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SUMMARY

A system to provide variable control power and augmented stability for a hovering VTOL aircraft is described. On the Bell X-14A aircraft the exhaust of the two jet engines is deflected to provide lift for hovering flight. Bleed air is taken from the engines and used for reaction control at the wing tips and tail for attitude control while hovering. Two sets of reaction nozzles are used on the modified X-14A. The original set is mechanically actuated by the pilot; the other set, which uses electric servo-driven nozzles, was added to generate variable-stability control moments about all axes. Variable-stability modes provided are rate damping, cross-coupling cancellation, augmented pilot control, and stiffness with maneuverability cutout. The control system for one axis is outlined, and criteria for dividing bleed air between the two sets of nozzles are presented. The development of the nozzles for constant flow and pilot safety controls are discussed.

INTRODUCTION

In aircraft handling qualities research, variable-stability aircraft make possible the exploration of a wide range of the fundamental aircraft parameters, such as damping and control power about each axis. Such a research aircraft enables the pilot to evaluate combinations of variables under realistic flight conditions involving specific tasks. Investigations of new handling qualities concepts, exploration of characteristics of aircraft in the design stage, or improvements of existing aircraft may all be accomplished.

The technique for varying control power and dynamic response depends on supplying commands to the control surfaces, or auxiliary control devices, in addition to the pilot's normal commands. These additional commands are determined on the basis of both pilot command and aircraft motions. Servomechanisms are convenient for introducing these commands into the control system. Variable-stability systems are quite flexible since commands can easily be varied in amplitude, mixed in various proportions, reversed, or even adjusted to put the aircraft into an unstable condition.

The concept of variable-stability aircraft is particularly valuable in the solution of VTOL stability and control problems. The inherent aerodynamic damping of the VTOL aircraft tends toward zero when flight velocity approaches zero and hovering is reached. Since control system research has shown that the ability of a pilot to control a system depends on the damping and control parameters, the choice of these parameters for VTOL aircraft is exceedingly important, first, to assure good handling qualities, and second, to limit total power requirements. Reference 1 reports typical results obtained with an aircraft equipped with variable-stability systems that allowed the control power and damping to be varied about all three axes. The present report describes the original aircraft control system, the variable-stability system, the servo system design with particular emphasis on the motorized nozzle and safety subsystem, simulation studies, and system performance.

DESCRIPTION OF THE ORIGINAL AIRCRAFT CONTROL SYSTEM

The Bell X-14 VTOL (fig. 1), a fixed-wing, jet-propelled, deflected-jet airplane, was the test bed for the research program (see ref. 2 for a complete aircraft description). The exhaust from the jet engines passes through cascade diverters which enable the pilot to select any condition between horizontal and vertical thrust, or to make a transition from one to the other in the air. During hovering flight, the pilot controls the attitude of the aircraft by one of two sets of air jet nozzles at the tail and wing tips; the other set of nozzles was added to provide the variable-stability characteristics. The air for all nozzles is bled from the compressors of the turbo-jet engines, thus diverting thrust from the propulsion system.

The nozzles used by the pilot are mechanically connected to his controls (the stick and the pedals). The pitch nozzle is on the tail and has exit areas top and bottom. Changing this differential area produces a pitching moment; the total nozzle exit area is a constant. The nozzles on the wing tips are used for roll and yaw control. Changing the difference between the left nozzle exit area and the right nozzle exit area generates a rolling moment. A yawing moment is created by rotating the thrust vectors from the left and right nozzles in opposite directions about the lateral axis of the aircraft. The total nozzle exit area of the roll-yaw system is constant.

The basic X-14 aircraft has gyroscopic cross coupling, as the result of the angular momentum of the engine rotors, between the pitch and yaw axes of the aircraft. For example, a pitch-up motion causes a left yawing moment proportional to the pitching rate of the aircraft. From a handling qualities standpoint, this coupling limits the pitch rate at which yawing can be controlled.

VARIABLE-STABILITY SYSTEM

The first research program using the X-14 aircraft was directed at examining the control power and damping relationship in the hovering mode to define boundaries of good, acceptable, and unacceptable control (ref. 3).

To carry out that research program, the following system requirements were established:

a. Change the existing maximum obtainable acceleration proportions from 10:3.2:2.8 for roll, pitch, and yaw, respectively, to 10:5:2.8.

b. Provide maximum control and damping capability consistent with the constraints of weight and safety.

c. Keep added weight low to conserve the already short flight time (time of hovering flight is now approximately 15 minutes).

d. Keep added system independent of the pilot's basic hovering control system. This meant that the variable-stability system operated in parallel with the pilot's basic hovering control system.

e. Provide the ability to vary the control and damping parameters in the various modes of operation.

To enable these requirements to be met, the two original engines were replaced with General Electric J-85-5A engines to provide a greater amount of bleed air for reaction control and to give 25 percent more thrust with about 400 pounds less weight. Using these engines also resulted in less angular momentum and associated cross coupling.

The system added to the basic aircraft was designed so that control power and damping could be varied independently for all three axes and to provide control cross coupling if desired. The pilot was given control of all modes and had the option of reverting to the basic aircraft at any time. For safety, electrically operated air shutoff valves were put in the added system.

The augmented system provided control moments proportional to the position of the pilot's control stick and pedals, but the moments were independently adjustable in magnitude and direction. The damping moments were linear functions of the aircraft angular rates.

A stiffness generator stabilized the aircraft about a given attitude during rapid disturbances, thus reducing the pilot's effort to maintain a given attitude in the presence of gusts. A maneuverability cutout automatically disables the stiffness circuit whenever the pilot moves the controls more than a preset percentage of the maximum control motion. This allows the aircraft to be maneuvered easily during the hovering task.

The safety of the aircraft and pilot were major considerations in the design. Because of its adjustable characteristics, a variable-stability system is basically more complex than a normal aircraft control system and this inherently leads to greater chance of failure.

Special safety considerations were required because the pilot may examine nearly uncontrollable conditions to establish boundaries of controllability. In the event these boundaries are exceeded, it is necessary that the pilot be able to return the aircraft to a safe flight condition quickly and without disturbance. It is also important that the transition from the normal aircraft to the variable-stability operation be smooth and that inputs to the variable-stability system not be sensed through force or motion at the pilot's controls.

Since the bleed air of the jet engines is used as the source of power for the pilot's basic control system, it is advantageous to use it as a source of power for the variable-stability system also. The characteristics of the jet engines made it essential that the total amount of bleed air be constant. The air flow through the basic control nozzles is constant; therefore, the added flow for the variable-stability nozzles also has to be constant. This assures constant flow conditions in the air ducts.

Check valves in the bleed air discharge lines from each engine prevented the bleed air from reversing the rotation of an engine if it should fail. The size of these check valves was increased to accommodate an increase in air flow.

Separate air ducts were provided for the variable-stability nozzles so that air to these nozzles could be shut off. This assures that a failure in the variable-stability system cannot offset any of the pilot's basic control moments. It was decided that the basic control nozzles should produce at least 10 percent more moment in each axis than the variable-stability control system. An automatic failure detection circuit also was included in the system. A button was provided on the stick grip so that the pilot can immediately transfer from the variable-stability mode to the normal aircraft configuration.

SYSTEM DESIGN

Bleed air was selected as the source of reaction power for the X-14A servo system for several reasons. (1) No weight need be added for the source although additional ducting was required. (2) Any other equivalent source, such as air bottles or reaction devices, would necessitate too large a weight addition. (3) A constant ratio of maximum acceleration for the pilot control to the maximum acceleration for the variable-stability system was assured for any axis, thus giving the pilot the same percentage of override capability no matter what engine speed was used or what failure might occur in the air supply.

The total air-flow rate for both engines was 85.0 pounds per second. A constant 10 percent, or 8.5 pounds per second, of the total engine air was to be bled off for all reaction control. This was greater than recommended by the manufacturer, but it was estimated to be permissible for short flight times. Engine data for 8.5 pounds per second of bleed air gave a pressure of 102.9 psia and a temperature of 516° F. Pressure drops through the engine bleed ports, check valves, shut-off valves, motorized valves and ducting, plus the drop necessary for establishing a flow of about 300 feet per second, resulted in an available nozzle pressure of 76.0 psia at 497° F. The total exit area of the nozzles was computed to be 7.17 square inches (appendix). The total reaction force is 500 pounds for this air pressure and exit area of the nozzles (appendix). This force was distributed among all the reaction nozzles to meet the previously stated requirements. To change the maximum force of any nozzle, its exit area must be changed, with an offsetting change in the area of other nozzles.

To maximize torques from the air forces available, the control nozzles were located on long moment arms. Because of space and ducting problems, the variable-stability roll nozzles were placed on the wing tips, while the pitch and yaw nozzles were placed at the tail. Only one independent force exists for each of the pilot's roll-yaw nozzles. All the variable-stability nozzles are separate so that no inadvertent cross-coupling effects will be introduced. The location and magnitude of individual nozzle forces are shown in figure 2. The corresponding maximum accelerations are computed in the appendix.

NOZZLE DESIGN

Extensive effort was expended in developing the nozzle design. To generate the variable-stability control moments proportional to inputs and to meet the system requirements, an independent motor-driven air reaction nozzle was needed. The design criteria for this nozzle were established by (1) the characteristics of the bleed air and the jet engine, (2) the requirement that the reaction force from each nozzle should be proportional to the command signal so as to adapt to a control system readily, and (3) the requirement that the nozzle should yield zero net force in event of loss of driving power.

An electric motor-driven nozzle was decided upon since there was adequate electrical power available on the basic aircraft. Since any weight unbalance for a hovering aircraft should be counterbalanced by weight rather than by reaction force and the nozzles at the tail require 3 to 1 counterbalancing to keep the center of gravity above the thrust vector, the weight of the small motor and gear drive had to be low. The driving torque was to be kept low enough to permit use of a low power servomotor.

The concept of the original X-14 pilot pitch control nozzle, in which air is discharged from diametrically opposed variable orifices, was used for all stability augmentation nozzles. Rotary valve motion was used to control the net area since the electric motor and gearing adapted easily to rotary motion.

Nozzle Torque

As a preliminary test, the torque needed to actuate the original pilot's pitch control nozzle under operating air pressure was measured. The results clearly indicated the need for a large reduction in the torque for a servomotor drive. There were two components of torque to be overcome; one, the bearing frictional torque caused by unbalanced forces on the nozzle rotor, and the other, the torque caused by the difference in the tangential forces on the rotor edge areas. These torques are illustrated by the cross-sectional representation of the nozzle (fig. 3).

With the rotor at center, the nozzle pressure inside the rotor acts on the unequal projected rotor areas and so produces a net side force on the rotor bearings. This can be minimized if an additional opening is cut in the rotor, as shown in figure 3, to equalize opposing projected areas. When the rotor is near its extreme position, the internal pressure distribution must balance the net reaction force output of the nozzle and, so again, produces a side force on the rotor bearings. The effects of this force can be minimized by antifriction bearings.

The tangential component of rotor torque is due to pressure distribution changes, as the orifices are opened or closed, causing force on the rotor edges, which can produce large torques on the rotor. These, fortunately, tend to center the rotor in most cases.

The rotor edge of the orifice proved to be the most important parameter in meeting the nozzle performance design criteria. The ability of the nozzle to center in a zero net thrust position, when driving power was lost, depended on the choice of edge for the orifice. The torque required to open was dependent on the orifice edge area. For full thrust in one direction, one rotor edge should nominally be covered by the outer shell. However, excessive torque would be required to recover from such a complete closure, so the orifice was not allowed to close completely. Several rotor edge modifications were tried (fig. 4): The 1/8-inch thick square edge orifice had high torque requirements. A chamfered edge reduced the torque required. An edge was then machined with ribs extending to and reinforcing a thin edge. The results of testing were promising so the ribs themselves were next chamfered to reduce further the rotor edge area.

Rotor edge area control was determined to be the answer to the centering problem by the following experiment. Solder was added to previously tapered edges to see whether it had the opposite effect, and indeed it did. When solder was added to the rotor edge of one orifice to increase the area facing the stream and the rotor edge of the opposite orifice was relieved, the nozzle could be made to stop at any desired position under zero torque. An additional test indicated that the major part of the pressure drop from nozzle pressure to ambient pressure occurs in a very short distance at the orifice edge.

Final Nozzle Design

For the final configuration, the rotor was made thinner (1/16 instead of 1/8 inch) to keep the driving torques low and to enhance the centering characteristics of the rotor edge area. The 120° segment of rotor between orifices was removed (see fig. 5) and the area balancing port in the rotor (see fig. 3) was eliminated. The bearing forces caused by the unequal opposing projected rotor areas were made small by nearly equalizing the pressures inside and outside the rotor. This was done by increasing the diametrical clearance a small amount giving a fairly large circumferential area increase for air distribution around both ends of the rotor.

Data in figure 6 for the final nozzle configuration show the typical rotor torque required to obtain a given orifice opening at the operating pressure level in the duct. The servo drive can operate satisfactorily at these torque levels. For servo application it is important that the torque be proportional to rotor displacement from center position which, in this case, means that area and output thrust are also proportional to torque.

Figure 7 is a photograph of the final nozzle configuration with the rotor in its centered position. All the nozzles had 30° of arc width of opening in the stator shell. They differed in length of opening to satisfy the area requirement.

The nozzle reaction force at the temperature and pressure ranges experienced on the X-14A follows approximately the formula (appendix):

$$F = C_V C_d A P_1 (1.268 - P_{atm}/P_1)$$

Using velocity coefficient $C_V = 0.95$, discharge coefficient $C_d = 0.90$, atmospheric pressure $P_{atm} = 14.7$ psia, nozzle pressure $P_1 = 64.7$ psia, and exit area $A = 0.80$ square inch in the formula gives a maximum thrust of 46 pounds for the roll nozzle. In actual test the thrust was 44.8 pounds. The linear relationship of reaction force with orifice area for a given duct pressure was confirmed by measurement.

Servo Drive

The servos position the variable-stability nozzles in response to various electrical signals. Type 0 position servos (see ref. 4) were used. For each servo an alternating current summing amplifier combines all input signals and amplifies the error signal if the nozzle is not in its commanded position. All channels have similar inputs. For the pitch channel, the inputs are:

1. Pitch rate gyro signal for damping

2. Roll rate gyro signal for cross coupling into pitch axis

3. Yaw rate gyro signal for cross coupling into pitch axis

4. Stick pitch position signal for control power

5. Pitch nozzle position signal for position feedback in the servo

6. Pitch nozzle tachometer generator signal for rate feedback in the servo

7. Pitch summing amplifier output signal for gain control

8. Stiffness generator signal for stiffness control

9. Spare input

A magnetic amplifier, accepting the output of the alternating current summing amplifier, drives the two-phase, ten-watt servomotor which operates the nozzle. The basic electrical diagram for a single axis of the servo system is shown in figure 8.

The nozzle and drive system were bench tested. Figure 9 shows the nozzle step response with and without air. The natural frequency of the unloaded nozzle was about 9.5 cps, which remained essentially unchanged with 75 psia air in the nozzle. This response introduces less than 20° phase shift for normal pilot inputs (which are less than 2 cps).

The transducers needed to get the electrical signals from the pilot, nozzle, and aircraft motions included position transducers, tachometer generators, and rate gyros. A 5-volt signal level was specified to drive the nozzle full open. For this value the system was not sensitive to electrical noise and yet great amplification of transducer signal was not necessary.

Rate gyros were used to produce a signal proportional to the aircraft angular velocity about each axis. The gyros were capable of measuring maximum rates of 60°/sec in pitch, 60°/sec in yaw, and 180°/sec in roll. The output voltage of each gyro was amplified and processed to give a maximum signal of 5 volts of each phase. Position pickoffs used were microsyns and inductive potentiometers with a linear range sufficient to supply the necessary 5-volt output without amplification.

At constant aircraft engine speed the engine rotor angular momentum is fixed and for any aircraft rotational rate in pitch or yaw a definite amount of torque is generated by the gyroscopic cross coupling. This torque is proportional to the angular rate and in a direction dependent upon the direction of aircraft rotation. Since the rate gyro output is also proportional to aircraft rate, a nearly exact cancelling of the cross-coupling effect can be achieved if a proper amount of this signal is fed into the proper axis with the proper sign.

The stiffness generator reduces the effect of short-period unwanted disturbances, such as gusts. This reduction could be obtained with an attitude reference as a control system feedback. However, for a desired new attitude, the attitude reference would oppose pilot control. Thus it was necessary to allow the attitude reference signal (an integrated rate gyro signal) to decay over a period of time. A first-order resistance-capacitance lag circuit gave an approximation of the desired characteristics. A time constant of 5 seconds gave adequate stiffness but restricted aircraft maneuvering response.

To cut out the stiffness effect if a control motion of over 25 percent was commanded by the pilot, a relay was operated whenever the stick position signal was over 1.25 volts (25 percent of maximum).

SYSTEM RESPONSE ON A SIMULATOR

In the study of a control system which involves new concepts, such as the one presented here, it is advantageous to study the system through simulation using as much actual flight hardware in the system as possible. Therefore, a low-friction, air-bearing supported, horizontally rotating beam was set up (fig. 10) to simulate the aircraft characteristics for one axis at a time. With this device it was possible to demonstrate the effectiveness of the control and damping of the aircraft and also to check the operation of the entire system before flight. Inertia weights and the lever arm for the nozzle were chosen so the acceleration would be identical to a desired axis of the aircraft. Air was piped through a rotating joint to the test stand and provided both air-bearing pressure and nozzle reaction forces. Nozzle air pressure was adjusted by a flow valve to obtain flow rates similar to those expected in the aircraft. The servo-drive components were mounted on the rotating unit and connected to control switches and signal sources by an overhead cable. This cable was brought to the center of rotation so minimum extraneous torque would be produced on the simulator. Without added servo damping the rotational speed of the beam decreased from an initial 40°/sec to 10°/sec in 3-1/2 minutes. This natural damping was considered satisfactory for the simulator tests.

When the nozzle servo system was used without damping, the angular position of the beam was relatively hard to control. As servo damping was added (by incorporating a signal from a rate gyro) the control characteristics improved and position changes could be easily controlled. When damping was reversed in sign, the system soon became uncontrollable.

SAFETY PROVISIONS

Safe systems are needed for any aircraft but they are extremely important for hovering aircraft which may have very little altitude for corrective maneuvers. Electronic failure may either make the servo inoperative or cause it to drive to full output. Mechanical failure may permit the nozzle to be free to air center or may cause it to jam in an open position and produce a large reaction force. A switch is needed for shutting off the power and a valve for shutting off the bleed air.

In order to implement an error detection subsystem for each axis, a duplicate summing amplifier with all nine inputs was used. This failure detection amplifier was designed so that if it failed or detected any type failure it would operate a cutoff relay. There is an error detection channel for each of the servo nozzles so only the faulty axis will be cut off, both roll nozzles going off if there is failure in either one.

The pilot-operated emergency controls and the sequence in which they are used are as follows. If the commands driving the variable-stability nozzles are suspect or have caused an unstable flight condition so that the pilot wishes to center all the nozzles, he may turn off the variable-stability system switch directly or use a button on the stick grip which electrically releases the same switch. All servos will then drive to the center, or zero force, position. If one of the nozzles is not driven to its center position, the automatic error detector will operate and remove excitation from this nozzle. Then, unless it is stuck, it will air center. If one of the nozzles does not air center, a second stick grip button removes all power from the variable-stability nozzles and also energizes the motors to close the air valves. A 2-inch valve is in each wing duct and a 3-1/2-inch valve is in the duct to the tail. The closing time for these valves is 2 and 4-1/2 seconds, respectively. If a nozzle is jammed in an off-center position the pilot must overcome the acceleration it produces with his basic control system until the air is shut off. He will then have his full basic control for landing. The pilot may operate the second button first to turn the system completely off with minimum delay.

SYSTEM PERFORMANCE IN THE AIRCRAFT

Electronic parts of the system (fig. 11) have given practically no trouble. One failure did occur in the power supply during a flight and the automatic safety system operated to turn the system off. After some flight experience, the automatic safety system was found to be too sensitive, and since the pilots did not like to lose the variable-stability control unless it was absolutely necessary, the safety system sensitivity was reduced by 50 percent. This has worked very well and is still much faster than the pilot's reaction. The motorized nozzles have performed well for over 150 hours of flight time. The nozzles are disassembled periodically for inspection and preventive maintenance. Several bearings have been replaced as they became noticeably rough and some of the roll pins holding gears on shafts have loosened, caused backlash, and been replaced. High temperature grease and silicone oil have both been used on the bearings, but neither seems to have much advantage on the basis of flight data so far. The nozzle's air centered position has stayed constant. There has been practically no deformation, pitting, or deposits from the bleed air on the critical rotor edge.

The pilots prefer the variable-stability system to be in operation as it makes the aircraft more stable and easier to fly in the hovering mode. First, it cancels out the unwanted cross coupling and, second, it provides damping and so makes the aircraft respond in hovering much like it does in normal flight. The pilot augmented control system permits a yaw rate of about 75°/sec and a pitch rate of about 60°/sec with no noticeable gyroscopic cross coupling between the pitch and yaw axes. The stiffness control augmentation makes it easier for the pilot to keep the aircraft steady under gusty conditions. The safety features have provided intangible benefits in terms of pilot confidence.

The system has operated as designed and provided a means of examining the control power and damping relationship in the hovering mode for the boundaries of good, acceptable, and unacceptable pilot control.

Ames Research Center
National Aeronautic and Space Administration
Moffett Field, Calif., Dec. 4, 1964

APPENDIX

FORMULAS AND CALCULATIONS FOR REACTION NOZZLES

In this appendix pertinent symbols and formulas are given. Calculations are made to establish the various nozzle forces.

SYMBOLS

A	cross-sectional area, sq ft
C_d	discharge coefficient ratio, A_*/A
C_v	velocity coefficient
F	reaction force of pilot nozzle, lb
F'	reaction force of variable-stability nozzle, lb
g	acceleration of gravity, 32.2 ft/sec^2
I	moment of inertia, slug-ft^2
L	moment arm, ft
$\dot{m}$	mass flow rate, slugs/sec
P	pressure, psia
R	universal gas constant
T	absolute temperature, $^\circ$R
V	flow velocity, ft/sec
$\dot{w}$	pounds flow rate, lb/sec
α	angular acceleration, rad/sec^2
γ	ratio of specific heats at constant pressure to constant volume and equal to 1.4 for air
ρ	density, slugs/cu ft

Subscripts

1 located upstream

2 located at orifice

3 downstream

* position of sonic velocity

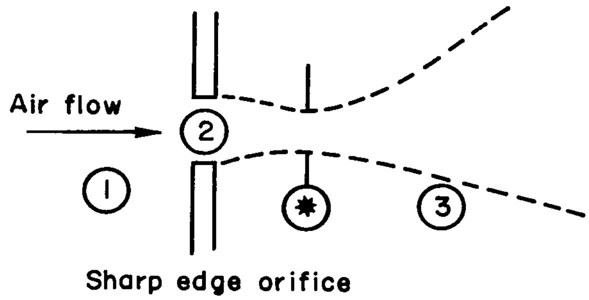
p pitch

r roll

t tail

w wing tip

y yaw



TOTAL NOZZLE EXIT AREA CALCULATED FOR ALL BLEED AIR

Mass flow rate can be expressed as:

$$\dot{m}_* = \rho_* V_* A_*$$

and

$$\dot{w}_* = g\dot{m}_* = g\rho_* V_* A_*$$

From the perfect gas laws it can be shown that:

$$\dot{m}_* = \frac{\sqrt{\gamma}P_1}{\sqrt{RT_1}}\left(\frac{P_*}{P_1}\right)^{\frac{\gamma+1}{2\gamma}} C_d A_2$$

Using an experimentally determined C_d of 0.90 and knowing the ratio of $P_*/P_1 = 0.5283$ at Mach 1.0 (see ref. 5)

$$A_2 = \frac{\dot{w}_* \sqrt{T_1}}{0.4767\ P_1}$$

From engine data, $\dot{w}_* = 8.5$ lb/sec, $P_1 = 76$ psia, and $T_1 = 939^\circ$ R, so solving: $A_2 = 8.5\sqrt{939}/0.4767(76) = 7.17$ sq in. This is the total allowable exit area to keep the bleed flow desired.

TOTAL REACTION FORCE AVAILABLE FROM ALL NOZZLES

The total reaction force, ideally, can be expressed as:

$$F = \dot{m}_* V_* + A_*(P_* - P_3)$$

Because of friction at the orifice edge, V_* will not be obtained over the whole area A_*, but will be less at the outer edges of the flow stream. This, in effect, reduces the sonic throat area to give an average area of sonic flow of $C_V A_*$, so actually,

$$F = \dot{m}_* V_* + C_V A_*(P_* - P_3)$$

Using the relationships $\dot{m}_* = \rho_* A_* V_* C_V$, $(V_*)^2 = \gamma R T_*$, and $P_* = \rho_* R T_*$, and keeping sonic flow:

$$F = C_V C_d A_2 P_1 \left(1.268 - \frac{P_3}{P_1}\right)$$

With $C_V = 0.95$ from test data and P_3 as atmospheric pressure, $F = (0.95)(0.90)(7.17)(76)[1.268 - (14.7/76)] = 500.7$ pounds total reaction force.

DIVISION OF TOTAL REACTION FORCE AMONG THE NOZZLES

For any nozzle: $F = \dfrac{I\alpha}{L}$

$$F_p = \frac{I_p \alpha_p}{L_t} \qquad F'_p = \frac{I_p\ 0.90\ \alpha_p}{L_t} \qquad \alpha_p = 0.50\ \alpha_r$$

$$F_y = \frac{I_y \alpha_y}{L_w} \qquad F'_y = \frac{I_y\ 0.90\ \alpha_y}{L_t} \qquad \alpha_y = 0.284\ \alpha_r$$

$$F_r = \frac{I_r \alpha_r}{L_w} \qquad F'_r = \frac{I_r\ 0.90\ \alpha_r}{L_w} \qquad \alpha_r = 1.0\ \alpha_r$$

The sum of the separate individual forces must equal the total available; F_r and F_y are not independent, and since F_r is the larger, it must be used.

$$F_{total} = F_p + F'_p + F_r + F'_r + F'_y$$

$$500.7 = \frac{1990(0.50\ \alpha_r)}{18.75} + \frac{1990(0.9)0.50\ \alpha_r}{18.75} + \frac{1170\ \alpha_r}{16.9} + \frac{1170(0.9)\alpha_r}{16.9} + \frac{2920(0.9)0.284\,\alpha_r}{18.75}$$

$$500.7 = 53.1\ \alpha_r + 47.8\ \alpha_r + 69.2\ \alpha_r + 62.3\ \alpha_r + 39.8\ \alpha_r = 272.2\ \alpha_r$$

$$\alpha_r = \frac{500.7}{272.2} = 1.84\ \text{rad/sec}^2$$

$\alpha_r = 1.84$ rad/sec^2 $\qquad$ $\alpha_r' = 0.9\ \alpha_r = 1.66$ rad/sec^2

$\alpha_p = 0.5\ \alpha_r = 0.92$ rad/sec^2 $\qquad$ $\alpha_p' = 0.9\ \alpha_p = 0.83$ rad/sec^2

$\alpha_y = 0.284\ \alpha_r = 0.52$ rad/sec^2 $\qquad$ $\alpha_y' = 0.9\ \alpha_y = 0.47$ rad/sec^2

$F_p = 97.8$ lb

$F_p' = 88.0$ lb

$F_r = 127.2$ lb $\qquad$ Non-independent $F_y = F_r \cos 45^\circ$

$F_r' = 114.5$ lb $\qquad$ $F_y = 127.2\ (0.707)$

$F_y' = 73.2$ lb $\qquad$ $F_y = 90.2$ lb

$\overline{\ 500.7 \text{ lb}}$

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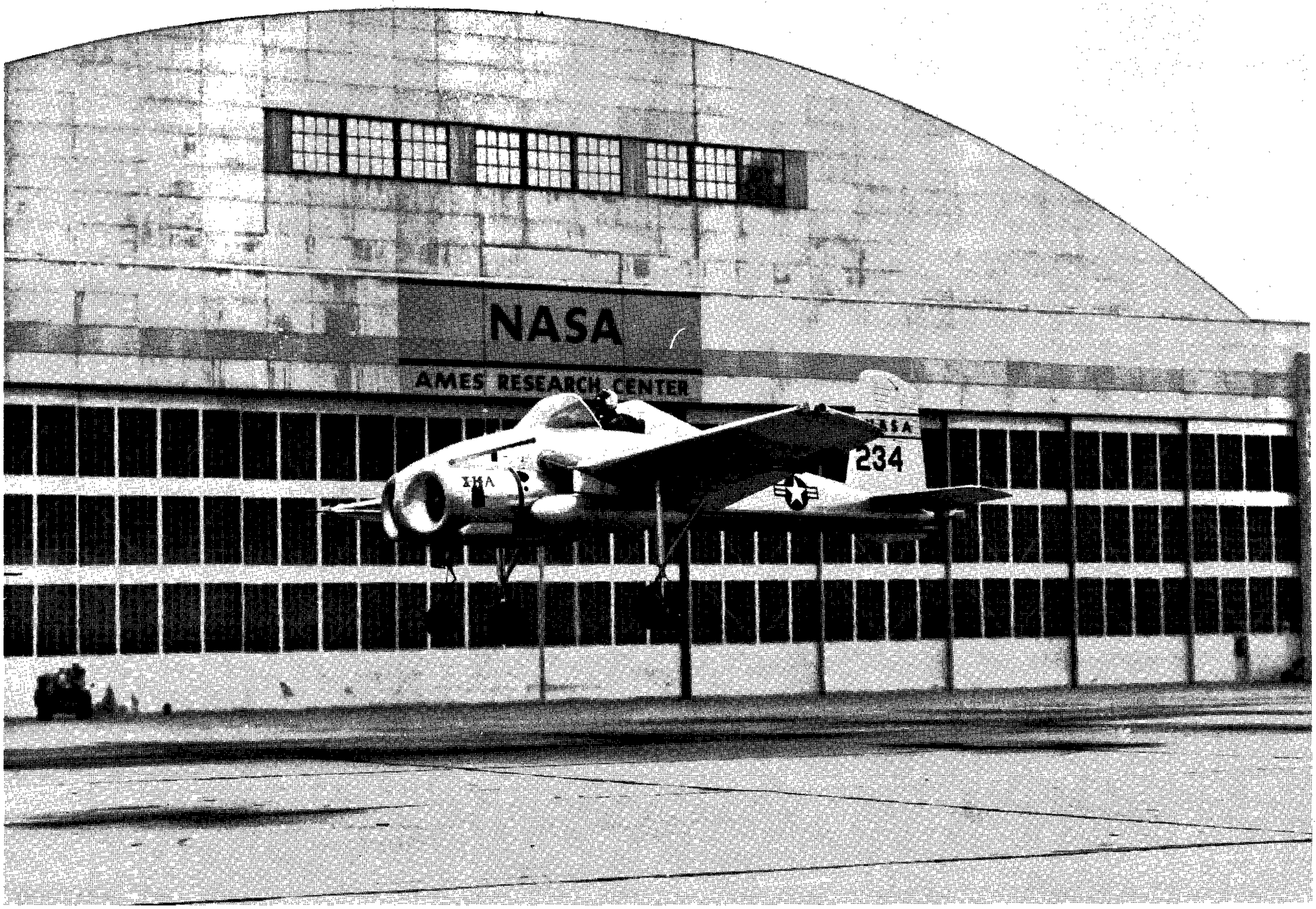

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Figure 1.- Photograph of X-14A hovering.

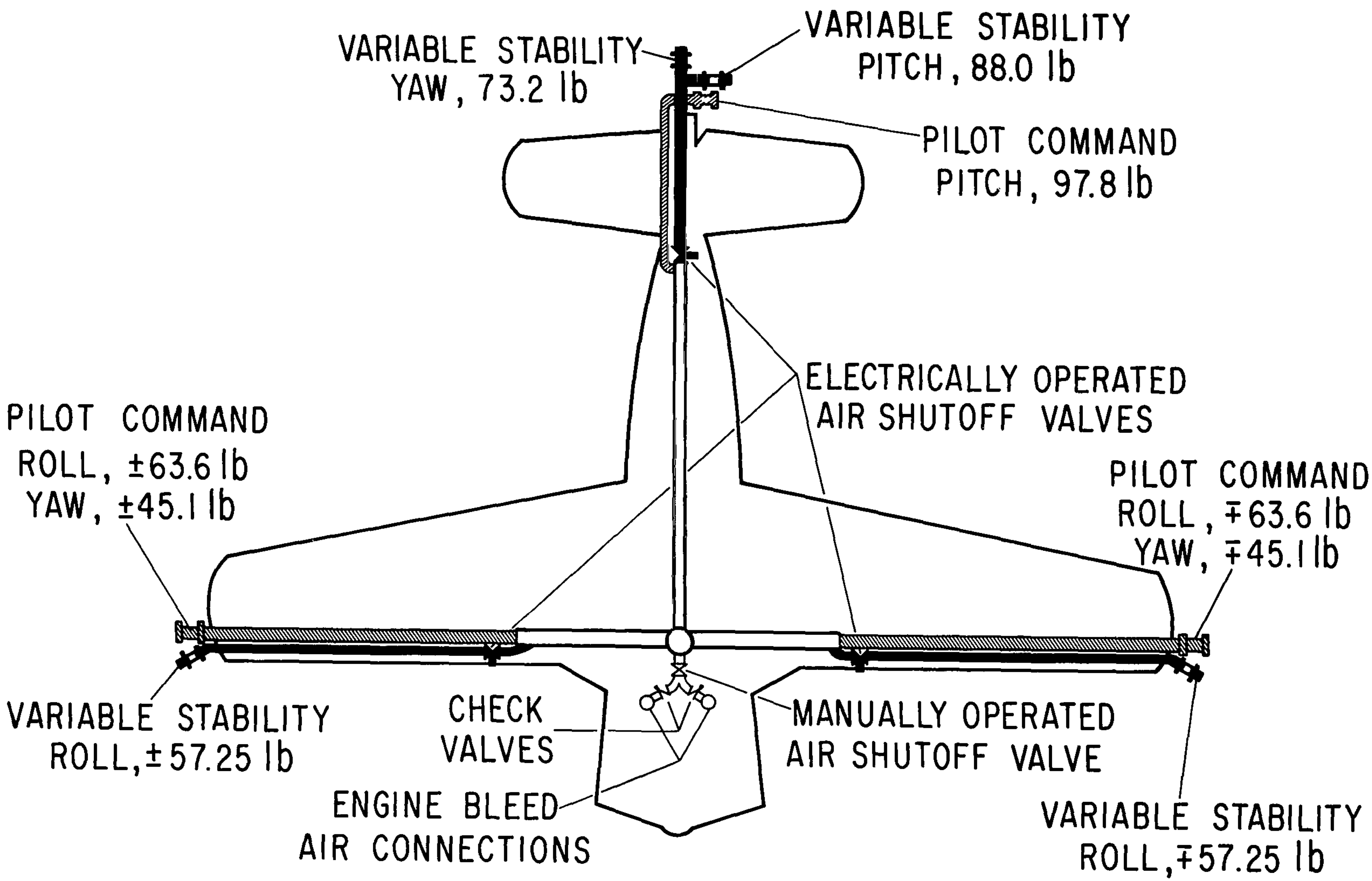
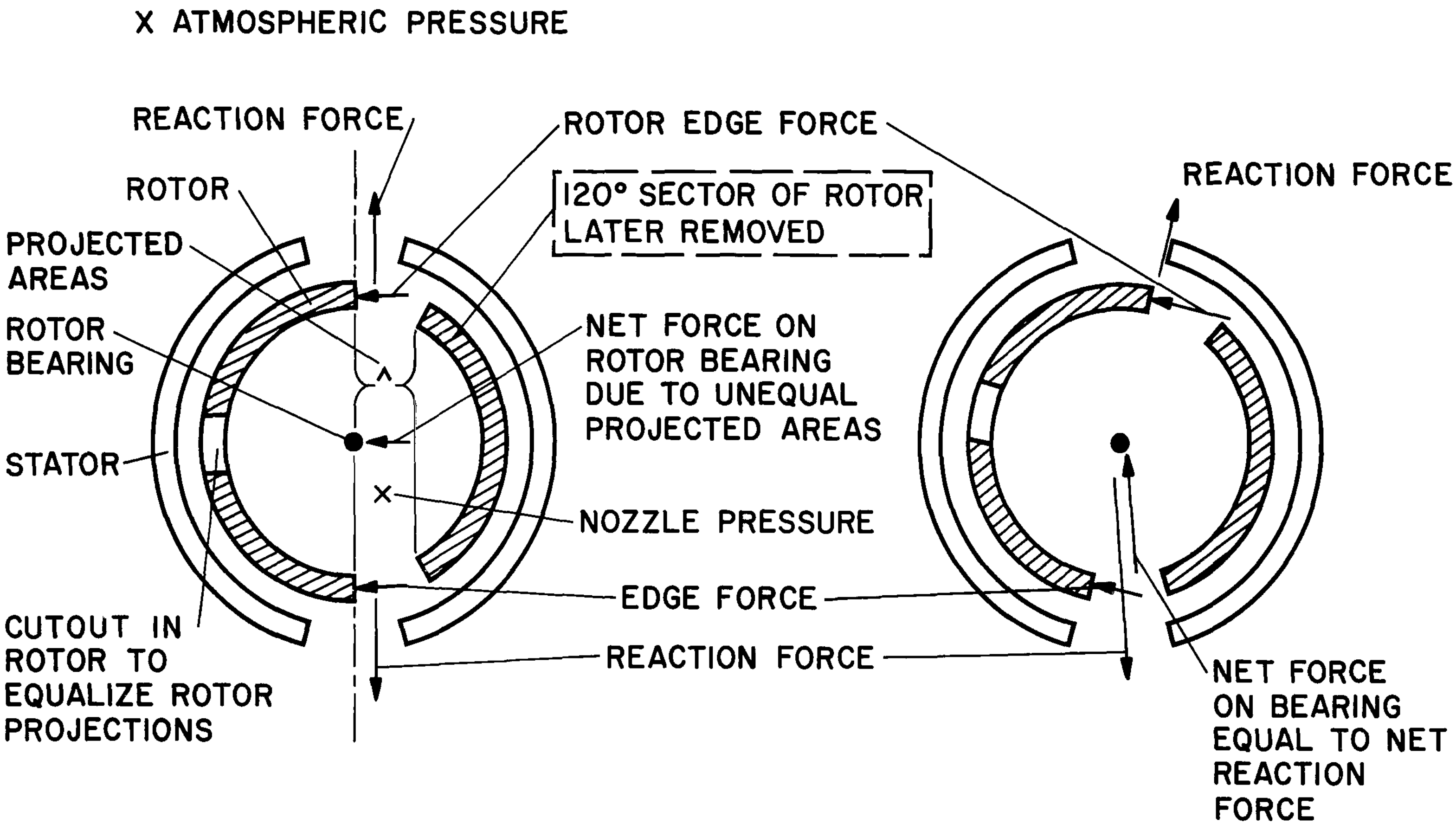


Figure 2.- Nozzle location and forces on X-14A aircraft.



(a) ROTOR AT CENTER

(b) ROTOR NEAR ONE EXTREME

Figure 3.- Forces acting on a prototype nozzle.

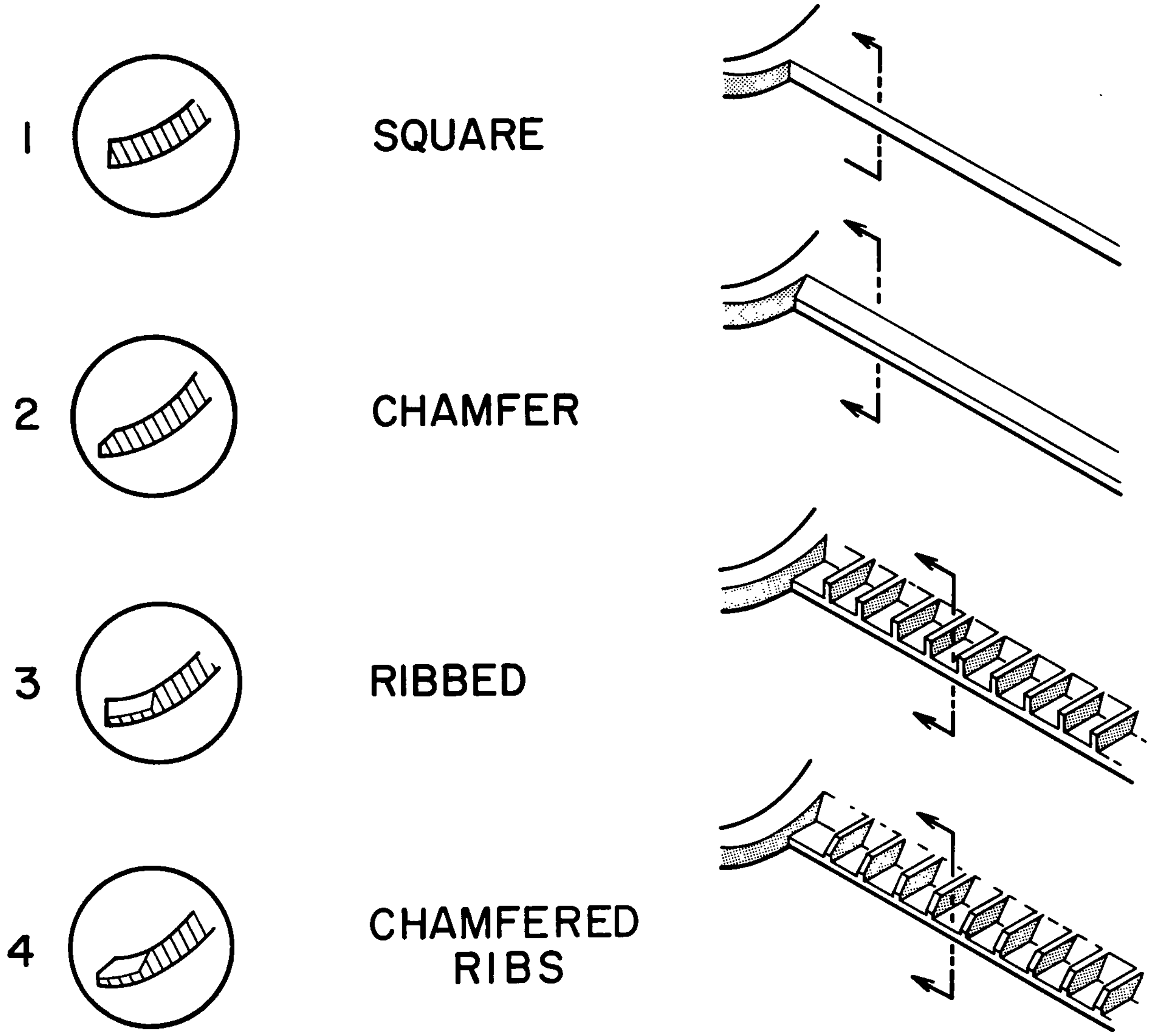


Figure 4.- Nozzle rotor edge modifications.

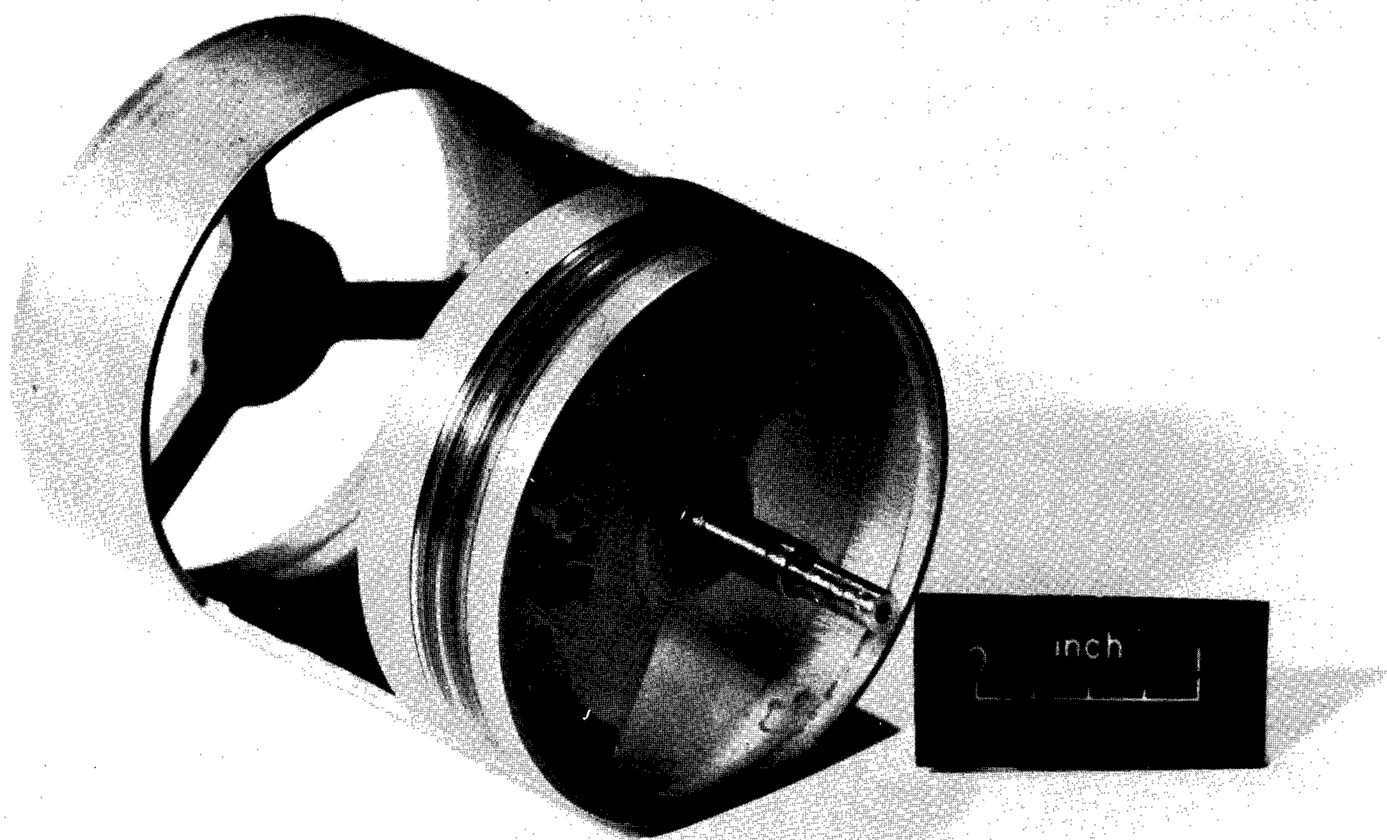


Figure 5.- Final nozzle rotor.

A-31765

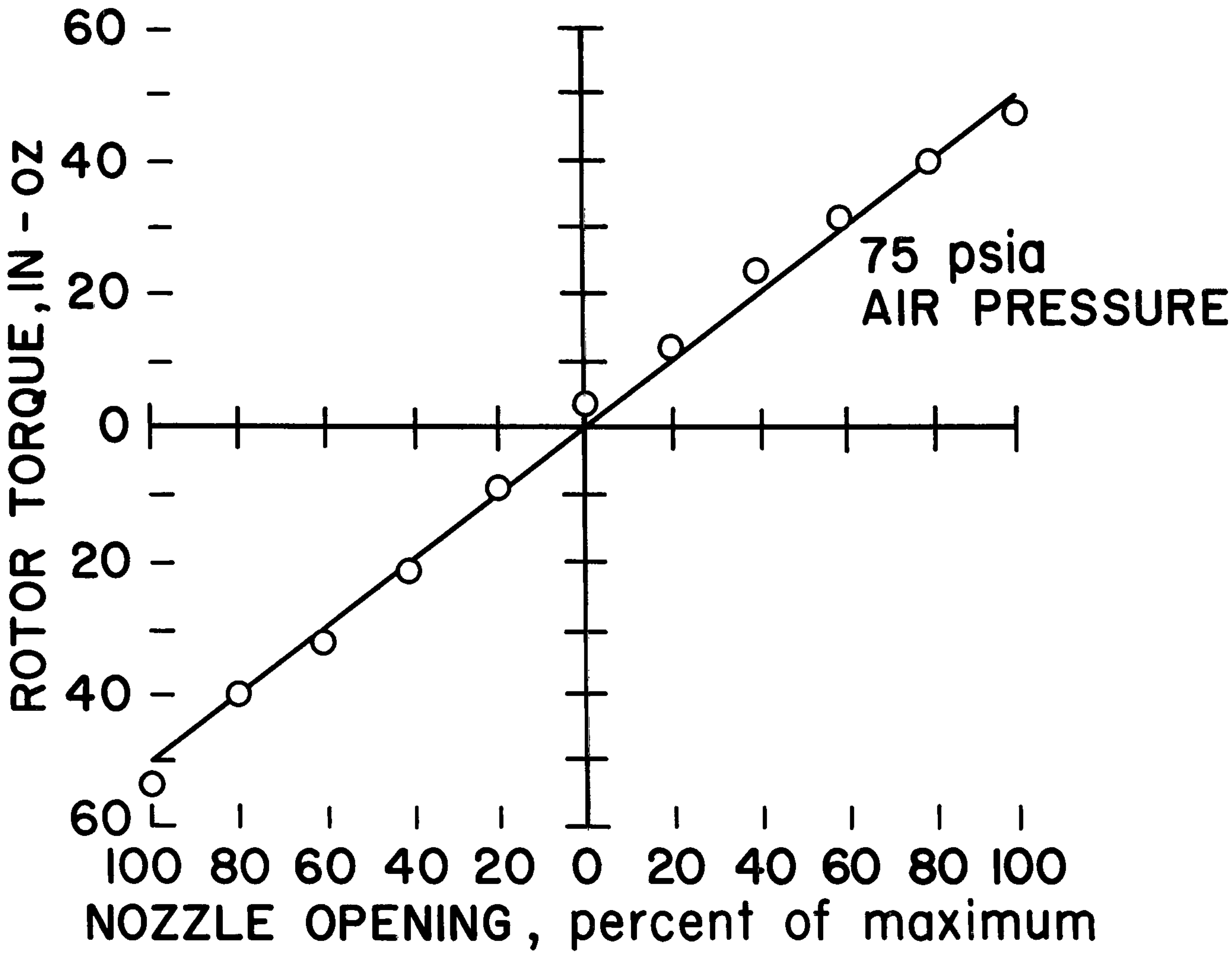


Figure 6.- Typical nozzle torque characteristics.

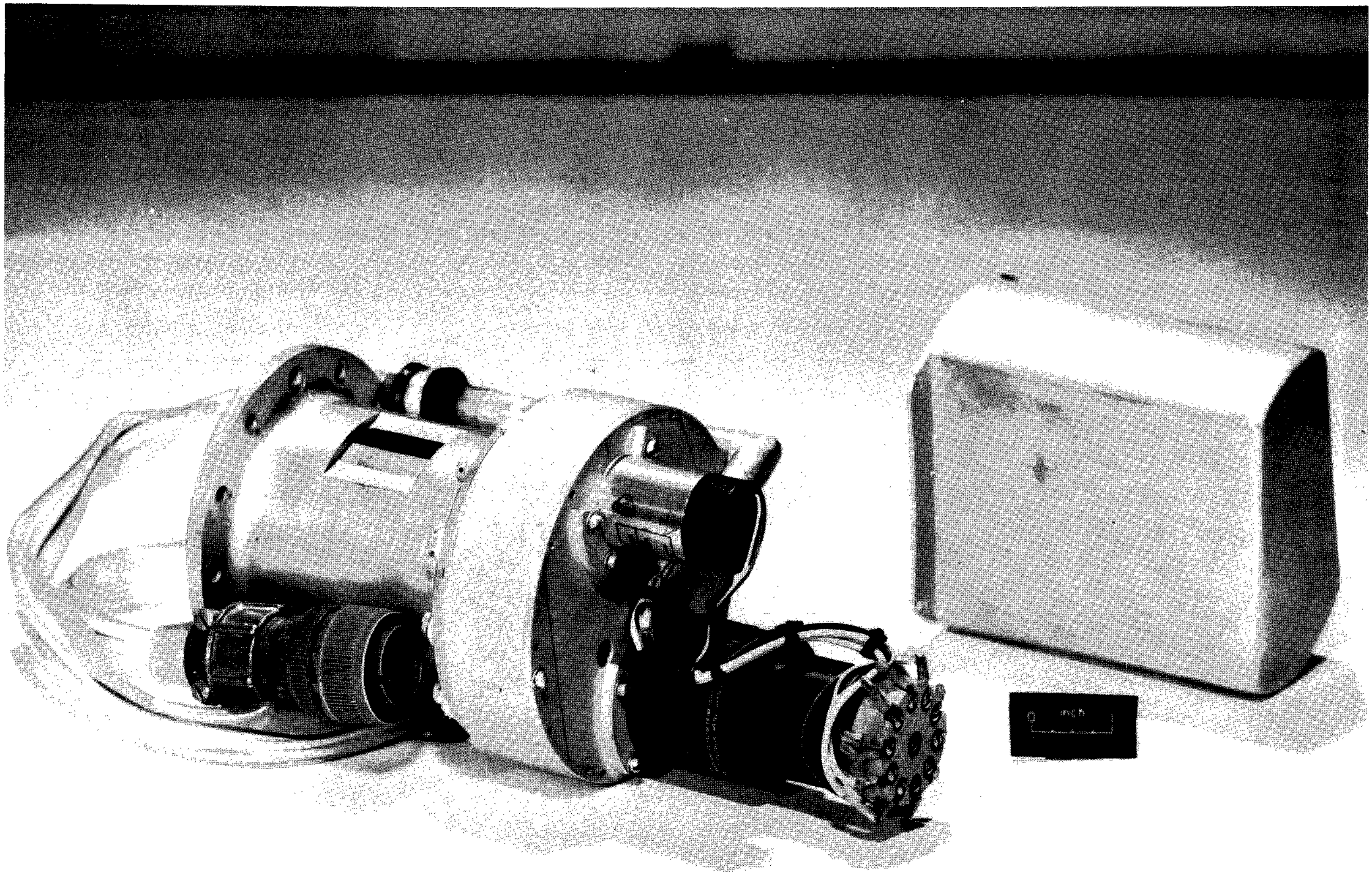


Figure 7.- Final nozzle assembly.

A-31764

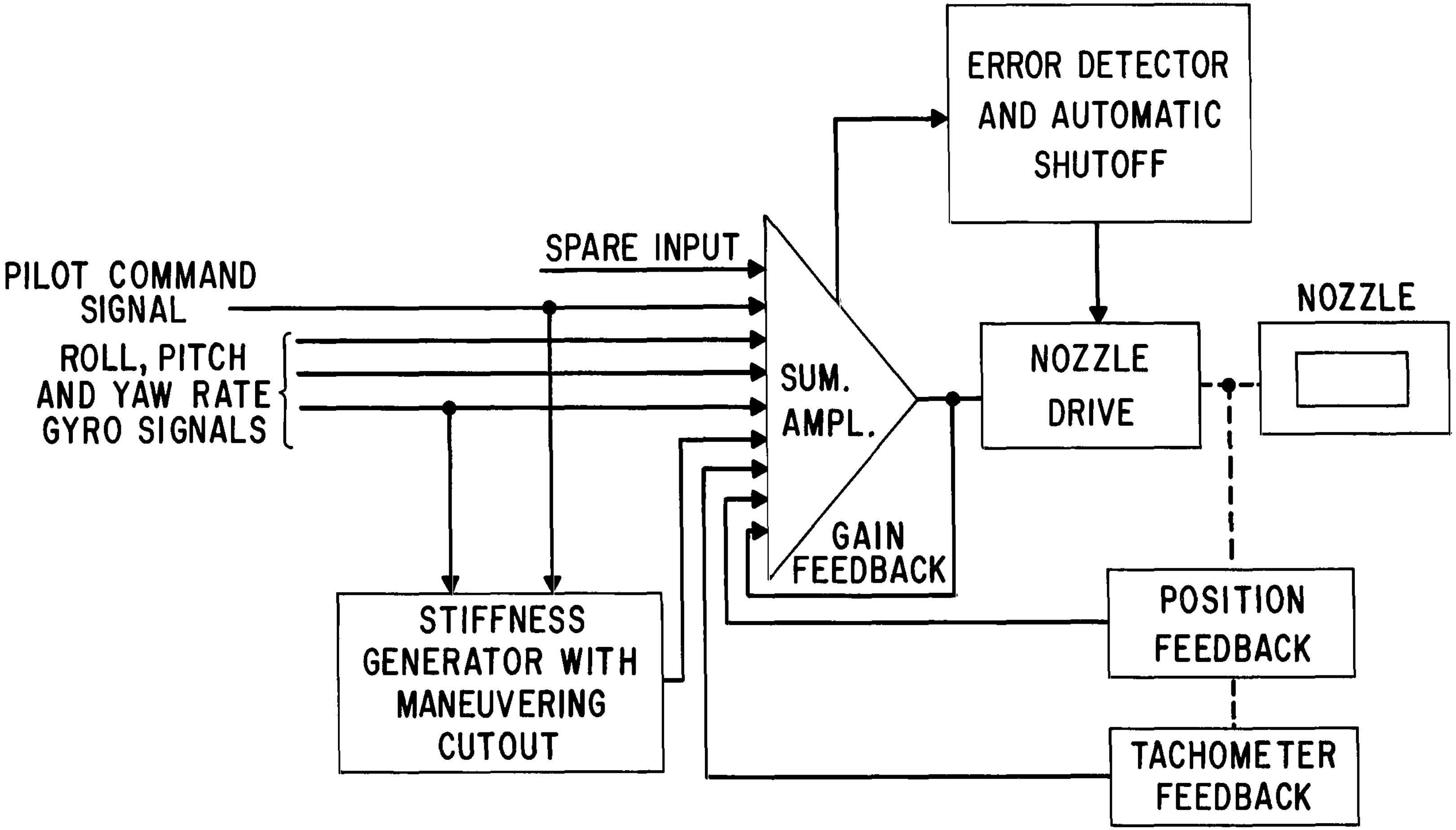


Figure 8.- X-14A control system diagram for one axis.

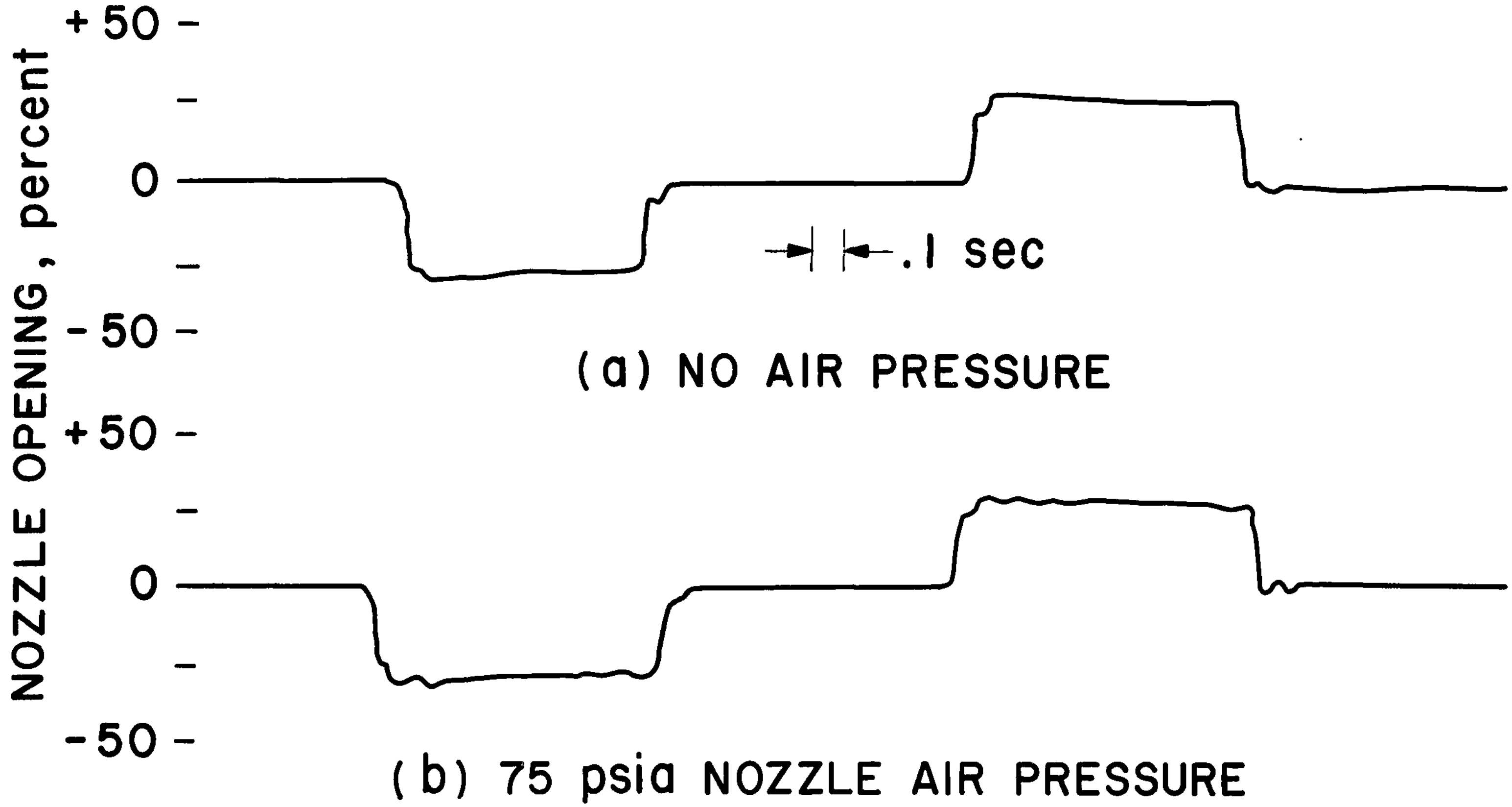


Figure 9.- Nozzle step response.

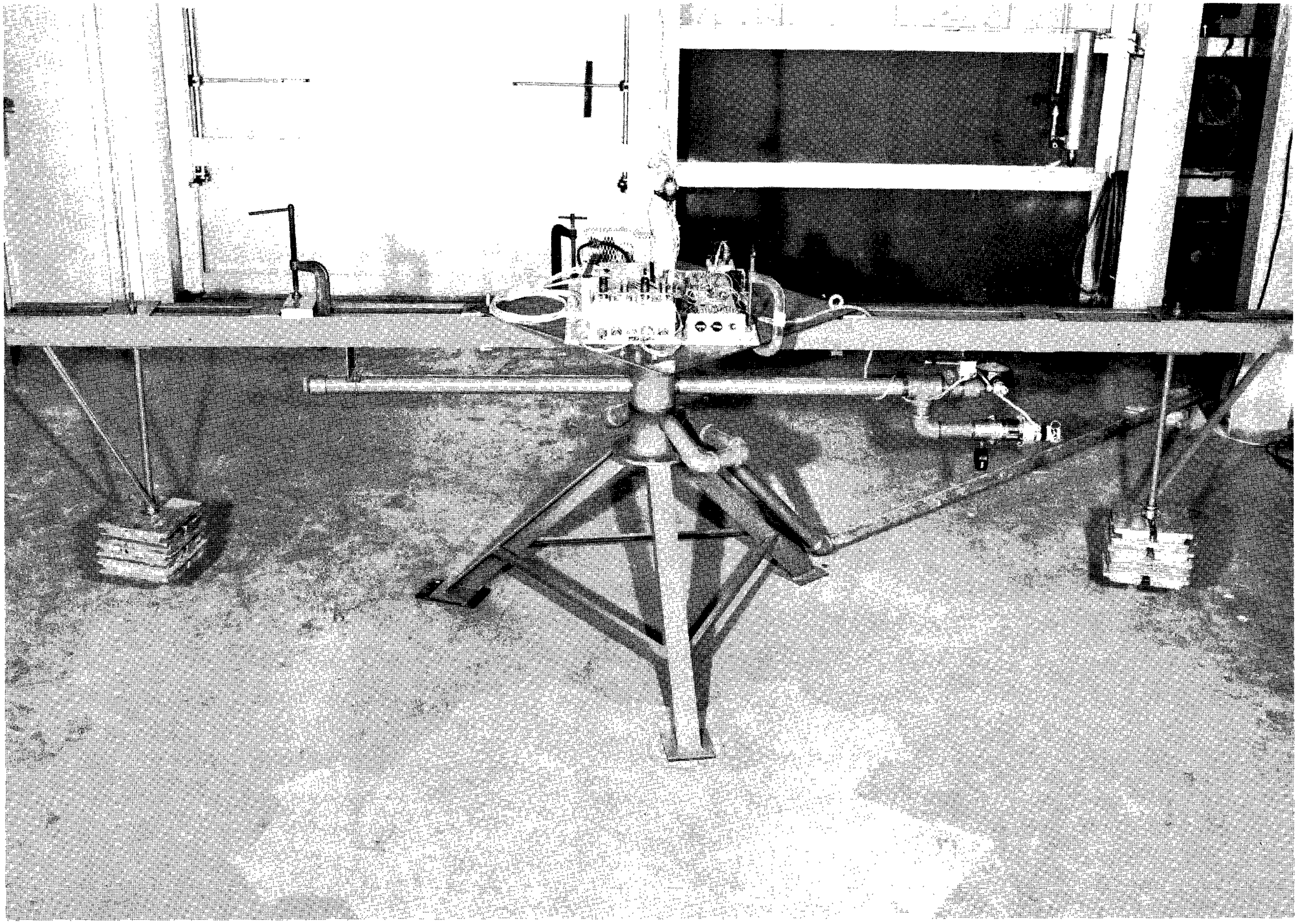


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Figure 10.- Single-axis aircraft simulator.

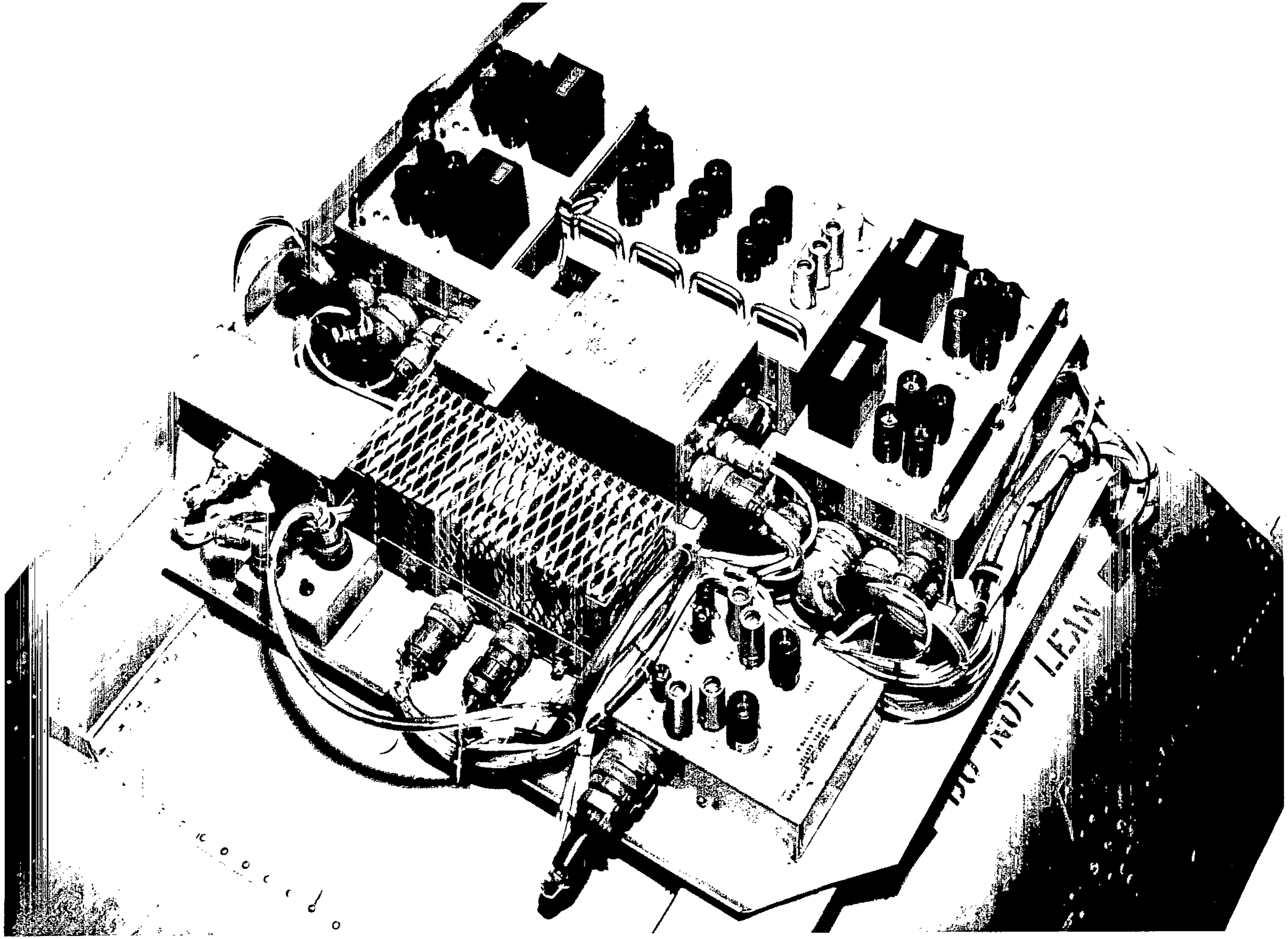

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Figure 11.- Mounting of electronic units.

NASA-Langley, 1965 A-816

2/22/85
76

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