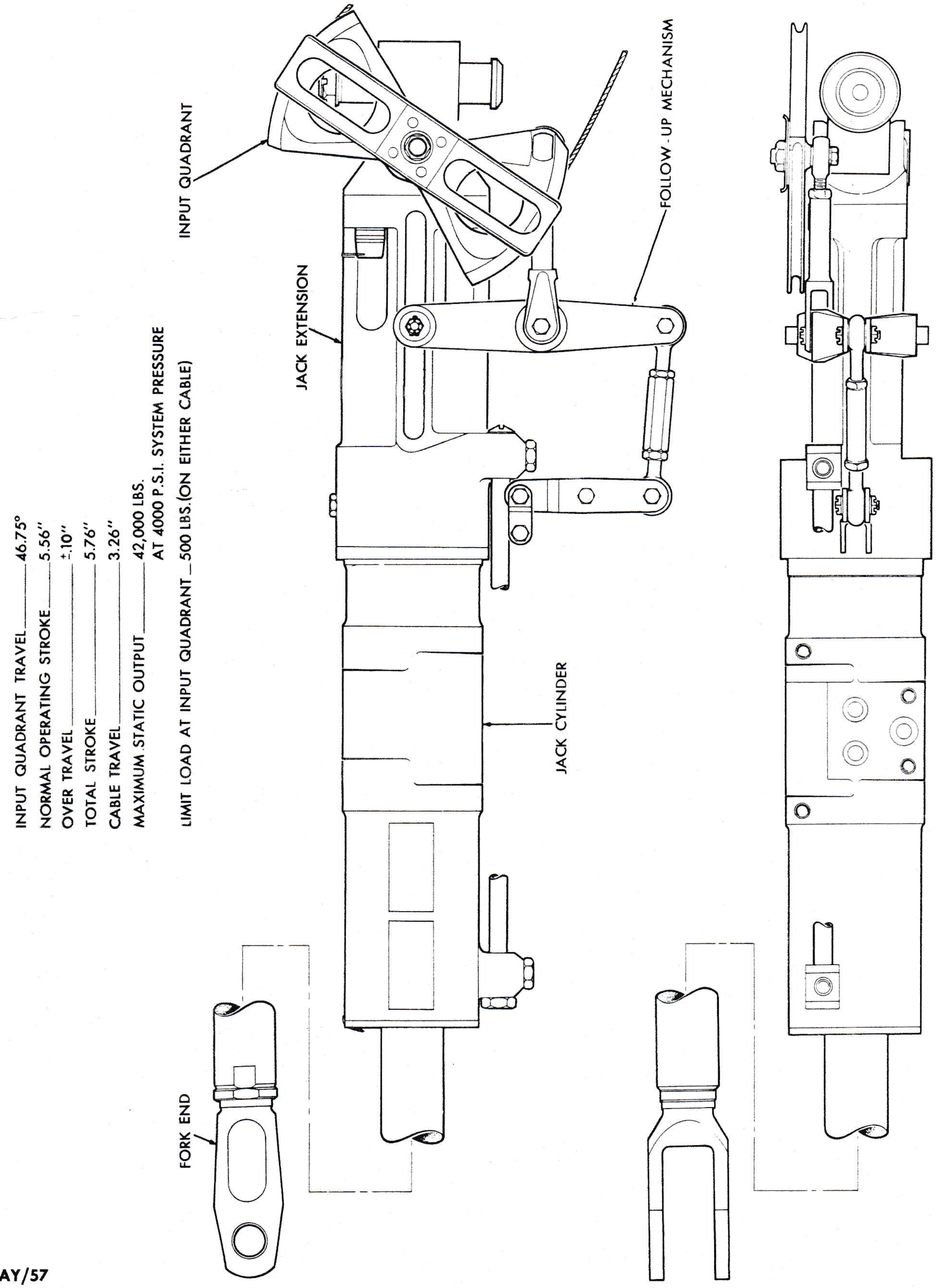


FIG. 17 POWER CONTROL ACTUATOR - AILERON

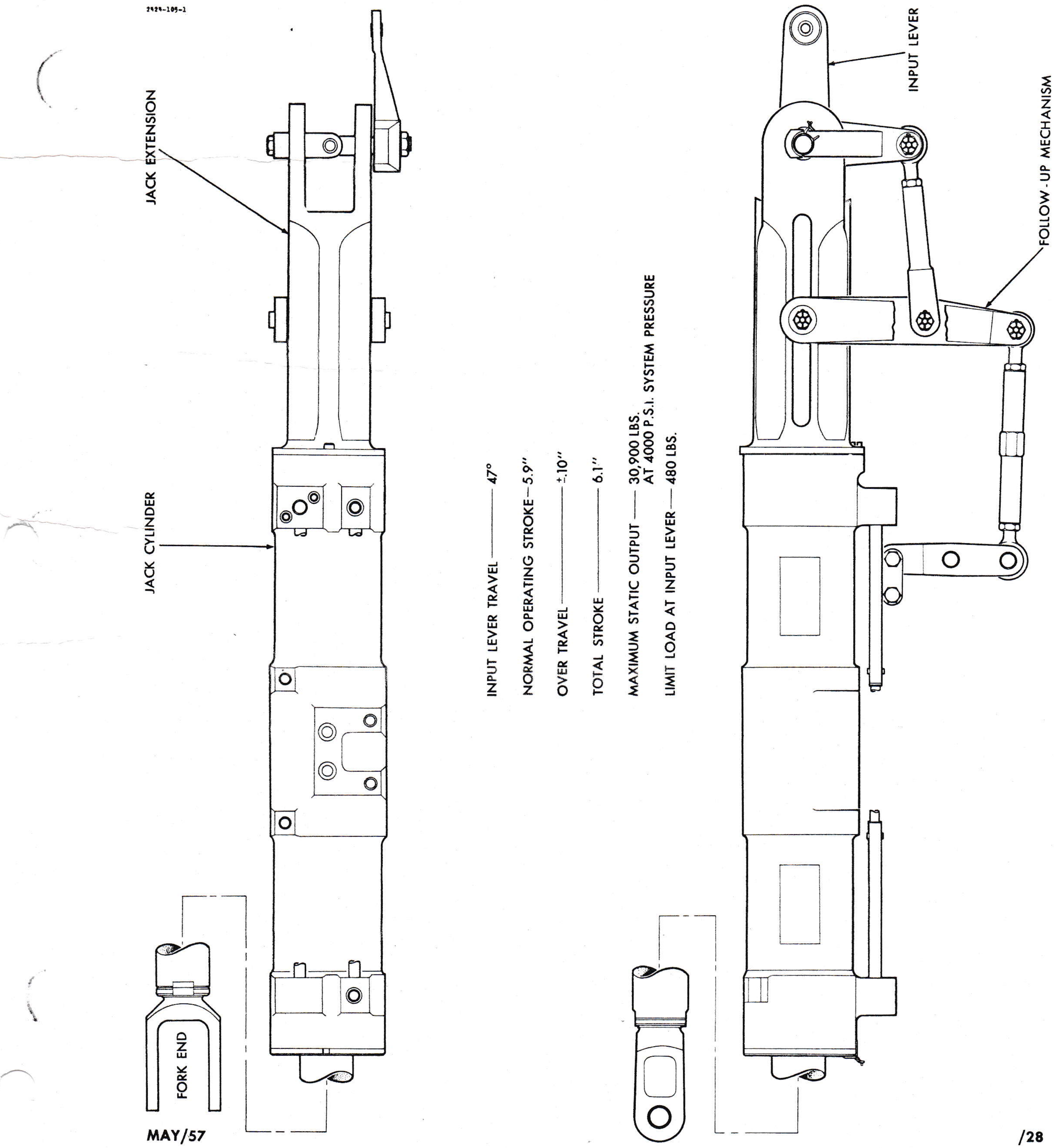


FIG. 18 POWER CONTROL ACTUATOR - RUDDER

250"-105-1

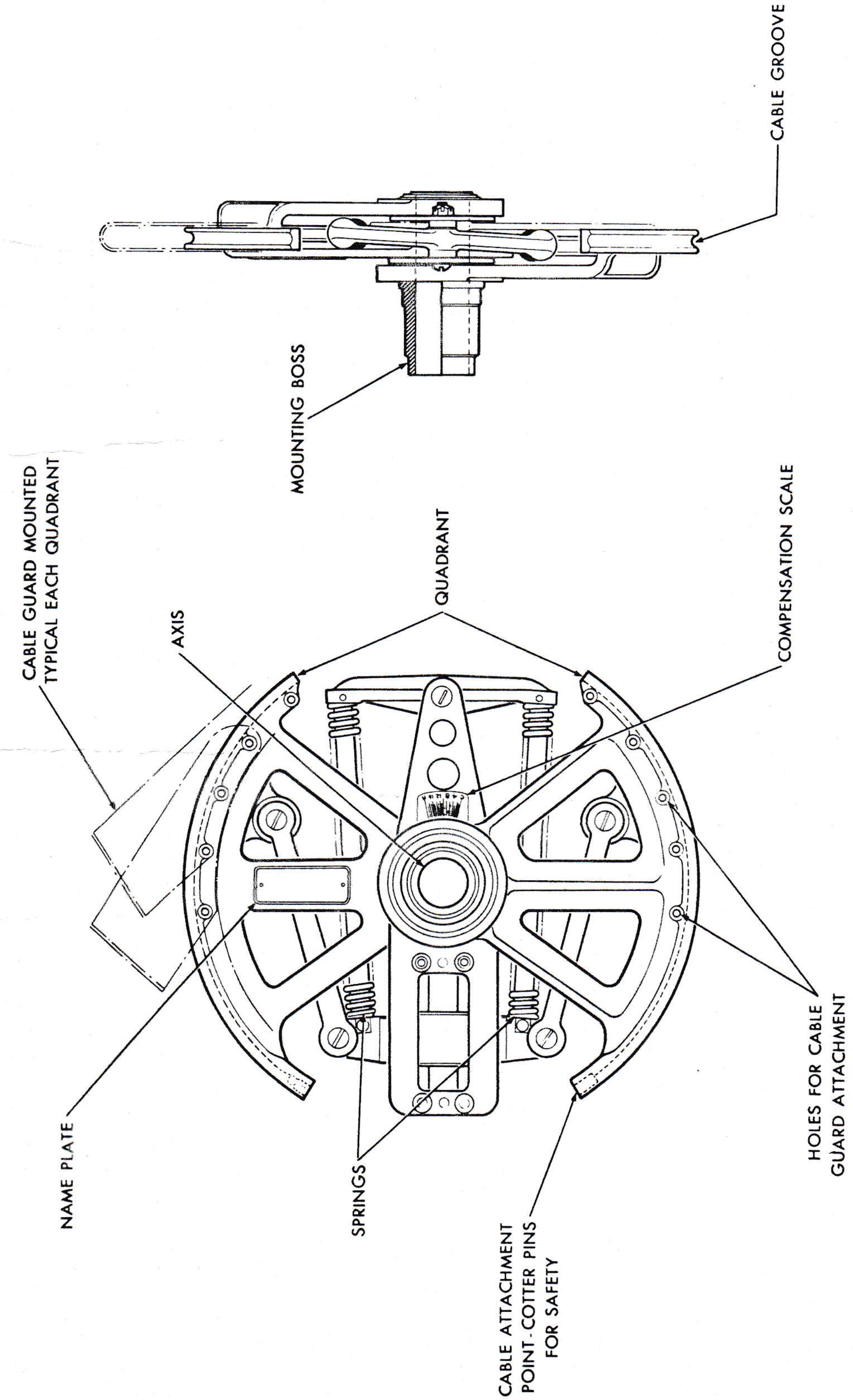


FIG. 19 TENSION REGULATOR QUADRANT-FRONT FUSELAGE

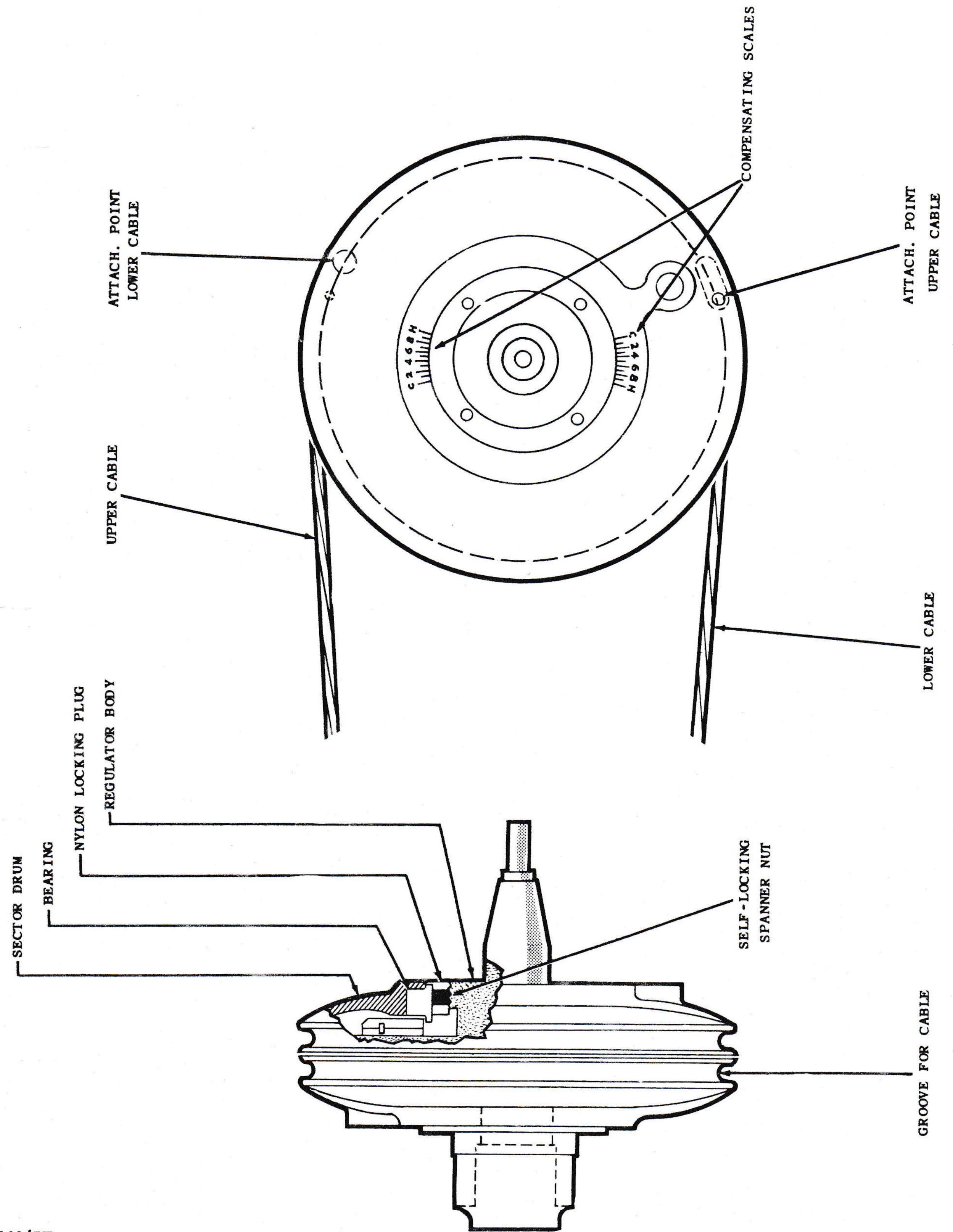
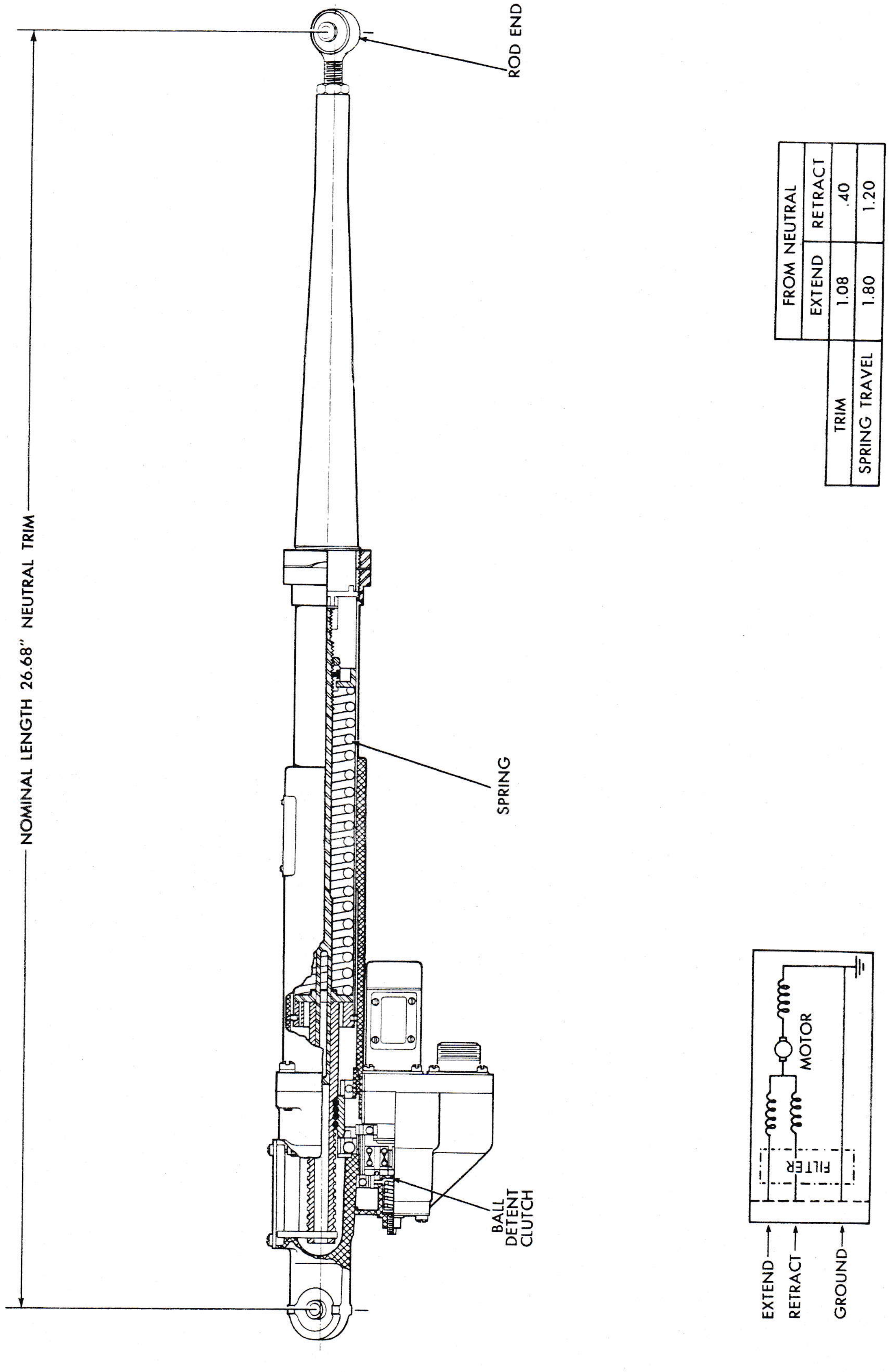


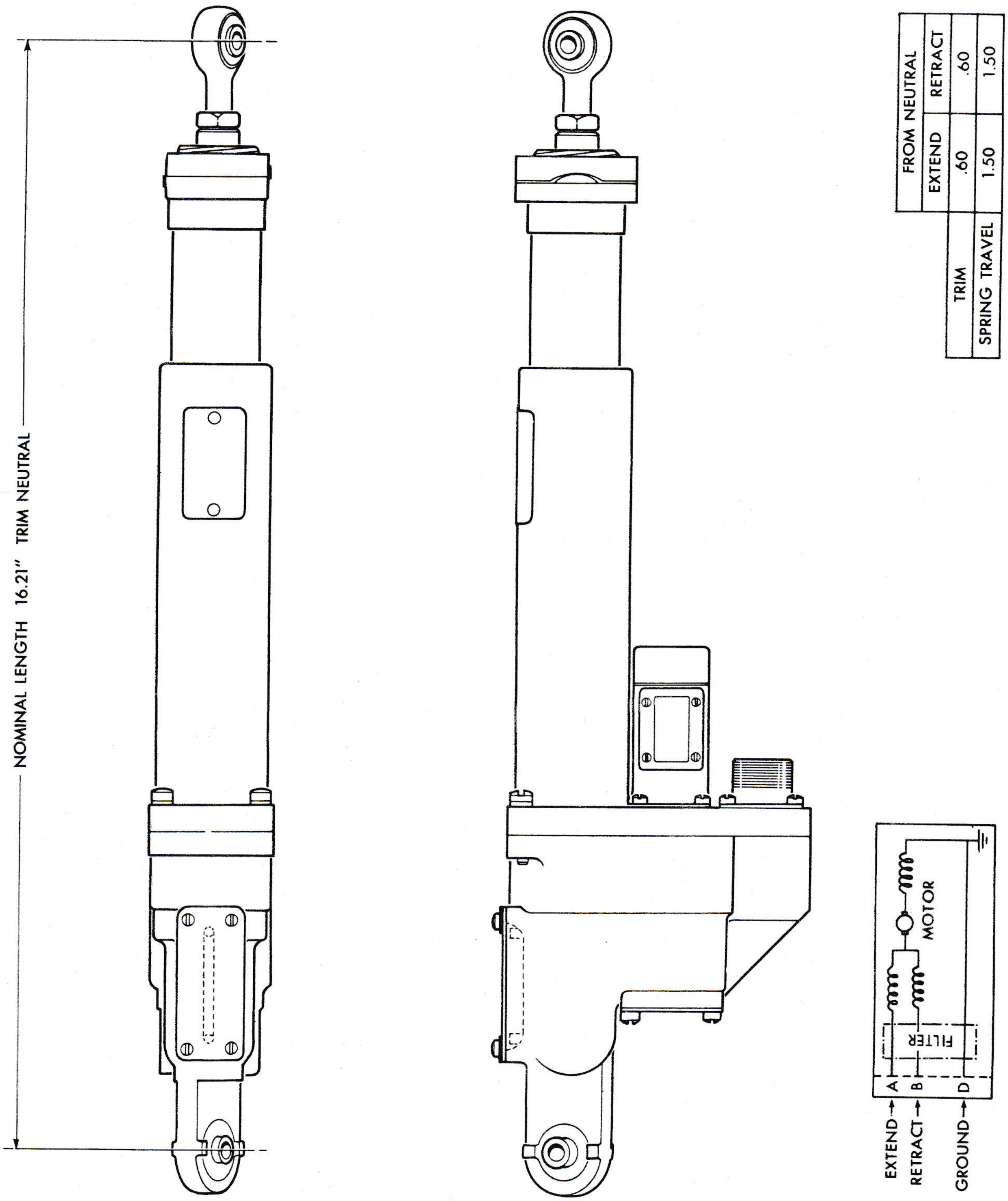
FIG. 20 TENSION REGULATOR QUADRANT - INNER WING



	FROM NEUTRAL	
	EXTEND	RETRACT
TRIM	1.08	.40
SPRING TRAVEL	1.80	1.20

FIG. 21 FEEL AND TRIM UNIT - ELEVATOR

2472-105-1

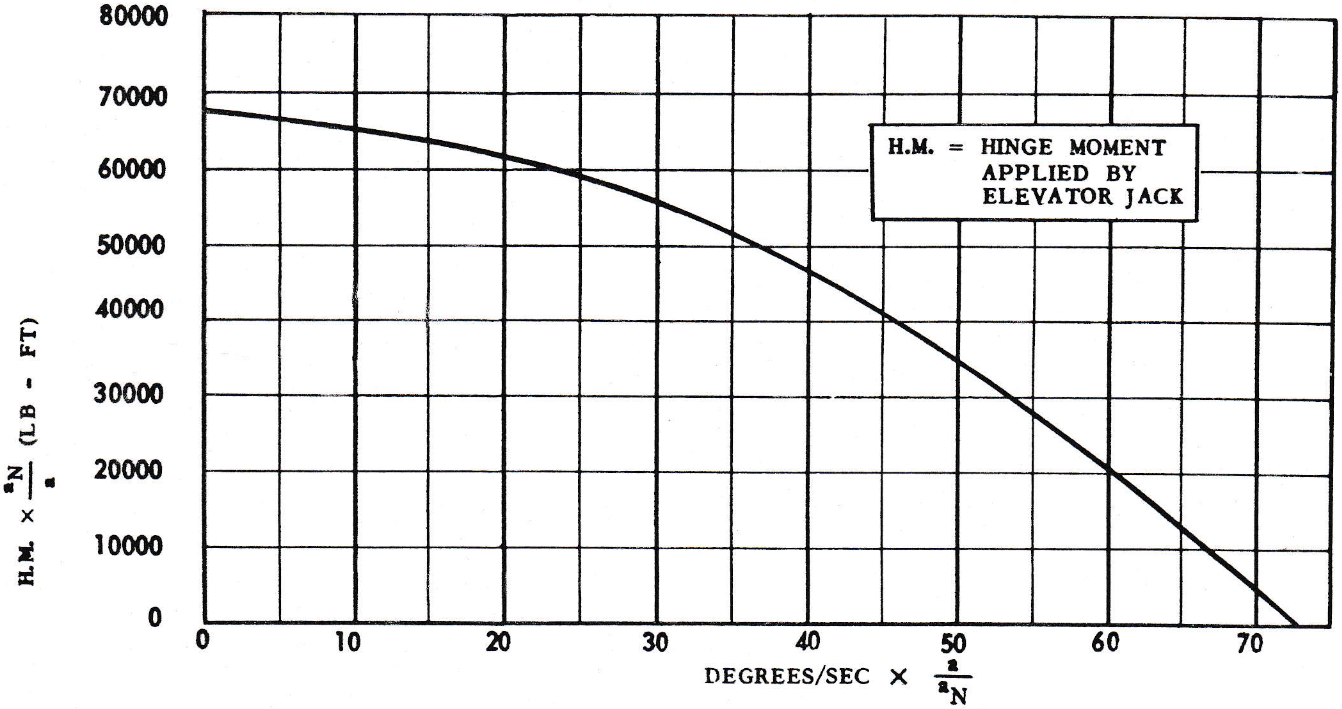


	FROM NEUTRAL	
	EXTEND	RETRACT
TRIM	.60	.60
SPRING TRAVEL	1.50	1.50

MAY/57

/28

FIG. 22 FEEL AND TRIM UNIT - AILERON



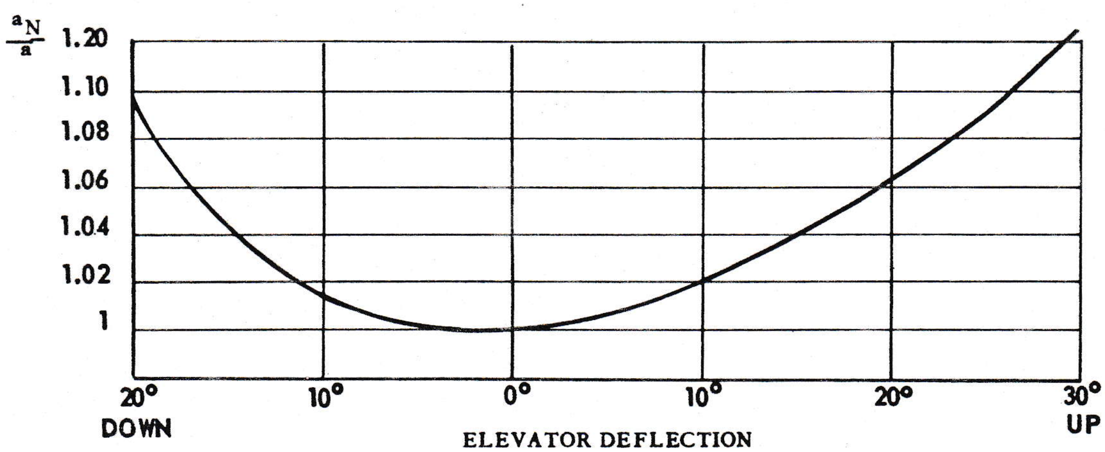
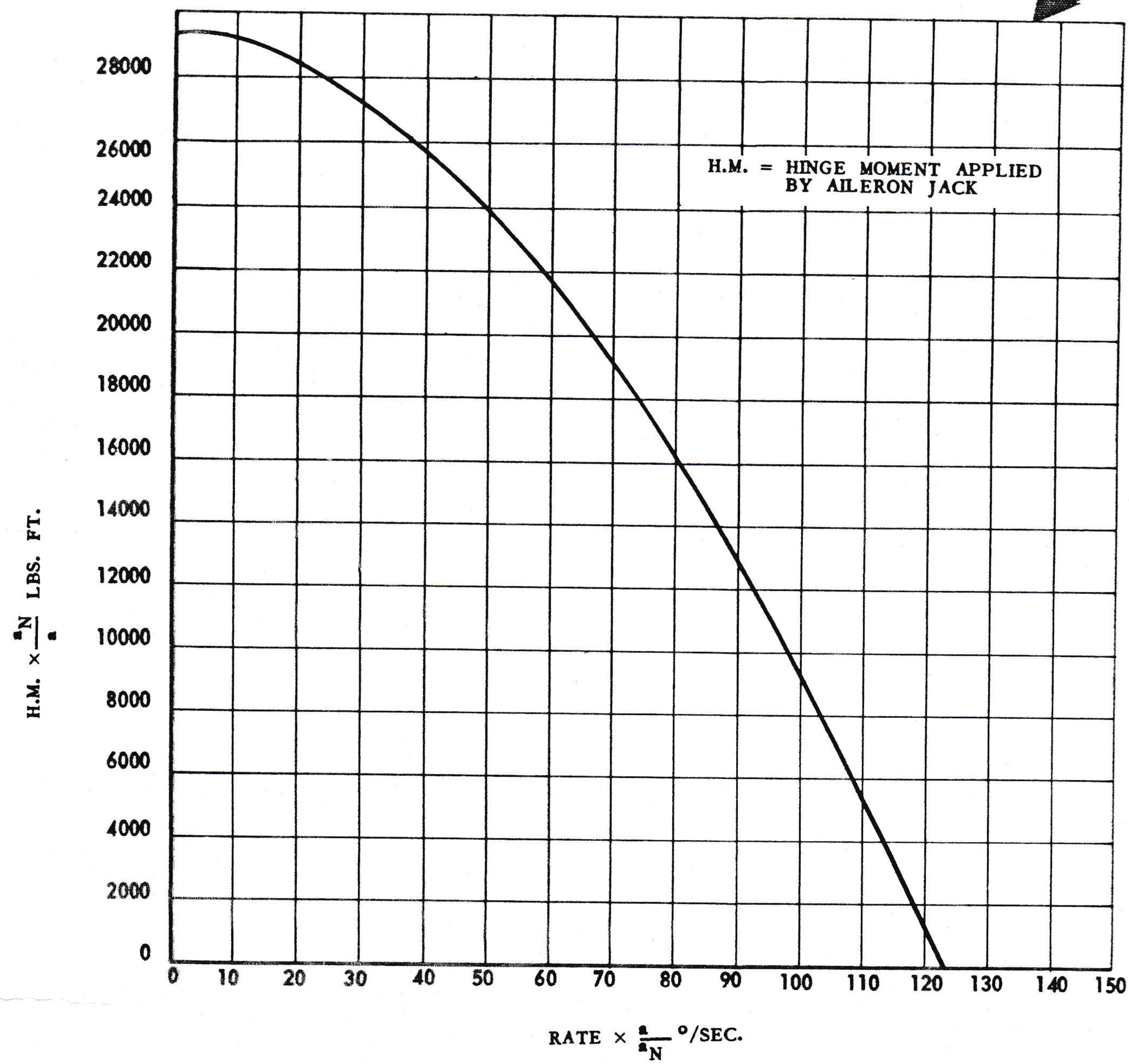


FIG. 23 RATE VS HINGE MOMENT ELEVATOR



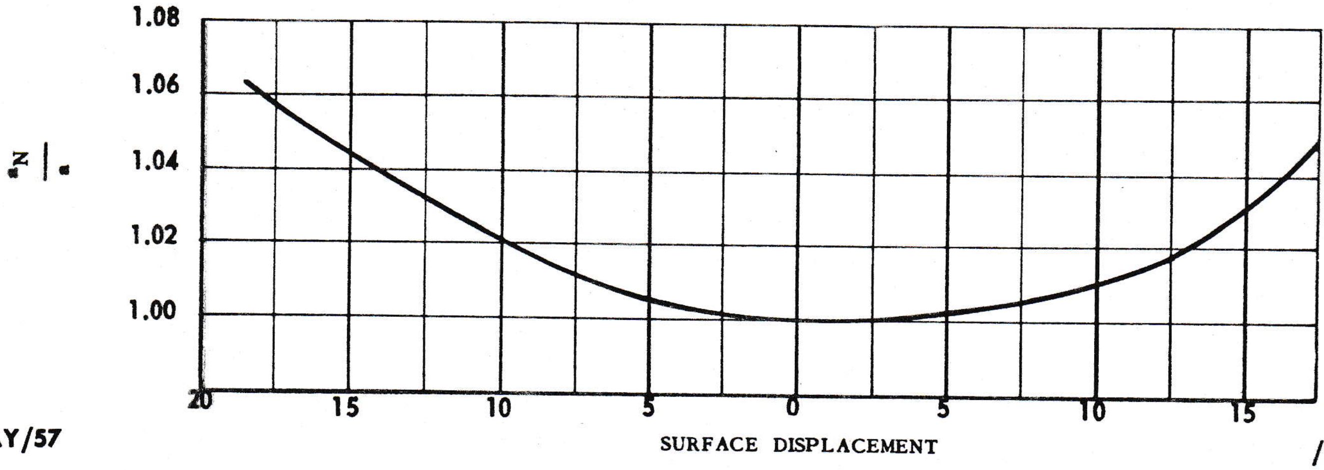


FIG. 24 RATE VS HINGE MOMENT AILERON

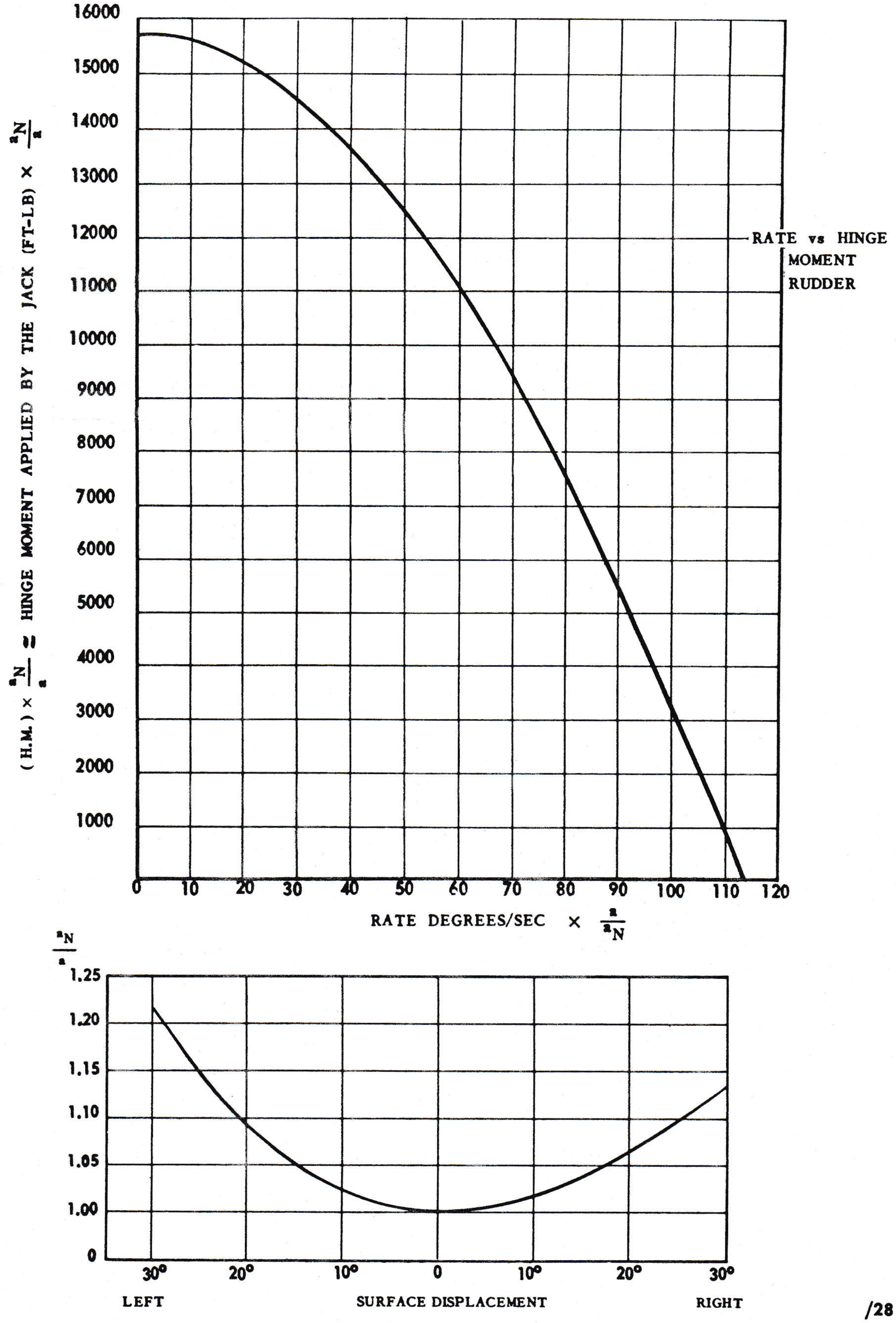


FIG. 25 RATE VS HINGE MOMENT RUDDER

RUDDER FEEL CHARACTERISTICS

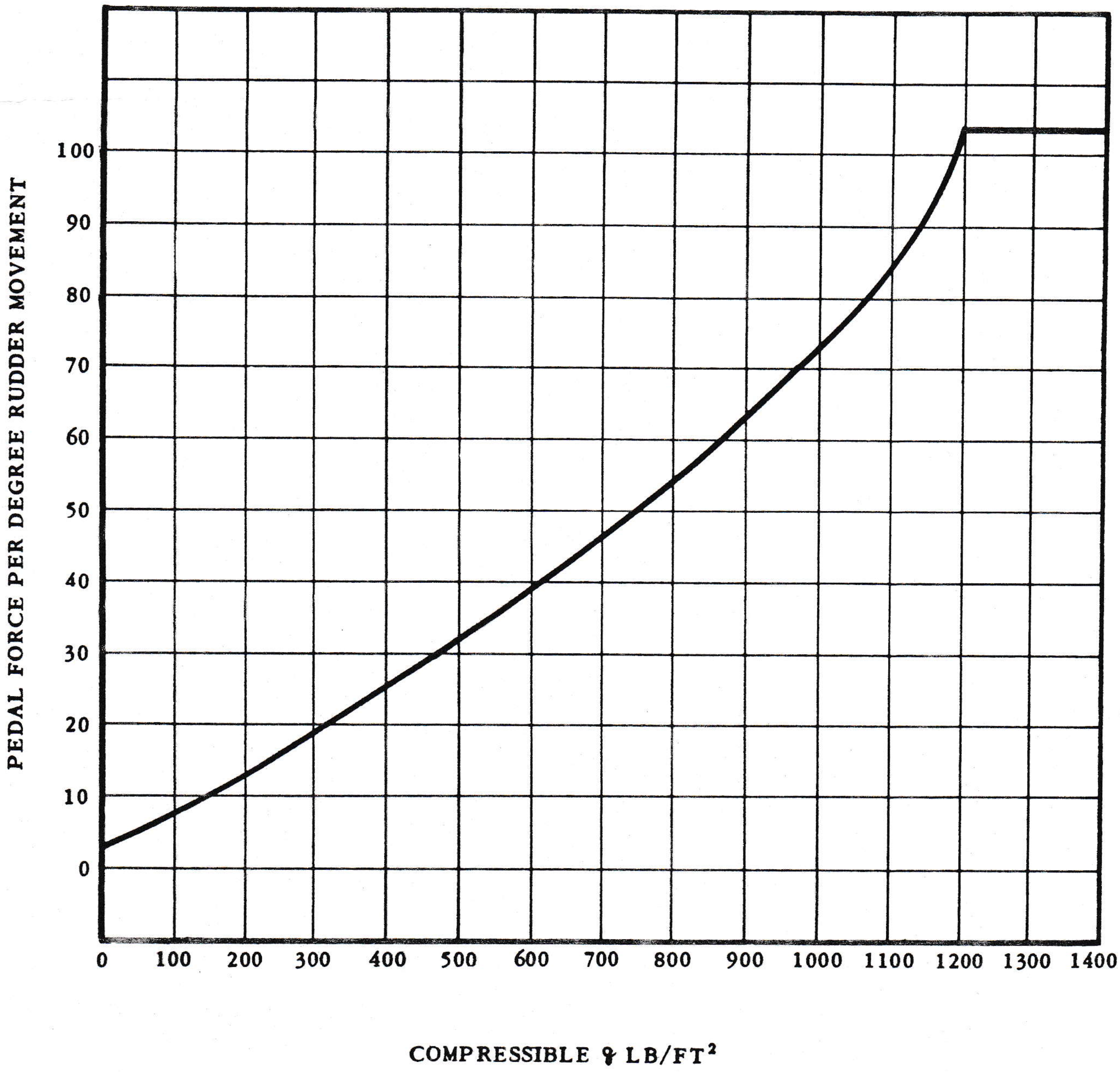


FIG. 26 CURVE OF RUDDER FEEL CHARACTERISTICS

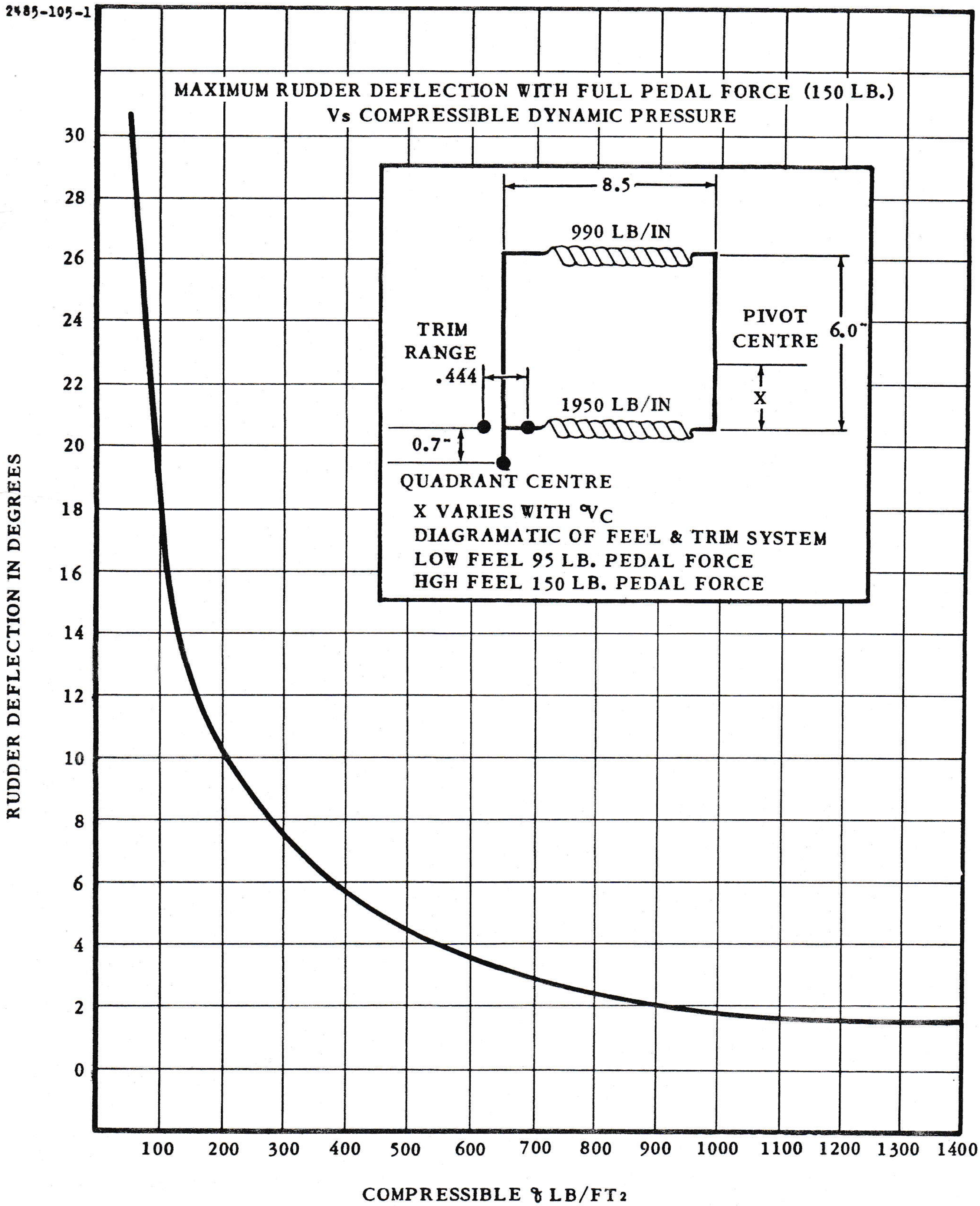
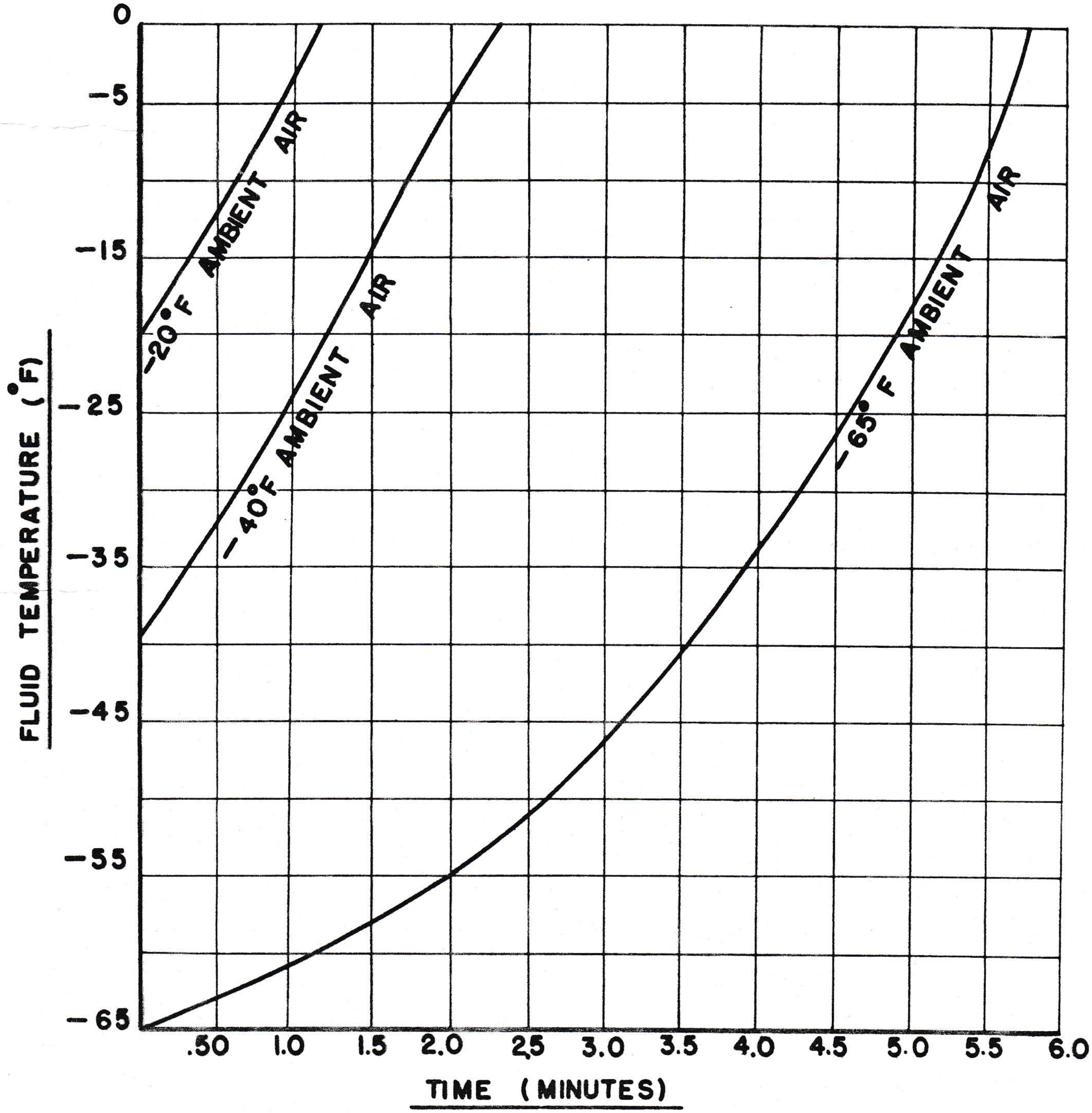


FIG. 27 CURVE OF RUDDER DEFLECTION VS COMPRESSIBLE DYNAMIC PRESSURE



TIME REQUIRED TO WARM UP HYDRAULIC FLUID

FIG. 28 FLYING CONTROLS - GRAPH OF TIME REQUIRED TO WARM UP HYDRAULIC FLUID VS HYDRAULIC FLUID TEMPERATURE

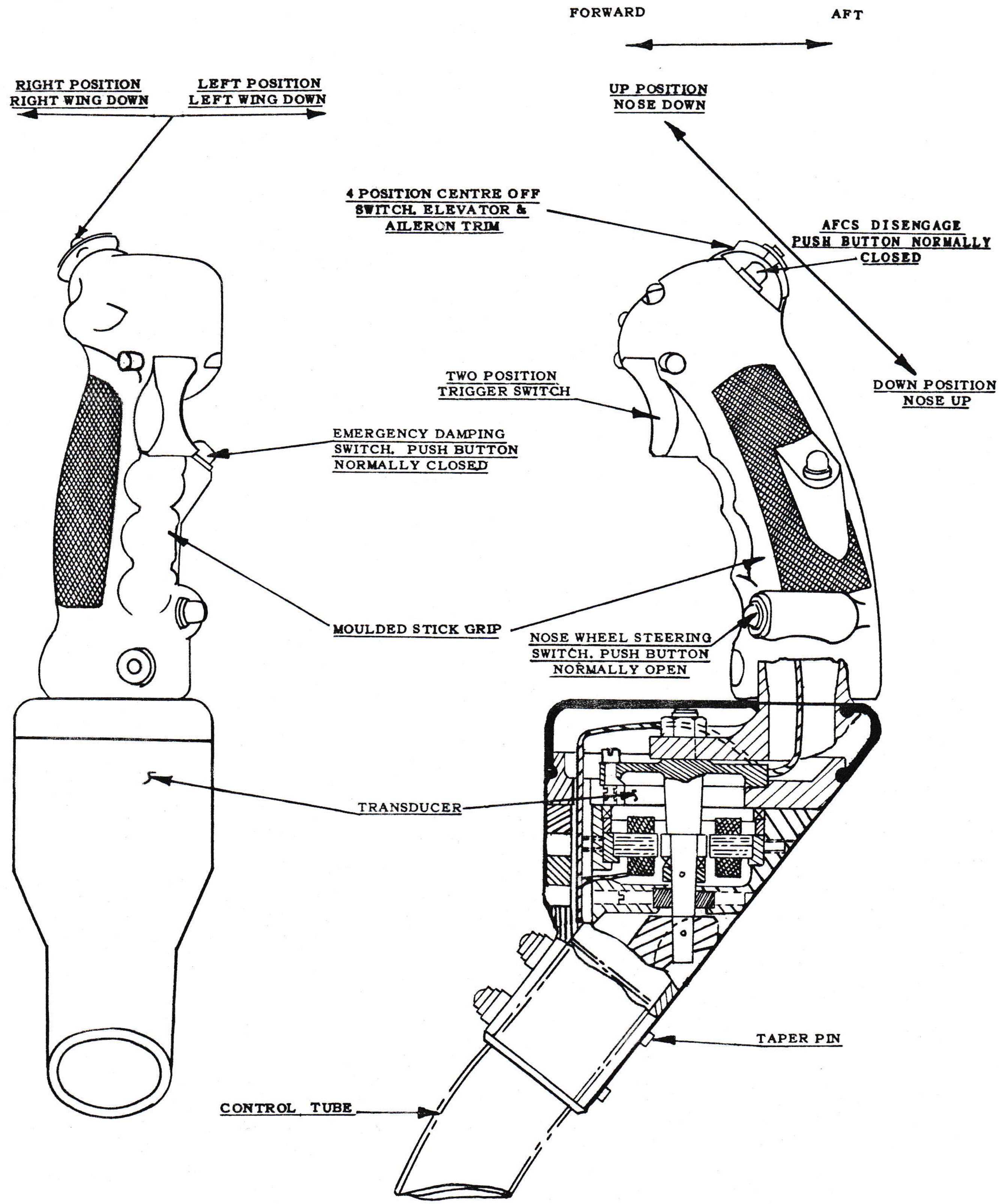


FIG. 29 STICK FORCE TRANSDUCER FLYING CONTROLS

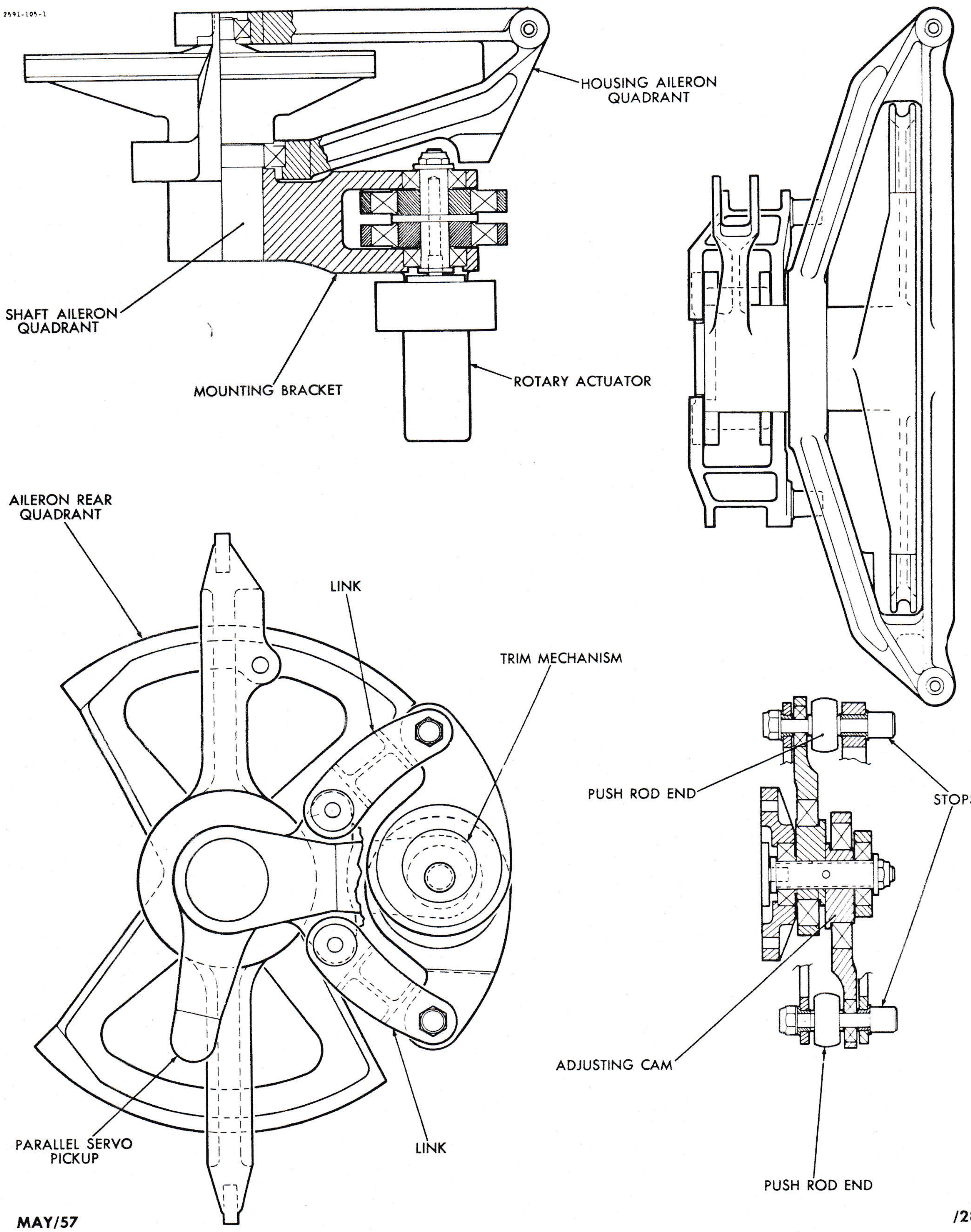


FIG. 30 AILERON PITCH-TRIM MECHANISM

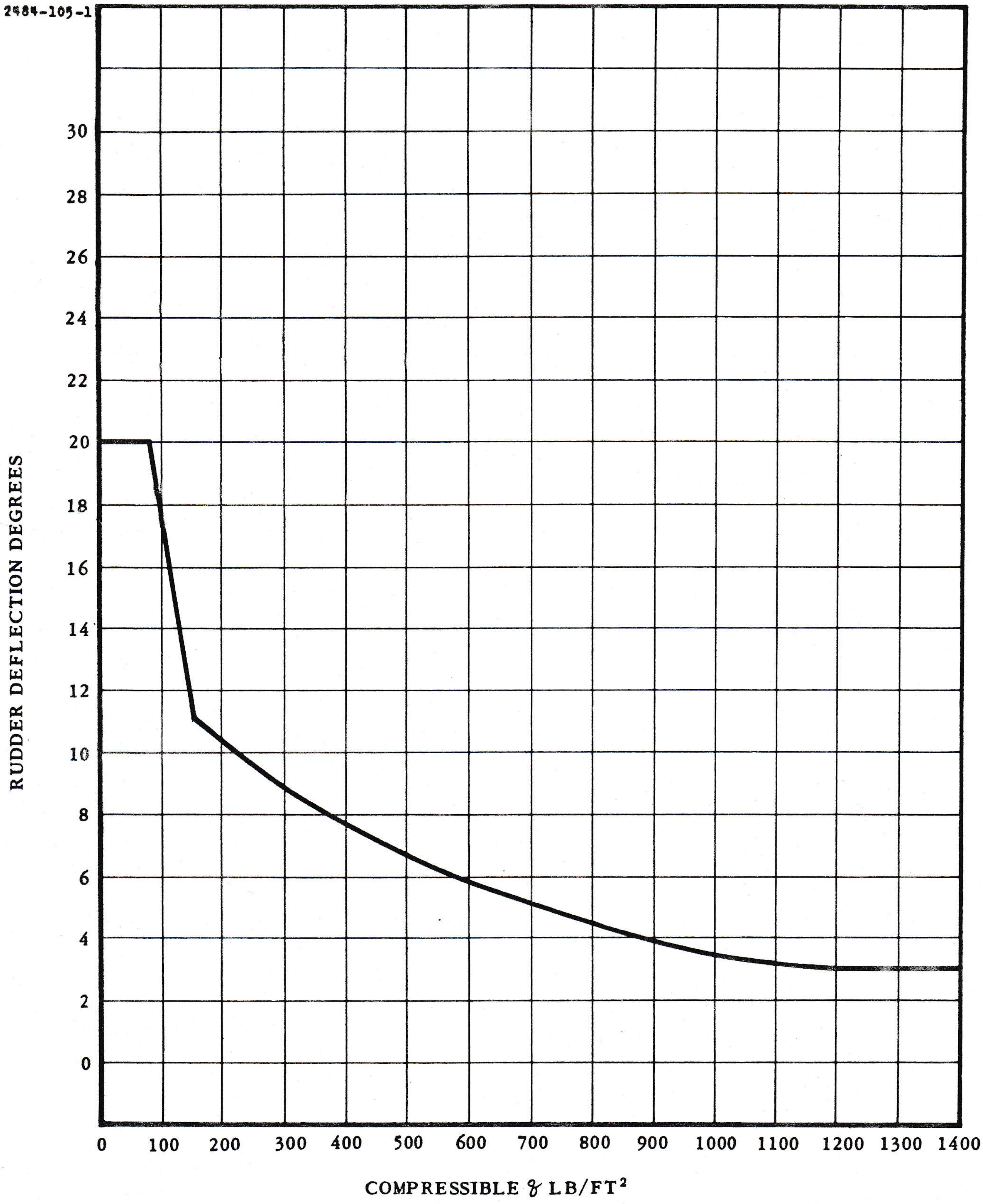


FIG. 31 MAXIMUM AVAILABLE TRIM VS COMPRESSIBLE DYNAMIC PRESSURE