USAAVLABS TECHNICAL REPORT 65-29

AIRCRAFT DESIGN
XV-9A HOT CYCLE RESEARCH AIRCRAFT
SUMMARY REPORT

Report HTC-AD 64-11 (385-X-05)

By
N. B. Hirsh

August 1965

U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

CONTRACT DA 44-177-AMC-877(T)

HUGHES TOOL COMPANY
AIRCRAFT DIVISION

DEPARTMENT OF THE ARMY
U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA 23604

This report was prepared by Hughes Tool Company, Aircraft Division, under the provisions of Contract DA 44-177-AMC-877(T), to permit design substantiation of the XV-9A aircraft. It is published for the dissemination of information and reporting of project results.

Task 1M121401A14403
Contract DA 44-177-AMC-877(T)

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Culver City, California

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FORT EUSTIS, VIRGINIA

ABSTRACT

This report discusses a research program, the purpose of which was to demonstrate the feasibility of the Hot Cycle Rotor System through the design, fabrication, and test of a flightworthy research VTOL aircraft. The XV-9A utilizes the Hot Cycle pressure jet rotor system which was developed by the U. S. Air Force. The report includes a discussion of concepts utilized in the aircraft design and information relating to the configuration, weight and balance, performance, stability and control, dynamics, and structural characteristics of the research aircraft.

PREFACE

This report was prepared in accordance with Contract DA 44-177-AMC-877(T) with the U. S. Army Aviation Materiel Laboratories. The contract became effective on 29 September 1962. The report summarizes the design of the XV-9A Hot Cycle Research Aircraft (U. S. Serial Number 64-15107).

The aircraft was designed by the Hughes Tool Company Aircraft Division in Culver City, California. The design was accomplished under the direction of Mr. H. O. Nay, Program Manager, Hot Cycle Programs, and under the direct supervision of Mr. C. R. Smith, Engineering Project Manager, Hot Cycle research aircraft. This report was prepared by Mr. N. B. Hirsh, Project Administrative Engineer, on the basis of contributions from the following personnel:

S. Cohan	-	Group Engineer, Propulsion Systems
J. F. Conlin	-	Weights Engineer
R. H. Heacock	-	Group Engineer, Controls and Equipment
W. J. Leas	-	Structures Engineer
J. R. Simpson	-	Group Engineer, Rotor System and Aircraft Structure
R. J. Sullivan	-	Chief, Performance and Dynamics Section

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ILLUSTRATIONS

1. SUMMARY

The design of the XV-9A Hot Cycle Research Aircraft (see Figure 1) has been accomplished in accordance with U. S. Army Aviation Materiel Laboratories Contract DA 44-177-AMC-877(T). This report includes a discussion of concepts utilized in design of the aircraft and also includes information relating to the configuration, weight and balance, performance, stability and control, dynamics, and structural characteristics of the research aircraft.

The aircraft incorporates the previously tested 55-foot-diameter Hot Cycle pressure jet rotor propelled by two gas generator versions of the YT-64 engine. The aircraft has a design gross weight of 15,300 pounds, with provision for future addition of an external payload-carrying capability to an alternate overload gross weight of 25,500 pounds. The aircraft is designed to be flown as a helicopter with a 150-knot maximum speed.

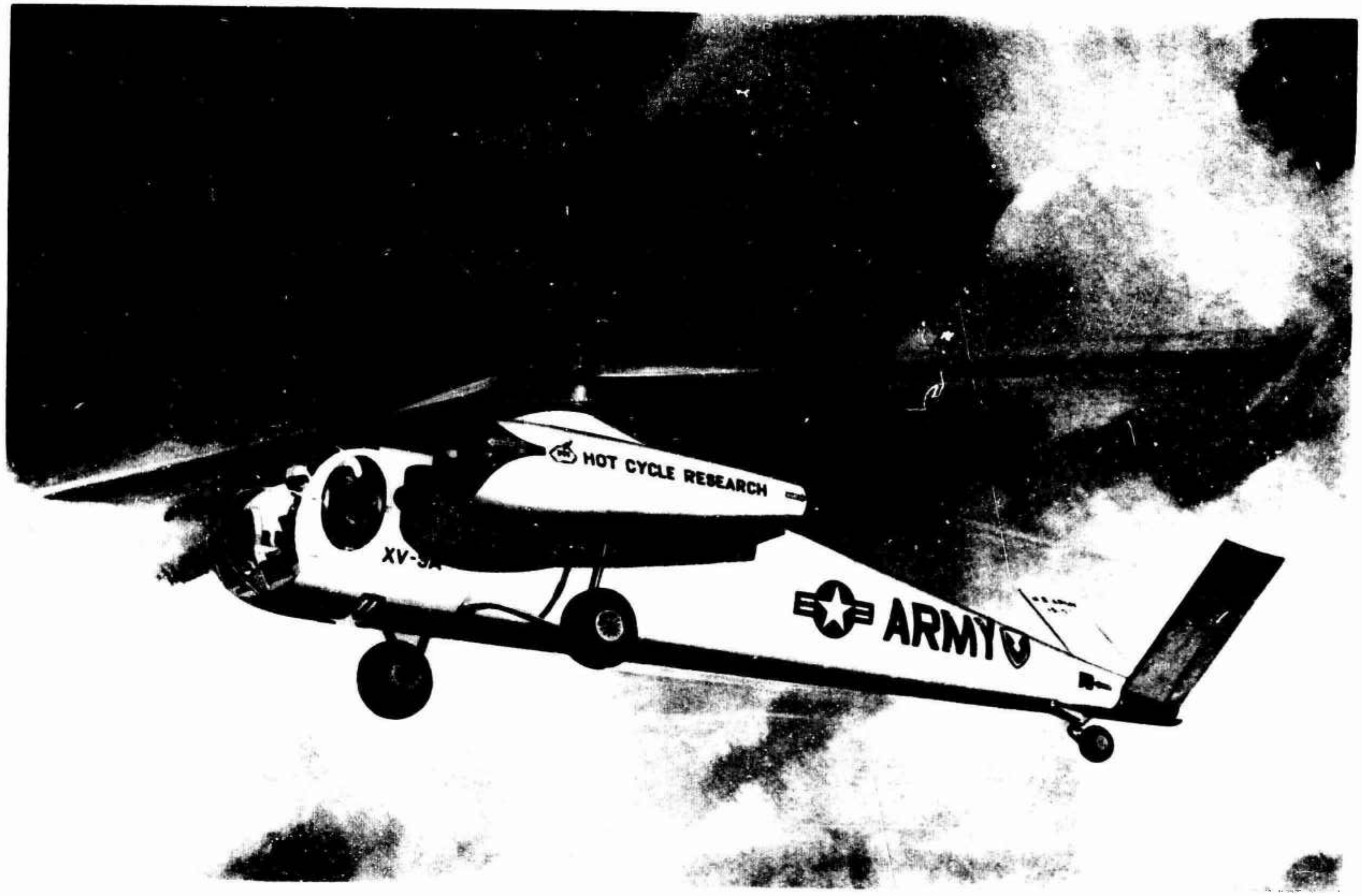


Figure 1. XV-9A Hot Cycle Research Aircraft

2. DISCUSSION

2.1 INTRODUCTION

The XV-9A Hot Cycle Research Aircraft (Hughes Model 385) has been designed under U. S. Army Contract DA 44-177-AMC-877(T). The principal objective of the contract is "to conduct a research program to demonstrate the feasibility of the Hot Cycle Rotor System through the design, fabrication, and test of one flightworthy research VTOL aircraft, incorporating the Hot Cycle Rotor System powered by two gas generator versions of the YT-64 engine." The aircraft utilizes the Hot Cycle pressure jet rotor system developed under U. S. Air Force Contract AF 33 (600)-30271, and has been designed in general accordance with the configuration established by the Preliminary Design Reports, References 1 and 2, prepared under U. S. Army Contract DA-44-177-TC-832 and with the Model Specification, Reference 3. The general arrangement of the aircraft is shown in Figure 2.

The XV-9A Hot Cycle Research Aircraft pressure jet rotor is driven by hot gases produced by two YT-64 gas generators. The gas generator exhaust gases are ducted through diverter valves, stationary ducts, a trifurcated rotating duct, and the blades to the blade-tip cascade nozzles. Due to the absence of significant rotor drive shaft torque, no tail rotor is required. A jet reaction yaw control valve, mounted at the aft end of the fuselage, is powered by the gas generator exhaust and will supply required stabilizing yaw force during hover and low-speed forward flight. Aerodynamic control surfaces will be used for yaw control at higher forward flight speeds.

The work to be done under the contract includes the design described in this report, whirl tests, component tests, aircraft fabrication, ground tests, and a 15-hour flight test program.

The design of the aircraft employed the simplest design and fabrication techniques consistent with the mission of the aircraft. Off-the-shelf components, such as J-85 diverter valves, CH-34A landing gear, and OH-6A cockpit section, were used wherever possible in order to reduce design complexity and to improve aircraft reliability.

The design incorporates a separable structural unit, known as the power module, containing the rotor system support structure, the propulsion system, the hydraulic system, and their attendant accessories. The use of the power module concept both simplified and improved whirl testing and, in addition, provided for accumulation of maximum experience, prior to flight, on the most complex portion of the aircraft.

The discussion of the design is broken into seven major areas covering aircraft characteristics, rotor system, aircraft structure, propulsion and gas transfer systems, controls system, aircraft equipment, and aircraft safety.

2.2 AIRCRAFT CHARACTERISTICS

2.2.1 Weight Summary

Empty weight	8,641 pounds
Design minimum gross weight	10,000 pounds
Design gross weight	15,300 pounds
Alternate overload gross weight (external cargo)	25,500 pounds

2.2.2 Performance

Condition	Gross Weight (Pounds)	Altitude and Temperature	Speed (Knots)
Helicopter maximum speed	15,300	SL Standard	140
	10,000	SL Standard	150
Helicopter maximum dive speed	15,300	SL Standard	200
	10,000	SL Standard	200

2.2.3 Rotor Characteristics

Number of blades	3
Rotor radius	27.6 feet
Blade area (total three blades)	217.5 square feet
Disc area	2,392 square feet
Rotor solidity	0.091
Blade chord	31.5 inches
Blade airfoil	NACA 0018
Blade twist	-8 degrees

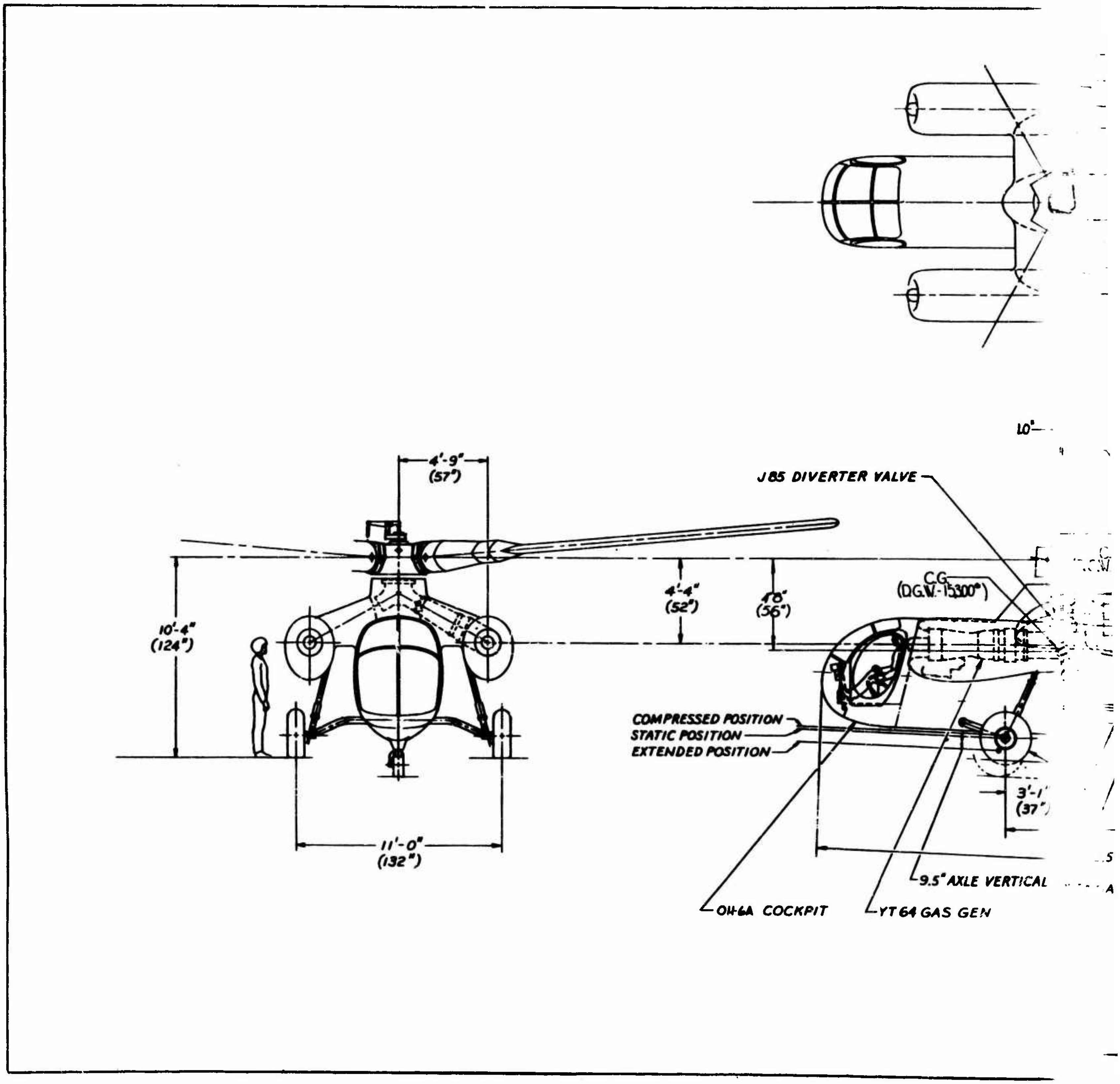
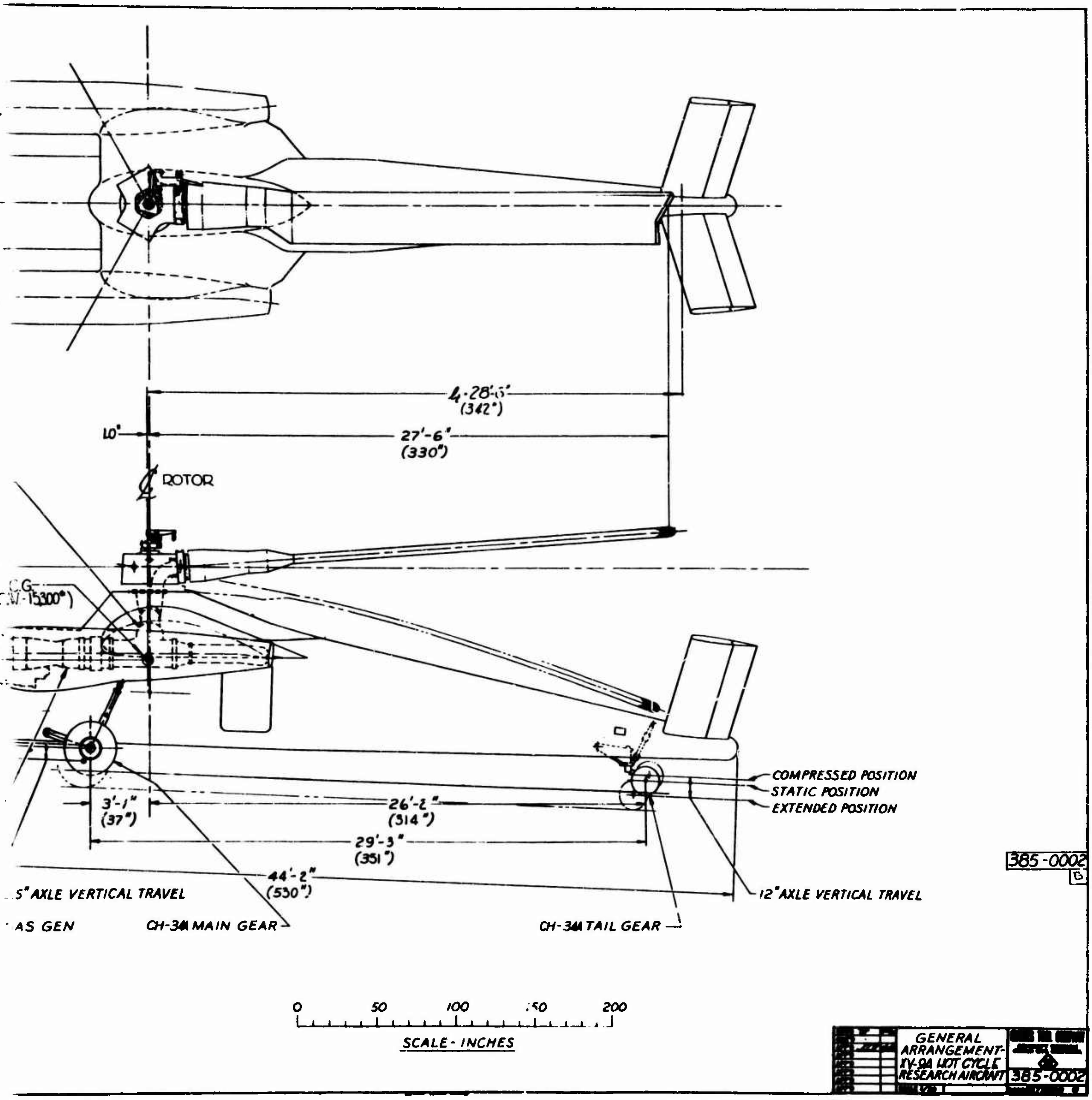


Figure 2. General Arrangement


28'-6"
(342")
27'-6"
(330")
10"
℄ ROTOR
C.G.
COMPRESSED POSITION
STATIC POSITION
EXTENDED POSITION
3'-1"
(37")
26'-2"
(314")
29'-3"
(351")
44'-2"
(530")
AXLE VERTICAL TRAVEL
12" AXLE VERTICAL TRAVEL
AS GEN
CH-34A MAIN GEAR
CH-34A TAIL GEAR
385-0002
B
0
50
100
150
200
SCALE - INCHES
GENERAL ARRANGEMENT- XV-9A HOT CYCLE RESEARCH AIRCRAFT
385-0002

B

Hot gas ducts

Number of ducts per blade	2
Total duct area per blade	54.8 square inches
Duct utilization $\frac{\text{duct area}}{\text{blade cross section area}}$	0.451
Tip nozzle area per blade (closure valve open)	37.5 square inches

2.2.4 Rotor Speed

	rpm	V_{tip} (fps)
Design operational, power-on or power-off	243	700
Design minimum, power-on	225	648
Design maximum, power-on (red line)	255	734
Design minimum, power-off	225	648
Design maximum, power-off (red line) (1.1 x maximum power-on rpm)	280	807
Rotor speed, limit, power-on or power-off (1.1 x maximum power-on rpm) x 1.05	295	848

2.2.5 Powerplant

YT-64 gas generators -- Government furnished -- 2 required.

The structural design criteria of Section 7.2, the preceding performance characteristics, and the gas conditions shown below are based on exhaust gas conditions of gas generator versions of T-64-GE-6 engines as defined by References 4 and 5.

	Temp (^{o}R)	Temp (^{o}F)	Pressure Ratio	Pressure (psig)	Mass Flow (lb/sec)
Maximum	1,643	1,183	2.87	27.5	24.6
Normal	1,575	1,115	2.60	23.5	23.0

2.2.6 Empennage

Area (true) (total)	54.00 square feet
Dihedral	45.0 degrees

Sweep	7.5 degrees
Incidence (with respect to rotor shaft)	1.0 degree ± 5.00 degrees adjustment
Chord	3.50 feet
Span (true)	15.40 feet
Aspect ratio (geometric)	4.35
Airfoil	NACA 0012
Rudder chord (37.5 percent, including overhang)	1.31 feet
Rudder span (true)	15.40 feet
Rudder area (true)	19.9 square feet
Rudder deflection	± 20.0 degrees

2.2.7 Overall Dimensions

Aircraft length (rotor turning)	59.7 feet
Fuselage length	44.17 feet
Tread of main wheels	11.00 feet
Height (to top of rotor hub)	12.40 feet
Width (across lateral pylons)	12.20 feet

2.2.8 Maximum Control Displacement

Cyclic control	
Longitudinal cyclic pitch travel	± 10 degrees
Longitudinal cyclic stick travel	13 inches, total
Lateral cyclic pitch travel	± 7 degrees
Lateral cyclic stick travel	12 inches, total
Collective	
Collective pitch travel (75 percent radius)	0 degrees to 12 degrees
Collective stick travel	7.5 inches
Rudder pedal (from neutral)	
Full left	3.0 inches
Full right	3.0 inches
Rudder deflection (± 3.0 inches at pedal)	± 20 degrees

2.3 ROTOR SYSTEM

The XV-9A rotor system consists of the Hot Cycle pressure jet rotor system fabricated and tested under U. S. Air Force Contract AF 33(600)-30271. The rotor system has been modified in accordance with the results of that test program. The three-bladed rotor system shown schematically in Figure 3 consists of a free-floating hub and three coning blades mounted on a shaft that is supported by an upper radial bearing and by a lower thrust bearing. A detailed discussion of the basic rotor system development and design may be found in Reference 6. A review of the overall system is presented below.

2.3.1 Blade Construction

The blade design incorporates two laminated steel spars (replacing the previously used solid titanium spars) running from the blade root to the tip, and separated chordwise by eighteen identical sheet metal duct segments. The segments are bolted to the spars and are joined together by bellows-type flexible couplings riveted to the outer skins. The ducts and skins of adjacent segments are slip-jointed. In this structural arrangement the spars are the only members that react to normal blade bending loads and centrifugal loads. Torsional and chordwise shear loads are carried by the assembly of segments.

2.3.1.1 Blade Constant-Section Segments

Blade constant-section segments are sheet metal assemblies consisting of two ducts contained within nine ribs and outer skins. Each segment is 12.50 inches spanwise and 15.00 inches chordwise. The ribs are die-formed with flanges matching the airfoil and duct contours. The ducts and the inner edges of the ribs are subjected to the full gas heat of the power system. René 41 alloy sheet was chosen as the material for these parts. The ribs and ducts were formed and spot welded together as a subassembly while in the solution heat-treated condition. This subassembly was then age hardened for maximum strength, and the segment was completed by spot welding outer skins of Type 301 corrosion-resistant steel sheet.

2.3.1.2 Flexible Couplings

At each joint between segments of the rotor blade there is a bellows-type flexible coupling riveted to the outer skins. This coupling performs a number of functions. It provides a pressure-tight

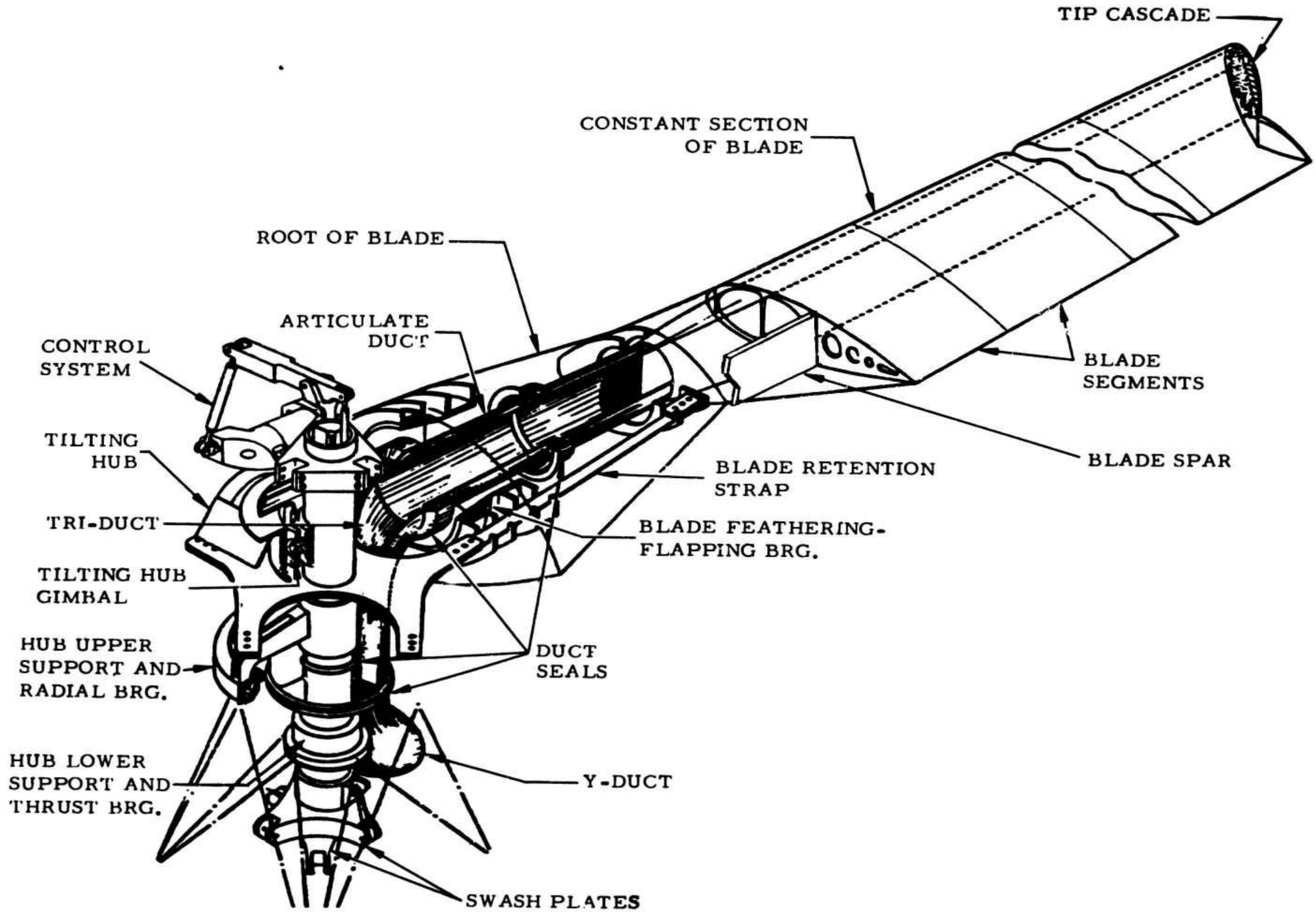


Figure 3. **Rotor System**

enclosure around the duct slip joints; it absorbs the thermal expansion and centrifugal load deflection differences between the segments and the spars; it transfers torsional and chordwise shear loads from segment to segment; and it incorporates a high degree of flexibility into the assembly of segments so that no appreciable blade bending loads are carried by any part of the structure other than the spars. This coupling is made up of two identical Inconel X drop hammer stampings welded together at the centerline of the blade, which is the point of minimum cyclic stress. The welded assembly is heat treated and glass peened for maximum fatigue strength. The riveted joints connecting the blade segments are sealed against gas leakage using Dow Corning Silastic RTV 601.

2.3.1.3 Blade Trailing Edge Segments

Interchangeable trailing edge segments of the blade are conventional sheet metal assemblies consisting of four ribs, a skin, and a spar-type channel section tying all members together at the forward end of the assembly. The channel section also functions as one wall of a tunnel for air flow to cool the blade rear spar during rotor operation. The segment is assembled by means of bonding. Skins of adjacent segments are slip jointed.

2.3.1.4 Blade Leading Edge Fairings

Leading edge fairings are identical roll-contoured sheets of Type 301 corrosion-resistant steel, each as long as a blade segment. Adjacent fairings are slip jointed. Additional roll-contoured sheets are attached internally to the fairings, for adjustment of blade chordwise balance.

2.3.1.5 Blade Root Structure

The blade root structure is made up of skin-covered ribs, frames, and webs that are bolted to the spars. As in the blade constant section, the spars are the only members reacting to normal bending loads. This is accomplished by dividing the root structure into seven sections, joined together with six electroformed nickel frames of hat-type cross section that readily deflect under bending loads. Torsional and chordwise shear loads are carried by the frames from section to section.

2.3.1.6 Blade Bearings

A feathering-flapping bearing is located on the inboard end of the blade root structure. This bearing consists of a chromeplated aluminum cast ball rotating in a teflon-lined ring attached to the hub. An opening through the ball provides clear passage for the hot gas ducts connecting the hub and blade ducts.

2.3.2 Rotor Hub and Shaft Assembly

The rotor hub and shaft assembly forms the central pivot for the rotor. Each blade is attached to the hub by a pair of tension strap packs. The free-floating hub ties the three rotor blades together and transfers the total resultant load to the shaft. The shaft in turn transfers the rotor load through an upper and lower bearing into the stationary supporting structure. The hub and shaft assembly provides support for the rotating portion of the control system. In general, 4130 or 4340 steel is used in the fabrication of the components.

2.3.2.1 Hub Structure

The free-floating hub structure is composed of a central hexagonal box with three sets of two vertical parallel beams extending from the hexagon to support the feathering bearing housings and pairs of blade retention strap shoe fittings. The radial strap loads from the three blades are balanced across the lower surface of the floating hub structure by two parallel plates. Vertical components of the strap loads are transferred from the shoe fittings through the parallel beams to the hexagonal box. The free-floating hub is gimbal-mounted at the upper end of the rotating shaft. The gimbal clevis transfers all hub loads to the shaft through the gimbal assembly.

2.3.2.2 Hub Tilt Stop

A hub tilt stop is provided for two separate conditions: two-degree tilt for low rpm and ground handling, and nine-degree tilt for normal flight maneuvers. The two-degree stop condition applies while the rotor is stationary and until it reaches 150 rpm . The stop mechanism is an overcenter linkage actuated by centrifugal forces on a weighted arm with a spring return. Above 150 rpm, the two-degree stop becomes disengaged, permitting nine degrees of tilt. As the rotor slows, the two-degree stop again engages at 90 rpm.

2.3.2.3 Blade Droop Stop

The blade droop stop is located at the lower inboard face of the blade structure and contacts the lower outboard face of the feathering bearing housing. The stop has two roller bearings with the surfaces ground to a 12-inch radius to provide for misalignment as the rollers contact the hub plate during the total blade feathering range without change in blade coning angle. Droop stop loads from a single blade are transferred through the feathering bearing support ring into the hub, where they are balanced by loads from the other two blades or are transferred by the tilt stop system into the mast.

2.3.2.4 Shaft Support

The rotating shaft is supported by two bearing assemblies. A lower bearing resists all of the vertical or thrust load. Moments are resisted by radial reactions on this bearing and on an upper bearing that is free to float vertically. The upper bearing outer housing is supported by three radial spokes attached to the shaft. The inner housing of the upper bearing is attached to one of the rotor support trusses. The lower bearing housing is attached to a similar support truss.

2.3.2.5 Upper Bearing

The upper bearing is a cylindrical roller bearing that can resist radial loads only. A circulating oil system is provided to ensure optimum lubrication and cooling (see Section 2.3.4.10).

2.3.2.6 Lower Bearing

The lower support bearing assembly consists of two tapered roller bearings mounted back to back. It carries all the vertical load and those radial loads due to moments. The bearing has a circulating oil system (see Section 2.3.4.10) coupled to the upper bearing lubrication system.

2.3.2.7 Hub Cooling

By using an air seal between the floating hub and the rotating race of the upper bearing, air is drawn through the hub by centrifugal pumping of the rotating blades. This air moves through the hub from three directions (down through the gimbal assembly, up between the mast and duct, and up inside the upper bearing stationary race) and flows outward through the feathering bearings, over the articulate ducts, and is exhausted at approximately blade section 60.00 (see Figure 4).

2.3.3 Ducts and Seals

The ducts receive gas from the gas generators and provide a passage through the free-floating hub and along the entire length of the blade to the tip cascades. A schematic of the duct system is shown in Figure 5. The hub portion of the ducting has been redesigned to reduce weight, and is discussed in Section 2.3.4.7. Blade ducts and seals are discussed below.

2.3.3.1 Blade Ducts

The duct from Station 15.50 to Station 42.50 is articulated to allow for hub float and blade coning. The inboard end of this duct is supported by a gimbal using Fabroid bearings for the coning motion and flexures for the chordwise motion. At the outboard end of the duct, freedom of motion is required, and the design of this point is discussed in Section 2.3.3.3. From the articulate duct inboard seal to Station 60.50 the duct is circular and made of Type 347 corrosion-resistant steel.

From Station 60.50 to 91.00 a transition duct starts with a circular shape at the inboard end and progresses to two roughly elliptical openings at the outboard end. Due to the noncircular shape, a relatively high-strength alloy, Inconel X, is used for this duct.

From Station 91.00 outboard to the blade-tip cascades the duct is contained in the constant-section segments, discussed in Section 2.3.1.1.

2.3.3.2 Carbon Seals

Carbon, with no supplemental lubrication, is used as the sealing material for rotating joints in the hub ducts. These seals are shown schematically in Figure 5. In the hub duct outer seal, two rows of carbon segments are held against the rotating duct by two garter springs. A wave spring holds the carbon segments against the surface of the seal housing. Gas pressure aids the springs in maintaining a tight seal.

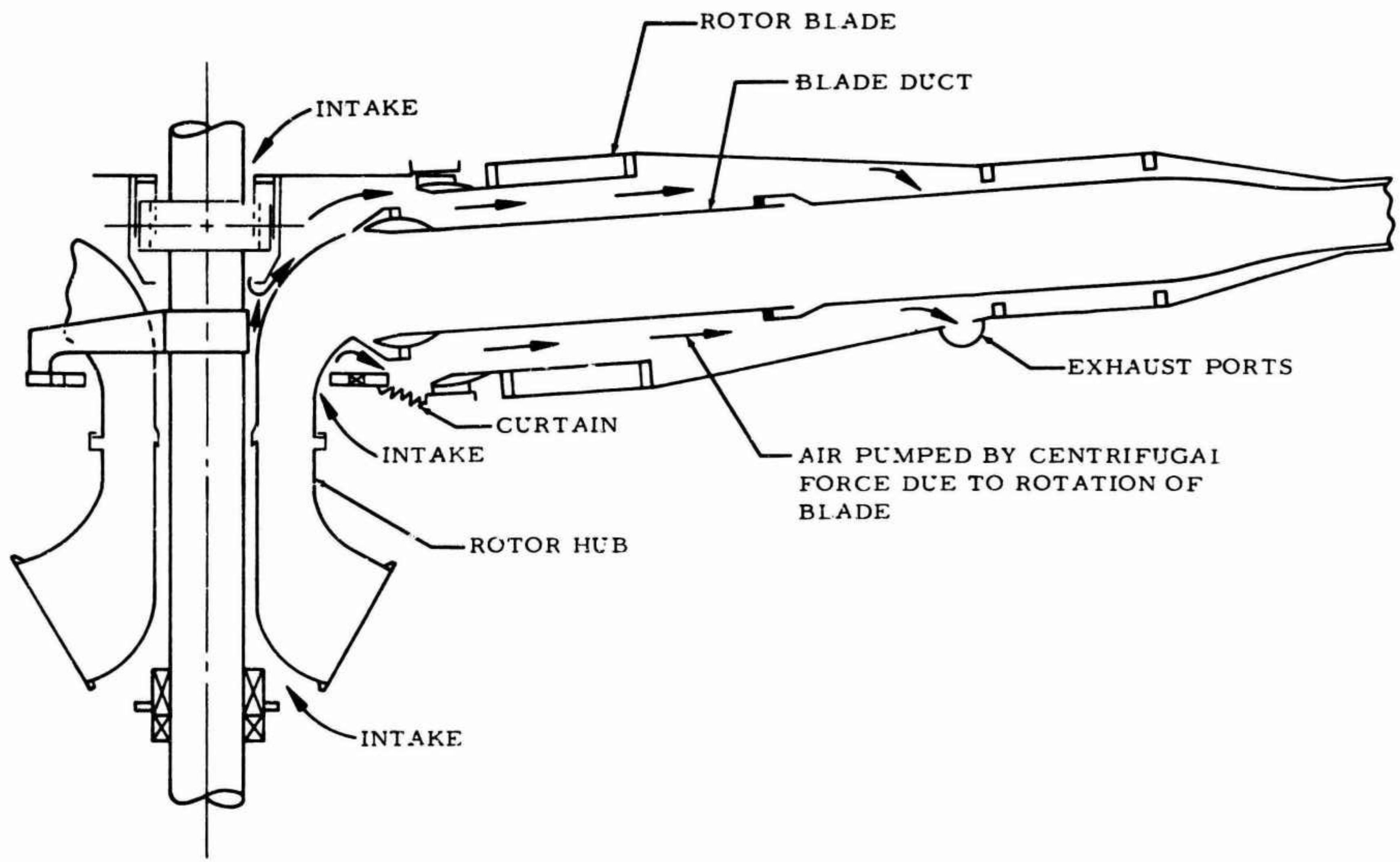


Figure 4. Rotor Hub and Blade Cooling

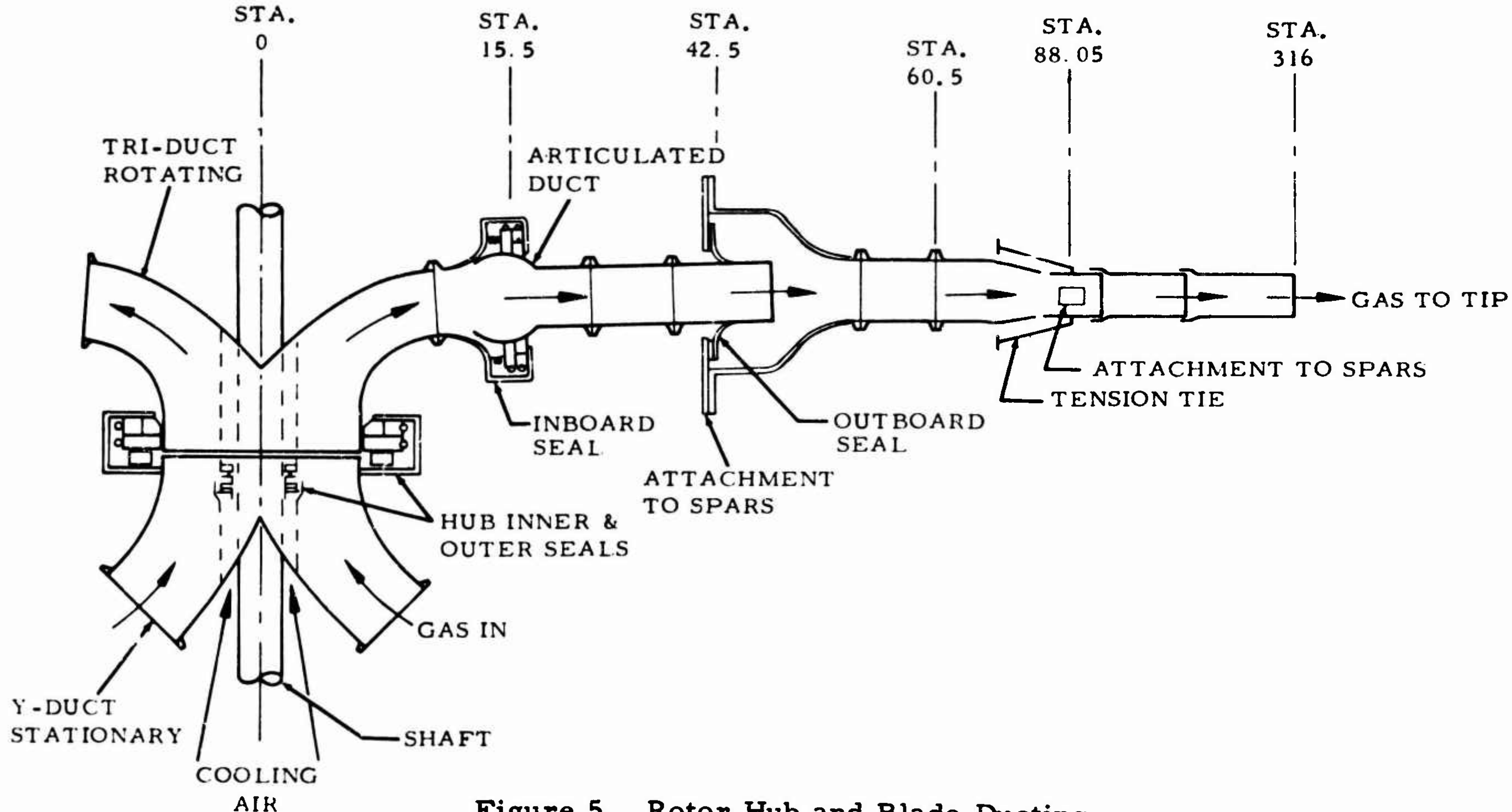


Figure 5. Rotor Hub and Blade Ducting

The hub duct inner seal uses a carbon face seal at the rotating face and two rows of carbon segments supported by two garter springs and a wave spring for the static seal, as shown in Figure 5. The seal allows relative movement between the upper (rotating) and the lower (stationary) duct without separation occurring at the face seal.

The articulate duct inboard seal configuration is approximately the same as that of the hub duct outer seal described above.

2.3.3.3 Articulate Duct Outboard Seal

The articulate duct outboard seal must seal against axial movement (due to hub float and blade coning), rotation (due to blade feathering), misalignment (due to hub float and blade coning), and side impact (due to a change from small positive to negative blade coning angles). Because of the necessity to carry side load and to accept reversal of loading, this seal consists of a nest of three slotted lip laminations riding on a tungsten carbide coated cylinder. Each lamination is formed from René 41 alloy. Slots in the laminations are staggered to eliminate continuous paths through which gas could leak. Two thicker overload leaves were added, at the top and bottom only.

2.3.4 Changes to the Rotor System

In preparation for flight testing of the XV-9A, various design revisions and additions to the existing rotor system have been accomplished, based on previous whirl test results. The major revisions to the rotor system consisted of the following:

a. New blade-tip cascades incorporating closure valves were designed.

b. Hub gimbal lugs were strengthened for in-plane loads, and thrust bearings were added to provide a direct load path for these loads.

c. Laminated steel spars have replaced the solid machined titanium spars previously used.

d. New blade retention straps have been provided.

e. Reinforced articulate duct clamps have been provided.

f. The stationary swashplate weight has been reduced.

g. The lower (stationary) Y-duct and the upper (rotating) triduct have been redesigned to decrease weight.

h. The rotor shaft and the radial bearing support spoke have been redesigned to increase strength. The shaft has also been redesigned to incorporate an output sprocket for driving the accessory gearbox.

i. A rotor accessory gearbox has been added.

j. The rotor lube system has been redesigned to provide a compact, flight-type, rotor-driven system.

2.3.4.1 Blade-Tip Cascades

The tip cascades previously used were welded assemblies of contoured sheet metal parts, consisting of two elbow ducts faired into the ducts of the blade segments, four hollow-section airfoil turning vanes in each duct, and an outer cover faired into the skins of the blade. The leading and trailing edges of the assembly each incorporated a discharge orifice for the centrifugally pumped air used to cool the spars during rotor operation. The components of the cascade were joined by Heliarc and spot welding.

The need for a blade-tip closure valve to provide the Research Aircraft with a single-engine flying capability were established in Reference 1. A review of various design approaches and a selection of the blade-tip closure valve was made, and was discussed in References 1 and 2. The design selected for installation on the aircraft was one in which the blade-tip cascade incorporates the closure valve. The following criteria were used in designing the blade-tip closure valve:

a. Exit area open, 110 square inches (total - 3 cascades)

b. Exit area closed, 55 square inches (total)

c. Maximum actuator pressure, 3,000 psig

d. Maximum temperature, 1, 200 degrees F

e. Actuation time, 0. 50 second (maximum)

The new blade-tip cascade and closure valve assembly employs three turning vanes per duct, instead of the four vanes used on the original blade-tip cascades. One of these vanes is the blade-tip closure valve, which can be rotated to close off one-half the exit area for single-engine operation. The material used for the tip-cascade assembly is Inconel 718. Actuation of the closure valve is accomplished by a pneumatic cylinder driving a series of push rods and bellcranks (see Figure 6).

The blade-tip closure valve actuators are energized with air supplied through tubing attached to the rotor forward spar. Two reservoir bottles, a fill valve, a gage, and a three-way control valve are mounted on the hub for a completely self-contained system. The schematic is shown in Figure 7. Operation of the three-way control valve is explained in Section 2. 5. 6. 2.

The supply tube to each actuator is a 1/8-inch OD stainless steel tube attached to the forward face of the rotor front spar. The tube is attached to one spar attach bolt on each blade segment. The tube is convoluted to avoid the excessive thermal and centrifugal stresses supported by the spar itself. These tubes are joined to the hub-mounted manifold by 1/8-inch OD stainless steel tube flexures, which are formed by putting five spring-like turns near each end of the tube.

The 3000-psi reservoir, consisting of two 50-cubic-inch bottles, contains sufficient gas to operate the blade-tip closure valves seven times without recharging. Only two cycles (one for maintenance and one for actual use in case of an emergency) are required per flight.

2. 3. 4. 2 Hub Gimbal Bearing Reinforcement

The hub gimbal system installation consists of a gimbal clevis, ring, and shaft trunnion. The design of the existing hub gimbal assembly was revised to provide for reinforcement of the hub gimbal lugs, in order to increase the strength of the hub gimbal system for in-plane loads, and by the addition of thrust bearings, to provide a direct load path for in-plane loads.

2.3.4.3 Laminated Steel Spars

The solid machined titanium spars used previously on the blades have been replaced by laminated steel spar assemblies. The new spars are made of laminations of AM355CRT stainless steel bonded together and bonded to a machined AM355CRT spar root fitting. The spar cross sectional area is tapered by dropping laminations off before reaching the outboard end. Great care was taken in the preparation of edges and holes in the material, to preclude the development of fatigue cracks. The laminations are bonded together and to the root fitting. Bonding is accomplished in a vacuum bag enclosed fixture and inside a 350-degree-F, 100-psi autoclave. The spar assemblies are then bolted to the blade segments and blade root sections. A shim of low-friction material is installed between the spars and the blade segments, to prevent fretting of the contacting surfaces. During rotor operation, spars are cooled by centrifugal movement of air outboard along the spars and through discharge orifices in the blade-tip cascades.

2.3.4.4 Blade Retention Straps

The chordwise natural frequency of the blades has been increased to move it further from the operating frequency range by design of new blade retention straps incorporating increased stiffness. The revised design provides for two strap packs per blade of 22 AM355CRT stainless steel laminations each and of increased width, in lieu of the original strap packs of 20 Type 301 stainless steel laminations each.

2.3.4.5 Reinforced Articulate Duct Clamps

Reinforced articulate duct clamps were designed and fabricated to Hughes Tool Company specifications. The reinforced clamps were required to eliminate duct leakage during severe maneuvers.

2.3.4.6 Redesigned Stationary Swashplate

The stationary swashplate has been redesigned to reduce weight and to provide for revised flight-type hydraulic rotor control actuators. A saving of approximately 30 pounds was realized by this change.

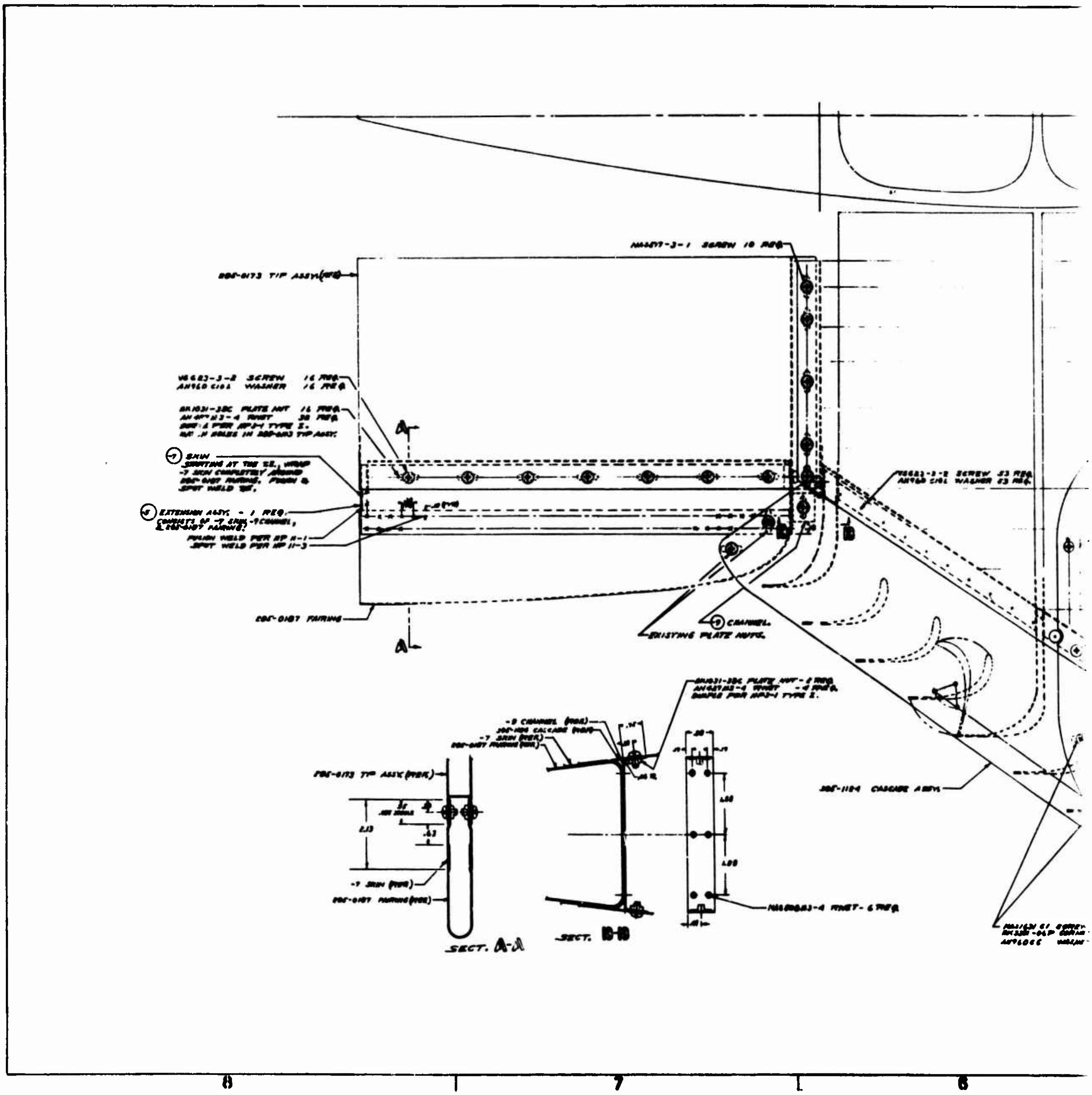


Figure 6. Blade-Tip Closure Valves

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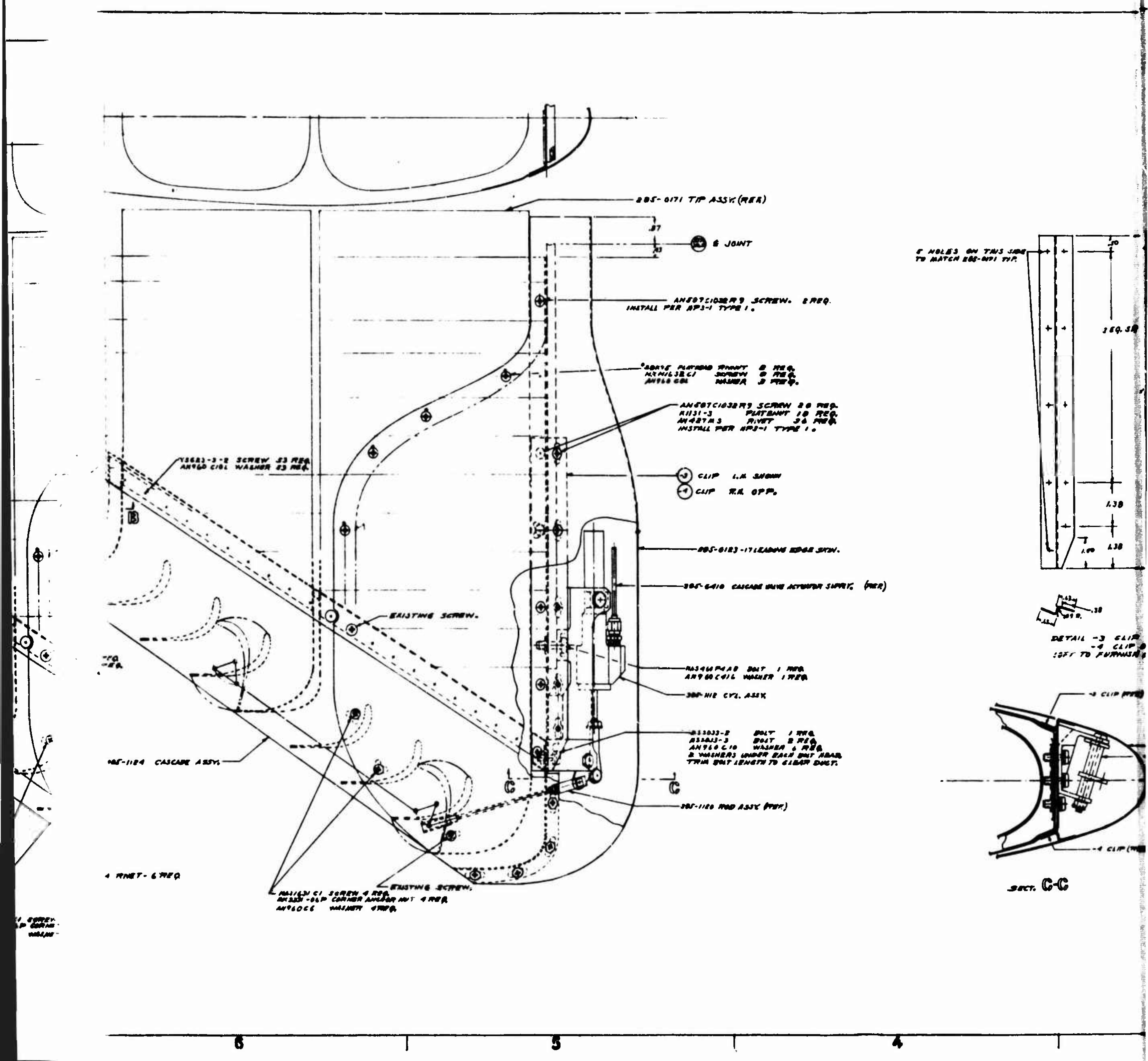

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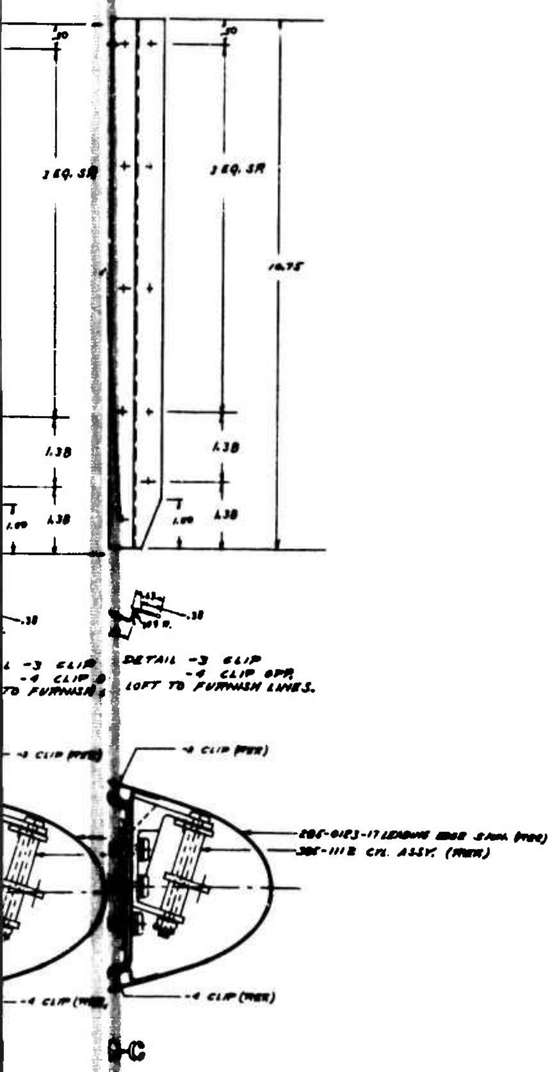

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			18	BR1031-3DC	PLATE NUT		KAYLOCK		3.8
			69	AN960C10L	WASHER				4.7
			69	V1683-3-8	SCREW		VOI SHAN		4.7
			8	AN960C6L	WASHER				
			8	MRH16EC1	SCREW		NYLOCK		5
			8	8DN76	RIVNUT		B.F. GOODRICH		5
			36	AN427 A3	RIVET				5
			10	KN31-3	PLATE NUT		KAYLOCK		5
			30	AN507C1032R4	SCREW				5
			1	AN960C416	WASHER				5
								385-1127	
			6	AN960C10	WASHER				4.5
			8	AS2033-3	BOLT				4.5
			1	AS2033-2	BOLT				4.5
			4	NAS1631C1	SCREW				6
			4	BR3331-06P	ANCHOR NUT				6
		1		-9	CHANNEL	[illegible]	[illegible]	[illegible]	7
		1		-7	SKIN	[illegible]	[illegible]	[illegible]	8
			1	-5	RIB ASSEMBLY				8
			1	-4	CLIP	[illegible]	301 COND. SHT.	[illegible]	5
			1	-3	CLIP	[illegible]	301 COND. SHT.	[illegible]	5
		-2	[illegible]	PART NO.	[illegible]	[illegible]	DESCRIPTION	SPECIFICATION	[illegible]

LIST OF MATERIAL

INSTALLATION - CASCADE VALVE.

385-1127

NOTES: LOFT TO FURNISH ALL LINES REQUIRED.

3 2 1

C

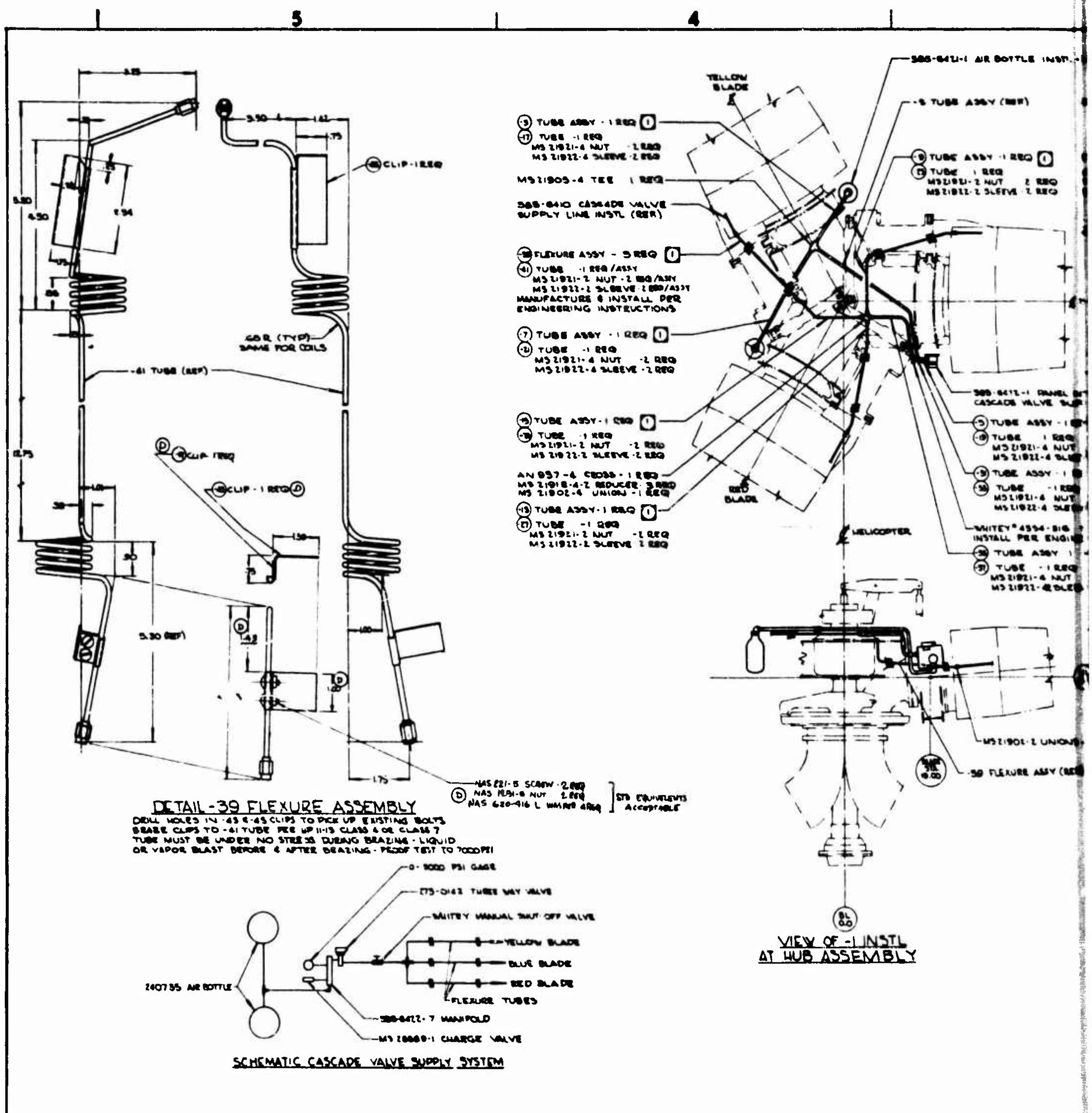
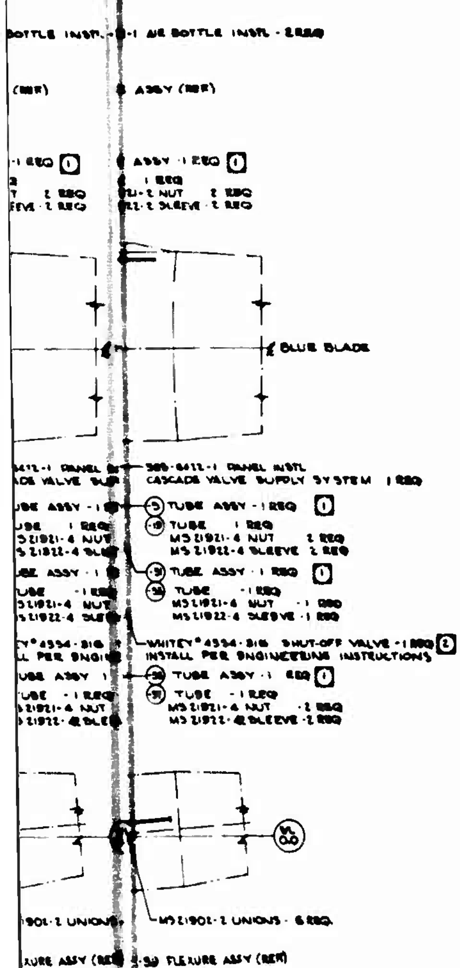


Figure 7. Blade-Tip Closure Valve Actuation System

A

REV		DESCRIPTION				
A	-	REMOVED [illegible]				
B	—	REPLACED MS 21906-4 CROSS WITH AN 917-4; MS 21916-4-2 [illegible]; DETAILED -39 FLEXURE ASSY				
C	-	ADDED [illegible] INFORMATION				
D	-	CHANGED -45 CLIP; ADDED -47 CLIP				



											1	585-6422-1	PANEL INSTL				C-3
											2	585-6421-1	AIR BOTTLE INSTL				D-3
2											1	4534-316	SHUT-OFF VALVE	WHITEY RESEARCH TOOL CO. OAKLAND CALIF.			C-3
D	2											NAS 1291-3	NUT				
D	4											NAS 620-3	WASHER				
D	2											NAS 221-9	SCREW				
		2	1					2	2	2		MS 21922-4R	SLEEVE				
	2			2	2	-	2					MS 21922-2D	SLEEVE				
		2	1					2	2	2		MS 21921-4	NUT				
	2			2	2	-	2					MS 21921-2	NUT				
											5	MS 21916-4-2	REDUCER				C-4
											1	AN 917-4	CROSS				C-4
											1	MS 21905-4	TEE				D-4
											6	MS 21901-2	UNION				B-3
D	1											-47	CLIP	1.0 x 2.0	321 STAINLESS	MIL-S-5059	C-4
	1											-45	CLIP	2.0 x 3.0	321 STAINLESS	MIL-S-5059	D-4
D	1											-43	CLIP	1.0 x 3.0	321 STAINLESS	MIL-S-5059	C-4
	1											-41	TUBE	[illegible]	TYPE 304 1/4 TUBING	MIL-T-6845	D-4
											5	-39	FLEXURE ASSY				D-4
		1										-37	TUBE	[illegible]	TYPE 304 1/4 TUBING	MIL-T-6845	C-3
											1	-35	TUBE ASSY				C-3
			1									-33	TUBE	[illegible]	TYPE 304 1/4 TUBING	MIL-T-6845	C-3
											1	-31	TUBE ASSY				C-3
				1								-29	TUBE	[illegible]	TYPE 304 1/4 TUBING	MIL-T-6845	C-4
					1							-27	TUBE	[illegible]	TYPE 304 1/4 TUBING	MIL-T-6845	C-4
	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	-1						

585-6420

D

					—							-25	— DELETED —			
						1						-23	TUBE	[illegible]	TYPE 304 1/4 TUBING	MIL-T-6845	D-3
							1					-21	TUBE	[illegible]			C-4
								1				-19	TUBE	[illegible]			C-3
									1			-17	TUBE	[illegible]	TYPE 304 1/4 TUBING	MIL-T-6845	D-4
										1		-15	TUBE ASSY				C-4
										1		-13	TUBE ASSY				C-4
										—		-11	— DELETED —				
										1		-9	TUBE ASSY				D-3
										1		-7					C-4
										1		-5					D-3
										1		-3	TUBE ASSY				D-4
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]		PART NO.					
REQUIRED PER DASH NO. ASSY														LIST OF MATERIAL			

2 SHUT-OFF VALVE IS SUPPLIED BY ENGINEERING WITH 1/4 INCH "SWAGELOK" TUBE END FITTINGS.
1 TUBE ROUTING & CLIPS TO BE DEVELOPED AT INSTALLATION.
NOTE:

INSTALLATION- CASCADE VALVE SUPPLY SYSTEM

02731

585-6420

B

2.3.4.7 Redesigned Hub Ducts

The hub two-branch stationary Y-duct and the three-branch rotating triduct previously used were made of Type 347 corrosion-resistant steel. By extensive use of drophammer formed Inconel 718 sheet, it was possible to both strengthen the ducts and effect a weight saving of approximately 105 pounds.

2.3.4.8 Redesigned Rotor Shaft and Spoke

The rotor shaft was redesigned to increase the strength and to incorporate a gear for use in driving the accessory gearbox (see Section 2.3.4.9). The three-armed spoke utilized in transmitting shaft radial loads to the upper bearing was redesigned to increase strength.

2.3.4.9 Accessory Gearbox

An accessory gearbox driven by a cogged-tooth timing belt has been added to the rotor system. Power for the gearbox is extracted from a drive gear added to the rotor shaft. The gearbox is a basic off-the-shelf unit with a 90-degree drive adaptor added. Three output pads on the accessory gearbox are used for driving the rotor lubrication system pump (see Section 2.3.4.10), the rotor tachometer generator, and the rotor speed governing system drive gearbox, which in turn drives a control system hydraulic pump and two rotor speed governing system hydraulic pumps. The accessory gearbox assembly is mounted to the rear spar of the power module.

2.3.4.10 Rotor Lubrication System

A flight-type rotor lubrication system was designed to supply circulating lubricant for the rotor upper radial bearing and the rotor lower thrust bearing. The system employs a combination pressure and scavenge pump driven by the accessory gearbox. An electrically driven scavenge pump has been provided and is used to prevent flooding of the bearings due to inadequate scavenge flow at low rotor speeds. A two-quart system reservoir is mounted in the fuselage immediately aft of the power module and has sufficient volume to permit deaeration. An oil cooler with an electrically driven blower is installed to maintain safe lubricant temperatures (see Figure 8).

2.4 AIRCRAFT STRUCTURE

The structure of the XV-9A consists of four major assemblies: the fuselage, the power module, the empennage, and the landing gear (see Figure 9). These major units of the aircraft structure are separable, to facilitate fabrication, test, and aircraft maintenance. The simplest structural design consistent with the mission of the aircraft was utilized. The structure was designed in a manner that permitted fabrication using a minimum of special purpose tools and fixtures.

2.4.1 Fuselage

For design and manufacturing purposes, the fuselage was broken down into three sections: the cockpit, the main fuselage, and the aft fuselage. The fuselage is of conventional riveted aluminum alloy, semimonocoque construction.

In order to minimize tooling and fabrication requirements, a simple fuselage cross section was chosen. The main fuselage is of constant cross section, allowing the use of identical frames and flat-wrapped skin. The aft fuselage is a truncated cone section covered with flat-wrapped skin. A transition section of compound curvature was required to fair between these two sections. In order to facilitate manufacturing, this section was kept to the minimum size.

2.4.1.1 Cockpit

The OH-6A (Hughes Model 369) cockpit structure is used on the XV-9A. It is of conventional riveted and spot welded sheet metal construction, with emphasis on lightweight structure, optimum visibility, and convenience for the pilot and copilot, seated side by side. Some structural revisions were made to accommodate the electrical system, instrumentation, rotor controls, and propulsion controls installations. Thickness of two of the windshield panels was increased to provide additional stiffness and safety.

2.4.1.2 Main Fuselage

This portion of the fuselage is of a constant cross section, matching that of the OH-6A cockpit, and extends from the cockpit aft approximately 15 feet. Structurally, this main section provides for

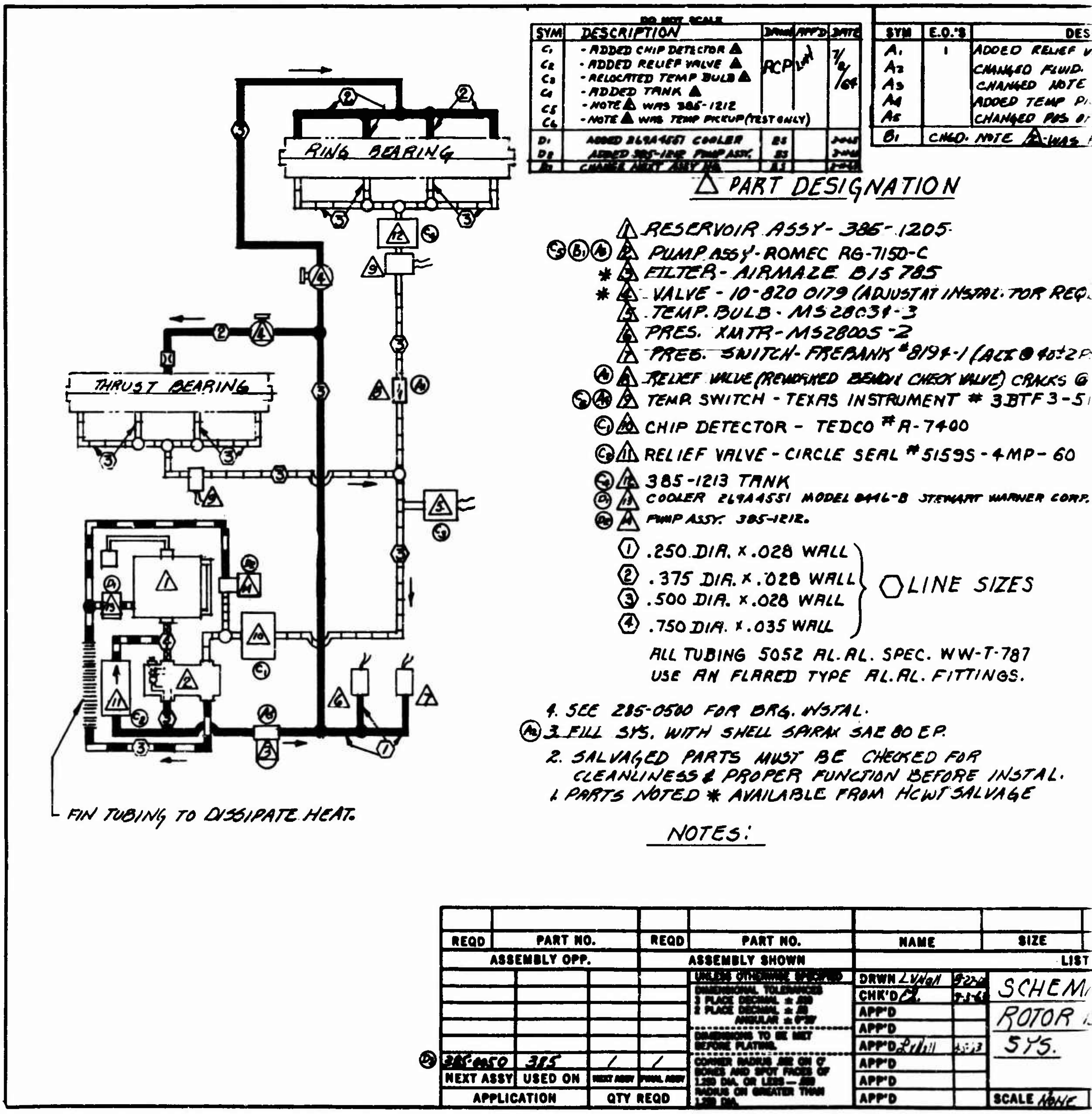
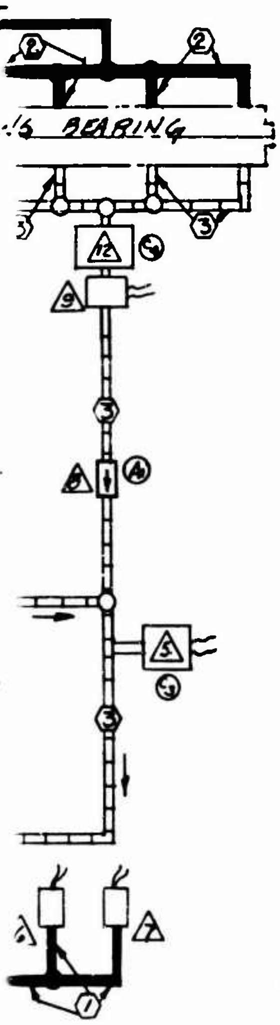


Figure 8. Rotor Lubrication System Schematic



DO NOT SCALE

SYM	DESCRIPTION	DRWN	APP'D	DATE
C1	- ADDED CHIP DETECTOR △	RCP		7/8/64
C2	- ADDED RELIEF VALVE △			
C3	- RELOCATED TEMP BULB △			
C4	- ADDED TANK △			
C5	- NOTE △ WAS 385-1212			
C6	- NOTE △ WAS TEMP PICKUP (TEST ONLY)			
D1	ADDED 269A4551 COOLER	BS		
D2	ADDED 385-1212 PUMP ASSY.	BS		
D3	CHANGED NEXT ASSY NO.	BS		

REVISIONS

SYM	E.O.'S	DESCRIPTION	DRWN	APP'D	DATE
A1	1	ADDED RELIEF VALVE △		P. Hall	3-3-64
A2		CHANGED FLUID			
A3		CHANGED NOTE △			
A4		ADDED TEMP PICK-UPS - 2 PLACES			
A5		CHANGED POS OF △ FILTER			
B1		CHGD. NOTE △ WAS ROMEC RG-7150-C	LVHall		9-24-64

△ PART DESIGNATION

- △ RESERVOIR ASSY - 385-1205
- C5 B1 A4 △ PUMP ASSY - ROMEC RG-7150-C
- * △ FILTER - AIRMAZE B15785
- * △ VALVE - 10-820 0179 (ADJUST AT INSTAL. FOR REQ'D. FLOW)
- △ TEMP. BULB - MS28034-3
- △ PRES. XMTR - MS28005-2
- △ PRES. SWITCH - FREBANK #8194-1 (ACT @ 40±2 PSI - DECREASING PRES.)
- A1 △ RELIEF VALVE (REWORKED BENDIX CHECK VALVE) CRACKS @ 9 PSID
- C3 A4 △ TEMP. SWITCH - TEXAS INSTRUMENT # 3BTF3-51
- C1 △ CHIP DETECTOR - TEDCO # A-7400
- C2 △ RELIEF VALVE - CIRCLE SEAL # 5159S-4MP-60
- C4 △ 385-1213 TANK
- D1 △ COOLER 269A4551 MODEL 8446-B STEWART WARNER CORP. WITH ELECT DRIVEN FAN
- D2 △ PUMP ASSY. 385-1212.

LINE SIZES:

- ① .250 DIA. x .028 WALL
- ② .375 DIA. x .028 WALL
- ③ .500 DIA. x .028 WALL
- ④ .750 DIA. x .035 WALL

ALL TUBING 5052 AL. AL. SPEC. WW-T-787
USE AN FLARED TYPE AL. AL. FITTINGS.

385-0051

D

4. SEE 285-0500 FOR BRG. INSTAL.
3. (A2) FILL SYS. WITH SHELL SPIRAX SAE 80 EP.
2. SALVAGED PARTS MUST BE CHECKED FOR CLEANLINESS & PROPER FUNCTION BEFORE INSTAL.
1. PARTS NOTED * AVAILABLE FROM HCWT SALVAGE

NOTES:

REQD	PART NO.	REQD	PART NO.	NAME	SIZE	DESCRIPTION	SPECIFICATION
ASSEMBLY OPP.		ASSEMBLY SHOWN		LIST OF MATERIAL			

NEXT ASSY	USED ON	NEXT ASSY	FINAL ASSY
385-0050	385	1	1
APPLICATION		QTY REQD	

UNLESS OTHERWISE SPECIFIED
DIMENSIONAL TOLERANCES
3 PLACE DECIMAL ± .010
2 PLACE DECIMAL ± .03
ANGULAR ± 0°30'
DIMENSIONS TO BE MET BEFORE PLATING.
CORNER RADIUS .032 ON C' BORES AND SPOT FACES OF 1.250 DIA. OR LESS — .060 RADIUS ON GREATER THAN 1.250 DIA.

DRWN	LVHall	8-27-64
CHK'D		9-3-64
APP'D		
APP'D		
APP'D	P. Hall	9-3-64
APP'D		
APP'D		
APP'D		

SCHEMATIC DIAG. ROTOR BRG. LUBE SYS.

SCALE NONE

HUGHES TOOL COMPANY
AIRCRAFT DIVISION
CULVER CITY, CALIFORNIA

385-0051

CODE 02731 SHEET OF

r Lubrication System Schematic

B

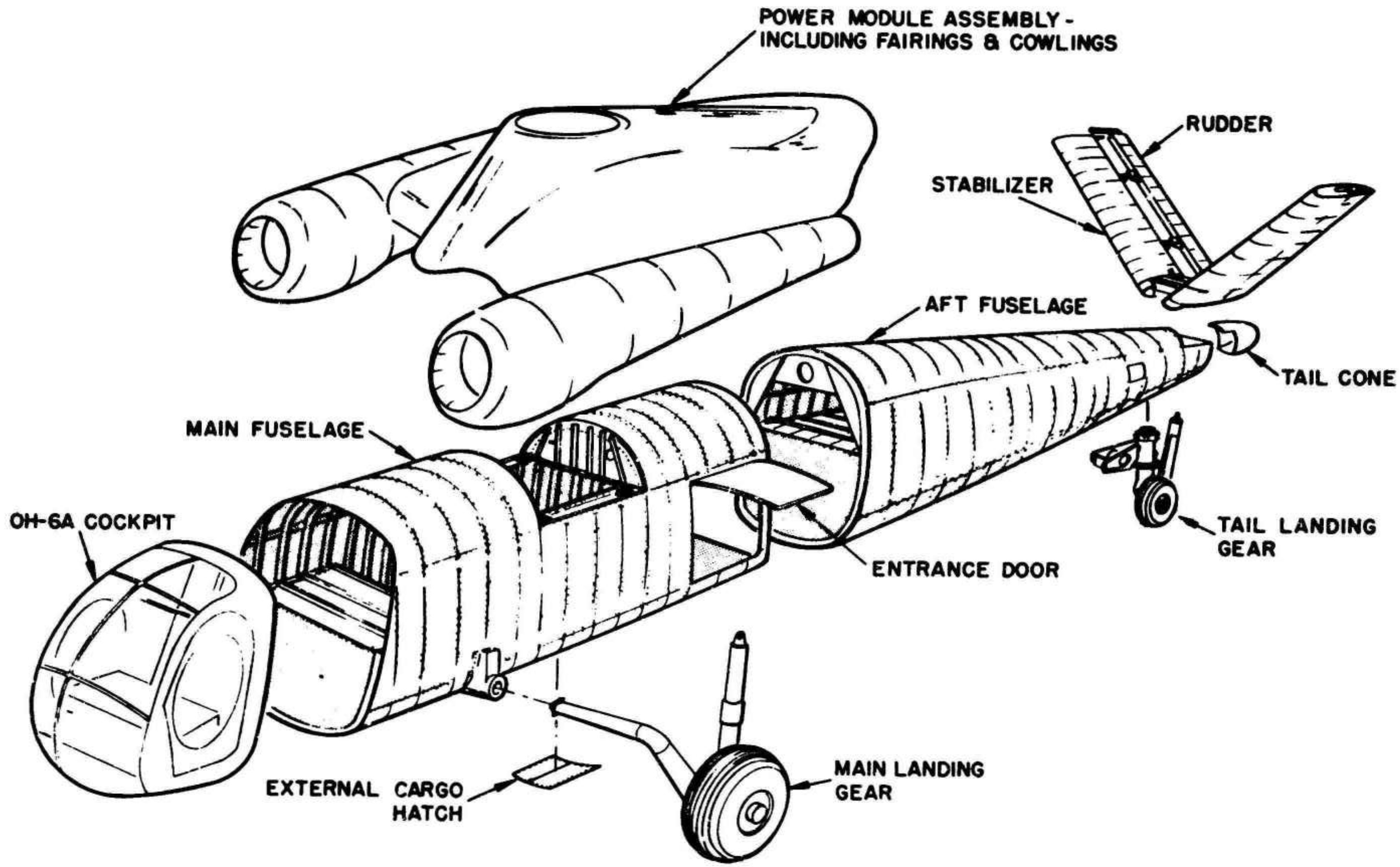


Figure 9. Aircraft Structural Components

the attachment of the power module, main landing gear, and forward fuel cell, and contains the payload compartment.

This structure is of typical longeron, skin, and frame construction. Typical frames are C-sections, approximately three inches deep, hydropressed out of 2024 aluminum alloy sheet and spaced at seven- to eight-inch intervals. Longerons are made from 7075 T6 aluminum alloy T-section extrusions, and the skins are 2024 T3 aluminum alloy sheets that vary in thickness from 0. 016 inch through 0. 040 inch. Two main frames provide for the attachment of the power module through tension fittings.

The main landing gear legs are inserted into a heat-treated steel tube that runs across the ship under the floor line. Landing gear leg bending stresses are taken across this tube, and only the resultant loads are taken by the adjacent stiffened frames and skin. In the forward end of the main fuselage section is one of the fuel cells, supported by an inner liner riveted to the flanges of the lower half of the frames. The cell end bulkheads are designed to react hydrostatic loading and are of a aluminum alloy honeycomb construction.

The remainder of the main fuselage section is floored with removable aluminum alloy honeycomb panels, and incorporates provisions for installation of flight test instrumentation equipment.

An upward swinging main entrance door (approximately 30 inches x 40 inches) is installed on the left-hand side of the fuselage and provides for access to the payload compartment. A hatch (approximately 24 inches x 24 inches) is installed directly under the rotor centerline and provides for future addition of an external payload carrying capability to the aircraft.

2. 4. 1. 3 Aft Fuselage

This section of the fuselage extends from the side entrance door aft to the tail cone fairing. Structurally, it must provide for the aft fuel cell, yaw control valve and supply ducting, tailwheel, and empennage attachment. As in the main fuselage section, this structure is made up of a continuation of the four longerons, similar C-section frames (on 10-inch spacing), and skin, riveted together. Access to the yaw control valve is through the tailwheel well. Fuel cell provisions are almost identical with those for the forward fuel cell. Bulkheads are provided for the attachment of the tailwheel and empennage.

2.4.2 Empennage

This assembly consists of stabilizers set at 45-degree dihedral and 7-1/2 degree sweep with aerodynamically and dynamically balanced rudder surfaces. The total empennage area is 54.0 square feet.

2.4.2.1 Stabilizers

The assembly consists of front and rear spars, hydro-pressed ribs, and skin, riveted together. The material is aluminum alloy throughout. The airfoil (NACA 0012) is a constant section from root to tip, greatly simplifying the structure. Sweep back and dihedral require a small amount of complexity in the root ribs and fittings. The stabilizer area is 32.4 square feet.

2.4.2.2 Rudders

The rudders are aluminum alloy structures made up of spars, ribs, and skin. They are hinged to the stabilizer by three hinges plus a torque tube support. Aerodynamic and dynamic balance are provided. The total rudder area is 21.6 square feet. Operation of the rudders is described in Section 2.6.2

2.4.3 Landing Gear

The landing gear installation consists of Government furnished CH-34A components.

2.4.3.1 Main Landing Gear

As previously noted in Section 2.4.1.2, provisions are made in the forward fuselage to permit installation of the main landing gear legs. A fitting on the power module at the intersection of the horizontal pylon front spar and both nacelles supplies the attachment point for the upper end of each oleo strut. The length of the oleo strut has been modified to adapt it for use on the XV-9A.

2.4.3.2 Tail Gear

The tail gear yoke casting attaches to a bulkhead at fuselage Station 581.00 and the tail gear oleo strut attaches to a bulkhead at fuselage Station 616.50 (see Figure 9). The tail gear assembly is fully castering for ground handling, and can be locked for flight. Operation of the tailwheel lock system is described in Section 2. 6. 4. 2.

2. 4. 4 Power Module

The power module structural assembly consists of the nacelles, the horizontal or lateral pylon, and the vertical or rotor pylon. The complete assembly is bolted to the fuselage at four points (see Figure 10). Structural requirements for this section consist of providing for engines, diverter valves, ducting, controls, hydraulic system, and auxiliary gearbox installations, in addition to providing for the rotor and the main landing gear oleo strut attachments. The high temperature environment surrounding the engine and ducting was one of the primary factors that influenced the design.

2. 4. 4. 1 Nacelles

The nacelle structure supports the engine, diverter valve, tailpipes, and tail cone. The engine mount support structure is composed primarily of welded 4130 steel tubing and A286 steel skin and formers. Nonstressed cowling panels are constructed of aluminum alloy sheet, with the exception of the one over the hot section of the engine, which is constructed of Type 347 stainless steel. The diverter valve is located between the two main nacelle frames, which are integral with the forward and rear spars of the horizontal pylon. The diverter-valve support yoke is attached to the intersection of the outboard or canted rib and the forward spar. The structure in this area is predominantly A286 steel formers, longerons, and skins. Some titanium structure is also used in this area. The stressed access door for installation of the diverter valve is made of aluminum alloy. The tail cone assemblies cantilever aft from the rear spar frames, and are made in two parts. The upper half is attached permanently to the frame and provides for the tailpipe support. The lower half is removable, to permit installation and removal of the tailpipe. The tail cone skins are titanium and the formers are Type 347 stainless steel.

2. 4. 4. 2 Horizontal Pylon

This structure attaches the nacelles and rotor support trusses to the fuselage, and also provides the reactions to large loads from the hydraulic power control cylinders. It is constructed almost entirely of aluminum alloy. The main structural members in this

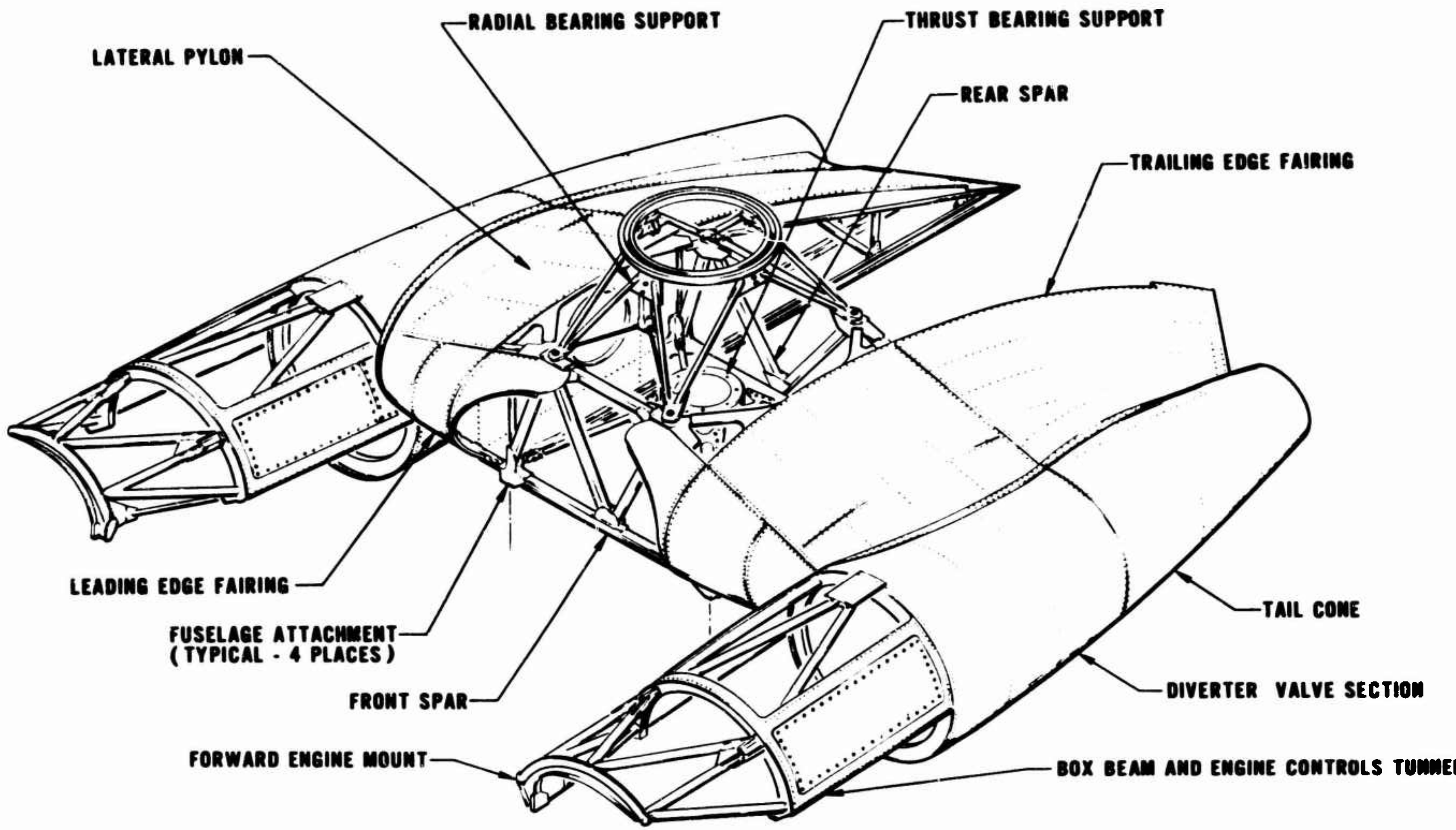


Figure 10. Power Module Structural Assembly

assembly are a front and rear spar, which run across the fuselage and terminate at their attachment to the nacelle main rings. These spars are in the form of a truss between the four fuselage attach points, and are made up of a shear web and caps between the fuselage and nacelle. Nonstructural leading and trailing edge fairings complete the horizontal pylon. The rotor is supported from the horizontal pylon by two welded steel tube trusses, one supporting the radial bearing and the other supporting the thrust bearing at the lower end of the rotor shaft.

2.4.4.3 Vertical Pylon

An aluminum alloy sheet metal assembly made up of formers and skin provides the nonstructural fairing around the rotor radial bearing truss, Y-duct, and yaw control duct. Portions of it are removable, to permit access to the hub ducting and to the auxiliary gearbox installation.

2.5 PROPULSION SYSTEM

The propulsion system installation includes those components required to provide the energy for the Hot Cycle pressure jet rotor system. The design, therefore, embraces the physical configuration of the gas generators, the attendant gas generator systems, the hot gas transfer system, the rotor system, and the helicopter jet reaction yaw control system. Description of the rotor system has been covered in Section 2.3 of this report. This section will deal with all other elements of the propulsion system.

The design features excellent inspection and maintenance accessibility, convenient gas generator and diverter valve handling, an aerodynamically clean gas generator air inlet, safety precaustions for the minimization of gas generator bay fire hazard, and, in the event of fire, complete coverage for both detection and extinguishment.

The general arrangement of the propulsion system is shown in Figure 11.

2.5.1 Thermal Characteristics

During operation of the aircraft, heat will be flowing from the hot gases into the adjoining hardware. In addition, heat will be generated in the gas generator lubrication circuit and in the hydraulic

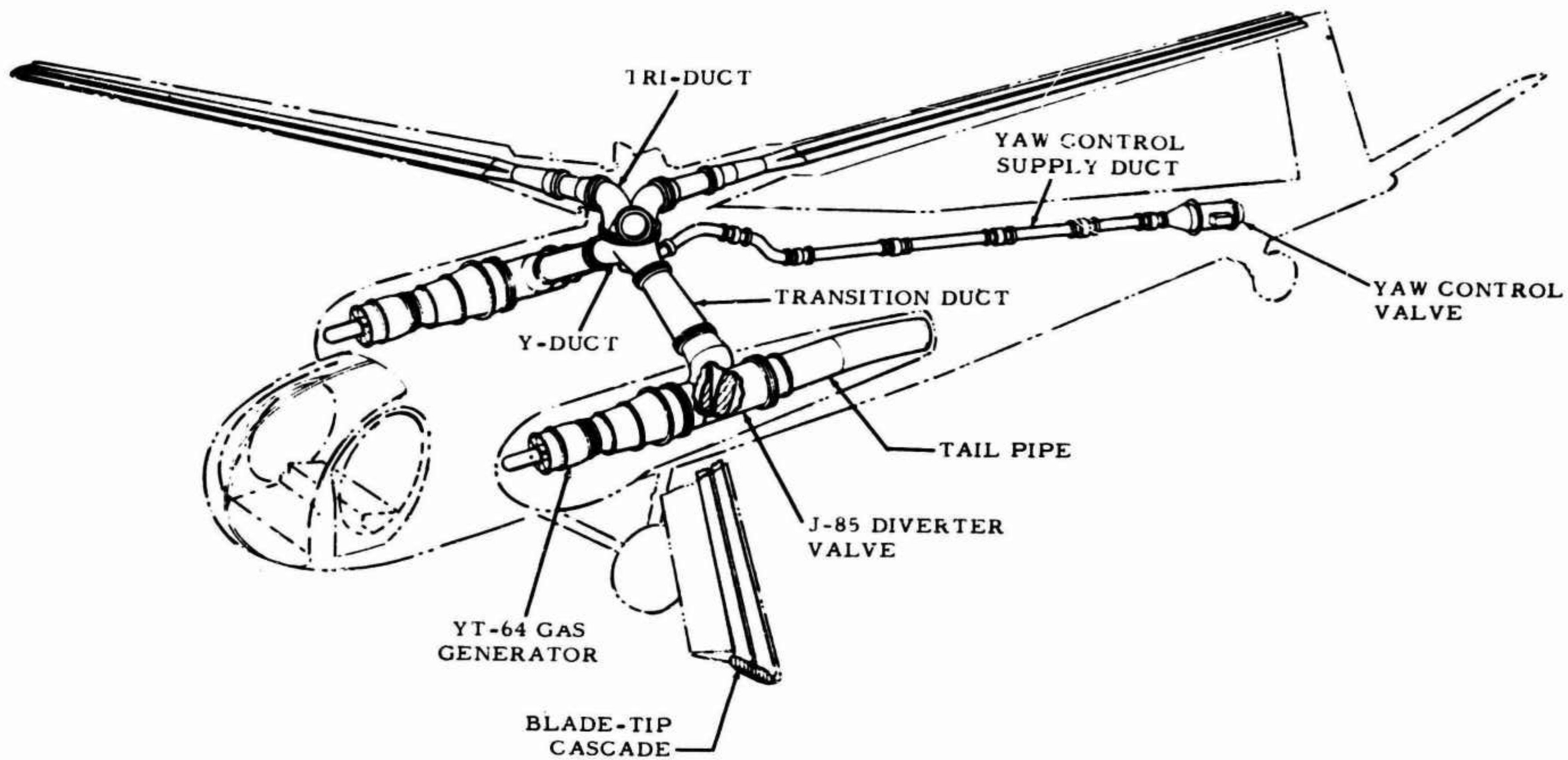


Figure 11. General Arrangement - Propulsion System

pumps and actuators. There are five areas in the aircraft having different cooling requirements: gas generator lubricant system, hydraulic fluid, gas generator bay, pylon and fuselage structure, and rotor system. These areas represent a wide range of thermal conditions and require various means to accomplish overall control of temperature. An extensive study was conducted on this subject before any particular insulation or cooling arrangement was finally selected. Simplicity of installation had high priority in these design considerations and is reflected in the selection of methods and materials concerning thermal problems. Whenever forced cooling was not readily available, one or more of the following methods were used to ensure safe operation of the system:

a. Insulation of hot ducts to reduce heat flux

b. Local protection for critical parts

c. Ventilating holes to induce air circulation

d. Leaving the area at elevated temperature and using temperature-resistant materials.

2.5.1.1 Thermal Criteria

Limit temperatures and heat transfer data are taken from the YT-64 gas generator data given in Reference 7. Additional data were supplied directly by the manufacturer.

Heat generation in the hydraulic system was estimated on the basis of system analysis and information supplied by the manufacturers of individual components.

Characteristics of heat exchangers and thermal properties of insulation were based on the test results supplied by the manufacturers.

2.5.2 Basic Powerplant

The basic powerplant is the YT-64-6 turboshaft engine modified to a gas generator for application to the XV-9A by the removal of the second or power turbine. This modified gas generator includes a 14-stage axial flow compressor, a through-flow annular combustion chamber, a two-stage axial flow turbine, a fixed area exhaust cone, and an integral control system. The gas generator control system, in turn, consists of a fuel pump and filter, a hydromechanical fuel control assembly, and a pair of compressor stator vane actuators. The first seven stages of compression include variable stator guide vanes. Angular position of these vanes is automatically controlled by the fuel control.

The quick engine change (QEC) or gas generator assembly includes the basic powerplant, the gas generator air inlet, the gas generator lubricating system, the gas generator starting system, the gas generator mounted accessories, the fuel inlet, the power control attachments, the gas generator mounts, the fire extinguishing manifold, the fire detection cabling, the gas generator flight instrumentation, the gas generator exit adaptor, and the required gas generator vents and drains. These items are discussed in the applicable paragraphs.

All gas generator mounted items such as accessories, ignition generator, ignitor, filters, and so on, can be inspected, cleaned, adjusted, removed, and/or replaced without the use of special tools or the removal of the engine or prime structure.

2.5.2.1 Accessories

The helicopter accessories mounted on and driven by each gas generator included a 6-gpm, 3,000-psi hydraulic pump, a 24-vdc, 150-amp electric generator, a tachometer generator and a governor drive hydraulic motor.

2.5.2.2 Starting System

The YT-64 gas generator features an integral air impingement starter (AIS), by way of a manifold on the turbine casing, passing compressed air from an external source to turn the gas generator turbine second stage up to starting speed. The AIS horn diameter is 1.75 inch; provision for attachment of a V-band coupling connector is included.

Each gas generator will be motored by a Government-furnished USAF Type MA-1 mobile air compressor (air at 45 psig and 360 degrees F approximately) for individual starting. Cross-bleed starting has not been considered necessary.

The hose connection from the MA-1 cart is made to each gas generator through an access door in the lower aft nacelle cowl panel by the mating of a quick-attach-detach coupling (attached to the hose) to a standard MS33740 nipple (attached to the gas generator).

Since the MA-1 supply line is 3.50 inches in diameter, a transition duct is mounted on the gas generator to reduce the diameter to the 1.75-inch diameter noted above. The duct and duct supports are fabricated of Type 347 corrosion-resistant steel, and are mounted to the forward and aft turbine flanges (see Figure 12.) Between the starter nipple and the transition duct, the system incorporates a flapper check valve. The purpose of this valve is to prevent backflow of the exhaust gas through the manifold. The flapper check valve assembly includes the MS33740 nipple upstream of the valve. It attaches to the transition duct by a V-band coupling.

2.5.2.3 Gas Generator Air Inlet

The gas generator inlet assembly is made of 6061-0 aluminum alloy formed and welded into a homogeneous unit. The complete assembly, which mounts on the gas generator compressor inlet, serves as the gas generator air inlet duct, the nacelle nose fairing, and the gas generator oil tank (see Figure 13).

The assembly consists of three formed pieces: a 360-degree inner skin forming the inlet duct and duct inlet lip, a 360-degree outer skin forming the nose fairing, and a closing frame for assembly rigidity. The inner skin and outer skin were fusion welded at the leading edge, ground, buffed, and reformed to maintain smooth aerodynamic contours. Nose ribs were used to maintain the shape. The closing frame and the nose ribs were spotwelded to the subassembly to complete the unit.

The closing frames and two solid nose ribs define the oil tank. Seam and fusion welding were used in this area. The tank cavity is designed to withstand a pressure differential of three pounds per square inch.

The gas generator inlet duct profile is sized to provide efficient inlet recovery for both hovering and forward flight operations. The outer nose fairing is contoured for low drag at the higher forward speeds.

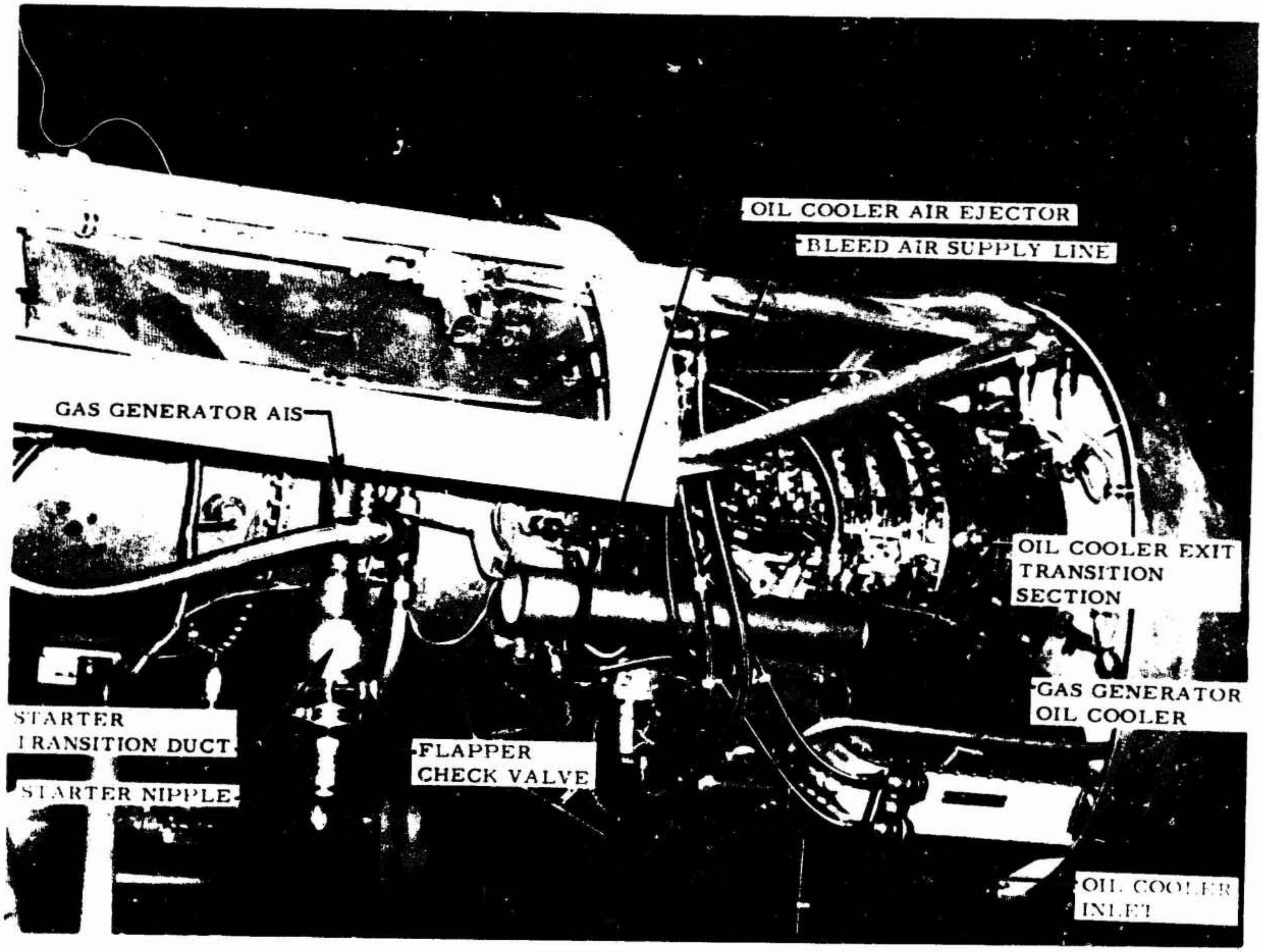


Figure 12. Gas Generator Installation

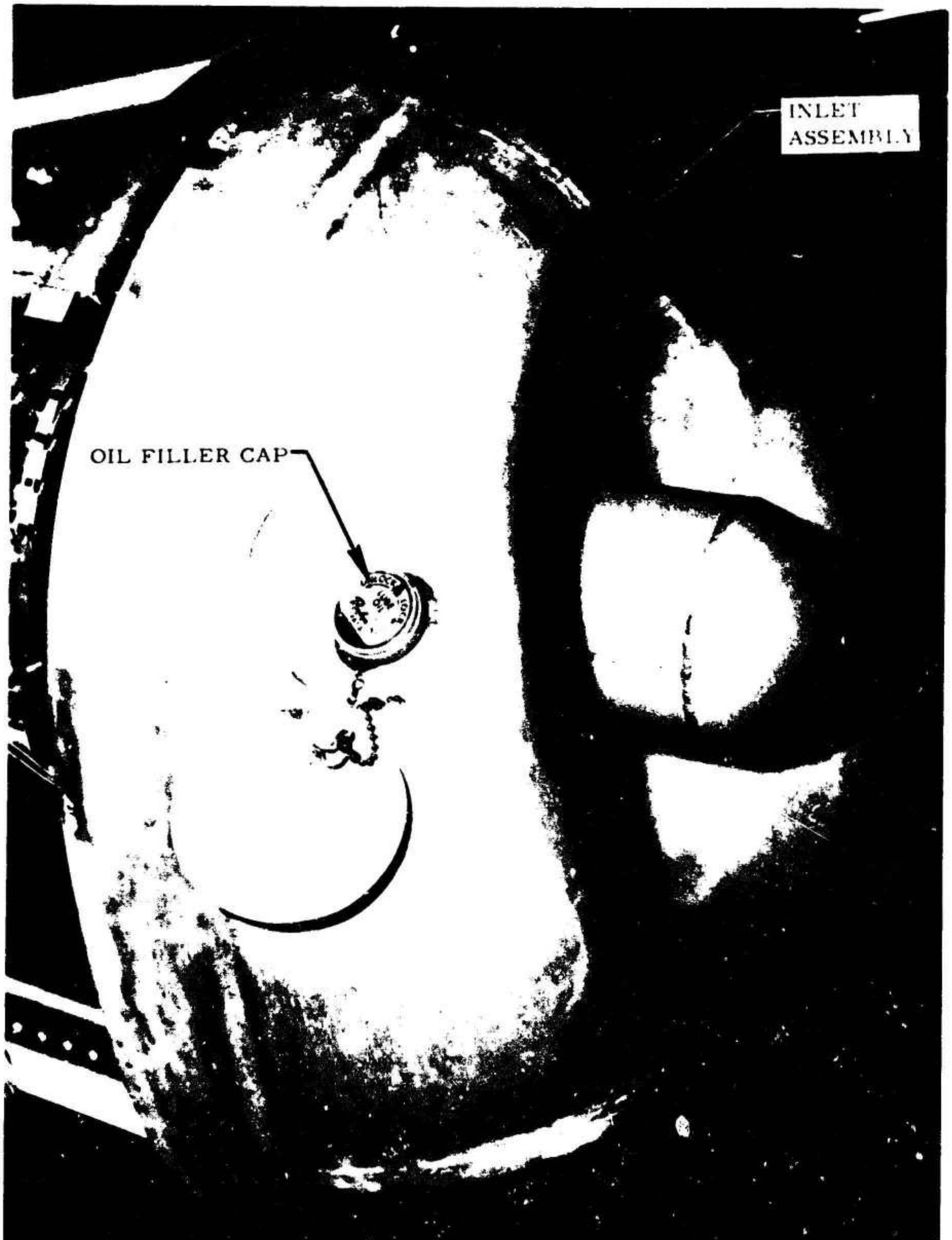


Figure 13. Gas Generator Inlet

The assembly is mounted to the gas generator inlet flange by a V-band coupling, and to one of the gas generator mount pads by a support bracket. Air loads in the axial direction are transmitted to and carried by the gas generator inlet flange. The support bracket serves only as a centering device for locating the inlet assembly and as a steadying link to resist inlet assembly rotation in the event of inadvertent loosening of the V-band attach coupling.

2.5.2.4 Lubrication System

The complete lubrication system consists of: (a) the internal or power unit system, which includes a supply system, a scavenge system, and a sump vent system; and (b) the external or airplane system, which includes the external supply and cooling systems.

The power unit system is shown schematically in Figure 10-1 of Reference 7, and the aircraft system is defined by Figure 14 herein.

The function of the aircraft system is to provide the power unit with an adequate supply of oil to lubricate and cool the gas generator components to meet the following installation requirements:

Gas generator oil	MIL-L-7808D
Pump inlet	
Minimum supply pressure	5 psia
Maximum aeration by volume	10 percent
Maximum supply temperature	225 degrees F
Optimum supply temperature	175 to 190 degrees F
Minimum flow	5.7 gpm (at 225 degrees F and 5 psia)
Pump scavenge	
Maximum discharge pressure	30 psig
Maximum aeration by volume	78 percent
Maximum temperature	360 degrees F
Maximum flow	25.5 gpm (at 360 degrees F and 30 psig)

Sump vent	
Maximum pressure at vent connections	5 psig at 100 percent gas generator rpm
Maximum inflow (air or air/oil vapor)	1.25 cfm
Maximum outflow (air or air/oil vapor)	1.00 cfm
Gas generator operating oil pressure	35 to 55 psig - military power 15 to 25 psig - ground idle power
Oil consumption (maximum average)	1.1 pph

The oil reservoir is an integral part of the gas generator air inlet assembly located on the left-hand side of the assembly. The reservoir has a total volume of five gallons (1,155 cubic inches), a total oil capacity of 3.6 gallons, and an expansion space equivalent to 40 percent of the total oil capacity, as detailed below:

	Pounds	Gallons	Cubic Inches
Usable oil (1.1 pph for 4 hours)	4.4	0.59	137
Nonusable oil (sump)	3.6	0.48	111
Transient residual oil	4.0	0.53	123
Dwell time oil	15.0	2.00	462
Subtotal	27.0	3.60	833
Expansion (40 percent oil capacity)	10.8	1.40	322
Total	37.8	5.00	1,155

To provide the required air-free oil to the engine, a dwell time of 20 seconds was established. At the rate of flow of approximately 6 gpm, a dwell time of 20 seconds necessitated an additional required oil capacity of 2.00 gallons.

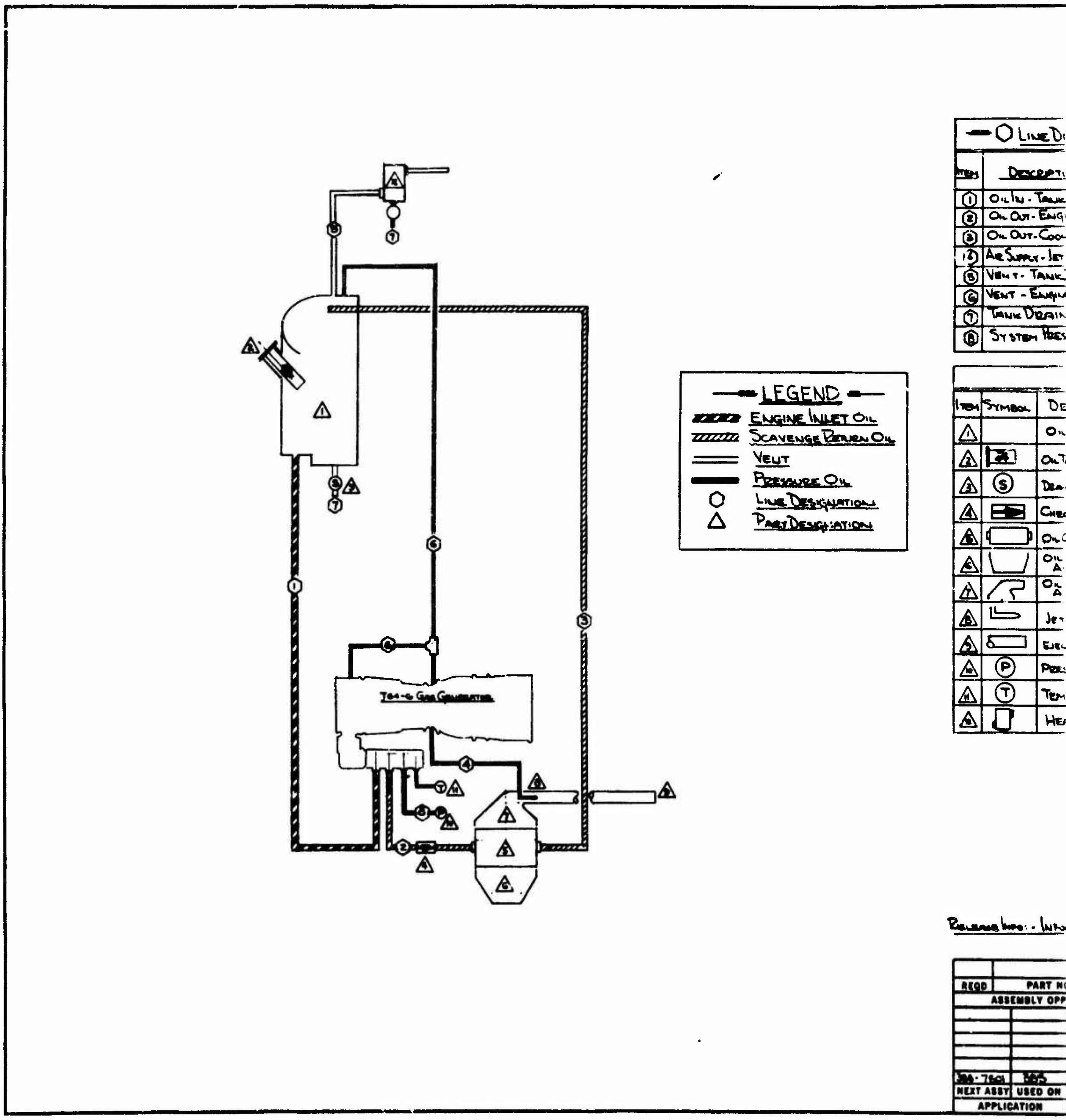


Figure 14. Gas Generator Lubrication System

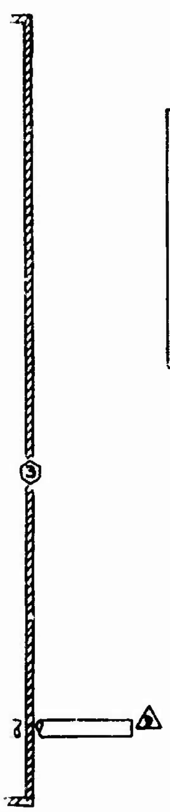
REVISIONS					
SYM	E.O.'S	DESCRIPTION	DRWN	APP'D	DATE
A		Revised For Header			

Item	Line Designation — Description	Basic Line Diameter
1	Oil In - Tank To Engine	.750
2	Oil Out - Engine To Cooler	.625
3	Oil Out - Cooler To Tank	.625
4	Air Supply - Jet Pump	.625
5	Vent - Tank To Atmosphere	.500
6	Vent - Engine To Tank	.375
7	Tank Drain - Header Drain	.250
8	System Pressure	.250

LEGEND

- Engine Inlet Oil
- Scavenge Return Oil
- Vent
- Pressure Oil
- ⬡ Line Designation
- △ Part Designation

Item	Symbol	Description	HTC P/N	Vendor P/N	Vendor
1		Oil Tank	385-7503	—	—
2		Oil Tank Filler	385-7503	9577	Romvu Inc.
3	S	Drain Valve	—	K-175-FT-4D	Kohler Co
4		Check Valve	—	2C6086	Cerrair Inc.
5		Oil Cooler	—	62165 A	Harrison Radiator
6		Oil Cooler - Air Inlet	385-7507	—	—
7		Oil Cooler Air Outlet	385-7508	—	—
8		Jet Pump	385-7510	—	—
9		Ejector Tube	385-7509	—	—
10	P	Pres. Transmitter	—	MS 28005-3	—
11	T	Temp. Bulb	—	MS 28034-3	—
12		Header Tank	385-7518	—	—



385-7502

A

Release Hooe: - Information Only

12-3-63

REQD	PART NO.	REQD	PART NO.	NAME	SIZE	DESCRIPTION	SPECIFICATION
ASSEMBLY OPP.		ASSEMBLY SHOWN		LIST OF MATERIAL			

NEXT ASSY	USED ON	APPLICATION	QTY REQD
385-7501	385	(REF)	(REF)

UNLESS OTHERWISE SPECIFIED: DIMENSIONAL TOLERANCES 2 PLACE DECIMAL ± .03, 3 PLACE DECIMAL ± .010, ANGULAR ± 0°30'. DIMENSIONS TO BE MET BEFORE PLATING. CORNER RADIUS ARE ON OF BORES AND SPOT FACES OF LESS DIA. OR LESS — RADIUS OR GREATER THAN LESS DIA.

DRWN, CHKD, APPROVED, APP'D, APP'D, APP'D, APP'D

Schematic Diagram
Model 385 -
Engine Lube System

HUGHES TOOL COMPANY
AIRCRAFT DIVISION
CULVER CITY, CALIFORNIA

385-7502

CODE 02731 SHEET OF

SCALE

ion System

B

The oil filler is located at the 3.6-gallon full line, and features a positive locking cap and dipstick assembly (see Figure 13). The dipstick is graduated to indicate quantity of oil to be added to bring the reservoir to the full line. A 10-mesh wire screen is provided in the filler neck.

All external lines are fire-resistant, teflon-lined flexible tubing, sized as shown in Figure 14. The oil flows, by gravity, from the reservoir directly to the engine. Return oil is directed back to the reservoir by engine scavenge pump pressure. Deaeration of this oil is ensured by directing the return flow tangentially against the tank forward wall. The shape of the tank itself and the amount of space above the solid oil level is such as to ensure successful deaeration of the return oil. The return flow is at a low velocity, so that deaeration is accomplished without splashing or further aeration.

Cooling of the oil is accomplished by means of a 68-square-inch air-to-oil heat exchanger located in the return side of the system. Airflow through the heat exchanger is induced by means of an ejector. Compressed air bled from the fourteenth stage of the gas generator compressor is used as the source of primary flow. One oil cooler with the associated equipment is provided for each gas generator. The estimated heat rejection to the lubricating oil for the YT-64 gas generator is:

	BTU per Hour	Oil Flow gph
Maximum power	89,000	354
75 percent power	83,000	350
60 percent power	77,500	342
Flight idle	24,000	300
Ground idle	10,000	280

The oil cooler design conditions are:

Maximum air temperature	$103^{\circ}F$
Air flow at 2 inches H_2O	65 pounds/minute
Design temperature	
OIL IN	$225^{\circ}F$ (maximum)
OIL OUT	$360^{\circ}F$ (maximum)

For oil cooler heat rejection data see page 10 of Reference 8. For air ejection data see Figure 15

3 In Dia Mixing Tube
0.08 Sq In. Primary Sonic Nozzle
$\Delta P = 12$ In. H_2O

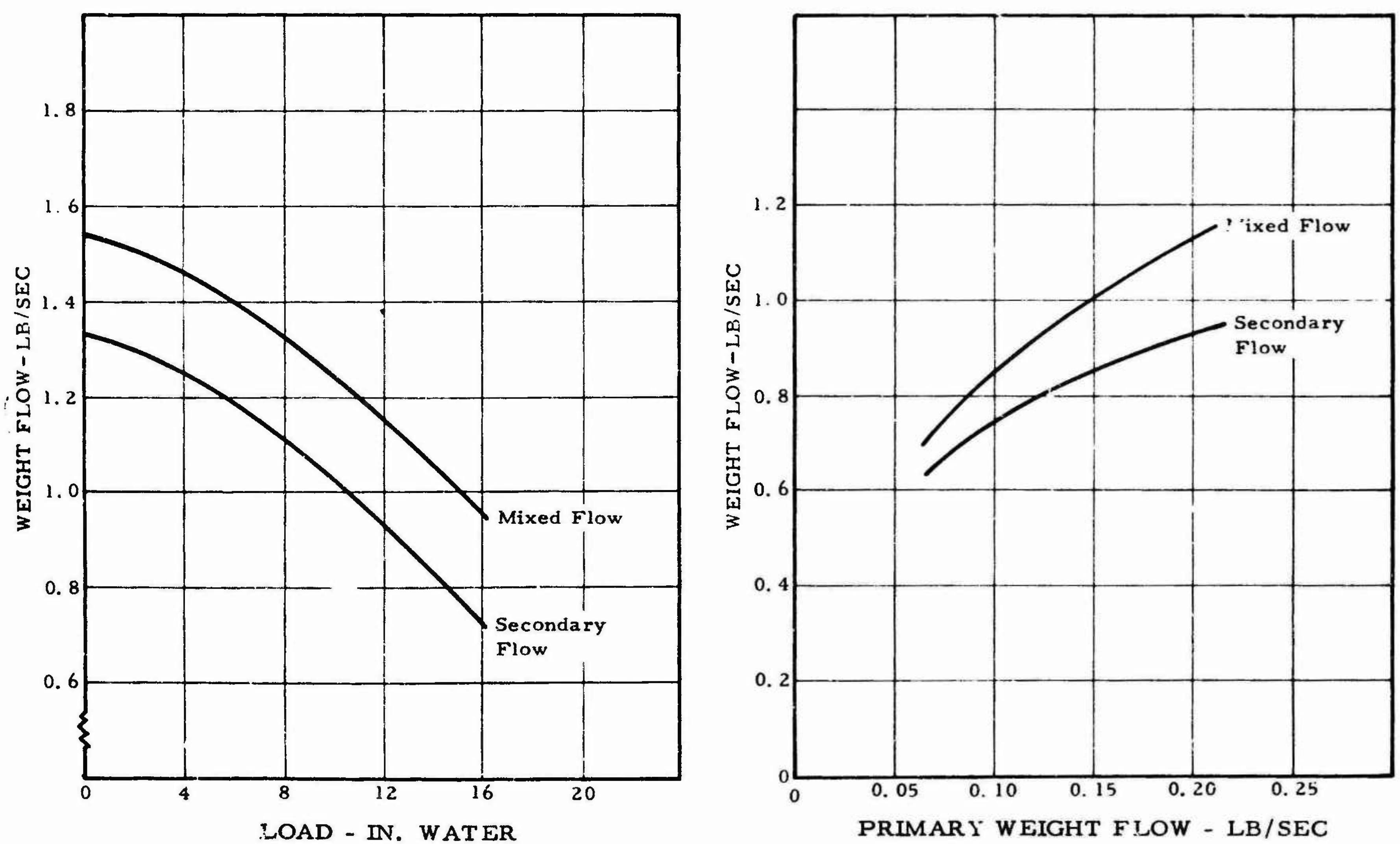


Figure 15. Air Ejector Performance

Figure 16 defines the gas generator bleed air ejector system geometry. Installation of components has been shown in Figure 12.

A check valve is located immediately upstream of the heat exchanger to preclude backflow to the gas genera' and prevent engine sump flooding during and after shutdown.

The gas generator front frame sump vent and midframe sump vent are manifolded and vented to the reservoir. The reservoir, in turn, is vented to atmosphere.

2.5.3 Mounting System

The load limitations of the YT-64 gas generator exhaust casing necessitated independent mounting of the diverter valve. Consequently, the mounting system consists of three independently mounted component systems (see Figure 17):

a. A gas generator support system

b. A diverter valve support system

c. A tailpipe support system

2.5.3.1 Gas Generator Mounting

The gas generator mount supports form a statically determinate system with three points of support. The forward supports located on the gas generator front frame are spherical-type bearings. The inboard mount on each gas generator is retained in a two-piece hinged socket rigidly attached to the gas generator mount structure. This support point reacts vertical, side, and thrust loads. The outboard mount is also retained by a two-piece hinged socket. However, this socket is attached to the truss by a ball rod end stabilizer link, and can resist vertical load only. The aft mount is a spherical universal linkage attached to the mounting points at the gas generator compressor rear frame. This support reacts vertical and side loads only, with no fore and aft restraint, so as to permit gas generator linear expansion. This fitting is part of the gas generator buildup and is attached to the structure at the time of gas generator installation.

The forward mounts are located in a relatively cool area and are fabricated from Type 410 corrosion-resistant steel heat treated

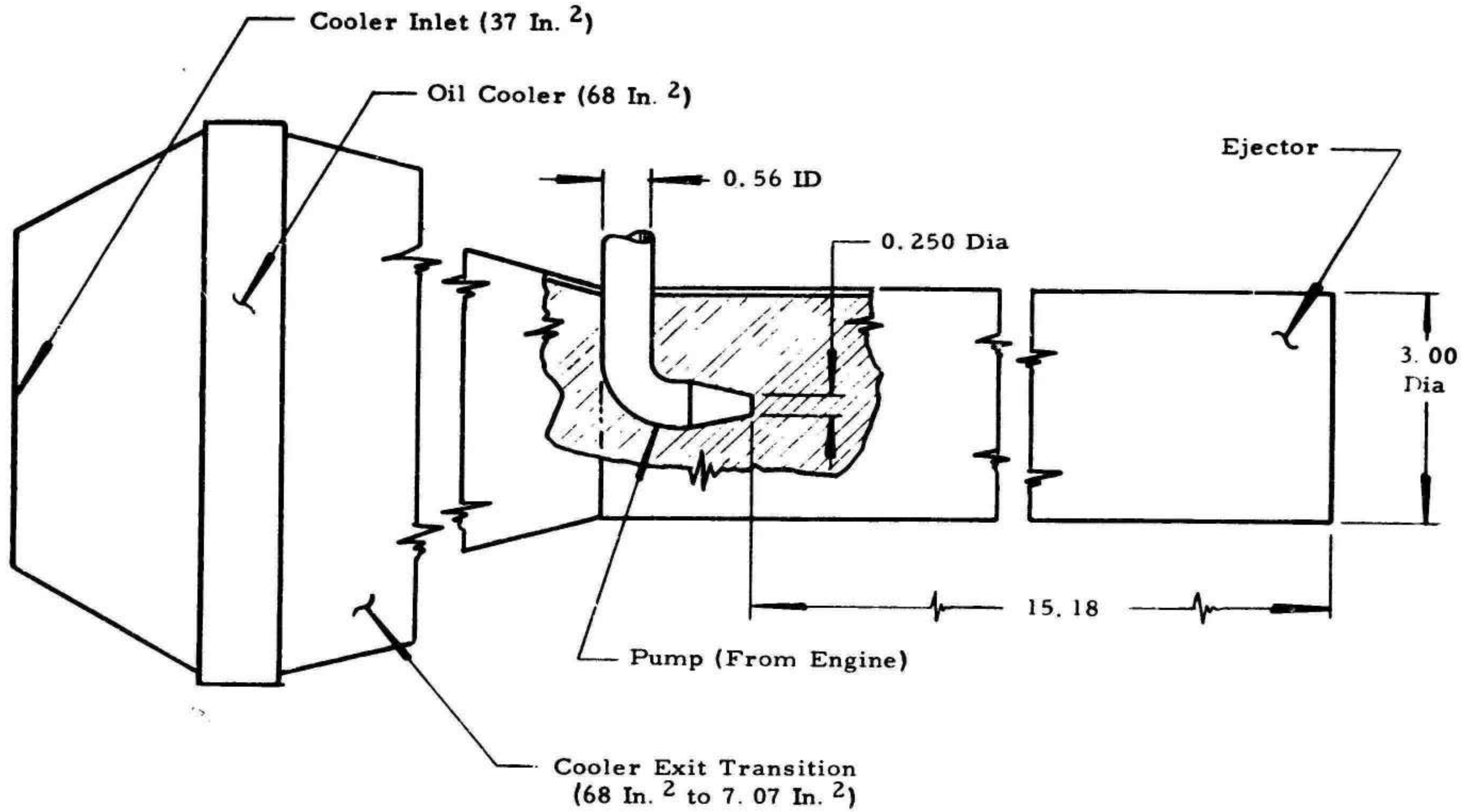


Figure 16. Air Ejector Configuration

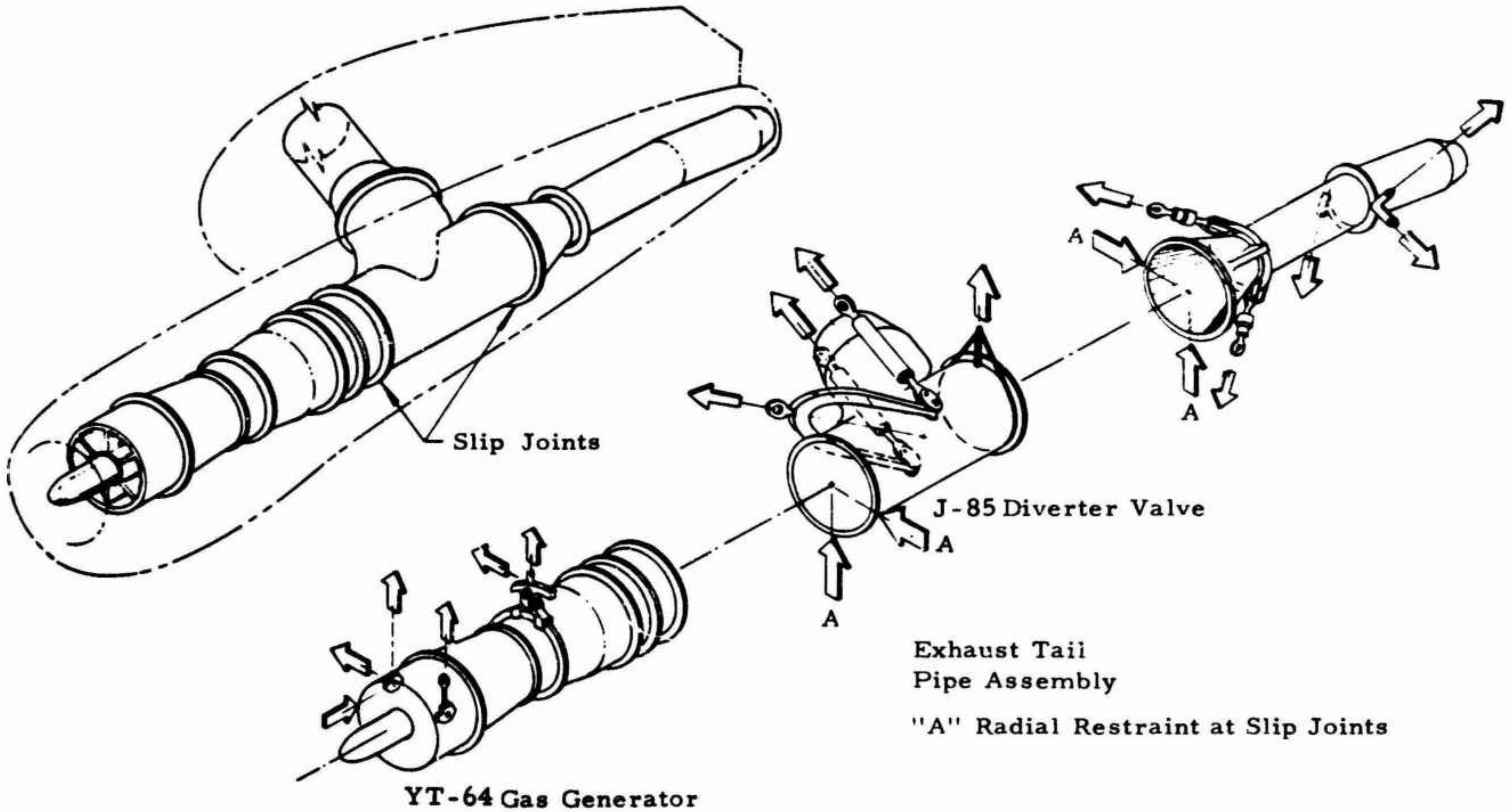


Figure 17. Propulsion System Mounting

to 180,000 to 200,000 psi. The aft mounts, located in a higher temperature region, are fabricated from Type 17-4 PH steel heat treated to 190,000 to 215,000 psi. The uniball bearings are 52100 steel, heat treated to 300,000 psi.

2.5.3.2 Diverter Valve Mounting

The configuration demands flexibility of the three components, to prevent transmission of loads as well as to permit relative motion and allow for axial thermal expansion. Comparison of several proposed mounting systems, their attachment to the structure, and the degree of freedom allowed indicated that a yoke-type support for the diverter valve was superior in action and simplicity.

The hot gas transition duct is sized and arranged so that the resultant of the diverter valve forces acts through the center supports of the yoke when engine gas is diverted to rotor position. When in straight-through or overboard flow, the induced loads are small and result only from the pressure loss through the valve and small changes in momentum. For the one-engine-out condition, the component of load out of the yoke plane is reacted by the two auxiliary support struts.

The weight of the diverter valve is supported by the gas generator exhaust adaptor at the forward end, and by a cable sling at the aft end. Up-loads are resisted by a link at the aft end.

The materials used in the support of the diverter valve are high-temperature, corrosion-resistant steel.

2.5.3.3 Tailpipe Mounting

The tailpipe assembly is supported from the structure at two points. The forward support consists of two struts and a gimbal ring assembly that carries the thrust loads into the power module rear spar. This arrangement is designed so that a portion of the tailpipe assembly weight is supported by the diverter valve, but prevents any diverter valve loads from being transmitted into the relatively thin walls of the tailpipe assembly.

The rear support consists of three struts to react side and vertical loading but not fore and aft loads, thereby permitting thermal expansion. Like the diverter valve, the materials used for the tailpipe assembly support are high-temperature, corrosion-resistant steel.

2.5.4 Component Removal

Each component of the propulsion system car e installed and/or removed easily without effect on the other component systems.

Points are provided on the gas generator, two at the forward mounts and one at the exhaust front frame, to attach the removal sling. Installation and removal of the gas generator are made vertically by means of a cable attached to the sling. The sling is removed when the gas generator is secure in the mounts or on the transportation dolly.

The diverter valve is installed and removed through the large access door incorporated in the lower portion of the nacelle between the power module front and rear spars. The yoke and linkage supports are permanently installed in the nacelle and are rigged for true gas generator and tailpipe alignment. Installation and/or removal of the diverter valve is made with the aid of an hydraulic lift and two cradle adaptors.

The tailpipe assembly is installed and removed through the large access door provided in the lower portion of the nacelle tail cone fairing. Installation and removal can be made without use of special fixtures. Supports are the two forward gimbal links and the three aft links, as well as the seal interconnect.

2.5.5 Interconnect Seals

The seal interconnect between the components is a metal-to-metal seal. The seal consists of an inner ring or adaptor attached to the component and an outer seal assembly that slides over the adaptor and is attached to the mating component after both components are secure. The seals consist of three layers of 0.010 René 41 high-temperature steel ground and cut to fit the inner ring circumference. The seal is angled to promote additional sealing from gas generator exhaust gas pressures. This type of seal will allow free axial and limited angular movement between the components while effecting a relatively gas-tight connection. Tests to date have demonstrated leakage to be negligible with this configuration. Friction loads are minimized by tungsten carbide surfaces on the inner ring and by moly-disulphide treatment of the ground sealing edges of the Rene 41.

Flight vertical and side loads are reacted by four studs, 90 degrees apart, located on the seal assembly. The seal configuration is shown in Figure 18.

2.5.6 Hot Gas System

The purpose of the hot gas system is to transfer the gas generator exhaust gases from each gas generator to the rotor blade-tip cascades in order to provide the driving force for the Hot Cycle pressure jet rotor. The system, exclusive of the gas generator and jet reaction yaw control system, consists of:

a. Diverter valves and controls

b. Transition ducts

c. Tailpipe assembly

d. Fixed Y-duct

e. Rotating triduct

f. Rotor system ducting, including blade-tip closure valves and controls

2.5.6.1 Component Description

The diverter valves are J-85 valves modified to meet Hughes Tool Company specifications. The gas generator diverter valve seal has been shown in Figure 18. This seal provides a smooth transition area from the gas generator to the diverter valve, as well as allowing for thermal expansion and unit misalignment. Flow through the diverter valves may be overboard for engine starting and operation or diverted for rotor operation. The valves are operated by hydraulic actuators and the hydraulic supply system described in Section 2.7.1.2.

The transition ducts are located in the lateral pylons. These ducts are fabricated from Inconel 718 corrosion-resistant steel, and contain bellows to compensate for thermal expansion or installation misalignment. The transition ducts are insulated to maintain acceptable structural temperatures.

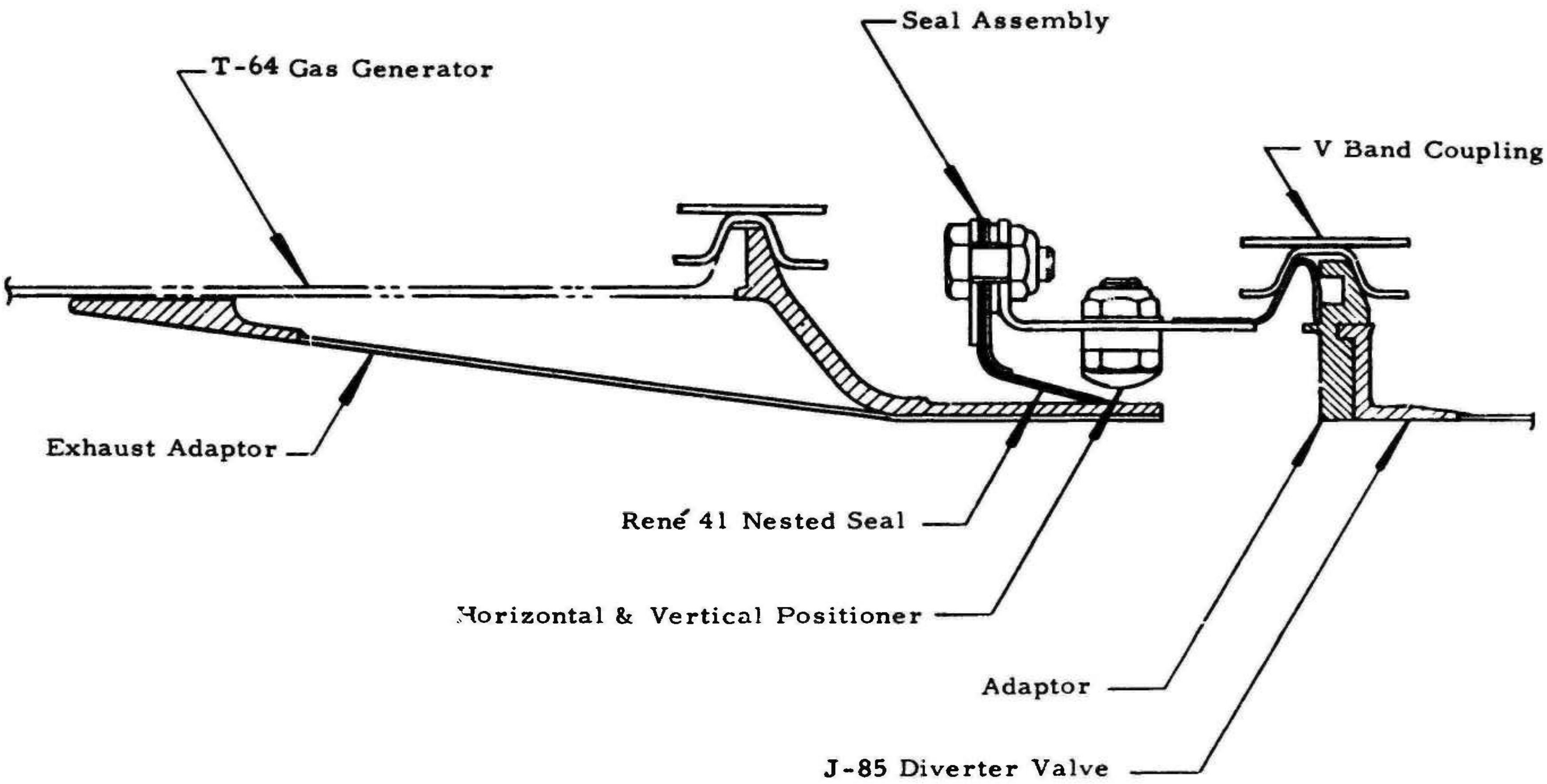


Figure 18. Gas Generator - Diverter Valve Seal

The attachment of the transition duct to the diverter valve is by V-band coupling. The flanges are of special design to ensure minimal leakage. Configuration is shown in Figure 19. This method of attachment is standard for the vehicle.

The tailpipe assembly, fixed Y-duct, and rotating triduct, as well as various details of the rotor ducting system, have been defined in other sections of this report.

2.5.6.2 Hot Gas System Controls

The hot gas system valves are all two-position devices that can be controlled by switches on the pilot's control console, or, at the pilot's option, can be controlled semiautomatically by the crossflow warning system (paragraph 2.5.6.3). To prevent inadvertent reduction in engine exit area, the blade-tip closure valves are interlocked with the diverter valve limit switches. Thus, the blade-tip closure valves cannot be closed unless either or both of the diverter valves are in the "gas overboard" position.

2.5.6.3 Crossflow Warning System Description

The Y-duct crossflow warning system is designed to sense an unbalance in the output of the gas generators, to visually display this unbalance, to give the pilot a visual and an aural warning of excessive unbalance, and to set up control circuits to aid the pilot in diverting the flow of a defective gas generator. These operations are accomplished as follows. An aerodynamically unbalanced vane installed in the intersection of the two gas streams (Figures 20 and 21) senses the relation between the output of the gas generators and drives a dual tandem potentiometer. The output of the potentiometer is detected by the warn-divert circuit, which drives the crossflow indicator and triggers the visual and aural warning signals (Figure 22). If the gas generator mismatch reaches an arbitrarily designated value, the warn-divert circuit triggers a blinking amber light and warbling tone in the crew headsets. The illuminated lights and the position of the crossflow indicator designate the gas generator with the low output. If the mismatch approaches the potentially dangerous level, the warn-divert circuit triggers a blinking red light and modifies the warbling tone in the headsets. Simultaneously, the divert circuit arms the collective-stick-mounted divert switch. If the pilot accepts the mismatch warnings, he simply pushes the divert switch, and the malfunctioning

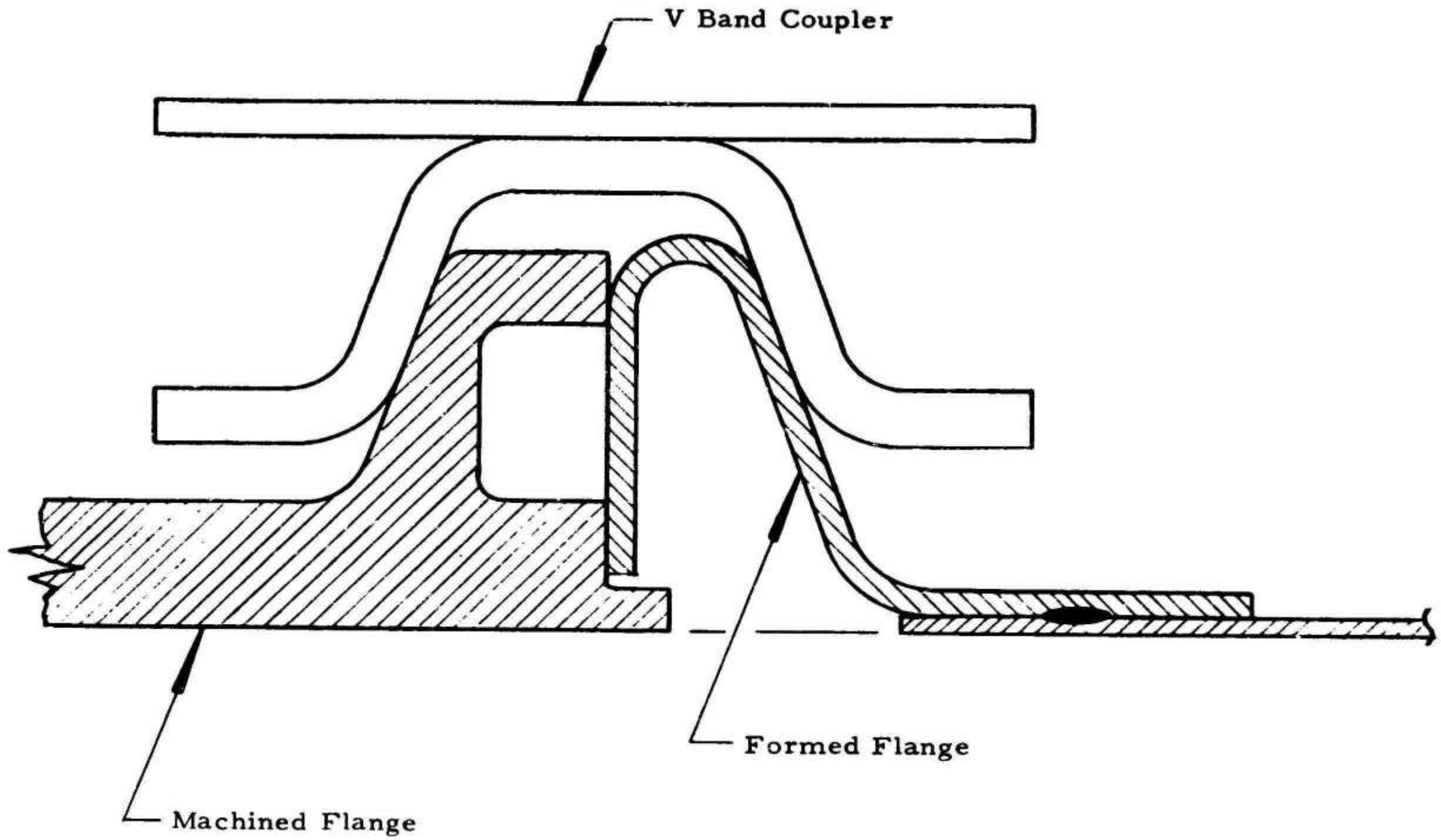


Figure 19. Hot Gas System Connection Configuration

engine is isolated by diverting its flow overboard. The blade-tip closure valves are then automatically closed, to maintain proper engine exit area. The pilot can utilize this semiautomatic divert system only if his manual control switches are initially set to automatic position and only if the warn-divert system indicates excessive mismatch. The valves may be returned to their normal flight position by use of the manual control switches or by opening and reclosing the system circuit breakers.

2.5.6.4 Warn-Divert System Components

The following listed components comprise the major units of the system:

a. Y-duct vane. A vane located in the Y-duct juncture from the two gas generators. It is positioned to center, and is deflected away from center if the exhaust gas flow from the two gas generators is not balanced. The angle of deflection is a measure of the magnitude of the unbalance.

b. Transducer. A dual-tandem potentiometer. It is coupled to the Y-duct vane by sprockets and a chain. The output of the transducer provides a signal of the Y-duct vane position.

c. Indicator, Y-duct vane position. A 270-degree dial, 2-inch indicator using a standard electrical meter movement. It is coupled to the warn-divert unit to indicate the position of the Y-duct vane.

d. Warn-divert unit. A completely solid-state signal conditioning unit. This unit accepts the signal from the position transducer and provides the following outputs:

(1) Analog position signal to operate the position indicator.

(2) Right or left amber light flashing simultaneously with a modulated tone in the radio headsets, if the vane moves $\pm$ 5 degrees from center.

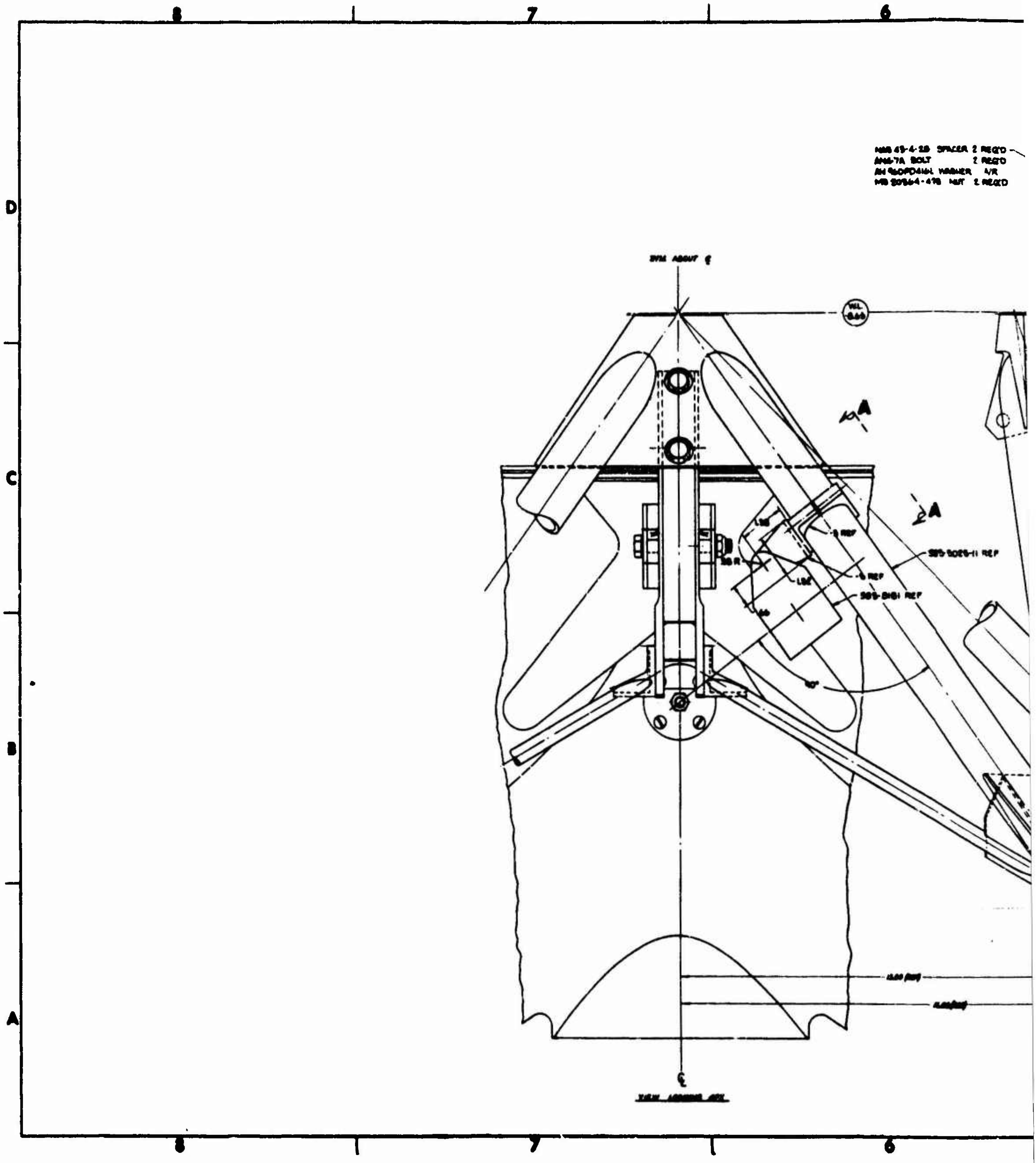


Figure 20. Y-Duct Crossflow Indication System

6 5 4

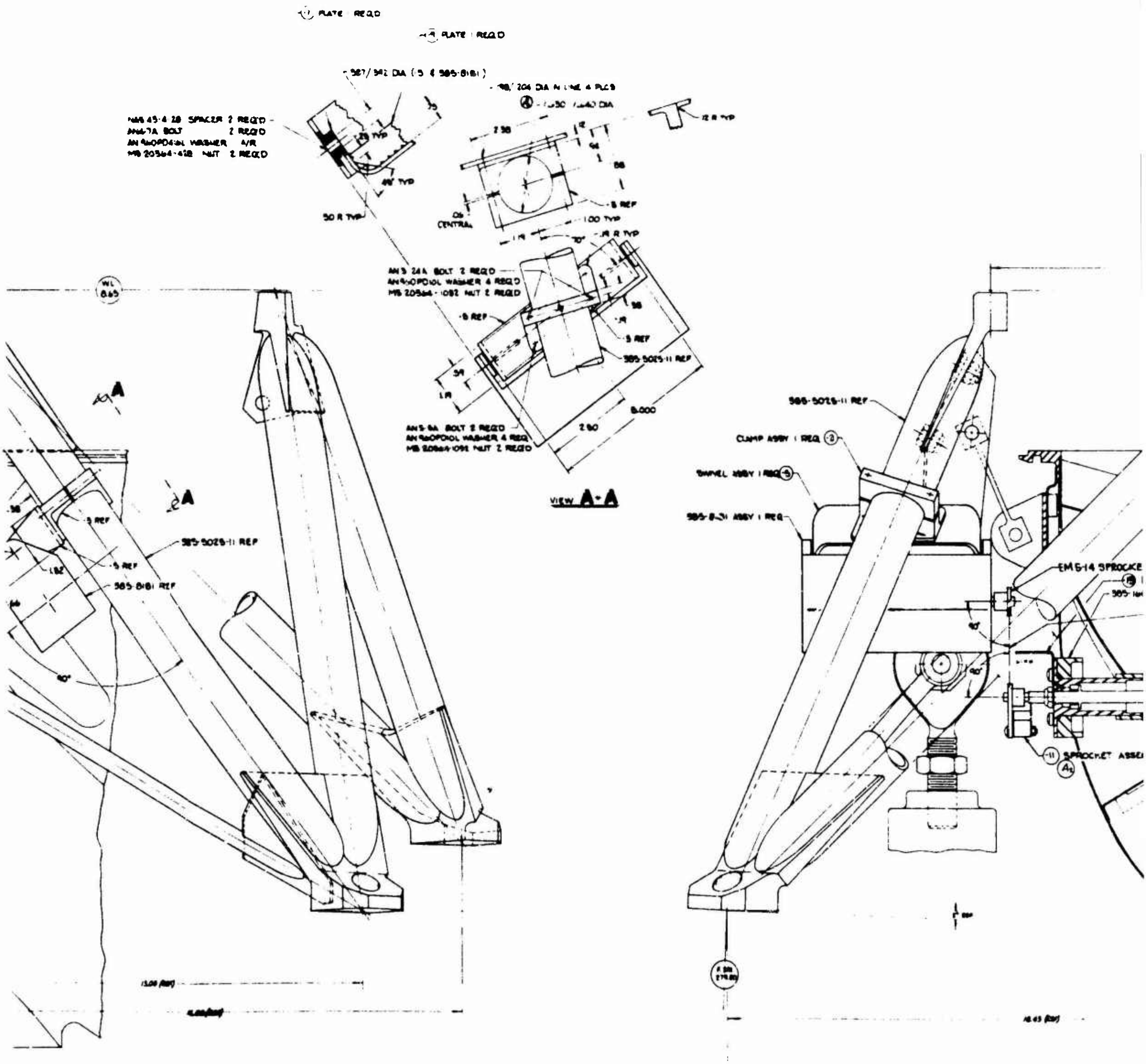
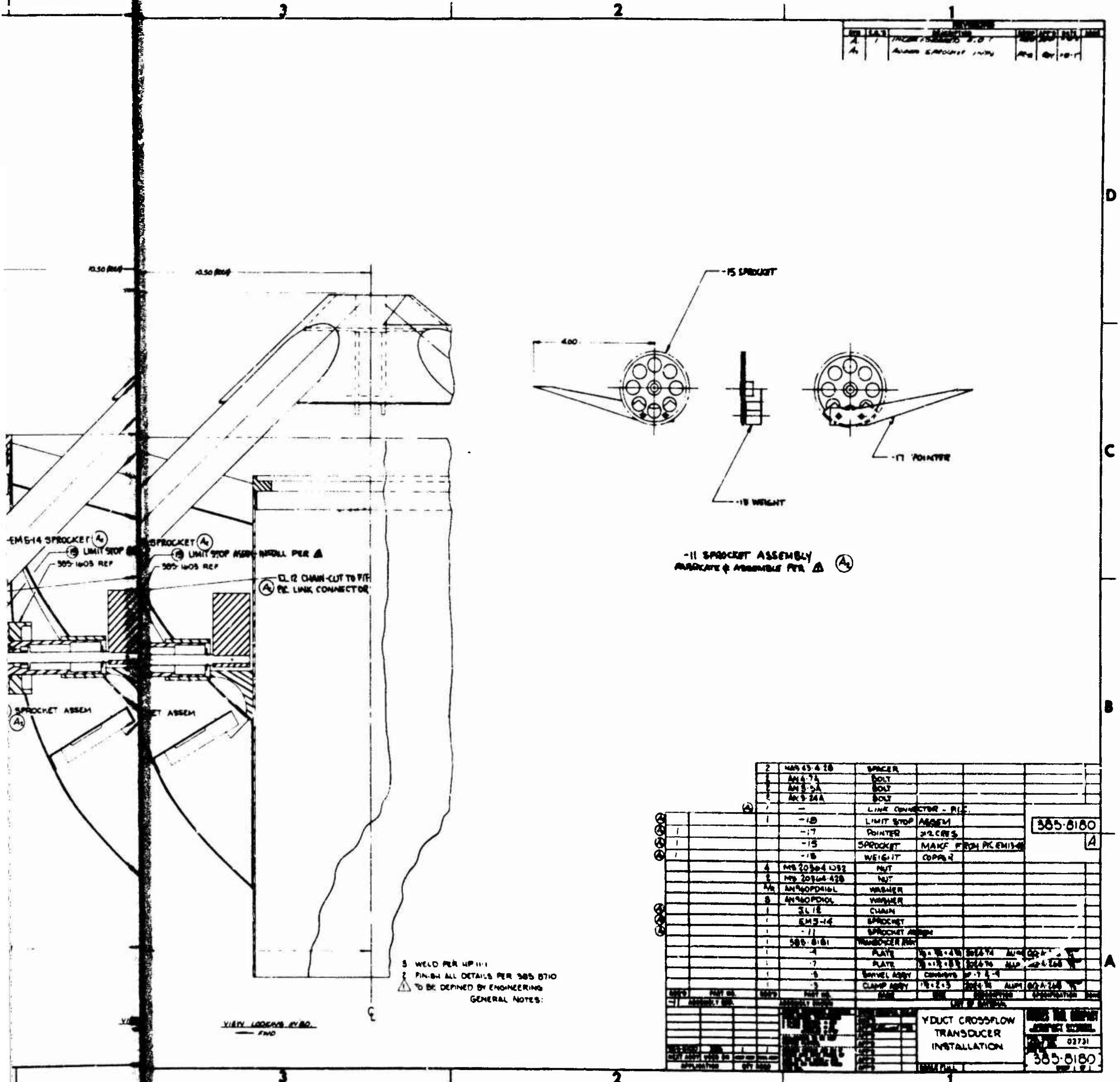


PLATE 1 REQ'D
NAS 45-4 28 SPACER 2 REQ'D
AN67A BOLT 2 REQ'D
MS 20364-428 NUT 2 REQ'D
VIEW A-A
585-5025-11 REF
585-8181 REF
CLAMP ASSY 1 REQ
SWIVEL ASSY 1 REQ
EM6-14 SPROCKET
SPROCKET ASSEM

6 5 4

B



2	NAS 43-4-28	SPACER		
2	AN 6-74	BOLT		
2	AN 5-5A	BOLT		
2	AN 5-24A	BOLT		
1	—	LINK CONNECTOR - P.C.		
1	-19	LIMIT STOP ASSEM		
	-17	POINTER	302 CRES	
	-15	SPROCKET	MAKE FROM PC EM5-14	
	-13	WEIGHT	COPPER	
4	MS 20364-1032	NUT		
2	MS 20364-428	NUT		
A/R	AN960PD416L	WASHER		
8	AN960PD10L	WASHER		
1	3L12	CHAIN		
1	EM5-14	SPROCKET		
1	-11	SPROCKET ASSEM		
1	385-8181	TRANSDUCER ASSY		
1	-9	PLATE	[illegible] 2024-T4 ALUM	[illegible]
1	-7	PLATE	[illegible] 2024-T4 ALUM	[illegible]
1	-5	SWIVEL ASSY	CONSISTS OF -7 & -9	
1	-3	CLAMP ASSY	[illegible] 2024-T4 ALUM	QQ-A-268

C

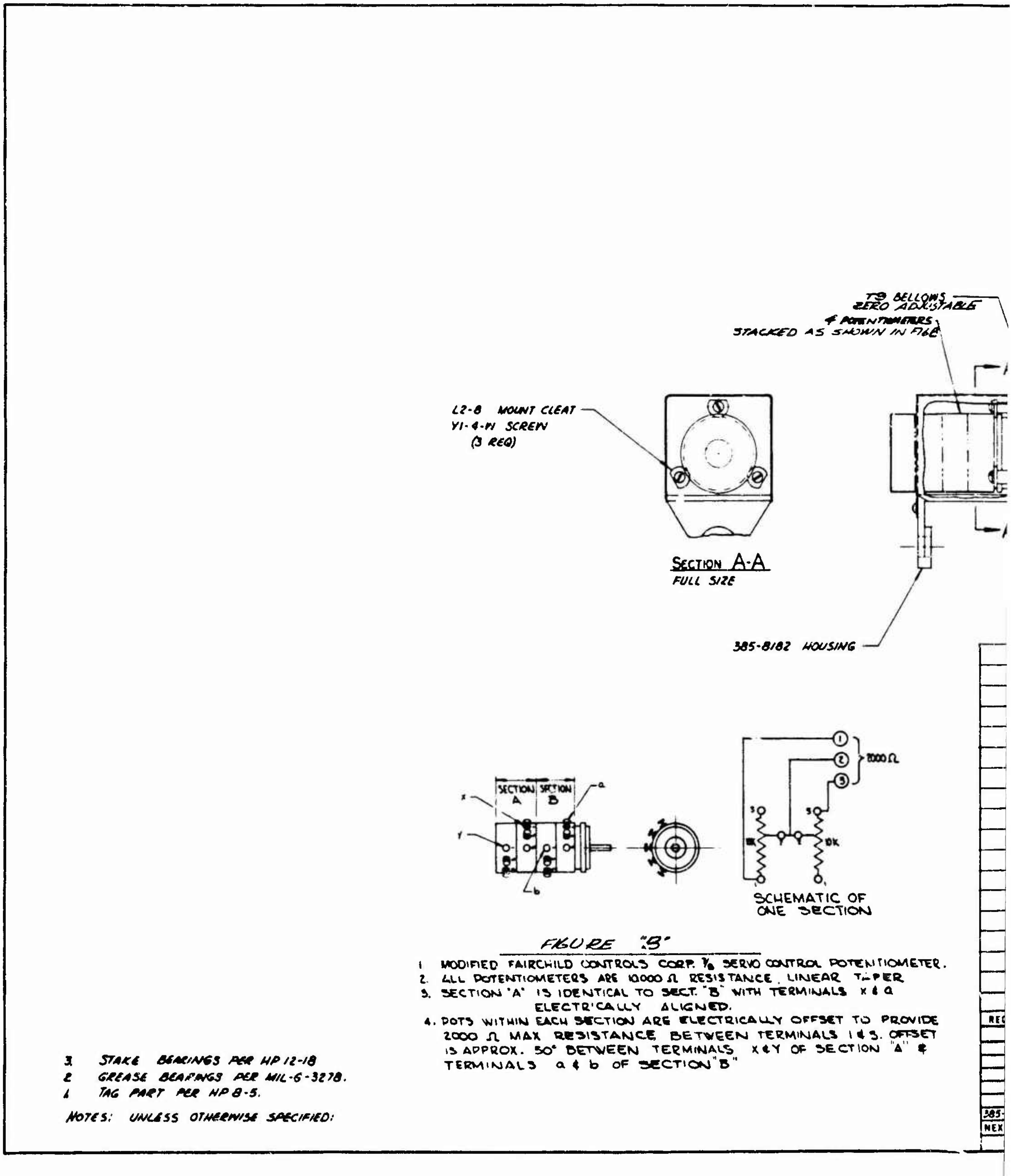
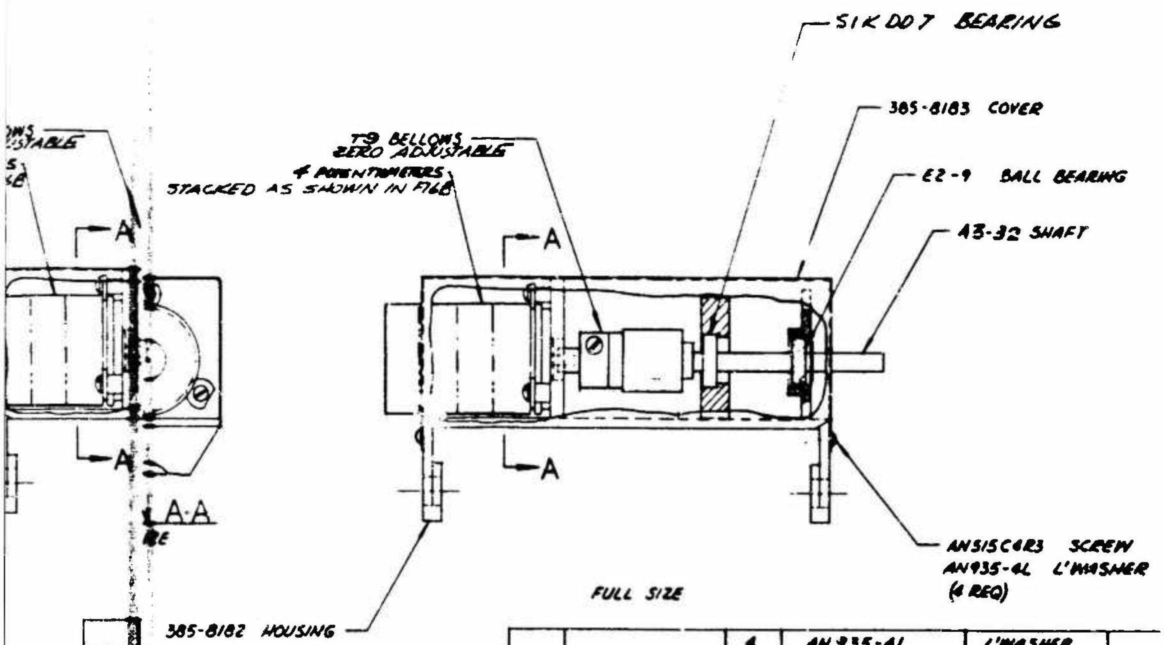
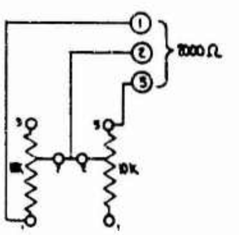


Figure 21. Crossflow Transducer Assembly

REVISIONS					
SYM	E.O.'S	DESCRIPTION	DRWN	APP'D	DATE
A.	1	CANCELLED FO 1			
A₁		MODIFIED ASSEMBLY			





SCHEMATIC OF ONE SECTION

TER.

SERVO CONTROL POTENTIOMETER.
TANCE, LINEAR TAPER
WITH TERMINALS X & Q

RICALLY OFFSET TO PROVIDE
EN TERMINALS 1 & 5. OFFSET
S X & Y OF SECTION "A" &

REQD	PART NO.	REQD	PART NO.	NAME	SIZE / DESCRIPTION	SPECIFICATION
		4	AN935-4L	L'WASHER		
		4	AN515C4R3	SCREW		
		3	Y1-4-W	SCREW	P.I.C. DESIGN CORP. VAN NUYS, CAL.	
		3	L2-8	MOUNT CLEAT	P.I.C. DESIGN CORP. VAN NUYS, CAL.	
		1	SKDO7	BEARING	FAFNIR	
		1	E2-9	BALL BEARING	P.I.C. DESIGN CORP., VAN NUYS, CALIF.	
		1	T9-.2500 x .1800 DIA	BELLOWS	P.I.C. DESIGN CORP., VAN NUYS, CALIF.	
		4	VARIOUS	POTENTIOMETERS	MAKE UP STACK AS SHOWN IN FIG "B"	
		1	385-8183	COVER		
		1	385-8182	HOUSING		
		1	A3-32	SHAFT	P.I.C. DESIGN CORP., VAN NUYS, CAL.	
ASSEMBLY OPP		ASSEMBLY SHOWN		LIST OF MATERIAL		

385-8181

A

NEXT ASSY	USED ON	NEXT ASSY	FINAL ASSY
385-8180	385	1	1
APPLICATION		QTY REQD	

UNLESS OTHERWISE SPECIFIED DIMENSIONAL TOLERANCES 2 PLACE DECIMAL ± .010 3 PLACE DECIMAL ± .005 ANGULAR ± 0°30' DIMENSIONS TO BE MET BEFORE PLATING. CORNER RADIUS ARE OR ... BORES AND SPOT FACES OF ... DIA. OR LESS — ... RADIUS ON GREATER THAN ... DIA.

DRWN, CHK'D, APP'D, APP'D, APP'D, APP'D, APP'D, APP'D

TRANSDUCER ASSY. - Y-DUCT CROSSFLOW TRANSD.

HUGHES TOOL COMPANY
AIRCRAFT DIVISION
CULVER CITY, CALIFORNIA

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SCALE 1/1 | CODE 02731 | SHEET 1 OF 1

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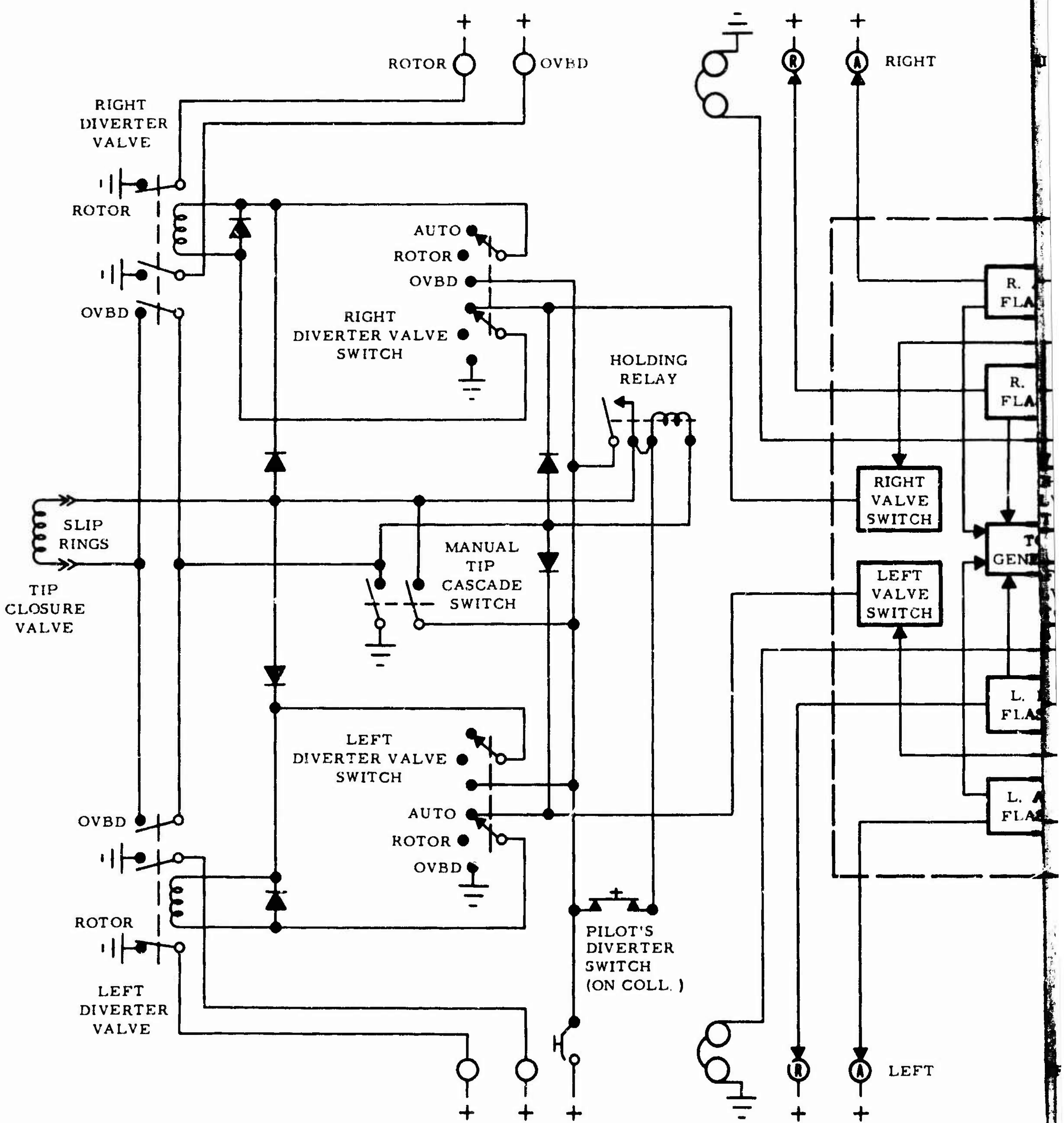
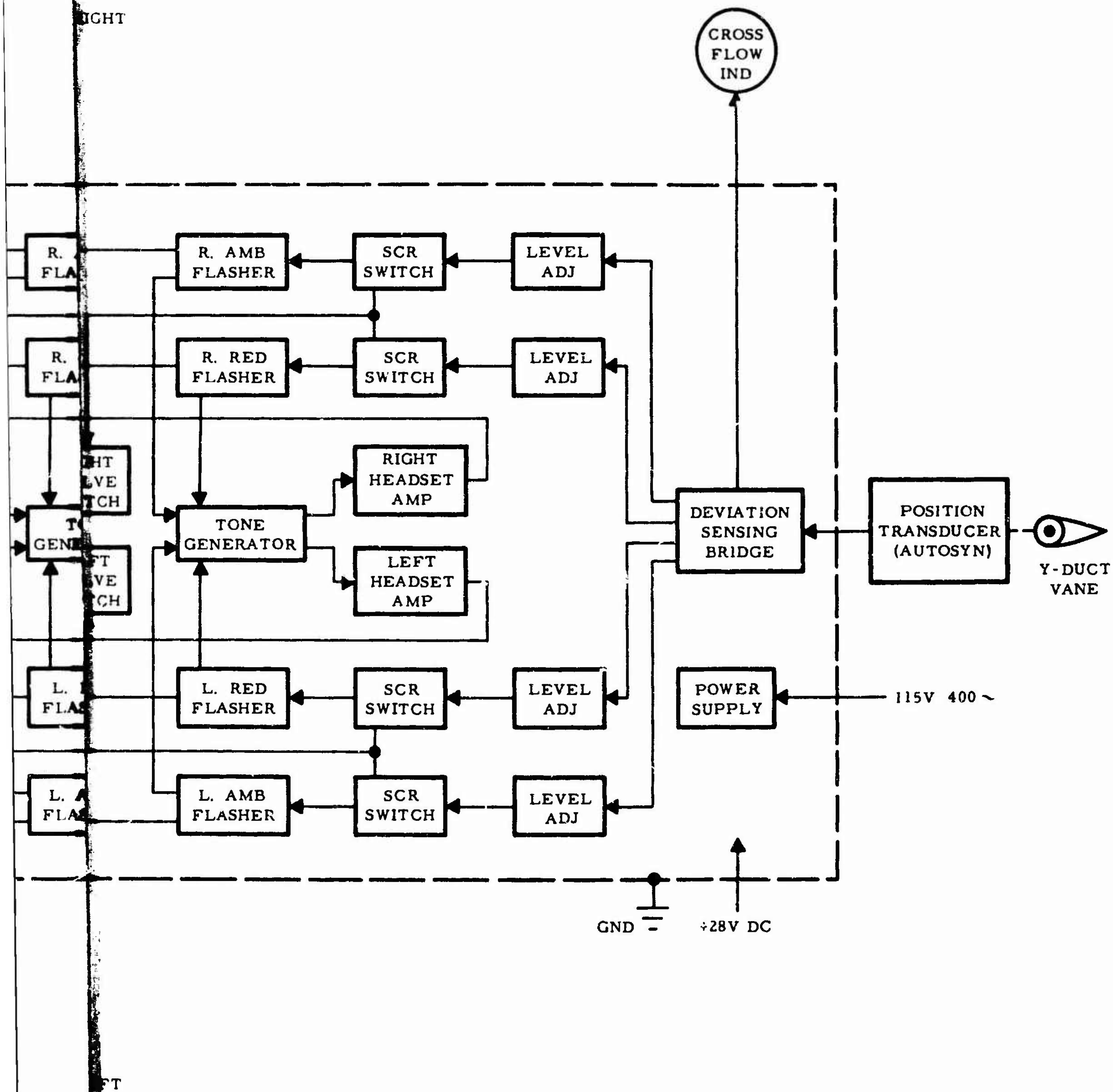


Figure 22. Block Diagram, Crossflow Warning System

A


CROSS FLOW IND
R. AMB FLASHER
SCR SWITCH
LEVEL ADJ
R. RED FLASHER
SCR SWITCH
LEVEL ADJ
RIGHT HEADSET AMP
TONE GENERATOR
LEFT HEADSET AMP
DEVIATION SENSING BRIDGE
POSITION TRANSDUCER (AUTOSYN)
Y-DUCT VANE
L. RED FLASHER
SCR SWITCH
LEVEL ADJ
POWER SUPPLY
115V 400 ~
L. AMB FLASHER
SCR SWITCH
LEVEL ADJ
GND
+28V DC

B

(3) Right-or-left red light flashing simultaneously with a drastic change in the tone in the radio headsets if the vane moves $\pm$ 10 degrees from center; in addition, provides a grounding circuit for the external control circuitry of the malfunctioning engine's diverter valve and for the blade-tip closure valves.

e. Signal balancing potentiometer. A potentiometer that provides capability for zero setting of the indicator and for pilot adjustment of the switch points for the warning light and the audible warning signal.

f. Pilot's activate switch. A momentary pushbutton switch on the collective stick. When a ground circuit has been set up by the warn-divert unit as described above, pushing this switch operates the solenoid of the diverter valve of the malfunctioning engine for overboard operation and the blade-tip solenoid valve for reduction of blade duct area.

2.5.7 Yaw Control System

In hovering and in low-speed forward flight, the directional control of the vehicle depends on the operation of the jet reaction yaw control system. This yaw control system utilizes the rotor system gases, and produces the required yaw force by discharging these hot gases through variable area nozzles located diametrically opposite each other at fuselage Station 598.85. The system has been shown in Figure 11, and consists basically of a supply system, a directional valve, and a control system.

2.5.7.1 Supply System

The supply system embraces the ducting required to carry the hot gases from the rotor Y-duct to the directional control valve. The ducts (5.00-inch and 7.00-inch diameter thin wall Type 347 corrosion-resistant steel tubing) are interconnected by stainless steel bellows, which compensate for duct thermal expansion. Loads induced into the bellows by thermal expansion are kept small by selected duct lengths and methods of support. The ducts and bellows are insulated to maintain safe operational temperature levels in the surrounding structure. The system employs the metal-to-metal V-band coupling configuration (Figure 19) with formed flanges on the ducts and the machined flanges on the bellows.

2.5.7.2 Directional Control Valve

The directional control valve assembly is shown in Figure 23. It consists essentially of a 9.56-inch-diameter rotor (cylindrical closure) contained in a 10-inch cylindrical housing (plenum chamber) that includes two ducted diametrically opposed outlets of 23.5 square inches each. The rotor, which is supported in the housing by sleeved carbon bushings, has corresponding cutouts, located so that rotation of $\pm$ 58 degrees from the neutral position will select the outlet and vary the flow from full closed to full open. The leakage from the plenum is controlled by carbon seals at each outlet. The seals are lapped in place for full coincidence with the rotor surfaces, and are designed to produce the maximum amount of sealing by taking advantage of the valve operating pressures. Rotor operation loads are minimized by the low coefficient of friction of the carbon used.

The valve body, the rotor, the valve end plate, and the side outlets are fabricated from Type 347 corrosion-resistant steel, welded and machined into homogeneous units. Springs used in the assembly are fabricated from Inconel X.

The valve assembly is insulated to ensure safe operational temperature levels in the adjacent structure.

2.5.7.3 Yaw Control Valve Design Criteria

Maximum yaw thrust required at the design point with the fully open valve	300 pounds
Maximum nozzle exit area	23.5 square inches
Gas power as required in hovering at gross weight	15,300 pounds

The thrust available for the adequate directional control of the XV-9A varies with flight conditions. Once the valve opening is fixed, yaw thrust becomes a function of gas power that is utilized at that moment in the rotor system. Whenever the rotor power increases,

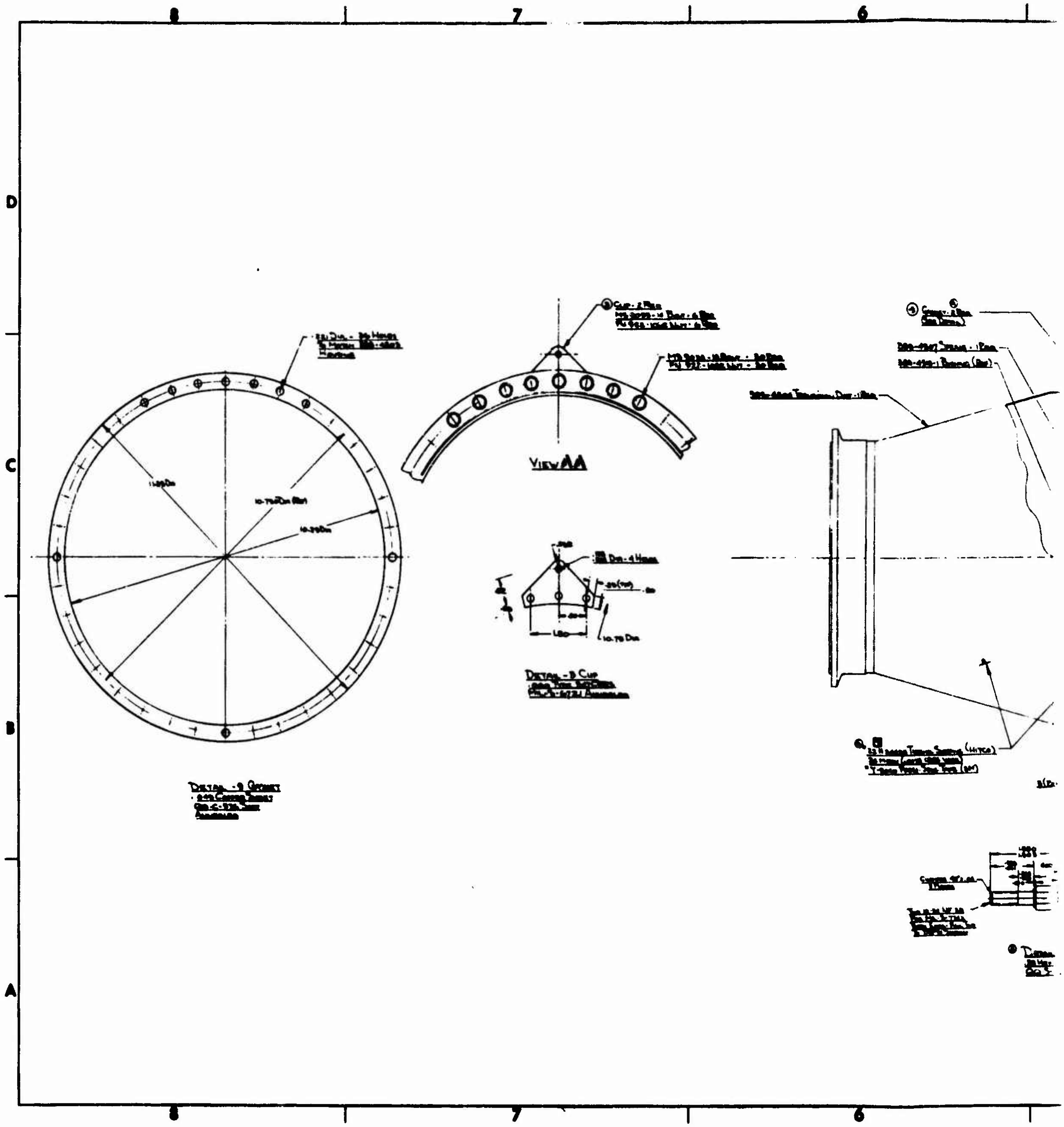
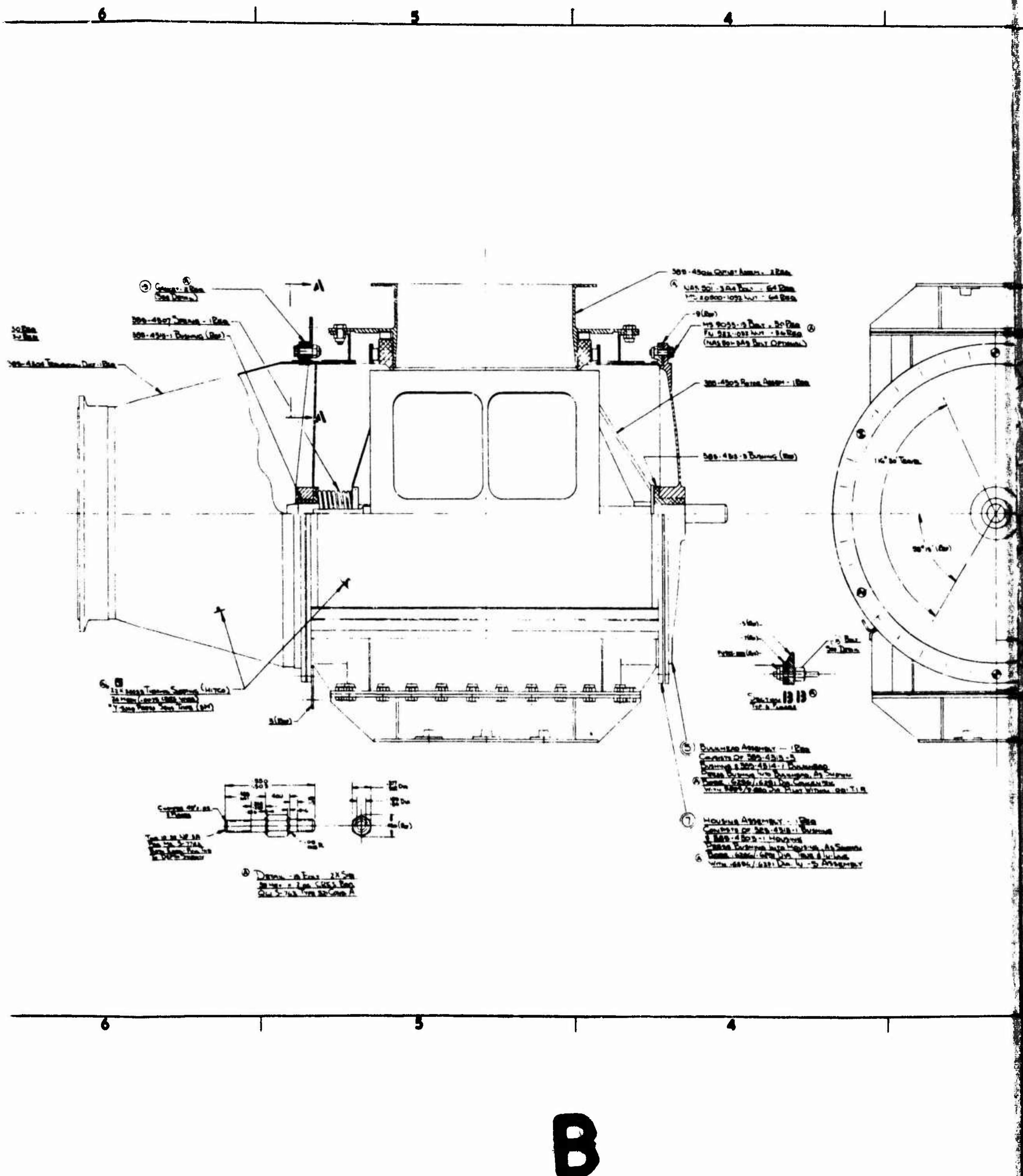
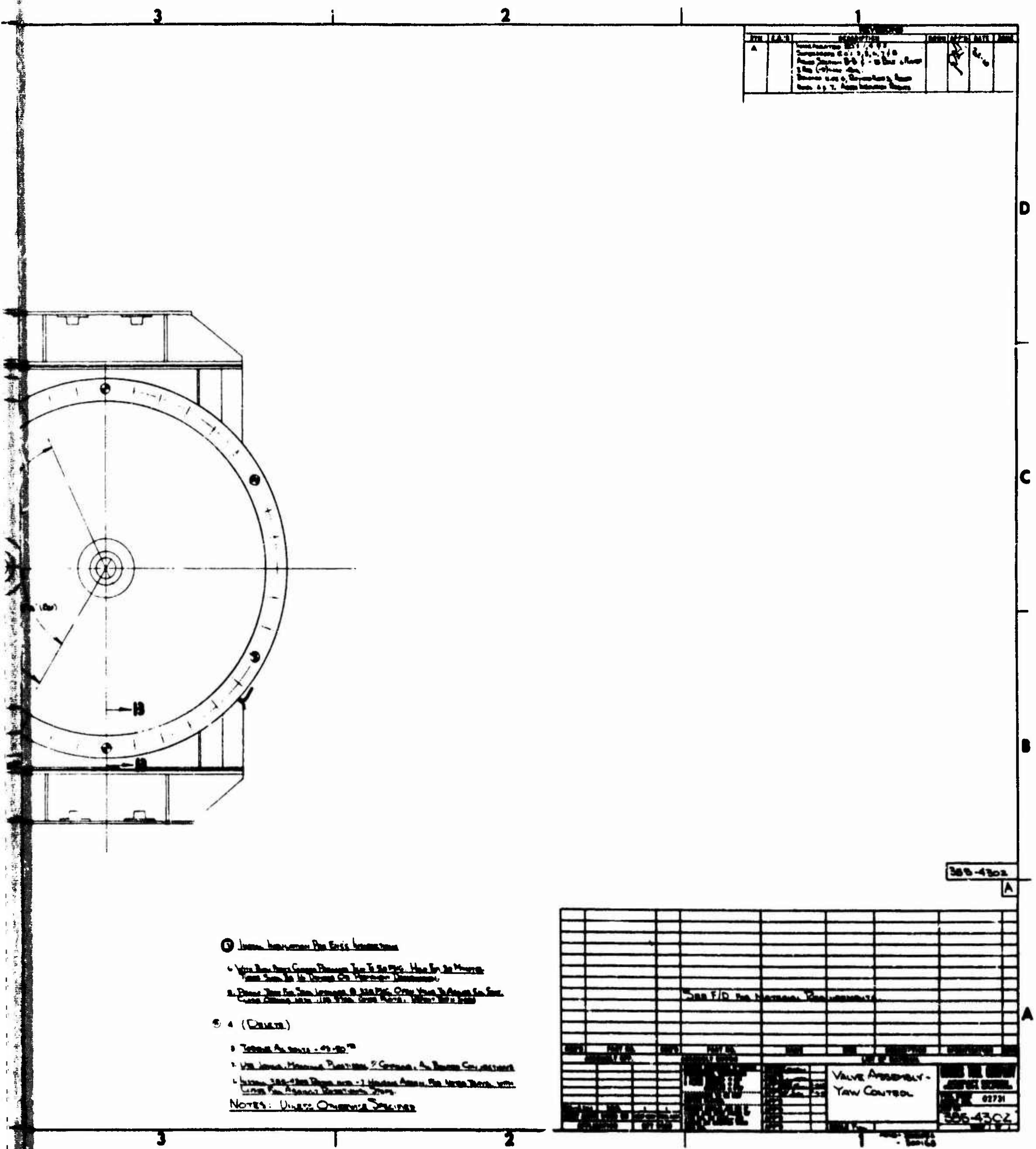


Figure 23. Yaw Control Valve Configuration


6
5
4
Bulkhead Assembly — 1 Req
Housing Assembly - 1 Req
B



See F/D for Material Requirements
Valve Assembly - Yaw Control
02731
386-4302
386-4302
4 (Deleted)
Notes: Unless Otherwise Specified

C

yaw control becomes more effective. Therefore, after exceeding a certain engine power level, only a partial opening of the yaw control valve is necessary to satisfy the specified yaw control requirements. This engine power level is approximately 60 percent of maximum power.

There is also an interdependence between the engine output and the operation of the yaw control system. Opening of the valve increases the total exit area used by the engines, resulting in a pressure drop in the system and a reduction of mass flow to the rotor. The subsequent decrease in power must be compensated by the adjustment of power lever angle (PLA).

2. 5. 7. 4 Control System

The yaw control valve is controlled through a cable and lever system by the pilot's yaw control pedals. It is also connected to the rudders by a cable and push-rod system so that the valve and rudders move together. For a more complete description, refer to Section 2. 6. 2.

2. 5. 8 Compartment Cooling

The front part of the gas generator bay is comparatively cool. Temperatures are rising toward the rear end of the gas generator, where the bay is subjected to intensive radiation and convection from the gas generator hot sections; namely, the combustor, the turbine, and the exhaust casings. In this area, fuel and hydraulic lines, electrical runs, and control cabling are routed into closed insulated compartmented areas. Ventilation is provided in these areas by holes at each end of the compartments.

Cooling air is introduced to the nacelle through a 1. 00-inch annular opening at the leading edge of the accessory cowling and then induced to flow through by the action of the ejector described in Section 2. 5. 2. 4. Additional cooling is provided by cutouts and louvers in the accessory cowl panels.

YT-64 gas generator heat rejection data are delineated on Figure 6-5 of Reference 7, and the estimated temperature profile for the nacelle skin and adjacent structure is shown in Figure 24 herein.

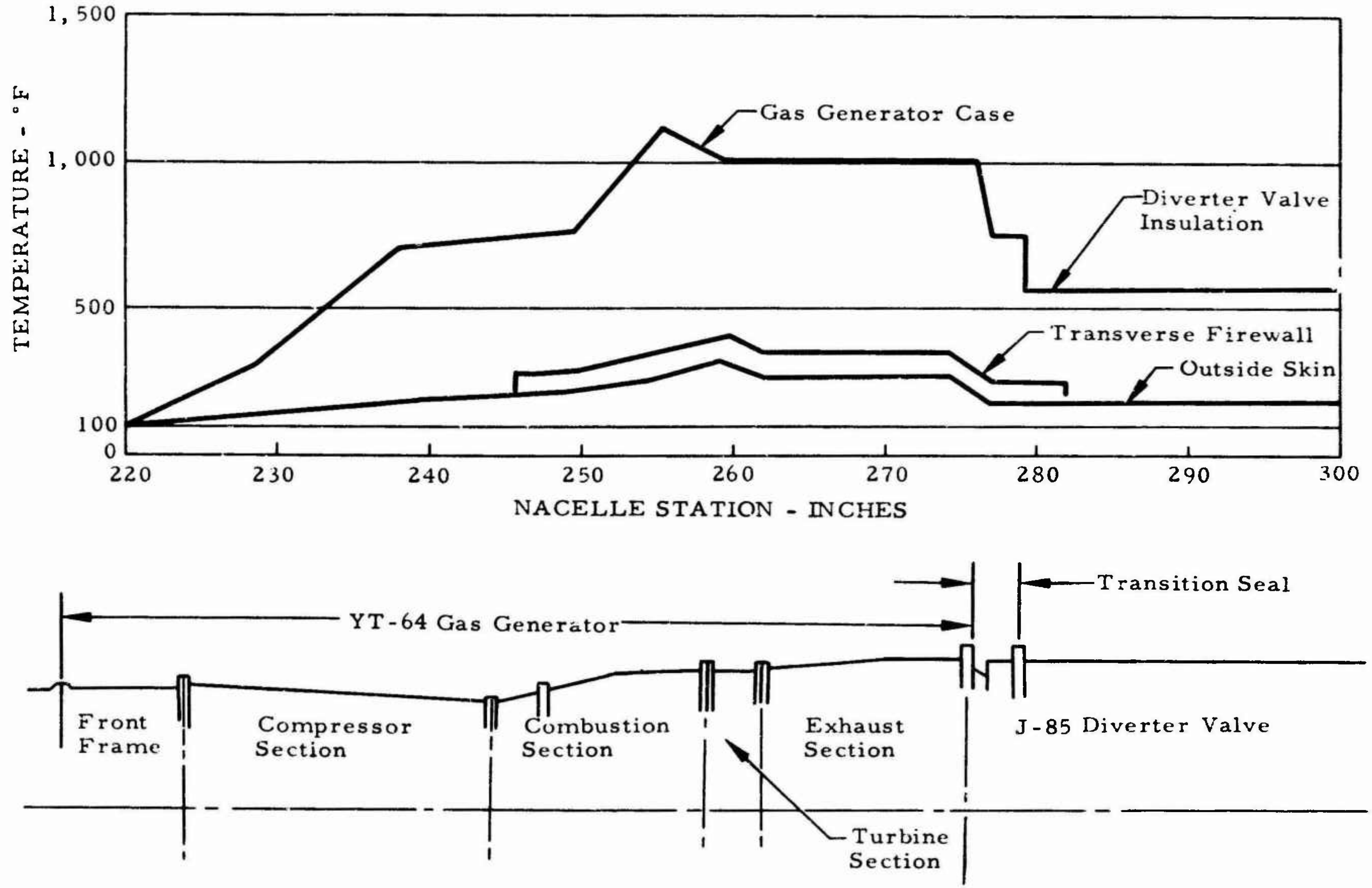


Figure 24. Estimated Nacelle Temperatures

The diverter valve is covered with an insulation blanket. This insulation, the airflow induced by the oil cooling ejector, and the use of high-temperature steel in the nacelle construction make air openings in this area unnecessary. The tailpipe section is designed to operate at elevated temperature.

The transition duct in the pylon is wrapped with insulating material combined with radiation shielding. The heat flow inside the pylon is reduced to a degree where natural convection is sufficient to maintain acceptable structural temperatures. Tests, to date, have demonstrated temperatures of the pylon skins to be within the allowable limits for safe operation. Ventilation holes are provided at the top and bottom skins of the pylon to permit circulation. The pylon is isolated from the nacelle by a firewall baffle that seals around the transition duct insulation.

2.5.9 Cowling

Four accessory cowl panels are utilized to cover each YT-64 gas generator. In addition to providing weather protection for the gas generators, the cowling serves as a means for introducing and directing the cooling airflow through the single-zone gas generator bay.

2.5.10 Fuel System

The airplane fuel system consists of two individual systems, one for each gas generator, with crossfeed feature. Under normal conditions, each gas generator operates with its own system; however, by pilot operation of the system valving, fuel can be made available to both gas generators from either fuel cell or to either gas generator from both fuel cells.

2.5.10.1 Description

Each system consists of a 250-gallon bladder-type rubberized fuel cell, boost pump, shutoff valves, strainers, vents, and drains. The cells are of urethane synthetic and comply with Hughes Tool Company dimensional requirements. The left-hand gas generator system cell is located between fuselage canted Station 200.00 and fuselage Station 256.37. The right-hand gas generator system cell is located between fuselage Stations 377.75 and 430.22. Cell support is conventional, by nylon chord and by bolted connections through the upper access door and sump. Each cell is vented to atmosphere by two 0.625-inch-diameter aluminum alloy lines. The vent outlets are safely located in areas isolated from gas generator exhaust or hot surfaces. The fillers for each cell are located on the left-hand side of the fuselage.

Fuel is supplied to the gas generator through 0.750-inch-diameter aluminum alloy lines in the fuselage and through a 0.750-inch fire resistant flexible line in each gas generator section. The supply to the engine is maintained at continuous pressure by the boost pumps located in each cell. Operation of boost pumps, as well as system shutoff valves, is by 28-volt dc. Each system employs two shutoff valves, one at the tank and one at the firewall. The firewall shutoff valve is also coordinated to close when the fire extinguishing switch is energized.

The system may be drained by gravity at the fuel strainers, or may be disconnected at the engine and pumped out by the boost pumps. The fuel system schematic is shown in Figure 25.

2.5.10.2 Fuel-Hydraulic Heat Exchanger

Gas generator fuel is used to remove the heat generated in the hydraulic system. Two hydraulic fluid-to-fuel heat exchangers are installed to provide independent cooling for primary and utility hydraulic systems. The hydraulic cooling requirements are discussed in Section 2.7.1.

2.6 CONTROL SYSTEMS

2.6.1 Rotor Pitch Control

The rotor pitch control system (see Figures 26 and 27) may be divided into three distinct installations:

a. Pilot linkage

b. Stationary power linkage

c. Rotating linkage

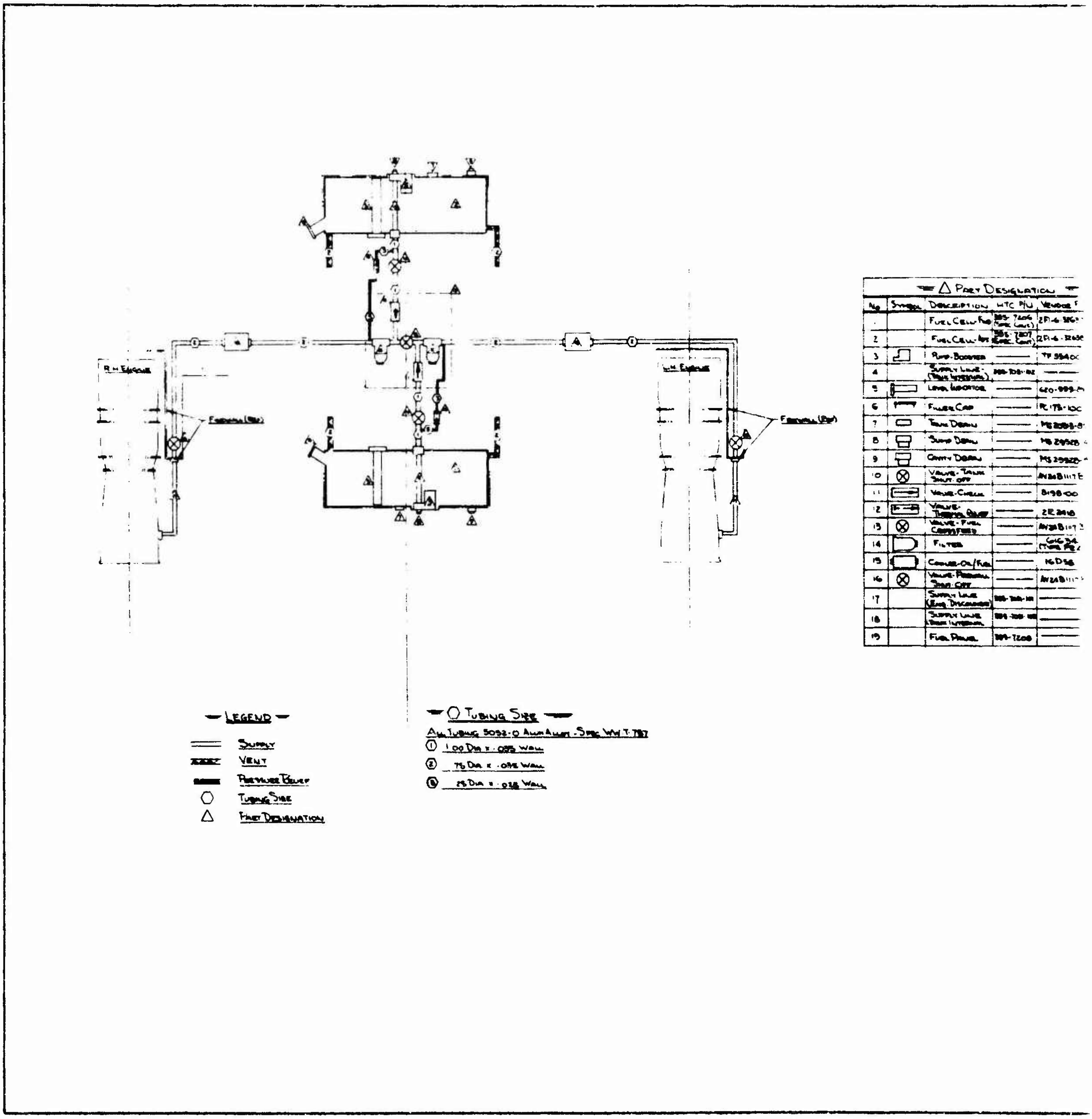
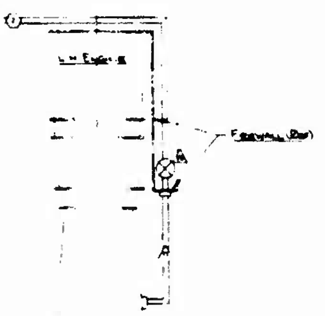


Figure 25. Fuel System Schematic

REVISIONS

A	Revised & Redrawn	
3	Rev as Thermal Relief	



△ Part Designation

No	Symbol	Description	HTC P/N	Vendor P/N	Vendor
		Fuel Cell-Fwd	385-7206 (Tank Unit)	2F-6-32637	Goodyear Tire & Rubber
2		Fuel Cell-Aft	385-7207 (Tank Unit)	2F1-6-32636	Goodyear Tire & Rubber
3		Pump-Booster	———	TF 33400	Thompson Prod.
4		Supply Line (Tank Internal)	385-7051-102	———	———
5		Level Indicator	———	620-999-2508	Liquidometer
6		Filler Cap	———	FC 178-100	Swiss Electro
7		Tank Drain	———	MS 2093-80	———
8		Sump Drain	———	MS 29523-4	———
9		Cavity Drain	———	MS 29523-4	———
0		Valve-Tank Shut-Off	———	AV24B1117B	General Controls
		Valve Check	———	3198-00	Dukes Aviation
2		Valve-Thermal Relief	———	2R 2418	Consave Inc.
3		Valve-Fuel Crossfeed	———	AV24B1117B	General Controls
4		Filter	———	G1G 34 (Type PE 235)	Purolator Inc.
5		Cooler-Oil/Fuel	———	16D58	Janitrol Inc.
6		Valve-Firewall Shut-Off	———	AV24B1117B	General Controls
7		Supply Line (Eng. Disconnect)	385-7051-101	———	———
8		Supply Line (Tank Internal)	385-7051-103	———	———
19		Fuel Probe	385-7208	———	———

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Released 1960 - Information Only

Schematic Diagram - Model 385 Fuel System

HUGHES TOOL COMPANY
AIRCRAFT DIVISION
CULVER CITY, CALIFORNIA

385-7220

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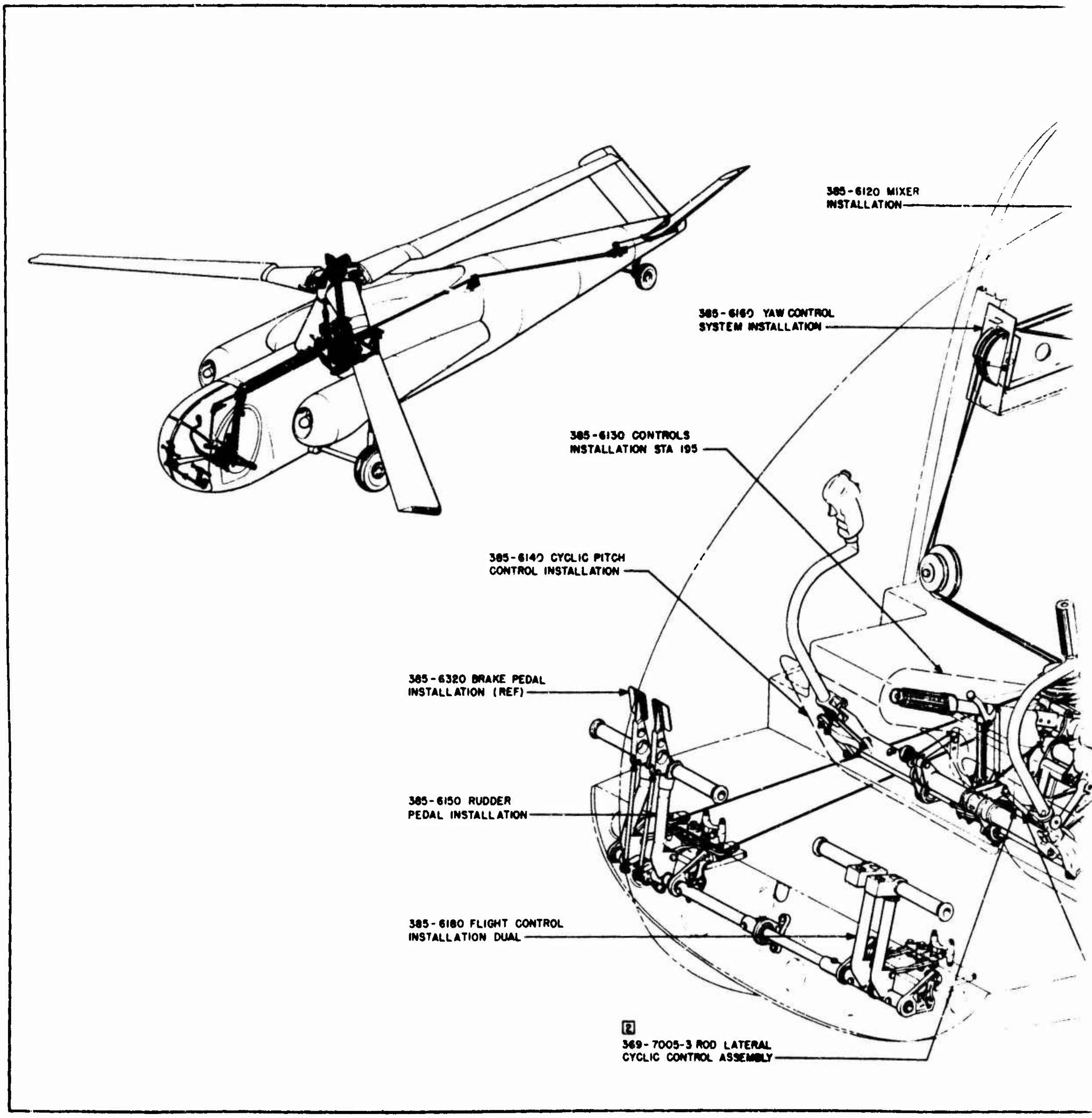
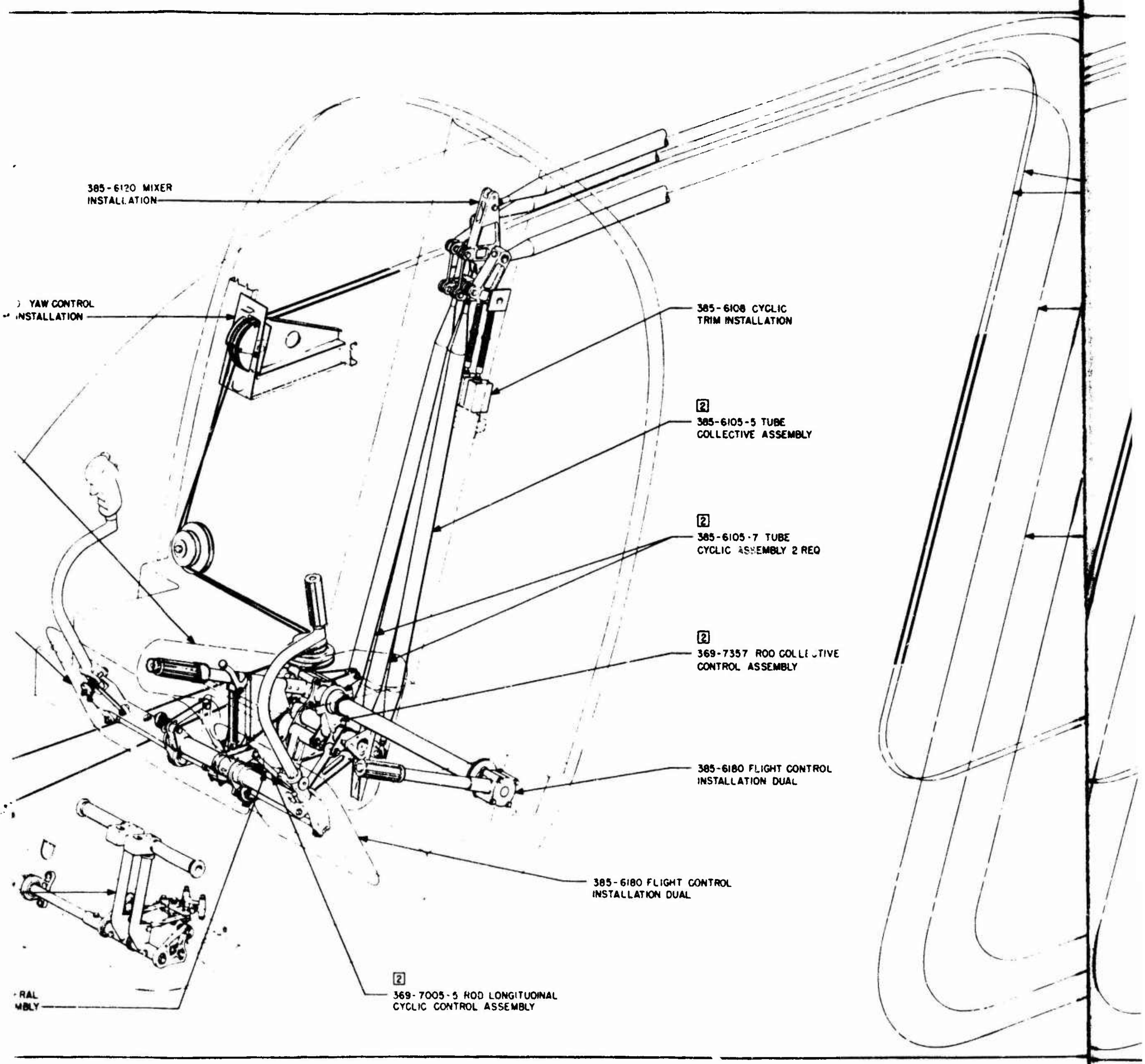


Figure 26. Flight Control System (Sheet 1 of 2)

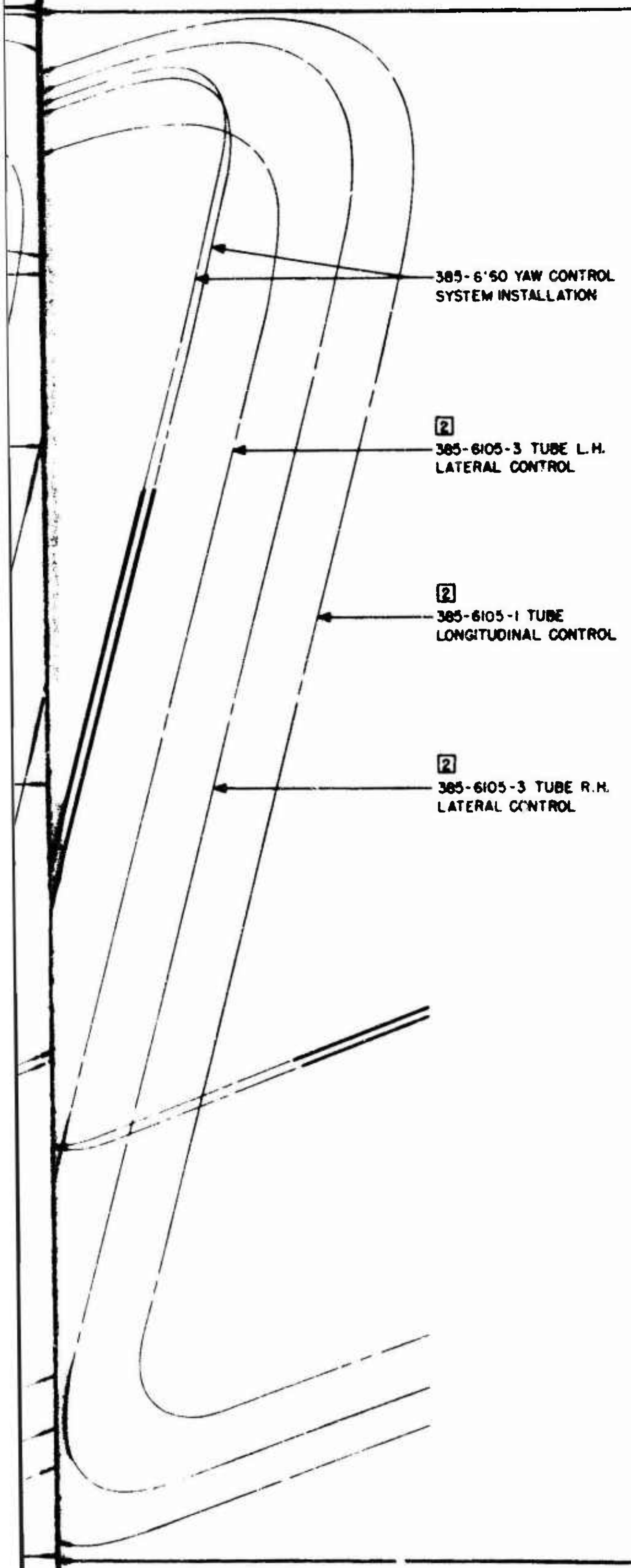
A



385-6120 MIXER INSTALLATION
YAW CONTROL INSTALLATION
385-6108 CYCLIC TRIM INSTALLATION
2
385-6105-5 TUBE COLLECTIVE ASSEMBLY
2
385-6105-7 TUBE CYCLIC ASSEMBLY 2 REQ
2
369-7357 ROD COLLECTIVE CONTROL ASSEMBLY
385-6180 FLIGHT CONTROL INSTALLATION DUAL
385-6180 FLIGHT CONTROL INSTALLATION DUAL
2
369-7005-5 ROD LONGITUDINAL CYCLIC CONTROL ASSEMBLY

B

REVISIONS



	REQ'D	PART NO	REQ'D	PART/NO	NAME
			1	HS626-4-335	BUSHING
			1	HS626-4-310	BUSHING
			9	HS626-4-275	BUSHING
			4	HS626-4-217	BUSHING
			3	HS626-4-165	BUSHING
			1	NAS464P4-19	BOLT
			1	NAS464P4-18	BOLT
			10	NAS464P4-17	BOLT
			2	NAS464P4-16	BOLT
			2	NAS464P4-15	BOLT
			1	NAS464P4-14	BOLT
			1	NAS464P4-11	BOLT
			36	AN960PD416-L	WASHER
			18	AN381-2-12	PIN
			18	AN320-4	NUT
			2	385-6108	CYCLIC TRIM INSTALLATION
			1	385-6106	DRAG LINK INSTALLATION
2			2	385-6105-7	TUBE CYCLIC
2			1	385-6105-5	TUBE COLLECTIVE
2			2	385-6105-3	TUBE LATERAL CONTROL
2			1	385-6105-1	TUBE LONGITUDIAL CONTROL
			1	385-6180	FLIGHT CONTROL INSTALLATION DUAL
			1	385-6170	RUDDER CONTROL INSTALLATION AFT LINKAGE
			1	385-6160	YAW CONTROL INSTALLATION
			1	385-6150	RUDDER PEDAL INSTALLATION
			1	385-6140	CYCLIC PITCH CONTROL INSTALLATION
			1	385-6130	CONTROLS INSTALLATION
			1	385-6120	MIXER INSTALLATION
1			1	385-6110	ROTOR CONTROL POWER LINKAGE INSTALLATION
2			1	369-7357	ROD COLLECTIVE
2			1	369-7005-5	ROD LONGITUDINAL
2			1	369-7005-3	ROD LATERAL
1			1	285-0300	UPPER CONTROLS INSTALLATION
	ASSEMBLY OPP		ASSEMBLY SHOWN		LIST OF MATERIAL

385-6100

NEXT ASSY | USED ON — APPLICATION — QTY REQD

FLIGHT CONTROLS INSTALLATION

HOT CYCLE HELICOPTER

HUGHES TOOL COMPANY — AIRCRAFT DIVISION

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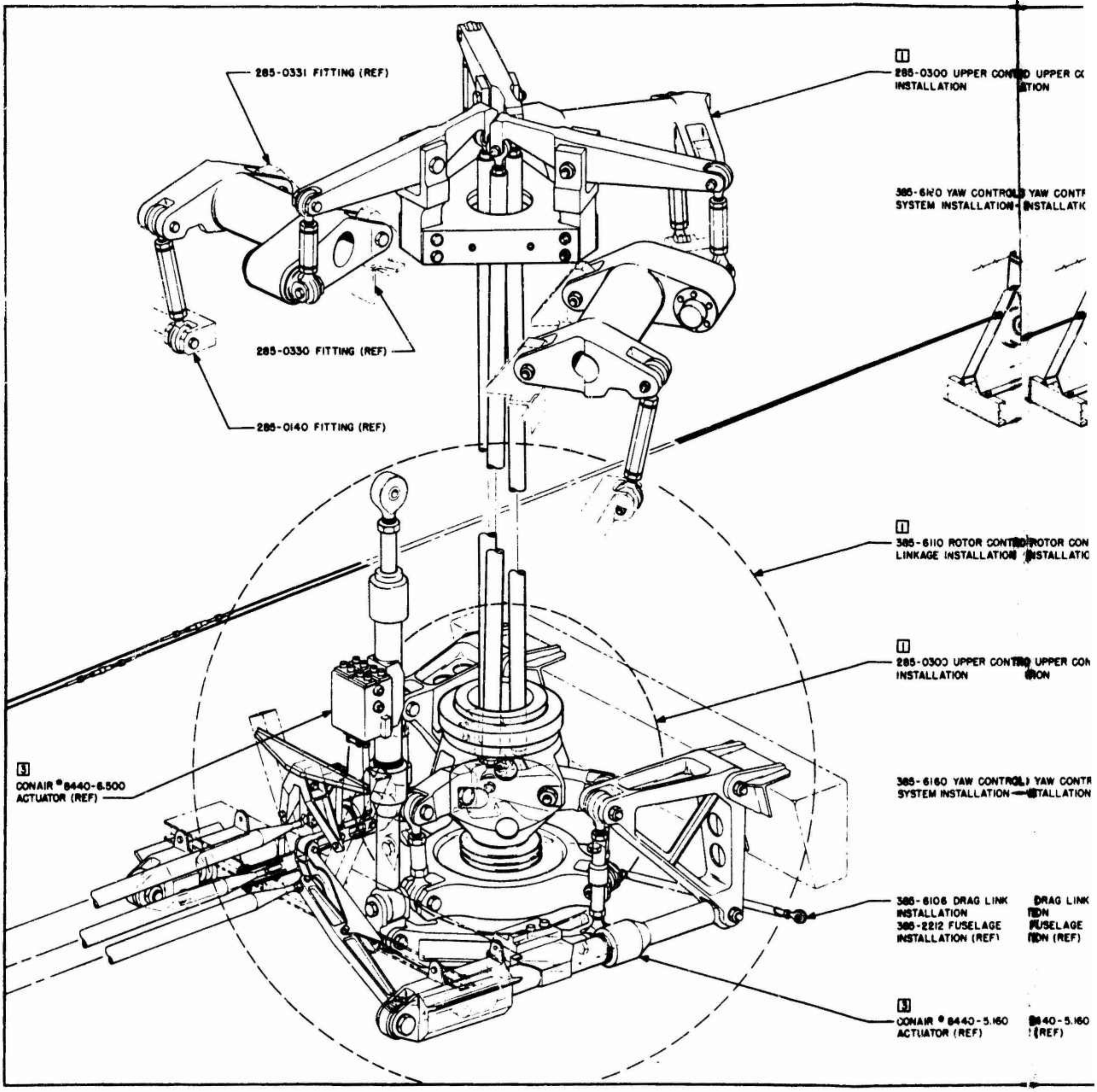
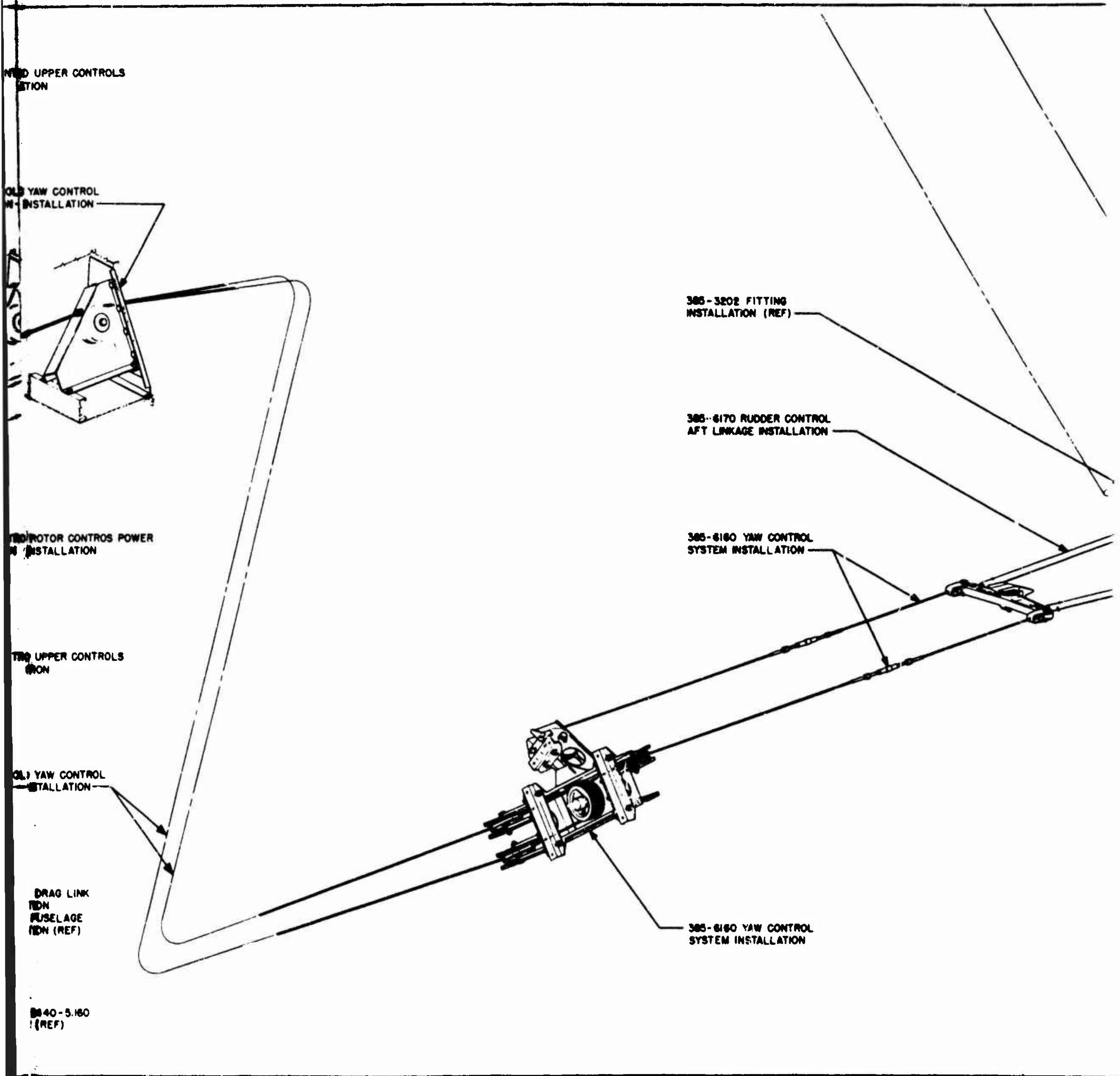
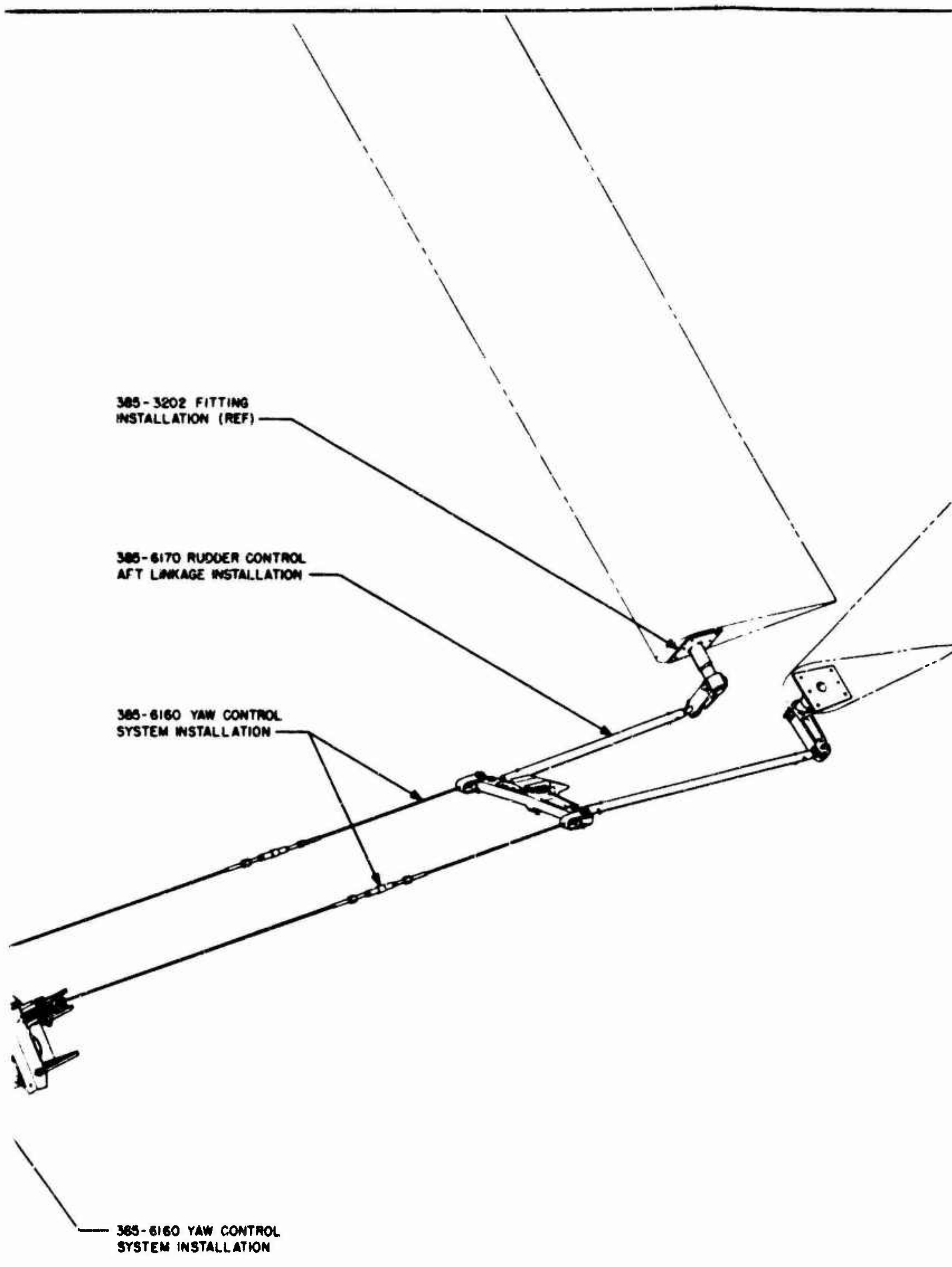


Figure 27. Flight Control System (Sheet 2 of 2)

A


UPPER CONTROLS
YAW CONTROL
INSTALLATION
385-3202 FITTING
INSTALLATION (REF)
385-6170 RUDDER CONTROL
AFT LINKAGE INSTALLATION
ROTOR CONTROS POWER
INSTALLATION
385-6160 YAW CONTROL
SYSTEM INSTALLATION
UPPER CONTROLS
YAW CONTROL
DRAG LINK
385-6160 YAW CONTROL
SYSTEM INSTALLATION
(REF)

B



NOTES:

1. FOR KINEMATICS STUDY SEE 385-6103 AND 385-6104

2. CONTROL RODS AND TUBES ARE TO BE INSTALLED WITH THE FOLLOWING HARDWARE

385-6105-1 385-6105-3			NAS464P4-17 HS626-4-275
385-6105-5	UPPER END		NAS464P4-17 HS626-4-275
	LOWER END		NAS464P4-15 HS626-4-165
385-6105-7	UPPER END	LAT & LONG	NAS464P4-17 HS626-4-275
	LOWER END	LAT	NAS464P4-17 HS626-4-217
	LOWER END	LONG	NAS464P4-18 HS626-4-217
369-7357	UPPER END		NAS464P4-16 HS626-4-335
	LOWER END		NAS464P4-11 HS626-4-217
369-7005-3	FWD END		NAS464P4-15 HS626-4-165
	AFT END		NAS464P4-16 HS626-4-217
369-7005-5	FWD END		NAS464P4-19 HS626-4-4-310
	AFT END		NAS464P4-14 HS626-4-165
COMMON TO ALL			AN320-4 AN960PD416-L AN381-2-12

3. THESE ACTUATORS PURCHASED IN ACCORDANCE WITH HTC SPEC CONTROL DRAWING 385-6109

4. FINISH PER 385-6710. REPAIR PAINT ON DAMAGED SURFACES PER

5. RIG CONTROLS PER ENGINEERING INSTRUCTIONS

C

NEMATICS STUDY SEE 385-6103 AND 385-6104

H THE OL RODS AND TUBES ARE TO BE INSTALLED WITH THE
WING HARDWARE

17 / 75	6105 - 1 / 6105 - 3		NAS464P4 - 17 / HS626-4-275	BOLT / BUSH	6 PLACES
17 / 75	6105 - 5	UPPER END	NAS464P4 - 17 / HS626-4-275	BOLT / BUSH	
15 / 5		LOWER END	NAS464P4 - 15 / HS626-4-165	BOLT / BUSH	
17 / 75	6105 - 7	UPPER END LAT & LONG	NAS464P4 - 17 / HS626-4-275	BOLT / BUSH	
17 / 17		LOWER END LAT	NAS464P4 - 17 / HS626-4-217	BOLT / BUSH	
18 / 17		LOWER END LONG	NAS464P4 - 18 / HS626-4-217	BOLT / BUSH	
16 / 35	7357	UPPER END	NAS464P4 - 16 / HS626-4-335	BOLT / BUSH	
11 / 7		LOWER END	NAS464P4 - 11 / HS626-4-217	BOLT / BUSH	
15 / 5	7005 - 3	FWD END	NAS464P4 - 15 / HS626-4-165	BOLT / BUSH	
16 / 7		AFT END	NAS464P4 - 16 / HS626-4-217	BOLT / BUSH	
19 / -310	7005 - 5	FWD END	NAS464P4 - 19 / HS626-4-4-310	BOLT / BUSH	
14 / 5		AFT END	NAS464P4 - 14 / HS626-4-165	BOLT / BUSH	

ON TO ALL .... AN320-4 NUT
-L AN960PD416-L WASH - 2 REQ
AN381-2-12 PIN

TC SP ACTUATORS PURCHASED IN ACCORDANCE WITH HTC SPECIFICATION
L DRAWING 365-6109

CES PER PER 385-6710. REPAIR PAINT ON DAMAGED SURFACES PER 385-6710

NTROLS PER ENGINEERING INSTRUCTIONS

			1	HS626-4-335	BUSHING
			1	HS626-4-310	BUSHING
			9	HS626-4-275	BUSHING
			4	HS626-4-217	BUSHING
			3	HS626-4-165	BUSHING
			1	NAS464P4-19	BOLT
			1	NAS464P4-18	BOLT
			10	NAS464P4-17	BOLT
			2	NAS464P4-16	BOLT
			2	NAS464P4-15	BOLT
			1	NAS464P4-14	BOLT
			1	NAS464P4-11	BOLT
			36	AN960PD416-L	WASHER
			16	AN381-2-12	PIN
			16	AN320-4	NUT
			2	385-6108	CYCLIC TRIM INSTALLATION
			1	385-6106	DRAG LINK INSTALLATION
2			2	385-6105-7	TUBE CYCLIC
2			1	385-6105-5	TUBE COLLECTIVE
2			2	385-6105-3	TUBE LATERAL CONTROL
2			1	385-6105-1	TUBE LONGITUDIAL CONTROL
			1	385-6180	FLIGHT CONTROL INSTALLATION DUAL
			1	385-6170	RUDDER CONTROL INSTALLATION AFT LINKAGE
			1	385-6160	YAW CONTROL INSTALLATION
			1	385-6150	RUDDER PEDAL INSTALLATION
			1	385-6140	CYCLIC PITCH CONTROL INSTALLATION
			1	385-6130	CONTROLS INSTALLATION
			1	385-6120	MIXER INSTALLATION
1			1	385-6110	ROTOR CONTROL POWER LINKAGE INSTALLATION
2			1	369-7357	ROD COLLECTIVE
2			1	369-7005-5	ROD LONGITUDINAL
2			1	369-7005-3	ROD LATERAL
1			1	285-0300	UPPER CONTROLS INSTALLATION
			REQ'D	PART NO.	LIST OF MATERIAL

385-6100

FLIGHT CONTROLS
INSTALLATION
HOT CYCLE HELICOPTER

02731

385-6100

2.6.1.1 Pilot Linkage

The pilot linkage consists of the cyclic and collective pitch levers, their associated mounting structures, bellcranks, and push rods, the mixer and its support structure, the artificial feel system, and the linkage attached to the power control actuator servo valve spools.

The pilot system in the cockpit is derived from the OH-6A helicopter, and is modified only as required to satisfy different movement and load specifications. The collective lever has been modified to provide the throttle-pitch coordination required by the propulsion control system.

A dual control installation has been incorporated into the cockpit system. The copilot stick installations do not incorporate the friction stops, cyclic trim switches, or complete radio intercom controls.

Stops are attached to the cyclic and collective levers to limit the control lever motion. The stops incorporate friction pads so the pilot can manually adjust stick friction. This is required because the servo spool dynamic forces and friction forces are greater than pilot linkage friction. In addition, the sticks are not weight balanced.

The rotor control mixer, designed specifically for the XV-9A, is used to: Mix collective and cyclic signals, match the otherwise incompatible motions of the OH-6A cockpit linkage and the power linkage, act as direction changing bellcranks at the junction of the cockpit and fuselage, and provide attachment for the artificial feel system.

The mixer consists of a double set of four-bar linkages pivoted on a frame that is acted upon by the collective signal. The configuration is shown in Figure 28. The ratio of mixer input to output motion through each cyclic path may be adjusted in order to obtain the following sets of control lever travel and rotor pitch motions.

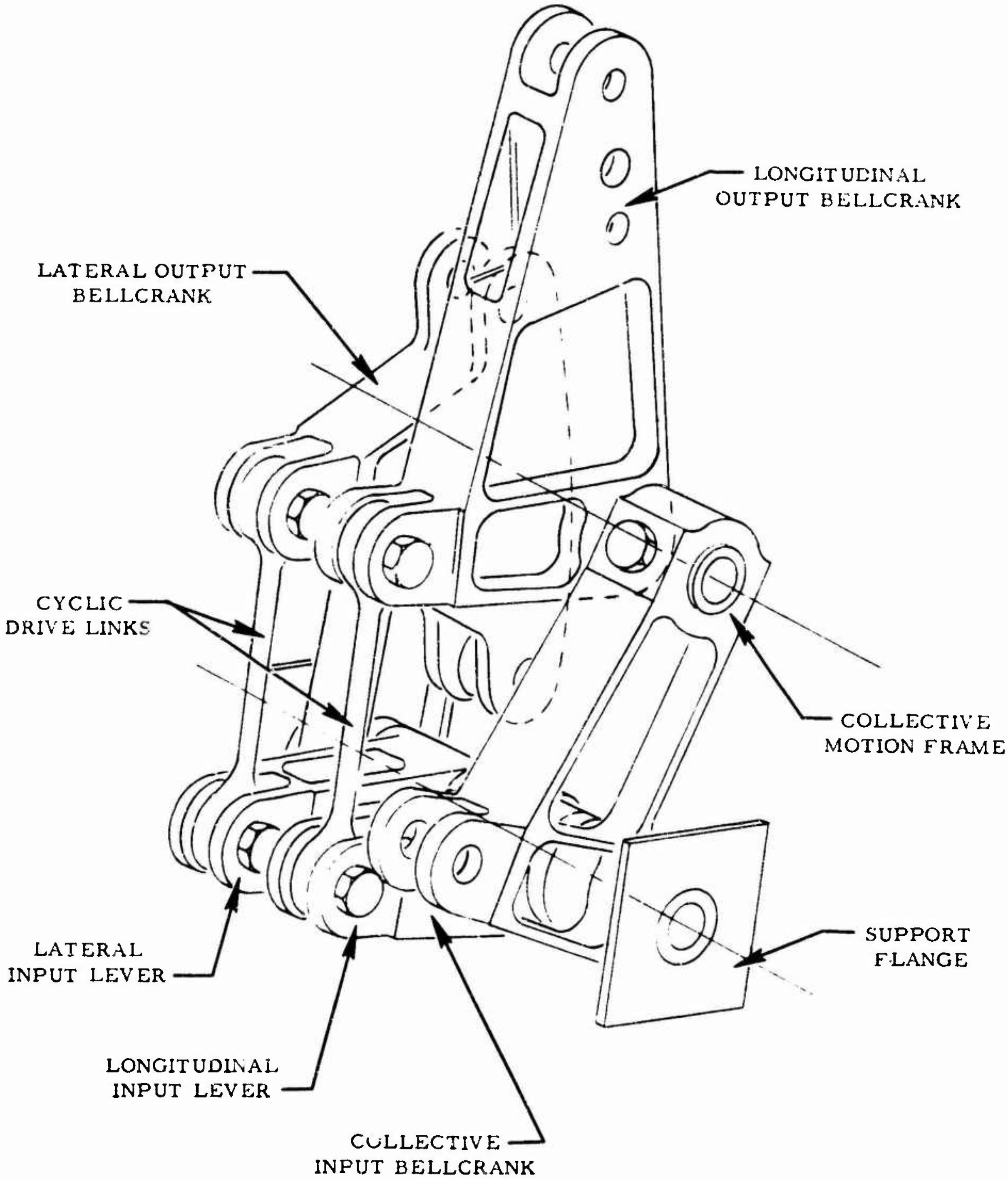


Figure 28. Rotor Control Mixer

	Pilot Lever (Inches)	Blade Pitch (Degrees)
Longitudinal cyclic	$\pm$ 4.5 $\pm$ 5.5 $\pm$ 6.5	$\pm$ 10
Lateral cyclic	$\pm$ 4.0 $\pm$ 5.0 $\pm$ 6.0	$\pm$ 7
Collective	7.5	0 to 12

The artificial feel system, mounted on the aft face of the cockpit bulkhead, applies forces to the two cyclic input levers on the mixer. The force derives from two opposed springs whose fixed end can be adjusted by an electric actuator. The pilot can adjust the stick trim position by use of a cyclic stick mounted switch.

The remainder of the pilot linkage is mounted on the power module front spar, and is used to increase the mixer output motion and adapt to the position and direction of travel of the power control actuator servo valves.

2.6.1.2 Stationary Power Linkage

The stationary power linkage consists of the hydraulic power control actuators and servo valves, the actuator attaching linkage and support structure, the stationary swashplate, and the swashplate drag link.

The forward (longitudinal) actuator, with a stroke of 6.50 inches, is attached to the forward edge of the stationary swashplate through a toggle joint that permits swashplate tilt but prevents rotation of the actuator and consequent misalignment of the servo spool rod. The lateral actuators, with a travel of 5.16 inches, act on the swashplate through offset bellcranks and push rods. This arrangement allows the use of the standard 90-degree T-arrangement of the stationary swashplate, and at the same time permits all the actuators to be supported entirely by the power module structure.

Both the lateral and longitudinal hydraulic actuators are of identical design, and differ only in the length of the replaceable stroke stops. They are balanced tandem cylinder actuators controlled by separate servo valves (see Section 2. 7. 1. 1) fed by independent hydraulic supply systems. The hydraulic power control actuators were designed and fabricated to Hughes Tool Company specifications.

In order to protect the rotor pitch control mechanism from unpredictably high blade pitching moments, a load relief device is placed across each piston of the tandem actuator. This pressure relief device limits the pressure across each piston so the actuating or resisting load is limited to 6, 000 pounds, and also limits the pressure applied across each piston to a nominal 1, 500 psi when both hydraulic supply systems are operative. If either of the two independent hydraulic systems should fail, the pressure limit is automatically reset to a nominal 3, 000 psi.

To restrict the maximum rate of hub tilt, the actuator stroke rate is limited to 5. 90 inches per second. This is accomplished by external hydraulic flow control valves in each supply line. They are set at a maximum flow of 2. 45 gpm. These performance limits are shown in Figure 29 for either or both hydraulic systems in operation.

2. 6. 1. 3 Rotating Linkage

The rotating linkage was designed and fabricated under Contract AF33(600)30271 and has been used with slight modification on the XV-9A. The rotating linkage consists of the rotating swashplate, the swashplate centering spindle, the lower linkage and support housing, the central push rods, the upper linkage and support structure, the hub mounted torque tubes, and the instrumented blade pitch links.

In designing the rotating controls, an attempt was made to make the system as conventional and maintenance-free as possible. The components were located so as to keep them accessible and as far as possible from hot gas components.

As the rotating system is subjected to high cyclic stresses, the control system bearings were selected on a very conservative basis. Available bearings were evaluated in terms of limit loads, weighted fatigue loads, and past field performance of the bearings. Reduced clearance, high-quality bearings were specified for all locations.

Bearing forks have been oriented to minimize misalignment in order to increase bearing life and permit the use of standard $\pm$ 10-degree misaligning bearings in most places. In order to avoid multiple

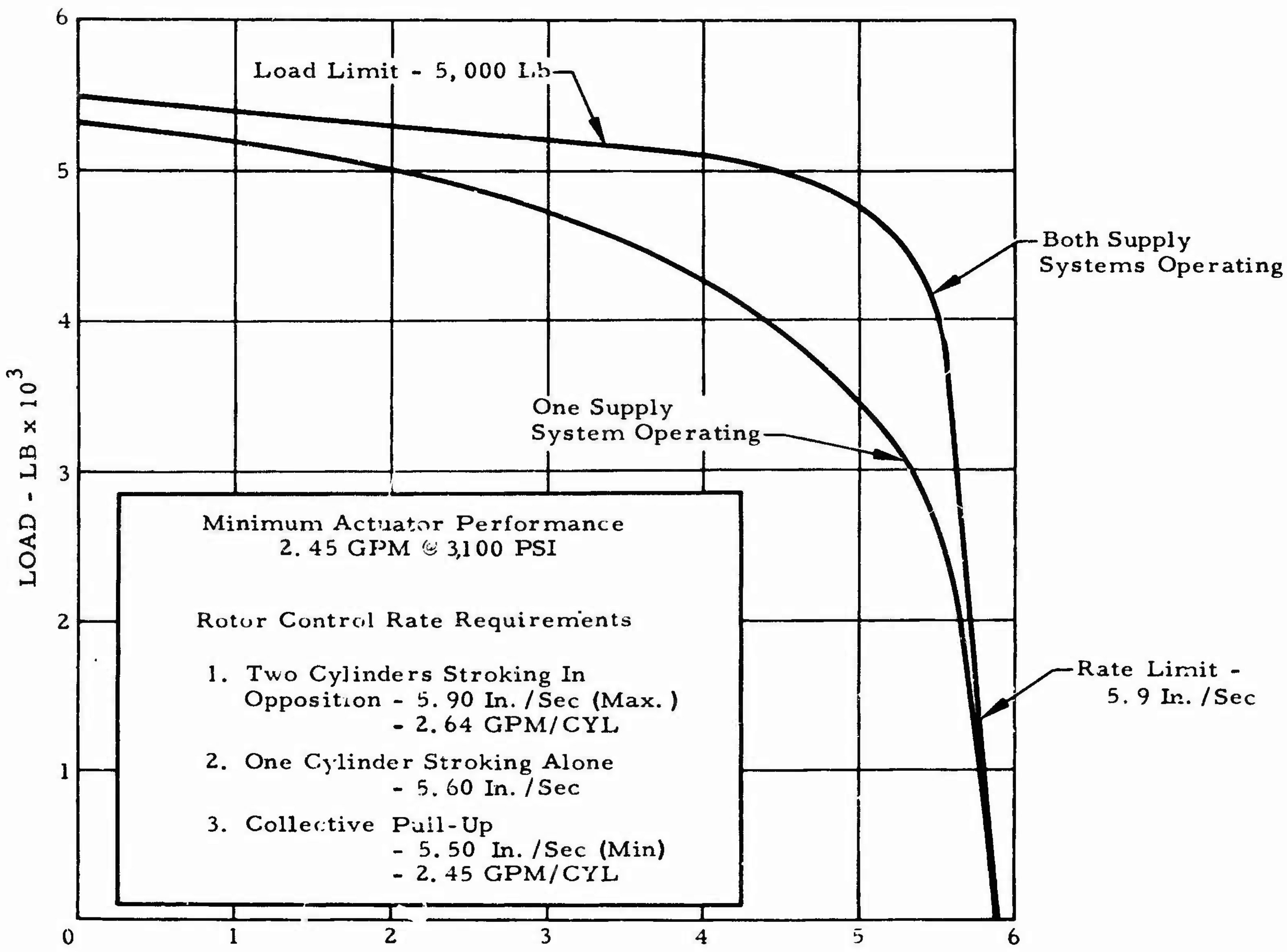


Figure 29. Hydraulic Power Control Actuator Performance

bearing installations, self-aligning roller bearings were used wherever possible. At only one point in the system was the misaligning angle too large for use of a single bearing; a multiple bearing joint was used in this instance. This is the bearing on the inboard end of the upper torque tube mounted onto the rotor hub.

A lubrication fitting is provided for each bearing. With the present configuration, none of the bearings operates at over 200°F. For this reason, conventional low-temperature greases are used. MIL-G-25537 lubricant, especially developed for helicopter bearings with small oscillations, is used on all but the swashplate bearings. MIL-L-7711 lubricant, compounded for rotating helicopter components, is used on the swashplate bearings.

2.6.2 Yaw Control System

The yaw control system consists of rudder pedals, support structure, pedal output linkage, cable system, valve drive installation, and rudder drive linkage (Figures 26 and 27). The yaw valve and the rudder are mechanically linked together. They move simultaneously, although yaw force is produced by the yaw valve during hovering and forward flight and by the rudder during forward flight only.

The yaw control system has been designed for $\pm$ 3.0 inches of rudder travel to obtain full valve rotation of $\pm$ 58 degrees and full rudder deflection of $\pm$ 20 degrees. There is no provision for ratio adjustment. Dual controls are provided.

2.6.3 Power Control System

Each gas generator incorporates a hydromechanical fuel control assembly that automatically regulates fuel flow in accordance with the throttle setting called for by the pilot. The gas generator power control system will enable the pilot to satisfy the hot gas requirements of the lifting rotor through the control of the outputs of the two gas generators. The control system provides for:

a. Independent (individual) gas generator power control

b. Simultaneous (twin) gas generator power control

c. Overriding of collective stick power input

d. Limitation of power reduction to preset minimum level

e. Separate power-matching capabilities for each gas generator

f. Power control by either pilot or copilot

2.6.3.1 Rotor Speed Governing System

The power control system featuring rotor speed governing consists of two principal installations: (a) the speed sensing feedback link (N_f link) to drive the fuel control governor (Figure 30), and (b) the mechanical power control linkage to set and adjust gas generator power and set rotor rpm (Figure 31). For the T-64 engine, N_f represents power turbine speed with respect to the fuel control; however, on the Hot Cycle system the rotor is the power turbine, and therefore N_f represents rotor speed with respect to the YT-64 gas generator fuel control.

2.6.3.1.1 Speed Sensing Feedback Link

There are two N_f links, one for each gas generator; except for minor installation differences, they are identical. Each N_f link consists of a hydraulic pump driven by the accessory gearbox, the transmission line, and the hydraulic motor that drives the gas generator N_f flyball governor (Figure 32).

The rotor driven pump and its hydraulic coupled motor are not synchronous, due to internal leakage of pump and motor and due to flow through the micrometer bleed valve. The rotor driven pump runs at 4,200 rpm for 100 percent rotor speed, while the governor driving motor operates at a nominal 3,660 rpm for 100 percent rotor speed. This speed difference, and consequent bypass flow, is adjusted by the bypass valve to provide governor speed adjustment and to make up for changes in pump and motor internal leakage. This system has been tested as a breadboard test assembly (Reference 9) in order to verify dynamic characteristics. Test results indicated that the system is mechanically sound in all respects.

2.6.3.1.2 Mechanical Power Control System

The mechanical power control linkage has been defined by Figure 31. It is a manually operated arrangement of driving and

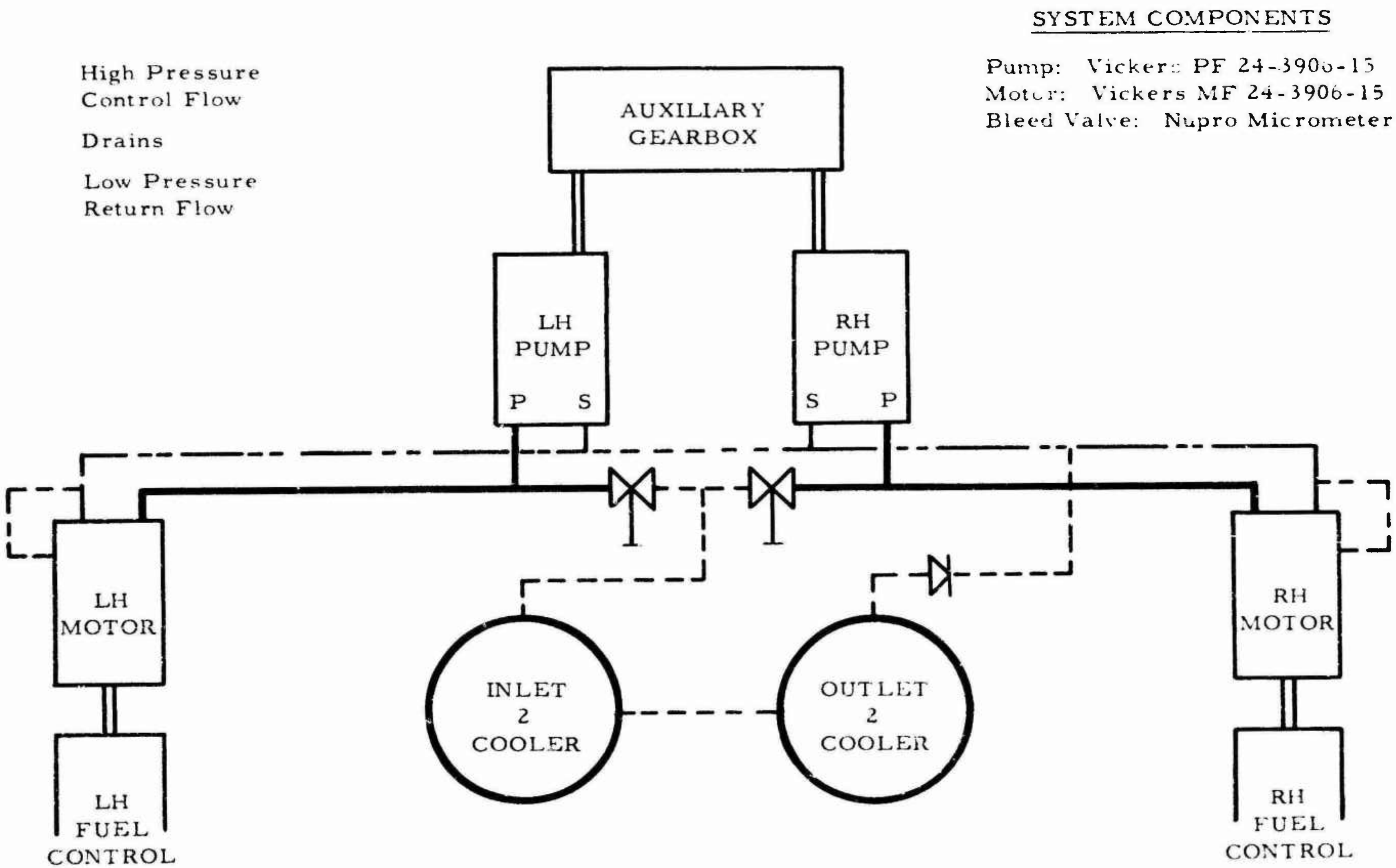


Figure 30. Schematic - Gas Generator Power Control System - N_f Link

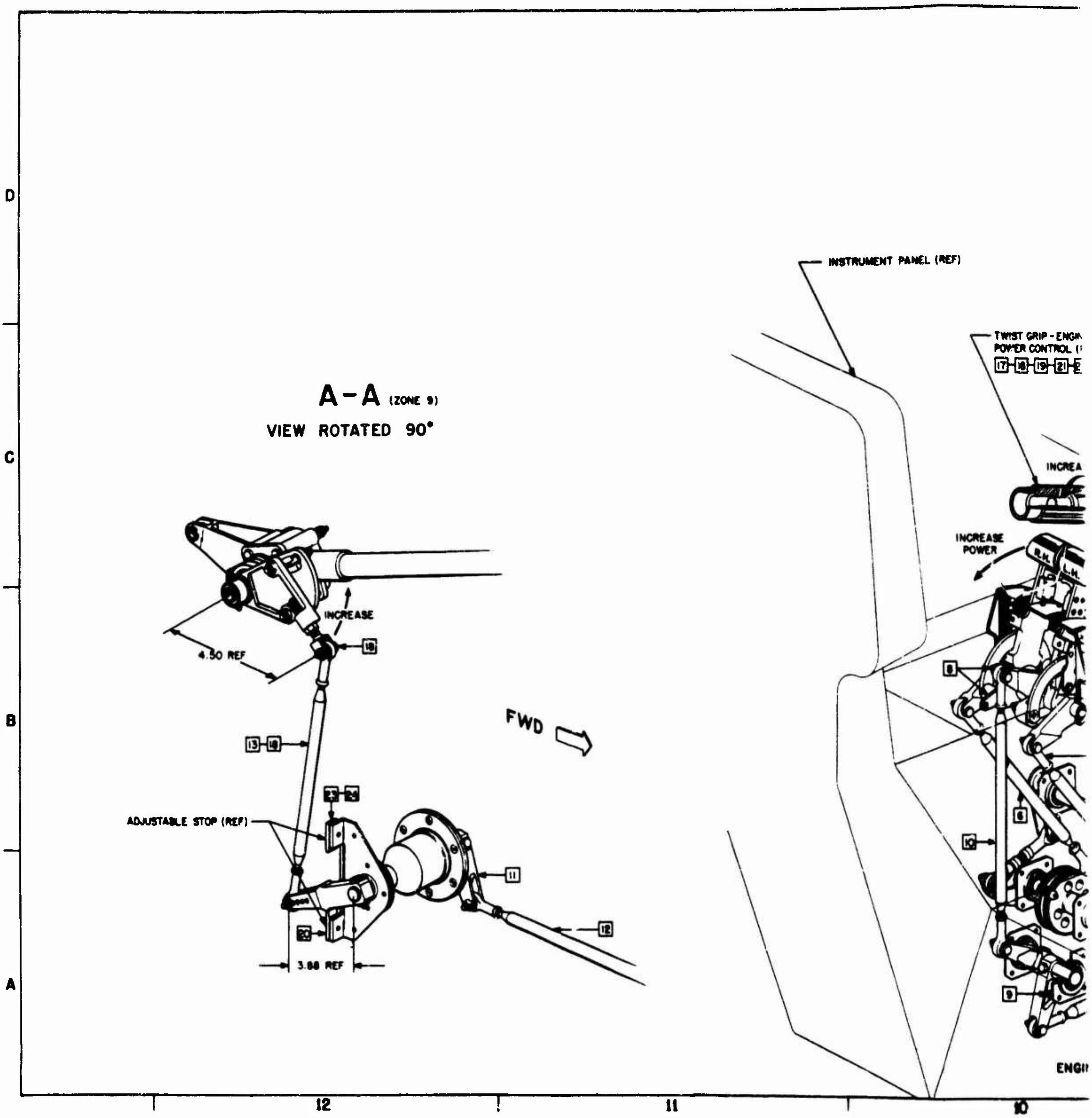
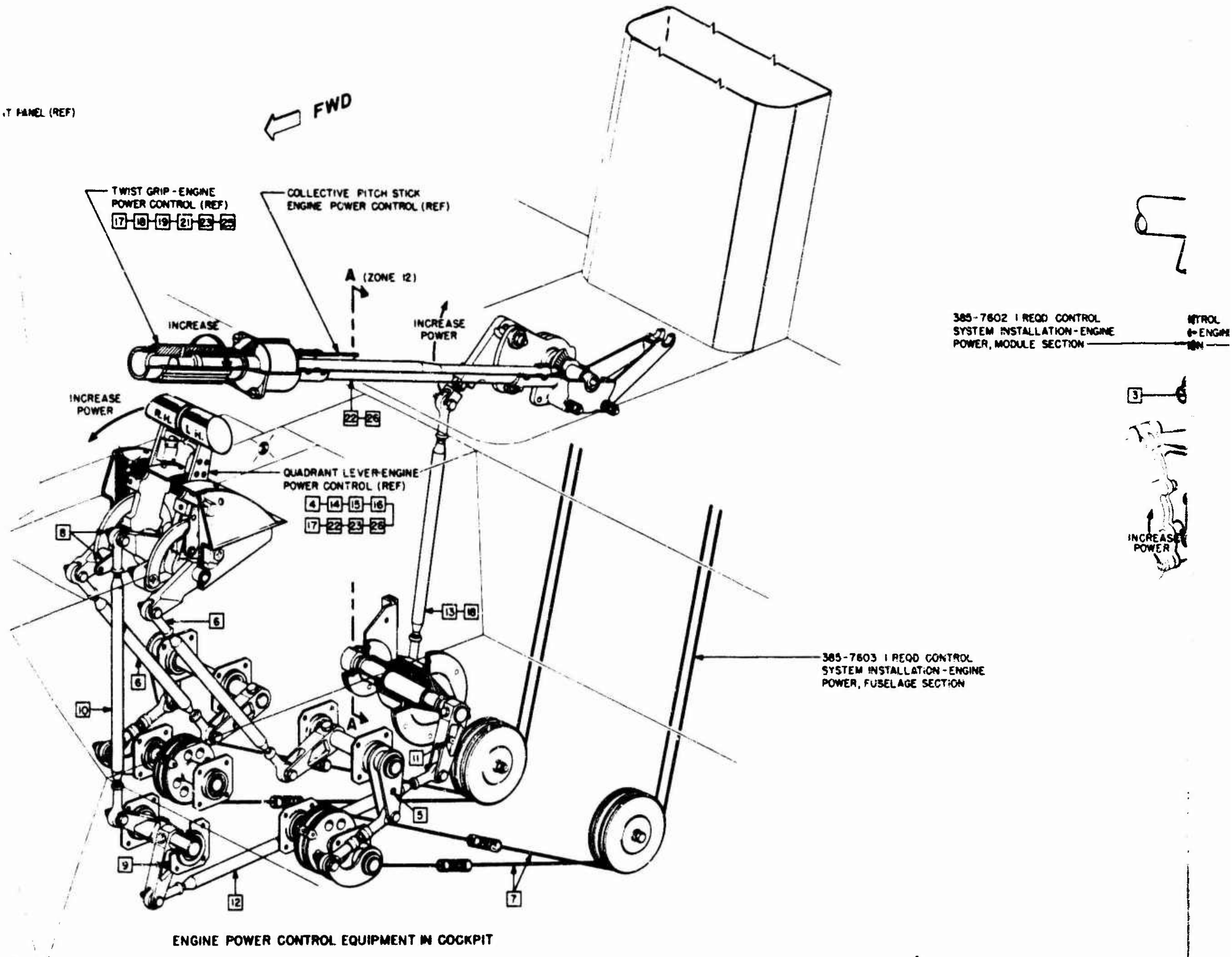


Figure 31. Gas Generator Power Control System - Mechanical Linkage

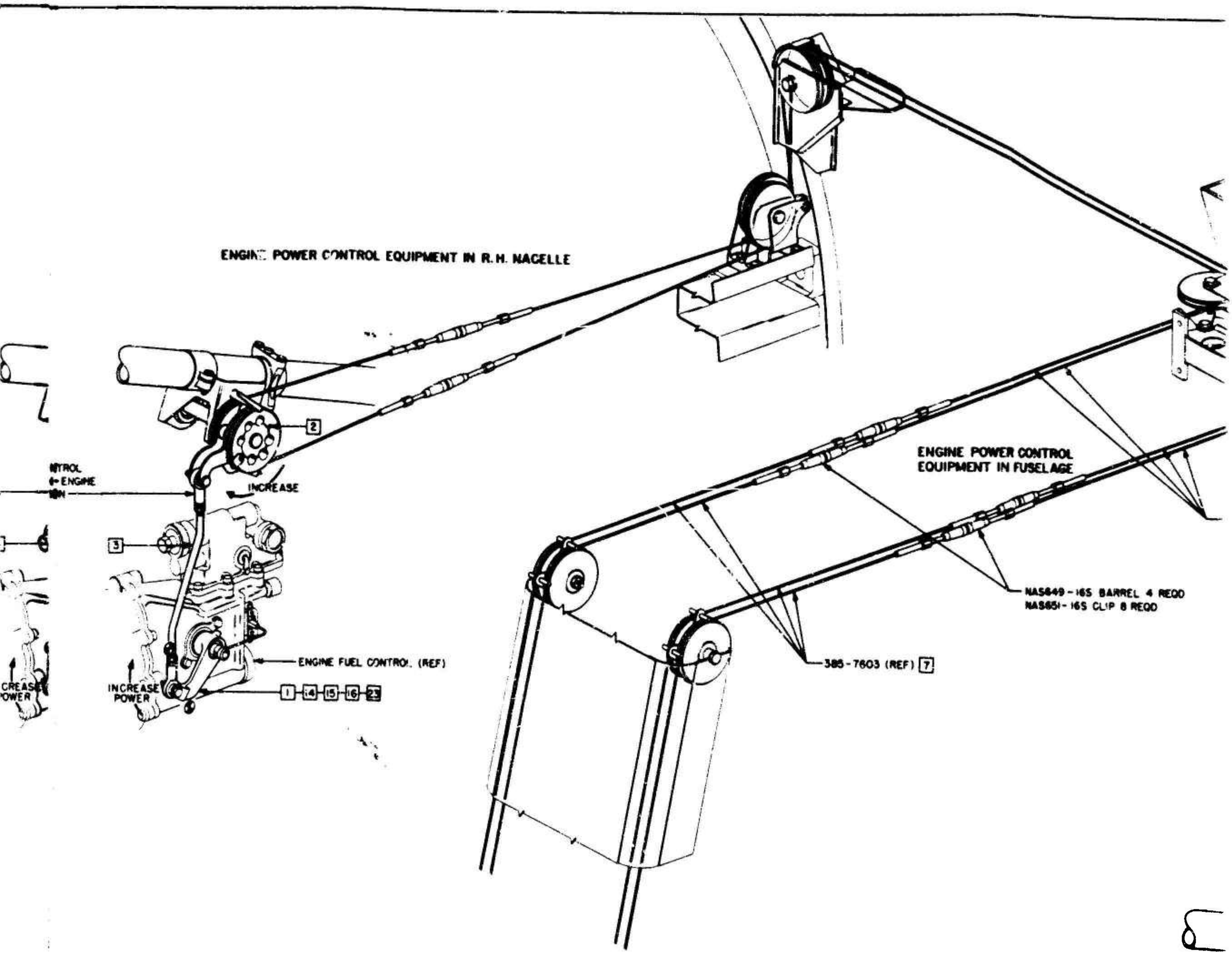
A



ENGINE POWER CONTROL EQUIPMENT IN COCKPIT

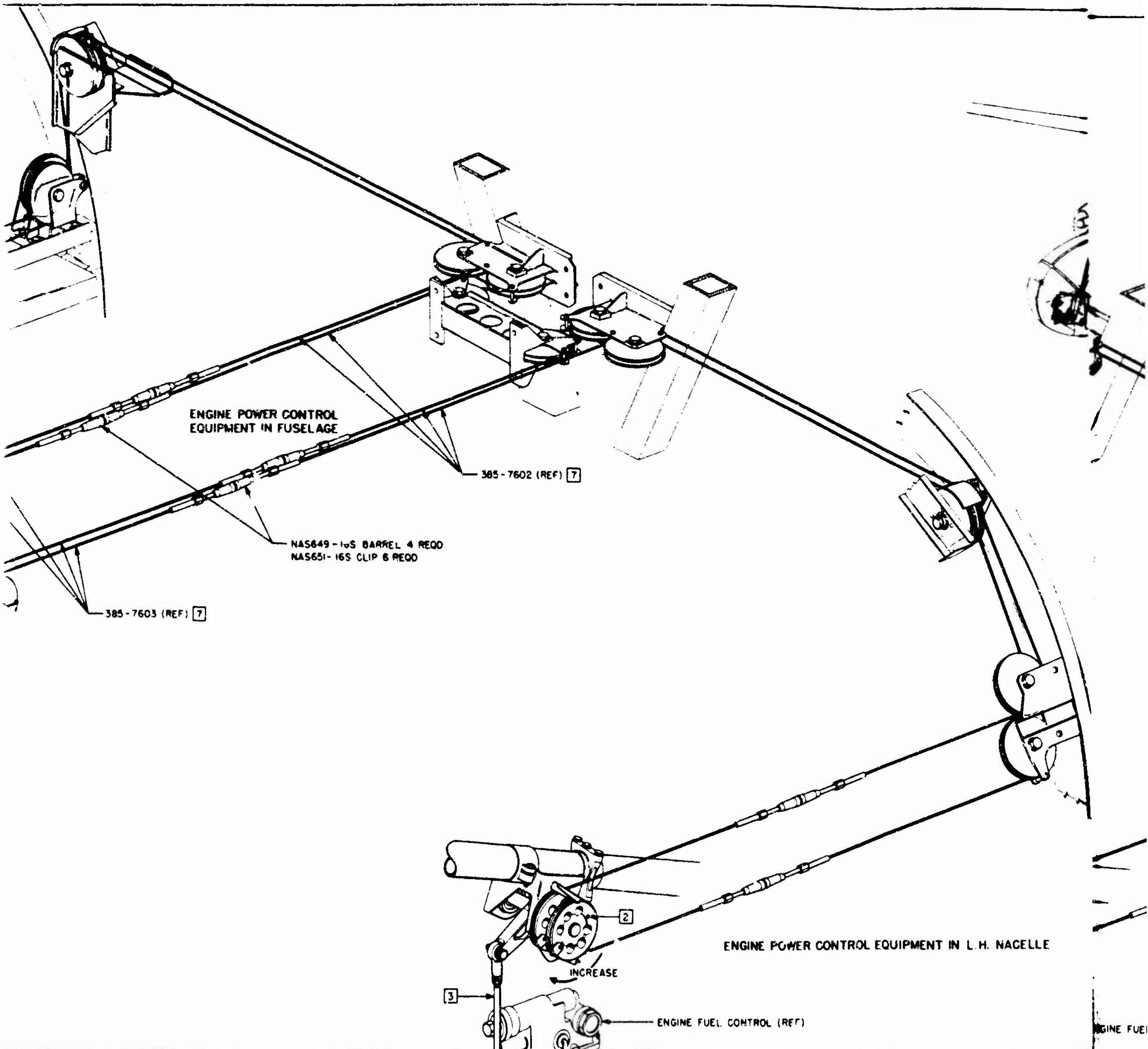
10 9 8

B


ENGINE POWER CONTROL EQUIPMENT IN R.H. NACELLE
2
INCREASE
3
ENGINE FUEL CONTROL (REF)
INCREASE POWER
1 14 15 16 23
ENGINE POWER CONTROL EQUIPMENT IN FUSELAGE
NAS649-16S BARREL 4 REQD
NAS651-16S CLIP 8 REQD
385-7603 (REF) 7
3

7	6	5

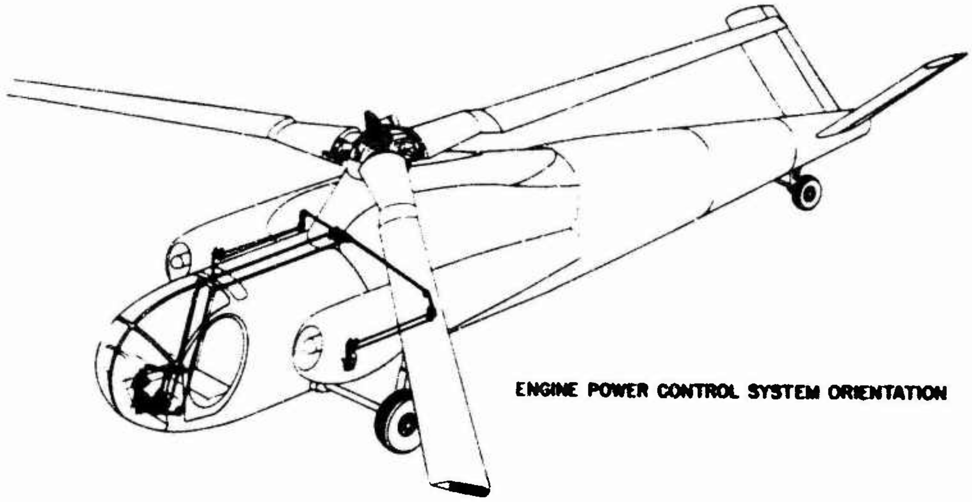
C



ENGINE POWER CONTROL EQUIPMENT IN FUSELAGE
385-7602 (REF) 7
NAS649-10S BARREL 4 REQD
NAS651-16S CLIP 6 REQD
385-7603 (REF) 7
2
ENGINE POWER CONTROL EQUIPMENT IN L.H. NACELLE
INCREASE
3
ENGINE FUEL CONTROL (REF)

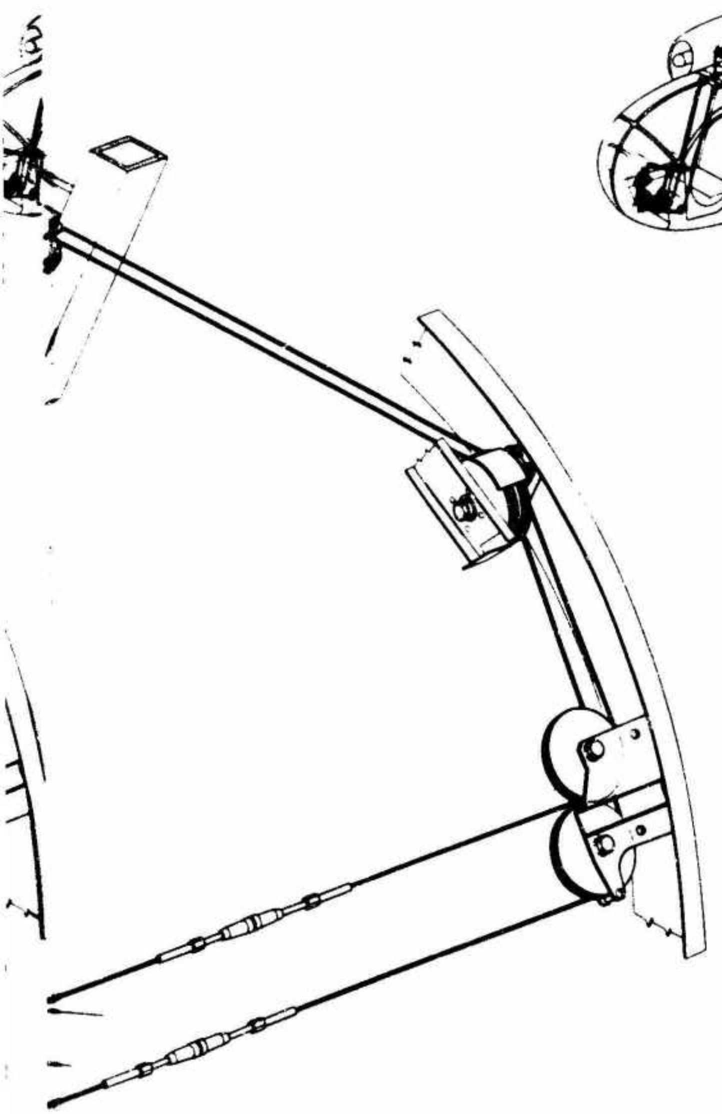
5 4 3

D



ENGINE POWER CONTROL SYSTEM ORIENTATION

REVISIONS						
SYM		DESCRIPTION			DATE	
C		SEE S-4E 5				



ENGINE POWER CONTROL EQUIPMENT IN L.H. NACELLE

NGINE FUEL CONTROL (REF)

NOTE: UNLESS OTHERWISE SPECIFIED

1. INSTALL CABLE PER SPECIFICATION HP-6
2. $\boxed{2}$ NUMBER IN BOX DENOTES RIGGING STEPS SEE SHEET 3 OF 3 FOR PROCEDURE
3. RIG CABLE INSTALLATION TO 46-50 LB TENSION AT 70° F

385-7601 C

		8	NAS651-168	CLIP		
		4	NAS649-16S	BARREL		
		1	385-7603	INSTALLATION - FUSELAGE		
		1	385-7602	INSTALLATION - MODULE		
			385-7601	INSTALLATION		
				LIST OF MATERIAL		

CONTROL SYSTEM INSTALLATION ENGINE POWER

HOT CYCLE HELICOPTER

02731

385-7601

SHEET 1 OF 3

3 | 2 | 1

D C B A

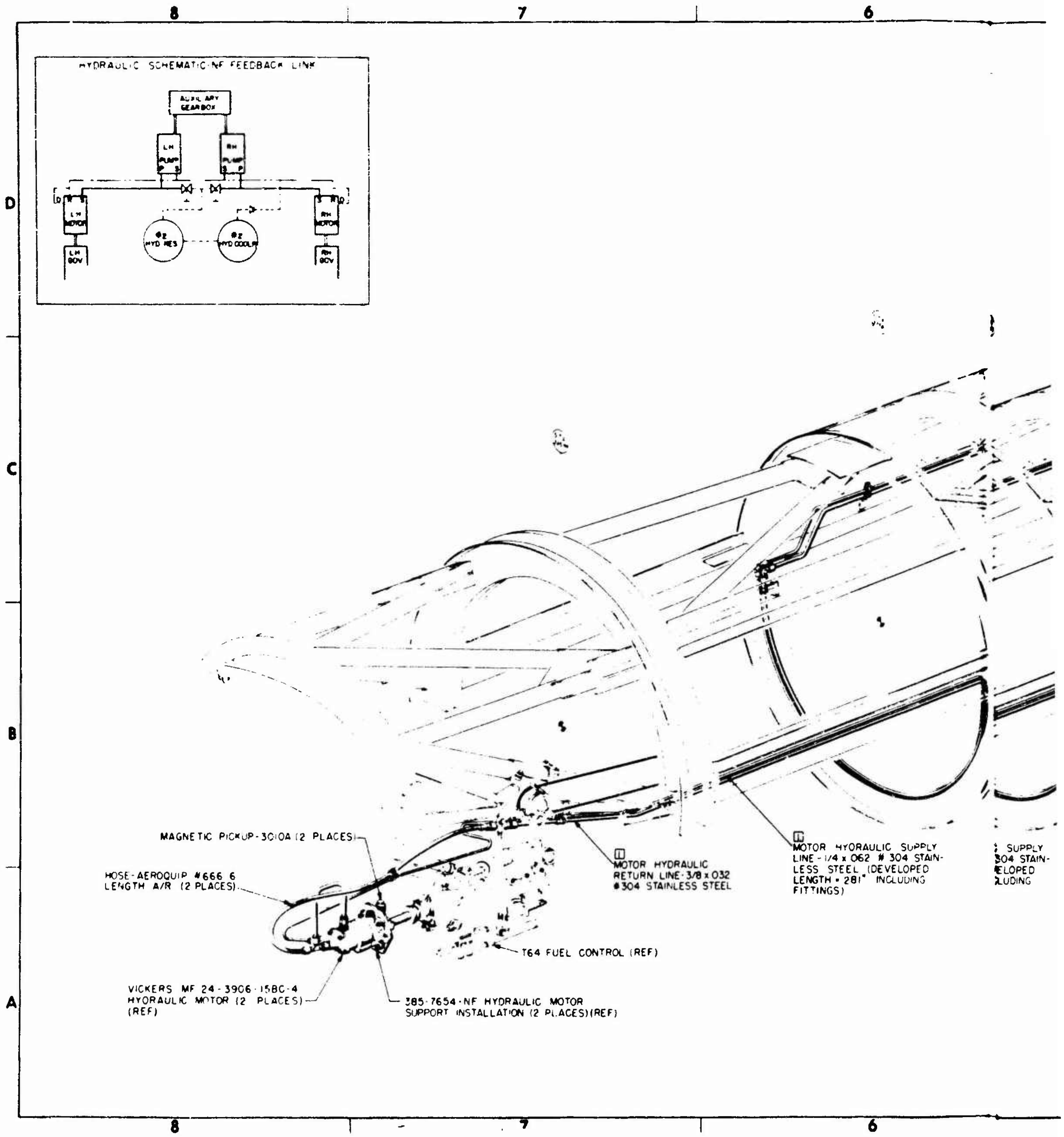
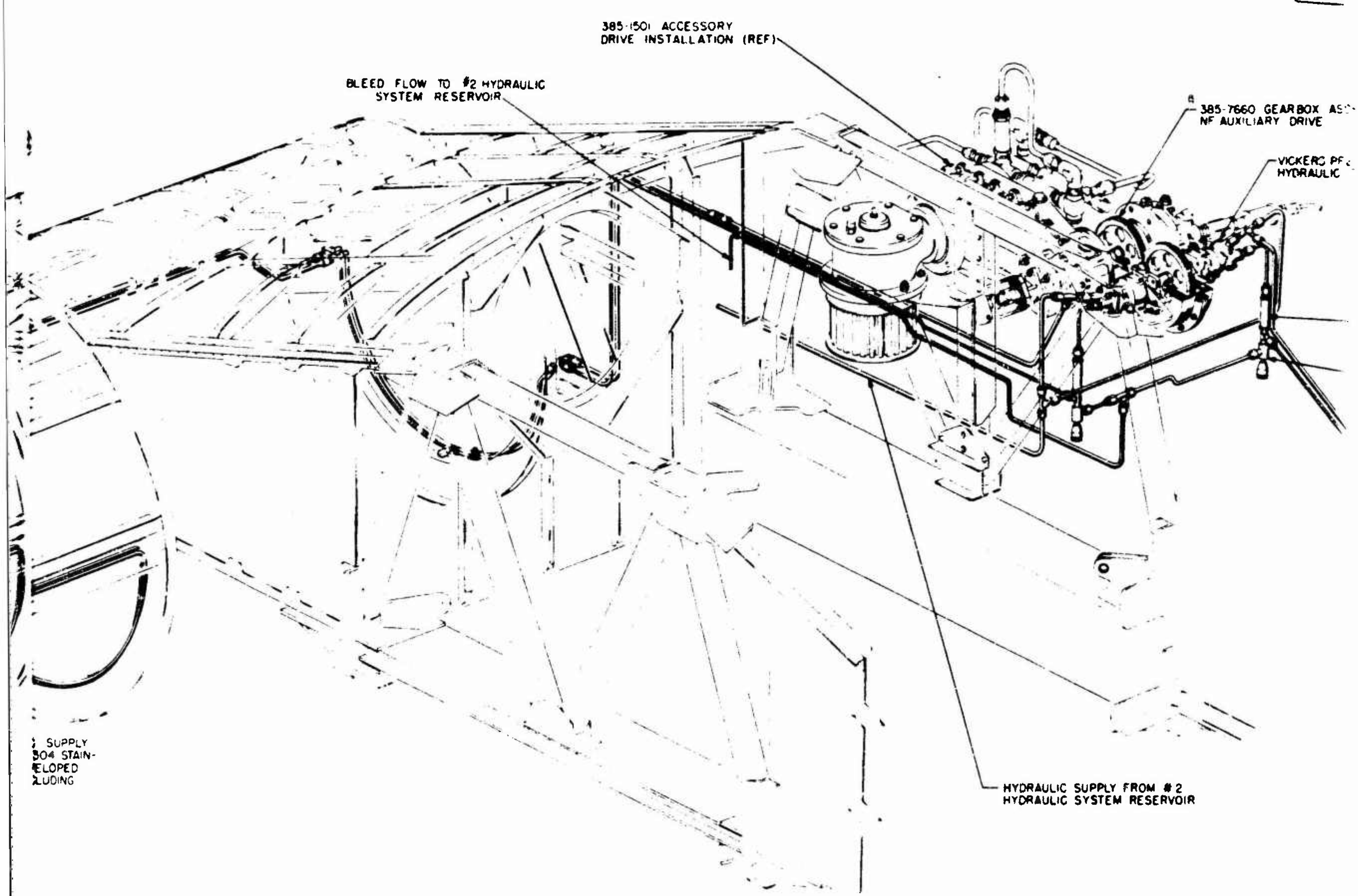


Figure 32. Gas Generator Power Control System - N_f Link

A

5 4 3

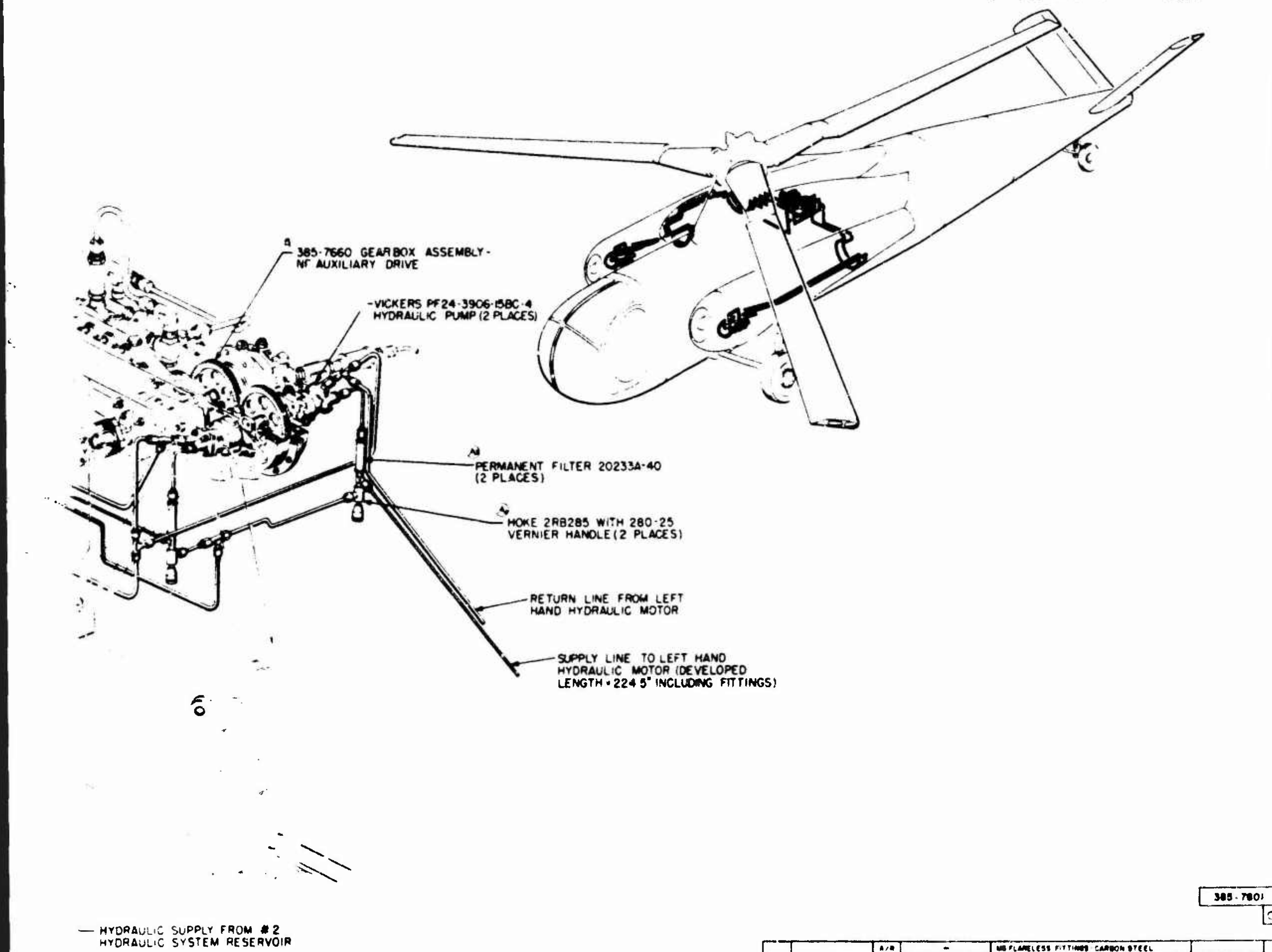


385-1501 ACCESSORY
DRIVE INSTALLATION (REF)
BLEED FLOW TO #2 HYDRAULIC
SYSTEM RESERVOIR
385-7660 GEARBOX AS
NF AUXILIARY DRIVE
VICKERS PF
HYDRAULIC
SUPPLY
304 STAIN-
ELOPED
LUDING
HYDRAULIC SUPPLY FROM #2
HYDRAULIC SYSTEM RESERVOIR

5 4 3

B

REVISIONS						
SYM	E.O.	DESCRIPTION	DRWN	APP'D	DATE	ZONE
A		WAS [illegible]				
A_1		[illegible]				
B_1		ADDED 385-7660 GEARBOX				
		SEE SHEET [illegible]				



385-7601

C

4. EQUIPMENT & HARDWARE PURCHASED ON AMO'S 300806, 300808, 300804, 300801, & PR 005012
3. ALL PUMP SEAL DRAINS TO BE MANIFOLDED AND A DRAIN LINE LED OVERBOARD
2. CLIP TUBES TO STRUCTURE PER ENGINEERING INSTRUCTIONS
1. TUBE ROUTING APPROXIMATELY AS SHOWN - DEVELOP ON INSTALLATION - USE MS FLARELESS FITTINGS THROUGHOUT EXCEPT FOR ENGINEERING APPROVED DEVIATIONS

NOTES UNLESS OTHERWISE SPECIFIED

REQ'D	PART NO	REQ'D	PART NO	NAME	SIZE	DESCRIPTION	SPECIFICATION	ZONE
		A/R	—	MS FLARELESS FITTINGS CARBON STEEL				
		A/R	—	AUXILIARY TUBES	3/8 x .020	304 SS SEAMLESS	MIL-T-6845	
		A/R		AUXILIARY TUBES	1/4 x .020	304 SS SEAMLESS	MIL-T-6845	
		A/R		RETURN TUBES	3/8 x .020	304 SS SEAMLESS	MIL-T-6845	
		A/R		SUPPLY TUBES	1/4 x .062	304 SS SEAMLESS	MIL-T-6845	
		A/R	666-6	HOSE	AEROQUIP, INC			
	A_1	8	20834-40	FILTER	20834-40	PERMANENT FILTER		
		2	3010-A	MAGNETIC PICKUP	ELECTRO			
	A	2	2RB869	NEEDLE VALVE	HOKE			
		1	385-7660	GEAR BOX N_F DRIVE				
		2	PF24-3906-49C-8	PUMP	VICKERS, INC			
ASSEMBLY OPP		ASSEMBLY SHOWN		LIST OF MATERIAL				

CONTROL SYSTEM INSTALLATION ENGINE POWER HOT CYCLE HELICOPTER

HUGHES TOOL COMPANY AIRCRAFT DIVISION

FED. CODE IDENT NO 02731

385-7601

SCALE NONE

SHEET 2 of 3

NEXT ASSY | USED ON | APPLICATION | QTY REQD

C

driven levers and linkages whose movements are transmitted by flexible cables to the power lever of the gas generator mounted fuel control. Movement of the power lever to a higher or lower power setting, as required by flight conditions, can be made by the pilot through either the gas generator control quadrant assembly, the collective stick, or the collective stick twist grip.

With the rotor speed governing system, the normal operating range of the engine fuel control is between power control shaft angles of 76.5 degrees and 121.5 degrees. In this range, the fuel control will automatically vary engine power between flight idle and military, to match load power at rotor speed. The gas generator fuel control load signal shaft is locked in the 90-degree position to obtain the maximum gas generator power for any throttle setting.

The gas generator power control quadrant assembly was manufactured to Hughes Tool Company specifications. This control

a. Transmits the desired power and/or speed setting to either or both of the fuel control units

b. Overrides any power setting initiated by the collective stick travel or twist-grip rotation

c. Limits power reduction to a minimum preset gas generator speed

d. Employs a self-locking principle to permit infinite positioning and prevent creepage or change of setting due to vibration or feedback forces

e. Provides a visual check of the gas generator control setting

The power control quadrant levers will be used to individually start, idle, and stop the gas generators, and to conduct operational checks. The assembly incorporates stops at gas generator idle speed for each engine. Once the rotor is in operation, changes to gas generator power setting will be normally made by collective stick or by collective stick twist grip. However, individual gas generator power may be adjusted by the quadrant. A reverse locking clutch mechanism is incorporated in the linkage from the quadrant to the collective stick. This clutch will transmit motion from the collective stick to the quadrant. The irreversible mechanism in the clutch assembly prevents motion originating at the quadrant from being transferred to either the collective stick or the twist grip.

Movement of the collective stick will accomplish two things; it will change rotor pitch angle, as well as power lever angle. Rotational movement of the twist grip will change engine power setting without affecting the rotor pitch angle.

The total movement of the collective stick is 30 degrees, while rotational movement of the twist grip is $\pm$ 120 degrees. Adjustable linkages are employed to vary total power lever angle travel, as required, from 30 degrees to 45 degrees in 5-degree increments. The operational stops noted above are duplicated on the collective stick position lever. Overtravel of the collective/twist-grip combination is absorbed by an overtravel spring incorporated in the collective stick assembly. This spring also acts as a decoupler to permit continued use of the flight control in the event of jamming of the power control linkage.

2.6.4 Ground Handling Controls

2.6.4.1 Brake System

The XV-9A uses all principal components of the CH-34A brake system. The wheel brake system consists of the toe brake control on each of the pilot's yaw control pedals, a brake cylinder attached to each of the pilot's pedals, a dual parking brake valve with a handle on the pilot's side of the cockpit, a wheel brake assembly at each main landing gear wheel, and interconnecting hydraulic tubing and hoses. Each main wheel can be braked separately. Depressing the toe brake control actuates the piston in the brake cylinder, to apply the wheel brake. Depressing both toe brake controls and pulling out the Parking Brake handle closes the dual parking brake valve and locks both wheel brakes. Depressing both brake pedals or the right pedal only will release the parking brakes. The parking brake valve incorporates a dual temperature compensator that provides for independent brake line operation. Access to the system is gained through the pilot seat hatch and through the cockpit floor hatch.

In order to meet the requirements of MIL-B-8584B (develop a coefficient of friction of 0.55 between tires and ground), the CH-34A brake hydraulic pressure must be increased by 34 percent to approximately 1,100 psi. This increase results in an actuator load of 290 pounds and a pedal load of 115 pounds. MIL-B-8584B specifies that a brake pedal load not exceeding 125 pounds shall develop a coefficient of friction of 0.31 between the tires and ground.

2.6.4.2 Tailwheel Lock

The CH-34A tailwheel assembly used on the XV-9A incorporates a shear pin to lock the full castering wheel in the centered position. This pin may be pulled by the pilot to unlock the tailwheel. Actuation is by a manual push-pull control (located on the right-hand side of the pilot's seat) that is connected to the locking pin by a cable running the length of the fuselage. A spring returns the pin to the locked position on release of the push-pull control.

2.7 AIRCRAFT EQUIPMENT

2.7.1 Hydraulic Systems

The two XV-9A hydraulic systems furnish power for operation of the helicopter flight controls and the hot gas diverter valves (see Figure 33). One system also furnishes fluid to the N_f signal feedback system (Section 2.6.3.1).

With the exception of the rotor control actuators and diverter valve actuators, all units used in the hydraulic systems are standard off-the-shelf items manufactured in accordance with the Military Specifications appropriate to the indicated service and function. The power control actuators are described below and in Section 2.6.1.2.

Two hydraulic systems are utilized, to provide at least one reliable source of power for the three tandem-cylinder hydraulic rotor control actuators and the diverter valve actuators. The two systems are entirely independent of each other. Both the primary system (1) and the utility system (2) are powered by engine-driven variable displacement pumps. System 1 is also supplied with oil by a rotor accessory gearbox driven pump whose prime function is to power the flight controls in the event of dual engine failure.

The hydraulic system flow capability has been based on the flow limitation imposed on the power control actuators by flight control dynamic considerations (Section 2.6.1.2). The total actuator flow is thus 2.45 x 3 = 7.35 gpm. The diverter valve actuators require

0.2 gpm each, or 0.4 gpm total. Thus the maximum flow requirement never exceeds 7.75 gpm. Each pump is capable of delivering 6.5 gpm at 4,400 rpm. The rotor pump, during autorotative descent, will deliver approximately 7.4 gpm. The maximum possible actuator requirements can thus be nearly met in the event of dual engine failure. Normal control requirements can easily be met during these conditions, so no loss of control capability is expected as a result of hydraulic flow limitations.

The heat generated in the hydraulic system as a result of hydraulic energy losses and hot bay heat transfer is transferred to the gas generator fuel by means of the oil-fuel heat exchanger. Each hydraulic system is cooled by one heat exchanger. The heat load for the "hottest" hydraulic system has been estimated to be:

Fixed load	7,000 btu/hour
Variable load	$85.2\ \frac{\text{btu}}{\text{gal.}}\ \left(\frac{\text{btu/min}}{\text{gal./min}}\right)$
Estimated maximum load	13,000 btu/hour

This load schedule, together with cooler performance data, is shown in Figure 34.

Test data indicate that the maximum temperatures to be expected on a 110°F day will be less than

Hydraulic oil system	175°F
Fuel at igniters	175°F
Hydraulic oil at diverter valve	200°F

2.7.1.1 Rotor Control Actuator Servo Valves

In normal operation, both System 1 and System 2 supply fluid at 3,500 psig to the flight control servos. The flight control servos incorporate sequence and relief valves so arranged that when the systems are operating a full pressure the low-pressure relief valves are operative and will allow only 1,600-psig pressure differential across the actuator pistons. When one of the system pressures drops below approximately 1,500 psig, the sequence valves shift so that the portion of the tandem servo connected to the remaining "good" system blocks off the low-pressure relief valves and connects the fluid to the high-pressure relief valves. These allow a pressure

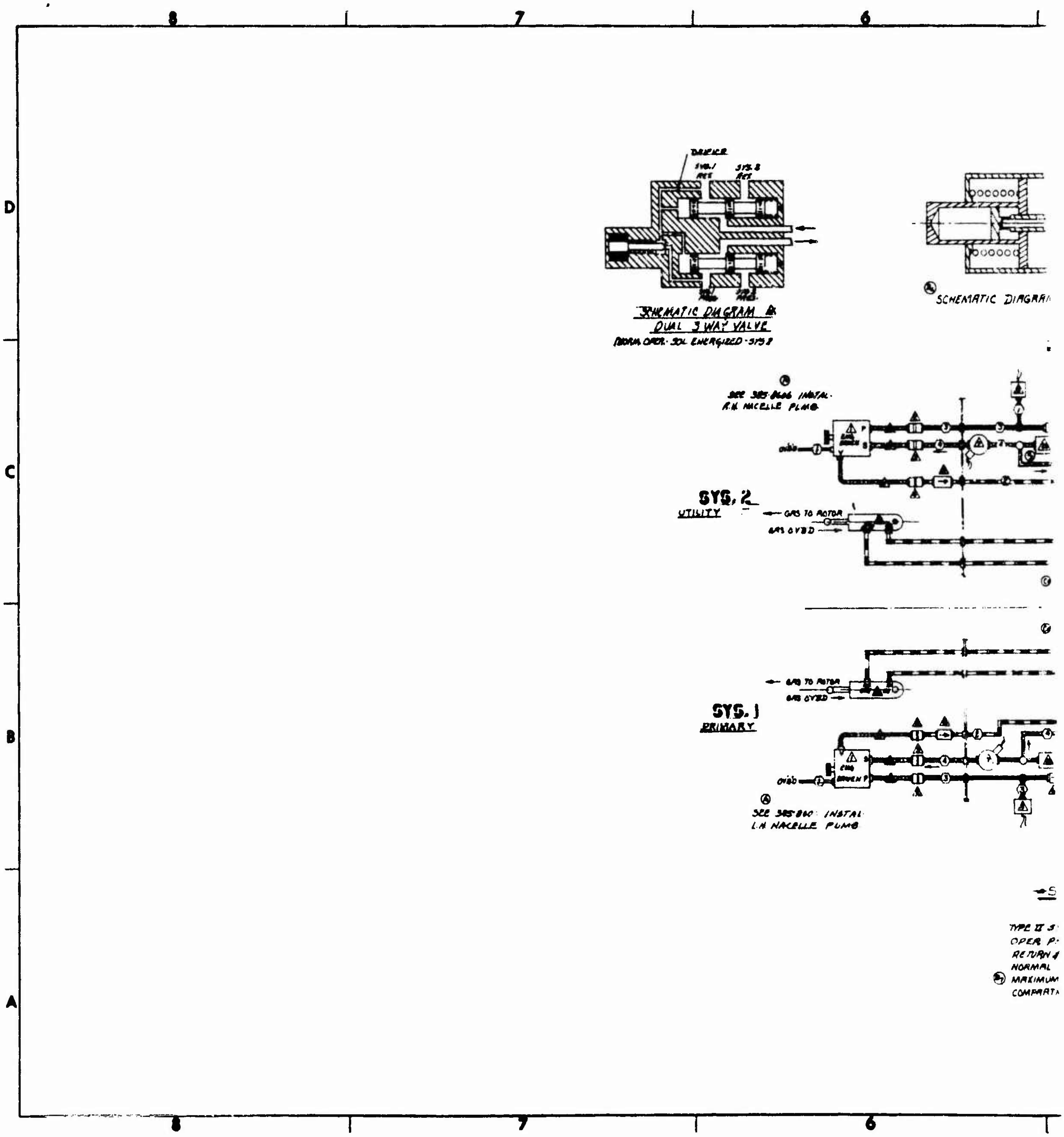
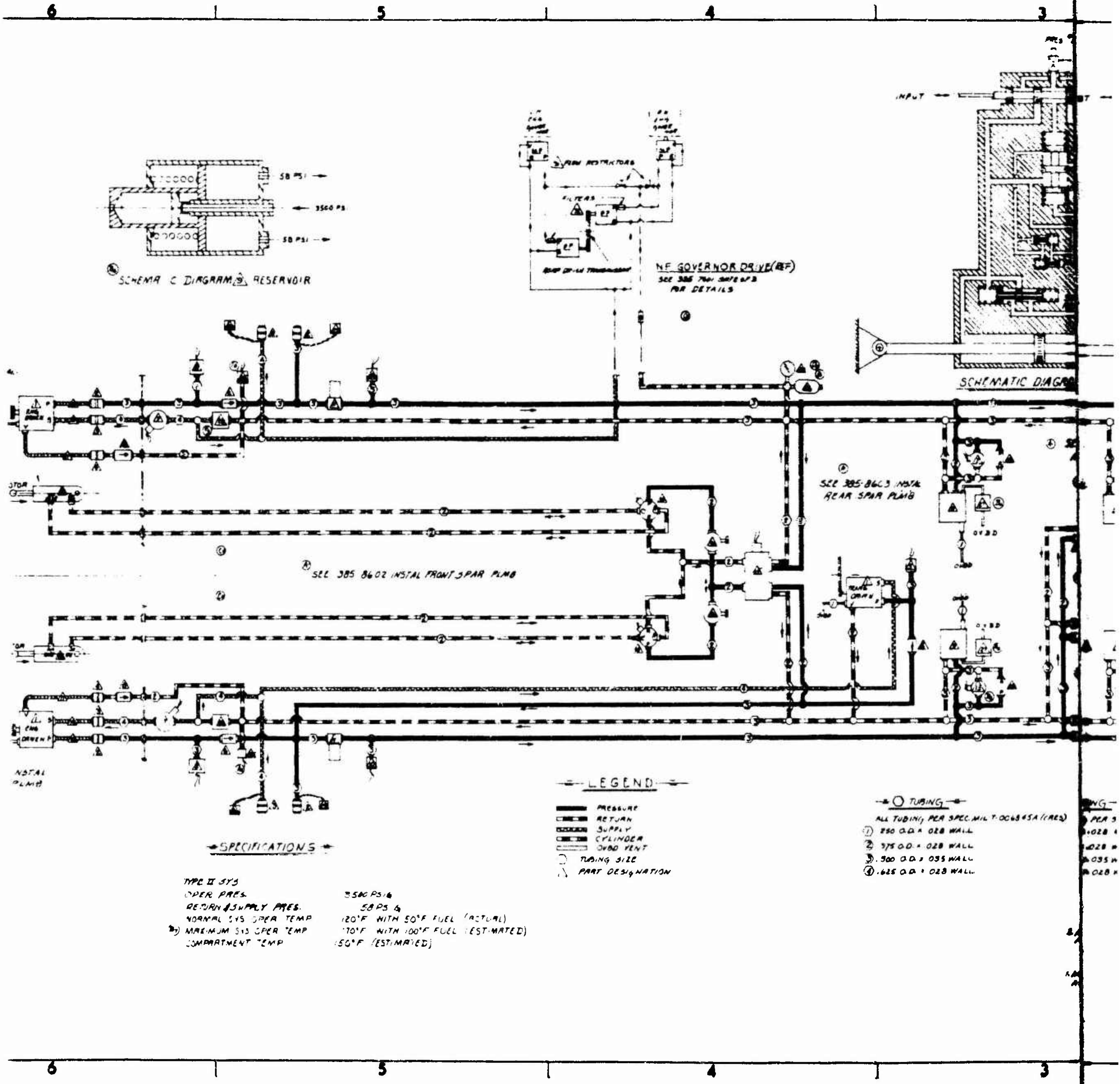
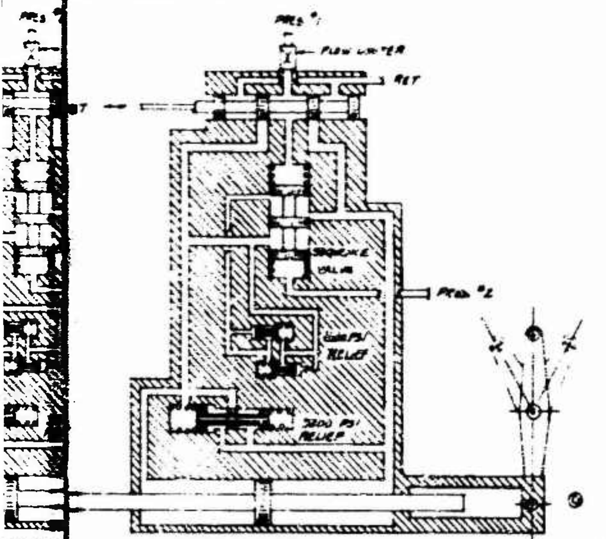


Figure 33. Hydraulic System Schematic

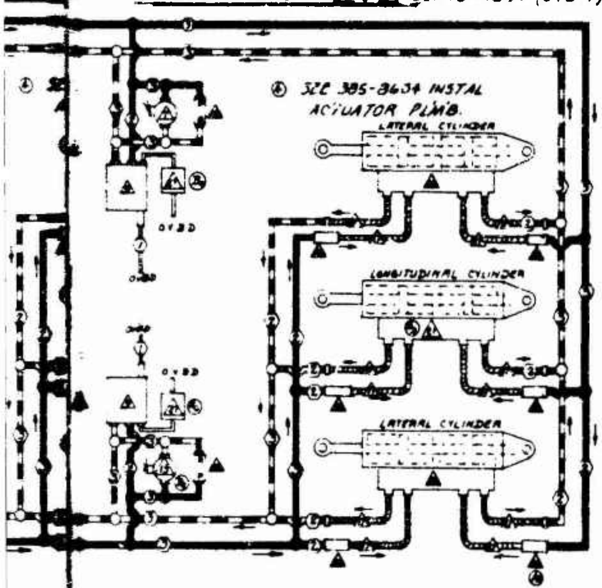
A


6
5
4
3
58 PSI
3500 PSI
58 PSI
SCHEMATIC DIAGRAM RESERVOIR
FLOW RESTRICTORS
FILTERS
NF GOVERNOR DRIVE (RF)
SEE 385 7601 SHTS 1 OF 3
FOR DETAILS
INPUT
SCHEMATIC DIAGR
SEE 385-8603 INSTAL
REAR SPAR PLMB
SEE 385 8602 INSTAL FRONT SPAR PLMB
INSTAL
PLMB
LEGEND
PRESSURE
RETURN
SUPPLY
CYLINDER
OVBD VENT
TUBING SIZE
PART DESIGNATION
SPECIFICATIONS
TYPE II SYS
OPER PRES 3500 PSIG
RETURN & SUPPLY PRES. 58 PSIG
NORMAL SYS OPER TEMP 120°F WITH 50°F FUEL (ACTUAL)
MAXIMUM SYS OPER TEMP 170°F WITH 100°F FUEL (ESTIMATED)
COMPARTMENT TEMP 150°F (ESTIMATED)
TUBING
ALL TUBING PER SPEC. MIL-T-006845A (CRES)
1 .250 O.D. x .028 WALL
2 .375 O.D. x .028 WALL
3 .500 O.D. x .035 WALL
4 .625 O.D. x .028 WALL

B



SCHEMATIC DIAGRAM SERVO ASSY (SYS 1)



PART DESIGNATION

- PUMP AMERICAN BRAKE SHOE # APZV-51 (DISP. .38 CIPR)(CUTOFF PRES. 3500 PSIG) (FLOW-ELEC DRIVEN - 7 GPM · TURBINE DRIVEN · 8.8 GPM)
- COUPLING - MS 24334-8
- COUPLING - MS 24334-10
- COUPLING - MS 24334-6
- PRES SWITCH - PREBANK # 8179 (ACT PRES 1800 PSI ON INCREASE - 1400 PSIG DECREASE)
- CHECK VALVE - CRISSAIR # 2C 5380
- SHUTOFF VALVE GEN CONTROLS # AV 18 17H (MOTOR OPER)
- CHECK VALVE - CRISSAIR # 2C 5340
- RESERVOIR - PNEUDRAULICS # 7070-160-6BR (VOL - 160 CU IN)(PRESSURE RATIO - 6:1)
- PRES TRANSMITTER PRINCETON MACHINE # 12N6
- FILTER - MS 28720-12
- RELIEF VALVE - MS 28893-08 (3750)
- SHUTOFF VALVE - GEN CONTROLS # AV 12 1H (MOTOR OPER)
- 4-WAY VALVE - WESTON # 14180-6X (SOL OPER. ENERGIZED POS. - GAS DRIVE)
- DUAL 3 WAY VALVE - VINSON # A-62366 (NORMAL POS. - SYS 2 - SOL. ENERGIZED)
- RESTRICTOR VALVE - REMCO # 107-6 (ADJUST FOR REQUIRED DIVERTER VALVE OPERATION)
- HOSE ASSY - MS 28762-6 / HOSE ASSY - MS 28762-8 / HOSE ASSY - MS 28762-10 — MAKE HOSES TO REQD LENGTH AT INSTALLATION
- LATERAL SERVO ASSY - CONVAIR # 9000-5.16 (EFFECTIVE AREA 1.72 SQ IN)(PISTON-STROKE - 5.16 IN)
- LONGITUDINAL SERVO ASSY - CONVAIR # 9000-6.50 (EFFECTIVE AREA 1.72 SQ IN)(PISTON-STROKE - 6.50 IN)
- DIVERTER VALVE CYL ASSY # 385-8607 (EFF AREA EXT .785 SQ IN, RET .590 SQ IN, STROKE 4.0 IN)
- DUST CAP ASSY - AEROQUIP CORP # 3207-8
- DUST CAP ASSY - AEROQUIP CORP # 3207-10
- TEMP BULB - MS 28036-3
- FLOW LIMITER CONVAIR # 3010 2455 (8.55 GPM)
- RELIEF VALVE - CIRCLE SEAL 5160T-12-TT-70
- COOLER ASSY - JANITROL # 16 D88 (OIL TO FUEL)
- ACCUMULATOR GREER # 30AP-15IN
- GAUGE - 100 PSI MARSHALL TOWN 3"
- FLOW RESTRICTOR - HOKE NEEDLE VALVE # 2RB 285
- FILTER - PERMANENT FILTER CO # 20231-40

} SEE 385-8604 SHEET 2 FOR DETAILS

385-8601

E

- PER SPEC MIL-T-6845A (PRES)
- .028 WALL
- .028 WALL
- .035 WALL
- .028 WALL

2. ALL TUBE ASSY'S SHALL BE PROOF TESTED BEFORE FINAL INSTAL AS FOLLOWS:

PRES LINES	7000 PSI
RETURN LINES	200 PSI
SUPPLY LINES	300 PSI

1. ALL HOSE ASSY'S SHALL BE PROOF TESTED BEFORE FINAL INSTAL AS FOLLOWS:

SIZE	PRES.
-6	7000 PSI
-8	7000 PSI
-10	8.. PSI

— NOTES —

SEE BODY OF DRAWING FOR LM

SCHEMATIC DIAGRAM - HYDRAULIC SYSTEM

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C

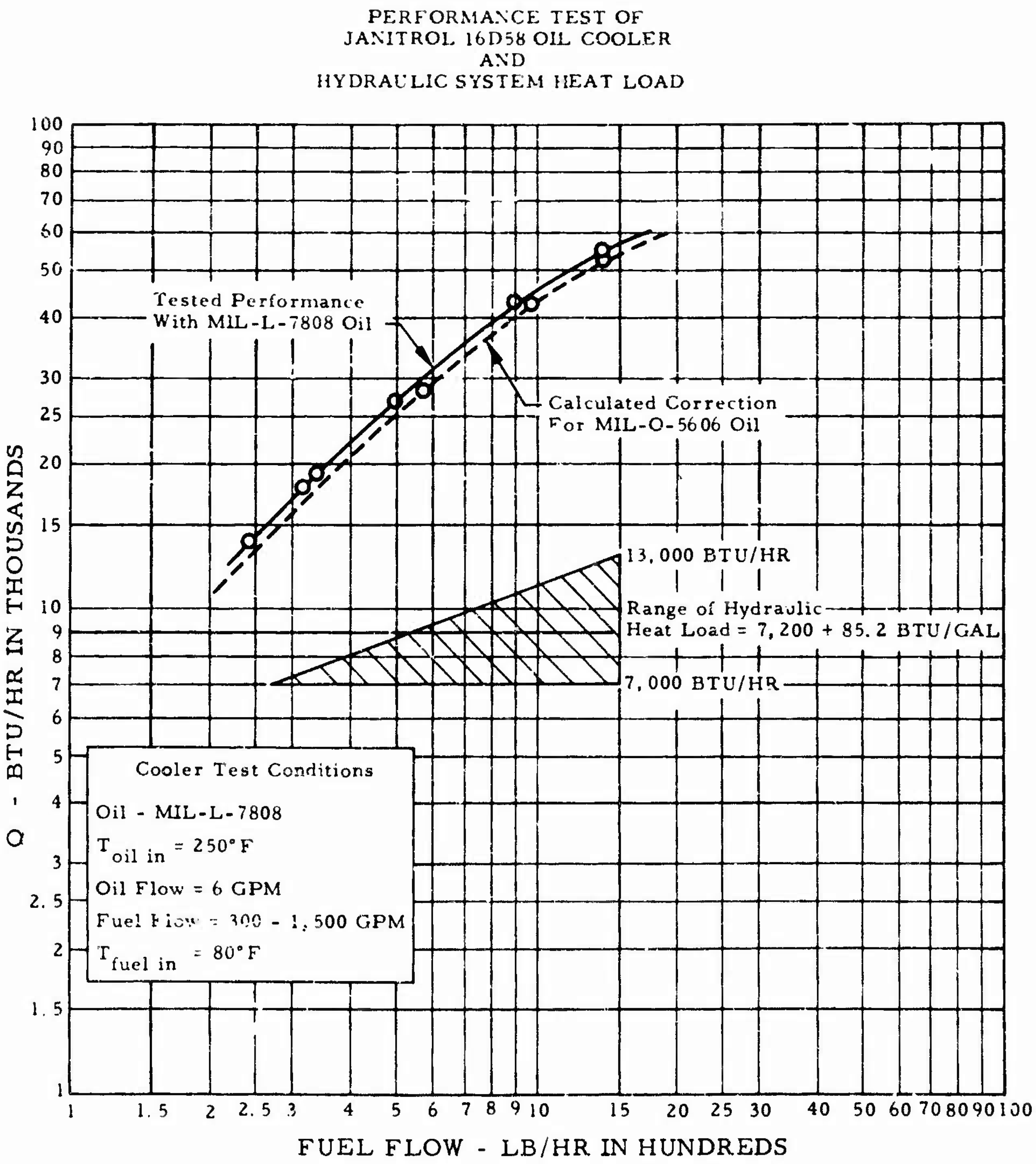


Figure 34. Hydraulic System Cooling

differential of 3, 200 psig across the actuator piston. At the same time, the sequence valve in the low-pressure portion of the tanden. servo shifts so that free flow is established from one side of the piston to the other.

Flow limiters, which are set to deliver 2. 45 gpm regardless of pressure, are installed in the pressure lines to the servo valves. Thus, all the servos are limited to the same stroke rate and force level, regardless of whether one or both hydraulic systems are operating. The shifting of the individual servos from low- to high-pressure output is accomplished automatically; thus, a loss of one of the two hydraulic systems during flight does not require the pilot to take emergency action in order to maintain hydraulic power.

2. 7. 1. 2 Diverter Valve Actuation System

As shown on the schematic diagram (Figure 33), the utility system (System 2) normally provides power for operation of the diverter valves. The diverter valve subsystem is made up of a dual three-way supply selector valve, two four-position selector valves, two rate limiting restrictor valves, and the diverter valve actuators.

The solenoid-operated dual three-way valve selects the hydraulic system that provides power for operation of the diverter valves. In normal operation, with both systems at full pressure, fluid from System 2 is directed to and from the subsystem.

In the event of loss of pressure in System 2, the unit may be switched by the pilot to supply fluid to the actuators from System 1. The pilot would switch the supply only in the event of an engine malfunction or for test or checkout purposes.

The diverter valve actuator is a Hughes Tool Company unit designed to replace the existing actuator. It is an unbalanced cylinder especially designed to operate in a high-temperature ambient atmosphere. It incorporates an orifice across the piston head that bleeds fluid at the rate of approximately 0. 2 gpm at 3, 000 psi.

This constant flow of fluid through the unit serves to absorb heat and carry it back to the oil-to-fuel heat exchanger. This fluid flowing from the diverter valve actuator may be considered to be the "hotspot" of the entire hydraulic system. Temperature recordings taken during whirl testing show that the maximum fluid temperature in this area will be less than 200°F on a 110°F day.

2.7.2 Electrical, Instruments, and Radio Installations

2.7.2.1 Supply System

The aircraft electrical system comprises two systems: the primary 28-volt d-c and the secondary 400-cps systems (see Figure 35).

The 28-volt d-c system is a regulated single-wire system with negative ground structural return and has three primary buses: (a) the main bus for power devices such as actuators, solenoids, motors, igniters, lights, and so on, (b) the bus for flight instruments, and (c) the bus for all aircraft warning lights.

All wiring is standard per MIL-N-5086 and MIL-N-5088. High-temperature wire is used in the hot bays. Where terminals are used, they are standard preinsulated, crimp type.

The 28-volt d-c system is supplied with power by two gas generator driven d-c generators and a 24-volt battery operating in parallel. The generators are rated at 150 amperes each over a speed range of 7,900 to 12,100 rpm. They are mounted by quick-attach-detach rings to the gas generator starter pads. The output of the generators is regulated by a transistorized voltage regulator. The regulation sensing circuit produces a transformed a-c signal that is compared with a diode reference voltage and thereby maintains generator voltage at the set level. Reverse current relays are provided to allow parallel operation of the generators.

The battery installation is composed of two 12-volt lead acid batteries of 24-ampere-hour capacity wired in series to obtain 24-volt dc.

External power is connected to the aircraft through a standard external power re eptacle mounted on the left side of the fuselage just forward of the entrance door. An external power relay is provided, wired into the interlocking portion of the external power receptacle so that if the external power is on the circuit is made and broken in the relay, not in the pins of the receptacle.

Cockpit controls are provided to adjust generator voltage, to open or close generator line and field, and to connect or disconnect either battery or external power.

The secondary electrical system is a 400-cps single-phase system with two subsystems, one at 115 volts and the other at 26 volts. They are used to power synchros, gyros, and portions of the flight test apparatus. Power for the 400-cps system is obtained through a transistorized inverter with capacity of 250 volt-amperes at 115 volts. A variable transformer is used to obtain an adjustable low voltage of approximately 26 volts. The a-c bus is energized anytime the d-c bus is energized, and may be disconnected only by circuit breaker.

2.7.2.2 Lighting Installation

For adequate flight safety, a lighting installation has been provided. Rotating and anticollision lights are installed on the aft surface of the vertical pylon and on the underside of the cockpit. Navigation and tail lights are also installed. A 450-watt land/hover light can be installed at an angle for best coverage. In the cockpit, a utility flood light is mounted on the upper canopy bow to serve as a cockpit light, portable trouble light, and instrument panel light.

2.7.2.3 Cockpit Installations

The arrangement of the cockpit control console, instrument panel, warning panel, and emergency panel is shown in Figure 36.

All controls have been grouped along the aircraft centerline for ready access by the pilot. The controls for the most important systems, those systems that must be manipulated during flight and especially during an emergency, have been placed in easy reach of both pilot and copilot, even though they may be restricted by their inertia reel harnesses.

The fuel system panel is directly aft of the power control quadrants. It is arranged in a simplified schematic representation of the dual fuel feed system. Rotary switches instantly show the actual fuel flow path from tanks, through crossfeed, to the gas generators. Alongside the fuel control panel are the hydraulic and gas system panels. There are five hydraulic control switches, for manual operation of the firewall valves, the bypass valves, and the diverter valve actuator supply selector valve. The gas system switches are for manual control of the diverter valves and blade-tip closure valves. These valves are

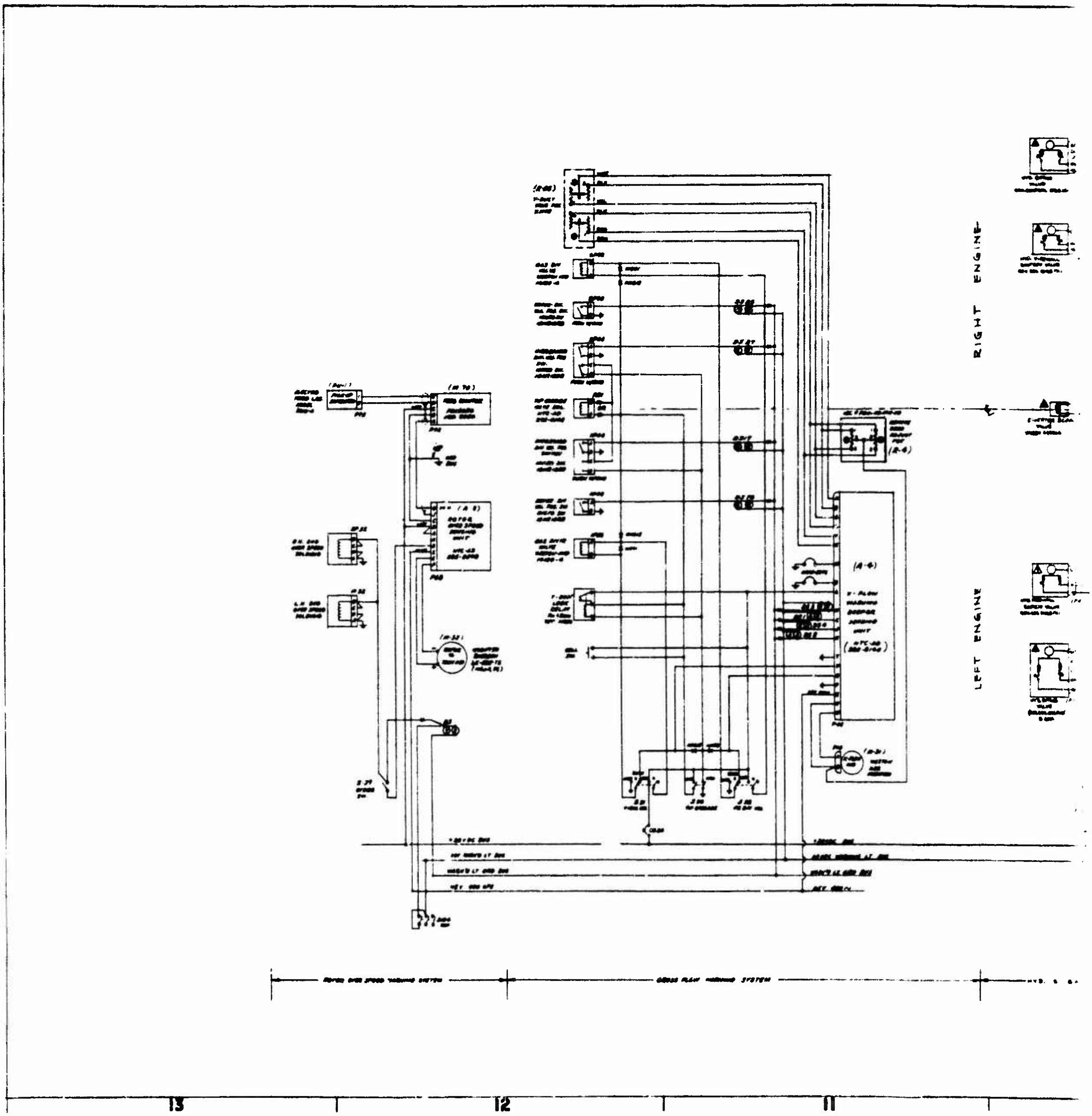
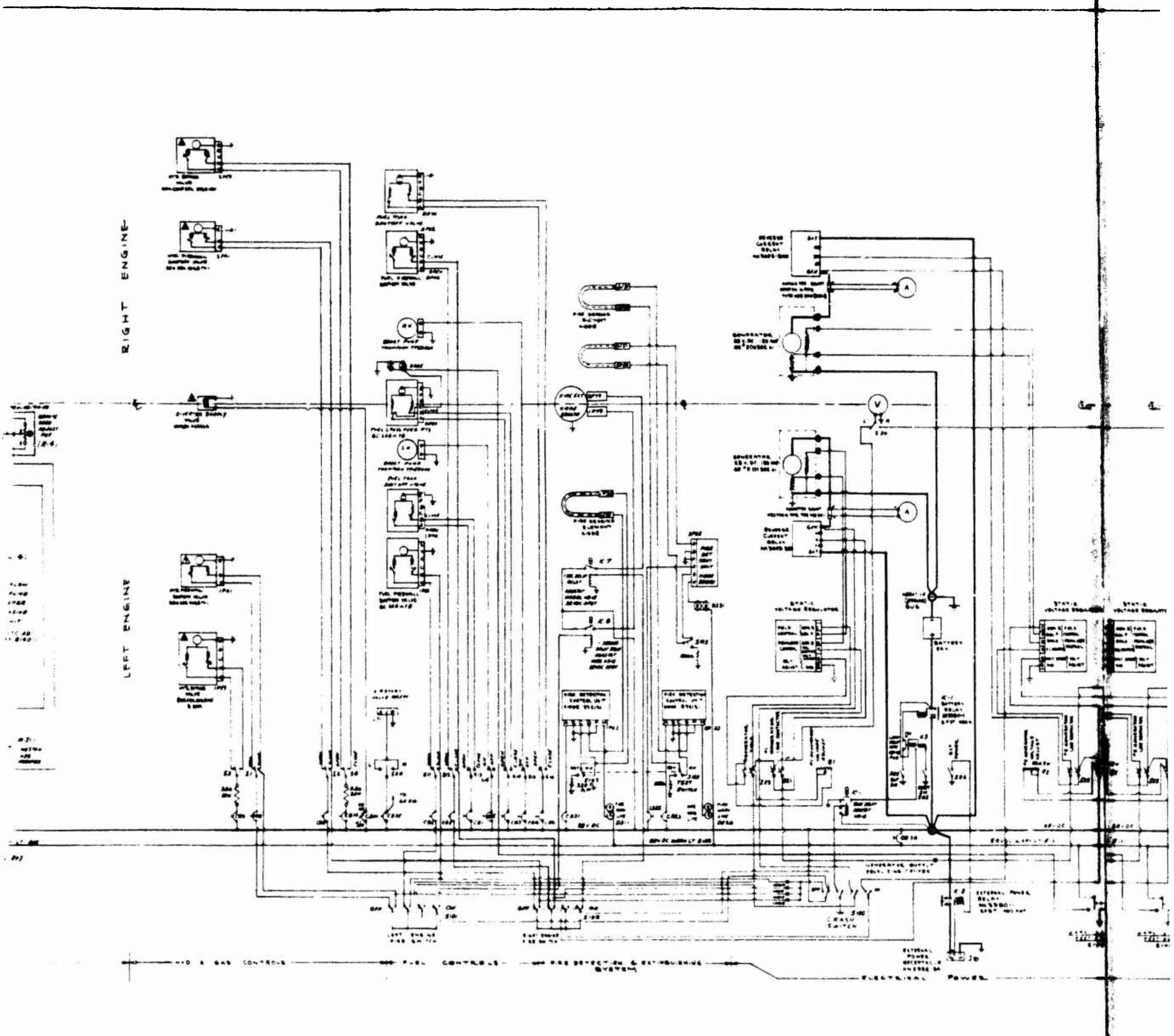
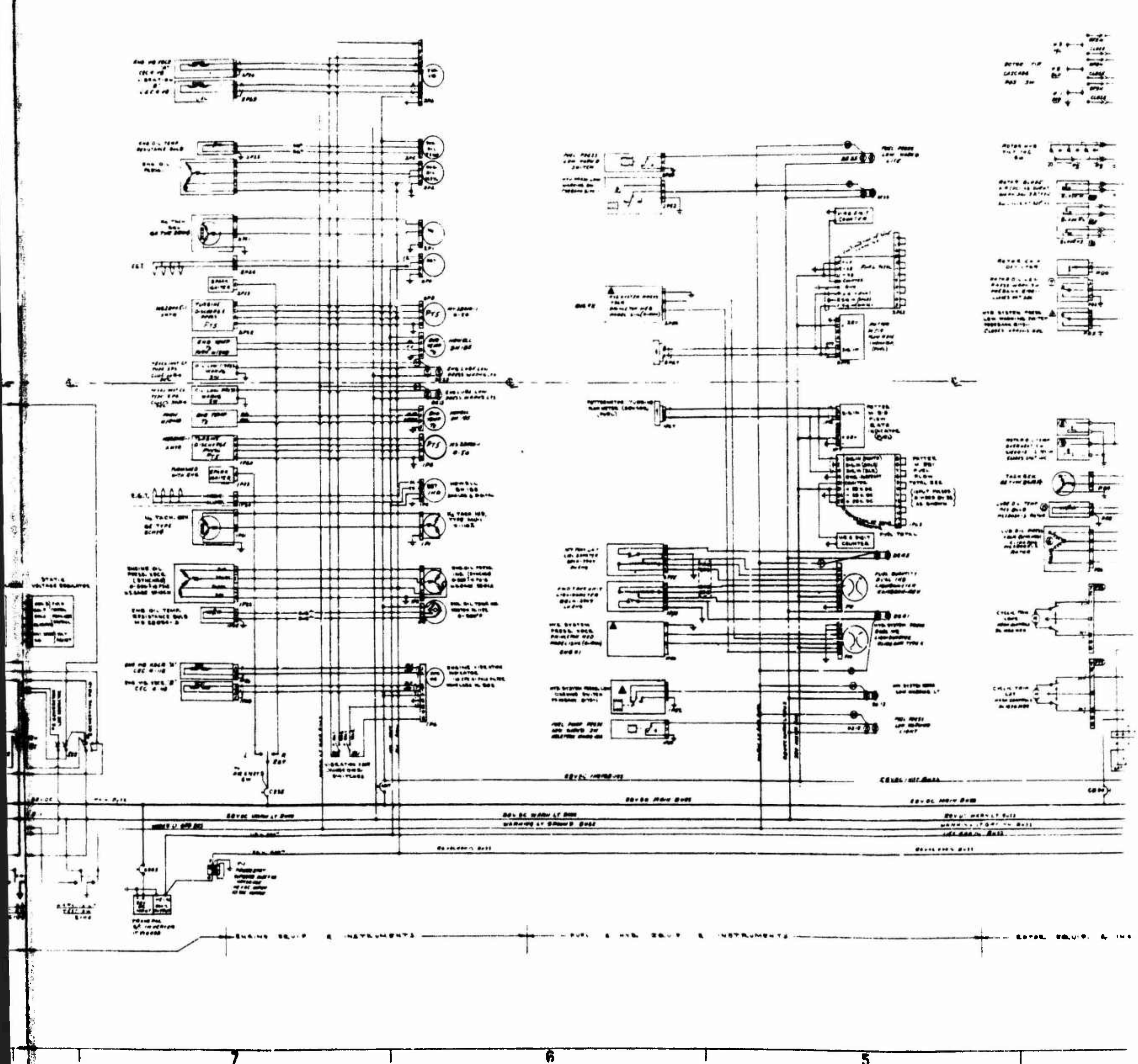


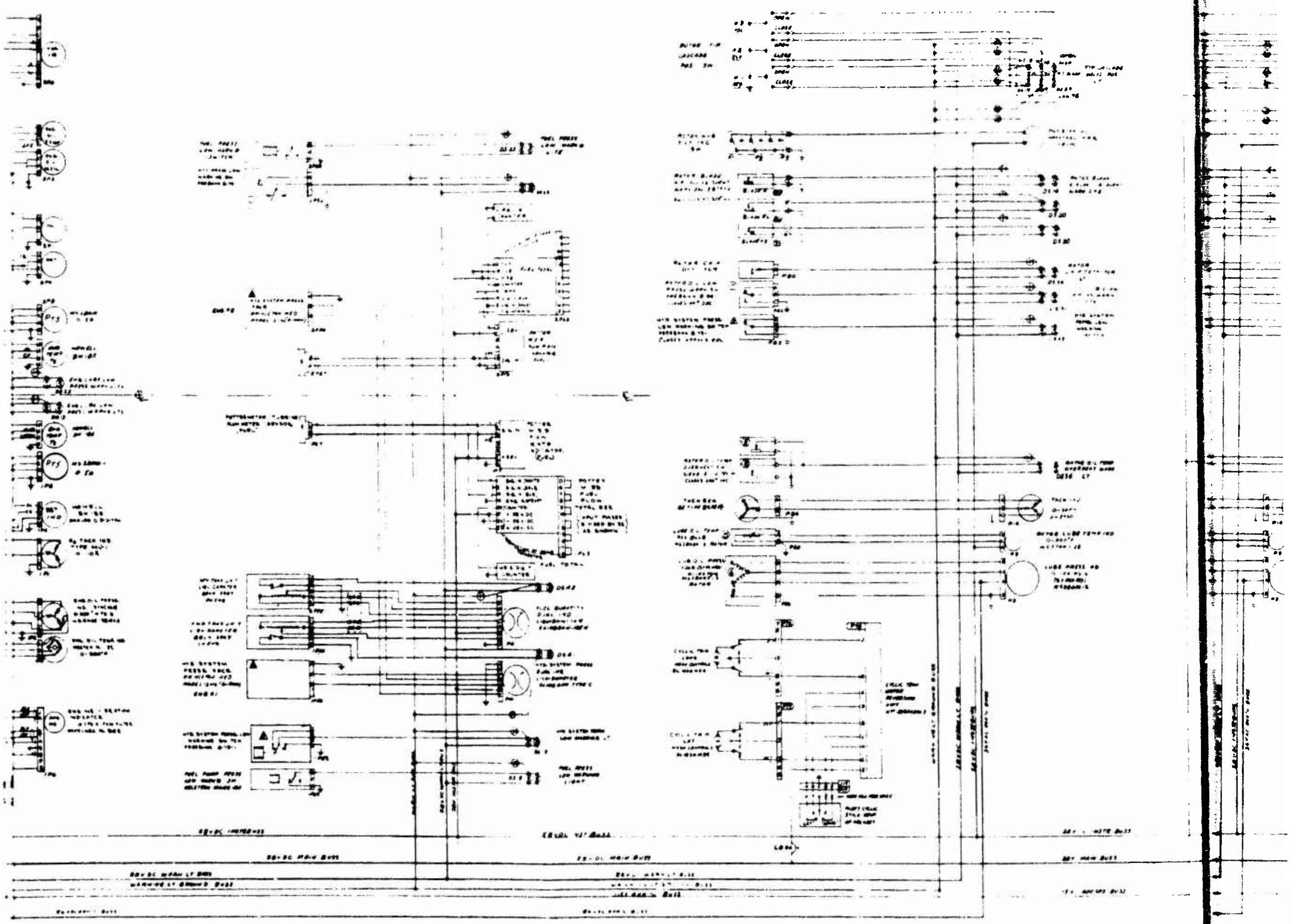
Figure 35. Electrical System Schematic


RIGHT ENGINE
LEFT ENGINE
STATIC VOLTAGE REGULATOR
CRASH SWITCH
FUEL CONTROLS
FIRE DETECTION & EXTINGUISHING SYSTEM
ELECTRICAL POWER
10
9

B


ENGINE EQUIP. & INSTRUMENTS
FUEL & HYD. EQUIP. & INSTRUMENTS
ROTOR EQUIP. & INS
7
6
5

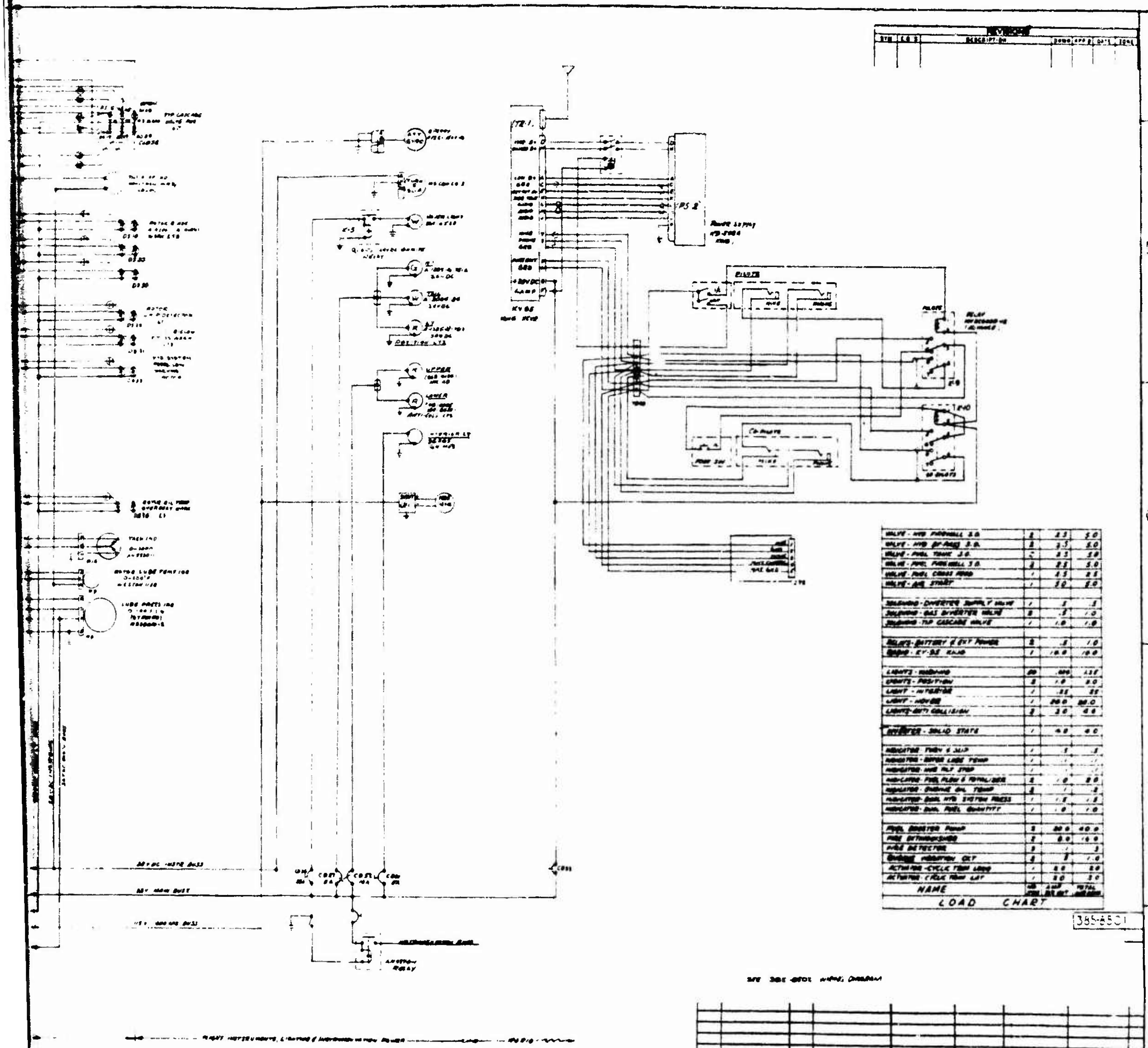
C



FUEL & HYD. EQUIP. & INSTRUMENTS — ROTOR, EQUIP. & INSTRUMENTS — RIGHT INSTRUMENT

6 5 4

D



NAME	NO. REQD	AMP PER UNIT	TOTAL AMPERES
VALVE - HYD FIREWALL S.O.	2	2.5	5.0
VALVE - HYD BY-PASS S.O.	2	2.5	5.0
VALVE - FUEL TANK S.O.	2	2.5	5.0
VALVE - FUEL FIREWALL S.O.	2	2.5	5.0
VALVE - FUEL CROSS FEED	1	2.5	2.5
VALVE - AIR START	1	5.0	5.0
SOLENOID - DIVERTER SUPPLY VALVE	1	.5	.5
SOLENOID - GAS DIVERTER VALVE	2	.5	1.0
SOLENOID - TIP CASCADE VALVE	1	1.0	1.0
RELAYS - BATTERY & EXT POWER	2	.5	1.0
RADIO - KY-95 [illegible]	1	10.0	10.0
LIGHTS - [illegible]	20	[illegible]	[illegible]
LIGHTS - POSITION	2	1.0	2.0
LIGHT - INTERIOR	1	.25	.25
LIGHT - HOVER	1	20.0	20.0
LIGHTS - ANTI COLLISION	2	2.0	4.0
INVERTER - SOLID STATE	1	4.0	4.0
INDICATOR - TURN & SLIP	1	.5	.5
INDICATOR - ROTOR LUBE TEMP	1	.1	.1
INDICATOR - HUB TILT STOP	1	.1	.1
INDICATOR - FUEL FLOW & TOTALIZER	2	1.0	2.0
INDICATOR - ENGINE OIL TEMP	2	.1	.2
INDICATOR - DUAL HYD SYSTEM PRESS	1	.5	.5
INDICATOR - DUAL FUEL QUANTITY	1	1.0	1.0
FUEL BOOSTER PUMP	2	20.0	40.0
FIRE EXTINGUISHER	2	8.0	16.0
FIRE DETECTOR	3	.1	.3
[illegible] IGNITION CKT	2	.5	1.0
ACTUATOR - CYCLIC TRIM LONG	1	2.0	2.0
ACTUATOR - CYCLIC TRIM LAT	1	2.0	2.0

LOAD CHART

385-6501

SCHEMATIC WIRING – DIAG

HUGHES TOOL COMPANY

SCALE NONE

E

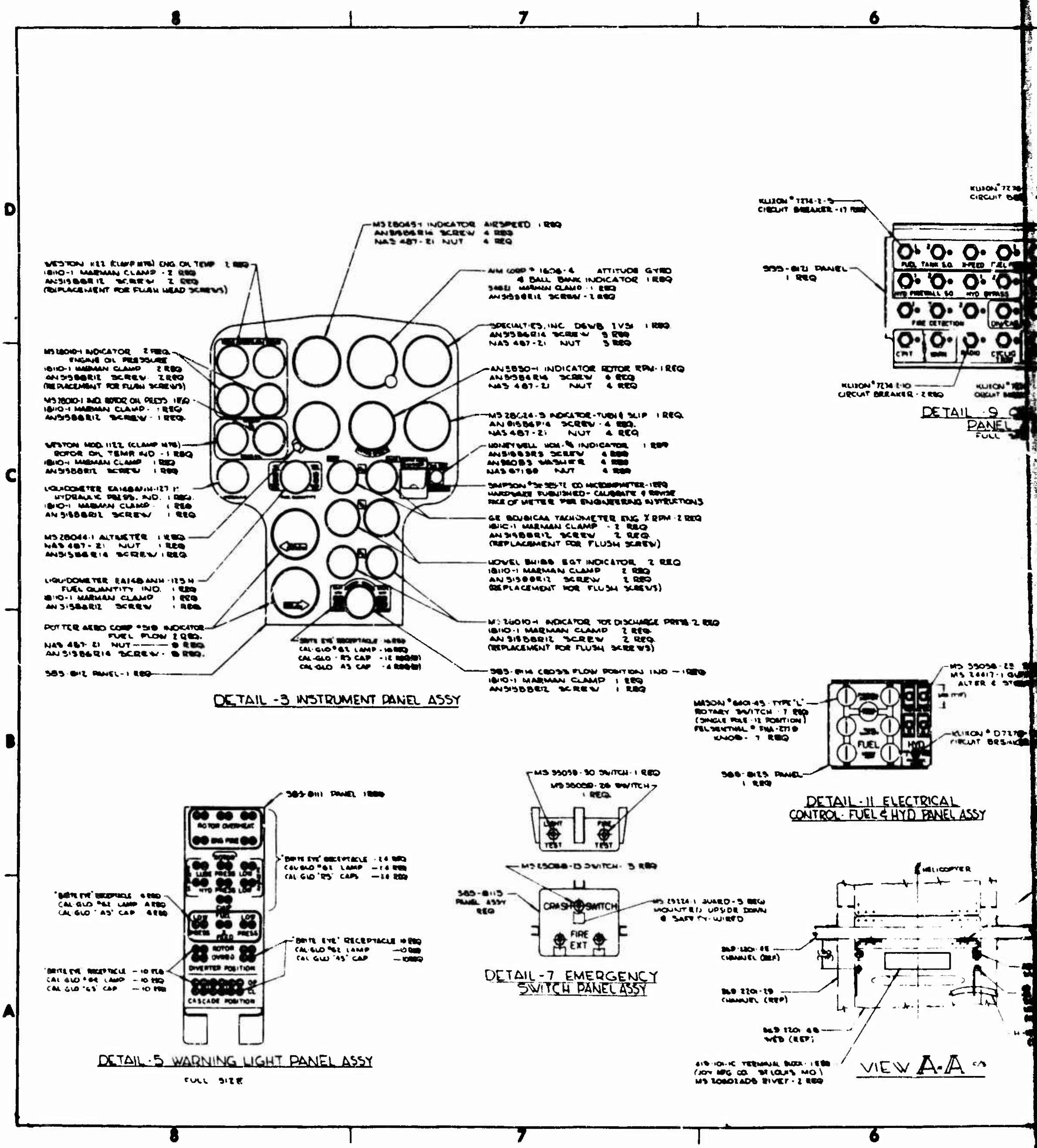
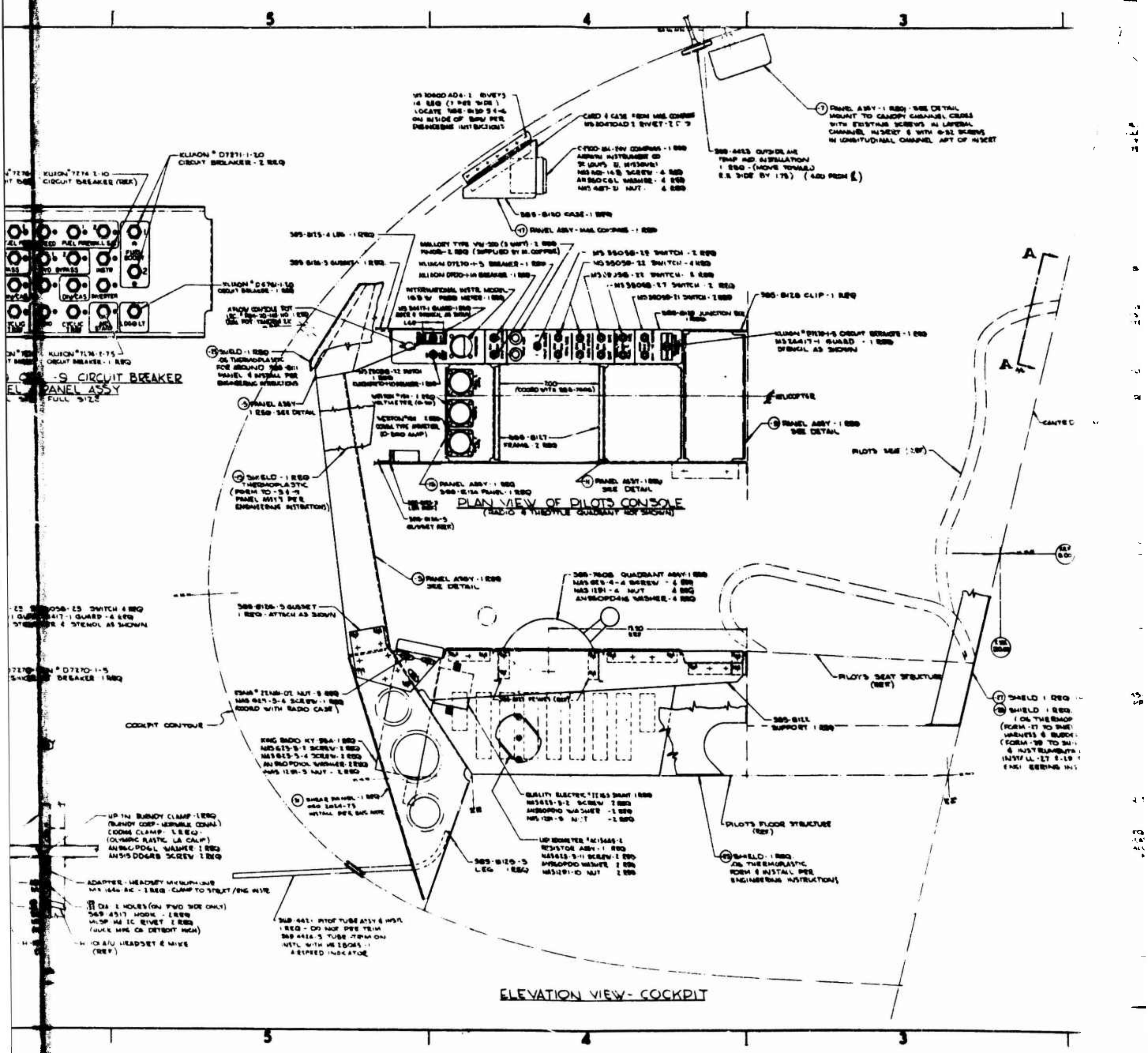
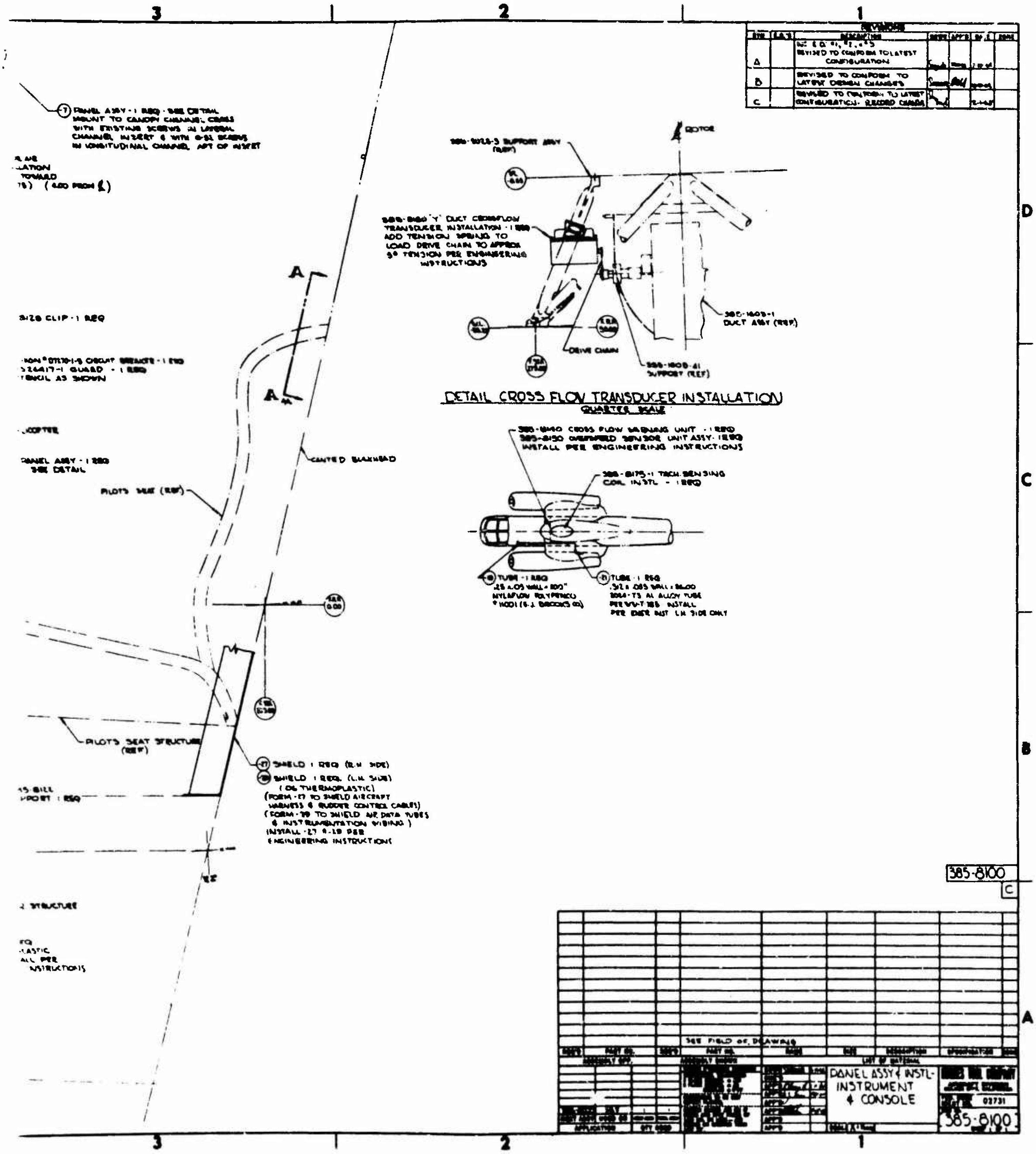


Figure 36. Cockpit Instrument Panel and Console

A


5
4
3
CIRCUIT BREAKER
PANEL ASSY
FULL SIZE
PLAN VIEW OF PILOTS CONSOLE
(RADIO & THROTTLE QUADRANT NOT SHOWN)
PILOTS SEAT STRUCTURE (REF)
PILOTS FLOOR STRUCTURE (REF)
COCKPIT CONTOUR
ELEVATION VIEW - COCKPIT
5
4
3

B



DETAIL CROSS FLOW TRANSDUCER INSTALLATION
QUARTER SCALE
CANTED BULKHEAD
PILOTS SEAT (REF)
PILOTS SEAT STRUCTURE (REF)
DRIVE CHAIN
ROTOR
385-1605-1 DUCT ASSY (REF)
SEE FIELD OF DRAWING
LIST OF MATERIAL
PANEL ASSY & INSTL- INSTRUMENT & CONSOLE
385-8100
385-8100
C

C

normally controlled semiautomatically by sensing engine power differences (Section 2. 5. 6. 2), but manual control is required for preflight and maintenance purposes.

Wherever possible, circuit breaker switches have been incorporated. Where the switching function will not permit this, separate circuit breakers are provided on a recessed panel directly under the pilot's collective lever.

Alongside the power control quadrant are the gas generator start switches. The gas generator air-start switch is wired to connectors in the external power connector panel for control of the MA-1 ground air start cart. Forward of these switches are the electrical system controls and instruments and the aircraft light switches. Adjustment is provided to match the voltage output of the two generators.

A VHF radio is installed for use during the local flight testing. It is controlled by a two-position press-to-talk switch incorporated into the cyclic grip. The first position is for intercom. This position energizes a radio relay to provide side tone intercom, but leaves open the transmit circuits. The second position is for transmit. The antenna for the radio is installed on the V-tail. At this frequency, no matching equipment is required. The headset and mike is a standard carbon mike-headphone set. An alternate headset-mike system is a part of the pilot's flight helmet. The aircraft intercom system has connectors at the external power connector panel to permit communication with ground crew during engine checkout and runup.

ARC-45(UHF) and ARC-73(VHF) radio installations are planned when the flight test requirements demand greater flexibility and greater range. The control boxes for these radios will be installed on the control console. The radios will be installed in the cargo compartment. Antennas will be installed as required.

The flight instrument installation is nonshock mounted and uses standard MIL types, FAA types, and standard aircraft instruments. It has been arranged to conform as nearly as possible to accepted practices and yet be as small as possible to reduce interference with the pilot's vision. As can be seen in Figure 36, the flight attitude group is quite conventional, except that the gas generator and rotor tachometers are separate instruments. This is preferred since the rotor and the gas generators are gas coupled rather than shaft coupled; thus, there is no fixed relationship between the speed of either gas generator and that of the rotor. The standard gas generator instruments, gas generator rpm, exhaust gas temperature, and discharge pressure are displayed under the flight attitude instruments. Additional propulsion and accessory instruments are arranged to the left, in a subsidiary position.

There are two instruments peculiar to the Hot Cycle system. The first is the rotor tilt stop indicator, which is located beneath the flight attitude group. This indicator tells the pilot when the hub-tilt limiters are disengaged and full rotor control motion can be applied. The second unique instrument is the crossflow indicator that indicates the relative gas flow from the two gas generators. The operation of this system has been more completely described in Section 2. 5. 6. 2.

On the instrument panel are several warning lights. These are the rotor lube overheat light, the fuel low-level lights, the rotor overspeed switch warning lights, and the crossflow warning lights (see Section 2. 5. 6. 2). In addition, on the aircraft centerline and atop the instrument panel is the aircraft warning light panel. This panel is as directly in line with the pilot's vision as possible, without interfering. High intensity dual bulb lights and glare shields assure that warning indications can be seen during all exterior light conditions.

The lights are arranged so that the most serious malfunction -- rotor spar overheat -- is at the top. The necessary system position indicating lights are at the bottom. The lights are also arrayed in columns -- left-hand gas generator, rotor, and right-hand gas generator -- to distinguish the source of trouble.

Associated with the warning panel is the emergency panel. It is located on the center cockpit canopy bow above the pilot. Its position is dictated by the reach of the pilot when restricted by his inertia reel. On this panel are fire extinguishing switches to control fire in either the left-hand or right-hand gas generator bays. These switches close the hydraulic and fuel firewall valves, disable the generator, open the line contactor, and discharge the fire extinguishing agent after a one-second delay. The crash switch does everything both the fire switches do except discharge the fire extinguishing agent. In addition, the crash switch disconnects the battery after a two-second delay. On the bottom of the emergency panel are located the fire detection test switch and the warning light test switch.

Also mounted on the cockpit canopy are the magnetic compass and the outside temperature indicator.

2.7.3 Aircraft Furnishings

The aircraft furnishings are all installed in the cockpit, and are items for the safety and convenience of the crew. These items are:

a. Seat cushion assemblies

b. Safety harness and inertia reels

c. Portable fire extinguisher

d. Map case

The seat cushions, safety harness, inertia reels, and map case were designed and tested for the OH-6A, and are used in the XV-9A with no modifications. The portable fire extinguisher is a 2.5-pound dry chemical type extinguisher.

2.8 AIRCRAFT SAFETY

Aircraft safety is promoted not only by complete structural integrity but by fail-safe operating systems, by a complete fire prevention program, by fire detection and extinguishing, and by provisions for crew protection. Each of these areas will be discussed separately in this section.

2.8.1 Failure of Operating Systems

2.8.1.1 Hydraulic Systems

Hydraulic power is obtained from two completely separate hydraulic systems. Power is supplied by a pump on each gas generator and by a rotor-driven pump. Both systems operate continuously at 3,500 psi, and all power actuators and their controls are designed to work with either or both hydraulic systems in operation. In the event of dual gas generator failure, the rotor-driven pump will adequately supply the system.

2.8.1.2 Electrical System

Electrical power is obtained from three sources: a generator on each gas generator and the 24-volt battery. If both generators fail, sufficient energy is available from the battery for approximately 20 minutes of normal aircraft operation (including test instrumentation). Most valves are motor operated, so they will stay in their normal position if the electrical failure is at the valve. Solenoid operated valves are spring loaded in the normal position when deenergized. With a complete electrical failure of all components, the aircraft can still be flown and landed in a normal manner, although instruments, radio, valves, and so on, are inoperative and not usable to cope with further emergencies. The gas generator fuel pump will sustain normal fuel flow under most conditions, should the electric fuel boost pumps become inoperative.

2.8.1.3 Gas System

The semiautomatic gas system controls protect the aircraft by instantly indicating a defective gas generator. This system allows the pilot to isolate a defective engine and close the tip closure valves without the hazard of erroneous interpretation of instrument data. Failure of the semiautomatic system can be corrected by manual pilot override.

The blade-tip closure valves are gas loaded and centrifugally loaded to the "Open" position, which is normally desired. Therefore, if pneumatic or electrical failure occurs, full rotor power can be obtained. The tip closure valves cannot be closed unless one gas generator is diverted overboard.

If a structural failure of the gas system occurs, the diverter valves are put into the overboard position to protect the damaged component from further and possibly catastrophic damage. If this occurs, the helicopter can be flown as an autogyro. The overboard jets provide sufficient thrust to maintain altitude at approximately 13,500 pounds, and would extend the aircraft glide at greater gross weights.

2.8.1.4 Flight Controls

The flight controls are a simple mechanical system with adequate safety margins, but could be disabled by complete hydraulic failure. Dual hydraulic supply systems, dual servo valves, and tandem actuators ensure against such disabling.

The pilot's collective lever is connected to the power control system through an override assembly, so that power control failures do not disable the collective pitch system.

The yaw control system has a yaw jet and rudders. If engine power failed completely, the helicopter could be autorotated with rudder control above forward speeds of approximately 30 to 40 knots.

2.8.2 Fire Prevention

The most important phase of fire protection is the minimization of the fire hazard to prevent the occurrence of fire. Recognizing this fact, the following commonsense safety precautions have been incorporated into the design and fabrication of the powerplant installation:

a. Combustibles have been isolated from sources of ignition. Fuel and hydraulic oil lines are isolated from the gas generators by compartmentation. Temperatures of these fluids are kept to a minimum by gold plating the compartment skins and by insulation blankets.

   All combustible-carrying-line runs in the gas generator compartments are kept as short as possible. If rigid, lines are stainless steel with generous bend radii. If flexible, lines are fire resistant and have ample slack for relative movements. All lines are adequately supported by cushion-type clamps.

b. Electrical wiring has been isolated from the hot portions of the gas generators, having been run through the above compartments.

c. Drain holes have been provided in the gas generator cowling, to assure drainage of combustibles from the aircraft for all attitudes of flight and ground operation. All drain lines are clear of the aircraft so as to prevent impingement and possible reentry.

d. High-temperature gas-carrying ducts have been insulated to reduce the heat flux, with resultant lowering of temperature. Heat shields installed over the ducts provide local protection in areas of possible hydraulic fluid leakage by reducing temperatures below the autogenous ignition temperature of the fluid.

e. Ventilation holes and louvers have been provided throughout to induce air circulation.

f. Drain lines have been extended below the exterior surfaces to negate impingement, and sealed to prevent entry back into the vehicle. Fuel vent lines are designed to prevent spillage, and are located in areas isolated from any exhaust or hot gas generator surface and away from electrical ignition sources.

g. All fuel system equipment, such as lines, valves, and so on, have been located remote from ignition sources or isolated by compartmentation. All electrical equipment in the fuel cells has been grounded.

h. Overheat and fire warning systems have been installed on each engine. An overheat system has been installed in the Y-duct bay.

2.8.3 Fire Containment

Fire is kept from spreading by the following features of the design:

a. Fuel and hydraulic oil supply lines passing through the firewall bulkheads incorporate shutoff valves located outside of fire areas.

b. There are multiple fuel and hydraulic shutoff devices.

c. Gas generator bays are ventilated and are isolated from the rest of the airplane so that fire cannot travel inward through the lateral pylon from the nacelles.

d. No absorbent materials or materials that could constitute a source of reignition are used.

2.8.4 Fire Detection System

The fire detection system installed on the gas generators is shown diagrammatically in Figure 37. The system will:

a. Indicate a heat hazard, either a fire or an abnormal temperature condition

b. Remain on for the duration of the overheat condition

c. Indicate when the overheat condition is resolved

d. Indicate reoccurrence of fire or abnormal temperature

The system consists of continuous-resetting temperature-sensing elements mounted on the gas generators, a fuselage-mounted control unit to monitor the sensing element, and a warning circuit to inform the pilot that a heat hazard is detected.

The sensing element assemblies mount on each gas generator in one continuous loop, covering all potential fire and high-temperature areas. The detector alarm will trigger when the temperatures are 150 degrees F above the maximum operating temperatures.

The overheat warning system in the Y-duct area is also noted in Figure 37. This system will indicate an abnormal temperature condition.

The system, like the fire detection system, consists of a temperature-sensing element loop, a control unit, and a warning circuit. The detector alarm will trigger when the temperatures are 100 degrees F above the maximum allowable ambient.

2.8.5 Fire Extinguishing System

The fire extinguishing system is a one-shot system consisting of a fire extinguishing agent container, distribution systems to each nacelle, and an electrically operated discharge valve for each distribution system. The selected agent, bromotrifluoromethane ($CBrF_3$), offers unusual advantages, particularly against Class B (flammable liquid) and Class C (electrical) fires. From a fire extinguishment standpoint, $CBrF_3$ is the most effective agent, is noncorrosive to aluminum, steel, or brass, and has a low toxicity in the natural condition as well as in the pyrolized state.

Eleven pounds of agent are carried in a single container located on the centerline of the airplane. The container has dual outlets, and makes use of a cartridge and a disc-type valve in each outlet. Switches in the pilot's compartment select one gas generator or the other as required to receive the entire charge. When such a switch is actuated, the cartridge in the applicable valve fires a slug that ruptures a frangible disc. The pressurized liquid agent is forced out of the container, and is directed to a stainless steel engine manifold through runs of 1.00-inch-diameter x 0.049-inch-wall aluminum alloy tubing. A flex line, to compensate for relative movement between the gas generator and the structure, connects the two line assemblies. The agent is discharged forward through a series of 0.055-inch-diameter holes in the 0.750-inch-diameter x 0.028-inch-wall gas generator manifold. Coverage is 360 degrees around the gas generator and forward to the gas generator inlet assembly. Total time for effective agent discharge, including a one-second time delay for closing fuel and hydraulic firewall valves, is calculated to be 3.8 seconds. The system is shown diagrammatically in Figure 38.

When the fire switch is pulled, the fuel and hydraulic firewall shutoff valves are automatically closed prior to the discharge of the agent to the selected nacelle. The generator is also disabled, to avoid the reignition by a defective or fire-damaged electrical system. The crash switch duplicates the action of both fire switches used together, except for discharging the fire extinguishing agent, and in addition disconnects the battery after a delay of two seconds.

MIL-E-5352 was used to determine the required quantity of agent. Air Force Technical Report 6430 was the basis for analyzing discharge times.

2.8.6 Cockpit Safety

Several items of furnishings (Section 2.7.3) contribute to cockpit safety in the event of crash. The crew is restrained by lap belt and by double shoulder inertia reel harness with standard quick-release feature. A 2-1/2-pound dry chemical fire extinguisher is available for small cockpit fires. A data case is installed for safe stowage of miscellaneous small items required by the crew. The pilot door has emergency release devices and transparent panels that can be knocked out if the door is jammed.

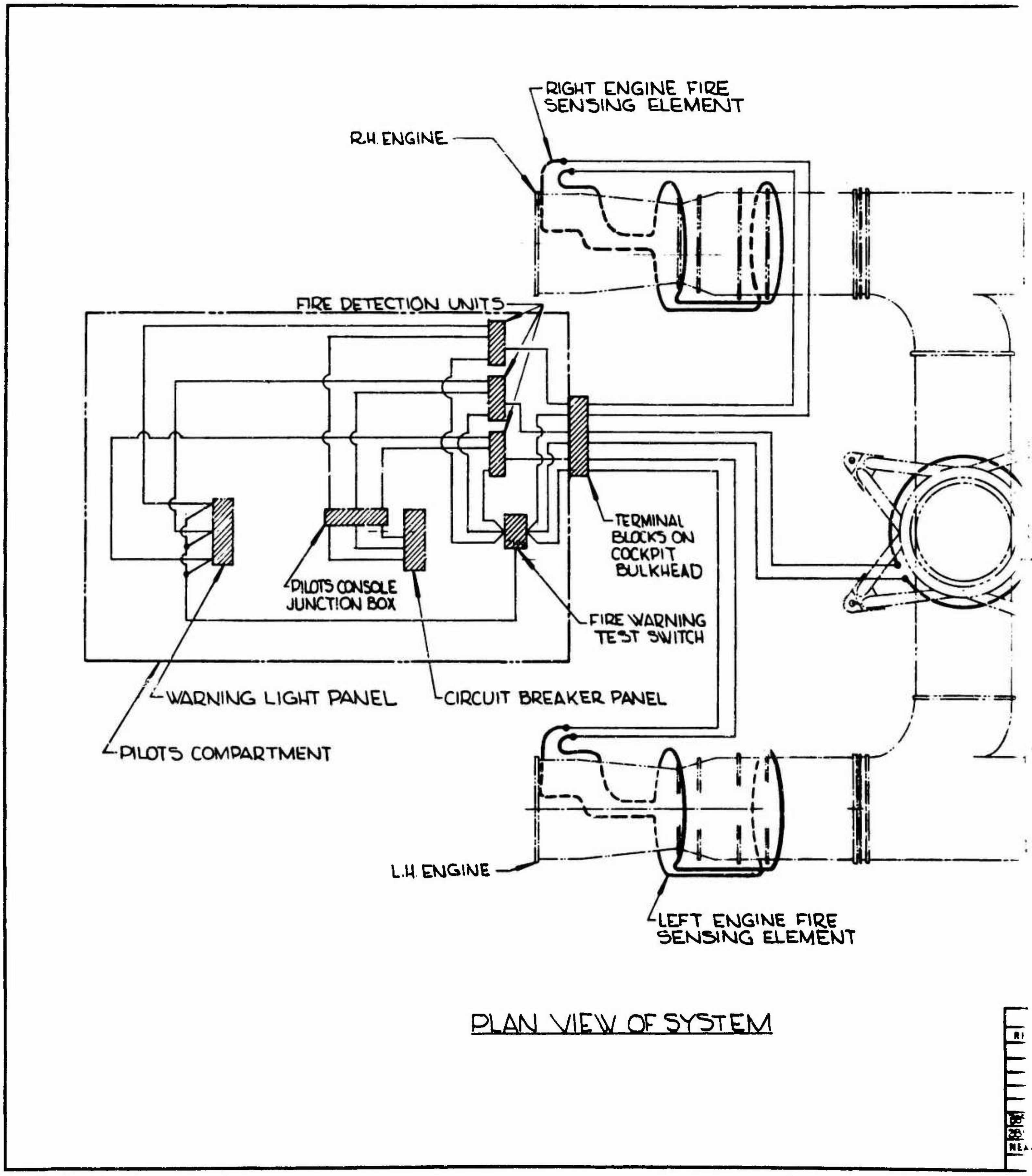
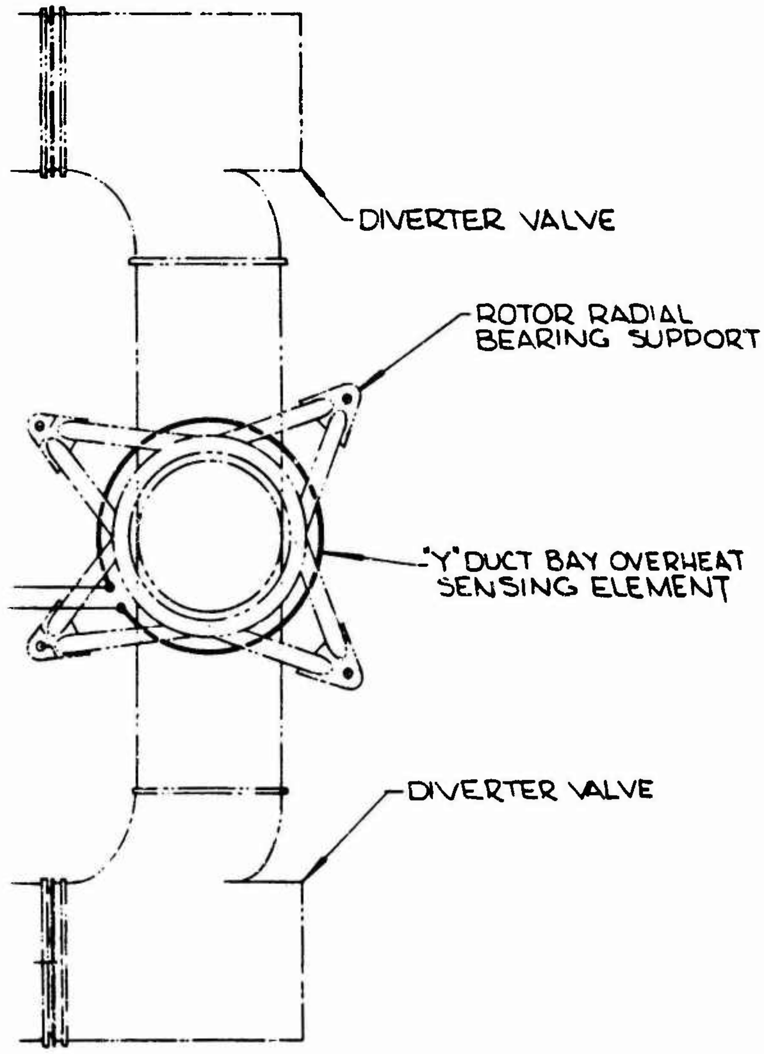


Figure 37. Fire Detection System

SYM	E.O.'S	DESCRIPTION	DRWN	APP'D	DATE

REVISIONS



385-7723

REQD	PART NO.	REQD	PART NO.	NAME	SIZE	DESCRIPTION	SPECIFICATION
ASSEMBLY OPP.		ASSEMBLY SHOWN		LIST OF MATERIAL			

NEXT ASSY	USED ON	NEXT ASSY	FINAL ASSY
385-7722	385	REF	REF
385-7721	385	REF	REF
APPLICATION		QTY REQD	

UNLESS OTHERWISE SPECIFIED
DIMENSIONAL TOLERANCES
3 PLACE DECIMAL ± .010
2 PLACE DECIMAL ± .03
ANGULAR ± 0°30'
DIMENSIONS TO BE MET BEFORE PLATING.
CORNER RADIUS .030 ON 0' BORES AND SPOT FACES OF 1.250 DIA. OR LESS — .030 RADIUS ON GREATER THAN 1.250 DIA.

DRWN	SARGENTI	6-8-64
CHK'D		
APP'D	[illegible]	8-4-64
APP'D	[illegible]	8-4-64
APP'D	[illegible]	8-6-64
APP'D	[illegible]	8-7-64
APP'D		
APP'D		

DIAGRAM - FIRE DETECTION SYSTEM

HUGHES TOOL COMPANY
AIRCRAFT DIVISION
CULVER CITY, CALIFORNIA

385-7723

SCALE NONE | POWER PLANT 2123 | CODE 02731 | SHEET 1 OF 1

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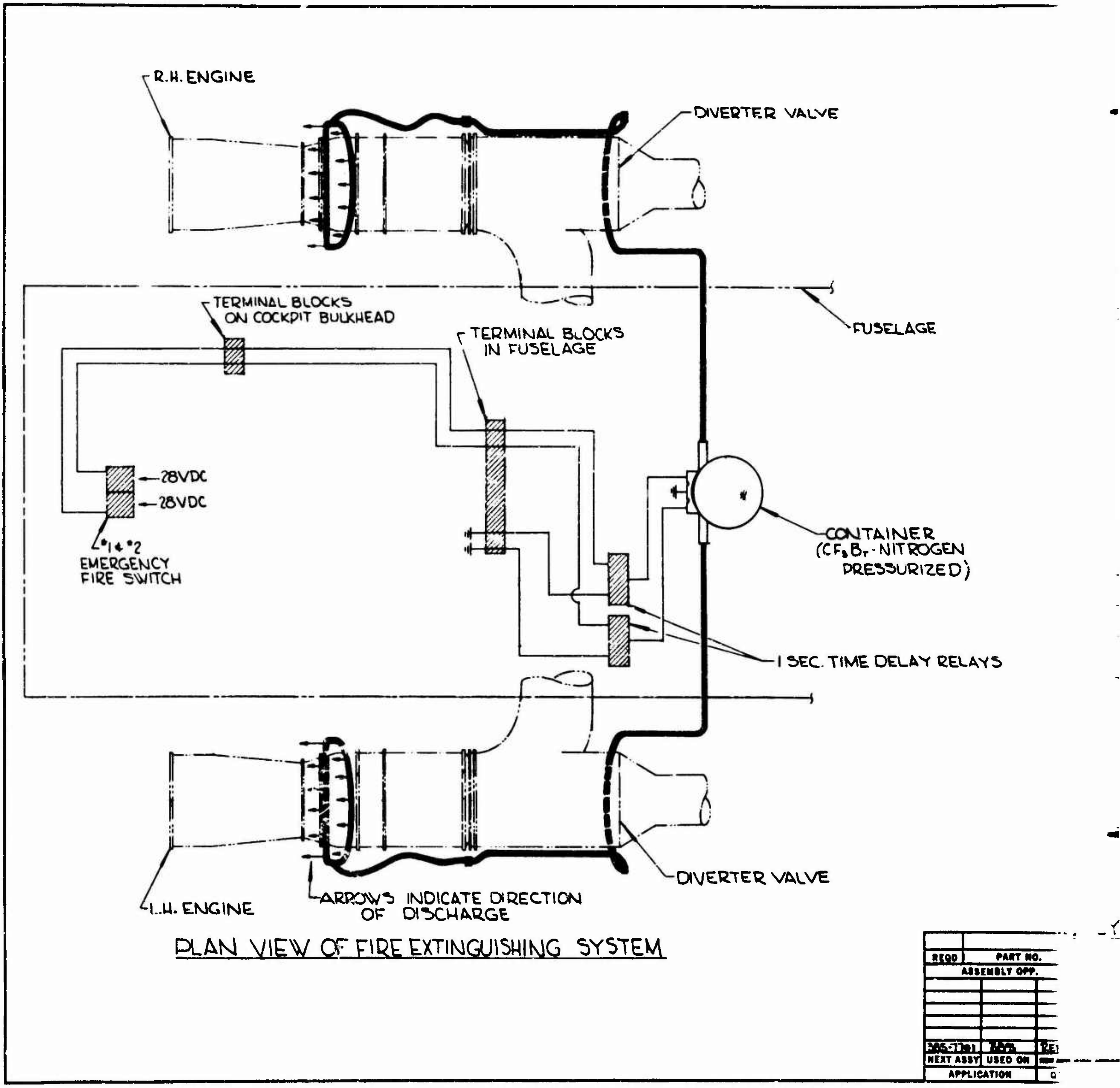
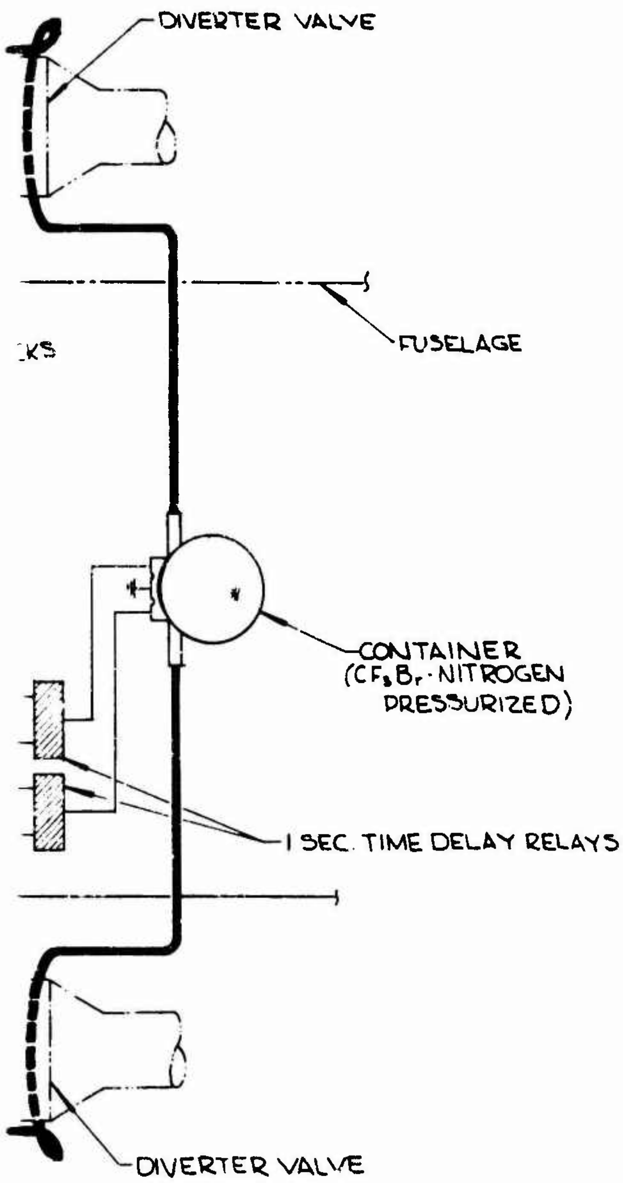


Figure 38. Fire Extinguishing System

REVISIONS

SYM	E.O.'S	DESCRIPTION	DRWN	APP'D	DATE
A		CORRECTED U.A. BLOCK			



385-7702

YSTEM

REQD	PART NO.	REQD	PART NO.	NAME	SIZE	DESCRIPTION	SPECIFICATION
ASSEMBLY OPP.		ASSEMBLY SHOWN		LIST OF MATERIAL			

NEXT ASSY	USED ON			TOLERANCES			
385-7701	385	REF	REF	UNLESS OTHERWISE SPECIFIED DIMENSIONAL TOLERANCES: 3 PLACE DECIMAL; 2 PLACE DECIMAL; ANGULAR. DIMENSIONS TO BE MET BEFORE PLATING. CORNER RADIUS ...	DRWN / CHK'D / APP'D / APP'D / APP'D / APP'D / APP'D / APP'D	DIAGRAM-FIRE EXTINGUISHER SYSTEM	HUGHES TOOL COMPANY AIRCRAFT DIVISION CULVER CITY, CALIFORNIA
APPLICATION		QTY REQD				SCALE NONE / POWER PLANT 1123	385-7702 / CODE 02731 / SHEET 1 OF 1

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3. WEIGHT AND BALANCE

Weight calculations have been made for all systems that contribute to the empty weight of the XV-9A. Additional calculations and estimates have been made to reflect the design gross weight and the alternate overload gross weight of the vehicle. The weight, balance, and inertial characteristics are presented in the following sections.

3.1 WEIGHT ANALYSIS

The design gross weight of the research vehicle is composed of two components: weight empty and useful load.

3.1.1 Weight Empty Determination

The empty weight of the XV-9A has been determined by compilations of actual, calculated, and estimated weights. Prior to actual weighing of the craft, the weights were approximately 55 percent actual, 43 percent calculated, and 2 percent estimated. The weight breakdowns follow the methods outlined in MIL-STD-451.

3.1.2 Inertia Characteristics

The roll, pitch, and yaw inertias about the helicopter center of gravity have been computed for the weight empty and gross weight conditions. These values are tabulated on the weight, balance, and inertia summary that follows in Sections 3.2 and 3.3. It will be noted that the inertia about the three conventional planes of the helicopter is higher for the 15,300-pound design gross weight than for the 25,500-pound alternate overload gross weight. The reason for this is that the design gross weight assumes use of full capacity of 500 gallons distributed in the forward and aft tanks. These are located at some distance from the cg and produce a large inertial moment about the cg. For the alternate overload gross weight, only 200 gallons of fuel are distributed in these tanks, which produces a much smaller inertial moment about the cg. The externally slung payload is assumed suspended from a point source at the cg, and therefore contributes no inertial moment about the cg.

3.1.3 Contract Weight Reductions

As outlined in this report, redesign was accomplished to reduce the weight of three major components that were carried over from the original Model 285 whirl test program. Total weight reduction realized was 133 pounds. The Y-duct and triduct were fabricated with lighter gage ducting, less flange material, and lighter stiffening, to reduce the weight 26 pounds for the triduct and 77 pounds for the Y-duct. The stationary swashplate was redesigned to reduce the sectional area and simplify the geometry. The new swashplate design resulted in a weight reduction of 30 pounds.

3.1.4 Sketches and Charts

During the design and development of this aircraft, stress and aerodynamic calculations required reference to load distributions in the nacelles, fuselage, and rotor blades. Appendix I includes figures that depict nacelle, fuselage, and rotor blade weight distributions as utilized during aircraft loads analysis.

3.2 WEIGHT STATEMENT

ROTORCRAFT
SUMMARY WEIGHT STATEMENT

	Weight Empty (Pounds)		
Rotor Group			2,797.4
Blade Assembly (3)		1,853.9	
Front Spar Installation	268.8		
Rear Spar Installation	212.5		
Interspar Structure	802.0		
Leading Edge	33.2		
Trailing Edge	55.8		
Hot Gas Inboard Duct	179.2		
Tension Straps	133.3		
Droop Stop	4.8		
Balance Weights	111.0		
Tip Cascade	17.3		
Cascade Valve and Mechanism	15.0		
Blade Root Bearings	21.0		

Hub Installations		658.6	
Structure	502.0		
Installation Hardware	9.8		
Bearings, Housing, etc	146.8		
Gimbal Assembly		124.5	
Shaft and Spoke		115.2	
Miscellaneous Spacers, Tubing, etc		45.2	
Tail Group (V-Tail)			134.7
Tail Cone Structure		27.6	
Fixed Stabilizer Surface (2)		78.7	
Rudder (Including 7.6-lb ballast) (2)		28.4	
Body Group			877.2
Fuselage Basic Structure		625.8	
Rotor Pylon and Fairing		187.4	
Cockpit Canopy		23.6	
Doors, Panels, Miscellaneous		40.4	
Alighting Gear Group, Land Type (CH-34A Landing Gear)			475.5

Location	Rolling Assembly	Structure	Controls	
Main Gear 11.00-12	166.6	231.3	13.7	411.6
Tailwheel 6.00-6	13.4	47.8	2.7	63.9

Flight Controls Group			954.7
Cockpit Controls		28.5	
System Controls - Linkage		91.6	
System Controls - Rotor Head		584.4	
Upper Beams	63.1		
Upper Rods	24.6		
Upper Torque Tubes	106.5		
Upper Supports	81.8		
Control Rods	36.5		
Rotating Swashplate	59.6		
Fixed Swashplate	44.0		
Spindle and Support	73.9		
Lower Links and Beams	47.5		
Hardware	46.9		

Hydraulic Cylinders Installation		88.7	
Yaw Control Installation		161.5	
Ducting	40.2		
Bellows	34.7		
Clamps, Supports, etc	27.7		
Yaw Control Valve	42.8		
Rudder-to-Valve Controls	16.1		
Nacelle Group			684.1
Engine Mounts		30.9	
Cowling, Structure, and Firewall		412.0	
Nacelle Support Pylons and Fairings		241.2	
Propulsion Group			2,170.7
Engine (2)		1,160.0	
Accessory Gearbox and Drive		74.2	
Air Induction System		32.2	
Exhaust System		78.2	
Tailpipe	27.0		
Connectors	26.2		
Supports	25.0		
Lubricating System		60.4	
Tanks	3.4		
Coolers	16.1		
Ducts	7.9		
Plumbing	19.2		
Rotor System	13.8		
Fuel System		255.2	
Forward Tank	92.2		
Aft Tank	94.4		
Sump Pumps	15.8		
Fill System	2.0		
Distribution System	26.4		
Vent System	3.7		
Drain System	0.3		
Valves and Miscellaneous	20.4		
Engine Controls		76.1	
Starting System (Ram Air)		11.2	
Rotor Drive System		423.2	
Diverter Valve (2)	173.3		
Diverter Valve Support	51.3		
Connector to Engine	39.5		
Duct, Seals, and Insulation	159.1		

Instrument and Navigational Group			50.0
Cockpit Instruments	27.0		
Transducers	22.0		
Navigational Equipment, Mapcase	1.0		
Hydraulic and Pneumatic Group			168.3
Pumps		27.6	
Reservoir		25.3	
Filter		5.4	
Cooler		12.2	
Valves and Plumbing		74.6	
Hydraulic Fluid		23.2	
Electrical Group			192.2
A-C System		11.9	
D-C System		180.3	
Generators (2)	51.4		
Battery and Support (2)	43.1		
Voltage Regulators (2)	2.7		
External Power	1.3		
Relays	14.3		
Lights	5.1		
Circuitry	57.4		
Supports	5.0		
Electronics Group			10.0
Furnishings and Equipment Group			75.9
Personnel Accommodations		14.0	
Instrument Panel		8.0	
Emergency Equipment, Fire Ext Sys		53.9	
Auxiliary Gear Group			50.0
Load Handling System		39.0	
Handling Gear		11.0	
TOTAL WEIGHT EMPTY			8,640.7

Useful Load and Gross Weight
(Pounds)

	Load Condition	
	Design	Alternate Overload
Crew (2)	400	400
Fuel (Type JP-4, Internal)	3,250 (500 gal.)	1,300 (200 gal.)
Oil (4-Hr Flight)	60	60
Payload	2,949	15,099
Total Useful Load	6,659	16,859
Weight Empty	8,641	8,641
Total Gross Weight	15,300	25,500

3.3 WEIGHT AND BALANCE STATEMENT

SUMMARY OF
WEIGHT, BALANCE, AND INERTIA

	Weight (Lb)	Center of Gravity: Inches Aft of X Ref Datum	Center of Gravity: Inches Above Z Ref Datum	Inertia, Slug Ft^2: Roll	Inertia, Slug Ft^2: Pitch	Inertia, Slug Ft^2: Yaw
Weight Empty	8,641	299.6	161.2	3,858	12,364	12,570
Design Gross Weight						
Useful Load	6,659					
Gross Weight						
Nominal	15,300	299.8	138.6	6,281	22,364	20,664
Most Aft CG	15,300	301.0	138.6	6,281	22,180	20,480
Most Forward CG	15,300	294.0	138.6	6,307	20,439	18,682
Alternate Overload Gross Weight						
Useful Load	16,859					
Gross Weight	25,500	298.7	139.7	5,989	18,613	16,907

Reference Data:

1. Reference datum (X-X axis) for horizontal center of gravity is 300 inches forward of rotor centerline.
2. Reference datum (Z-Z axis) for vertical center of gravity is 200 inches below waterline 0.0 (waterline 0.0 is rotor plane).

WEIGHT EMPTY SUMMARY

Item and Remarks	Weight (Lb)	Horizontal Distance (In.)	H Moment (In. -Lb) (x 10^{-3})	Vertical Distance (In.)	V Moment (In. -Lb) (x 10^{-3})
Rotor Group	2,797.4	300	839.2	199.6	558.5
Blades (27.5-ft radius, 31.5-in. chord)(3)	1,853.9	300		200	
Gimbal Assembly	124.5	300	37.4	206	25.6
Hub Structure	658.6	300	197.6	200	131.7
Shaft	115.2	300	34.6	189	21.8
Miscellaneous	45.2	300	13.6	189	8.5
Tail Group	134.7	648.3	87.3	117.3	15.3
Body Group	877.2	336.0	294.7	129.7	113.7
Fuselage					
Forward Section	70.7	183.6	129.8	126.4	89.3
Center Section	456.0	319.0	145.5	120.4	54.9
Aft Section	137.0	525.4	72.0	110.2	15.1
Carry-Thru Structure	26.1	297.8	7.8	143.0	3.7
Rotor Pylon	146.7	300.0	44.0	165.0	24.2
Pylon Fairing	40.7	307.6	12.5	168.7	6.9
Alighting Gear Group	475.5	302.4	143.8	96.5	45.9
Main Rolling Gear	166.6	263.0	43.8	88.0	14.7
Main Chassis	161.4	260.8	42.1	104.6	16.9
Brake Operating Mechanism	13 7	194.3	2.7	110.6	1.5
Tailwheel	13.4	616.5	8.3	69.4	0.9
Tailwheel Chassis	39.5	601.6	23.8	86.2	3.4
Tailwheel Locking Mechanism	3.0	402.7	1.2	101.3	0.3
Main Gear Structure	69.9	245.5	17.2	106.8	7.5
Tailwheel Structure	8.0	605.5	4.8	91.1	0.7

Item and Remarks	Weight (Lb)	Horizontal Distance (In.)	H Moment (In. - Lb) (x 10^{-3})	Vertical Distance (In.)	V Moment (In. - Lb) (x 10^{-3})
Starting System	11.2	256.9	2.9	138.3	1.5
Gas Starters (2) 9.5					
Controls 1.7					
Powerplant Controls	76.1	257.9	19.6	147.3	11.2
Drive System - Rotor Duct	423.2	293.1	124.0	161.3	68.4
Diverter Valves (2) 224.6					
Connectors 39.5					
Duct & Seals in Hub 159.1					
Instrument & Navigation Group	50.0	212.0	10.6	136.0	6.8
Instruments	27.0	160.0	4.3	128.0	3.4
Transducers	22.0	275.1	6.0	144.6	3.2
Map Case	1.0	190.0	0.2	114.0	0.1
Electrical Group	192.2	262.5	50.4	133.5	25.6
Generators (2)	51.4	226.0	11.6	135.0	6.9
Batteries (2)	40.0	329.0	13.2	119.0	4.8
Battery Supports	3.1	329.0	1.0	117.5	0.4
Ext. Power Receptacle	1.3	341.3	0.4	113.0	0.1
Voltage Regulators	2.7	346.0	0.9	160.3	0.4
Circuitry	76.9	260.3	20.0	144.5	11.1
Lights	5.1	214.7	1.1	108.0	0.6
Inverter & Transformer	11.7	185.0	2.2	115.0	1.3
Electronics Equipment Group	10.0	200.0	2.0	115.0	1.2
Hydraulic System Group	168.3	289.5	48.7	152.0	25.6
Pumps	27.6	258.6	7.1	144.3	4.0
Reservoir	25.3	318.0	8.0	154.0	3.9
Filter	5.4	276.0	1.5	152.0	1.8
Cooler	12.2	275.0	3.4	152.0	1.8
Plumbing	97.8	293.4	28.7	153.3	15.0

Item and Remarks		Weight (Lb)	Horizontal Distance (In.)	H Moment (In. - Lb) (x 10^{-3})	Vertical Distance (In.)	V Moment (In. - Lb) (x 10^{-3})
Flight Controls Group		954.7	322.9	308.3	169.3	161.7
Cockpit		28.5	181.9	5.2	118.[illegible]	3.4
Intermediate Linkage		91.6	287.5	26.3	149.6	13.7
Rotor Head Controls		584.4	299.0	174.7	189.2	110.5
Hydraulic Cylinders & Servos		88.7	287.9	25.5	149.3	13.2
Yaw Jet Control - Pipe		102.6	423.8	43.5	142.9	14.7
- Tail Unit		42.8	598.1	25.6	100.0	4.3
- Controls		16.1	460.7	7.4	116.4	1.9
Nacelle Group		684.1	280.8	192.2	155.3	106.2
Engine Mounts		30.9	229.0	7.1	154.8	4.8
Firewall		2.6	300.0	0.8	148.0	0.4
Cowling & Structure		409.4	271.4	111.1	152.1	62.3
Support Stubs (Power Module)		241.2	303.5	73.2	161.0	38.8
Propulsion Group		2,170.7	265.7	576.8	145.1	315.0
Engines (YT-64)(2)		1,160.0	241.5	280.1	146.4	169.8
Air Inlets		32.2	210.0	6.8	148.0	4.8
Exhaust Pipes		78.2	335.8	26.3	147.9	11.6
Accessory Gearbox & Drive		74.2	310.9	23.1	139.7	10.4
Lubrication System		60.4	244.4	14.8	139.9	8.4
Cooler & Supports	16.1					
Pumps & Air Ducts	7.9					
Distribution System	36.4					
Fuel System		255.2	311.3	79.4	113.1	28.9
Tanks (500-gal.) & Supports	186.6					
Pumps	15.8					
Distribution System	52.8					
Starting System		11.2	256.9	2.9	138.3	1.5
Gas Starters (2)	9.5					
Controls	1.7					

ALTERNATE HEAVY LIFT GROSS WEIGHT

Item and Remarks	Weight (Lb)	Horizontal Distance (In.)	H Moment (In. - Lb) ($x\ 10^{-3}$)	Vertical Distance (In.)	V Moment (In. - Lb) ($x\ 10^{-3}$)
Weight Empty	8,641		2,588.7		1,392.7
Pilot	200	190	38.0	130	26.0
Copilot	200	190	38.0	130	26.0
Oil	60	215	12.9	146	8.8
Fuel, Fwd Tank (100 gal.)	650	227	147.6	114	74.1
Fuel, Aft Tank (100 gal.)	650	404	262.6	110	71.5
Payload	15,099	300	4,529.7	130	1,962.9
Useful Load - Heavy Lift	16,859		5,028.8		2,169.2
Gross Weight - Heavy Lift	25,500	298.7	7,617.5	139.7	356.9

Item and Remark	Weight (Lb)	Horizontal Distance (In.)	H Moment (In. -Lb) (x 10^{-3})	Vertical Distance (In.)	V Moment (In. -Lb) (x 10^{-3})
Furnishings & Equipment Group	75.9	255.1	19.4	141.4	10.7
Seat Cushions	10.0	195.0	2.0	126.0	1.3
Belts & Harnesses	4.0	195.0	0.8	126.0	0.5
Instrument Board	8.0	165.0	1.3	120.0	1.0
Fire Extinguisher System	53.9	284.0	15.3	148.6	8.0
Auxiliary Gear	50.0	305.9	15.3	120.9	6.0
Ground Handling and Tie-Down	11.0	321.4	3.5	141.6	1.6
Cargo Handling	39.0	301.5	11.8	115.0	4.5
Total Weight Empty	8,640.7	299.6	2,588.7	161.2	1,392.7

DESIGN GROSS WEIGHT

Item and Remark	Weight (Lb)	Horizontal Distance (In.)	H Moment (In. -Lb) (x 10^{-3})	Vertical Distance (In.)	V Moment (In. -Lb) (x 10^{-3})
Weight Empty	8,641		2,588.7		1,392.7
Pilot	200	190	38.0	130	26.0
Copilot	200	190	38.0	130	26.0
Oil	60	215	12.9	146	8.8
Fuel, Fwd Tank (250 gal.)	1,625	227	368.9	114	182.5
Fuel, Aft Tank (250 gal.)	1,625	404	656.5	110	178.8
Payload	2,949	300	884.7	103	303.7
Total Useful Load	6,659		1,999.0		728.5
Gross Weight	15,300	299.8	4,587.7		138.6 2,121.2

3.4 WEIGHT COMPROMISES

As appropriate in the design and fabrication of a research aircraft, where only one of a kind is to be built, compromises in weight of some XV-9A components have been made in the interest of schedule and cost considerations. Compromises of this type are entirely appropriate and are universally applied for a research aircraft such as the XV-9A, but would be eliminated in a prototype or operational aircraft, where the additional cost and time required for minimization of weight is justified.

A weight analysis of the XV-9A has identified the specific weight compromises listed below, which account for a total of 630 pounds.

a. Rotor system — 171 pounds

Excessive weight of built-up hub structure and blade root structure, excess blade segment sealant, excessive weight of blade inboard ducts, and overweight segment flexures

b. Fuselage and empennage structure — 56 pounds

Lack of taper of spar caps in V-tail and of fuselage longerons, no lightening holes in frames, and overweight lower rotor thrust bearing support structure

c. Landing gear — 20 pounds

Use of the CH-34A landing gear is not optimum for the XV-9A

d. Flight control group — 144 pounds

Mounting of flight control actuators compromised for existing rotor system, overweight condition of rotor head controls and supports

e. Gas generator support and nace... structure — 210 pounds

Independent mounting of gas generators, diverter valves, and tailpipes; excessive size and weight of J-85 diverter valves; nonoptimum structure in nacelle area.

f. Hydraulic and electrical systems — 29 pounds

4. PERFORMANCE

4.1 HOVERING FLIGHT

Calculations of the hover performance of the XV-9A are based on the NACA procedures summarized in Reference 10. The rotor profile drag coefficient was adjusted to allow for the 18 percent thickness of the XV-9A rotor blade. Both the induced power and the profile power required were increased, for the effect of a linear (rather than ideal) twist of -8 degrees. A design condition of 6,000 feet and 95 degrees F was chosen. A down-load interference between rotor and fuselage of two percent of thrust was assumed. For these conditions, the design gross weight of the XV-9A aircraft was taken as 15,300 pounds. The alternate overload gross weight at sea level standard was taken at 25,500 pounds, using comparable procedures.

4.2 LEVEL FLIGHT

The total equivalent parasite drag area of the XV-9A aircraft was estimated to be 22.0 square feet, based on the size and shape as shown on the general arrangement drawing, Figure 2. A breakdown of the equivalent parasite drag area of the individual components is presented in Table 1. References 11 and 12 are used to estimate the parasite area of the remaining components.

The power required for level flight at sea level was computed using the parasite area established in Table 1 and standard NACA performance calculation procedures outlined in Reference 13. Figure 39 shows power required versus speed at sea level for the design gross weight of 15,300 pounds, and also for 10,000-pound gross weight. For test purposes, it has been established that a gross weight of 10,000 pounds will be sufficient to allow reasonable instrumentation and fuel loads for short missions.

The rotor power available at sea level is also shown on Figure 39 for takeoff, military, and normal continuous power conditions. Each level of power is shown to be constant versus forward speed. This is approximately true at speeds below 200 knots because the extra power that is developed as a result of ram pressure rise at the engine is just about compensated for by the power required to overcome the ram drag of accelerating the air up to the helicopter forward speed.

TABLE 1
PARASITE DRAG BREAKDOWN

Components	Applicable Areas as Indicated (Sq Ft)	Drag Coefficient	Equivalent Drag Area (Sq Ft)
Rotor head			
Hub	6.25 (max. frontal area)	0.75	4.70
Blade shanks	12.2 (max. frontal area)	0.25	3.05
Fuselage	23.0 (max. frontal area)	0.085	1.96
Landing gear			
Wheels	5.00 (max. frontal area)	0.30	1.50
Tailwheel	2.00 (max. frontal area)	0.50	1.00
Struts	4.30 (max. frontal area)	1.20	5.20
Empennage	54.0 (surface area)	0.02	1.08
Pylon	14.00 (max. frontal area)	0.06	0.84
Nacelles	13.20 (max. frontal area)	0.05	0.66
			19.99
Interference, roughness,	10%		
and miscellaneous	10%		2.00
Total equivalent parasite area			21.99

Using the takeoff power rating, which is good for ten minutes (long enough to get stabilized speed data), a maximum helicopter speed of 156 knots is calculated at sea level at 10,000-pound gross weight. To be conservative, the estimated maximum speed was reduced to 150 knots in the Model Specification (Reference 3) for 10,000 pounds (and to 140 knots for 15,300 pounds).

POWER VS VELOCITY
LEVEL FLIGHT AT SEA LEVEL

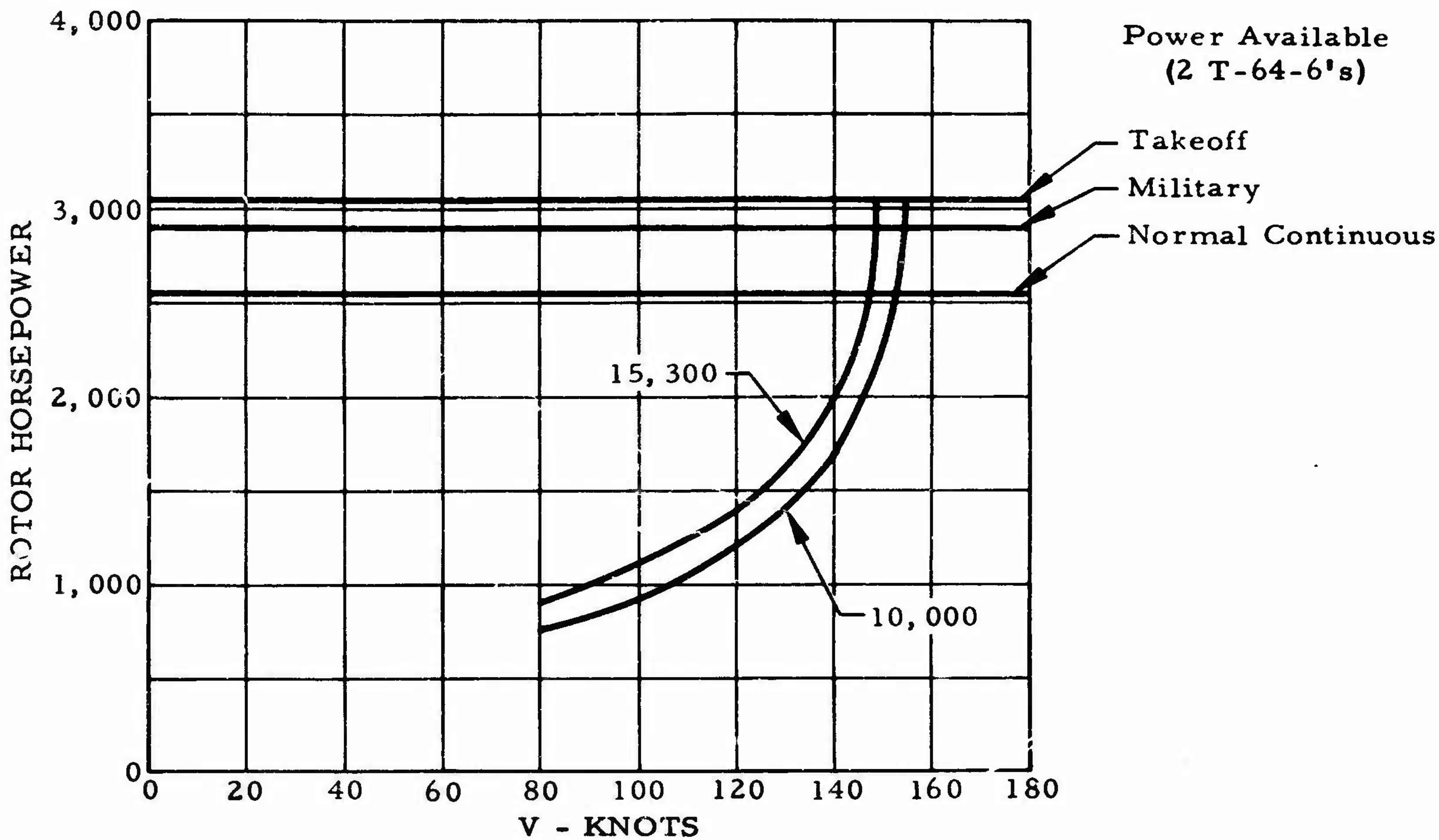


Figure 39. Performance Characteristics

5. STABILITY AND CONTROL

The requirements of Specification MIL-H-8501A (Reference 14) were used as the stability and control objectives. Theoretical analysis, based on the configuration shown in Figure 2, indicates that the Research Aircraft, although not meeting certain requirements of Reference 14, will have adequate stability and control characteristics to investigate safely the required flight conditions.

5.1 HOVERING FLIGHT

The expected handling characteristics of the Research Aircraft in hovering flight for 15,300-pound and 25,500-pound gross weight are shown in Figure 40 as the solid symbols. Also shown are the boundaries of acceptable handling characteristics as defined in References 14 and 15.

5.1.1 Handling Characteristics in Pitch

It is seen in Figure 40 that the handling characteristics in pitch are below the requirements of Reference 14 at 15,300-pound gross weight, primarily due to the low damping characteristics of the aircraft. The characteristics are improved, but still remain below the requirements at 25,500 pounds. However, it is felt that the aircraft can be flown in the hover condition without undue pilot effort. This conclusion is based on the flight results of Reference 15.

The tests reported in Reference 15 were conducted using an S-51 helicopter to determine the effects of various combinations of damping and control power on helicopter handling characteristics for visual and instrument flights.

The basic damping and control power levels of the S-51 in pitch and roll are shown in Figure 40. It can be seen that the S-51, with its basic damping and control power, is not able to meet the handling requirements of Reference 14. However, the helicopter was flown under the critical handling requirements of instrument flight without noticeable difficulties. Further, the helicopter was flown under instrument flight with one-half the basic damping and original control power. Pilots' comments indicated that the handling characteristics were poorer than with the original damping and control power, but the helicopter could be flown without excessive pilot effort.

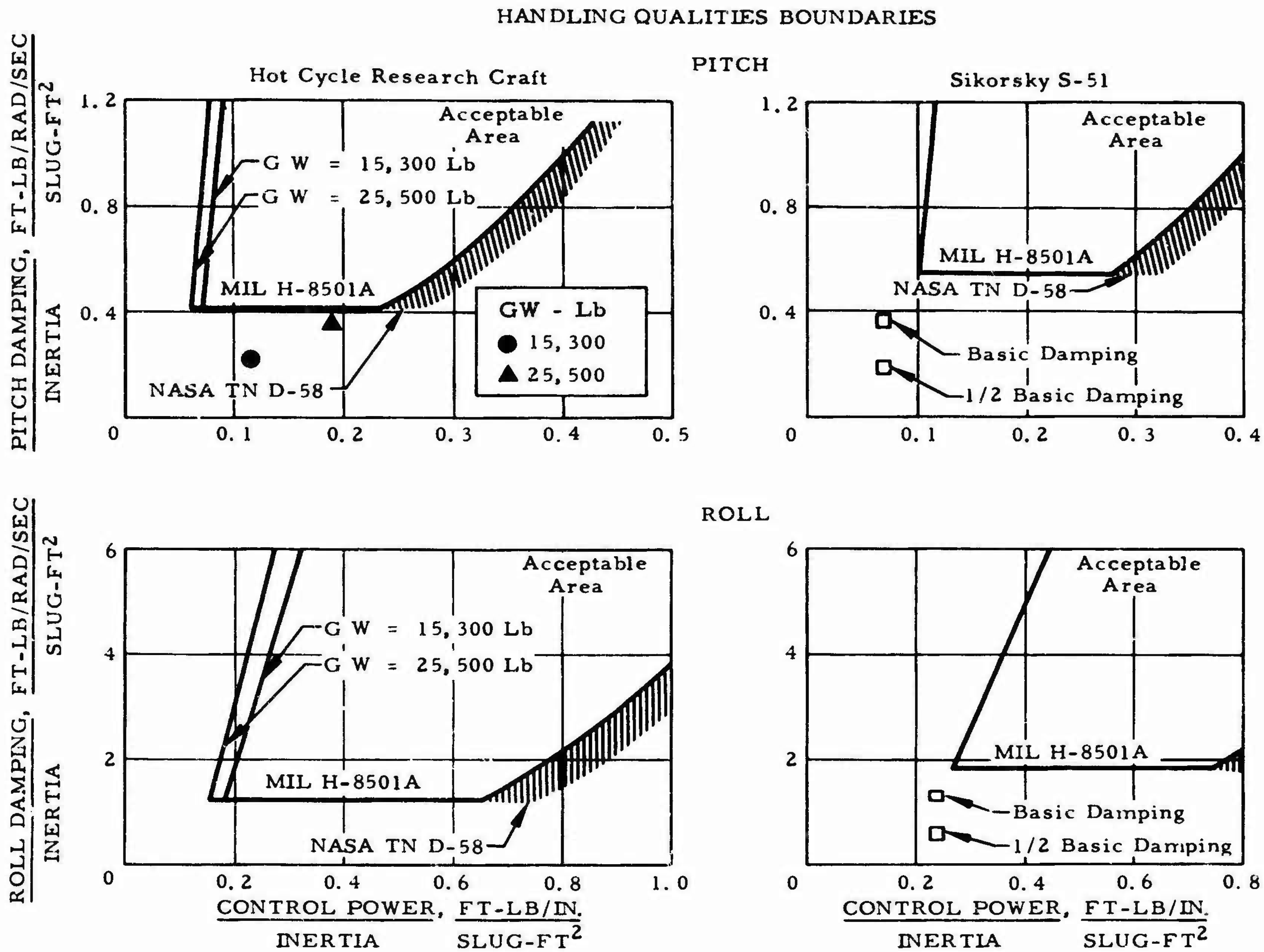


Figure 40. Handling Characteristics

Figure 40 shows that, for the one-half basic damping case, the S-51 has approximately 0.17/0.55, or 31 percent, of the relative damping per the requirements of Reference 14. The XV-9A aircraft has, at 15,300-pound gross weight, a relative damping of 0.22/0.42, or 52 percent, which is approximately 70 percent more relative damping than in the S-51 case. Therefore, the XV-9A should have superior handling characteristics compared with the S-51 for the one-half damping case.

5.1.2 Handling Characteristics in Roll

Based on reasoning similar to the pitch case above, the relative damping in roll of the XV-9A at 15,300-pound gross weight is 58 percent, compared with 32 percent for the S-51 for the one-half damping case. Therefore, it is expected that the XV-9A, with approximately 80 percent more relative damping in roll, will have superior handling characteristics over the S-51 for the one-half damping case.

5.1.3 Handling Characteristics in Yaw

The XV-9A meets the hovering yaw response characteristics per Reference 14, as shown by the tabulation below (based on an estimated maximum yaw thrust of ±300 pounds total).

	Specification MIL-H-8501A (Degrees)	XV-9A (Degrees)
Yaw displacement at the end of one second per inch of pedal deflection	3.77	4.1
Yaw displacement at the end of one second for full pedal deflection	11.3	12.4
Yaw displacement at the end of one second from the most critical azimuth position during a 35-knot wind for full pedal deflection	3.77	5.2

The yaw angular velocity damping of the XV-9A is essentially zero (due to the absence of a tail rotor). This characteristic, which is typical for all tip-driven helicopters, does not violate the requirements of Reference 14, because damping in yaw is discussed there in the sense of being "preferred", rather than being specifically required as for the pitch and roll case.

It should also be noted that the yaw damping criterion was included in Reference 14 chiefly because of the gust sensitivity in yaw of single-rotor helicopters with tail rotors. Due to the absence of a tail rotor, the XV-9A will be less sensitive to gusts in yaw. Therefore, the yaw damping criterion would not be a major consideration in the design of the XV-9A.

5.1.4 Summary

The Hot Cycle Research Aircraft, although not meeting the damping characteristics of Reference 14 in both pitch and roll except at the high gross weight of 25,500 pounds in roll, is expected to have superior handling characteristics in pitch and roll over those of an experimental version of the S-51, which could be flown without difficulty with one-half of the original damping. Thus, the XV-9A should have reasonable handling characteristics.

5.2 FORWARD FLIGHT

5.2.1 Directional Stability

According to Reference 14, "the helicopter shall possess positive, control fixed, directional stability and effective dihedral in both powered and autorotative flight at all speeds above 50 knots, $0.5\ V_{max}$, or the speed for maximum rate of climb, whichever is lowest". With this requirement in mind, an analysis (based on Reference 16) was made of the research aircraft with the configuration of Figure 2. It was found that a V-shaped tail, with a true area of 54 square feet and 45 degrees of dihedral, produces the required directional stability.

5.2.2 Longitudinal Maneuver and Dynamic Stability

Paragraphs 3.2.11.1, 3.2.11.2, and 3.2.12 of Reference 14 are concerned with longitudinal maneuver with dynamic stability. Calculations show that the V-tail with 45 degrees of dihedral sized at 54 square feet, for adequate directional stability (see 5.2.1), will have a proper amount of projected horizontal area to meet adequately the longitudinal maneuver and dynamic stability requirements.

5.2.3 Stick Position Versus Speed

Paragraph 3.2.10 of Reference 14 specified that "the helicopter shall at forward speeds possess positive static longitudinal control force and control position stability with respect to speed".

Because the XV-9A aircraft uses servos, the stick force requirement will be met by means of an artificial "feel" system. Mechanical springs giving a longitudinal and lateral stick force gradient of one-half to one pound per inch of stick travel will be used. Trim will be accomplished by the use of two electrical actuators that reset the feel springs to zero force. A four-position switch mounted on the cyclic stick will provide beep control for operation of the actuators.

Stable control position stability is shown in Figures 41 and 42 for various stabilizer incidences for gross weights of 10,000 and 15,300 pounds. This represents the range from design gross weight to the minimum weight for test purposes.

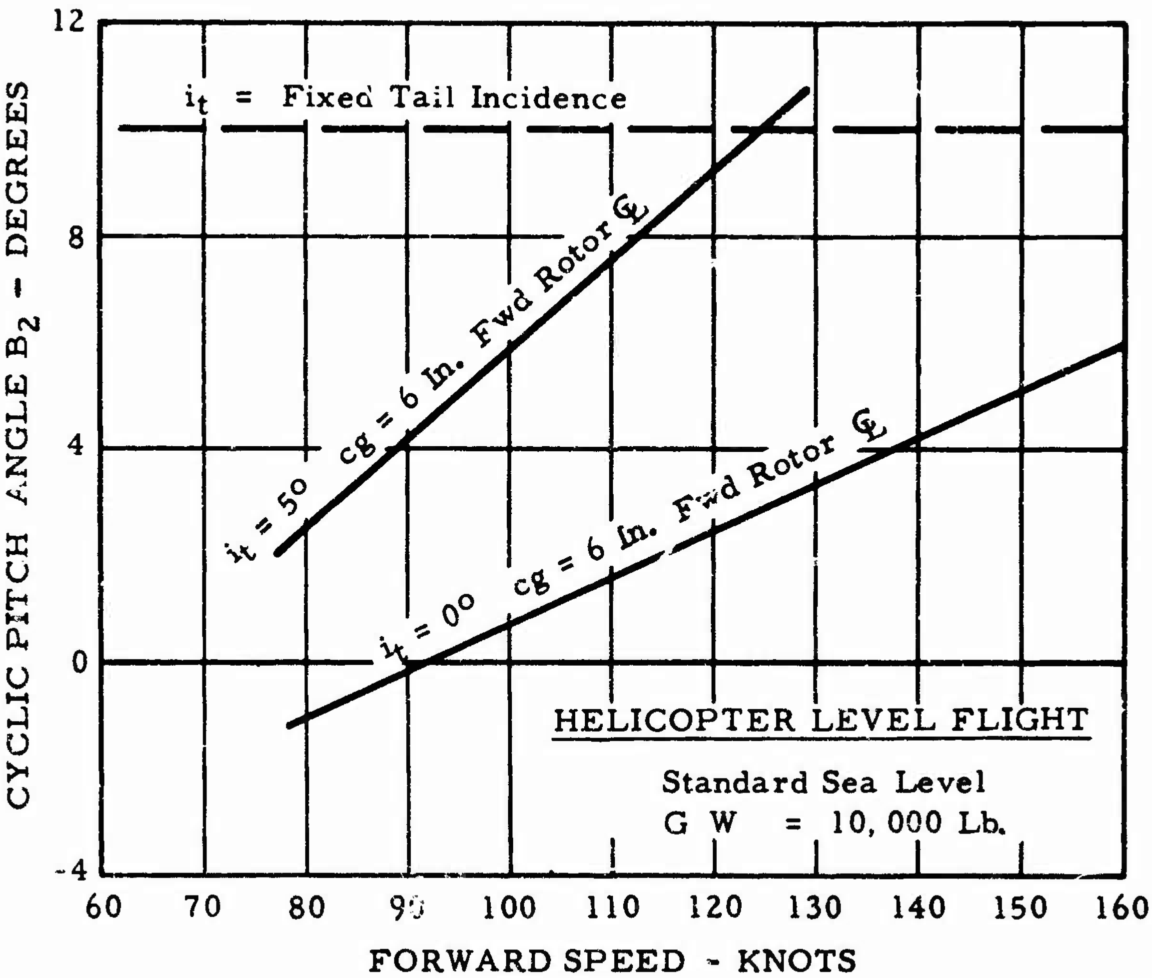


Figure 41. Cyclic Pitch Characteristics at
10,000-Pound Gross Weight

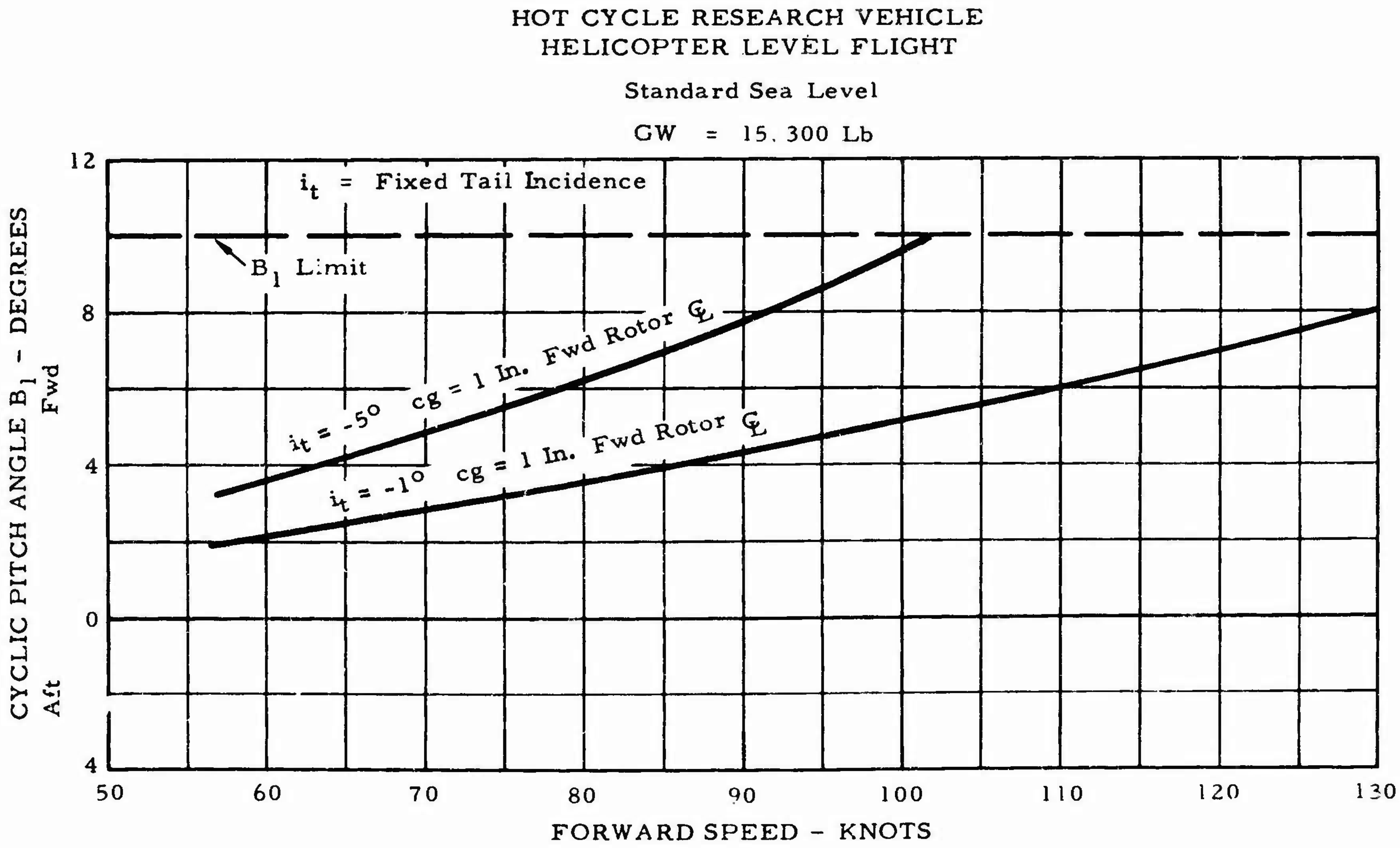


Figure 42. Cyclic Pitch Characteristics at 15,300-Pound Gross Weight

6. DYNAMICS

6.1 ROTOR DYNAMICS

The flapwise and chordwise natural frequencies of the XV-9A rotor are shown in Figures 43 and 44. It is seen that no resonances are expected to occur near the operating range of the rotor. Figure 43 shows the collective modes, which are those that respond to 3, 6, and 9 per rev, and Figure 44 shows the cyclic modes, which respond to 1, 2, 4, 5, 7, and 8 per rev. The mode shapes corresponding to these two sets of modes are shown in Figure 45.

The flapwise collective modes are computed with the blade pinned at Station 19 (the flapping hinge), and they assume that the hub, which is free to tilt about its gimbal mount, does not participate in the flapping motion. This will only result when equal blade root shear forces are applied in phase to the hub by each blade. For a three-bladed rotor, this condition is obtained with 3, 6, and 9 per rev excitation (multiples of the number of blades). Similarly, in the chordwise direction, the hub will act as a pin joint when the blades are all bending in the same direction at the same time. For a three-bladed rotor, this condition is obtained by a 3, 6, and 9 per rev excitation.

The flapwise cyclic mode is excited when unequal or out of phase shear forces are applied to the hub by the blades. The hub will then tilt, and the length of the blade is assumed to extend to the rotor centerline. The flapping hinge will continue to function as a hinge, so that the blade effectively has two hinges, one at the rotor centerline and one at the flapping hinge. This mode will be excited by any harmonic other than even multiples of the number of blades; that is, 1, 2, 4, 5, 7, and 8 per rev for a three-bladed rotor. Also, in the chordwise direction the blades will act as cantilever beams extending from the rotor centerline whenever they are excited unsymmetrically by harmonics other than even multiples of the number of blades. For this three-bladed rotor, this corresponds again to 1, 2, 4, 5, 7, and 8 per rev.

The natural frequencies shown in Figures 43 and 44 are a combination of test and calculation. Computations of natural frequency are carried out using a matrix procedure described in Reference 17. These calculations provide both the nonrotating frequencies and the variation of natural frequency with rotor speed. The value of nonrotating

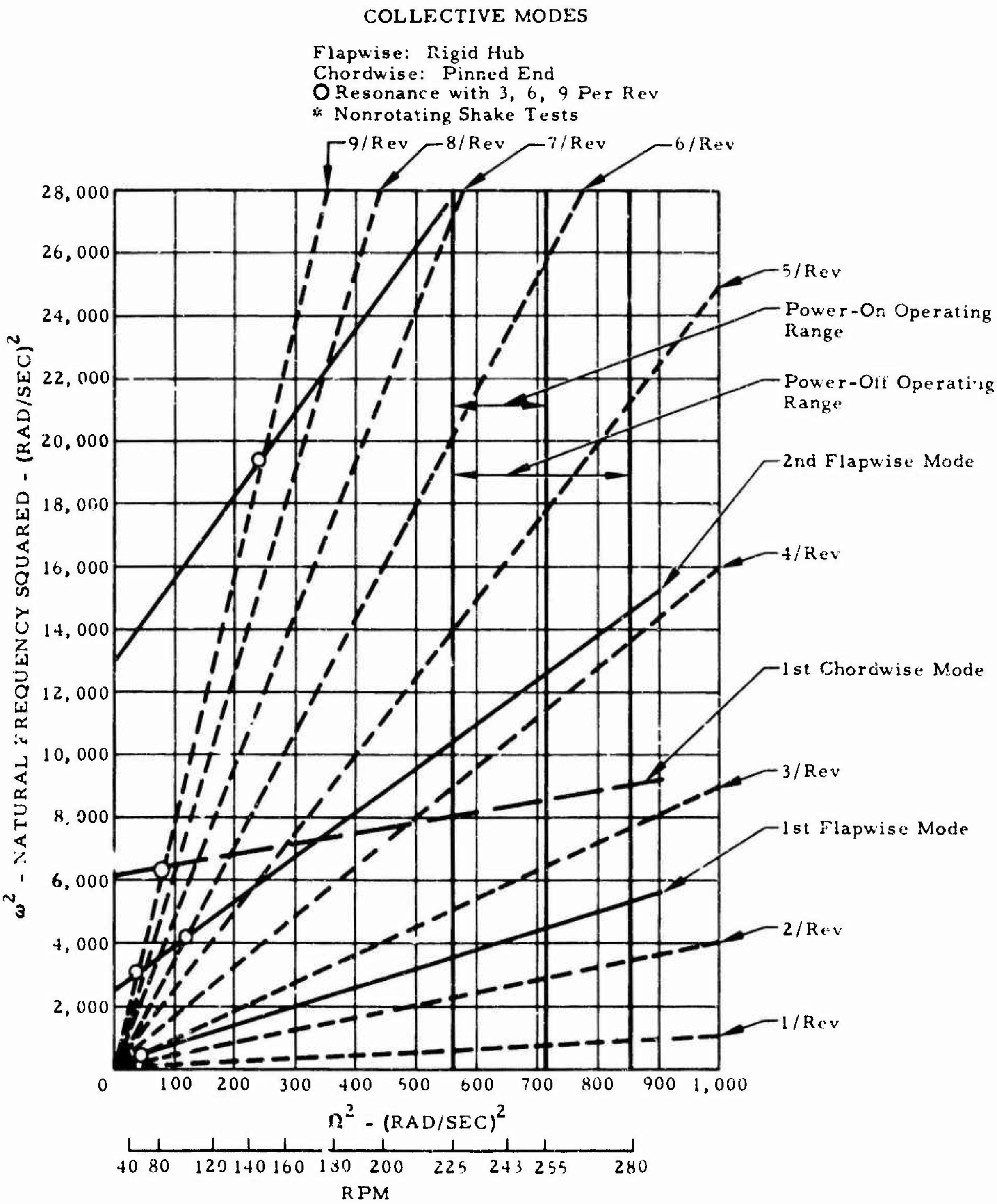


Figure 43. Collective Mode Resonances

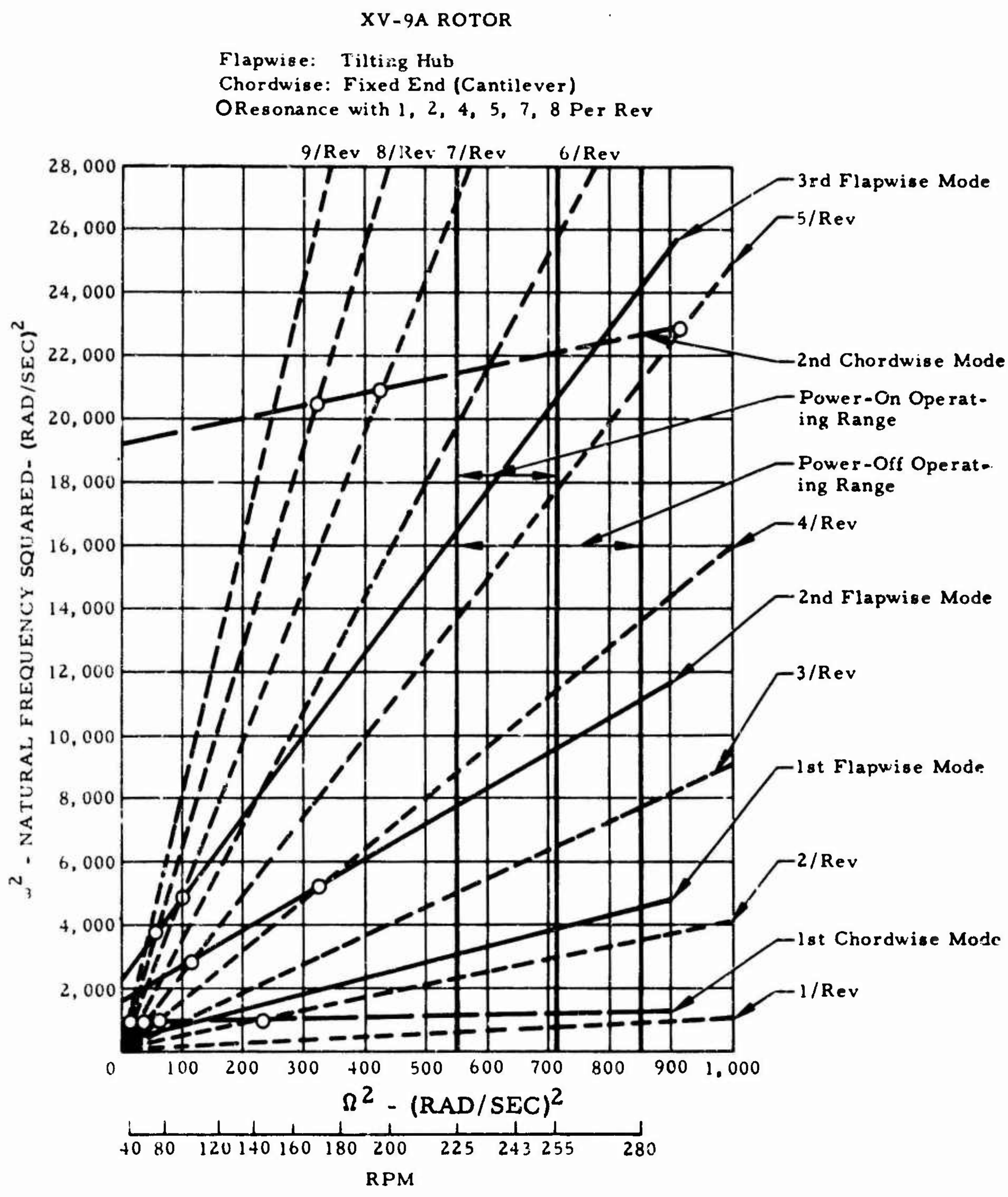


Figure 44. Cyclic Mode Resonances

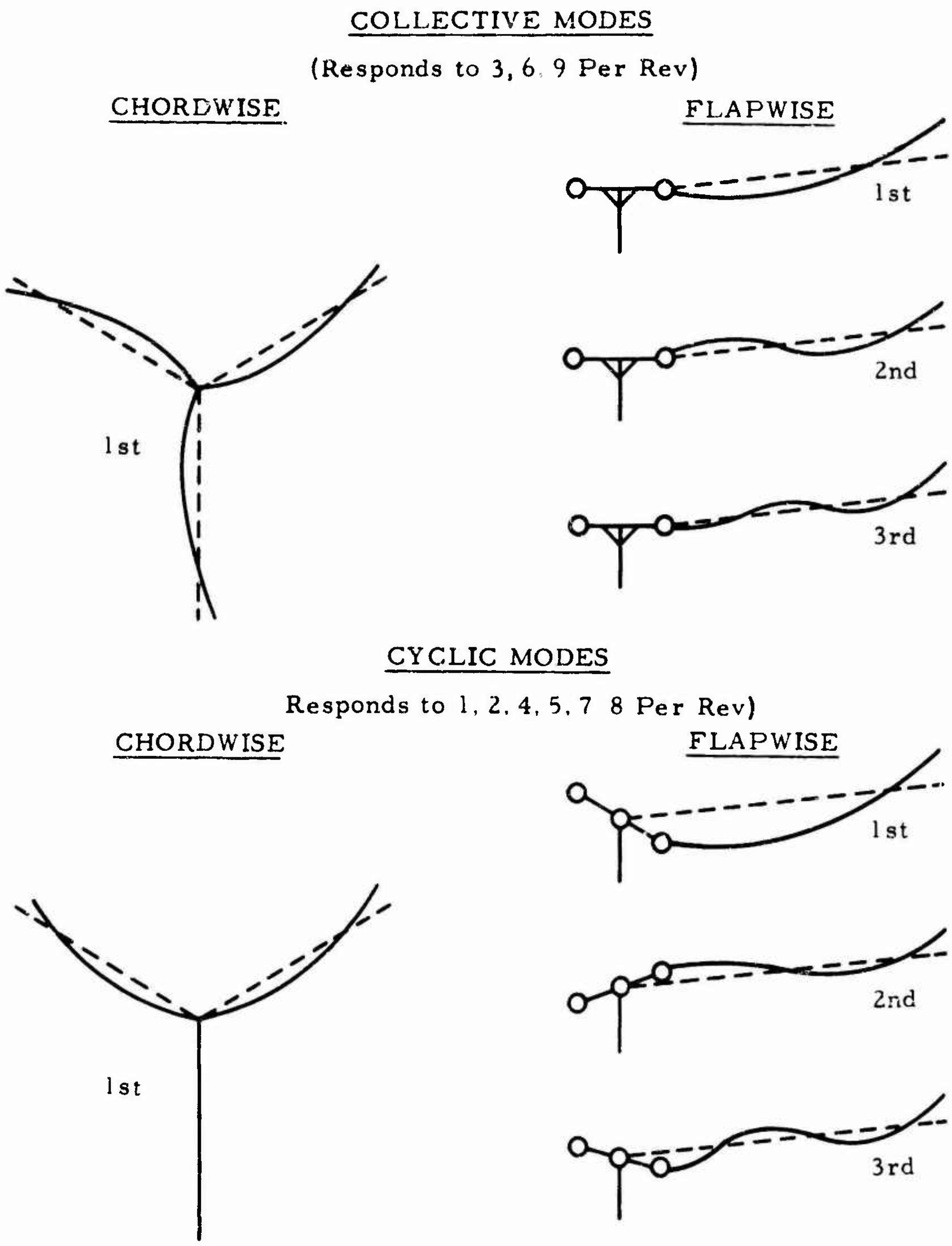


Figure 45. Collective and Cyclic Mode Shapes

frequency for the first three collective flapwise modes was adjusted to equal values obtained by shake tests of the XV-9A blade. These values, plus the slope of the lines from the matrix calculations, permit drawing the curves shown in Figure 43.

The cyclic chordwise first and second mode frequency curves use the same slope as that derived by calculation; however, the nonrotating frequencies cannot easily be obtained from test, because of the difficulty of maintaining adequate tension in the strap retention. Instead, the calculated values of the cyclic chordwise frequency are adjusted, based on a comparison of calculations and tests for the Model 285 rotor tested earlier and reported in Reference 18. The Model 285 rotor was found to have a first mode cyclic chordwise (cantilever) frequency of 1.25 per rev at 100-percent rpm, which was 0.78, the calculated value. Applying the same factor to the XV-9A rotor first mode calculated frequency, the predicted first mode cyclic chordwise frequency is expected to be 1.31 per rev.* This is approximately 5 percent higher than that of the Model 285 rotor. (This frequency increase, which reduces the response to 1-per-rev excitation, plus the redesign of the rotor to higher loads than were used for the Model 285 rotor design, is expected to provide a blade with a satisfactory life.)

After the predicted value of first mode cyclic chordwise frequency at 100-percent rpm is known, it is possible to use the computer-derived slope to draw the first mode chordwise frequency curve shown in Figure 44. A similar procedure was used for second mode cyclic chordwise frequency and for collective first mode chordwise frequency.

It is seen that no chordwise or flapwise resonance (circled points) occur in the operating range of the rotor. The closest possible resonances occur as shown in Table 2. Except for one resonance (the last) that occurs above the power-off operating range, all these resonances occur at less than 90 percent of the minimum operating rpm (225 rpm). This margin of 10 percent or more is considered adequate, since the rotor system structure has some inherent damping. Even if resonance of the cyclic chordwise second mode with 5 per rev is found to produce high loads, it should be noted that the occurrence of this resonance will probably be for only four cycles at a time. If the loads

---

*The blade first mode cyclic chordwise frequency was measured as 1.43 per rev during the whirl tests reported in Reference 19.

encountered are deemed to be too high, the power-off operating range can be reduced.

TABLE 2
RESONANCES CLOSEST TO OPERATING RANGE

Mode	Excitation	RPM at Resonance
Collective chordwise first mode	6/rev	130
Collective flapwise third mode	9/rev	150
Cyclic chordwise first mode	2/rev	145
Cyclic chordwise second mode	7/rev	195
Cyclic chordwise second mode	5/rev	295
Cyclic flapwise second mode	4/rev	175

Note: Power-on operating range = 225-255 rpm. Power-off operating range = 225-280 rpm.

6.2 FUSELAGE VERTICAL, LATERAL, AND TORSIONAL NATURAL FREQUENCIES

Because of the rather long and slender proportions of the fuselage of the XV-9A, the possibility exists that a coupling might occur between the natural bending frequencies of the fuselage and pertinent harmonics of the rotor speed. Estimates were made of the vertical, lateral, and torsional natural frequencies of the fuselage, taking into account the fuselage stiffness properties. Results are presented in Table 3.

The rotor for the XV-9A aircraft will operate at 243 rpm. The primary excitation of the fuselage by the three-bladed rotor is 3 per rev. Examination of the estimated frequencies of the modes listed in Table 3 shows that the natural frequencies are well removed from the primary rotor exciting frequency. Fuselage shake tests will be performed prior to flight, to establish the actual frequencies of the important fuselage modes.

TABLE 3
ESTIMATED FREQUENCIES

Mode	Cycles Per Second	Frequencies Per Rev (At 243 rpm)
First vertical bending	33	8.15
First lateral bending	30	7.40
First torsional	18	4.44

7. STRUCTURAL DESIGN CRITERIA

7.1 ROTOR BLADE, HUB, POWER MODULE, AND FUSELAGE LOADS AND LOAD FACTORS

Paragraphs 7.1.1 through 7.1.12 list the conditions that were investigated in the design and stress analysis of the rotor system, power module, and fuselage (empennage and aft fuselage section loads are covered in Section 7.3). The loads and load factors are basically those of Reference 20, plus revisions to include loads information from the whirl test results.

Design parameters of the rotor system are based on the design gross weight of 15,300 pounds.

The maximum design maneuver limit load factor is 2.5 g.

Maximum design loads are to be considered in combination with maximum temperature and pressure. Rotor blade tip speeds are as follows:

a. Hovering, cruise, and maneuver 700 ft/sec

b. Overrev for limit load
[(1.1 x maximum power-on rpm) 1.05] 848 ft/sec

7.1.1 Flight Design Criteria

Maneuver	Fwd Speed (knots)	Rotor RPM	Limit Load Factor	CG Ref Rotor ℄	Tilt of Rotor Plane & Lift Vector Tilt (degrees)	Tail Load (pounds)	Angular Acceleration Pitch (rad/sec^2)	Angular Acceleration Roll (rad/sec^2)	Angular Acceleration Yaw (rad/sec^2)
Symmetrical pullout	100	243	2.5 (1)	1 in. fwd	10° aft 10° fwd or aft	-318 (3) 0	±1.93 ±1.80	0	0
Rolling pullout	100	243	2.0 (2)	1 in. fwd	7.2° aft & 7.0° right or left	-318 (3)	±1.26	±3.14	±1.17 (4)
Maximum yaw	200	(See Section 7.3)							

Notes: (1) 2.5-g limit at design gross weight, per Reference 21, paragraph 3.1.10.
(2) 2.0-g (0.8 x 2.5 g), per Reference 21, paragraph 3.2.3.1.
(3) - sign indicates download.
(4) Maximum pedal displacement is assumed during the rolling pullout.

7.1.2 Load Factor in Ground Flapping

a. Blade droop stop and hub 9° tilt stop 2.5 g limit

b. Hub 2° tilt stop 2.0 g limit

7.1.3 Wind Loads

Wind loads shall be those resulting from a 40-knot wind from horizontal direction (per Reference 21, paragraph 3.4.6.2).

7.1.4 Rotor Starting Condition

Rotor starting condition is: static thrust (maximum) of 500 pounds per blade at blade tips reacted by rotational inertia rotor; blades in -2-degree 1-g drooped position. Rotational speed is zero.

7.1.5 External Chordwise Pressure Distribution, Cruise and 2.5-g Maneuver Condition

Use data in Reference 22, pages 45 and 46, Figures 25, 26, and 27, and increase values by ratio of tip speed squared

$$\frac{(700)^2}{(650)^2} = 1.16$$

and add 2.1 pounds per square inch from 55 percent to 85 percent chord (inertia loads are included). In addition, buffeting fatigue of blade aft skins must be guarded against, by comparing gages and panel sizes with those of existing high-speed aircraft.

7.1.6 Blade Torsion Loads

a. Cruise condition (coning = 2.24°, tilt = 0° to 3° aft)

6,550 ±13,860 inch-pound limit

b. Weighted fatigue condition (Figure 46) (coning = 2.24°, tilt = 0° to 6° aft)

13,100 ±25,140 inch-pound limit

BLADE TORSION LOADS
WEIGHTED FATIGUE CONDITION

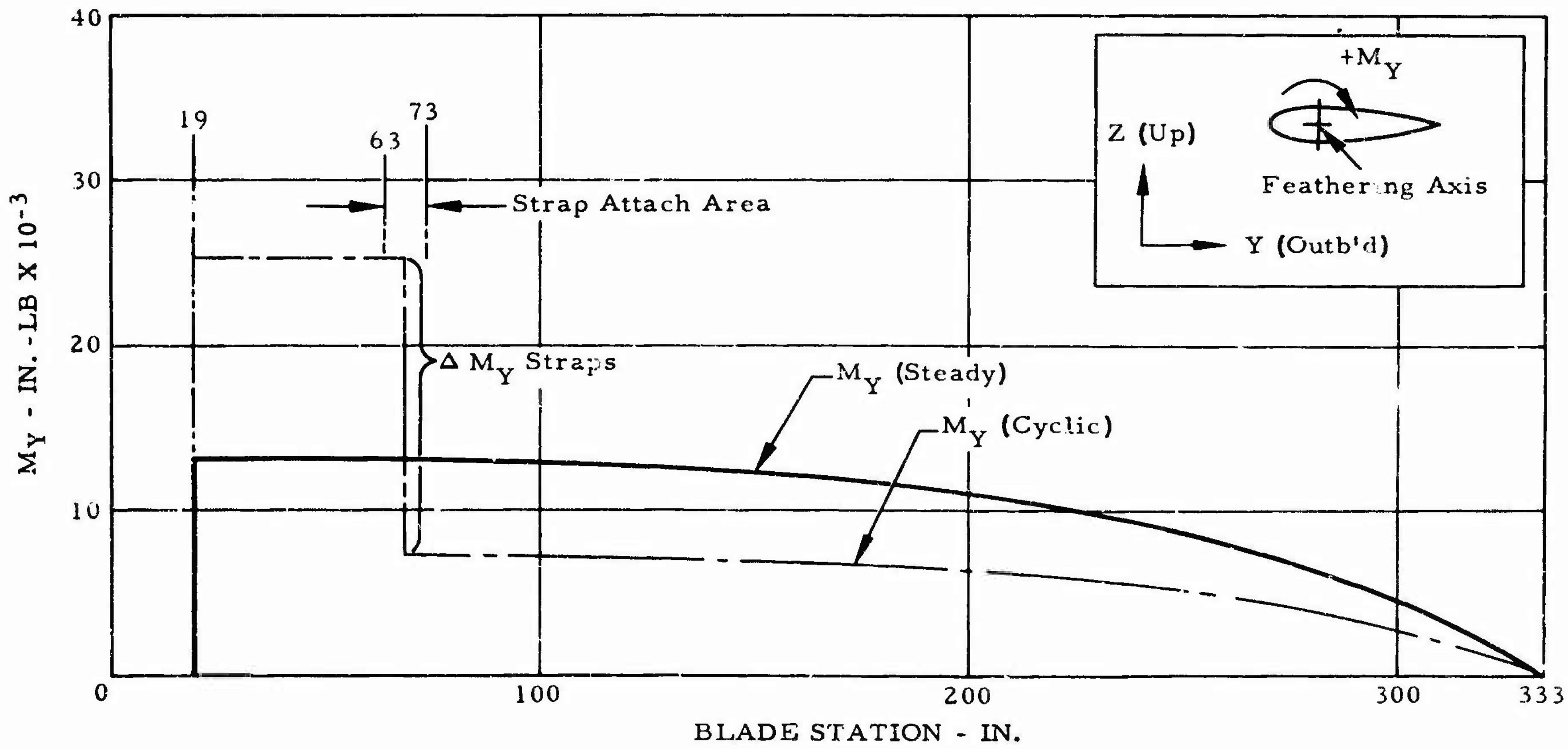


Figure 46. Blade Torsion Loads - Weighted Fatigue Condition

c. Maneuver, 2-1/2-g (Figure 47) (coning = 5.6°, tilt = 10° aft)

20,170 ±32,300 inch-pound limit recovery

Notes:

(1) Positive value indicates blade nose down.

(2) Values given include strap torsion.

(3) Steady torsion should be checked in both directions.

(4) When analyzing swashplate and lower controls, critical phasing of above loads from each of the three blades should be used.

(5) A dynamic (limit) factor of 1.25 shall be used for the ultimate conditions of blade root torsion (item c above). This factor may be reduced to 1.10 between actuating cylinders and the top of the shaft. The usual 1.5 ultimate factor is also required.

(6) The hydraulic cylinder load input shall be capable of supplying sufficient load to actuate the rotor blades under the design maneuvers (item c above).

(7) The hydraulic servo system shall be capable of rotating the swashplate at least 26.7 degrees per second but shall be restricted so that the swashplate shall not rotate faster than 40 degrees per second.

7.1.7 Blade Shear Loads

a. Normal shear

See Figures 48 and 49.

b. Chordwise shear just outboard of blade strap fittings

(1) Weighted fatigue = 200 ±866 pounds (derived condition from Figure 50)

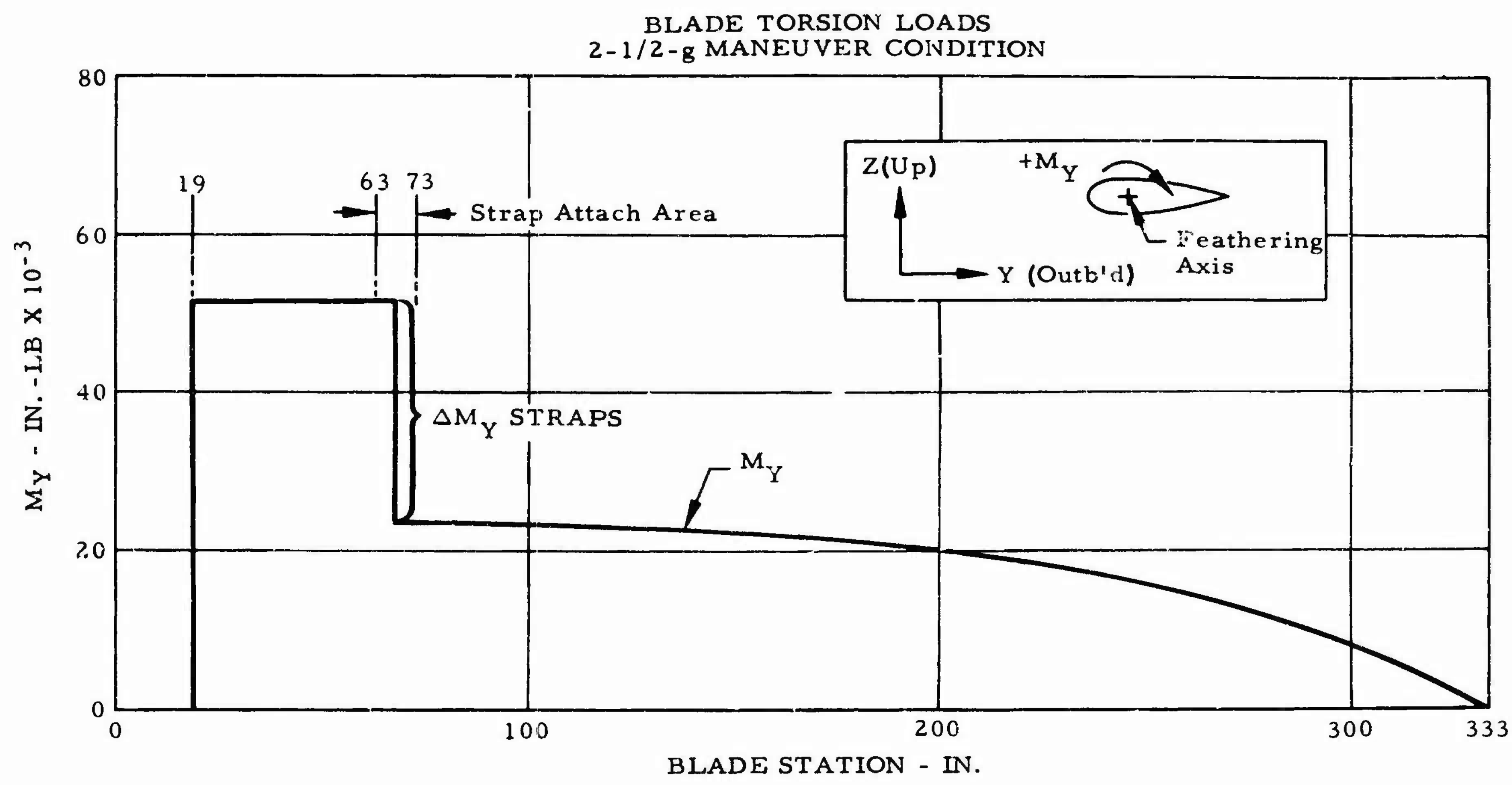


Figure 47. Blade Torsion Loads - 2-1/2-g Maneuver Condition

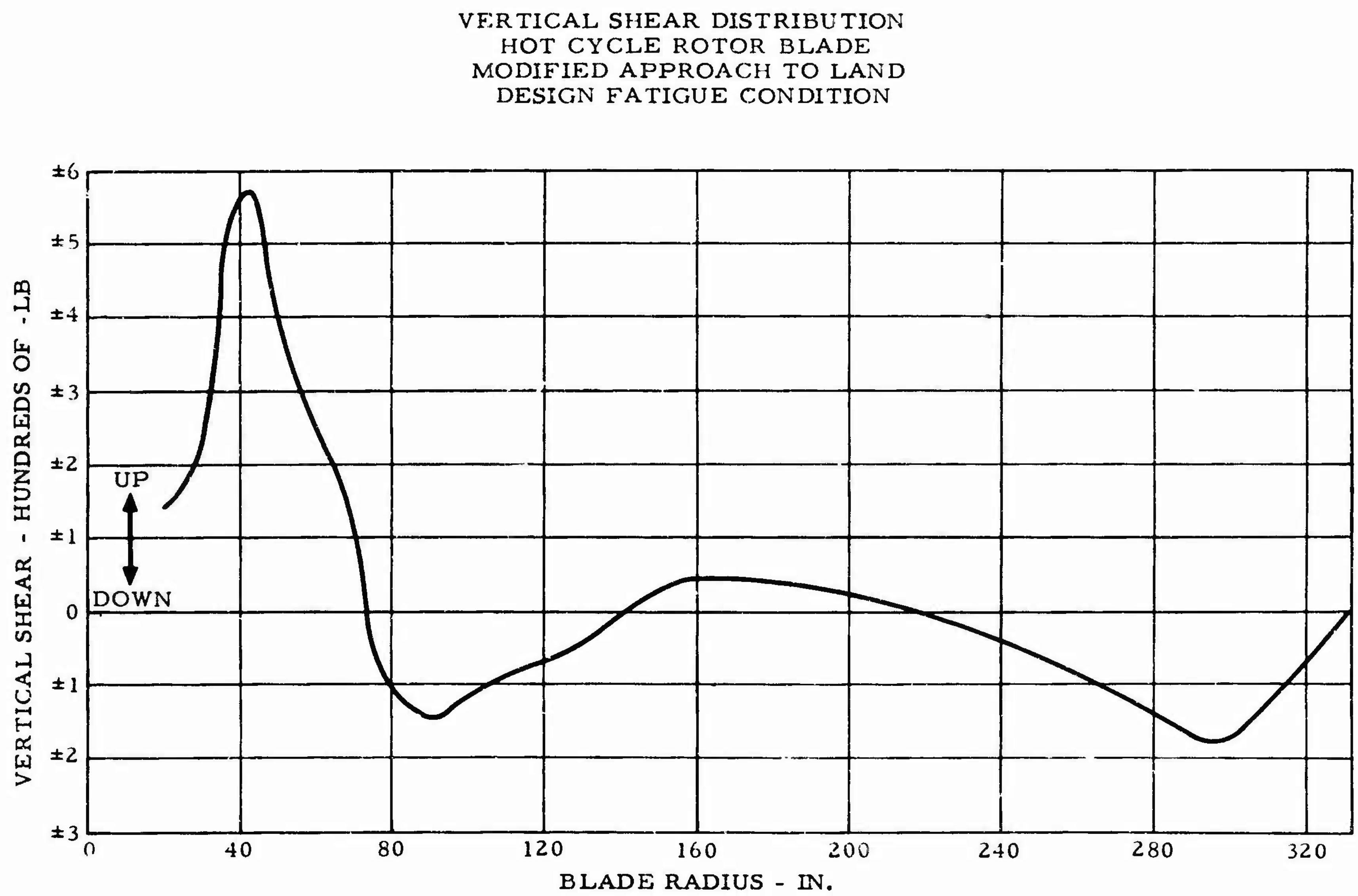


Figure 48. Vertical Shear Distribution - Modified Approach to Land

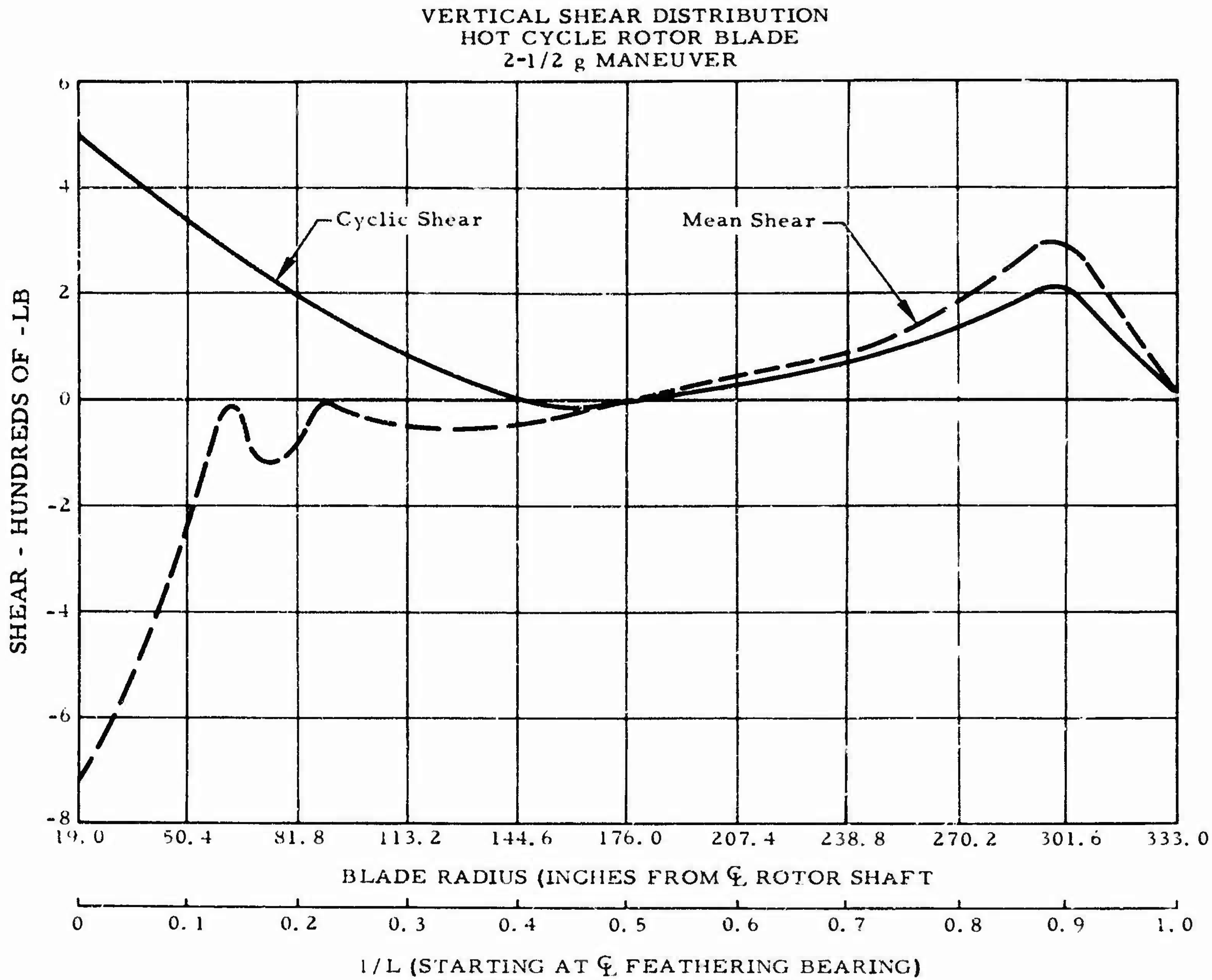


Figure 49. Vertical Shear Distribution - 2-1/2-g Maneuver

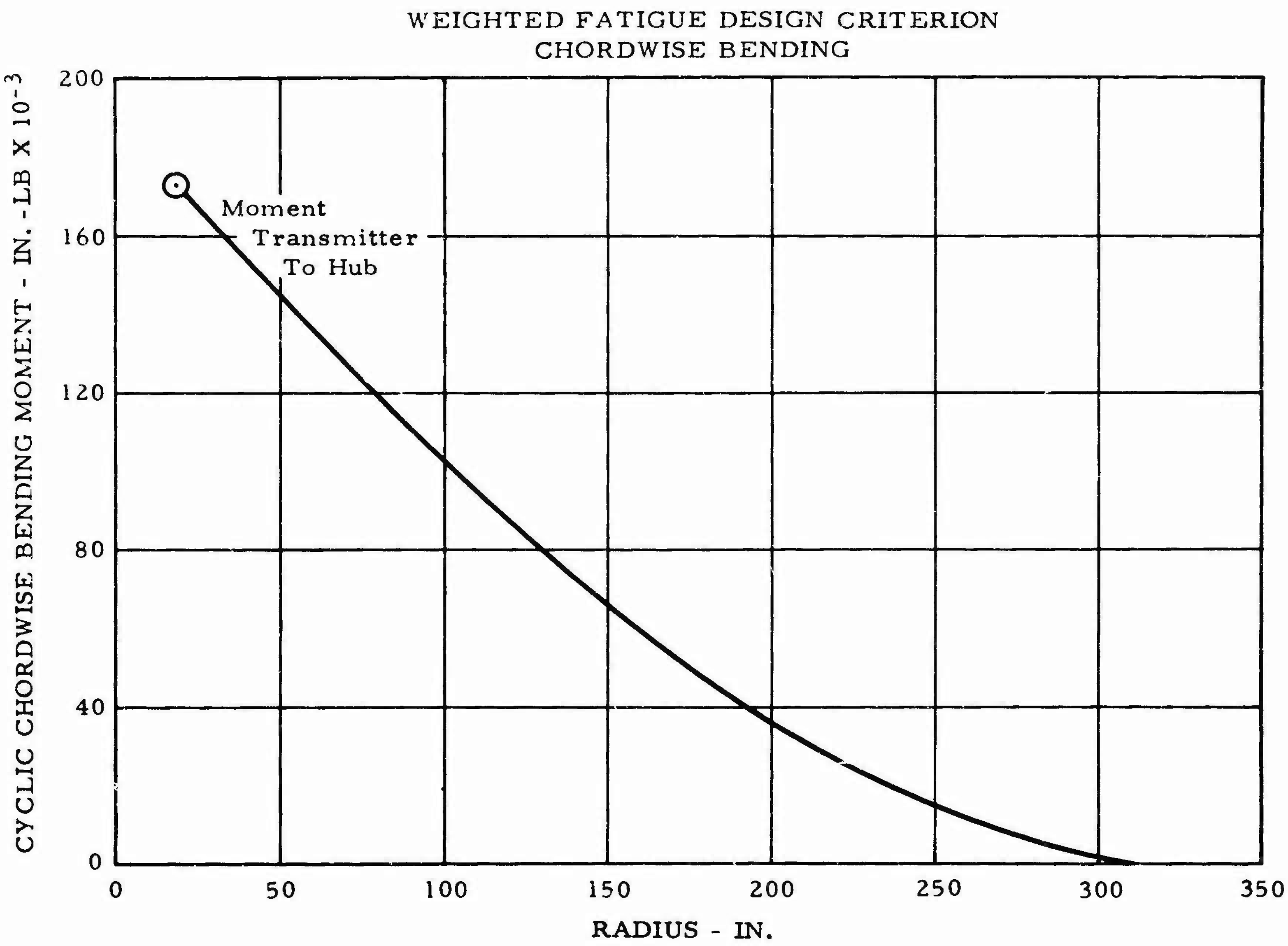


Figure 50. Weighted Fatigue Criterion

(2) 2-1/2-g maneuver = 100 ± 1,550-pound limit condition

Notes:

(1) Positive loads are up and aft on hub

(2) Normal shears do not include control forces

7.1.8 <u>Blade Bending Moments</u>

a. Chordwise Bending Moments
Chordwise moments are given in the chord plane of the blade

(1) Weighted fatigue
Blade spanwise variation of cyclic rotor blade chordwise bending is given in Figure 50

(2) 2-1/2-g maneuver
Blade spanwise variation of cyclic rotor blade chordwise bending is given in Figure 51

(3) Overrev
No significant bending stresses

b. Flapwise bending moments

(1) Weighted fatigue (modified approach to land) (Figure 52)

Note:
Cyclic bending moment shown in Figure 52 should also be used as the steady bending moment for this condition.

(2) 2-1/2-g maneuver (Figure 53)

(3) Overrev - 2.5-g autorotation maneuver at 100 knots; assume the total flapwise blade bending equals the cyclic bending used for the design fatigue limit (Figure 52)

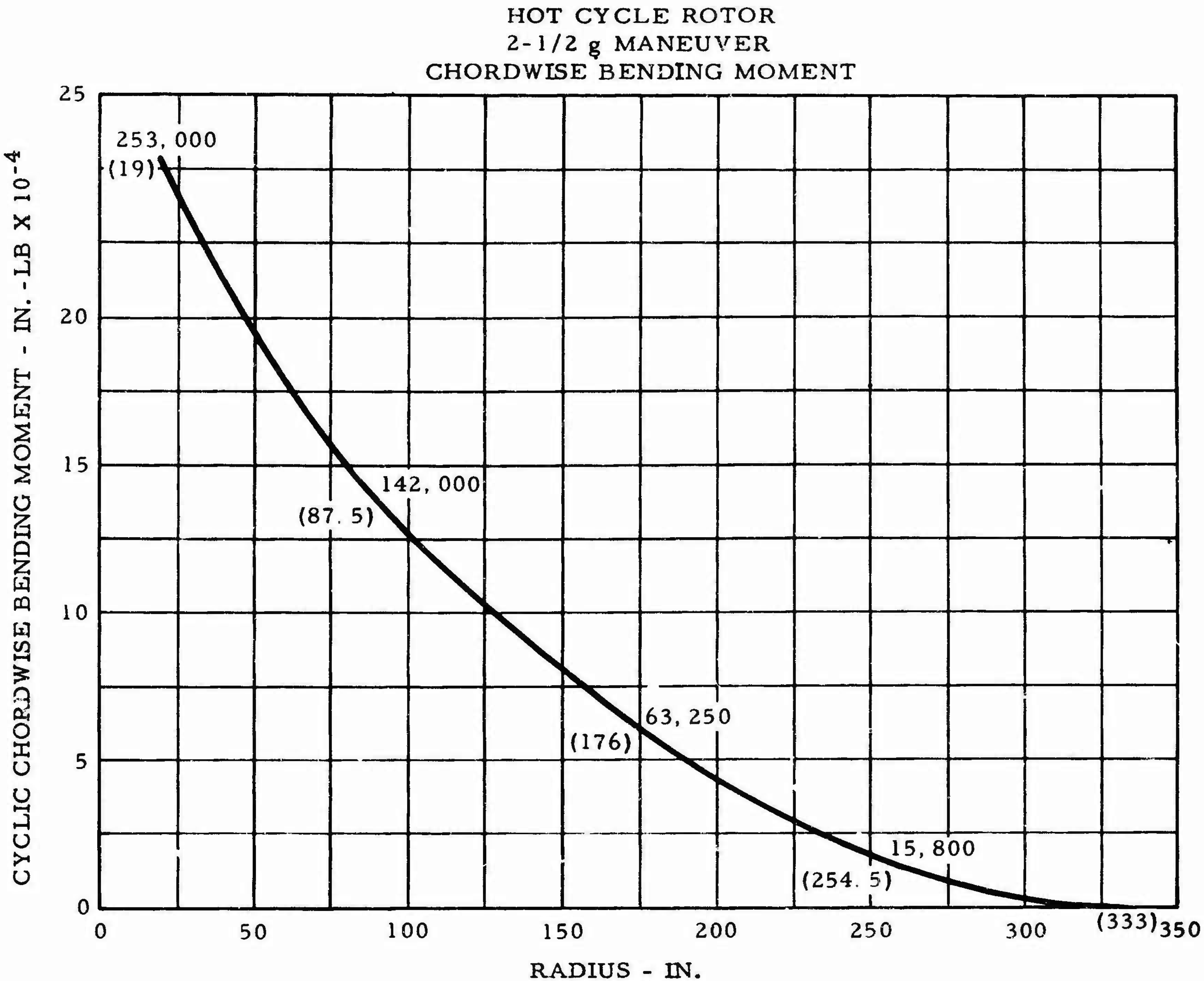


Figure 51. Chordwise Bending Moment

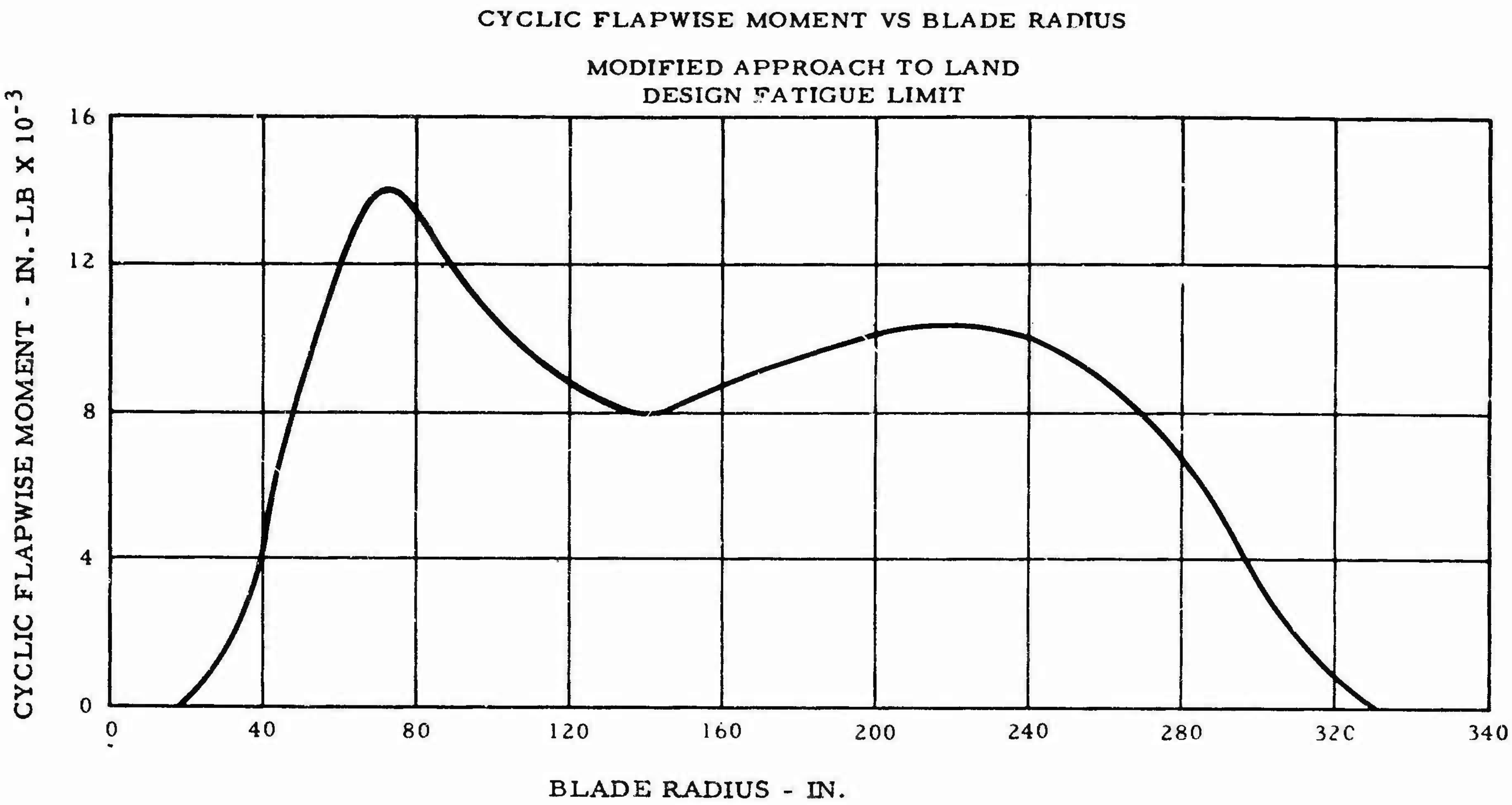


Figure 52. Cyclic Flapwise Moment

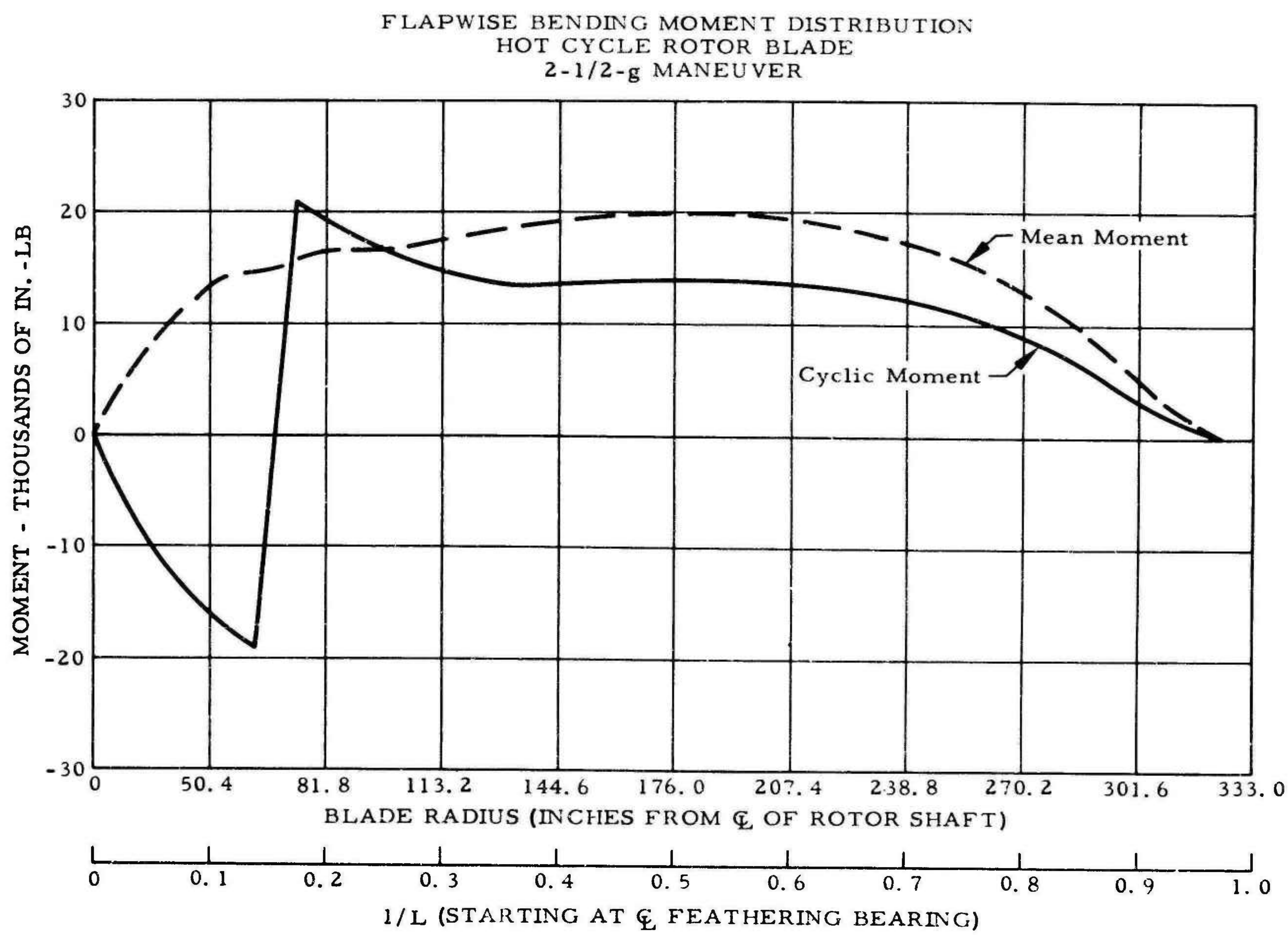


Figure 53. Flapwise Bending Moment

7. 1. 9 Duct Operating Pressure and Temperature

a. 910 hours of life:

Design	1, 117°F	26. 9 psig

b. 90 hours of life:

Design	1, 184°F	29. 0 psig

c. Power off, rotor rotating — 800°F — -4. 0 psig

7. 1. 10 Hub In-Plane Loads

a. Weighted fatigue condition
Use a 1. 0-g thrust with the vector at 6 degrees to the shaft and with the hub inclined 5 degrees to the shaft shaft, or same lateral component with 1. 5-g thrust.

b. 2. 5-g maneuver (ultimate condition)

(1) Fore and aft
Use a 2. 5-g thrust with the vector at 10 degrees to the shaft and with the hub inclined 8 degrees to the shaft.

(2) Left and right
Use a 2. 0-g (2. 5-g x 0. 80) thrust with the vector at 10 degrees to the shaft and with the hub inclined 8 degrees to the shaft.

7. 1. 11 Chordwise Pressure Distribution Over the Rotor Pylon

The design condition is yawed flight at maximum autogyro speed (V = 200 knots). Figure 54 shows the boundaries of the positive and negative pressures on the pylon for yaw angles of 0 to ± 20 degrees. The pressures include an approximate correction for the effect of the nacelle pylon.

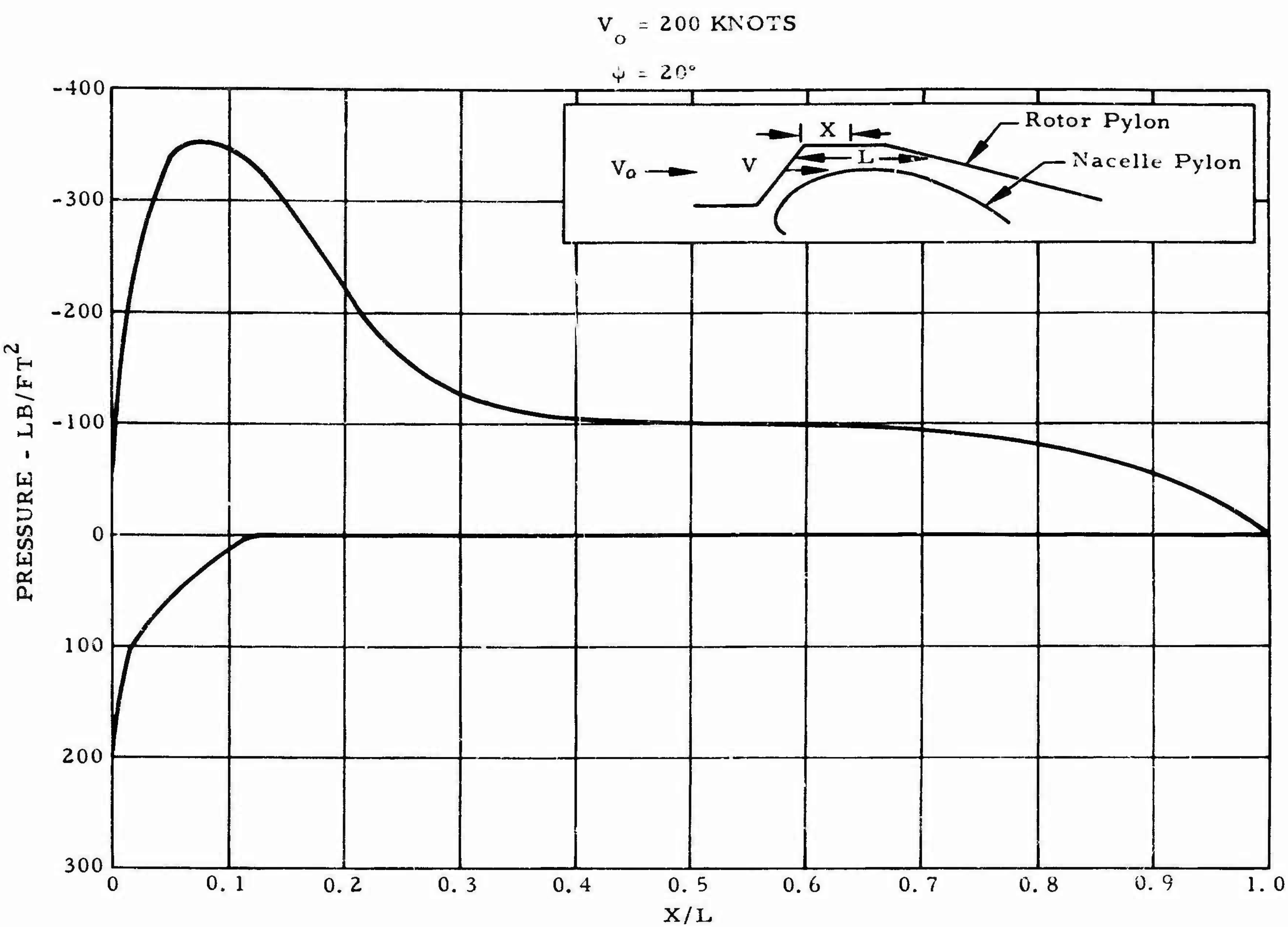


Figure 54. Rotor Pylon Pressure Distribution

7.1.12 Chordwise Pressure Distribution Over the Lateral Pylon and Nacelles

The design condition is maximum autogyro speed (V = 200 knots) during a maneuver with the tail producing a $C_L = 1.0$ (2.5-g load factor). The pressure distributions are based on data presented in Reference 23 and Reference 24. Figure 55 presents total chordwise pressure distribution over the nacelle pylon for an estimated fuselage angle of attack of 23 degrees. Figure 56 presents the estimated chordwise pressure distribution over the upper and lower surfaces of the engine nacelles. For this figure, the data in Reference 24 were extrapolated to nacelle angles of attack of +23 and -9 degrees. The data were also corrected to include the effects of the airflow through the YT-64 gas generators on nacelle leading edge pressures.

7.2 DESIGN CRITERIA FOR ROTOR SYSTEM POWER MODULE AND FUSELAGE

a. Cyclic pitch is defined as $\theta_{1_s} \sin \psi + \theta_{2_s} \cos \psi$,

   where ψ = blade azimuth location measured from the blade aft position, and θ_{1_s} and θ_{2_s} are measured with respect to the neutral swashplate position.

b. Under dynamic transient conditions, hub lag relative to the swashplate may be as much as 2.88 degrees beyond the steady state tilt. It will be restricted to this value by hydraulic flow restriction. (See Note 7, Section 7.1.6).

c. See Figures 57, 58, 59, and 60.

7.2.1 Clearance Condition

Hub tilt - relative to mast:
- At normal rpm, 9° in all azimuth positions
- At zero rpm, 2° in all azimuth positions

Blade coning - relative to hub:
- 15° up, 2° down

Blade collective pitch at 3/4 radius:
- 0° to 12°

Blade cyclic pitch - relative to mast:
- $\theta_{1_s} = \pm 10^\circ, \theta_{2_s} = \pm 7^\circ$

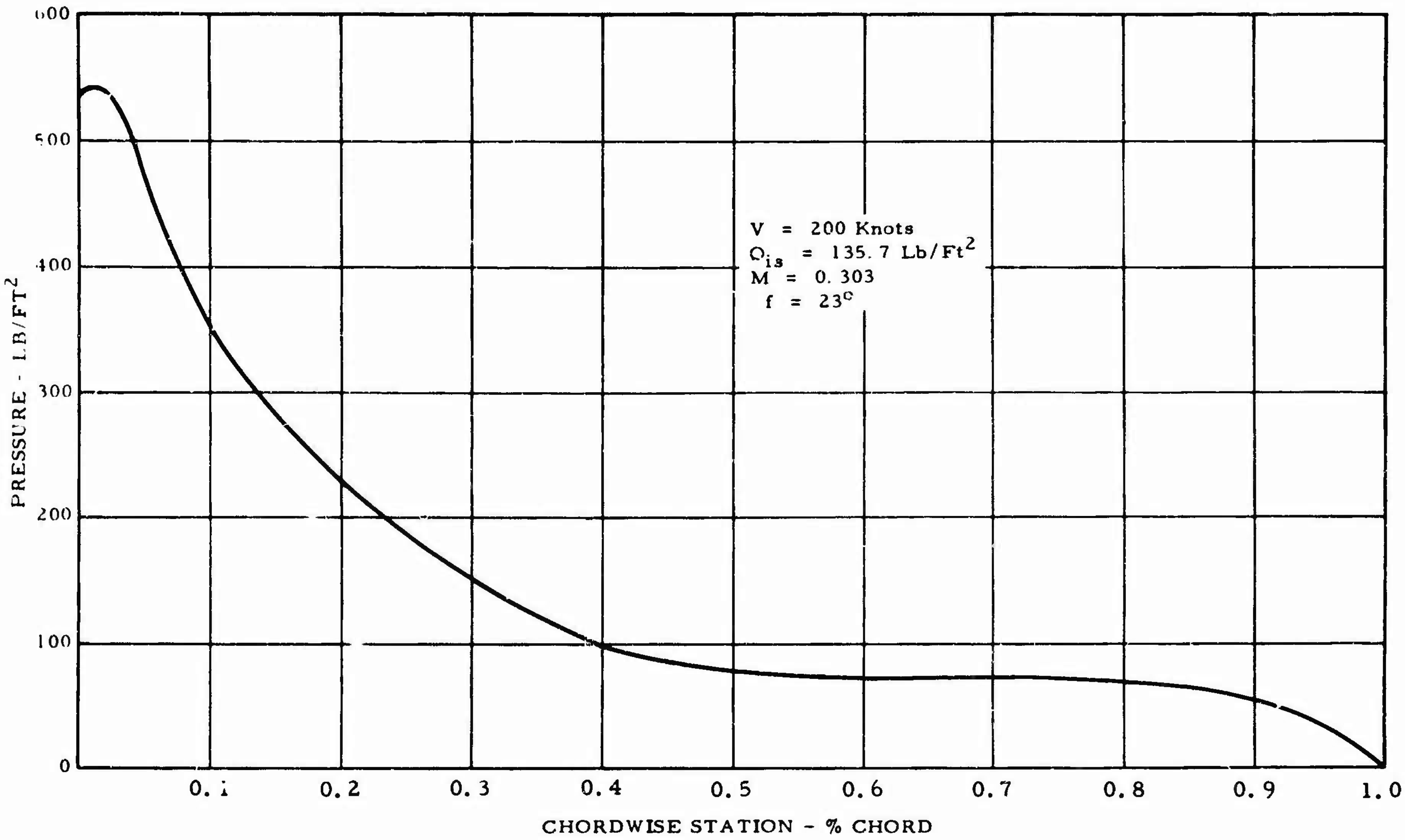


Figure 55. Lateral Pylon Pressure Distribution

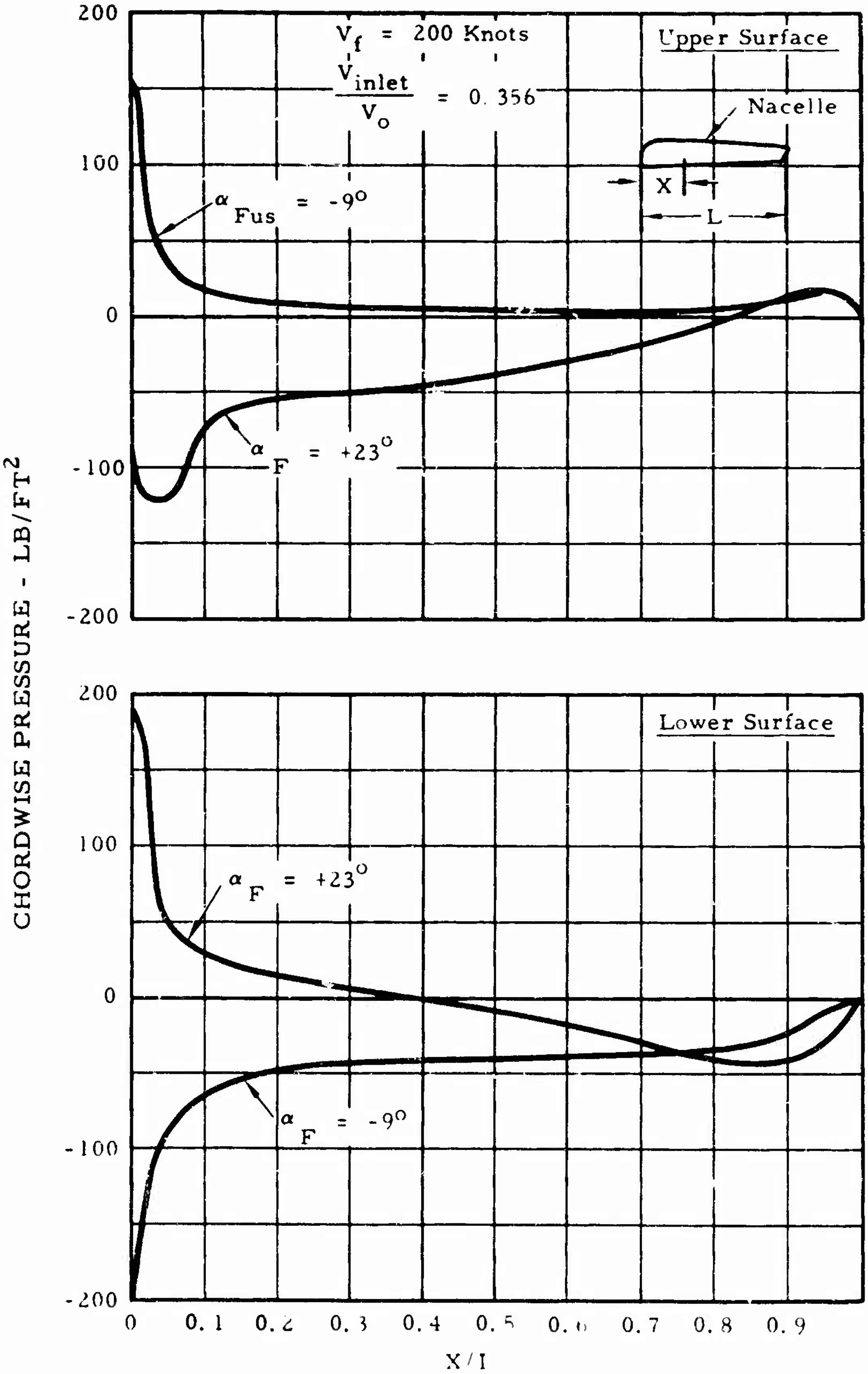
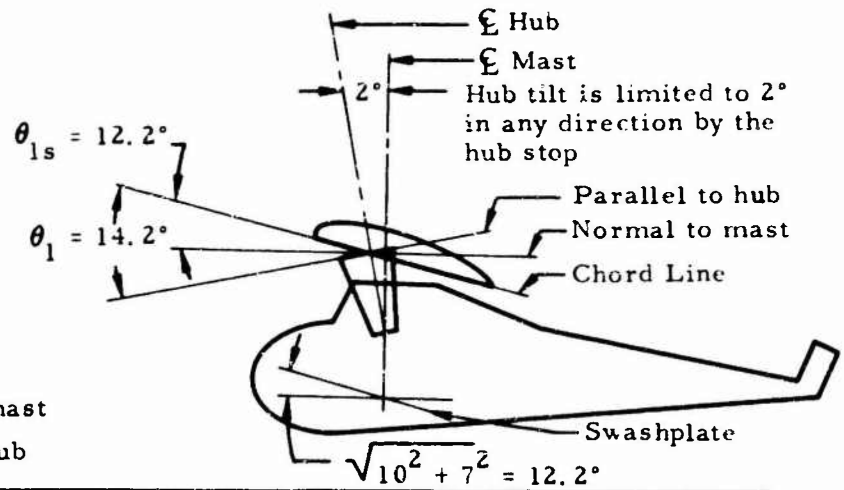


Figure 56. Nacelle Pressure Distribution

HUB AND ROTOR BLADE GROUND CLEARANCE CHECK

View shows an advancing blade at approximately 135° azimuth



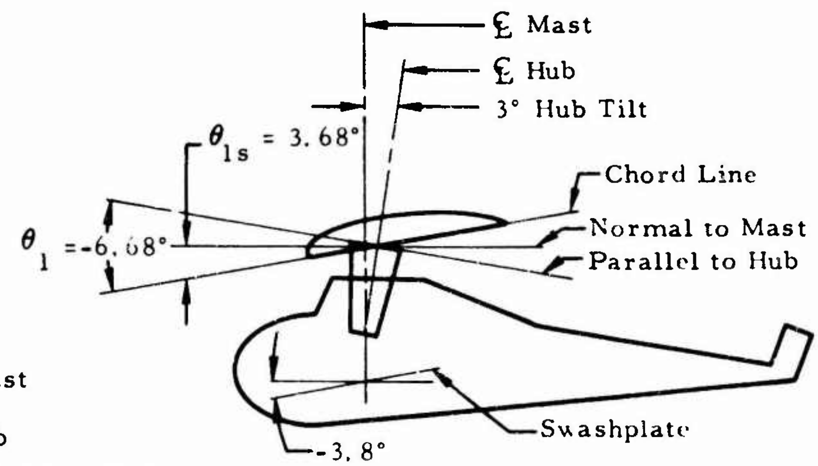
θ_{1s} = Pitch relative to mast

θ_1 = Pitch relative to hub

Collective Pitch	Maximum Strap Windup	
	Advancing Blade	Retreating Blade
12° -7.6° = 4.4° at pitch arm	+14.2° + 4.4° =+18.6° (Blade Nose Up)	-14.2° + 4.4° =-9.8° (Blade Nose Down)
0° -7.6° = -7.6° at pitch arm	+14.2° - 7.6° =+6.6° (Blade Nose Up)	-14.2° - 7.6° =-21.8° (Blade Nose Down)

ENTRY INTO AN AUTOROTATION MANEUVER FROM A CRUISE CONDITION

View shows an advancing blade at 90° azimuth



θ_{1s} = Pitch relative to mast

θ_1 = Pitch relative to hub

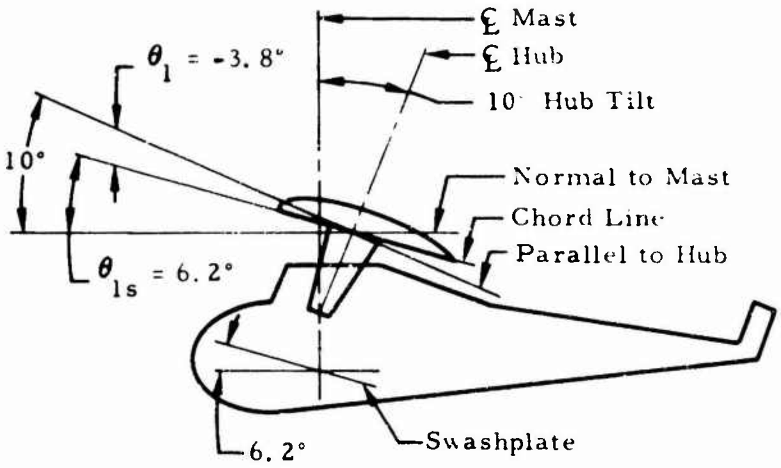
Collective Pitch	Max. Strap Windup	
	Advancing Blade	Retracting Blade
0° -7.6° = -7.6° at pitch arm	-6.68 - 7.6° = -14.28° (Blade Nose Down)	+6.68°-7.6° = -0.92° (Blade Nose Down)

Figure 57. Strap Windup Characteristics

2.5-G MANEUVER CONDITION AT 100 KNOTS

STEP 1 - CYCLIC STICK PULLBACK

View shows an advancing blade at 90° azimuth



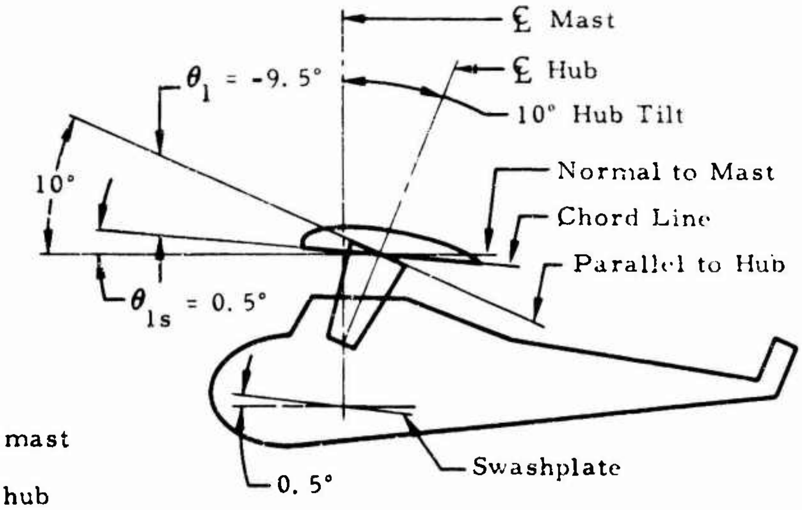
θ_{1s} = Pitch relative to mast

θ_1 = Pitch relative to hub

Collective Pitch	Maximum Strap Windup	
	Advancing Blade	Retreating Blade
7.6° -7.6° =0° at Pitch Arm	-3.8° (Blade Nose Down)	+3.8° (Blade Nose Up)

STEP 2 - APPLICATION OF FULL COLLECTIVE PITCH

View shows an advancing blade at 90° azimuth



θ_{1s} = Pitch relative to mast

θ_1 = Pitch relative to hub

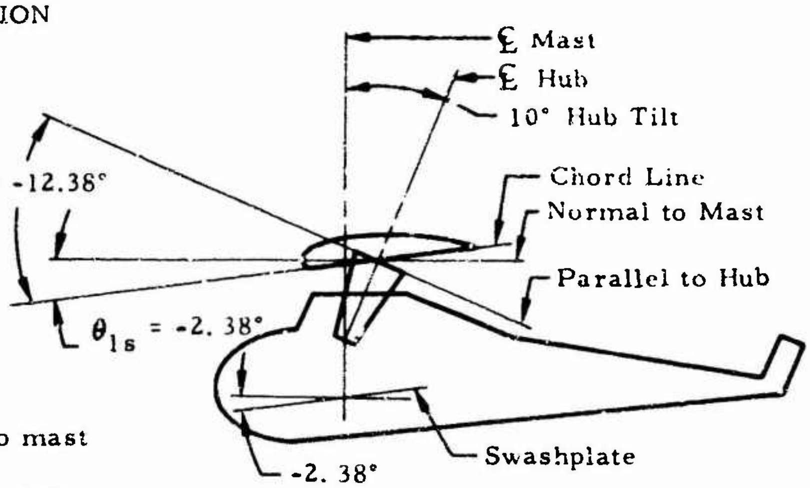
Collective Pitch	Maximum Strap Windup	
	Advancing Blade	Retreating Blade
12° -7.6°=4.4° at pitch arm	-9.5° + 4.4° = 5.1° (Blade Nose Down)	+9.5° + 4.4° =+13.9° (Blade Nose Up)

Figure 58. Strap Windup Characteristics

2. 5-G MANEUVER CONDITION AT 100 KNOTS

STEP 3 - RECOVERY PORTION

View shows an advancing blade at 90° azimuth



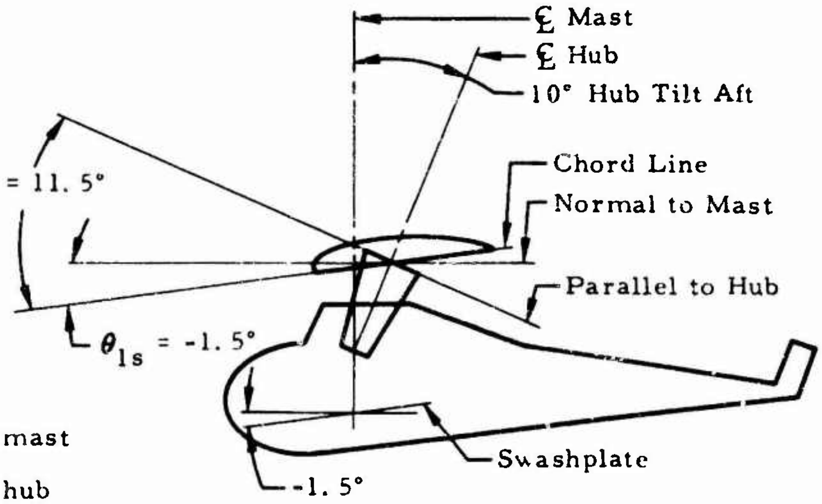
θ_{1s} = Pitch relative to mast

θ_1 = Pitch relative to hub

Collective Pitch	Maximum Strap Windup	
	Advancing Blade	Retreating Blade
12° - 7.6° = 4.4° at pitch arm	-12.38° + 4.4° = -7.98° (Blade Nose Down)	+12.38° + 4.4° = +16.78° (Blade Nose Up)

2. 5-G AUTOROTATION MANEUVER AT 100 KNOTS - FLAREOUT

View shows an advancing blade at 90° azimuth



θ_{1s} = Pitch relative to mast

θ_1 = Pitch relative to hub

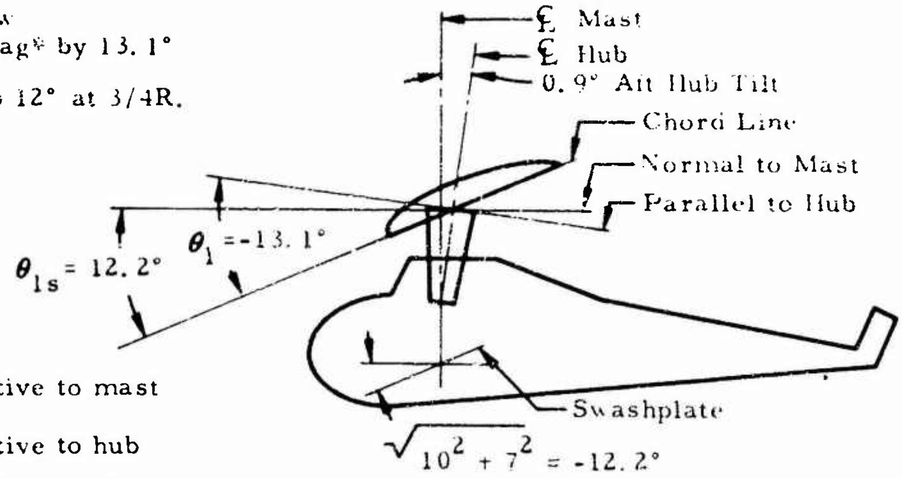
Collective Pitch	Maximum Strap Windup	
	Advancing Blade	Retreating Blade
3° - 7.6° = 4.6° at pitch arm	-11.5° + 4.6° = 16.1° (Blade Nose Down)	+11.5° - 4.6° = +6.9° (Blade Nose Up)

Figure 59. **Strap Windup Characteristics**

IDLING CONDITION - HUB TILT AFT

Hub Tilt Aft; to follow swashplate but may lag* by 13.1°
Cyclic pitch - 12.2°
Collective pitch 0° to 12° at 3/4R.

View shows an advancing blade at approximately 135° azimuth



θ_{1s} = Pitch relative to mast

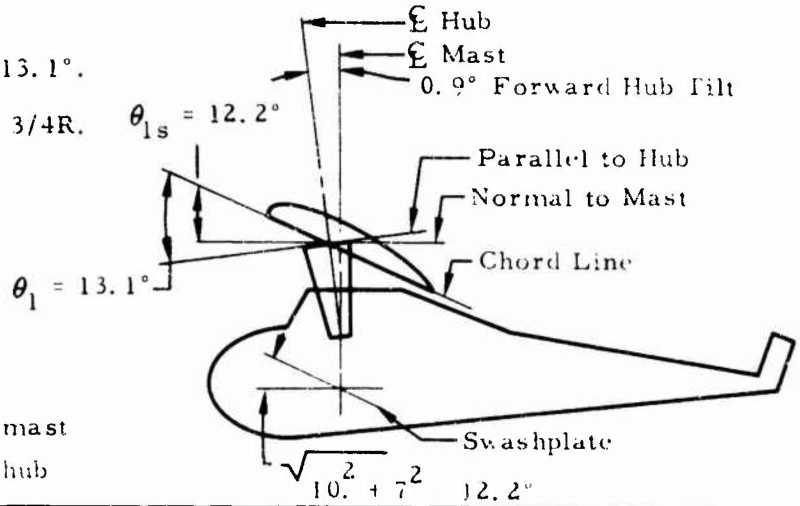
θ_1 = Pitch relative to hub

Collective Pitch	Maximum Strap Windup	
	Advancing Blade	Retreating Blade
0° - 7.6° = -7.6° at pitch arm	-13.1°-7.6° = 20.7° (Blade Nose Down)	+13.1°-7.6° = +5.5° (Blade Nose Up)
12°-7.6° = 4.4° at pitch arm	-13.1°+4.4° = -8.7° (Blade Nose Down)	+13.1°+4.4° = +17.5° (Blade Nose Up)

IDLING CONDITION - HUB TILT FORWARD

Hub Tilt Forward; to follow swashplate but may lag* by 13.1°.
Cyclic pitch +12.2°
Collective pitch 0° to 12° at 3/4R.

View shows an advancing blade at approximately 135° azimuth



θ_{1s} = Pitch relative to mast

θ_1 = Pitch relative to hub

Collective Pitch	Maximum Strap Windup	
	Advancing Blade	Retreating Blade
0° - 7.6° = -7.6° at pitch arm	+13.1° -7.6° = +5.5° (Blade Nose Up)	-13.1° -7.6° -20.7° (Blade Nose Down)
12° -7.6° = +4.4° at pitch arm	+13.1° + 4.4° = 17.5° (Blade Nose Up)	-13.1 + 4.4° = -8.7° (Blade Nose Down)

*Hub Lag = $2.88° \times \frac{\text{Normal RPM}}{\text{Idle RPM}} = 2.88° \times \frac{243}{53.3} = 13.1°$

Figure 60. **Strap Windup Characteristics**

7.2.2 Level Flight, 100-Knot Cruise

Gross weight	15,300 pounds
Rotor rpm	243
Centrifugal force per blade	130,766 pounds
Load factor	1 g
Hub tilt - relative to mast	0° to 3° aft
Blade coning - relative to hub	2.24°
Blade flapping - relative to hub	$\pm 0.25^\circ$ at 2/rev
Blade collective pitch at 3/4 radius	$\pm 7.6^\circ$
Blade cyclic pitch - relative to hub	$\theta_1 = 0^\circ$ to -3.8°, $\theta_2 = 1.7^\circ$
Blade cyclic pitch - relative to mast	$\theta_{1_s} = 0^\circ$ to -0.8°, $\theta_{2_s} = 1.7^\circ$

7.2.3 2.5-g Maneuver Condition at 100 Knots

Gross weight	15,300 pounds
Rotor rpm	243
Centrifugal force per blade	130,766 pounds

Maneuver description (this condition is a dynamic maneuver; therefore, its description is presented in three parts.)

a. Cyclic stick pullback

Helicopter load factor	1.0 g
Hub tilt - relative to mast	10° aft
Blade coning - relative to hub	$+2.24^\circ$
Blade flapping - relative to hub	$\pm 0.25^\circ$ at 2/rev
Blade collective pitch at 3/4 radius	$+7.6^\circ$
Blade cyclic pitch - relative to hub	$\theta_1 = -3.8^\circ$, $\theta_2 = +1.7^\circ$
Blade cyclic pitch relative to mast	$\theta_{1_s} = +6.2^\circ$, $\theta_{2_s} = +1.7^\circ$

b. Application of full collective pitch and decrease in feathering angle

Helicopter load factor	2.5 g
Hub tilt - relative to mast	10° aft

Blade coning - relative to hub	$\pm 5.6^\circ$
Blade flapping - relative to hub	$\pm 0.6^\circ$ at 2/rev
Collective pitch at 3/4 radius	12°
Blade cyclic pitch - relative to hub	$\theta_1 = -9.5^\circ$, $\theta_2 = +4.25^\circ$
Blade cyclic pitch - relative to mast	$\theta_{1_s} = +0.5^\circ$, $\theta_{2_s} = +4.25^\circ$

c. Recovery (cyclic pitch stick moved an additional 2.88 degrees forward; see Item b, Section 7.2).

Helicopter load factor	2.5 g
Hub tilt - relative to mast	10° aft
Blade coning - relative to hub	$+5.6^\circ$
Blade flapping - relative to hub	$\pm 0.6^\circ$ at 2/rev
Blade collective pitch at 3/4 radius	$+12^\circ$
Blade cyclic pitch - relative to hub	$\theta_1 = -12.38^\circ$, $\theta_2 = +4.25^\circ$
Blade cyclic pitch - relative to mast	$\theta_{1_s} = -2.38^\circ$, $\theta_{2_s} = +4.25^\circ$

7.2.4 Weighted Fatigue Condition

Gross weight	15,300 pounds
Rotor rpm	243
Centrifugal force per blade	130,766 pounds
Load factor (hub and shaft only)	1.5 g
Hub tilt - relative to mast	0° to 6° aft
Blade coning - relative to hub	$+2.24$ to $+4.48^\circ$ (whichever is critical)
Blade flapping - relative to hub	$\pm 0.5^\circ$ at 2/rev
Blade collective pitch at 3/4 radius	$+7.6^\circ$
Blade cyclic pitch - relative to hub	$\theta_1 = 7.6^\circ$, $\theta_2 = +3.4^\circ$

Blade cyclic pitch - relative to mast	$\theta_{1_s} = 1.6^\circ$, $\theta_{2_s} = +3.4^\circ$

7.2.5 Entry Into Autorotation From Cruise

Gross weight	15,300 pounds
Rotor rpm	295
Centrifugal force per blade	192,720 pounds
Load factor	1 g
Hub tilt - relative to mast	3° aft
Blade coning - relative to hub	$+1.52^\circ$
Blade flapping - relative to hub	$\pm 0.25^\circ$ at 2/rev
Blade collective pitch at 3/4 radius	0°
Blade cyclic pitch - relative to hub	$\theta_1 = 6.68^\circ$, $\theta_2 = +1.7^\circ$
Blade cyclic pitch - relative to mast	$\theta_{1_s} = 3.68^\circ$, $\theta_{2_s} = +1.7^\circ$

7.2.6 2.5-g Autorotation Maneuver at 100 Knots (Flareout)

Gross weight	15,300 pounds
Rotor rpm	295
Centrifugal force per blade	192,720 pounds
Helicopter load factor	2.5 g
Hub tilt - relative to mast	10° aft
Blade coning - relative to hub	$+3.8^\circ$
Blade flapping - relative to hub	$\pm 0.6^\circ$ at 2/rev
Blade collective pitch at 3/4 radius	$+3^\circ$
Blade cyclic pitch - relative to hub	$\theta_1 = -11.5^\circ$, $\theta_2 = 0^\circ$
Blade cyclic pitch - relative to mast	$\theta_{1_s} = -1.5^\circ$, $\theta_{2_s} = 0^\circ$

7.3 DESIGN CRITERIA FOR THE EMPENNAGE AND AFT FUSELAGE

Section 7.3.4 summarizes the limit loads that are to be considered in the design and stress analysis of the empennage and aft fuselage. These critical loads are derived as follows; where:

GW = gross weight

C_L = lift coefficient

V = velocity

q = dynamic pressure

η_t = tail efficiency factor $\frac{qt}{q}$

ℓ_t = distance to tail from cg

τ = dihedral angle

L = lift

L_v = vertical lift

L_h = horizontal lift

M = moment

S_t = total tail area

$\ddot{\alpha}$ = angular acceleration

I = moment of inertia

ψ = yaw angle

α = tail angle

ρ = density

C_d = drag coefficient

7.3.1 Maximum Autogyro Level Flight - Symmetrical Loading

GW = 10,000 pounds

$C_{L_{tail}}$ = 1.0 (per FAA)

V = 200 knots

q = 135.7 pound/feet^2

η_t = 0.90

ℓ_t = 28.5 feet

τ = 45°

Maximum lift perpendicular to each half of the V-tail

$$L_{max} = (1.0)(135.7)(0.9)(27) = 3,300 \text{ pounds}$$

Total tail load in the vertical plane

$$L_{V_{total}} = 2(3,300)(\cos \tau) = \quad 670 \text{ pounds}$$

Pitch acceleration

$$C_{L_{trim}} \cong 0.20$$

$$M = (C_{L_{max}} - C_{L_{trim}}) \ell_t \, q \, \eta_t \, S_t \cos \tau$$

$$= (1.0 - 0.20)(28.5)(135.7)(0.9)(54)(0.707)$$

$$M = 106,200 \text{ foot-pounds}$$

$$\ddot{\alpha} = \frac{M}{I} = \frac{106,200}{20,048} = 5.30 \text{ radians/sec}^2$$

7.3.2 Maximum Autogyro Level Flight - Asymmetrical Loading

Maximum lift perpendicular to each half of the V-tail

$$L_{max} = (3,300)(K) \quad K = 0.712 \text{ (correction factor for asymmetrical loading of a V-tail, Reference 23)}$$

$$L_{max} = (3,300)(0.712) = 2,350 \text{ pounds}$$

Total tail load in the horizontal plane

$$L_{H_{total}} = 2 \; (2,350 \sin \tau) \quad = 3,320 \text{ pounds}$$

Yaw acceleration

$$\Delta M = (2,862)(28.5) = 94,600 \text{ foot-pounds}$$

$$\ddot{\psi} = \frac{\Delta M}{I} = \frac{94,600}{18,545} = 5.11 \text{ rad/sec}^2$$

7.3.3 Maximum Chordwise Load

For an NACA 0012 section for $C_L = 1.0$, the tail drag coefficient is calculated to be $Cd_{tail} = 0.088$.

$$\alpha_{tail} = \frac{C_{L_{tail}}}{\frac{(dC_L)}{d\alpha_{tail}}} = \frac{1.0}{0.061} = 16.4 \text{ degrees}$$

With α_{tail}, Cd_{tail}, and $C_{L_{max}}$ known, the chordwise force coefficient is calculated to be $C_f = 0.197$.

$$\text{Chordwise load} = C_f\, q\, \eta_t \frac{S_t}{2} \text{ per side}$$

$$= (0.197) \left[\frac{\rho}{2} (337.8 \times \cos \alpha_t)^2\right] 0.9\ (27)$$

$$= 595 \text{ pounds}$$

The chordwise load is acting forward in the chord plane and assumed acting at the tail midspan.

7.3.4 Empennage and Aft Fuselage Limit Load

Condition: maximum autogyro level flight for $C_{L_{tail}} = 1.0$

	Load	Inertia Relief
Vertical (total)	4,670 pounds	5.30 radians/sec^2
Horizontal (total)	3,320 pounds	5.11 radians/sec^2
Chordwise/side	595 pounds (acting forward in chord plane)	0

7.4 LANDING CRITERIA

The XV-9A utilizes a CH-34A main landing gear and a full swiveling CH-34A tailwheel.

Per Reference 3, paragraph 3.4.2.1, the ultimate ground landing load factor for the XV-9A shall be 3.5 at the basic design gross weight of 15,300 pounds, or as limited by the CH-34A landing gear. The landing gear was originally designed for 8-foot-per-second vertical contact velocity and 11,400-pound landing weight, per Reference 25. For the CH-34A, the ultimate ground landing load factor is calculated to be 3.5. Thus, the structural limitation of the CH-34A gear actually determines the ultimate ground landing load factor for the XV-9A.

However, the aircraft has been designed with an ultimate ground landing load factor of 3.5 and a limit ground landing load factor of 3.5/1.5 = 2.33. This was done in order to provide a higher aircraft margin of safety and to permit possible future use of a higher strength landing gear.

The following table presents a summary of the ultimate load factors and accelerations about the aircraft cg for the conditions investigated in the loads analysis (see Appendix I).

TABLE 4
LANDING CRITERIA
SUMMARY OF ULTIMATE INERTIA FACTORS

Landing Condition	Gross Weight (Pounds)	Rotor Lift Load Factor at CG	Helicopter Total Load Factor at CG: Vertical	Drag	Side	Acceleration (Rad/Sec²): Pitch	Yaw	Roll
1. 3-point level landing	15, 300	0. 667	4. 17	0	0	0	0	0
2. 2-point level landing on main gear (no drag or side load)	15, 300	0. 667	3. 90	0	0	6. 525	0	0
3. 2-point level landing on main gear (with drag load on one wheel)	15, 300	0. 667	4. 04	0. 76	0	4. 220	2. 88	0
4. 2-point level landing on main gear (with side load on one wheel)	15, 300	0. 667	4. 04	0	0. 76	6. 845	1. 80	11. 448
5. 1-wheel banked landing (with drag load)	15, 300	0. 667	1. 99	0. 493	0	0. 485	2. 238	13. 980
6. Tail first landing	15, 300	0. 667	0. 869	0	0	-4. 00	0	0
7. Braking condition	15, 300	--	1. 80	1. 44	0	-2. 218	0	0

7.5 GROUND HANDLING DESIGN CRITERIA

The ground handling limit load factors for the basic design gross weight of 15,300 pounds are as follows.

7.5.1 Hoisting. To remove the rotor or hoist the helicopter, attachment points are provided on the vertical faces of the hub. A vertical limit load factor of 2.0 g is assumed with the helicopter at the design gross weight of 15,300 pounds. Horizontal loads are assumed to be zero.

7.5.2 Mooring. Mooring fittings and the structure to which they are attached shall be designed for limit loads resulting from a 40-knot wind from any azimuth position.

7.5.3 Jacking. Jacking loads for the primary flight structure jacking points, for level fuselage attitude, for the design gross weight of 15,300 pounds shall be:

2.0 g	vertical
0.5 g	fore or aft
0.5 g	lateral

7.5.4 Towing. A tow bar will be provided for towing the helicopter by the tail landing gear. Towing loads shall be per Reference 26, Chapter 4, at a design gross weight of 15,300 pounds.

7.6 CRASH CONDITION

For crash conditions, the following ultimate load factors shall be the design objective in the design of the seat installation and attachment of equipment and useful load items (and their carry-through structure) that might injure the crew if they became loose in a minor crash landing. The load factors are applied independently, at the design gross weight of 15,300 pounds:

Downward	10 g
Forward	10 g
Sideward	4 g
Upward	2 g

7.7 PRIMARY CONTROL SYSTEM LOADS

7.7.1 Lower Rotor Controls and Flight Controls

7.7.1.1 Pilot Loads. Since the lower flight control system from the pilot to the servo valves utilizes the OH-6A flight control system, the control loads criteria of the OH-6A shall be applicable. These control loads are as follows.

From the pilot compartment to the stops that limit the range of motion of the pilot's controls, the control system shall be designed to withstand the limit applied forces shown in column 1 of the following table. Dual control loads shall be 75 percent of the values of column 1 applied at each pilot station, either in conjunction or in opposition.

TABLE 5
PILOT CONTROL SYSTEM LOADS

	Limit Pilot Forces	
Controls	(1) To Stops (Pounds)	(2) Beyond Stops (Pounds)
Collective pitch control	100	60
Longitudinal cyclic	100	60
Lateral cyclic	67	40
Yaw controls	130	78

7.7.1.2 Nonrotating Power Linkage. For the nonrotating portion of the rotor control system, apply a cyclic load of ± 675 pounds to the pitch arm of one blade with the rotor stopped. Check stresses at the most critical rotor azimuth position. Design for infinite life.

The cyclic fatigue load is to be applied in addition to the normal steady load used in the design of the system. The limit loads for static strength should be based upon blade loads as given in Section 7.1.6.

7.7.2 Rotating Controls

7.7.2.1 Limit Loads. The limit loads shall be the torsion loads imposed by the rotor blades shown in Section 7.1.6.

7.7.2.2 Fatigue Loads. See Section 7.1.6.

7.8 PROPULSION SYSTEM

7.8.1 Ducting

Limit pressures and temperatures are taken from the T64 gas generator data of Reference 27 and additional information from the engine manufacturer concerning growth versions of the T64.

A 1.33 limit factor is applied to advanced engine pressures at maximum continuous power of the reference reports.

Limit pressure = 1.33 x 27.0 = 36.0 psig
Limit temperature = 1184°F
For crash loads, limit pressure shall be 29.0 psig
Target service life of hot parts under operating conditions is 1,000 hours

7.8.2 Engine Mounts and Nacelles

7.8.2.1 Limit Torque

Gas generator starting torque = 60 foot-pounds
Transient torque load factor on engine mount = 2
Limit torque = 2 x 60 = 120 foot-pounds

7.8.2.2 Gyroscopic Moment (Relative to Engine Axis)

Designing to the maximum gyroscopic moment of engine criterion of steady angular velocity of 2.5 radians/sec in yaw and 2.0 radians/sec in pitch, at maximum rated engine speed, the gyroscopic moment is calculated as follows:

Gyroscopic moment = $I_p \Omega \omega$
Ip gas generator = 22.8 pound-foot^2
Maximum allowable transient overspeed limit rpm = 18,330 rpm

For yaw angular velocity of 2.5-rad/sec:

$$\text{Gyroscopic moment} = \frac{22.8}{32.2} (18,330) \frac{\pi}{30} (2.5)$$
$$= 3,396 \text{ foot-pounds}$$

For pitch angular velocity of 2.0 rad/sec:

$$\text{Gyroscopic moment} = \frac{22.8}{32.2}\ (18,330)\ \frac{\pi}{30}\ (2.0)$$

$$= 2,720 \text{ foot-pounds}$$

7.8.2.3 Crash Loads. See Section 7.6

7.9 SUMMARY - MINIMUM MARGINS OF SAFETY

Part Title	Part Number 385-	Type of Loading	Margin of Safety Static	Margin of Safety Fatigue
ROTOR GROUP				
Front spar	1108	Cyclic bend	--	+0.20
Rear spar	1108	Cyclic bend	--	+0.19
Rear spar fitting	1115	Bending	+0.21	--
Strap assembly	0121	Bolt shear	+0.09	--
Tip cascade valve rod	1112	Tension	+0.02	--
Tip cascade rod and cranks	1124	Tension	+0.02	--
Lower hub plates, Drawings 285-0564 and 285-0565		Bending	+0.02	--
Hub feathering bearing ring, Drawing 285-0532		Bolt shear	+0.03	--
Main rotor shaft, Drawing 285-0517 EO 3		Bending	--	+1.43
POWER MODULE				
Fitting assembly - rotor thrust bearing support	5033-5039	Bolt shear	+0.46	--
Upper rotor bearing support	5025	Column	+0.11	--
Lower rotor bearing support	5018	Tension	+0.24	--
Power module front spar	5007	Tension	0.00	--
Skin assembly - lateral pylon	5015	Shear	+0.26	--
Mounting diverter valve	5014	Shear	+0.15	--
Fitting assembly canted rib	5028	Tension	+0.45	--
Canted rib	5014	Compression	+0.01	--
Fitting installation main gear strut	5029	Shear	+0.03	--
Fitting assembly front spar lower cap	5008	Bearing	+0.47	--

Part Title	Part Number 385-	Type of Loading	Margin of Safety Static	Margin of Safety Fatigue
Fitting assembly rear spar lower cap	5010	Bearing	+0. 37	--
Fitting assembly front spar upper cap	5009	Bearing	+0. 03	--
Fitting assembly rear spar upper cap	5011	Bearing	+0. 19	--
Fitting front spar lower center	5020	Bearing	+0. 11	--
Fitting rear spar lower center rib	5026	Bearing	+0. 83	--
Installation BL 22	5013	Compression	0. 00	--
Engine support truss assembly	5005	Tension	+0. 04	--
Forward engine mount clamp	7313	Bending	+0. 13	--
Aft engine mount brace	7306	Bending	+0. 07	--
Frame assembly - Nacelle Station 245. 83	5006	Bending	+0. 12	--
HOT GAS TRANSFER SYSTEM				
Duct assembly, lower stationary	1603	Tension	+0. 01	--
Duct assembly, upper rotating	1607	Bending	+0. 25	--
Assembly, engine exhaust tail pipe	4202	Tension	+0. 06	--
Transition duct assembly, hot gas system	4112	Tension	+0. 17	--
Duct assembly, yaw control supply S section	4323	Bending	+0. 66	--
Duct assembly, yaw control supply	4322	Tension	+0. 32	--
FUSELAGE				
Fuselage longeron	2001	Tension	+0. 44	--
Cutout edge member (bottom skin)	2200	Compression	+0. 10	--
Fuselage skin	2200	Shear	+0. 96	--
Cutout (side skin) rivets	2200	Shear	+0. 10	--
Aft fuselage skin	2300	Shear	+0. 33	--
Main frame assembly, Stations 279. 8 and 317. 5	2201	Bending	+0. 20	--

Part Title	Part Number 385-	Type of Loading	Margin of Safety Static	Fatigue
Side beam fuselage	2200	Shear	+0. 25	--
Fuselage power module attachment fitting	2202-2207	Shear	+0. 46	--
Landing gear support installation, main gear	2209-2208	Bending	+0. 21	--
Bulkhead fuselage, Station 616. 50	2304	Flange crippling	0. 00	--
TAIL ASSEMBLY				
Stabilizer	3100	Compression	0. 00	--
Tail plane center section	3006	Bearing	0. 00	--
Tail plane attachment fitting forward, lower fitting	3003	Bearing	+0. 12	--
Tail plane attachment fitting forward, upper fitting	3002	Tension	+0. 19	--
Tail plane attachment fitting aft, lower fitting	3005	Tension	+0. 64	--
Tail plane attachment fitting aft, upper fitting	3004	Bearing	+0. 69	--
Rudder assembly	3200	Compression	+0. 03	--
Fitting - torque tube	3202	Bending	+0. 20	
Fitting - rudder hinge	3201	Shear	+0. 05	--
Rudder mass balance	3203	--	High	--
Tail plane attachment fitting	2305	Bending	0. 00	--
Angle of incidence adjustment rod	2003	Compression	+0. 04	
CONTROLS				
Stationary swashplate	0313	Interaction	+0. 01	--
Spindle	1008	Bending	+0. 09	--
Bellcrank assembly	6132	Bending	+0. 96	--
Lateral follower bellcrank	6125	Bending	High	--
Collective drive	6121-6128	Shear	+0. 02	--
Assembly - flange support mixer control	6129	Shear	+0. 08	--
Collective driver bellcrank	6121	Bearing	+0. 25	--
Collective and follower	6127	Bearing	+0. 25	--
Lateral driver lever	6122	Bending	High	--

Part Title	Part Number 385-	Type of Loading	Margin of Safety Static	Margin of Safety Fatigue
Longitudinal driver lever	6123	Bending	High	--
Longitudinal follower bellcrank	6126	Bending	High	--
Interconnect tube	6128	Bending	+0.15	--
Rod assembly	6105	Compression	+0.26	--
Longitudinal lever assembly	6116	Bending	High	--
Bellcrank assembly	6115	Shear	High	--
Bellcrank assembly	6117	Bending	High	--
Module ℄ bracket	6194	Bending	0.00	--
Actuator support bracket	6191	Shear	+0.18	--
Lateral control bellcrank	6111	Bending	+0.52	--
Bellcrank support bracket assembly	6119	--	+0.77	--
Rudder control lever assembly	6171	Bending	+0.21	--
Rudder control tube assembly	6173	Column	0.00	--
Rudder operating bellcrank	6174	Bending	+0.16	--
Rudder control drive plate	6151	Bending	High	--
Stationary swashplate drag link	6106	--	High	--

Part Title	Part Number *369-	Type of Loading	Margin of Safety Static	Margin of Safety Fatigue
Torque tube assembly	7109	Bending	0.01	--
Torque tube fitting (copilot)	7806	Bending	0.47	--
Support housing (lh)	7105	Bending	0.60	--
Support housing (rh)	7139	Bending	0.77	--
Retainer, support bracket	7118	Bending	0.53	--
Cyclic pitch stick socket	7141	Bending	0.31	--
Tube, cyclic pitch stick	7142	Bending	0.20	--
Bellcrank assembly (lateral cyclic)	7101	Shear	1.68	--
Bellcrank assembly (longitudinal cyclic)	7201	Column	0.13	--
Bellcrank assembly (lateral cyclic)	7202	Shear	0.10	--
Rod assembly-lateral control	7102	Column	1.36	--
Rod assembly-lateral control (dual)	7805	Column	2.90	--

*369 numbers are control system components from the OH-6A (Model 369) helicopter that are used without change on the XV-9A.

Part Title	Part Number 369-	Type of Loading	Margin of Safety Static	Margin of Safety Fatigue
Tube assembly-collective pitch (rh)	7342	Rivet Shear	0. 26	--
Stick assembly-collective pitch (dual)	7807			--
Housing-pilot's collective stick	7347	Shear	0. 10	--
Housing-copilot's stick	7820	Shear	0. 02	--
Fitting-collective control gear	7327	Shear	0. 84	--
Tube-collective torque	7326	Torsion	0. 08	--
Bellcrank assembly-collective pitch mixer	7602	Bending	0. 00	--
Fitting-collective pitch stick	7354	Shear	High+	--
Strut-collective pitch stick	7355	Shear	High+	--
Plate-collective stop support	7358	Column	0. 03	--
Guide-collective pitch stick	7303	All	High+	--
Pedal arm-tail rotor control	7501	Bending and torsion	0. 52	--
Pedal-tail rotor control	7502	Bending	0. 83	--
Tube-rudder pedal support	7503	Bending and torsion	0. 74	--
Bracket assembly - rudder pedal support	7505	Shear	0. 64	--
Bracket assembly - rudder torque tube	7512	Shear	0. 40	--
Link assembly-pedal bus	7506	Column	0. 87	--
Bellcrank assembly-pedal bus	7507	Shear	0. 07	--
Stop-T/R control system	7513	Bending	0. 37	--

APPENDIX I
LOADS ANALYSIS

In calculating the basic loads for the XV-9A the Weight and Inertial Properties Status Report Number 6 was utilized. This report is included for reference.

The loads given in this analysis are limit values. Landing conditions are numbered as Cases 1 through 7. The crash condition is defined as Case 8. The flight conditions are numbered as Cases 10 through 13.

XV-9A Weight and Inertial Properties

Based on Status Report Number 6 as of 4 June 1963, weight and inertia characteristics for the XV-9A Hot Cycle Research Aircraft are listed below. The inertia of each item is calculated about the center of gravity of the item.

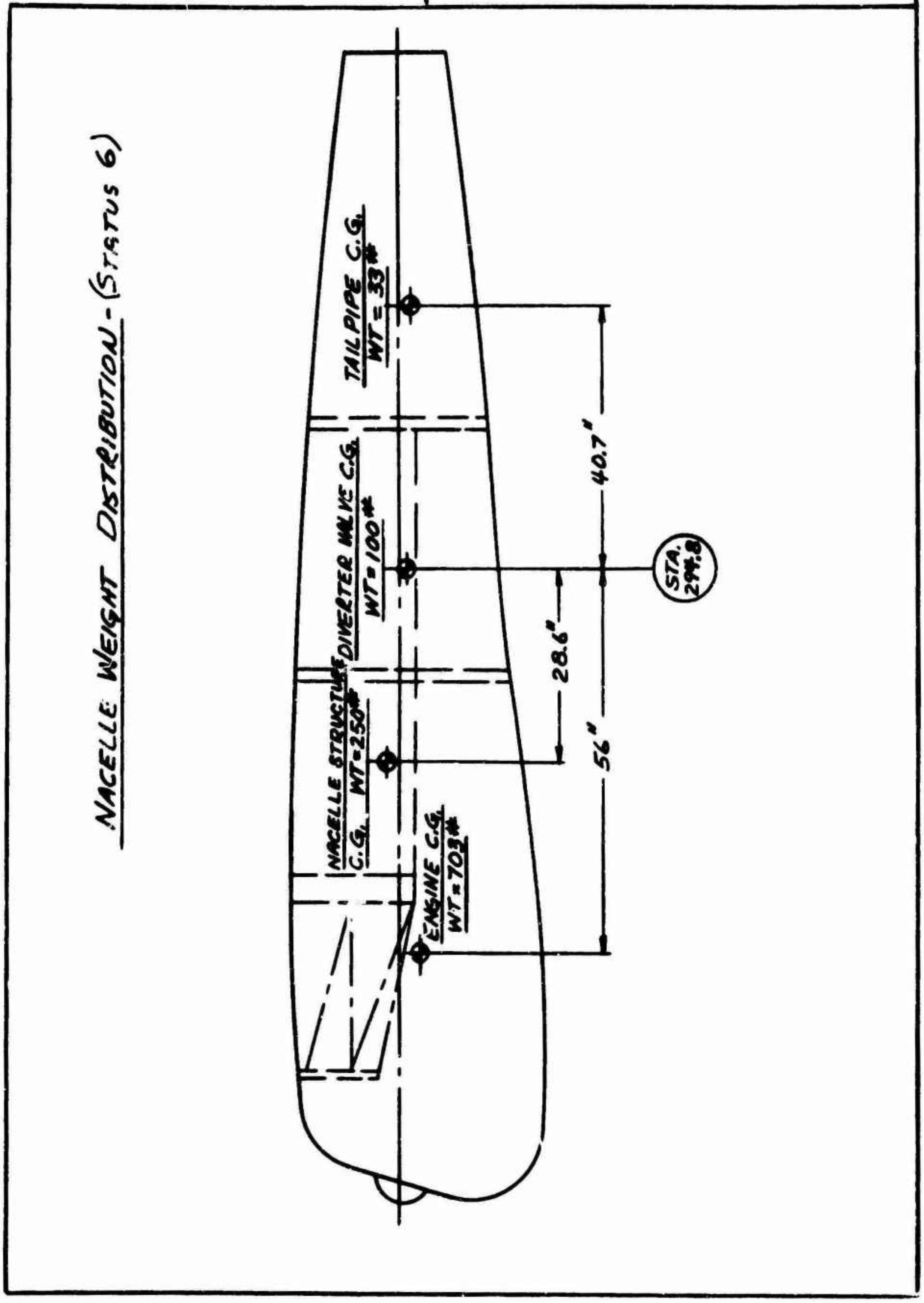
		Center of Gravity		Moment of Inertia		
Item	Weight Lb	Dist From Rotor ℄ Inches	Below Rotor Plane Inches	I_{o_x} Roll Slug Ft^2	I_{o_y} Pitch Slug Ft^2	I_{o_z} Yaw Slug Ft^2
Rotor module	3, 369	0. 0	2. 0	-	-	-
Fuselage module	2 086	35. 9 Aft	83. 4	617. 5	8, 235. 2	8, 388. 0
Power module	3, 338	27. 9 Fwd	50. 1	1, 739. 8	1, 159. 2	2, 773. 6
Weight empty	8, 793	2. 0 Fwd	39. 5	4, 318. 2	12, 516. 2	12, 322. 5
Useful load	6, 507	0. 5 Aft	79. 3	-	-	-
Design gross wt	15, 300	1. 0 Fwd	56. 5	6, 001. 3	20, 616. 1	19, 045. 7
Useful load, alternate overload	16, 707	1. 6 Fwd	72. 2	-	-	-
Gross weight, alternate overload	25, 500	1. 7 Fwd	61. 0	6, 036. 9	17, 322. 0	15, 603. 7

Note: Moment of inertia of the 15, 300-lb gross weight helicopter about the rotor centerline is 19, 068 slug ft^2. This represents 11, 931 pounds of weight (excluding rotor, hub, shaft, and all rotating controls about rotor shaft).

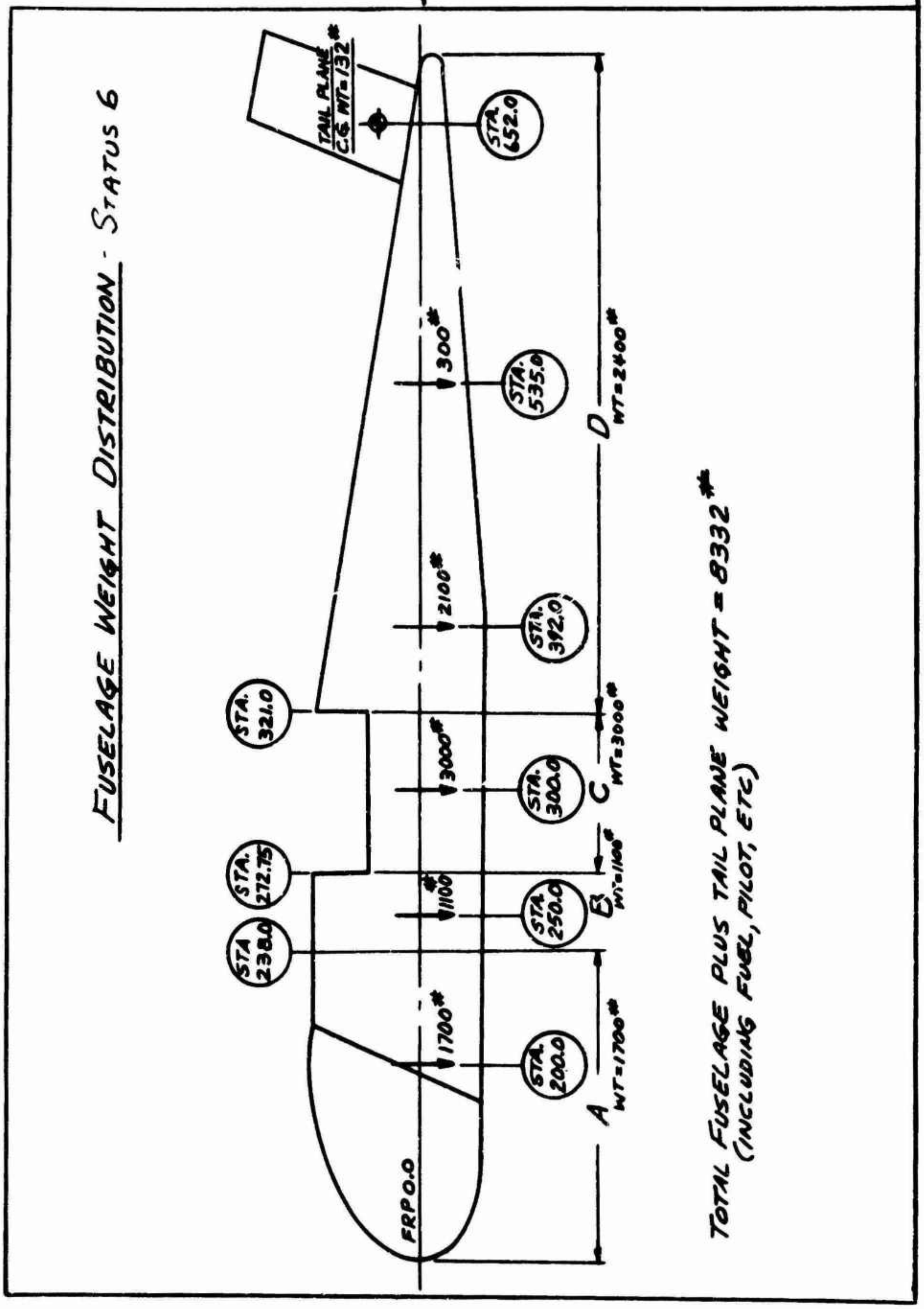
Weight Breakdown by Module

Weight Empty			8,793
Rotor Module		3,369	
Blade Assembly (3)	1,851.2		
Gimbal Assembly	124.8		
Hub Structure	649.5		
Shaft Assy	113.4		
Seals and Retainers	38.2		
Rotor Head Controls (rotating)	517.5		
Rotor Drive Ducting (rotating)	75.2		
Round-off adjustment	- 0.7		
Fuselage Module		2,086	
Structure, Forward (cockpit)	78.1		
Structure, Center Section	775.0		
Structure, Tail Section	120.0		
Alighting Gear, Main Installation (2)	357.2		
Alighting Gear, Tailwheel Installation	60.0		
Flight Controls, Cockpit	26.3		
Flight Controls, Linkage	7.2		
Flight Controls, Yaw Control	105.0		
Propulsion, Fuel Cells (2)	200.0		
Propulsion, Fuel Plumbing	20.0		
Instruments and Navigation	50.0		
Electrical System	73.0		
Electronics System	10.0		
Furnishings	22.0		
Auxiliary Gear	50.0		
Tail Group, Tail Cone	23.4		
Tail Group, Fixed Surface (2)	83.3		
Tail Group, Control Surface (2)	25.5		
Power Module		3,338	
Nacelle Section (2)		2,242	
Nacelle Structure	500.0		
Engines	1,160.0		
Air Induction System	34.6		
Exhaust System	76.2		
Lube System	68.9		
Engine Controls	50.0		

Start System	12.0	
Fuel System	10.0	
Diverter Valves and Connectors	251.1	
Hydraulic System	20.0	
Electrical System	59.0	
Pylon (box beam) Section		1,096
Structure, Box Beam	196.9	
Structure, Rotor Pylon	80.2	
Structure, Pylon Fairing	43.7	
Structure, Leading Edge	4.2	
Structure, Trailing Edge	26.6	
Rotor Drive Ducting (fixed)	100.6	
Electrical System	83.0	
Hydraulic System	160.0	
Fire Extinguishing	50.0	
Accessory Gearbox	72.4	
Rotor Lube System	6.8	
Fuel Distribution	50.0	
Alighting Gear Structure	20.0	
Flight Controls, Cylinders	82.2	
Flight Controls, Linkage	76.4	
Flight Controls, Swashplate	42.9	



NACELLE WEIGHT DISTRIBUTION - (STATUS 6)


FUSELAGE WEIGHT DISTRIBUTION - STATUS 6
TAIL PLANE C.G. WT = 132#
STA. 652.0
300#
STA. 535.0
D WT=2400#
2100#
STA. 392.0
STA. 321.0
3000#
STA. 300.0
C WT=3000#
STA. 272.75
1100#
STA. 250.0
B WT=1100#
STA 238.0
1700#
STA. 200.0
A WT=1700#
FRP 0.0
TOTAL FUSELAGE PLUS TAIL PLANE WEIGHT = 8332#
(INCLUDING FUEL, PILOT, ETC)

BASIC LOADS

FLIGHT & CRASH CONDITIONS — LOAD FACTORS, ANGULAR ACCELERATIONS AND TAIL LOADS

LIMIT LOAD TABLE

CASE No.	LOAD FACTORS n_x	n_y	n_z	ANGULAR ACCELERATIONS $\ddot{\theta}$ (PITCHING) RAD/SEC²	$\ddot{\psi}$ (YAWING) RAD/SEC²	$\ddot{\phi}$ (ROLLING) RAD/SEC²	ROTOR THRUST COMPONENTS F_{GX} #	F_{GY} #	F_{GZ} #	TAIL LOADS VERTICAL #	HORIZONTAL #
8,a	0	0	+10	0	0	0	0	0	-33690*	0	0
8,b	-10	0	0	0	0	0	+33690*	0	0	0	0
8,c	0	-4	0	0	0	0	0	+13476*	0	0	0
8,d	0	0	-2	0	0	0	0	0	+6738*	0	0
10,a	—	0	+2.5	+1.93	0	0	-5340	0	+37900	-318	0
10 b	—	0	+2.5	±1.80	0	0	+7960	0	+37500	0	0
11	—	—	+2.0	+1.26	±1.17	±3.14	-3830	±3730	+30200	-318	0
12	0	+.22	+1.0	0	+5.00	+1.10	0	0	+15300	0	+3320
13,a	0	0	+.63	+7.75	0	0	0	0	+14366	-4670	0
13,b	0	0	+1.24	-5.16	0	0	0	0	+14366	+4670	0

* FOR CASE 8,a,b,c & d ROTOR THRUST COMPONENTS ARE F_{RX}, F_{RY} & F_{RZ}

DESCRIPTION

CASE 8, a- 10G DOWN CRASH
b- 10G FOWARD CRASH
c- 4G SIDE CRASH
d- 2G UP CRASH

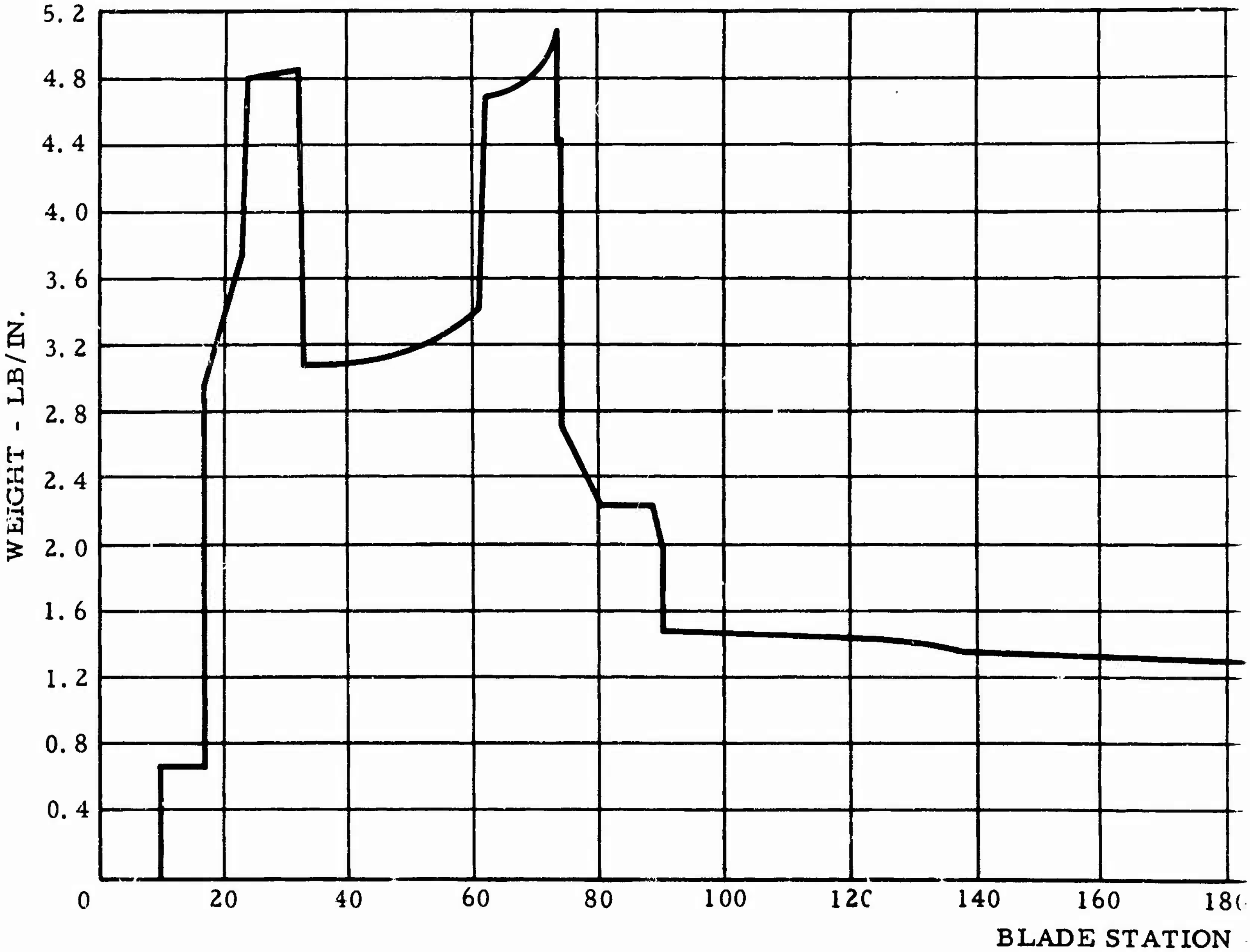
CASE 10, a- 2½G MANEUVER, ROTOR THRUST VECTOR AFT 10° TO SHAFT. PITCHING MOMENT REACTED BY TAIL LOAD
b- 2½G MANEUVER, ROTOR THRUST VECTOR FOWARD OR AFT 10° TO SHAFT. PITCHING MOMENT REACTED BY INERTIA

CASE 11, (.8)(2½G) MANEUVER WITH ROLLING PULL OUT, THRUST VECTOR 7.2° AFT & 7.0° TO SIDE (FROM DESIGN CRITERIA). ROLLING MOMENT REACTED BY INERTIA

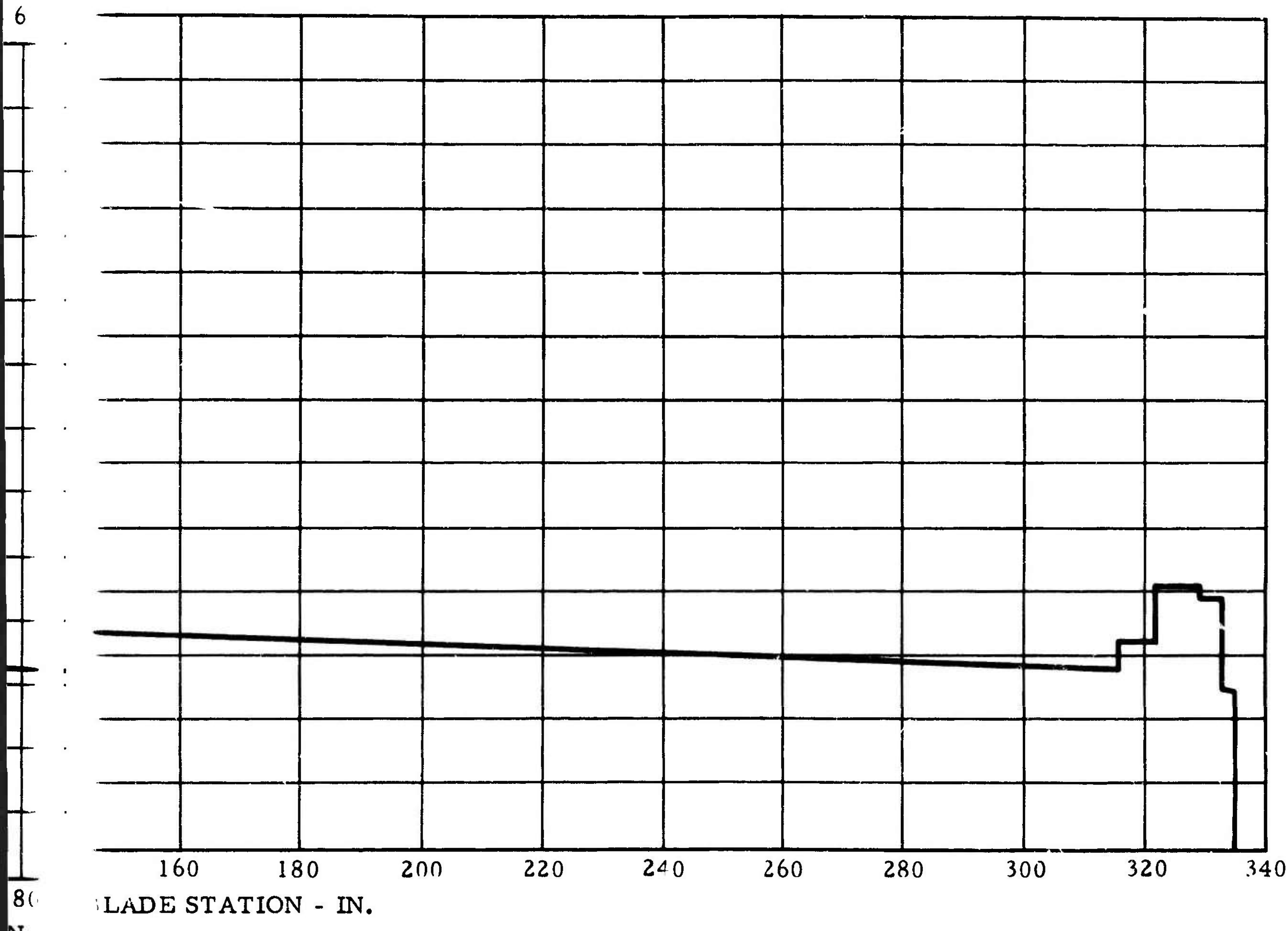
CASE 12, +1G CONDITION WITH HIGHEST ASYMMETRICAL LOAD 3320# } REF. SECT. 7. DESIGN CRITERIA

CASE 13, +1G CONDITION WITH HIGHEST SYMMETRICAL TAIL LOAD
a-MAXIMUM SYMMETRICAL DOWN LOAD -4670#
b-MAXIMUM SYMMETRICAL UP LOAD +4670#

MODEL XV-9A HELICOPTER
ROTOR BLADE WEIGHT
DISTRIBUTION - STATUS 6

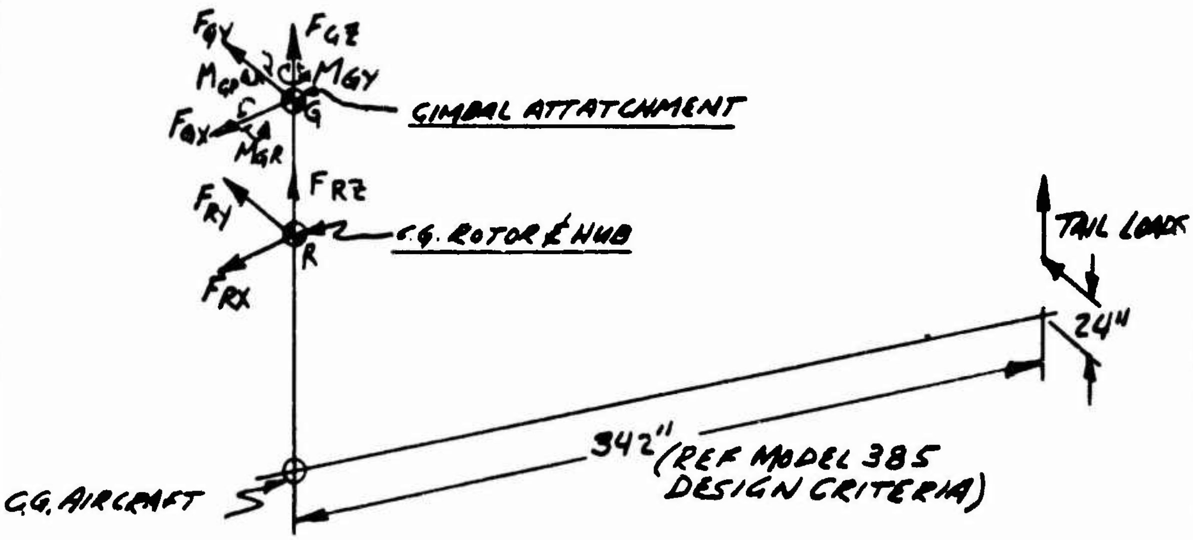


9A HELICOPTER
LADE WEIGHT
ION - STATUS 6



B

BASIC LOADS



CASE 12 - +1G CONDITION WITH HIGHEST ASYMMETRICAL LOAD 3320# FORCES RESISTED BY INERTIA

ROTOR THRUST = 15300 - 934 = 14366#

TRIM TAIL LOAD = +934#

$F_{GX} = 0$ $\qquad\qquad n_x = 0$

$F_{GY} = 0$ $\qquad\qquad n_y = \frac{3320}{15300} = +.22$

TAIL LOAD = 3320# HORIZ. $\qquad n_z = +1.00$

+934# VERTICAL

$F_{GZ} = 14366$

(PITCH) $M_{GP} = 0$ $\qquad \ddot{\theta} = 0$

(YAW) $M_{GY} = 3320(342) = +1135000$ IN#, $\ddot{\psi} = \frac{1135000}{18046(12)} = +5.00 \frac{RAD}{SEC^2}$

(ROLL) $M_{GR} = 3320(24) = +79700$ IN#, $\ddot{\phi} = \frac{79700}{6001(12)} = +1.10 \frac{RAD}{SEC^2}$

CASE 13,a – +1G CONDITION WITH HIGHEST SYMMETRICAL DOWNTAIL LOAD −4670#. FORCES RESISTED BY INERTIA.

TRIM LOAD @ 1G = +934#

DOWN TAIL LOAD @ $C_{L_{TAIL}} = -1$ IS −4670#

@ 1G ROTOR THRUST = 15300 − 934 = 14366#

$F_{GX} = 0$ $\qquad$ $n_X = 0$

$F_{GY} = 0$ $\qquad$ $n_Y = 0$

$F_{GZ} = 14366$# $\qquad$ $n_Z = \frac{14366 - 4670}{15300} = +.63$

$M_{GP} = (4670 + 934)(342) = +1916000$ IN# $\qquad$ $\ddot{\theta} = \frac{1916000}{20616(12)} = +7.75 \frac{RAD}{SEC^2}$

$M_{GY} = 0$ $\qquad$ $\ddot{\psi} = 0$

$M_{GR} = 0$ $\qquad$ $\ddot{\phi} = 0$

CASE 13,b – 1G CONDITION WITH HIGHEST SYMMETRICAL UP TAIL LOAD 4670#. FORCES RESISTED BY INERTIA.

TRIM LOAD @ 1G = +934#

@ 1G ROTOR THRUST = 14366# $\qquad$ UP TAIL LOAD @ $C_{L_{TAIL}} = +1$ IS +4670#

$F_{GX} = 0$ $\qquad$ $n_X = 0$

$F_{GY} = 0$ $\qquad$ $n_Y = 0$

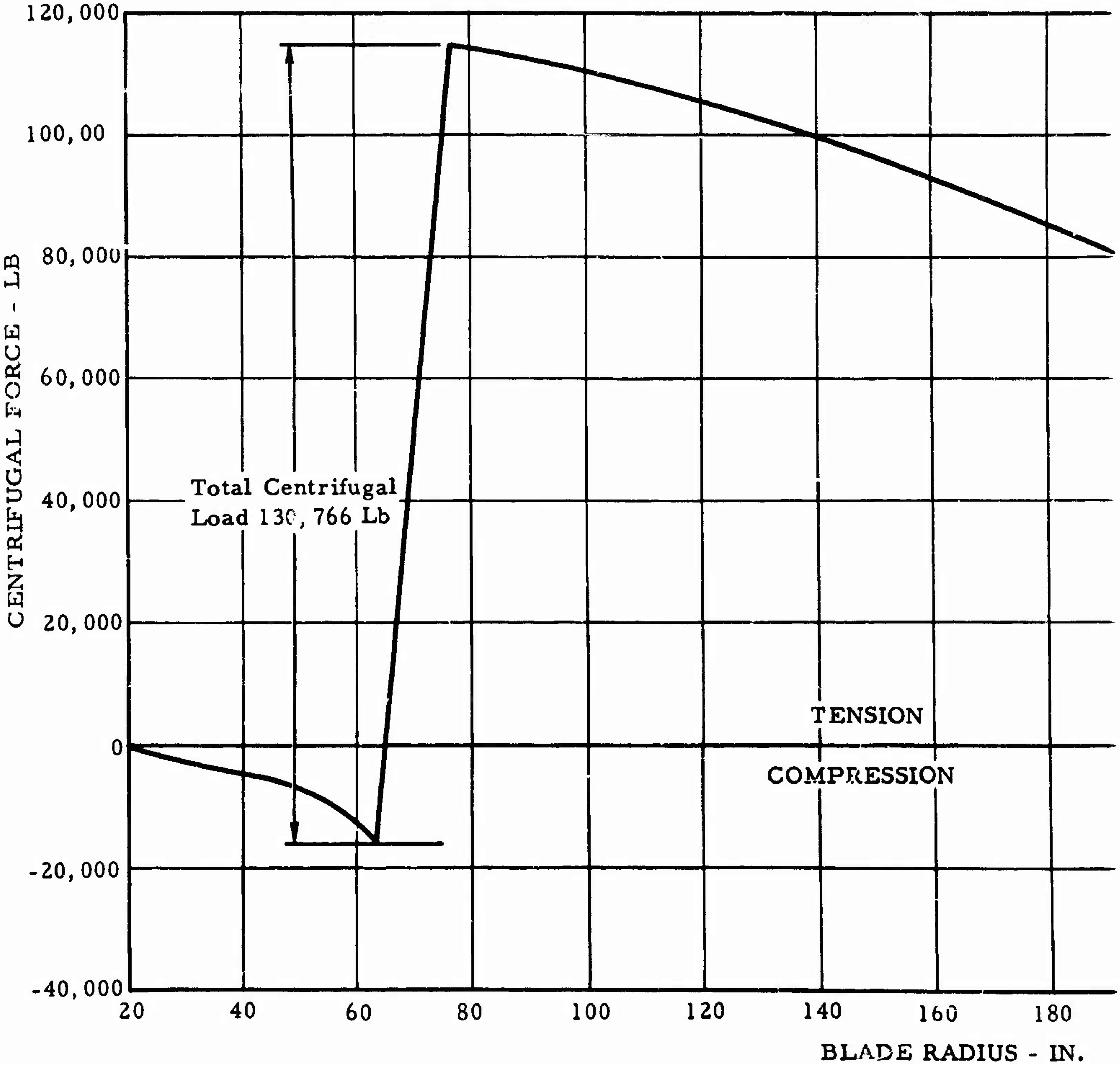
$F_{GZ} = 14366$# $\qquad$ $n_Z = \frac{14366 + 4670}{15300} = +1.24$

(PITCH) $M_{GP} = (4670 + 934)(342) = -1277000$ IN# $\qquad$ $\ddot{\theta} = \frac{-1277000}{20616(12)} = -5.16 \frac{RAD}{SEC^2}$

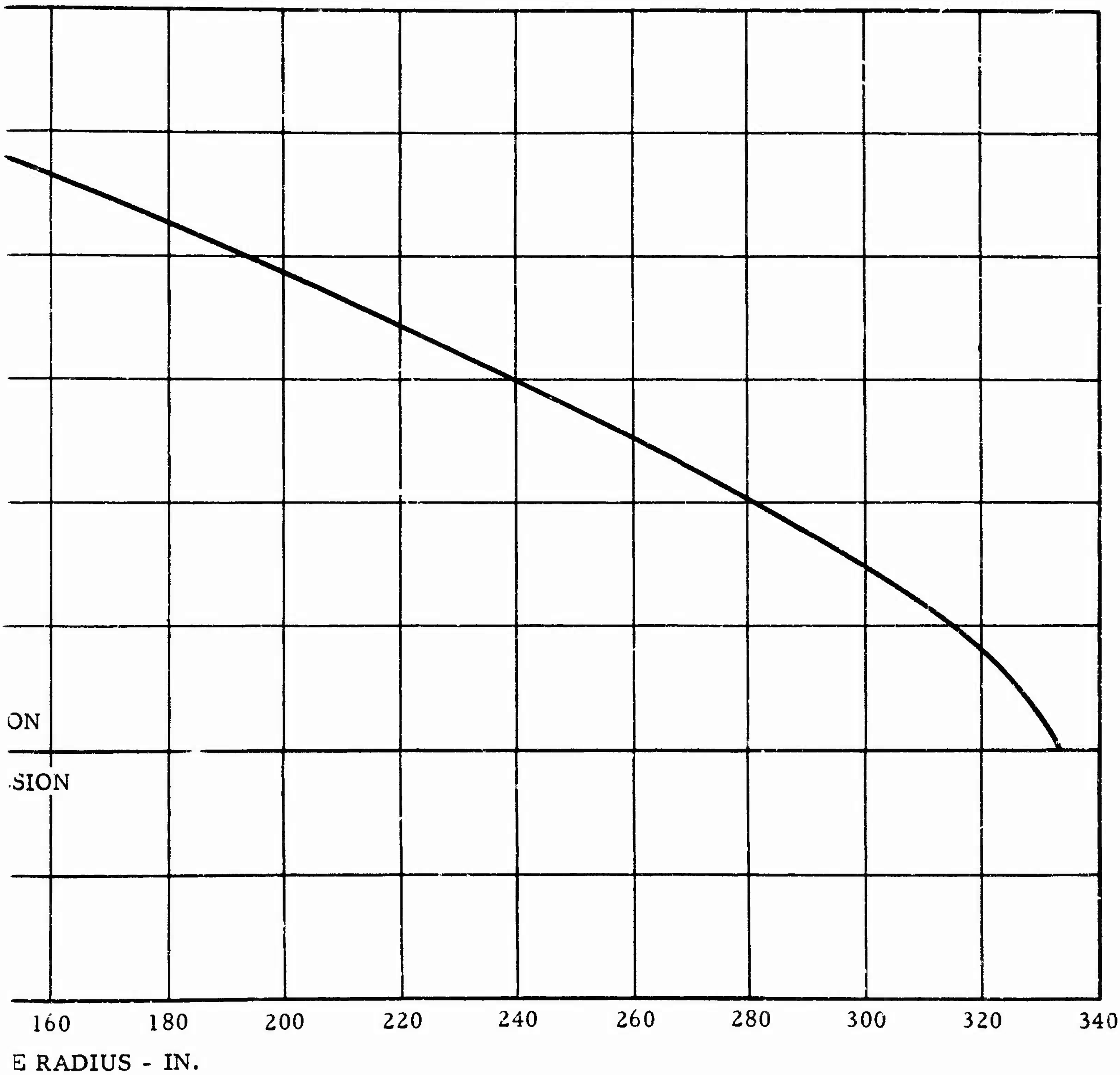
(YAW) $M_{GY} = 0$ $\qquad$ $\ddot{\psi} = 0$

(ROLL) $M_{GR} = 0$ $\qquad$ $\ddot{\phi} = 0$

MODEL XV-9A HELICOPT
CENTRIFUGAL LOADING ON ROT
BASED ON NORMAL RPI
N= 243 RPM
V= 700 FT/SEC STAT

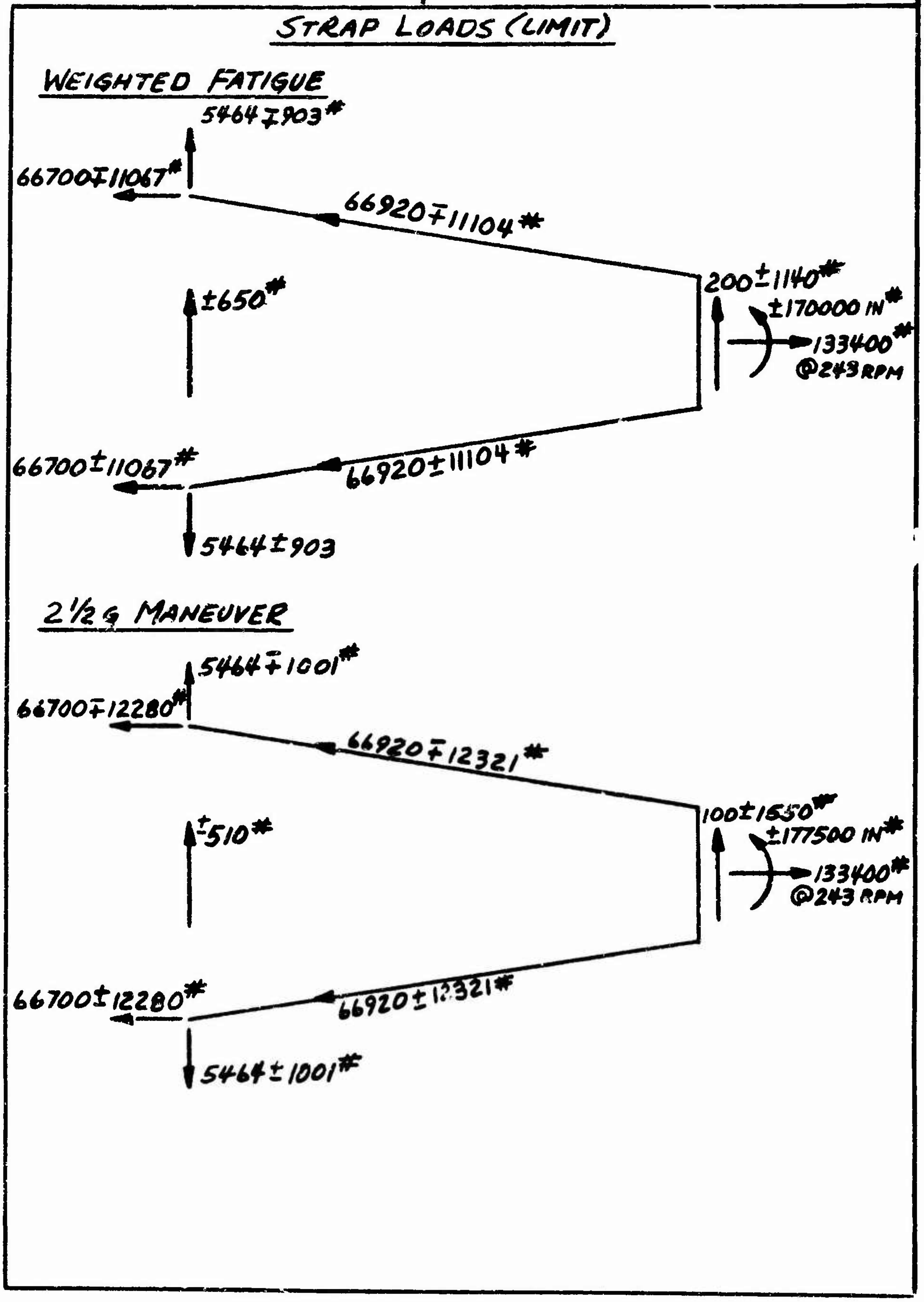


XV-9A HELICOPTER
LOADING ON ROTOR BLADE
ON NORMAL RPM
RPM
FT/SEC STATION 330



B

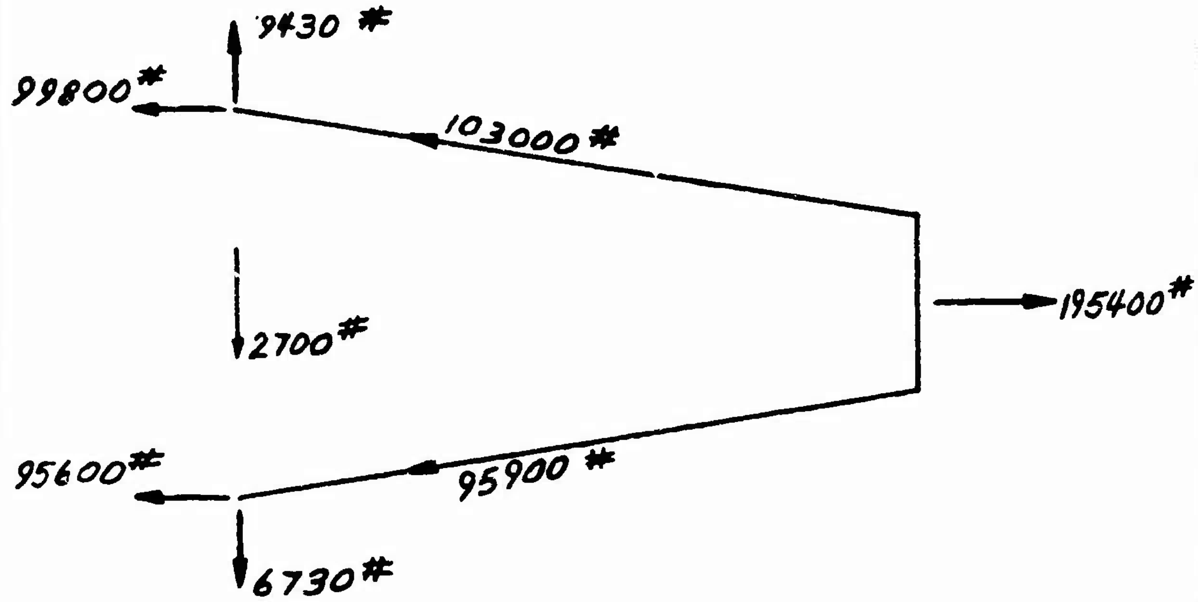
BASIC LOADS



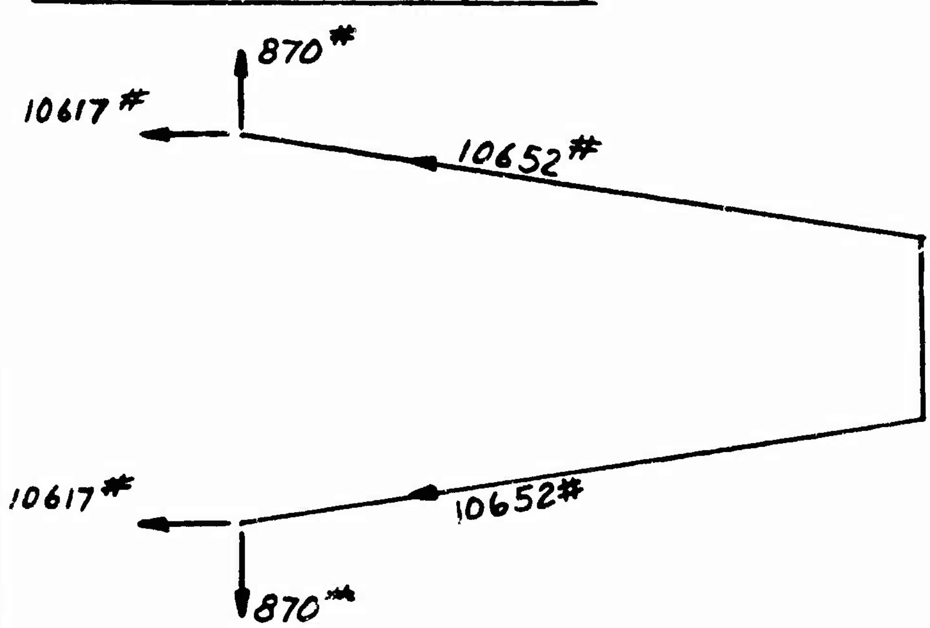
STRAP LOADS (LIMIT)
WEIGHTED FATIGUE
5464∓903#
66700∓11067#
66920∓11104#
±650#
200±1140#
±170000 IN#
133400#
@243 RPM
66700±11067#
66920±11104#
5464±903
2½ g MANEUVER
5464∓1001#
66700∓12280#
66920∓12321#
±510#
100±1550#
±177500 IN#
133400#
@243 RPM
66700±12280#
66920±12321#
5464±1001#

BASIC LOADS

OVER-REV. CONDITION (NO CHORDWISE SHEAR OR BENDING)

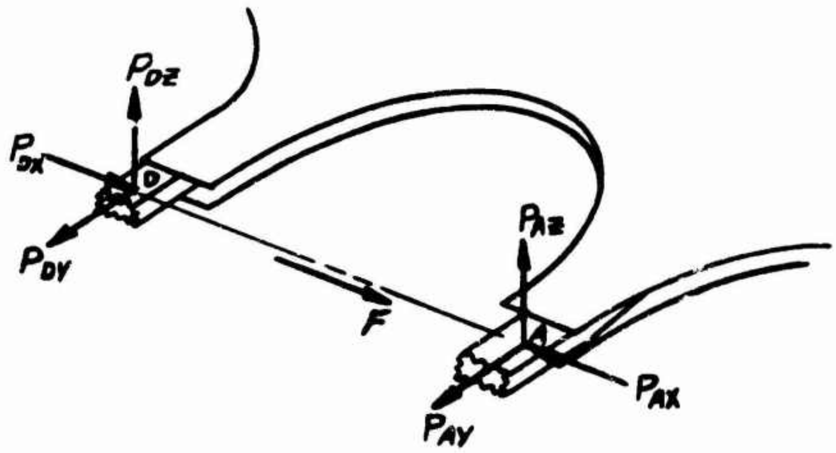


GROUND FLAPPING (2½ G)



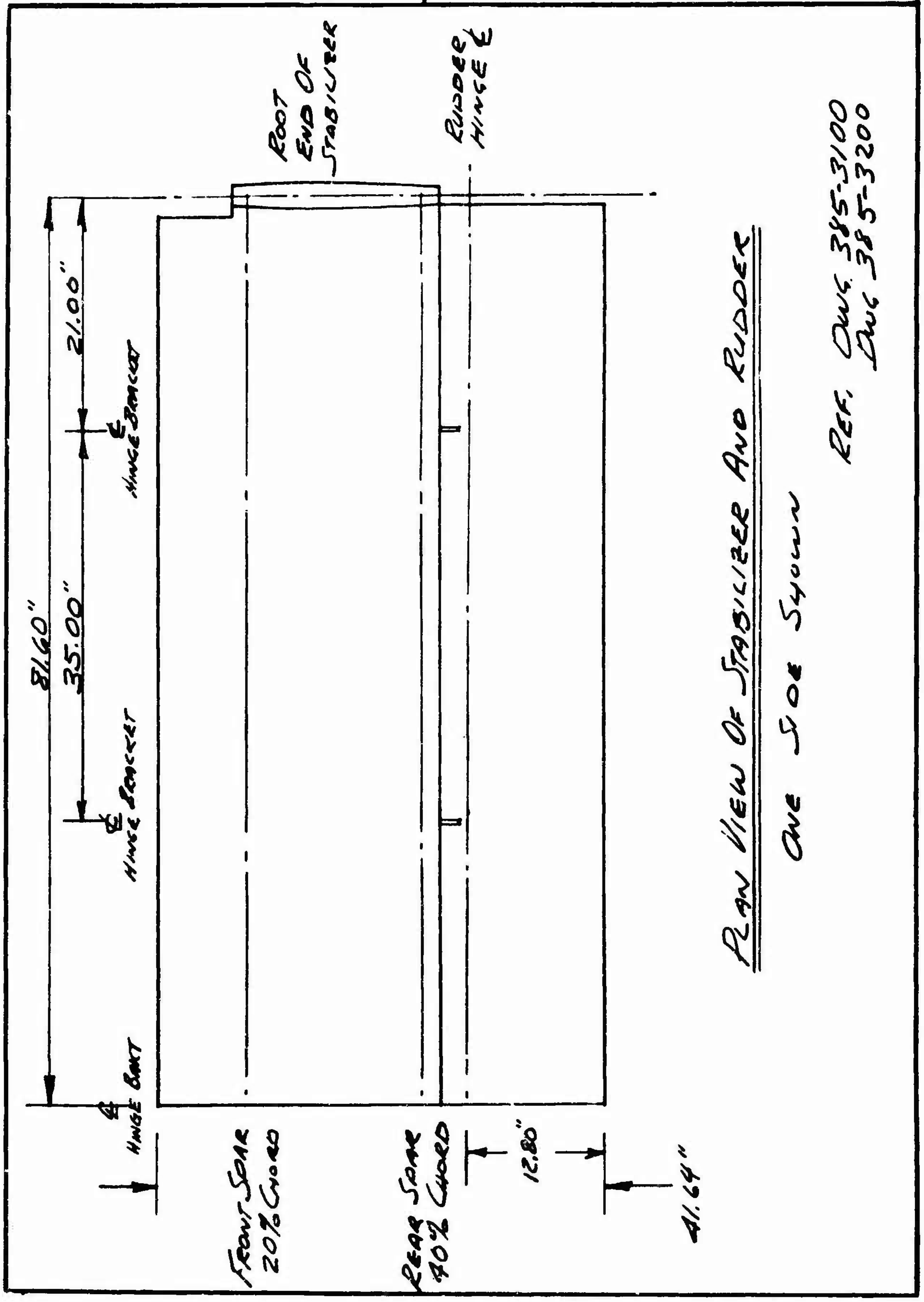
BASIC LOADS

BLADE LOADS APPLIED TO HUB

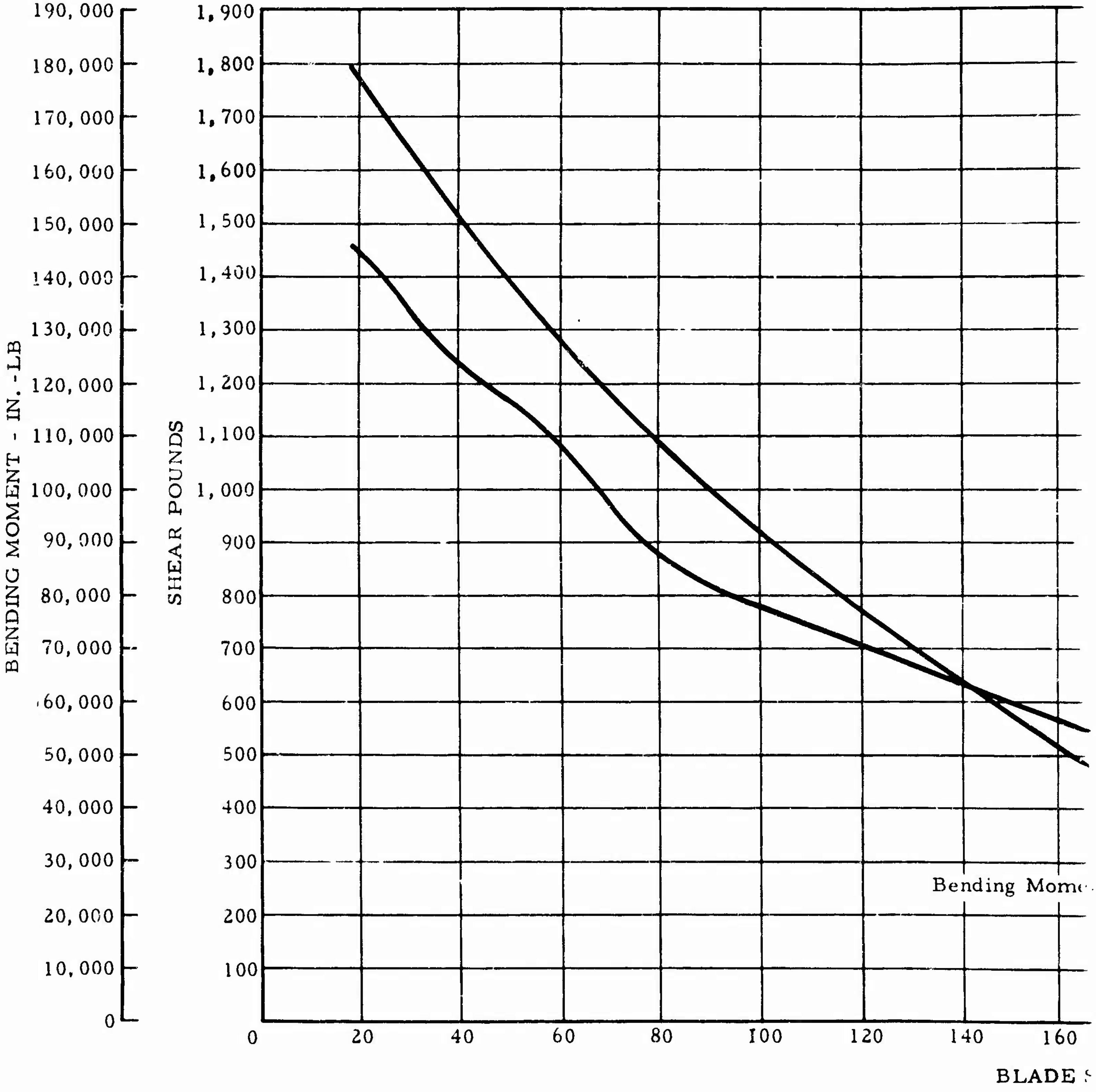


CONDITION \ LOADS *	P_A *	P_{AX}	P_{AY}	P_{AZ}	P_D *	P_{DX}	P_{DY}	P_{DZ}	F
2 1/2 G MANEUVER (LIMIT LOADS)	+79241	+6465	+78980	+10300	+54599	+4463	+54420	+3112	+510
WEIGHTED FATIGUE	+66920 ∓11104	+5464 ∓903	+66700 ∓11067	+5681 ∓943	+66920 ±11104	+5464 ±903	+66700 ±11067	+4510 ±742	±650
OVER-REV. 295 RPM (LIMIT LOADS)	+96000	+7830	+95600	+9500	+98023	+8170	+99800	+2560	+2700

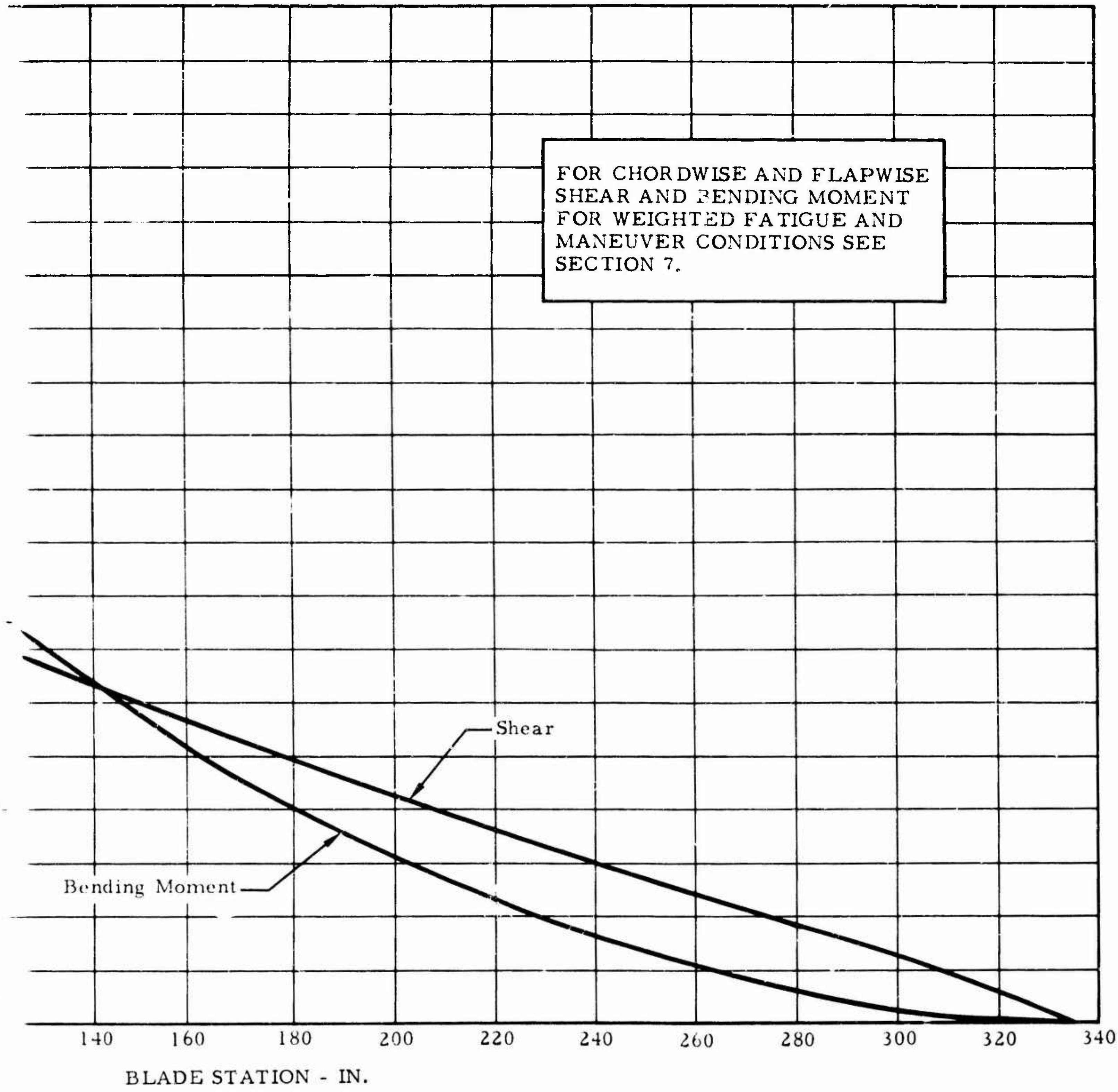
* STRAP LOAD


ROOT END OF STABILIZER
RUDDER HINGE ℄
81.60"
21.00"
35.00"
℄ HINGE BRACKET
℄ HINGE BRACKET
℄ HINGE BRKT
FRONT SPAR 20% CHORD
REAR SPAR 40% CHORD
12.80"
41.64"
PLAN VIEW OF STABILIZER AND RUDDER
ONE SIDE SHOWN
REF. DWG. 385-3100
DWG 385-3200

MODEL XV-9A HELICO
SHEAR AND BENDING MOMENT O
FOR 2-1/2-g LOADING – GROU

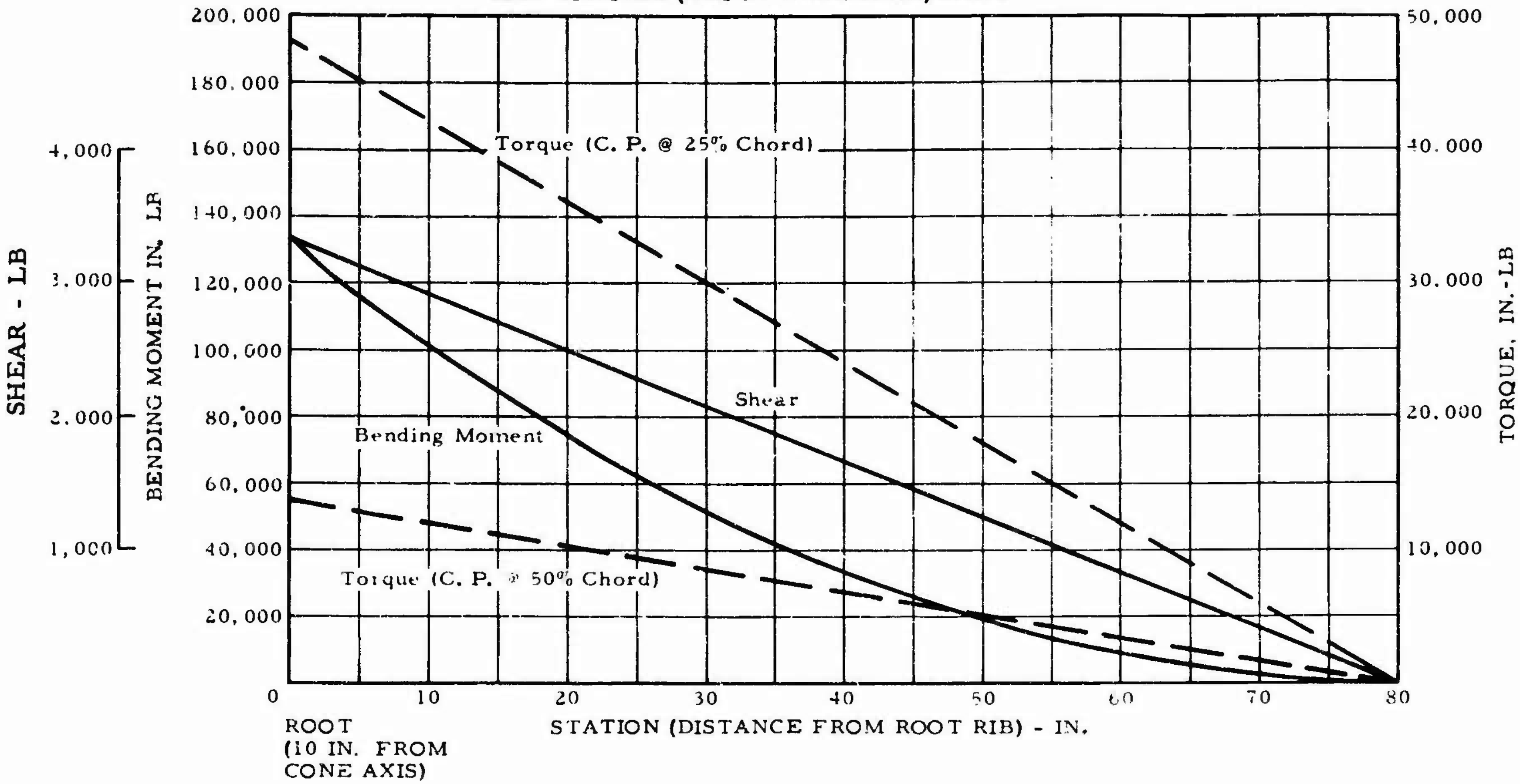


.L XV-9A HELICOPTER
.DING MOMENT ON ROTOR BLADE
LOADING – GROUND FLAPPING

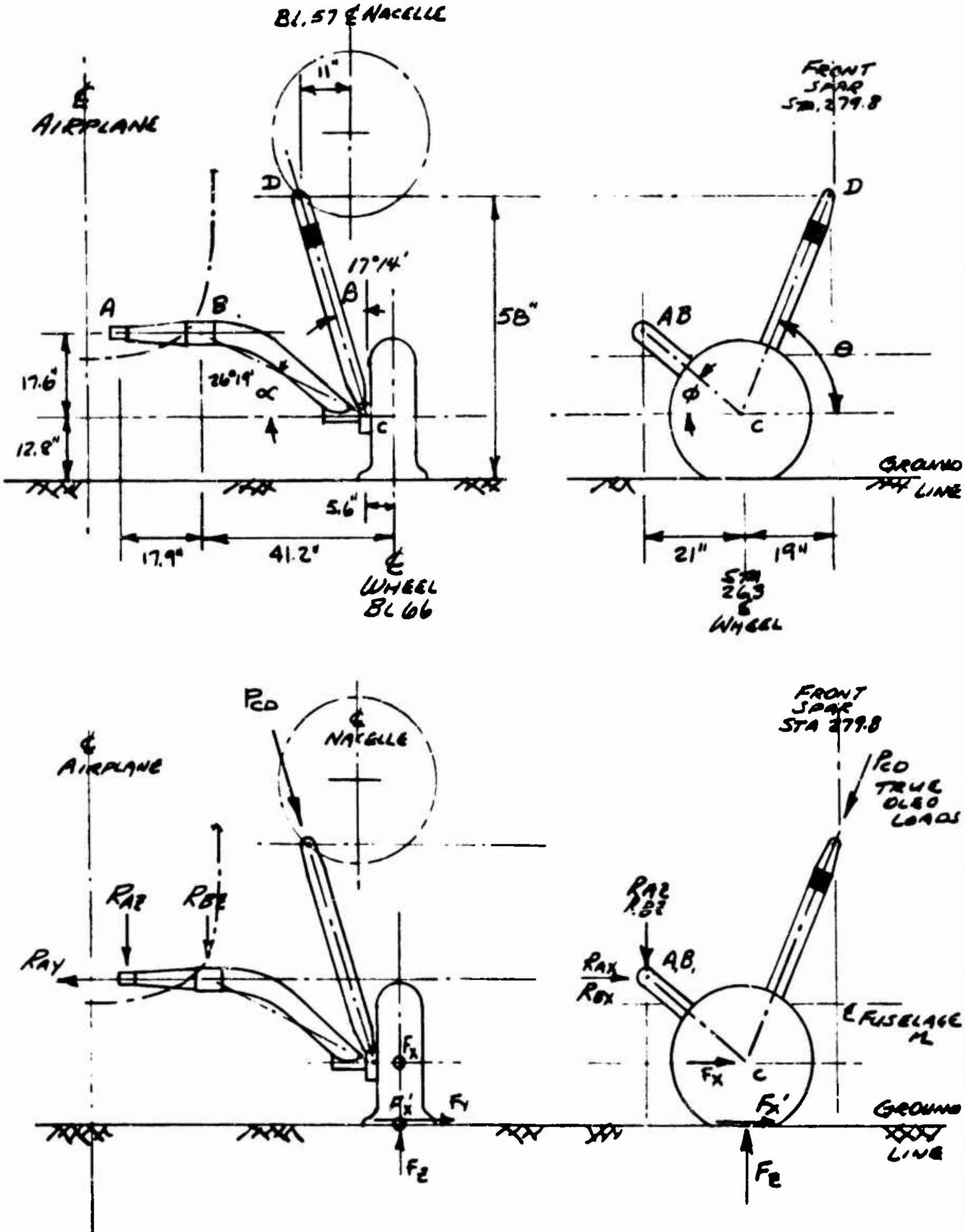


B

MODEL XV-9A HELICOPTER
TAIL SURFACE
BENDING MOMENT AND SHEARS (NORMAL TO SURFACE)
AND TORQUES (ABOUT REAR SPAR) LIMIT

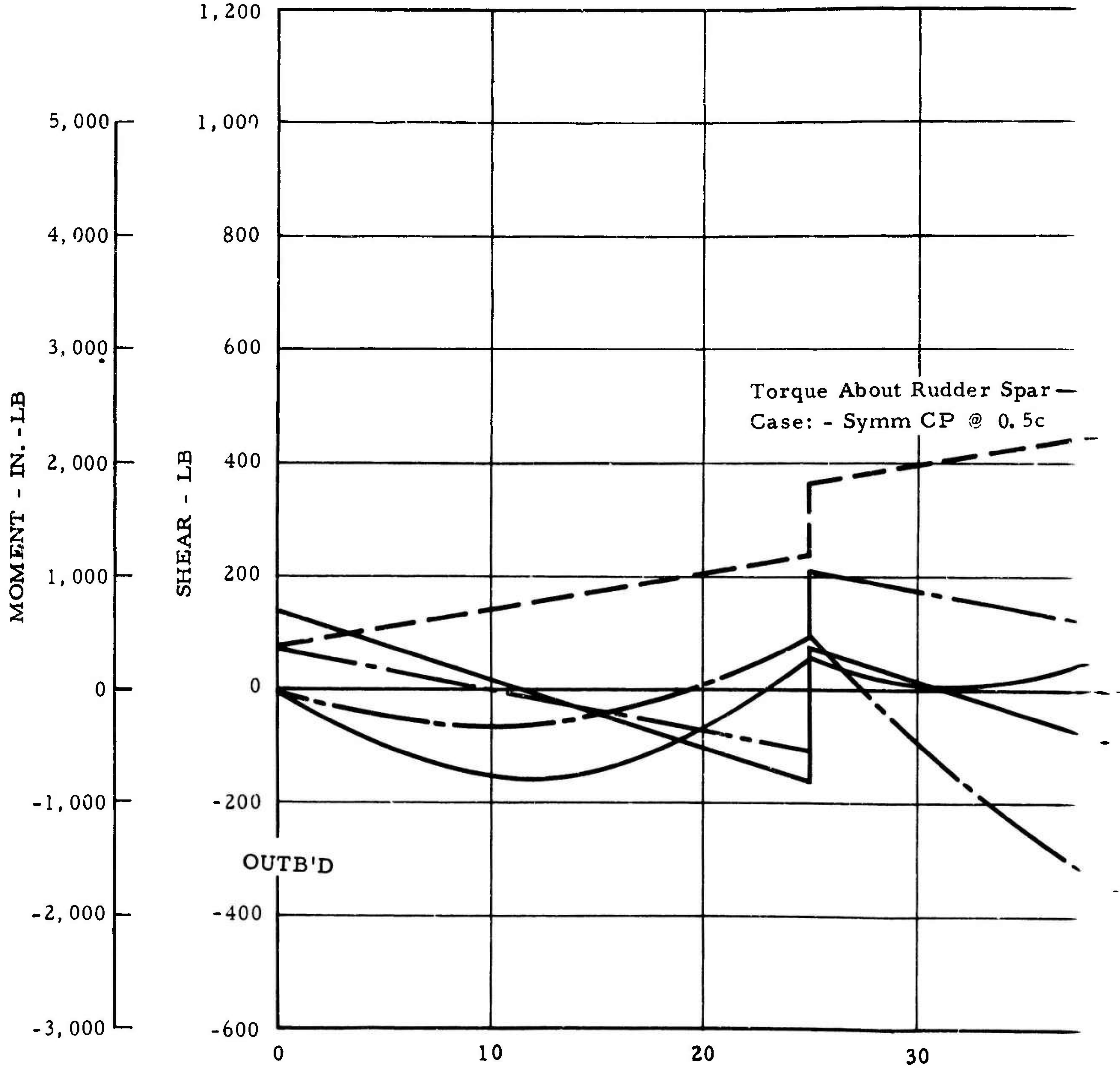


MAIN LANDING GEAR - GEOMETRY AND LOAD POINTS



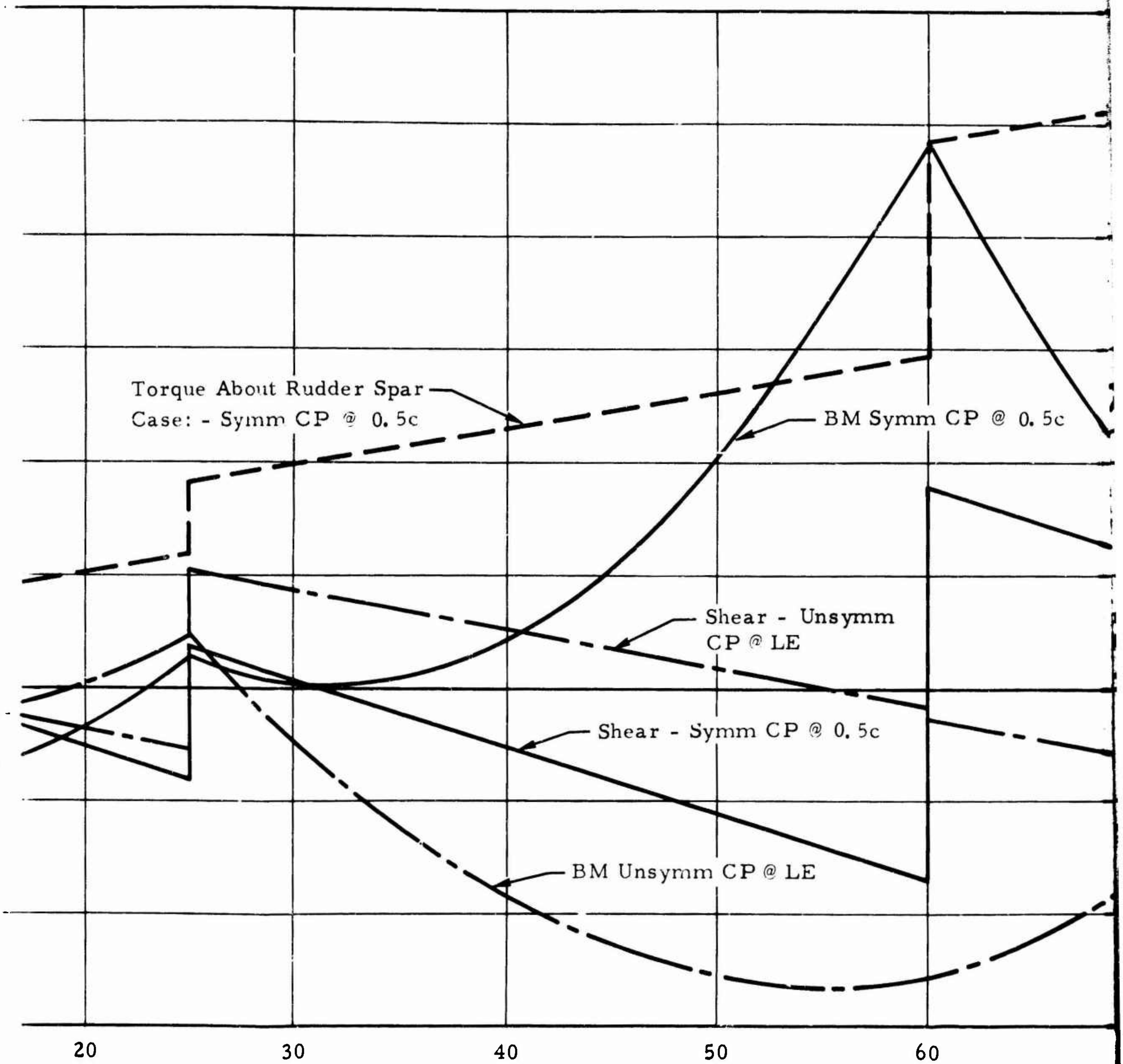
NOTE - STATIC POSITION OF GEAR SHOWN.

MODEL XV-9
SHEARS, BENDING MC



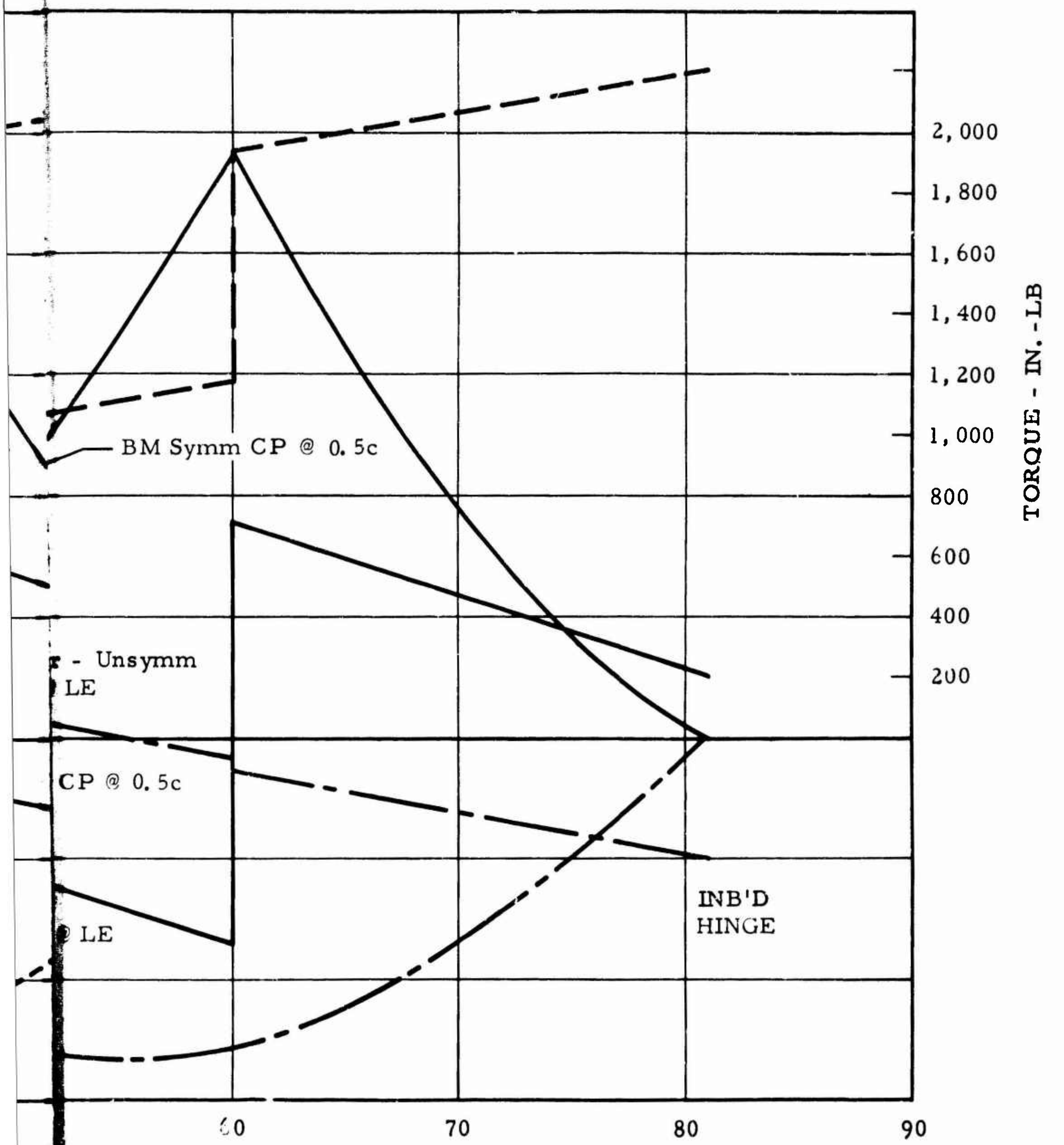
A

MODEL XV-9A HELICOPTER RUDDER
SHEARS, BENDING MOMENTS, AND TORQUE ABOUT SPAR
LIMIT LOADS



B

OUT SPAR



2,000
1,800
1,600
1,400
1,200
1,000
800
600
400
200
TORQUE - IN.-LB
BM Symm CP @ 0.5c
- Unsymm
LE
CP @ 0.5c
LE
INB'D
HINGE
60
70
80
90

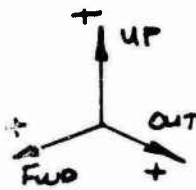
C

BASIC LOADS

LANDING CONDITIONS - ULTIMATE LOAD TABLE

	LOAD FACTORS			ROTATIONAL ACCELERATION			MAIN LANDING GEAR LOADS - POUNDS						TAIL WHEEL
							LIGHT LOADED SIDE			HEAVY LOADED SIDE			
CASE	n_x	n_y	n_z	$\ddot{\theta}$	$\ddot{\psi}$	$\ddot{\phi}$	F_x	F_y	F_z	F_x	F_y	F_z	F_z
1.	0	0	+4.50	0	0	0	0	0	+24000	0	0	+24000	+5450
2.	0	0	+3.90	+6.53	0	0	0	0	+22200	0	0	+22200	0
3.	-.76	0	+4.04	+4.22	+2.88	0	0	0	+23300	-11650	0	+23300	0
4	0	-.76	+4.04	+6.85	-1.80	-11.45	0	0	+23300	0	-11650	+23300	0
4'	0	+.76	+4.04	+6.85	+1.80	+11.45	0	0	+23300	0	+11650	+23300	0
5	-.49	0	+4.99	+.49	+2.24	+13.98	0	0	0	-7595	0	+15090	0
6	0	0	+1.20	-4.00	0	0	0	0	0	0	0	0	+3100
7	-1.44	0	+1.80	-2.22	0	0	-11016	0	+13770	-11016	0	+13770	0

CASES
1. THREE POINT LEVEL LANDING
2. TWO POINT LEVEL LANDING ON MAIN GEAR
3. TWO POINT LEVEL LANDING WITH DRAG ON ONE WHEEL
4. TWO POINT LEVEL LANDING INWARD LOAD ON ONE WHEEL
4'. TWO POINT LEVEL LANDING OUTWARD LOAD ON ONE WHEEL
5. ONE WHEEL LANDING WITH DRAG LOAD
6. TAIL FIRST LANDING
7. BRAKING CONDITION



n_x n_y n_z ARE LOAD FACTORS @ C.G.
F_x F_y F_z ARE LOADS FOR ONE WHEEL MAIN GEAR REF PAGE 200
LOADS ARE CALCULATED PER ANC-2

$+\ddot{\theta}$ = PITCHING ACCELERATION NOSE UP RAD/SEC^2
$+\ddot{\psi}$ = YAWING ACCELERATION NOSE LEFT RAD/SEC^2
$+\ddot{\phi}$ = ROLLING ACCELERATION L.H. UP RAD/SEC^2

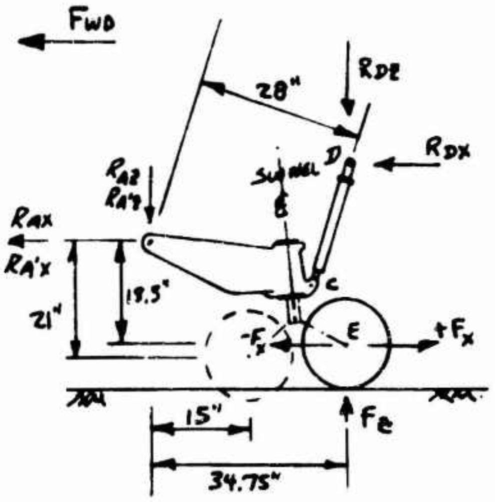
BASIC LOADS

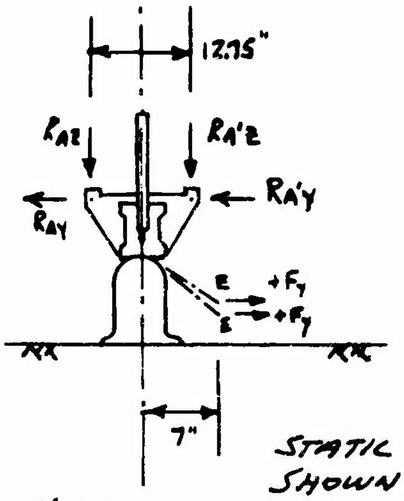
MAIN LANDING GEAR - REACTIONS TO NACELLE AND FUSELAGE
SHOWN ARE ULTIMATE LOADS - POUNDS

LANDING CASE NO.	LIGHTLY LOADED MAIN GEAR						HEAVILY LOADED MAIN GEAR					
	R_{AX}	R_{AY}	R_{AZ}	R_{BX}	R_{BZ}	P_{CD}	R_{AX}	R_{AY}	R_{AZ}	R_{BX}	R_{BZ}	P_{CD}
1.	-21450	+5650	-25500	+29000	+31750	+11950	-21450	+5650	-25500	+29000	+31750	+17900
2.	-19700	+5200	-23400	+26550	+29200	+16500	-19700	+5200	-23400	+26550	+29200	+16500
3.	-20800	+5480	-24800	+28100	+30800	+19500	-2760	+7790	-12660	+1585	+11380	+27650
4.	-20800	+5480	-24800	+28100	+30800	+19500	-7180	-6150	-5000	+14420	+11080	+19480
4'.	-20800	+5480	-24800	+28100	+30800	+19500	-34500	+17140	-44500	+41750	+50500	+19480
5.	0	0	0	0	0	0	-1790	+5050	-8200	+900	+7460	+17900
6.	—	—	—	—	—	—	—	—	—	—	—	—
7.	+1240	+7000	+5110	-539	-13300	+24800	+1240	+7000	+5110	-539	-13300	+24800

REF. PAGE 203 FOR DESCRIPTION OF LANDING CASES AND LOADS.

TAIL WHEEL - GEOMETRY, LOAD POINTS AND LOADS





STATIC POSITION SHOWN

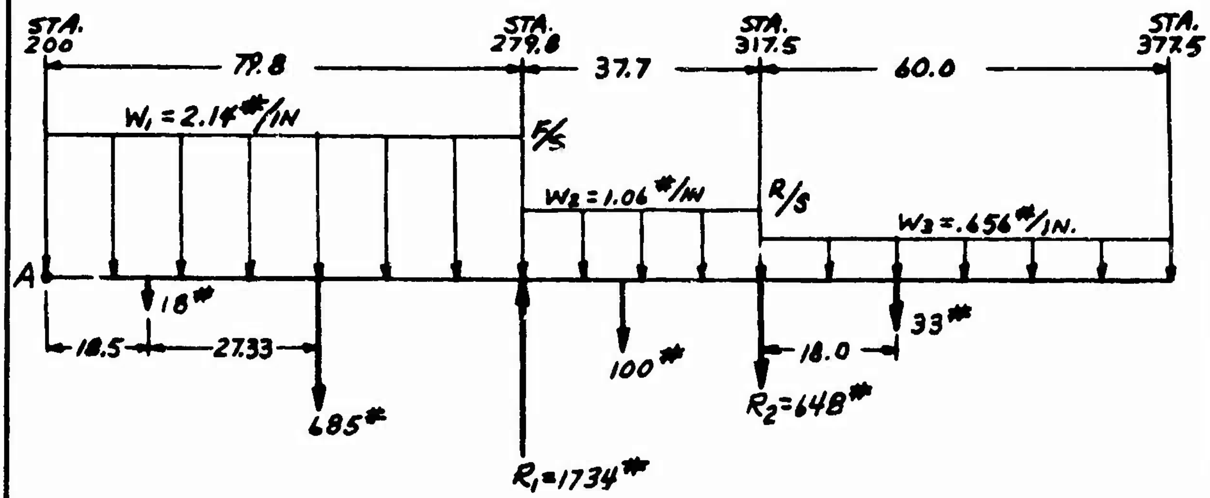
SUMMARY OF REACTIONS LIMIT VALUES

CASE	P_{cd} OLEO	R_{DX}	R_{DZ}	R_{AX}	R_{AY}	R_{AZ}	$R_{A'X}$	$R_{A'Y}$	$R_{A'Z}$
LANDING CASE 6(iii)	+2590	+776	+2465	+3560	+724	-3035	-4340	+723	+2630
TOW a(ii)	-2535	-763	-2415	-1915	0	+2060	-1915	0	+2060
TOW b(i)	+4990	+1498	+4750	+1546	0	-1578	+1545	0	-1575
TOW c(i)	-88	-27	-84	+2380	+805	-2585	-3980	+812	+4340
TOW d(i)	+2975	+896	+2845	+3545	+813	-3860	-2820	+812	+2685
TOW d(ii)	+833	+252	+796	-4125	-811	+1876	+2245	-812	-1002

LANDING CASE 6(iii) - TAIL FIRST LANDING WITH SIDE LOAD
TOW a (ii) - TAIL WHEEL SWIVELED FWD. WITH FWD PUSH
TOW b (i) - TAIL WHEEL SWIVELED AFT WITH PULL AFT
TOW c (i) - TAIL WHEEL SWIVELED 45° FROM FWD POSITION WITH A FWD. PUSH IN THE PLANE OF TAIL WHEEL.
TOW d (i) - TAIL WHEEL SWIVELED 45° FROM AFT POSITION WITH PULL AFT IN THE PLANE OF TAILWHEEL
TOW d(ii) - TAIL WHEEL SWIVELED 45° FROM AFT POSITION WITH PUSH FORWARD IN THE PLANE OF TAIL WHEEL

NACELLE — TOTAL VERTICAL LOAD DISTRIBUTION

1G DOWN (INERTIA LOADS ONLY)



$$\Sigma F_V = 0 = 18 + 685 - R_1 + 100 + R_2 + 33 + 2.14(79.8) + 1.06(37.7) + .656(60)$$

$$R_1 = 1086 + R_2$$

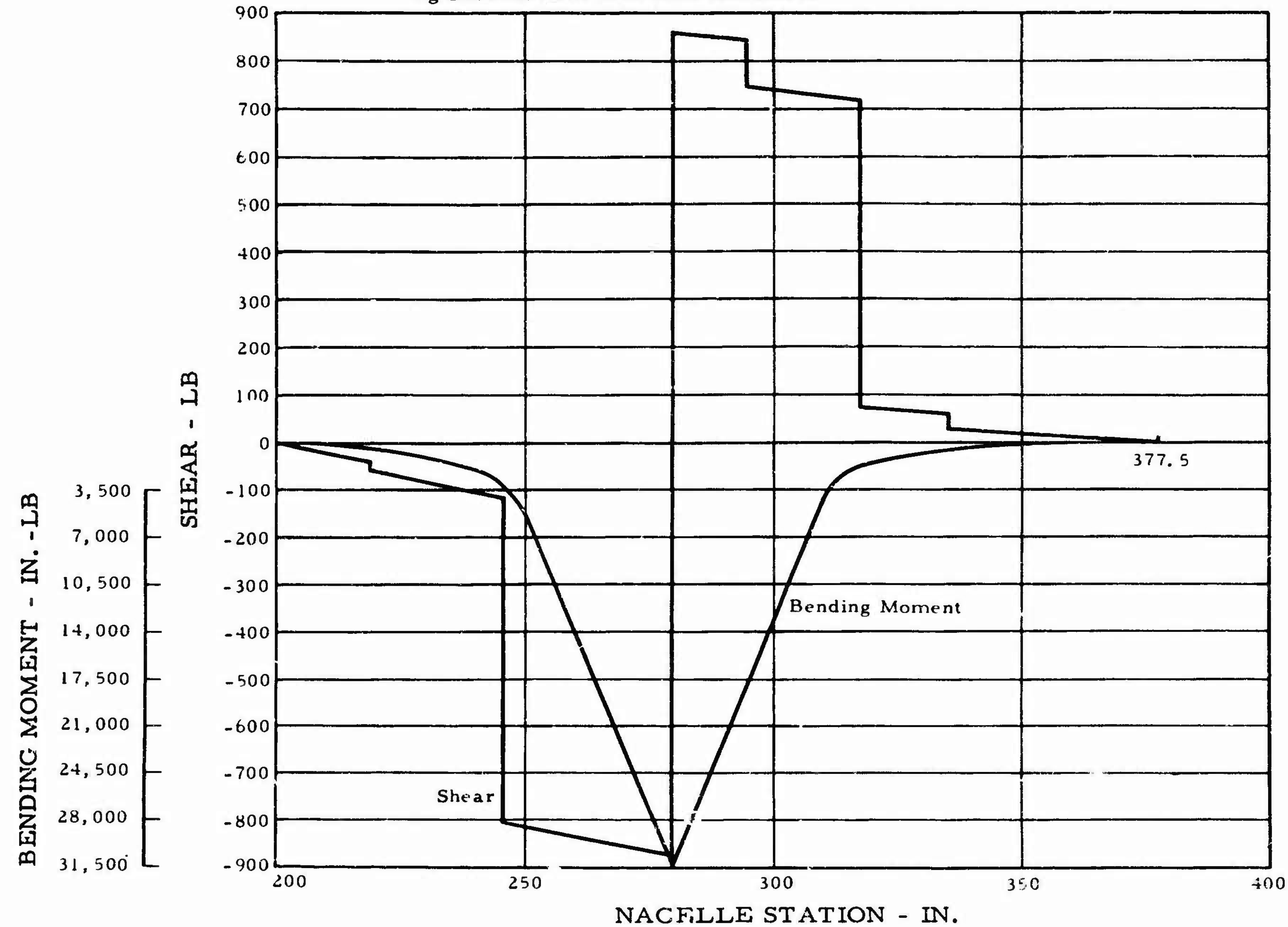
$$\Sigma M_A = 0 = 18(18.5) + 685(45.83) - 79.8R_1 + 100(94.8) + 117.5R_2 + 33(135.5)$$
$$+ 2.14(79.8)(39.9) + 1.06(37.7)(98.65) + .656(60)(147.5)$$

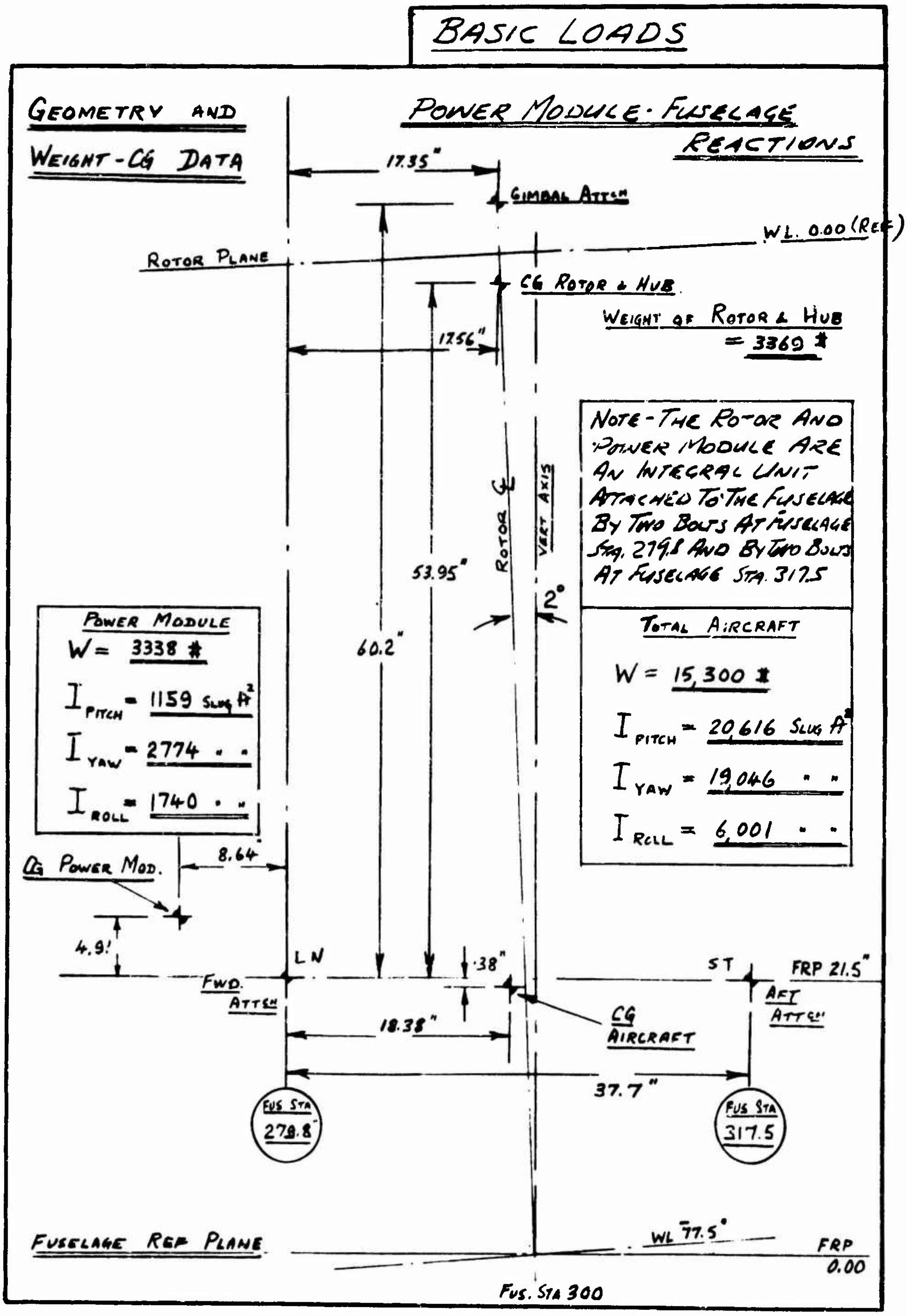
$$79.8(1086 + R_2) - 117.5R_2 = 62233$$

$$R_2 = \frac{62233 - 86650}{79.8 - 117.5} = 648\#$$

$$R_1 = 1086 + 648 = 1734\#$$

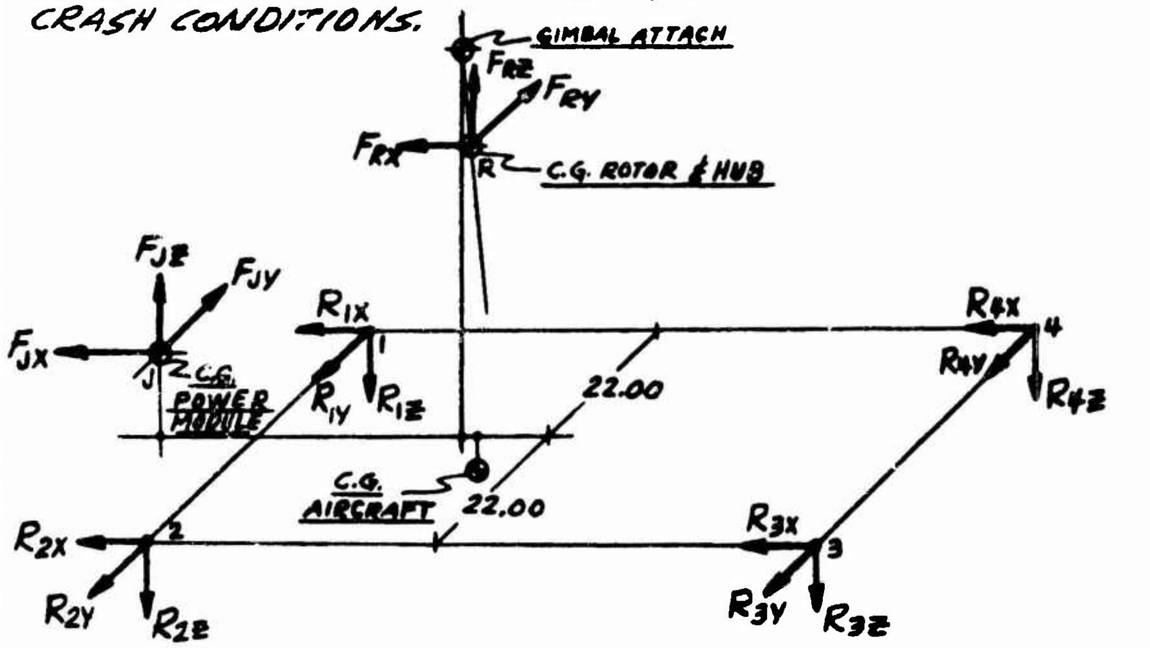
MODEL XV-9A HELICOPTER
NACELLE
1g SHEARS AND BENDING MOMENTS




BASIC LOADS
GEOMETRY AND WEIGHT-CG DATA
POWER MODULE · FUSELAGE REACTIONS
17.35"
GIMBAL ATTCH
W.L. 0.00 (REF)
ROTOR PLANE
CG ROTOR & HUB
WEIGHT OF ROTOR & HUB = 3369 #
17.56"
NOTE - THE ROTOR AND POWER MODULE ARE AN INTEGRAL UNIT ATTACHED TO THE FUSELAGE BY TWO BOLTS AT FUSELAGE STA. 279.8 AND BY TWO BOLTS AT FUSELAGE STA. 317.5
ROTOR ℄
VERT AXIS
53.95"
2°
60.2"
POWER MODULE
W = 3338 #
I PITCH = 1159 SLUG FT²
I YAW = 2774 " "
I ROLL = 1740 " "
TOTAL AIRCRAFT
W = 15,300 #
I PITCH = 20,616 SLUG FT²
I YAW = 19,046 " "
I ROLL = 6,001 " "
CG POWER MOD.
8.64"
4.9"
LN
.38"
ST
FRP 21.5"
FWD. ATTCH
AFT ATTCH
CG AIRCRAFT
18.38"
37.7"
FUS STA 279.8
FUS STA 317.5
FUSELAGE REF PLANE
WL 77.5"
FRP 0.00
FUS. STA 300

BASIC LOADS

THE CRITICAL LOADS FOR THE POWER MODULE TO FUSELAGE ATTACHMENTS AND FOR THE ROTOR SUPPORT STRUCTURE HAVE BEEN FOUND BY INVESTIGATION TO OCCUR IN THE CRASH CONDITIONS.



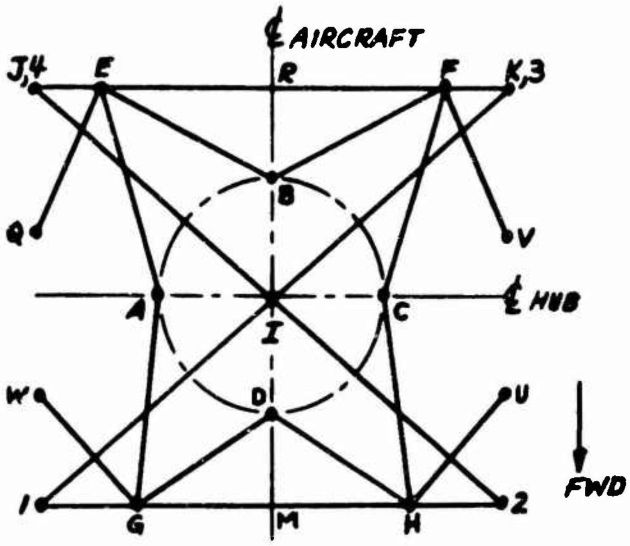
ULTIMATE LOADS

* THE MOMENT ABOUT THE Z AXIS IS ASSUMED TO BE REACTED BY THE X REACTIONS ONLY.

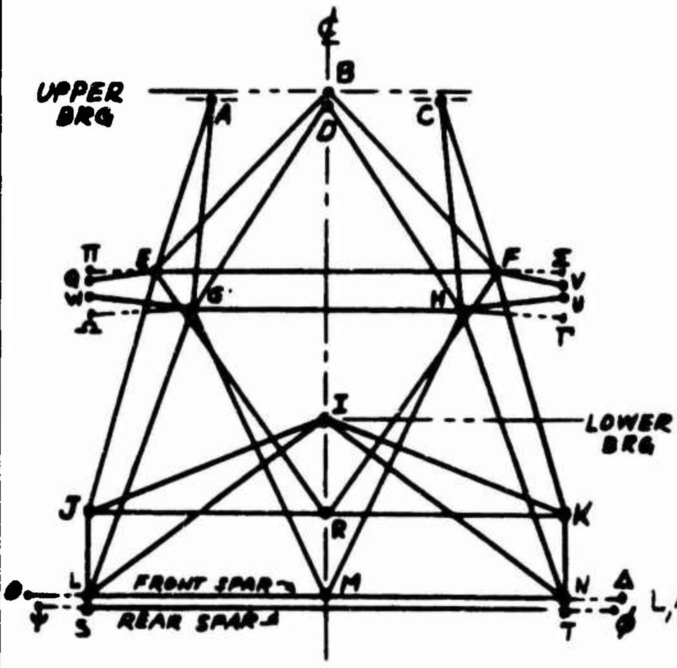
	CASE 8,b 10 G FWD CRASH			CASE 8,a 10G DWN CRASH			CASE 8,c * 4G SIDE CRASH			CASE 8,d 2G UP CRASH		
LOAD / REACT.	F_{RX}= 33690#	F_{JX}= 33380#	TOTAL #	F_{RZ}= -33690#	F_{JZ}= -33380#	TOTAL #	F_{RY}= 13476#	F_{JY}= 13352#	TOTAL #	F_{RZ}= 6738#	F_{JZ}= 6676#	TOTAL #
R_{1X}	-8423	-8345	-16768	—	—	0	198	4171	4369	—	—	0
R_{2X}	-8423	-8345	-16768	—	—	0	-198	-4171	-4369	—	—	0
R_{3X}	-8423	-8345	-16768	—	—	0	-198	-4171	-4369	—	—	0
R_{4X}	-8423	-8345	-16768	—	—	0	198	4171	4369	—	—	0
R_{1Y}	—	—	0	—	—	0	3369	3338	6707	—	—	0
R_{2Y}	—	—	0	—	—	0	3369	3338	6707	—	—	0
R_{3Y}	—	—	0	—	—	0	3369	3338	6707	—	—	0
R_{4Y}	—	—	0	—	—	0	3369	3338	6707	—	—	0
R_{1Z}	-24106	-2174	-26280	-9000	-20515	-29515	-8262	-745	-9007	1800	4103	5903
R_{2Z}	-24106	-2174	-26280	-9000	-20515	-29515	8262	745	9007	1800	4103	5903
R_{3Z}	24106	2174	26280	-7845	3825	-4020	8262	745	9007	1569	-765	804
R_{4Z}	24106	2174	26280	-7845	3825	-4020	-8262	-745	-9007	1569	-765	804

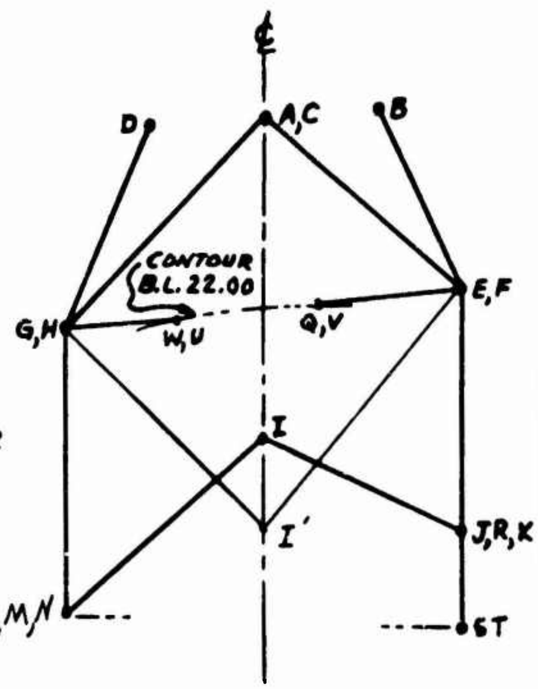
BASIC LOAD

RO:OR SUPPORT STRUCTURE

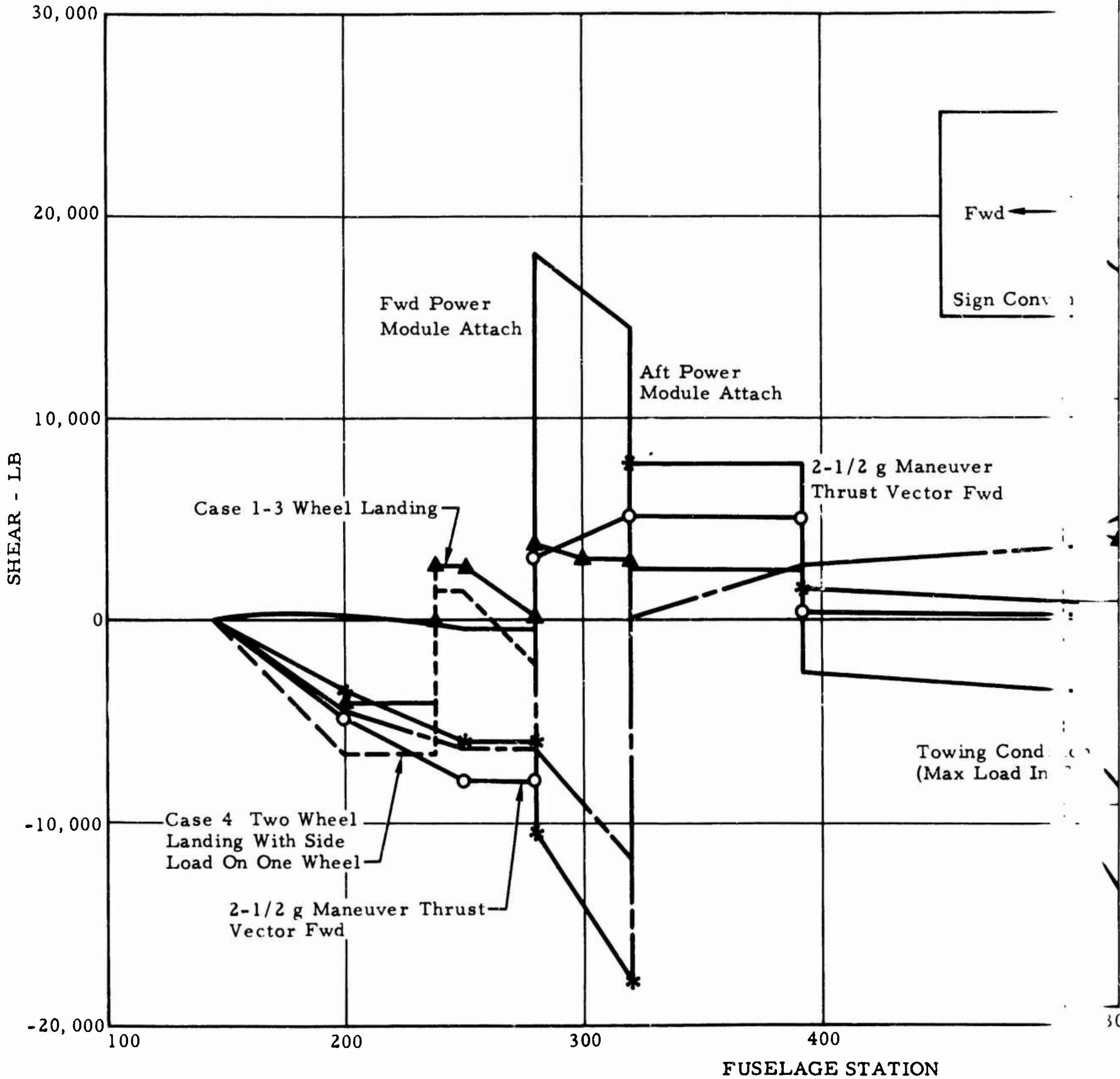


NOTE - MEMBERS I'G, I'H, I'E, AND I'F ARE SHOWN IN LOWER VIEW ONLY TO MAINTAIN CLARITY OF OTHER VIEWS

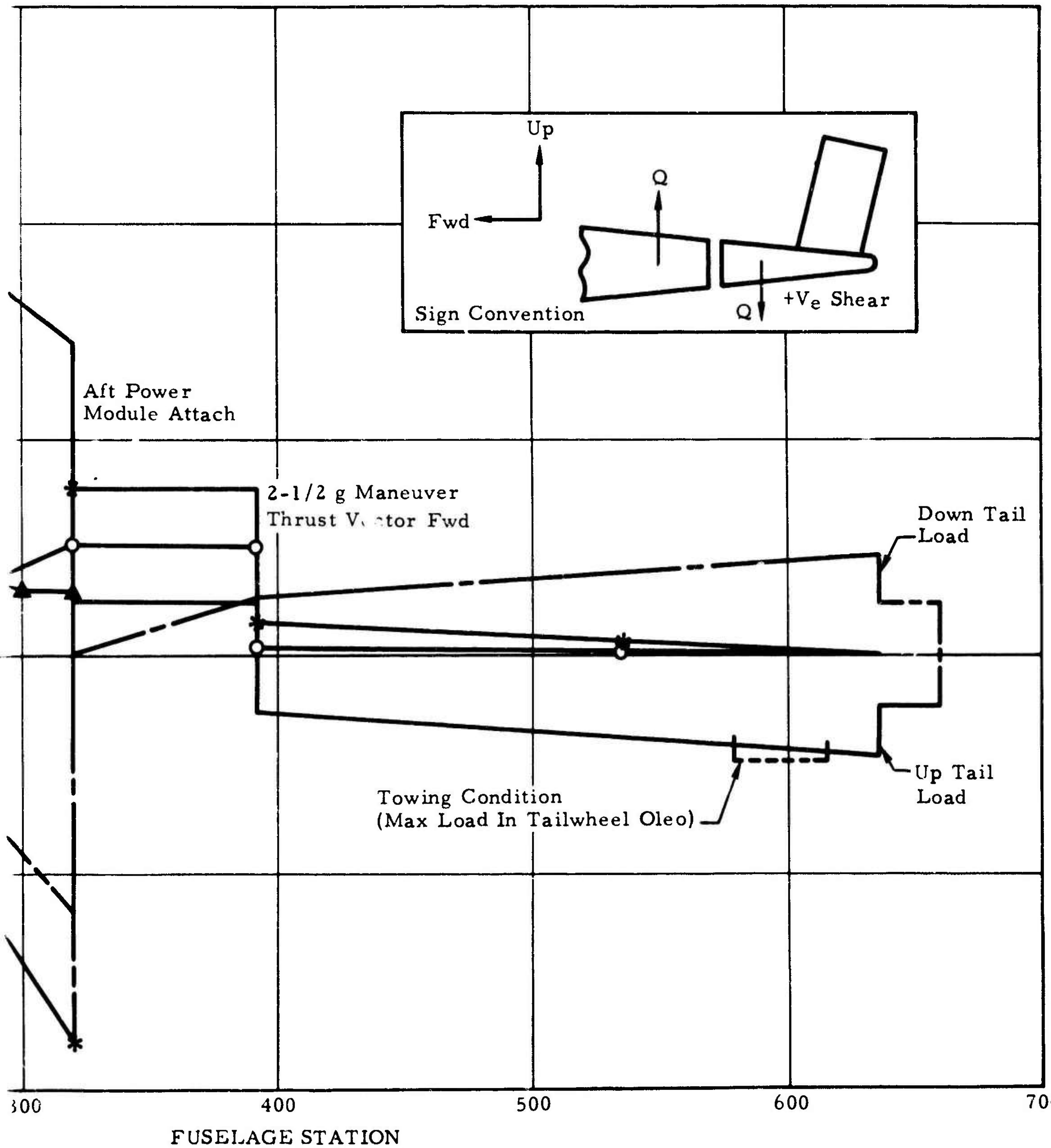




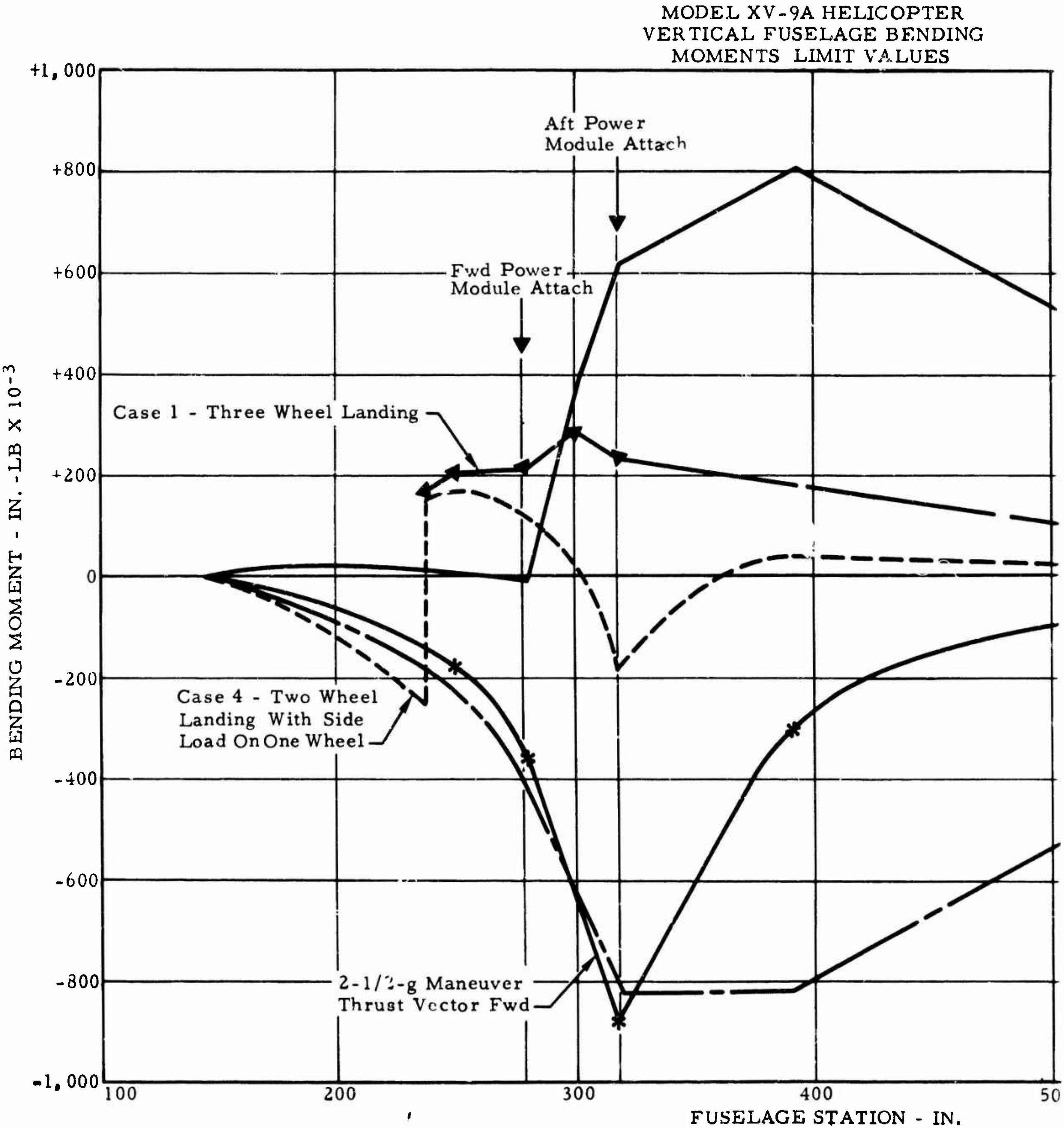
MODEL XV-9A HELICOPTER
VERTICAL FUSELAGE SHEARS
LIMIT ALUES



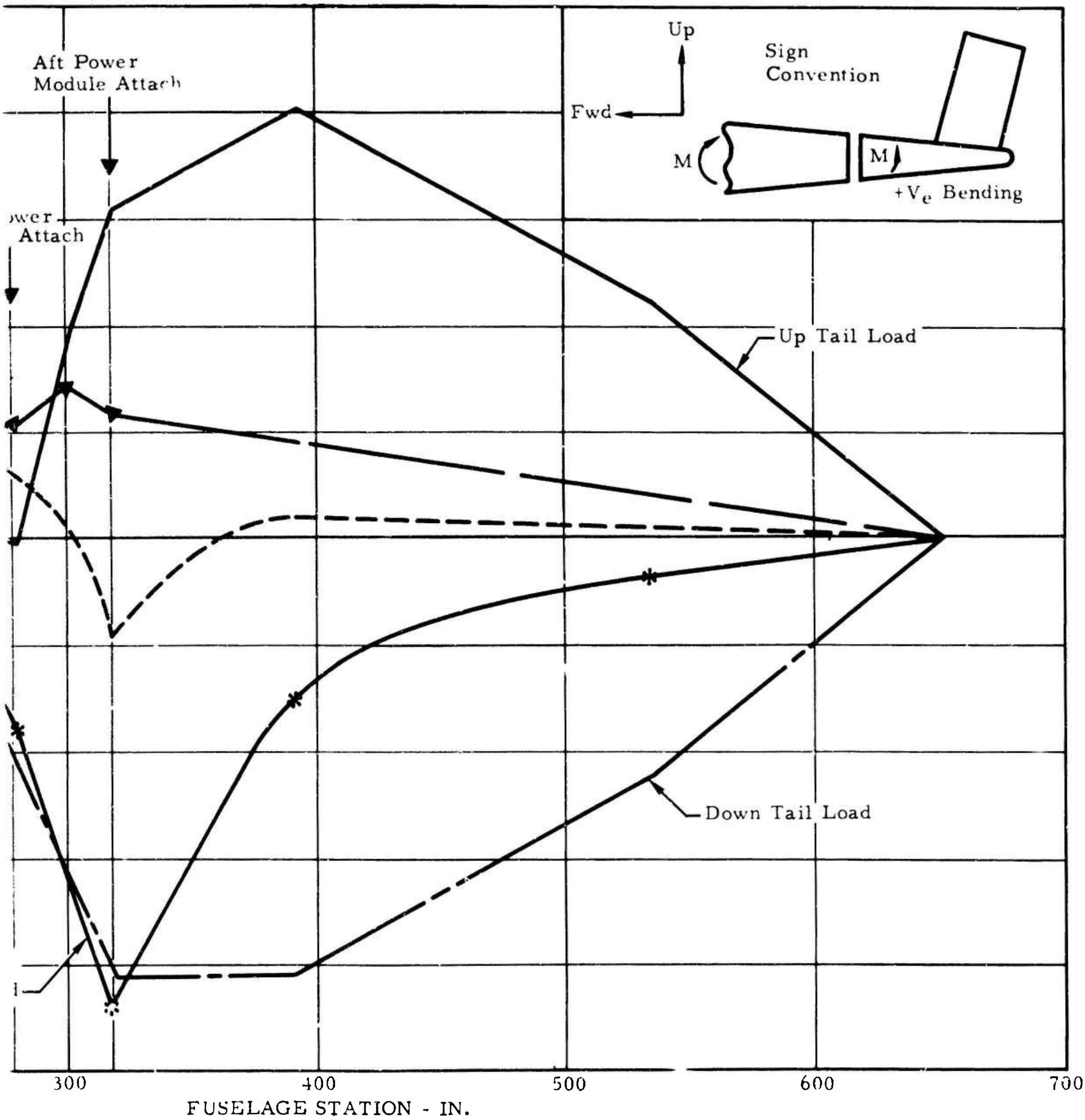
MODEL XV-9A HELICOPTER
VERTICAL FUSELAGE SHEARS
LIMIT VALUES



B


MODEL XV-9A HELICOPTER
VERTICAL FUSELAGE BENDING
MOMENTS LIMIT VALUES
BENDING MOMENT - IN. -LB X 10-3
+1,000
+800
+600
+400
+200
0
-200
-400
-600
-800
-1,000
100
200
300
400
50
FUSELAGE STATION - IN.
Aft Power
Module Attach
Fwd Power
Module Attach
Case 1 - Three Wheel Landing
Case 4 - Two Wheel
Landing With Side
Load On One Wheel
2-1/2-g Maneuver
Thrust Vector Fwd

MODEL XV-9A HELICOPTER
VERTICAL FUSELAGE BENDING
MOMENTS LIMIT VALUES



B

BASIC LOADS

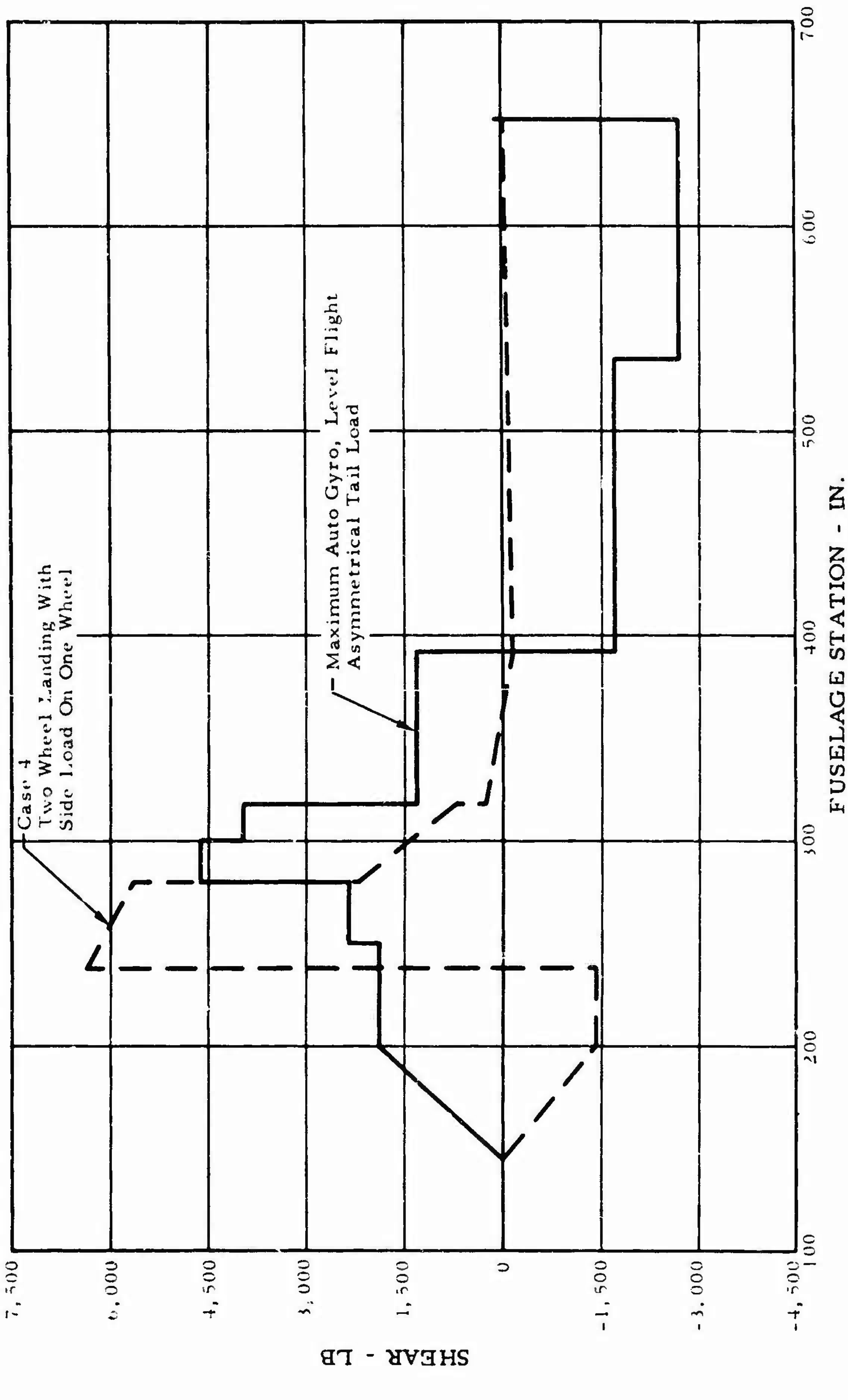
ROTOR SUPPORT STRUCTURE - LOAD SUMMARY

ALL LOADS ARE LIMIT EXCEPT CRASH CONDITIONS

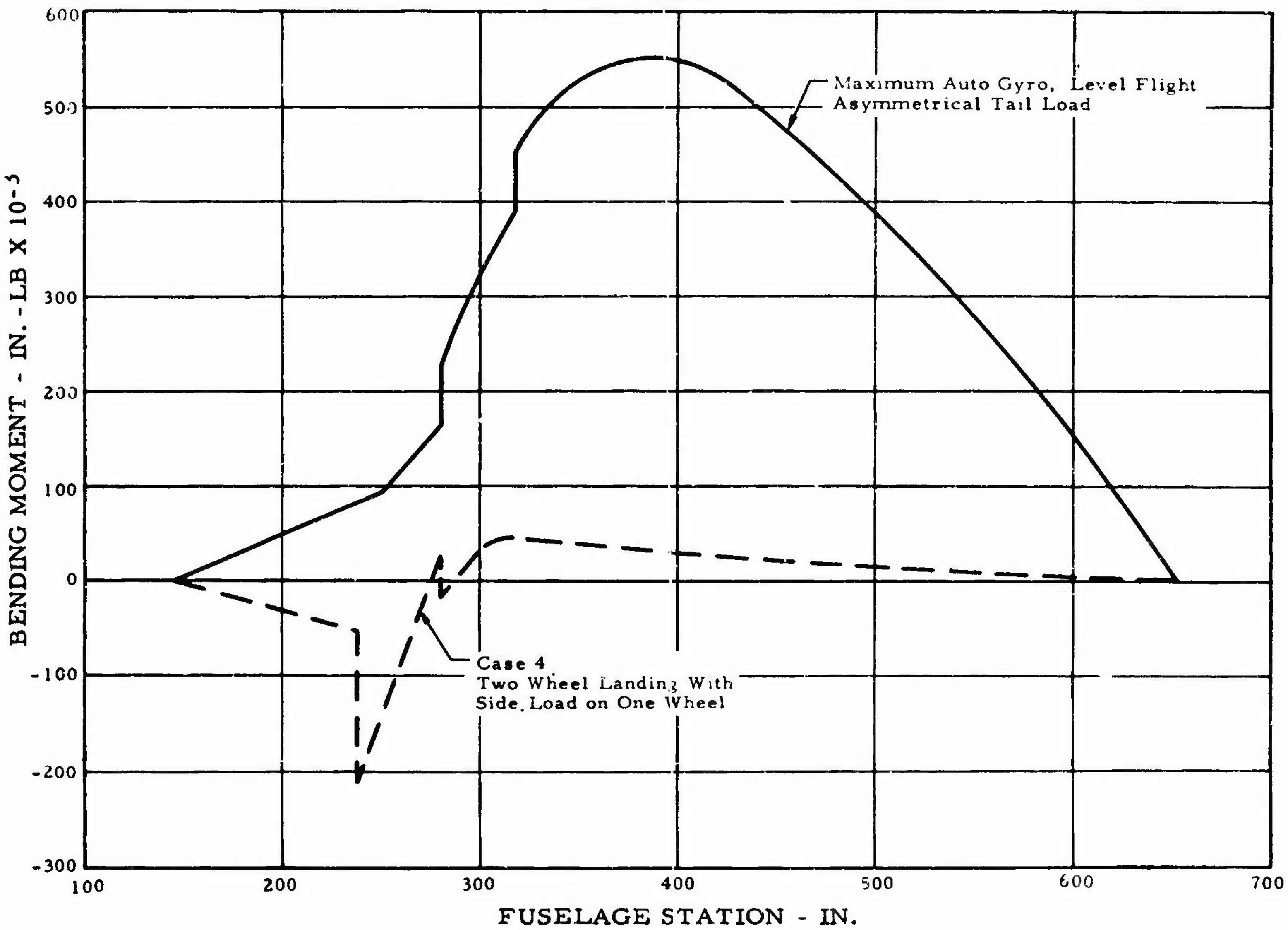
MEMBER	CASE 3	CASE 4'	CASE 5	CASE 8b	CASE 8C	CASE 10a	CASE 10b
AE	+1710	+1290	+685	+15368	0	-1820	+2600
AG	-1410	-1060	-563	-12648	0	+1495	-2135
CF	+1710	+1290	+685	+15368	0	-1820	+2600
CH	-1410	-1060	-563	-12648	0	+1495	-2135
DC	0	+1515	+3200	0	+7442	0	0
DH	0	-1515	-3200	0	-7442	0	0
BE	0	+1275	+2700	0	+6282	0	0
BF	0	-1275	-2700	0	-6282	0	0
EF	-6000	-4701	0	-8590	0	+17751	+15487
EJ	+2797	+3558	+3237	+8720	+8639	+8115	+10792
ER	+706	-66	+363	0	-5456	0	0
FK	+8472	+2269	+1826	+8720	-8639	+8115	+10792
FR	+706	+66	-363	0	+5456	0	0
GH	-3502	+1878	+360	+4105	0	+20145	+20853
GL	-3038	+7869	+4271	-10274	+10895	+6402	+3166
GM	+3010	-6352	-349	0	-4815	0	0
HM	-3020	+6352	+345	0	+4815	0	0
HN	+4617	-10401	-1090	-10274	-10895	+6402	+3166
LM	-6733	+263	-3008	+1020	+2156	+13572	-13560
MN	-4083	-5376	-3347	+1020	-2156	+13572	-13560
IJ	-159	-315	+310	-1915	-725	+7323	+6336
IK	-159	+169	+1334	-1915	+725	+7323	+6336
JS	+1700	+2805	+2051	+3975	+8128	+10573	+10971
KT	+3792	+2024	+2709	+3975	-8128	+10573	+10971
QE	-1211	-1357	-331	-14030	-2765	+11211	+7321
WG	+1310	+483	+825	+10245	-2554	+9109	+11661
FV	-1211	-333	+1833	-14030	+2765	+11211	+7321
HU	+1310	+1529	+2822	+10245	-3666	+9109	+11661
IL	+407	+95	+581	+1078	-791	+7460	+7900
IN	+407	+623	+1697	+1078	+791	+7460	+7900
JR	-429	+317	+858	+155	+3291	-9853	-9318
KR	-872	-1468	-3240	+155	-3291	-9853	-9318
I'E	-656	-753	-2277	-1640	-512	-12660	-13307
I'F	-656	-411	-1555	-1640	+512	-12660	-13307
I'G	+6	-263	-1858	+1745	-569	-11760	-10863
I'H	+6	+117	-1056	+1745	+569	-11760	-10863

CASE 3 LANDING ON MAIN WHEELS DRAG ON ONE WHEEL
CASE 4' LANDING ON MAIN WHEELS OUTBOARD LOAD ON ONE WHEEL.
CASE 5 LANDING ON ONE MAIN WHEEL WITH DRAG
CASE 8 CRASH CONDITION 8.b IS 10g FWD 8C IS 4g SIDE
CASE 10 2½g MANEUVER 10a ROTOR THRUST AFT 10b ROTOR THRUST FWD

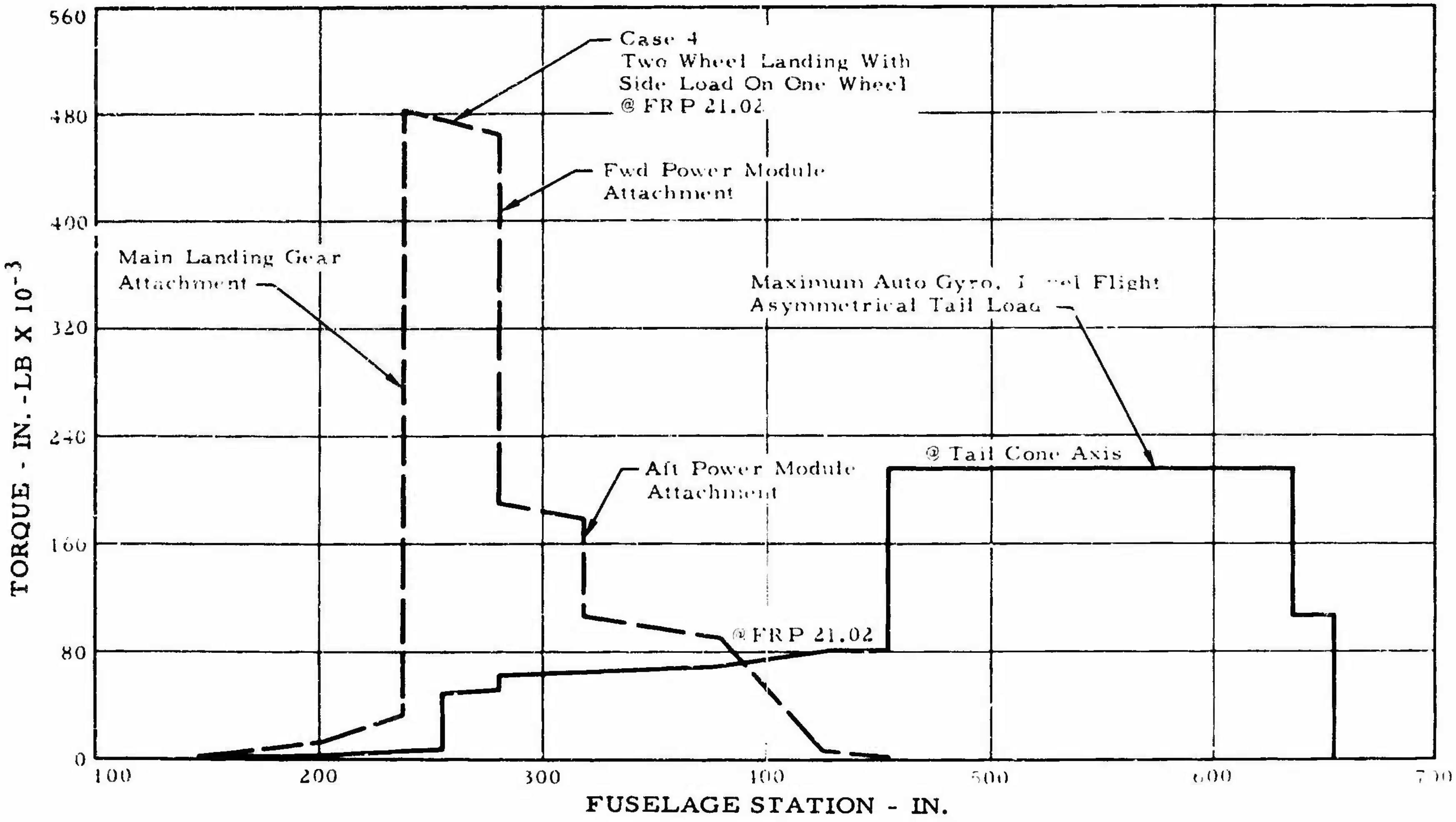
MODEL XV-9A HELICOPTER
FUSELAGE
SIDE SHEARS
LIMIT VALUES



MODEL XV-9A HELICOPTER
FUSELAGE
SIDE BENDING MOMENTS
LIMIT VALUES

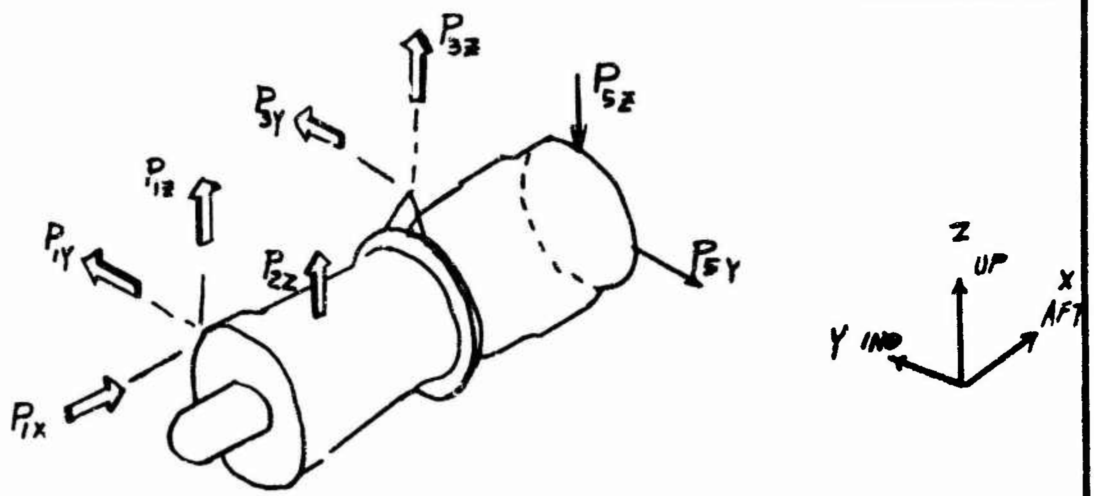


MODEL XV-9A HELICOPTER
FUSELAGE TORQUES
LIMIT VALUES



FORWARD ENG. MTG. LOADS (REVISED)

ILLUSTRATION SHOWS REACTIONS TO ENGINE LOADS
EXCEPT FOR P_{5Y} & P_{5Z} WHICH ARE LOADS FROM DIVERTER VALVE



LOADING FACTORS USED

i) Pressure Loading Only:
27 psig x 1.33 = Limit Load;
Limit Load x 1.5 = Ult. Load

ii) Pressure + inertia Loading:
27 psig x 1.00 = Limit Load;
Limit Load x 1.5 = Ult. Load

iii) Crash Condition
29 psig x 1.00 + Ult. inertia Factors

(±4 g side, 10 g dn, 10 g fwd. are considered to act separately)

REF. PAGE 273 FOR ENGINE MOUNT GEOMETRY.

ENGINE MOUNTING LOADS CRITICAL CONDITIONS

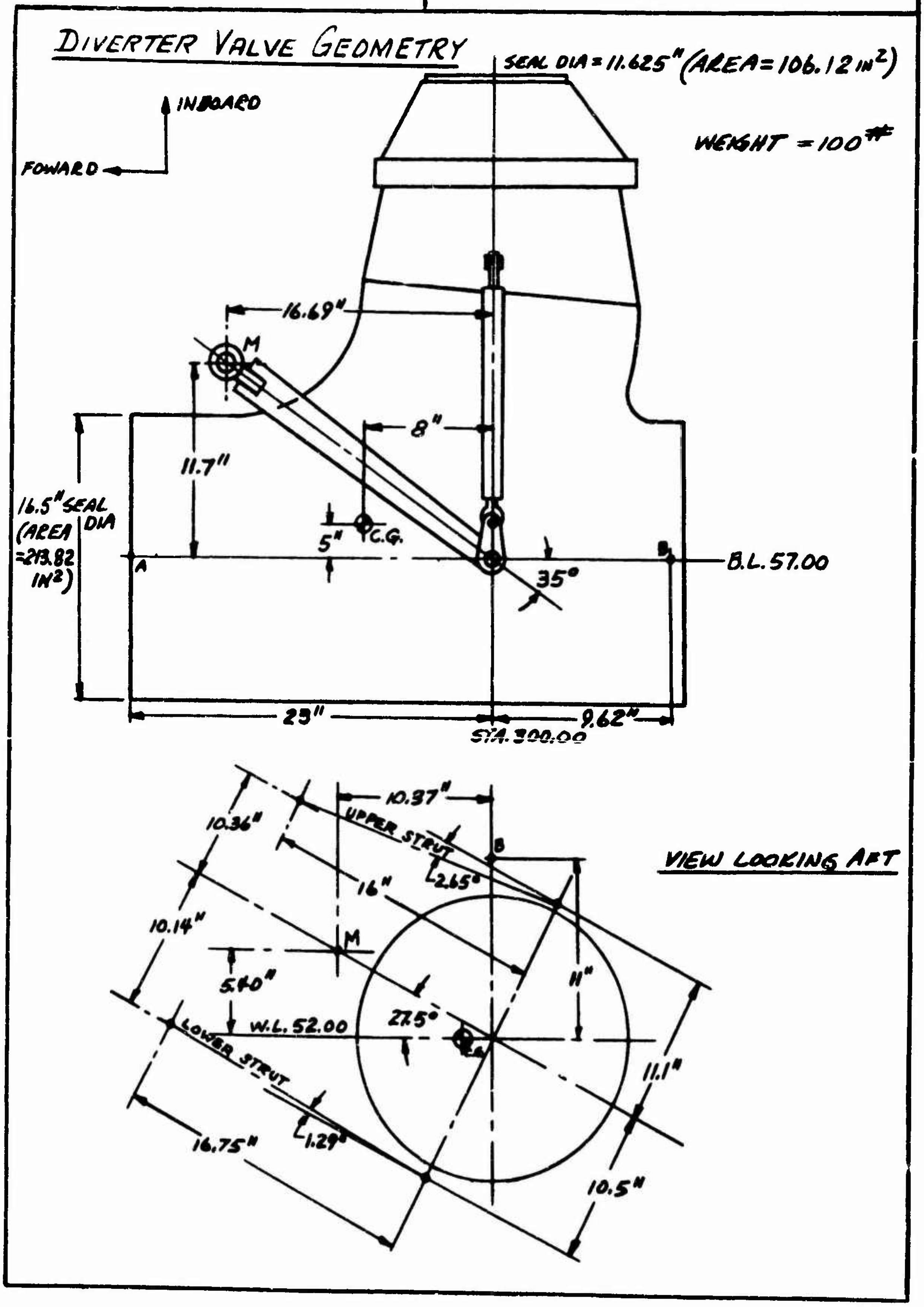
LOADS ARE LIMIT VALUES EXCEPT CRASH CONDITIONS - CRASH LOADS ARE ULTIMATE

CONDITION	P_{1X}	P_{1Y}	P_{1Z}	P_{2Z}	P_{3Y}	P_{3Z}	P_{5Z}	P_{5Y}
27(1.33)PSIG ONLY	+7820	-2695	+419	-1740	+2697	+1320	±140	±140
CRASH # 8,C	+6300	-1499	+2133	-3196	+4310	+1063		
FLIGHT # 13	+6005	-2661	+1496	-1986	+3212	+3119	±147	±100

	P_{1X}	P_{1Y}	P_{1Z}	P_{2Z}	P_{3Z}	$P_{2'X}$	$P_{2'Y}$	$P_{2'Z}$
CRASH # 8b	+5966	+2054	-1203	- 51	+2675	+7362	+2054	+1421

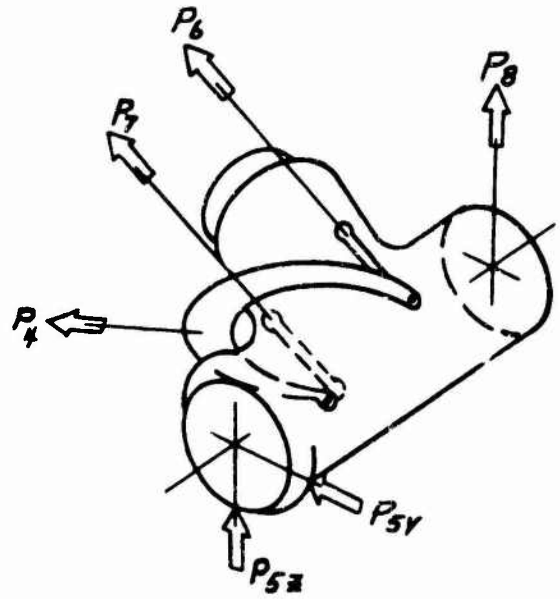
BASIC LOADS

DIVERTER VALVE GEOMETRY



BASIC LOADS

DIVERTER VALVE LOADS



LOADS SHOWN ARE POSITIVE AND ARE REACTING ENGINE LOADS.

A) VALVE CLOSED - NORMAL OPERATION (GAS DIVERTED TO ROTOR)

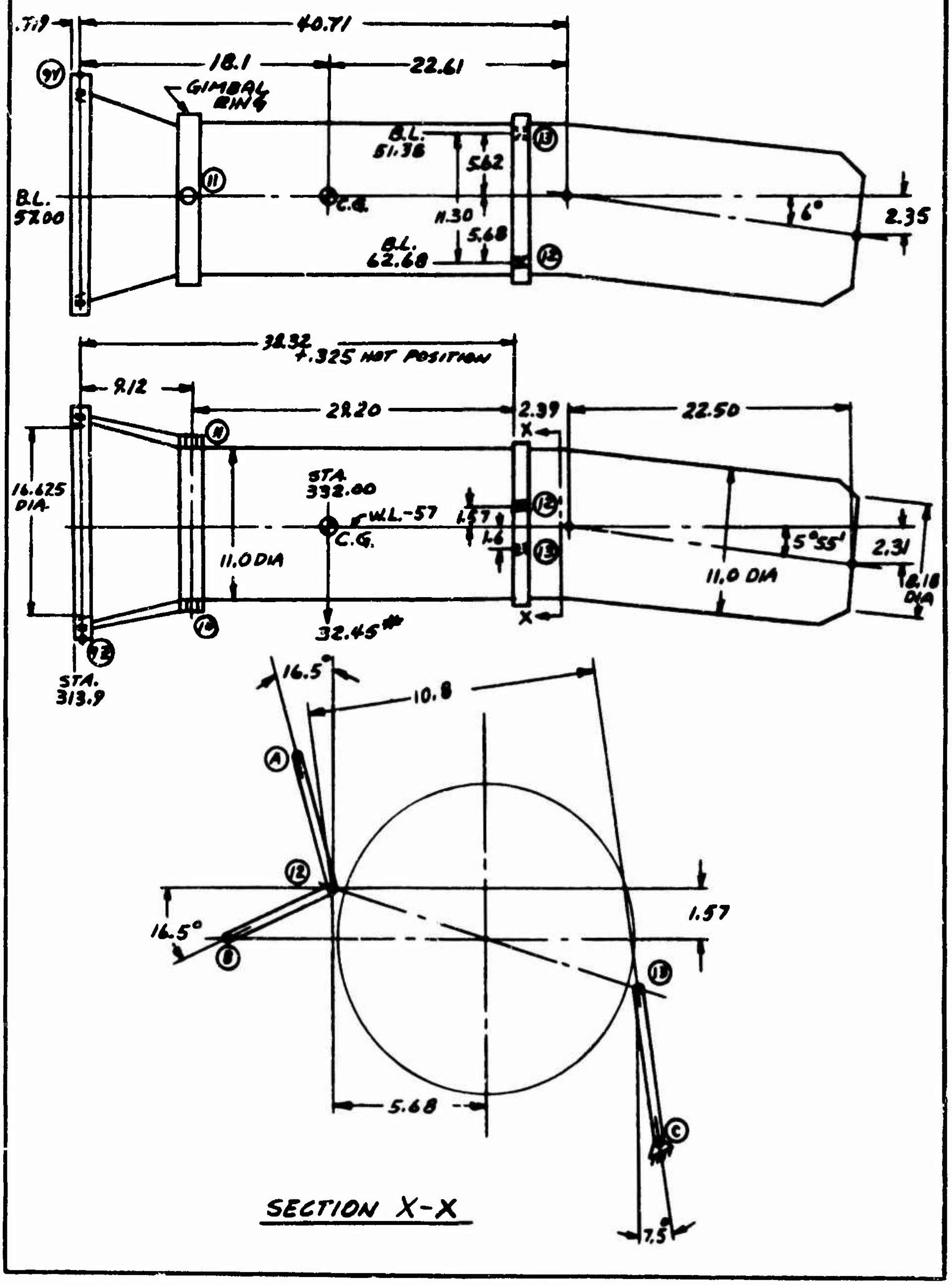
CONDITION \ LOADS #	P_4	P_{5Y}	P_{5Z}	P_6	P_7	P_8
35.91 PSIG PRESSURE ALONE (LIMIT) **	+9350	± 137	± 96	-729	-771	± 96
CRASH * (ULTIMATE)	+6340	± 251	± 618	-855	-777	+538 -174

B) VALVE CLOSED - ONE ENGINE OPERATION

CONDITION	P_4	P_{5Y}	P_{5Z}	P_6	P_7	P_8
35.91 PSIG PRESSURE + 1G DOWN FLIGHT COND. (LIMIT)	—	—	+54	+1850	+1960	+46

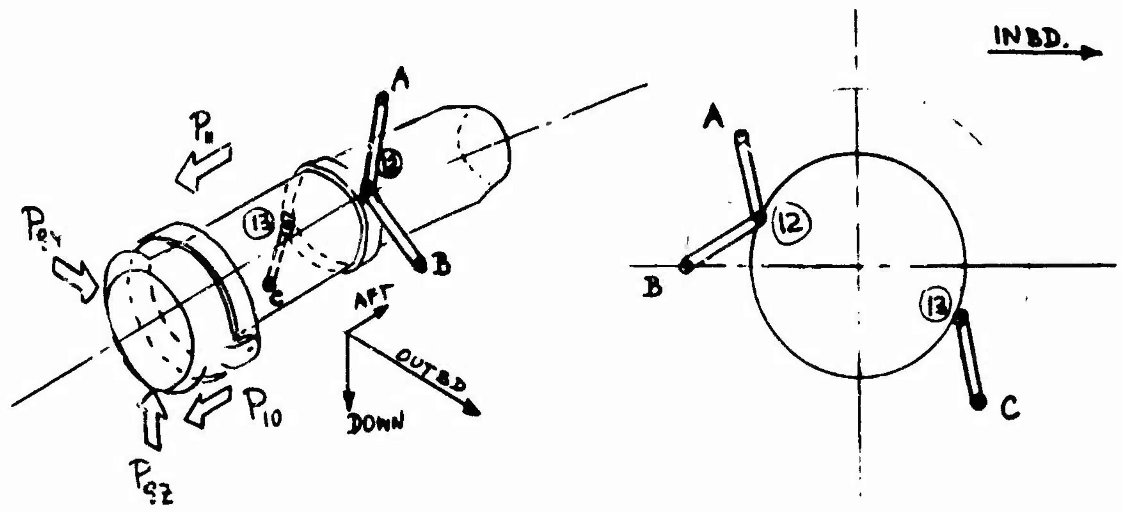
* IN CRASH CONDITION ALL LOADS SHOWN DO NOT ACT AT ONCE.
** REF. XV-9A DESIGN CRITERIA SECTION 7.

TAIL PIPE GEOMETRY (REF DWGS 385-4202 & -7106)



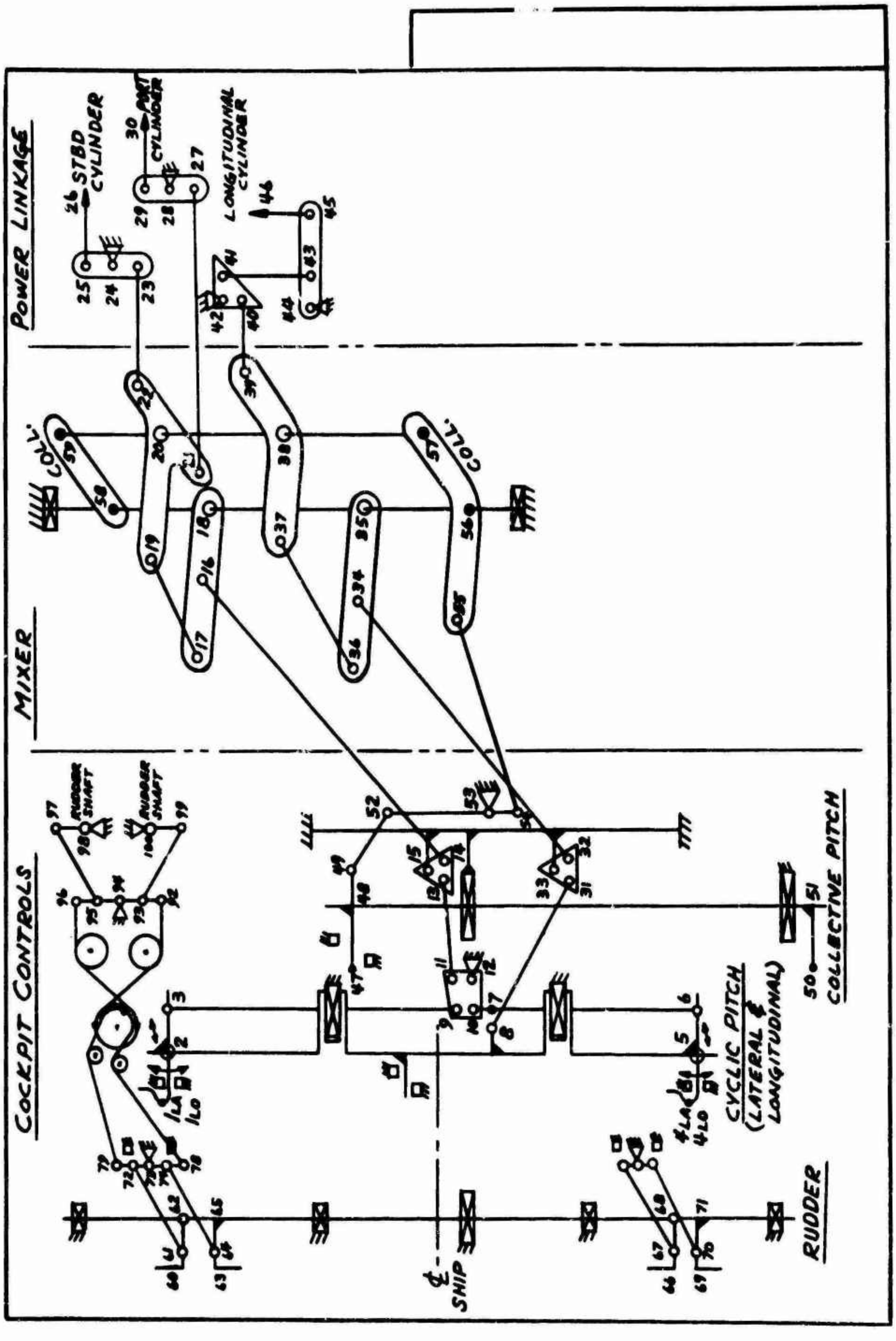
TAILPIPE MOUNTING

POSITIVE REACTIONS TO TAIL PIPE SHOWN:



SUMMARY OF REACTIONS [LB]

CONDITION	P_{9Y}	P_{9Z}	P_{10}	P_{11}	P_{12A}	P_{12B}	P_{13C}
a. 35.91 PSIG PRESSURE ALONE (LIMIT)**	- 12	+ 12	+2945	+2945	6 (c)	218 (t)	140 (t)
b. LANDING-CASE Nº 4 (ULTIMATE)	- 14	+ 111	+3327	+3327	37 (t)	235 (t)	113 (t)
c. CRASH COND. -POWER ON-	+ 62	+ 181	+2382	+2382	14 (t)	236 (t)	109 (c)
d. CRASH COND -POWER OFF-	± 72	+ 171	- 162	-162	97 (t)	87 (c)	77 (c)


POWER LINKAGE
STBD CYLINDER
PORT CYLINDER
LONGITUDINAL CYLINDER
MIXER
COLL.
COLL.
COCKPIT CONTROLS
RUDDER SHAFT
RUDDER SHAFT
CYCLIC PITCH
(LATERAL & LONGITUDINAL)
COLLECTIVE PITCH
RUDDER
℄ SHIP

COLLECTIVE CONTROL LOADS (LIMIT)

CASE No.	P_{47} #	P_{50} #	47-48 (50-51) IN.	48-49 IN.	P_{49-52} #	52-53 IN.	53-54 IN.	P_{54-56} #	55-56 IN.	56-57 IN.	P_{57-59} #
II,a	75	75	18.27	4.50	609	4.28	1.46	1,785	2.80	3.50	1,428

LATERAL CYCLIC CONTROL LOADS (LIMIT)

CASE No.	P_{1LA} #	P_{4LA} #	1LA-2 (4LA-5) IN.	2-3 (5-6) IN.	$P_{3-9}+P_{6-10}$ #	9-12 (10-12) IN.	11-12 IN.	P_{11-13} #	13-15 IN.	14-15 IN.	P_{14-16} #	16-18 IN.	17-18 IN.	P_{17-19} #
II,a	50	50	17.84	3.13	570	4.35	2.30	1,078	3.50	3.00	1,258	1.80	2.89	784

19-20 IN.	20-22 (20-21) IN.	P_{22LA} *# $-P_{21LA}$ #	$P_{20CO}/2$ **#	P_{22-23} #	P_{21-27} #	P_{22-23} #	23-24 IN.	24-25 IN.	P_{25-26} #	P_{21-27} #	27-28 IN.	28-29 IN.	P_{29-30} #
2.89	2.06	1100/2	1,428/2	1,264	164	1,264	4.30	8.60	632	164	4.30	8.60	82

STBD — PORT

* P_{22LA} = FORCE AT 22 DUE TO MOVEMENT OF LAT. STICK

** P_{20CO} = FORCE AT 20 DUE TO MOVEMENT OF COLL. STICK

LONGITUDINAL CYCLIC LOADS (LIMIT)

CASE No.	P_{1LO} #	P_{4LO} #	1LO-3 (4LO-6) IN.	7-8 IN.	P_{8-31} #	31-33 IN.	32-33 IN.	P_{32-34} #	34-35 IN.	35-36 IN.	P_{36-37} #
II,b	75	75	20.90	2.87	1,092	4.61	3.00	1,754	1.50	2.89	910

37-38 IN.	38-39 IN	P_{39-40} #	40-42 IN.	41-42 IN.	P_{41-43} #	43-44 IN.	44-45 IN.	P_{45-46} #
2.89	3.50	751	3.25	3.25	751	3.25	6.50	376

CASE II-HYDRAULIC SYSTEM ON
a) BOTH COLL. STICKS ARE PULLED BACK AND BOTH CYCLIC STICKS ARE PULLED TO THE LEFT. PORT AND STBD CYLINDERS HIT THEIR STOPS. b) BOTH CYCLIC STICKS ARE PULLED BACK AND THE LONG. CYLINDER HITS IT'S STOP. THE COLL. STICK IS HELD STATIONARY ($P_{57-59} = P_{39-40}$).

RUDDER CONTROL LOADS (LIMIT)

P_{60} #	P_{66} #	60-62 (66-68) IN	61-62 (67-68) IN	$P_{61-72}+P_{64-74}$ #	72-73 (73-79) IN	73-79 IN	P_{79-92} #	92-94 IN	94-95 IN	P_{95-97} #	97-98 IN	M98 IN#
98	98	8.95	1.88	933	1.50	3.17	441	4.95	4.00	700	3.00	1638

THIS ASSUMES ONE CONTROL SURFACE REACTS TOTAL LOAD.

HUB CONTROL LOADS (LIMIT)

THE HUB CONTROL LOADS ARE TABULATED IN REPORT No. 285-13(62-13) SECTION 5.4 WITH THE FOLLOWING ADDITIONS AND CHANGES. * POINT r IS LOCATED AT THE CENTER OF THE ROTATING SWASHPLATE.

ADDITION TO TABLE 5.4.2.3-1 (PG 5.4.2.3)	MN #	nr* IN.	r_8* IN.	P #
WEIGHTED FATIGUE COND. (LIMIT)	1,400 ±2,970	8.32	11.08	3,774

ADDITION TO TABLE 5.4.2.3-2 (PG 5.4.2.3)	MN #	nr* IN.	r_8* IN.	P #
2½G MANUEVER COND. (LIMIT)	2,580 ±4,125	8.32	11.08	6,194

ROTOR SPAR ANALYSIS

REAR SPAR STEADY & CYCLIC STRESSES

BLADE STATION	C.F. # (FIG. 65)	P = C.F. to F.S. + DUCT PRES.	A IN²	$f_c = \frac{P}{A}$ STEADY @N.A. #/in²	M_F "#" (FIG. 53)	Z IN³	$f_b = \frac{M_F}{Z}$ #/in² STEADY	f_T STEADY @EXT. FIBER #/in²	M_{CY} (FIG. 51)	$P^{\#}_{CYCLIC} = \frac{M_C}{15.38}$	$f_c = P_c/A$ CYCLIC @N.A. #/in²	f_{TC} = CYCLIC AT EXT. FIBER #/in²
90	112,000	55,200	.964	57,200	11,800	.979	12,050	69,250	146,000	±9500	±9860	±21,910
100	111,000	55,100	.964	57,200	10,500	.753	11,000	68,200	136,000	±8850	±9180	±20,180
120	106,000	51,840	.825	62,800	8,900	.855	10,400	73,200	115,000	±7480	±9070	±19,470
140	100,000	48,200	.757	63,700	8,000	.817	9,790	73,490	96,000	±6250	±8260	±18,050
160	93,000	43,980	.619	71,100	8,700	.716	12,150	83,250	79,000	±5140	±8300	±20,450
180	86,000	39,675	.619	64,200	9,500	.716	13,250	77,450	63,000	±4100	±6630	±19,880
200	78,000	35,520	.550	64,600	10,100	.652	15,500	80,100	49,000	±3180	±5780	±21,280
220	70,000	31,300	.472	66,500	10,300	.648	15,900	82,400	36,000	±2340	±4970	±20,870
240	60,000	26,830	.428	62,700	10,000	.677	14,800	77,500	25,000	±1625	±3790	±18,590
260	50,500	22,315	.338	66,000	8,800	.660	13,300	79,300	15,000	±975	±2880	±16,190
280	41,000	18000	.250	72,000	6700.	.652	10,300	82,300	8,000	±520	±2080	±12,380
300	29,500	13050	.175	74,500	3400	.662	5,150	79,650	3,000	±195	±1110	±6,260
320	16,500	7635	.076	100,500	900	.634	1400	101,900	0	0	0	0

NOTES.—
DUCT PRESSURE LOAD = 1200# REF. 29. PG. 5.2.2.10
$M_{F\,STEADY} = M_{C\,CYCLIC}$, SEC. 7.2.1.8.
15.38" = DISTANCE BETWEEN SPARS, FACTOR 41.76 REF. A 258

f_{STEADY} @N.A. $= \frac{P}{A}$ DUE TO C.F + DUCT PRESSURE
f_{STEADY} @EXTREME FIBER $= \frac{P}{A} + \frac{M_F}{Z}$
f_C CYCLIC @N.A. $= \frac{P_C}{A}$ DUE TO CHORDWISE CYCLIC BENDING
f_{TC} CYCLIC @EXTREME FIBER $= \pm \frac{P_C}{A} \pm \frac{M_F}{Z}$

APPENDIX II
STRESS ANALYSIS - ROTOR

In this appendix, an analysis is included of the rotor blade spars, the rotor blade retention straps and attachments, the tip cascade valve, the rotor hub, and the rotor shaft.

In the design of the XV-9A rotor blade spars, both stiffness and strength were design requirements. The stiffness criterion was established relative to the titanium spars used on the Model 285 Hot Cycle rotor system. The stiffness criterion required the same flapwise stiffness and required that chordwise stiffness increase by 30 percent at Station 90 and decrease linearly to the original stiffness at Station 140 and outboard.

Increased chordwise stiffness was required to reduce the response to one-per-revolution loads that were experienced in the whirl testing of the Model 285 Hot Cycle rotor system.

The blade bending moment criterion was revised to accommodate the loads measured on the whirl test. The flapwise blade bending moment curve was altered to the greater value of either 1.25 times the maximum cyclic flapwise bending moment measured in the whirl tests or the original design flapwise bending moment. Whirl test values altered the design curve only between Station 46 and Station 120. This resulted in an increase of the cyclic flapwise peak bending moment at Station 73 from 9,400 in.-lb to 14,000 in.-lb, or a 49 percent increase.

The shape of the chordwise bending moment curve was maintained, but it was made to pass through a value of 1.25 times the maximum chordwise moment measured at Station 83 on the Model 285 whirl tests. This resulted in 153,000 in.-lb/73,000 in.-lb = 2.10 times the old value, or a 110-percent increase, as the new chordwise bending moment criterion.

The stiffness requirements of the steel spar with respect to the titanium spars is expressed by the following relationships:

<u>Titanium Spars</u>

I_T = spar flapping moment of inertia at Station 90

Front spar = 1.11 in.^{4}

Rear spar = 1.17 in.^{4}

Blade = 2.28 in.^{4}

A_T = spar cross sectional area at Station 90

Front spar = 1.86 in.^{2}

Rear spar = 0.94 in.^{2}

E_T = modulus of elasticity (Reference 28, pages 5.2.2.8, 5.2.2.10, and 5.2.2.13)

= $0.87 \times 16 \times 10^6$ at 400° F

= 13.9×10^6

<u>Steel Spars</u> (Refer to page 233)

I_S = spar flapping moment of inertia at Station 90

Front spar = 0.68 in.^{4}

Rear spar = 0.79 in.^{4}

Blade = 1.47 in.^{4}

A_S = spar cross sectional area at Station 90

Front spar = 0.95 in.^{2}

Rear spar = 0.96 in.^{2}

E_S = modulus of elasticity

= $0.9 \times 29 \times 10^6$ at 400° F

= 26.1×10^6

ρ_S = radius of gyration of the steel spar

Stiffness:

(1) Flapping stiffness

Maintain the same stiffness.

$$E_T I_T = E_S I_S, \quad I_S = I_T \frac{E_T}{E_S}$$

$$I_S = I_T \frac{13.9 \times 10^6}{26.1 \times 10^6} = 0.53\, I_T$$

$$I_S = 0.53\, I_T \tag{1}$$

(2) Chordwise stiffness

Increase stiffness by 30 percent at Station 90 for the steel spar.

$$1.30\ A_T E_T \left(\frac{b}{2}\right)^2 2 = A_S E_S \left(\frac{b}{2}\right)^2\ 2$$

$$A_S = \frac{1.30\ (A_T) E_T}{E_S} = 1.30\ (A_T) \frac{13.9 \times 10^6}{26.1 \times 10^6}$$

$$A_S = 0.69\ A_T \qquad (2)$$

A comparison of actual stiffness with the theoretical stiffness is as follows:

$$\frac{I_S}{I_T} = \frac{1.47}{2.28} = 0.64 \text{ actual} > 0.53 \text{ from Equation (1)} \qquad (3)$$

$$\frac{A_S}{A_T} = \frac{0.96}{0.94} = 1.02 \text{ actual} > 0.69 \text{ from Equation (2)} \qquad (4)$$

The actual flapwise stiffness exceeds the criteria requirement ($\frac{0.64}{0.53}$ = 1.21, or 21 percent) as shown by Equation (3).

The 1962 Model 285 whirl tests showed no flapwise resonance condition, so this increase in stiffness was acceptable.

The actual chordwise stiffness exceeds the criteria requirement ($\frac{1.02}{0.69}$ = 1.48, or 48 percent) as shown by Equation (4). This increase is caused by the area required to keep the chordwise bending stresses within acceptable bounds, as there was an increase of 110 percent in chordwise bending moment. This additional area increased the chordwise stiffness that was desirable, as the 30 percent stiffness was rather a minimum stiffness requirement.

The most desirable spar cross section was next investigated. The required ρ_s of the steel spar was calculated:

$$\rho_s = \sqrt{\frac{I_S}{A_S}}$$

$$A_S = 2.10\left(\frac{4,000 \text{ psi}}{10,000 \text{ psi}}\right) A_T = 0.84\ A_T$$

based on the stress requirement for increased chordwise bending moment.

Cyclic allowable design stresses at bolt hole are as follows:

Titanium spars = 4,000 psi
Steel spars = 10,000 psi
h = depth of spar
= 3 in., front spar (refer to dwg 265-0170)
= 3.25 in., rear spar

$$\rho_s = \sqrt{\frac{0.53\, I_T}{0.84\, A_T}} = \sqrt{\frac{0.53\ (1.11)}{0.84\ (1.86)}} = 0.62;$$

$$\frac{0.62\ h}{3} = 0.20\ h, \text{ front spar}$$

$$\rho_s = \sqrt{\frac{0.53\, I_T}{0.84\, A_T}} = \sqrt{\frac{0.53\ (1.17)}{0.84\ (0.94)}} = 0.89$$

$$\frac{0.89\ h}{3.25} = 0.27\ h, \text{ rear spar}$$

Nothing is gained by using an I-cross section, as a rectangular spar cross section ($\rho/h = 0.29$) is more than adequate. The cross section chosen for the front spar was a solid trapezoidal section with the top and bottom surfaces chamfered for clearance to the leading edge fairing. The rear spar section was made a rectangular cross section.

The blade from Station 19 to Station 73 is inboard of the blade retention strap attachment. This section is not as highly loaded by flight loads as it is by ground flapping. In addition to strength, stiffness is required in this area, to minimize blade droop. The root end fittings are the primary bending material in this area. The maximum depth of bending section is limited by blade thickness and the typical cross section is an I-section, for maximum strength and stiffness per weight.

Outboard of Station 73, both a solid spar and a laminated spar configuration were studied. The laminated spar was chosen, for the following reasons:

1. These sections could be more easily fabricated from sheet stock by using a number of laminates to give the required spar area. The laminates were tapered off along the spar as permitted by strength requirements.

2. The numerous laminates in the spar also provided a fail safe feature that a solid spar would not have. A crack in any laminate does not propagate immediately to adjacent laminates. A crack can be found on inspection of the spar and corrective action taken long before danger of ultimate failure.

   The laminates are adhesive-bonded together. This stabilizes the laminates so that there can be no buckling of the laminates for the ground flapping condition, when the stabilizing centrifugal force is absent. The bonding does not eliminate the fail safe feature of the spars, as borne out by the fatigue tests.

An extensive fatigue testing program was conducted for material selection. It was decided to use AM 355 CRT corrosion-resistant steel, since it had a relatively high ultimate strength, good fatigue characteristics, and good elongation properties. The AM 355 CRT corrosion-resistant steel could be purchased in sheet stock with the desired strength properties, thus eliminating costly and impracticable heat treatment.

Selection of laminate thicknesses was based on the availability of gages and the minimum thickness to provide a reasonable number of laminates and yet not be so thin as to cause high concentrated or localized bearing stresses on the spar bolts.

The root end of the front spar is made up of two laminations each 0. 050 inch thick and 11 laminations each 0. 025 inch thick. The spar is tapered by dropping off laminations between Station 106. 75 and Station 231. 75 until the front spar is reduced to two laminations each 0. 050 inch thick and six laminations each 0. 025 inch thick.

The rear spar at the root is made up of one lamination 0. 050 inch thick and twelve laminations each 0. 025 inch thick. The spar is tapered by dropping off laminations outboard of Station 112. 75 until at the tip only the one 0. 050-inch lamination remains.

The 0.050-inch laminations are placed in the spar face next to the blade segment. This was done to provide a thicker laminate at the shear face where loads are transferred from the blade segments to the spars.

The chordwise bending stresses shown for the blade are based on higher bending moments than are shown in Section 7.2. The chordwise bending moment curve shown in Section 7.2 is 76 percent of the original curve used in design because of correction for a calibration error in the Model 285 whirl test data.

The allowable stresses used in the design of the spar are based on 80,000 psi $\pm$ 25,000 psi in the sections having no bolt holes, and 80,000 psi $\pm$ 10,000 psi in the sections having bolt holes. Steady and cyclic stresses are shown for the weighted fatigue condition. The rear spar fitting is analyzed for the ground flapping condition.

The rotor blade strap assembly is made up of twenty-two 0.025-inch thick laminations of AM 355 CRT corrosion-resistant steel. These straps connect the rotor blades to the hub, one set being attached to the front spar and one set to the rear spar. These straps are subject to axial loads from the rotor blade centrifugal force and chordwise bending and to flapwise bending from blade coning and feathering.

The rotor hub is the same hub used on the original whirl tests. The margins of safety have been revised to account for increased centrifugal forces and increased chordwise bending moment for the weighted fatigue condition.

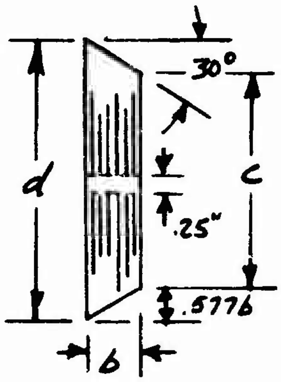
The rotor shaft has been redesigned by increasing the shaft wall thickness to increase the stiffness of the shaft. The rotor shaft supports the rotor hub through the gimbal attachment. The rotor shaft in turn is supported by an upper radial bearing and a lower thrust bearing also capable of taking lateral load.

The bolted connection originally attaching the spoke to the shaft is eliminated. The spoke and hub are positioned on the shaft by spacers, and are secured in place by the retention nut at the top of the shaft.

The shaft is subjected to axial loads from rotor thrust, from Y-duct and triduct gas pressure, and from control loads. The shaft is also subject to bending resulting from control forces and from the lateral component of rotor thrust.

ROTOR SPAR ANALYSIS

SECTION PROPERTIES   REF. DWG 385-1108



FRONT SPAR

$c = d - 1.154b \quad d = 3.0''$

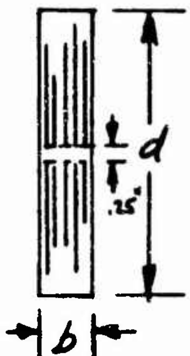
$Area = \left(\frac{d+c}{2}\right)b - A_{BOLT\ HOLE}$

$= bd - .577b^2 - \frac{b}{4}$

$= b(2.75 - .577b)$

$I = \frac{bd^3}{12} - .577b^2\left\{\frac{d}{2} - \frac{.577b}{3}\right\}^2$

$= 2.25b - .577b^2\{1.5 - .192b\}^2$



REAR SPAR

AREA =

$bd - \frac{b}{4} = b(d - .25)$

$I = \frac{bd^3}{12}$

d = 3.00" FROM STA 210

BLADE STATION	FRONT SPAR b INCHES	FRONT SPAR AREA IN^2	FRONT SPAR I IN^4	FRONT SPAR $Z = I/d/2$ IN^3	REAR SPAR b INCHES	REAR SPAR AREA IN^2	REAR SPAR I IN^4	REAR SPAR $Z = I/d/2$ IN^3
90	.375	.952	.680	.454	.350	.964	.788	.525
100	.350	.892	.643	.428	.350	.964	.788	.525
120	.325	.834	.606	.405	.300	.825	.675	.450
140	.325	.834	.606	.405	.275	.757	.619	.412
160	.300	.773	.567	.378	.225	.619	.507	.338
180	.300	.773	.567	.378	.225	.619	.507	.338
200	.275	.773	.527	.352	.200	.550	.450	.300
220	.250	.652	.487	.325	.175, d=2.95	.472	.374	.249
240	.250	.652	.487	.325	.175, d=2.70	.428	.287	.191
260	.250	.652	.487	.325	.150, d=2.50	.338	.195	.130
280	.250	.652	.487	.325	.125, d=2.25	.250	.118	.079
300	.250	.652	.487	.325	.100, d=2.00	.175	.067	.045
320	.250	.652	.487	.325	.050, d=1.78	.076	.023	.015

ROTOR SPAR ANALYSIS

FRONT SPAR STEADY & CYCLIC STRESSES

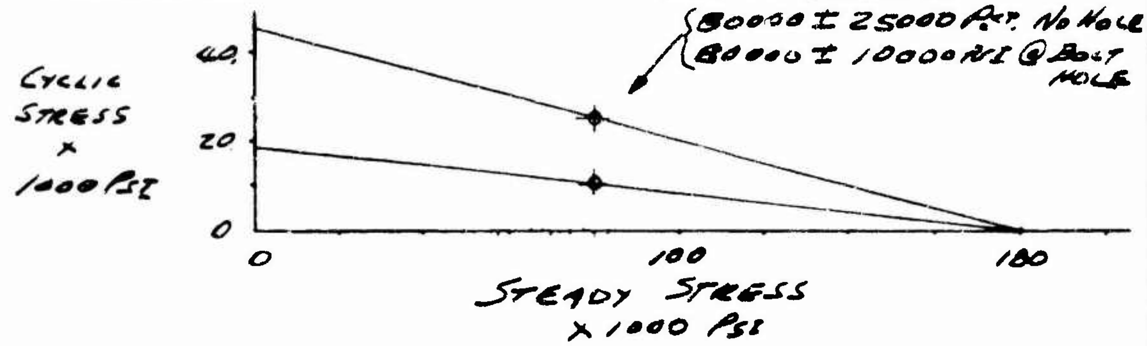
BLADE STATION	C.F. # (FIG. 65)	P = C.F. ÷ F.S. + DUCT PRESS	A IN²	$f_c = \frac{P}{A}$ STEADY @ N.A. #/IN²	M_F ## (FIG. 63)	Z IN³	$f_b = \frac{M_F}{Z}$ #/IN² STEADY	f_T, STEADY AT EXT. FIBER #/IN²	M_{CX} ÷ 76 (FIG. 51)	P_{CYCLIC} =	$f_c = P_c/A$ CYCLIC	f_{EX} = CYCLIC AT EXT. FIBER #/IN²
90	112,000	59,200	.952	62,000	11,800	.979	12,050	74,050	146000	±9500	±10000	±22,050
100	111,000	58,200	.892	65,200	10,500	.953	11,000	76,300	136,000	±8850	±9900	±20900
120	106,000	56,200	.834	67,400	8,900	.855	10,400	77,800	115,000	±7450	±9000	±19,400
140	100,000	54,200	.834	65,000	8,000	.817	9,790	74,790	96,000	±6250	±7500	±17,290
160	93,000	51,200	.773	66,200	8,700	.716	12,150	78,350	79,000	±5140	±6650	±18,800
180	86,000	49,200	.773	63,600	9,500	.716	13,250	76,850	63,000	±4100	±5300	±18,550
200	78,000	45,200	.773	58,400	10,100	.652	15,500	73,900	49,000	±3180	±4100	±19,600
220	70,000	40,200	.652	61,700	10,300	.575	18,000	79,700	36,000	±2340	±3600	±21,600
240	60,000	35,200	.652	54,000	10,000	.515	19,400	73,400	25,000	±1625	±2500	±21,900
260	50,500	30200	.652	46,400	8,800	.455	19,350	65,750	15,000	±975	±1500	±20,850
280	40,000	24200	.652	37,200	6700	.403	16,600	53,800	8,000	±520	±800	±17,400
300	29,500	18200	.652	28,000	3400	.369	9,200	37,200	3,000	±195	±300	±9,500
320	16,500	10670	.652	16,400	900	.340	2,650	19,050	0	0		±2,650

NOTES.—
DUCT PRESSURE LOAD = 1200# REF. 29, PG. 5.2.2.10
M_F STEADY = M_F CYCLIC, SEC. 7.2.1.8
15.88" = DISTANCE BETWEEN SPARS, FACTOR OF .76 SEE PG 258

f_{STEADY} @ N.A. = $\frac{P}{A}$ DUE TO C.F. + DUCT PRESSURE
f_{STEADY} @ EXTREME FIBER = $\frac{P}{A} + \frac{M_F}{Z}$
f_c CYCLIC @ N.A. = $\frac{P_c}{A}$ DUE TO CHORDWISE CYCLIC BENDING
f_{EX} CYCLIC @ EXTREME FIBER = $\pm\frac{P_c}{A} \pm \frac{M_F}{Z}$

GOODMAN DIAGRAM FOR AM-355 SPAR

MATERIAL



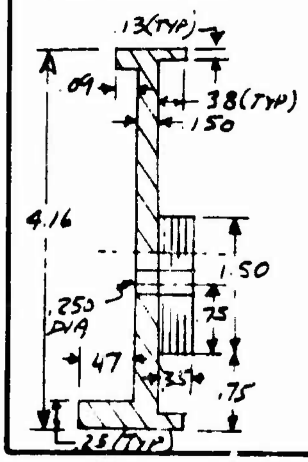
FRONT SPAR DWG No. 385-1108

STA.	LOCATION	STRESS	ALLOW. STRESS	M.S.
100	@ BOLT HOLE	65000 ± 10000 PSI	12000	+.20
120	@ BOLT HOLE	67500 ± 9000 PSI	11500	+.27
90	@ EXT. FIBER	74000 ± 22000 PSI	27500	+.25
220	@ EXT. FIBER	77500 ± 21500 PSI	26000	+.21

REAR SPAR

STA.	LOCATION	STRESS	ALLOW. STRESS	M.S.
90	@ BOLT HOLE	57000 ± 10000 PSI	13000	+.30
160	@ BOLT HOLE	71500 ± 8500 PSI	11500	+.35
90	@ EXT FIBER	69000 ± 22000 PSI	28000	+.27
220	@ EXT. FIBER	80000 ± 21000 PSI	25000	+.19
160	@ EXT. FIBER	83000 ± 20500 PSI	24500	+.20

REAR SPAR FITTING - STATION 46.34 (DWG. No. 385-1115)



FOR GROUND FLAPPING

$I = 1.929\ IN^4,\ A = 1.263\ IN^2,\ C = 2.412\ IN$

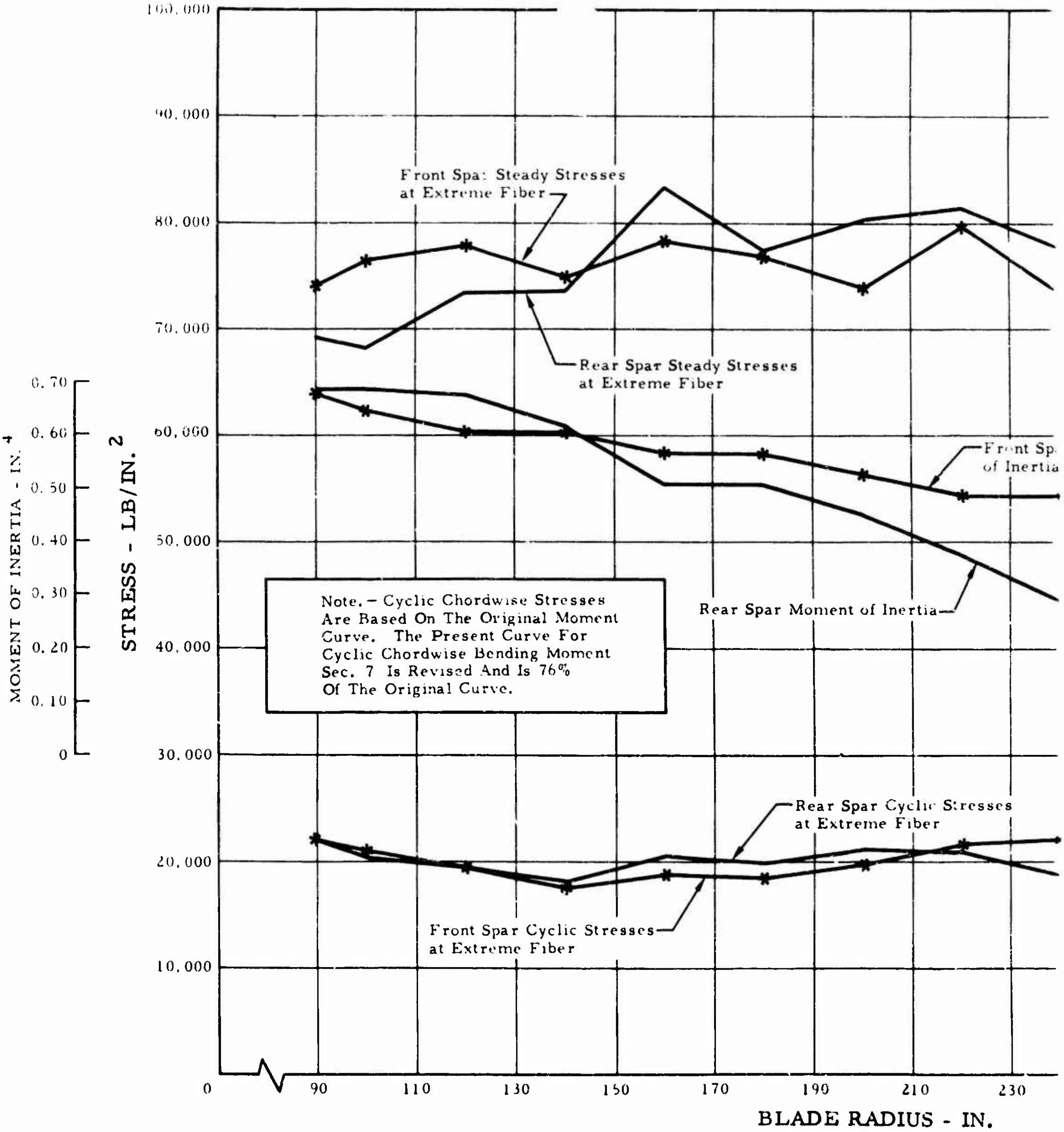
$P = 18{,}047^{\#}(ULT),\ M = 112{,}500\ IN^{\#}(ULT)$

$$f_t = \frac{MC}{I} - \frac{P}{A} = 126{,}380\ PSI$$

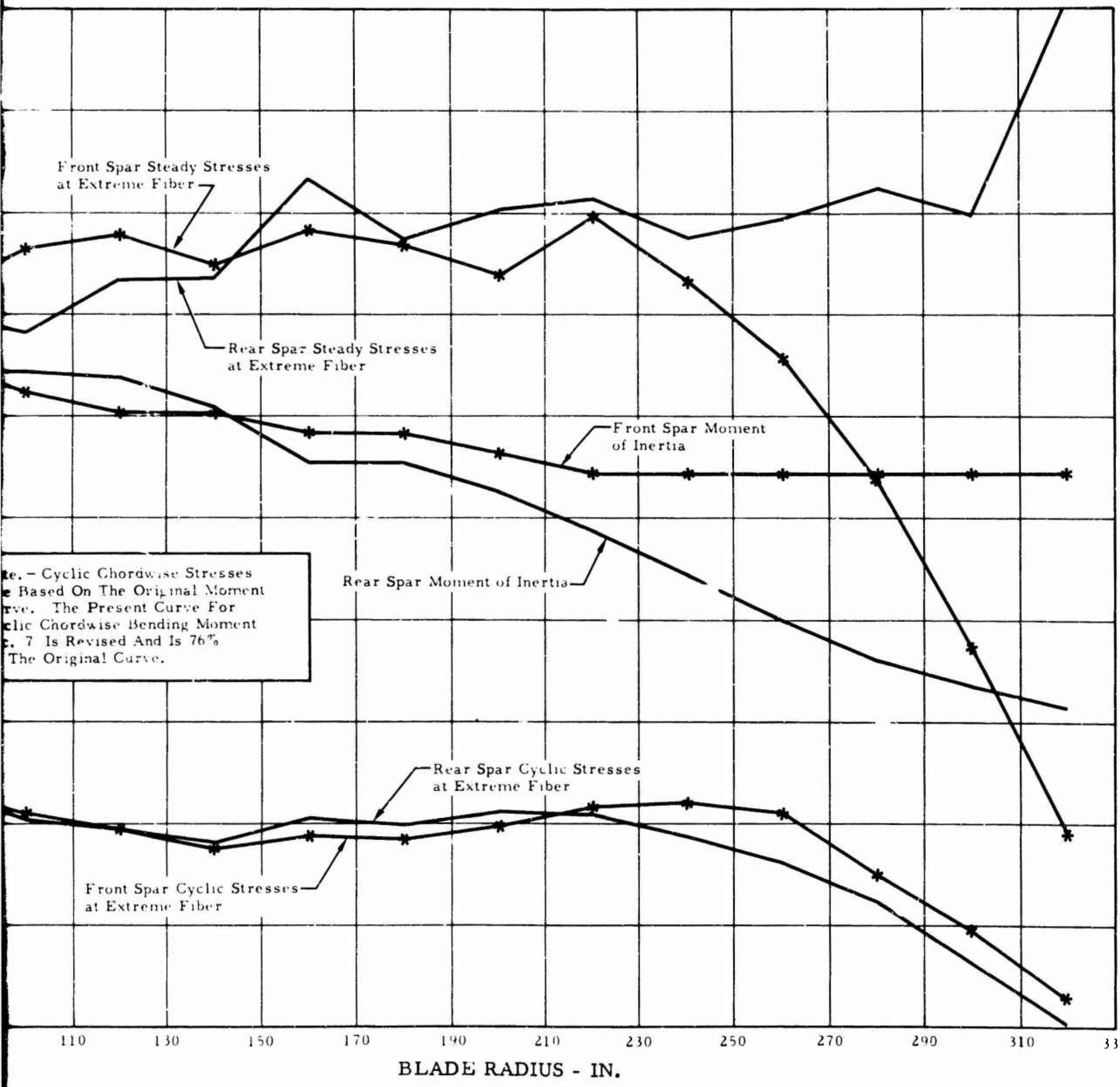
$F_{tu} = 153{,}000\ (@\ 400°F)$

$$M.S. = \frac{153000}{126380} - 1 = +.21$$

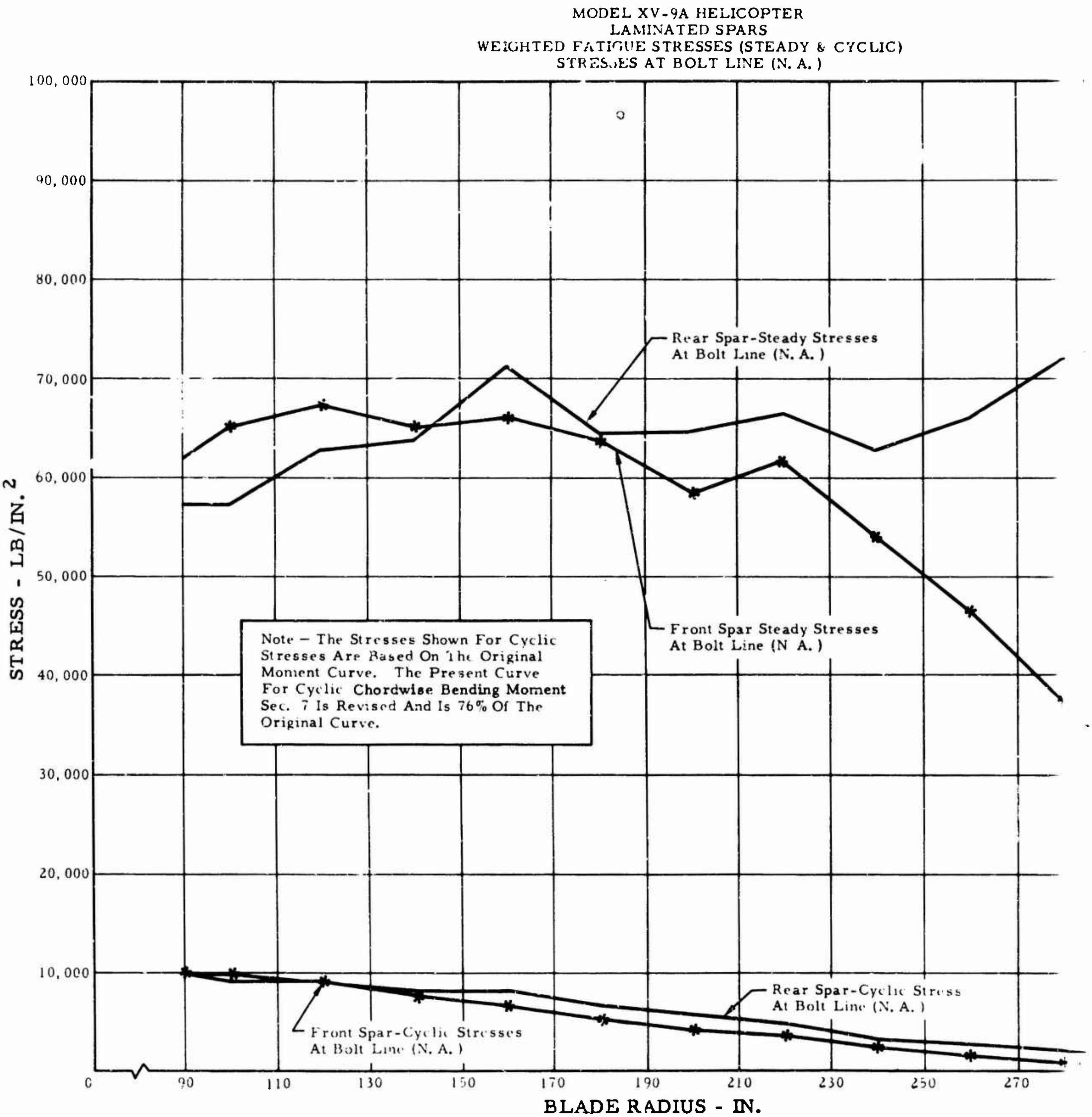
MODEL XV-9A HELICOPTER
LAMINATED SPARS
WEIGHTED FATIGUE STRESSES (STEADY AND CYCLIC)
STRESSES AT EXTREME FIBER AND FLAPPING MOMENTS OF INERTI



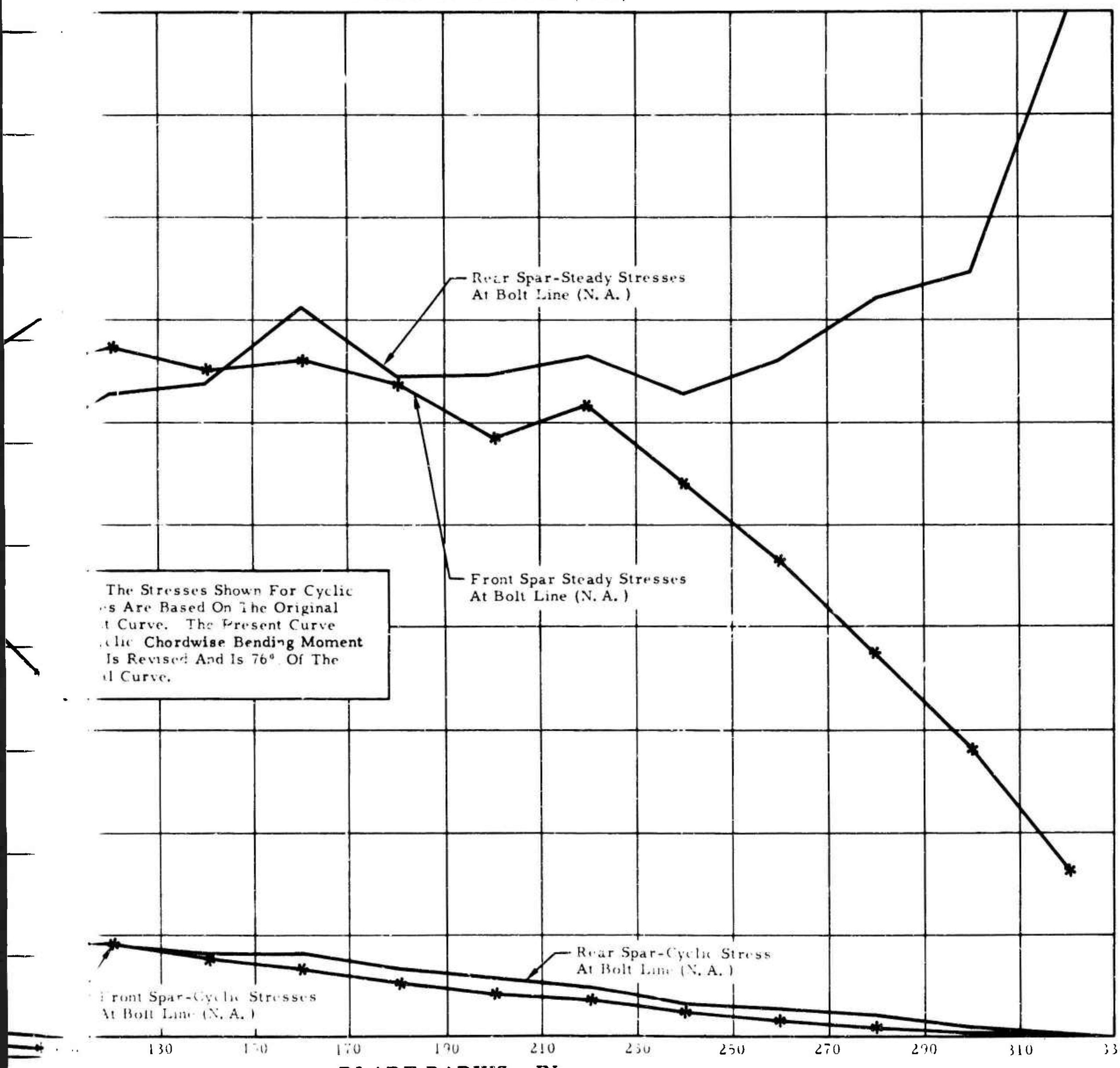
MODEL XV-9A HELICOPTER
LAMINATED SPARS
WEIGHTED FATIGUE STRESSES (STEADY AND CYCLIC)
STRESSES AT EXTREME FIBER AND FLAPPING MOMENTS OF INERTIA



B


MODEL XV-9A HELICOPTER
LAMINATED SPARS
WEIGHTED FATIGUE STRESSES (STEADY & CYCLIC)
STRESSES AT BOLT LINE (N. A.)
STRESS - LB/IN.²
100,000
90,000
80,000
70,000
60,000
50,000
40,000
30,000
20,000
10,000
0
90
110
130
150
170
190
210
230
250
270
BLADE RADIUS - IN.
Rear Spar-Steady Stresses At Bolt Line (N. A.)
Front Spar Steady Stresses At Bolt Line (N. A.)
Rear Spar-Cyclic Stress At Bolt Line (N. A.)
Front Spar-Cyclic Stresses At Bolt Line (N. A.)
Note – The Stresses Shown For Cyclic Stresses Are Based On The Original Moment Curve. The Present Curve For Cyclic Chordwise Bending Moment Sec. 7 Is Revised And Is 76% Of The Original Curve.

MODEL XV-9A HELICOPTER
LAMINATED SPARS
WEIGHTED FATIGUE STRESSES (STEADY & CYCLIC)
STRESSES AT BOLT LINE (N. A.)

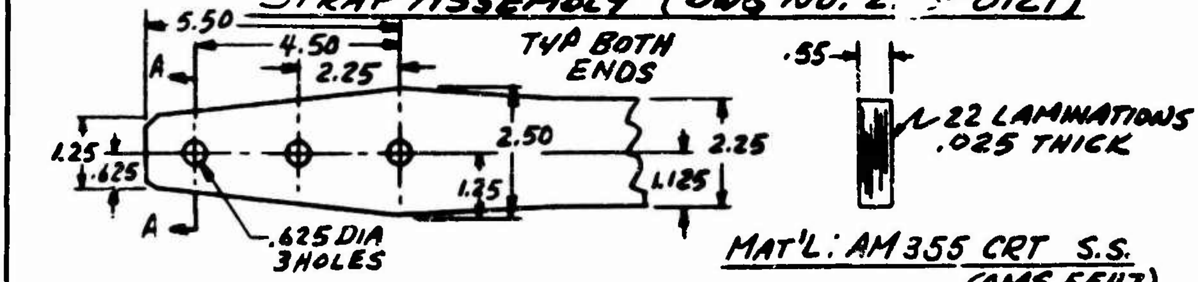


BLADE RADIUS - IN.

B

BLADE RETEN[illegible]ON STRAPS

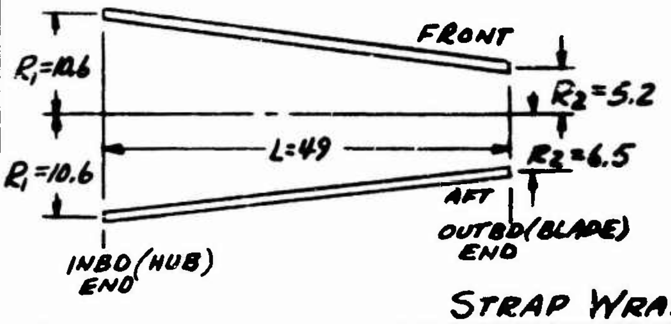
STRAP ASSEMBLY (DWG NO. 2[illegible]-0121)



MAT'L: AM355 CRT S.S. (AMS 5547)

$F_{tu} = 202{,}000$ PSI, $E = 27.2(10)^6$ PSI @ 415°F

STRAP LAYOUT



STRAP WRAP

CONDITION	PITCH CHANGE	STRAP	ϕ *	θ *	TOTAL INBOARD STRAP WRAP *	α *	γ *	TOTAL OUTBD STRAP WRAP *	NET EFFECT β
$2\frac{1}{2}$ G MANEUVER	4.4° ±13.2°	FRONT	5.6°	.47° ±1.40°	6.07° ±1.40°	4°	.95° ±2.85°	4.95° ±2.85°	1.12° ∓1.45°
		AFT	5.6°	−.58° ∓1.75°	5.02° ∓1.75°	5°	−.95° ∓2.85°	4.05° ∓2.85°	.97° ±1.10°
WEIGHTED FATIGUE	0° ±8.35°	FRONT	4.48°	±.89°	4.48° ±.89°	4°	±1.81°	4.0° ±1.81°	.48° ∓.92°
		AFT	4.48°	∓1.11°	4.48° ∓1.11°	5°	∓1.81°	5° ∓1.81°	−.52° ±.70°
OVER-REV CONDITION	−4.6° ±11.5°	FRONT	3.8°	−.49° ±1.22°	3.31° ±1.22°	4°	−1° ±2.49°	3° ±2.49°	.31° ∓1.27°
		AFT	3.8°	.61° ∓1.53°	3.19° ∓1.53°	5°	1° ∓2.49°	6° ∓2.49	−2.81 ±.96

* DEFINITION OF SYMBOLS

ϕ = CONING ANGLE

$\theta = \dfrac{\text{PITCH CHANGE} \times R_2}{L}$ (+ = WRAP ON UPPER SHOE; − = WRAP ON LOWER SHOE)

TOTAL INBD STRAP WRAP = $\phi + \theta$

α = BUILT IN STRAP WRAP − SHOE WRAP

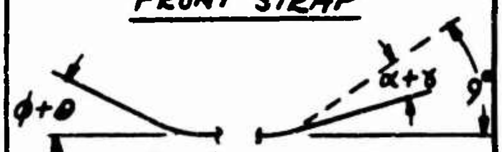
BUILT IN STRAP WRAP = 9° FRONT STRAP; 0° AFT STRAP

$\gamma = \dfrac{\text{PITCH CHANGE} \times R_1}{L}$ (+ = MORE STRAP WRAP; − = LESS STRAP WRAP)

TOTAL OUTBD STRAP WRAP = $\alpha + \gamma$

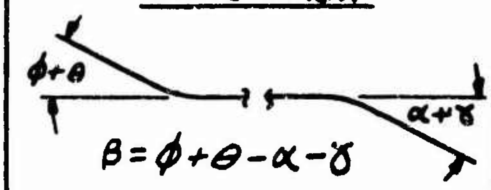
β = TOTAL STRAP WRAP

FRONT STRAP



$\beta = \phi + \theta - \alpha - \gamma$

REAR STRAP



$\beta = \phi + \theta - \alpha - \gamma$

OUTBOARD STRAP END AT TANGENT POINT
LIMIT LOADS AND STRESSES

CONDITION	P * #	STRAP	M_2 * IN.#	β RAD.	f_1 * PSI	f_2 * PSI	f_3 * PSI	f_4 * PSI	f_{TOTAL} PSI
$2\frac{1}{2}$ G MANEUVER	79,241	FRONT	11,950	.045	61,200	6,870	18,890	23,501	110,461
		AFT	12,350	.036	61,200	5,495	18,890	24,288	109,873
WEIGHTED FATIGUE	66,920 ±11,104	FRONT	2,800 ±450	.0084 ±.016	51,600 ±8,575	1,282 ±2,442	9,445 ±9,445	5,507 ±883	67,834 ±21,345
		AFT	2,910 ±460	.0091 ±.012	51,600 ±8,575	1,389 ±1,832	9,445 ±9,445	5,723 ±944	68,157 ±20,796
OVER-REV. CONDITION	98,023	FRONT	10,980	.028	75,600	4,274	18,890	21,593	120,357
		AFT	11,360	.066	75,600	10,075	18,890	22,341	126,906

* DEFINITION OF SYMBOLS

P = AXIAL TENSION IN STRAP (REF: PG 195)

M_2 = MOMENT DUE TO STRAP TWIST AT OUTBOARD END

$$P\psi_1 = \frac{1}{j}\left[\frac{M_2 - M_1 \cosh\left(\frac{L}{j}\right)}{\sinh\left(\frac{L}{j}\right)}\right] - \frac{1}{L}(M_1 - M_2)$$

$$P\psi_2 = \frac{1}{j}\left[\frac{M_1 - M_2 \cosh\left(\frac{L}{j}\right)}{\sinh\left(\frac{L}{j}\right)}\right] + \frac{1}{L}(M_1 - M_2)$$

$$j = \sqrt{\frac{EI}{P}} \;;\; \psi_1 = \frac{R_2(1-\cos W)}{L} \;;\; \psi_2 = \frac{R_1(1-\cos W)}{L}$$

W = PITCH CHANGE ANGLE

M_1 = MOMENT DUE TO STRAP TWIST AT INBOARD END

f_1 (TENSILE STRESS) $= \frac{P}{A}$; $A = 1.295\ IN^2$

f_2 (PACK BENDING STRESS) $= \frac{Et\beta}{2L}$; t = THICKNESS OF PACK = .55 IN

f_3 (LAMINATE BENDING STRESS) $= \frac{Et}{2R}$; t = THICKNESS OF LAMINA = .025 IN. ; R = SHOE RADIUS OF CURVATURE = 18.00 IN.

f_4 (STRAP TWIST STRESS) $= \frac{M_2 C}{I}$; C = 1.178 IN
I = .599 IN

WEIGHTED FATIGUE (FRONT STRAP, OUTBOARD END)

$$f_{TOTAL} = 67{,}834 \pm 21{,}345 \text{ PSI}$$

FOR $f_{STEADY} = 67{,}834$: $F_{CYC} = 27{,}535$ PSI

$$M.S. = \frac{27535}{21345} - 1 = \underline{+.29}$$

TENSION & BENDING (AFT STRAP, OUTBD END, OVER-REV COND.)

$$f_t = f_1 + f_2 = (75600 + 10075)(1.5) = 128{,}513 \text{ PSI (ULT)}$$

$$f_b = f_3 + f_4 = (18890 + 22341)(1.5) = 61{,}847 \text{ PSI (ULT)}$$

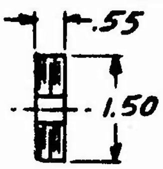
FORM FACTOR = 1.5

$$R_b = \frac{f_b}{1.5 F_{tu}} = \frac{61847}{1.5(202000)} = .204$$

$$R_t = \frac{f_t}{F_{tu}} = \frac{128513}{202000} = .639$$

$$M.S. = \frac{1}{R_b + R_t} - 1 = \underline{+.19}$$

STRAP END IN TENSION AT SECTION A-A



ASSUME BOLT TAKES 40% OF LOAD (P) AND $\frac{1}{3}$ OF PACK BENDING STRESS (f_2)

$$\underset{\text{(WEIGHTED FATIGUE)}}{f_t} = .40\frac{P}{A} + .33 f_2 = \frac{.40(66920 \pm 11104)}{.55(1.50 - .625)} + .33(1282 \pm 2442)$$

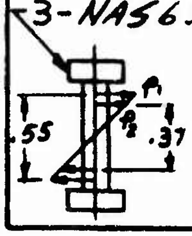
$$= 56{,}044 \pm 10{,}042 \text{ PSI}$$

(OVER-REV. COND.) $f_t = \dfrac{.40(98023)(1.50)}{.55(.875)} + .33(10075)(1.5) = 127{,}196$ PSI (ULT)

$$M.S. = \frac{202000}{127196} - 1 = \underline{+.59}$$

BOLT IN SHEAR (OVER-REV. COND.)

3-NAS630 BOLTS ON EACH END ; $P_{su} = 33{,}100^{\#}$ (SINGLE SHEAR)



$$P_2 = \left(\frac{f_2}{3}\right)\left(\frac{A}{2}\right) = \frac{10075(1.5)}{3} \times \frac{.55(.875)}{2} = 1{,}212^{\#} \text{(ULT)}$$

$$P_1 = \frac{.37 P_2}{.55} = 815^{\#}$$

$$P_S = \frac{.40(98023)(1.5)}{2} + P_1 = 30{,}222^{\#} \qquad M.S. = \frac{33100}{30222} - 1 = \underline{+.09}$$

Blade Tip Cascade

The purpose of the tip cascade is to turn the flow of the duct gases, producing the force that turns the rotor. The tip cascade has been redesigned to incorporate two movable vanes, which when placed in the closed position restrict the orifice size. These movable vanes are operated by a pneumatic actuator through a mechanical linkage, as shown in the sketch on the next page.

The static vanes are less critically loaded than the movable vanes, so only the latter are analyzed. The loads on the cascade are due to pressure and centrifugal effects. The pressure loading is derived from Section 7 of this report. Temperatures of the various components are also from Section 7 of this report. Two cases are considered for centrifugal loads:

a. Maximum rotor speed, N_R, = 295 rpm (825 g at tip)

b. Minimum rotor speed, N_R, = 225 rpm (479 g at tip)

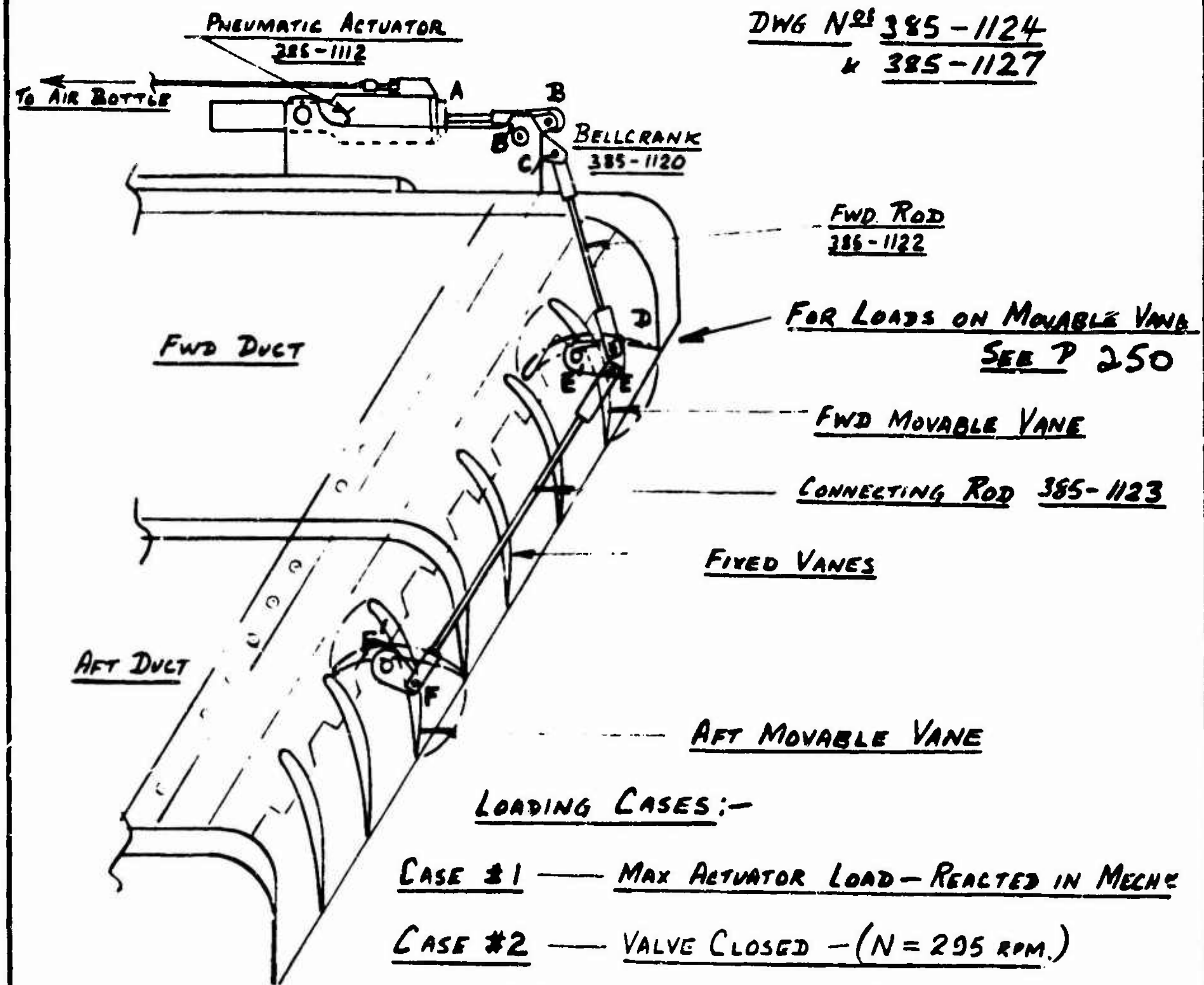
The mechanism is also designed to withstand the full actuator load with 3,000-psi pressure, assuming the linkage jammed at any point. Because this greatly exceeds the hinge moments caused by pressure and so forth on the movable vanes, this case is only carried as far as the arms that operate these vanes.

The cascade vanes and attaching structure are made from Inconel 718, heat treated after welding. Various steels, and alloy A-286 are used for the operating mechanism, the chief requirements being high strength and corrosion resistance at elevated temperatures.

CASCADE VALVE

TIP CASCADE

DWG Nos 385-1124 & 385-1127



LOADING CASES:—

CASE #1 —— MAX ACTUATOR LOAD — REACTED IN MECH^{M}

CASE #2 —— VALVE CLOSED — (N = 295 RPM.)

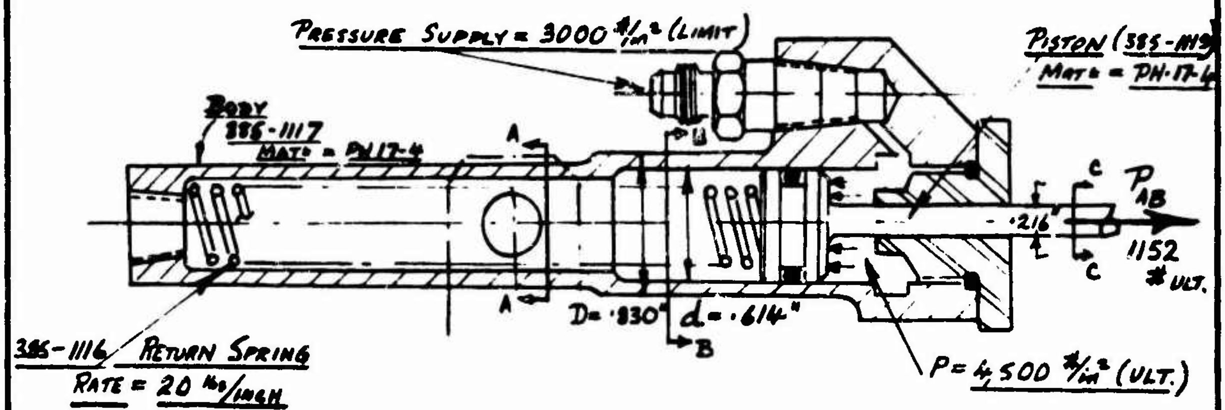
CASE #3 —— VALVE OPEN — (N = 295 RPM.)

TABLE OF LOADS IN MECHANISM:— (ULTIMATE LOADS; +Ve = TENSION)

CASE \ LOAD	P_{AB} (lbs)	$M_{B'}$ (lbs ins)	P_{CD} (lbs)	$M_{E'}$ (lbs ins)	P_{EF} (lbs)	$M_{F'}$ (lbs ins)
CASE #1	+1152	+864	+1257	+754	+1257	+754
CASE #2	+437	+328	+477	+286	+477	+286
CASE #3	0	0	0	−26	−41	−52

CASCADE VALVE

ACTUATING CYLINDER — CASCADE VALVE DWG. № 385-1112



SECTION A-A. NET A = ·050 in² P = 1152 # P/A = 23,000 #/in²

TEMP = 400°F MAT'L 17.4 CRES. BAR COND A f_{tu} = 100,000 #/in² M.S. HIGH

SECTION B-B WALL THICKNESS t_w = ·108" f = 15,400 #/in²

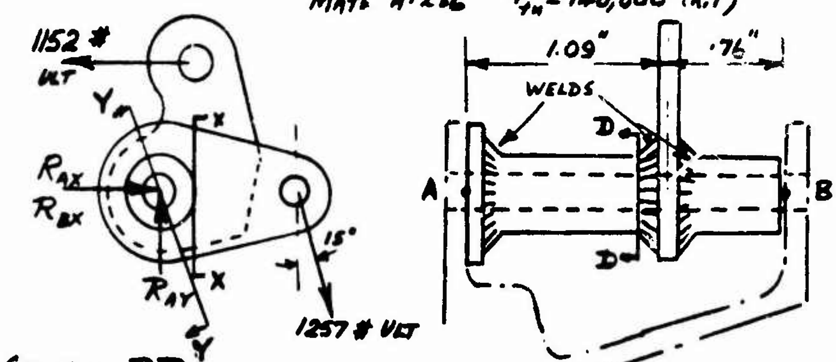
f_{tu} = 100,000 #/in² M.S. HIGH

SECTION C-C AREA = ·018 P = 1152 P/A = 64,000 #/in²

TEMP. = 400°F MAT'L PH·17·4 H925 f'_{tu} = 152,000 #/in² M.S. HIGH

BELLCRANK — DWG. № 385-1120

MAT'L A·286 f_{tu} = 140,000 (R.T)



R_{AX} = 148 #
R_{BX} = 679 # ULT.
R_{AY} = 1213 #

M_{DD} = 516 # ins

T_{DD} = 864 # ins

$Z_{BENDING}$ = ·00785 in³

$Z_{TORSION}$ = ·01570 in³

SECTION DD

BENDING f_t = 65,700 #/in²

TORSION f_s = 55,000 #/in²

COMBINED STRESSES

PRINCIPAL f_t = 96,900 #/in² @ 106,400 #/in² M.S. + ·098

PRINCIPAL f_s = 64,000 #/in² @ 69,100 #/in² M.S + ·080

TEMP = 400°F WELD EFFECT = 80%

CASCADE VALVE

BELLCRANK – (385-1120) – CONT'D

SECTION XX (ON END CRANK) $M_{XX} = 644$ # INS $P_{XX} = 325$ #

$M/Z + P/A = 5{,}300 + 93{,}300 = 98{,}600$ #/in² $f_{tu} = 106{,}400$ #/in² **M.S. + .08**

SECTION YY (ON MIDDLE CRANK) $M_{YY} = 432$ # INS $f = 60{,}000$ #/in²

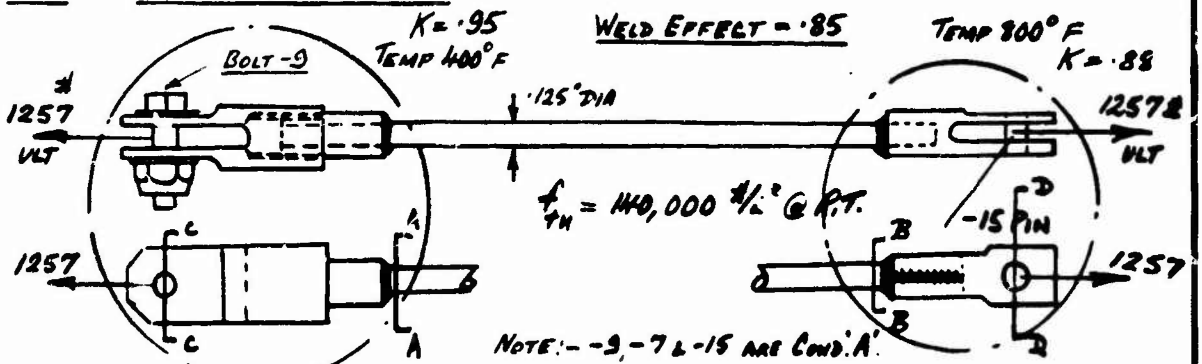
$f_{tu} = 106{,}400$ #/in² **M.S. + .77**

CENTER BOLT

MAX. SHEAR = 1450 # NAS 1003-34 STRENGTH = 2250 #

M.S. + .55

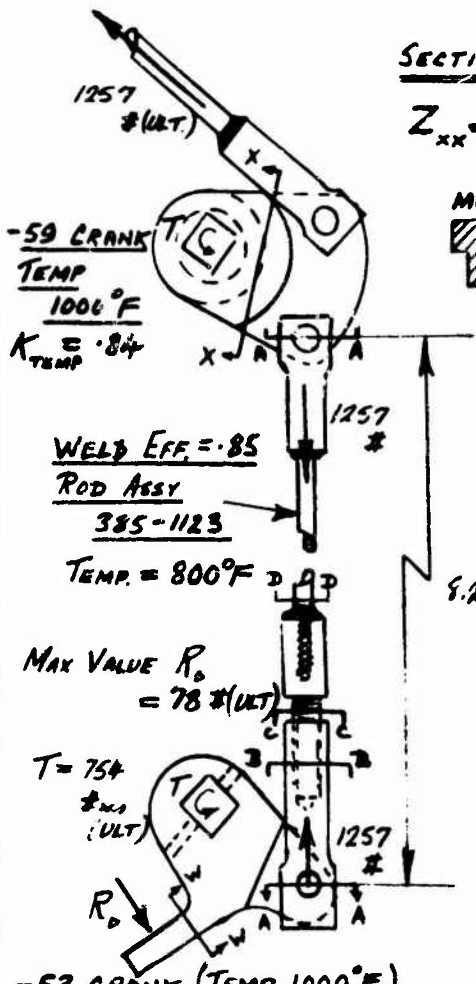
ROD – DWG. Nº 385-1122



SECTION A-A	NET A = .0123 in²	P/A = 102,200 @ 113,000	M.S. + .10
SECTION BB	NET A = .0123 in²	P/A = 102,200 @ 104,700	M.S + .02
STRENGTH OF BOLT -9	.132" DIA = 1654 # DOUBLE SHEAR		M.S + .31
WELD AT AA	.06" BEAD	69,000 #/in² SHEAR STRENGTH = 1615 # 104,700 #/in² TENSION " = 2586 #	M.S. + .28
WELD AT BB	LENGTH = .312" t = .06	SHEAR STRENGTH = 2244 @ 60,000	M.S. + .78
-15 PIN	.153" DIA. A = .0189 in²	DOUBLE SHEAR STRENGTH = 1984 @ 60,000 f_{su}	M.S + .58
SECTION CC	P = 1257 lbs A = .0263 in²	P/A = 47,800 #/in² f_{tu} = 140,000 × .95 = 133,000	M.S. HIGH
SECTION DD	P = 1257 A = .015	P/A = 83,800 #/in² f_{tu} = 140000 × .95 × .85 = 113,000	M.S. + .35

CASCADE VALVE

CONNECTING ROD & OPERATING CRANKS   DWG Nº 385-1124



SECTION XX $M_{xx} = 503$ # ins. $P_{xx} = 943$ # (Comp)

$Z_{xx} = \cdot 006 \text{ in}^3$ $A_{xx} = \cdot 052 \text{ in}^2$ $f_{xx} = 101{,}900$ #/in²

MAT'L INCONEL 718 HMC @ ·1075 $T_{ty} = 140{,}000$ (RT) = 117,600 #/in² @ 1000°F

·075"

Y Y

M.S. + ·15

SECTION YY TORSIONAL $f_s = 35{,}000$ #/in²

ALLOWABLE $f_s = 76{,}000$ #/in²   M.S. HIGH

SECTION WW $M_{ww} = 15.6$ # ins.

$Z = \cdot 000432 \text{ in}^3$ $f = 36{,}100$ #/in²   M.S. HIGH

S.25' ALLOWABLE $f = \cdot 84 \times 140{,}000 = 117{,}600$ #/in²

SECTION ZZ $M_{zz} = 377$ # ins $P_{zz} = 792$ lb

$Z = \cdot 0031 \text{ in}^3$ $M/Z = 121{,}600$ #/in²   M.S + .23

$A = \cdot 038 \text{ in}^2$ $P/A = 20{,}800$

$f = 142{,}400$ #/in²

Z Z

PLASTIC BENDING FACTOR 1.50

∴ $f_{ALLOW} = 1.5 \times 117{,}600 = 176{,}000$ #/in²

ROD ASSY 385-1123 PIN STRENGTH = 1984 lbs. DOUBLE SHEAR   M.S. + ·58

SECTION AA $P = 1257$ lbs $A = \cdot 0156 \text{ in}^2$ $f_t = 80{,}576$ #/in²   M.S + ·24

TEMP = 950°F $K = \cdot 84$ ∴ $f_{tu} = 100{,}000$ #/in²

$K_w = \cdot 85$

SECTION BB $P = 1257$ lbs $A = \cdot 0413$ $f_t = 30{,}400$ #/in²   M.S. HIGH

ALLOWABLE $f_{tu} = 100{,}000$

SECTION CC $P = 1257$ lbs STRENGTH OF THREAD = 1842 lbs.   MS + ·46

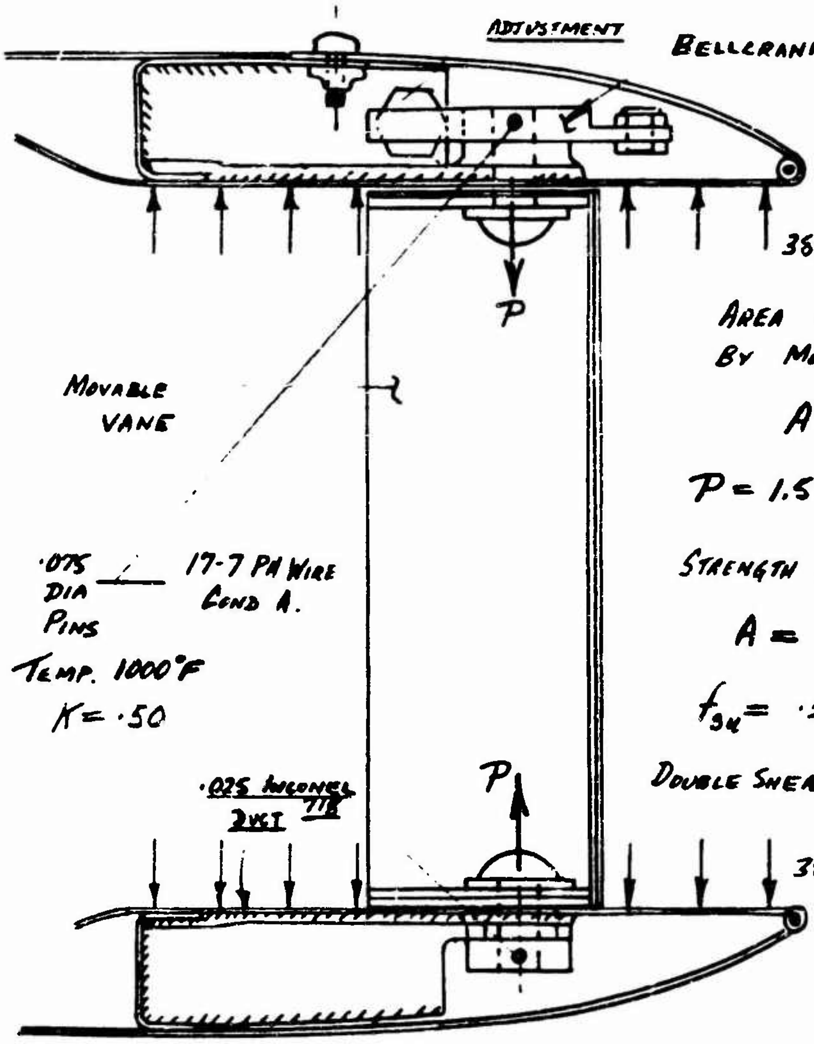
WELD STRENGTH (BOTH ENDS) LOAD = 1257 lbs STRENGTH = 2196 lbs @ 60,000 #/in² SHEAR   M.S + ·75

SECTION DD $P = 1257$ lbs $A = \cdot 0123 \text{ in}^2$ $f_t = 102{,}200$ #/in²   M.S. + .02

TEMP = 800°F $K = \cdot 88$ $f_{tu} = 140{,}000 \times \cdot 88 \times \cdot 85 = 104{,}700$ #/in²

WELD EFF. ·85

MOVABLE CASCADE VANES (AFT. VANE SHOWN)   DWG. № 385 – 1124



AREA OF DUCT TIED DOWN BY MOVABLE VANE

$A = 2 \times 3 = 6 \text{ in}^2$

$P = 1.5 \times 6 \times 38 = 342\#$

STRENGTH OF .075 DIA — 17-7

$A = .0044179$

$f_{su} = .50 \times 100,000 = 50,000$

DOUBLE SHEAR $= 2 \times .0044 \times 50,000 = 440\#$

M.S. +.29

.025" DUCT.   LIMIT PRESSURE = 38 psi.   ULT. PRESSURE = 57 psi

PANEL SIZE $b = 1.75$, $a = 3.00$, $t = .025"$ | MAX OPER. TEMP = 1183°F

TEMP. RED. FACTORS. $f_{tu} = 95,000 \times .66 = 62,700\ \#/\text{in}^2$

$a/b = 1.71$ $E = 31 \times 10^6 \times .74 = 22.94 \times 10^6$

* $\frac{P}{E}\left(\frac{b}{t}\right)^4 = 59.62$ FROM WHICH $\frac{\sigma_1}{E}\left(\frac{b}{t}\right)^2 = 12.0$ $\sigma = 58,800\ \#/\text{in}^2$

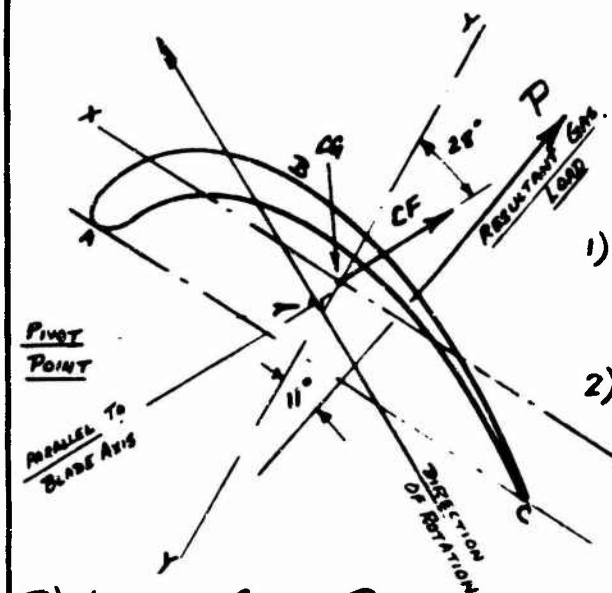
M.S. +.066

* REF 29 PG. 224

MOVABLE CASCADE VALVE

DWG N° 385 - 1124

A) VANE IN OPEN POSITION



CF = CENTRIFUGAL FORCE

= 265 lbs. (LIMIT) (N = 295 rpm)

& 180 lbs (LIMIT) (N = 243 RPM)

P = 167 lbs (LIMIT) REF.

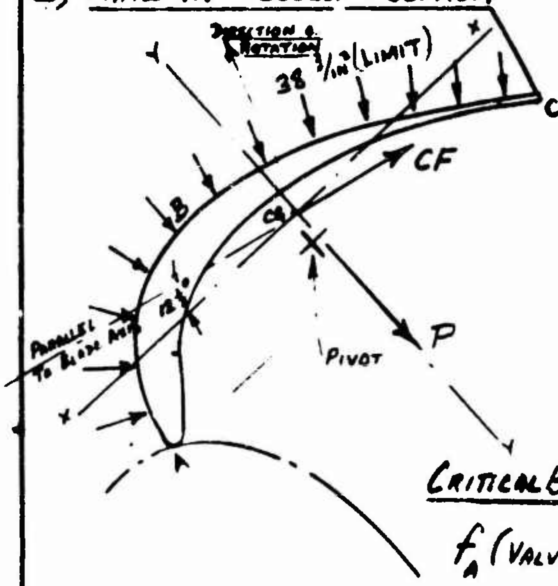
1) N = 295 RPM $\quad M_{XX}$ = 366 lbs. ins (ULT)

M_{YY} = 134 lbs ins (ULT)

2) N = 243 RPM $\quad M_{XX}$ = 174 lbs ins (LIM)

M_{YY} = 63 lbs. ins (LIM)

B) VANE IN CLOSED POSITION



CF = CENTRIFUGAL FORCE

= 265 x 1.5 = 398 lbs (ULT)

P = PRESSURE LOAD

= 456 x 1.5 = 684 lbs (ULT)

M_{XX} = 415 #ins $\quad M_{YY}$ = 209 #ins

SECTION ON MID SPAN OF VANE

CRITICAL BENDING STRESSES :— (ULTIMATE)

f_A (VALVE OPEN) = 18,000 #/in² COMP.

f_B (VALVE CLOSED) = 15,200 #/in² COMP

f_C (VALVE OPEN) = 15,200 #/in² COMP $\quad$ M.S. + 1.60

FLANGE CRIPPLING (TEMP 1100°F) @ 39,500 #/in²

CREEP RUPTURE (VALVE OPEN) $\quad f_B$ = 6,400 #/in² $\quad$ M.S. HIGH

$f_{ALLOWABLE}$ = 50,000 #/in² @ 1100°F

ROTOR HUB

REVISED MARGINS OF SAFETY — HUB STRUCTURE

NEW LOADS AS PER SECTION I – BASIC LOADS

REFERENCE:— REPORT Nº 285-13 (62-13) "HUB & CONTROL SYSTEM ANALYSIS" VOL III (REF. 30) MARCH 1962

BASIC ASSY Nº 285-0511 — HUB STRUCTURE

CRITICAL SECTIONS ARE LISTED BY PART Nº, TITLE, AND PAGE NUMBER IN THE ABOVE REPORT.

CHANGES TO LOADS:— CENTRIFUGAL LOADS INCREASED (HEAVIER BLADES)
INCREASED CHORDWISE BM. (WEIGHTED FATIGUE)

a) LOWER HUB PLATES (285-0564 & 0565)

SECTION B-B – PAGE 5-3-2-3-0 THRU 5-3-2-4-1

(PREVIOUS ANALYSIS REVISED TO ACCOUNT FOR REDUNDANCY OF LOAD PATH)

i) OVER-REV CONDITION (295 RPM) M.S. WAS:- NOT QUOTED *M.S. IS:- +.06

ii) $2\frac{1}{2}$G MANEUVER M.S WAS:- +.44 *M.S. IS:- +.02

iii) WEIGHTED FATIGUE **CRITICAL STRESSES WERE:— 40,000 ± 5540 lbs/in²
" " ARE:— 33,400 ± 14,300 "

NOTE * THIS STRUCTURE IS REDUNDANT, AND THE STRESSES WERE MEASURED BY A STRAIN GAUGE DURING WHIRL TESTING. THE MARGINS OF SAFETY WERE THEN BASED ON EXTRAPOLATED VALUES.
** THE WEIGHTED FATIGUE STRESS LEVELS ARE BASED ON CALCULATION & ARE GIVEN FOR COMPARISON PURPOSES WITH OLD VALUES. THE WHIRL TESTS INDICATE $f = 65,000 \pm 6,000$ #/in²

b) 285-0562 FITTING

LOWER BOLT ATTCH – PAGE 5-3-2-8-1 M.S. WAS:- +.13
(REVISED ANALYSIS →) M.S NOW:— > +1.00

UPPER BOLT ATTCH – PAGE 5-3-2-8-1 M.S. WAS:— +.05
(REVISED ANALYSIS →) M.S. NOW:— > +1.00

ATTCH TO WEB — PAGE 5-3-2-8-1 M.S. WAS:— +.97
M.S NOW:— +.81

REVISED MARGINS OF SAFETY — HUB STRUCTURE (CONTINUED)

c) 285-0529 FITTING PAGE 5-3-2-12-2

$2\frac{1}{2}$G MANEUVER CONDITION — NO CHANGE TO M.S.

d) 285-0532 FEATHERING BEARING RING

INCREASED LOADS $2\frac{1}{2}$G GROUND FLAPPING DUE TO HEAVIER BLADES

PAGE 5-3-2-13-1 - STRENGTH OF RING — M.S. WAS:- +1.08 M.S. IS:- +.18

" 5-3-2-14-1 — BOLT GROUP — M.S. WAS:- +.21 M.S. IS:- +.03

e) 286-0527 TRUNNION

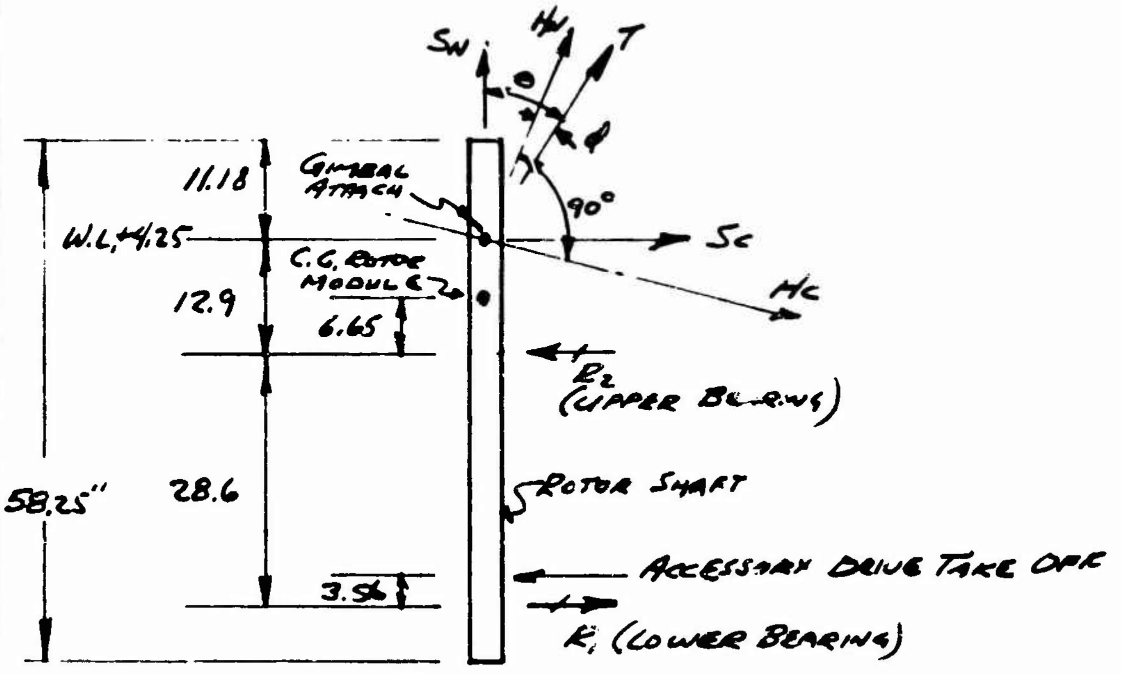
PAGE 5-3-4-6-2 — SECTION C-C — $2\frac{1}{2}$G MANEUVER — NO CHANGE

f) 285-0528 GIMBAL RING

PAGE 5-3-4-7-3 - SECTION AA — $2\frac{1}{2}$G MANEUVER — NO CHANGE

ROTOR SHAFT LIFT & CRASH LOADS

REF. DWG 385-1200
DWG 285-0517 E.O. #3 SHEET 2/2



$R_V = S_N$ (ALL VERTICAL LOADS ARE REACTED AT LOWER BEARING)

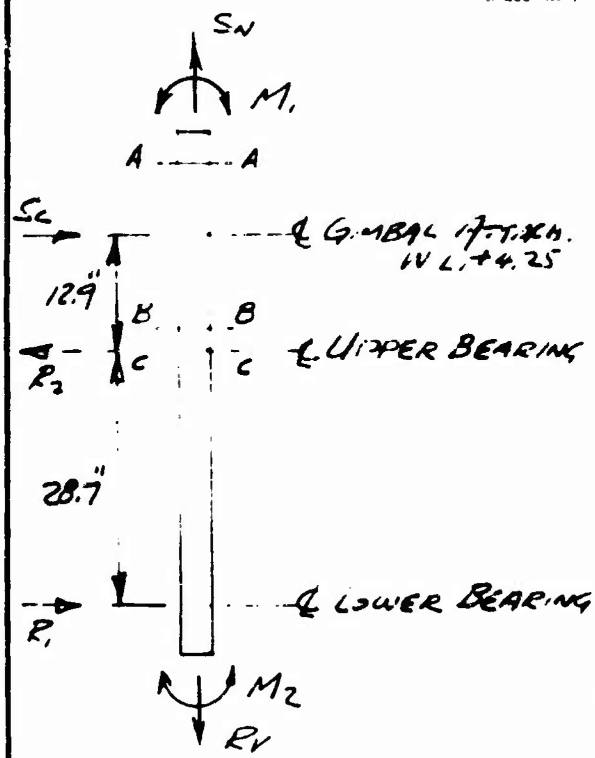
LIMIT VALUES —

CONDITION	θ (1)	φ (1)	T (1)	ROTOR INERTIA	HUB LOADS H_N	HUB LOADS H_C (1)	SHAFT LOADS S_N	SHAFT LOADS S_C	SHAFT LOADS R_1	SHAFT LOADS R_2
2½ g MANEUVER	10°	2°	38200	8425	38200	±1332	29175	±6640	±2985	±9625
WEIGHTED	6°	1°	15300	3369	15300	±4170	11831	±5502	±2480	±7982
FATIGUE	4°	1°	22950	5050	22950	±4170	17850	±5360	±2415	±7775
2½ g GROUND FLAPPING	—	—	—	—	—	—	—	25850	7000	7000
10g CRASH FWD (ULT)	—	—	—	33690	—	—	—	33690 @ C.G.	7840	41530
10g CRASH DN (ULT)	—	—	—	33690	—	—	33690	—	—	—
4g CRASH SIDE (ULT)	—	—	—	13500	—	—	—	13500 @ C.G.	3140	16640
ACCESSORY DRIVE (2)	—	—	—	—	—	—	—	2000 @ DRIVE	1750	250

(1) REF SECTION 7.
(2) 17 H.P AT SHOCK FACTOR OF 3 LIMIT

MAIN ROTOR SHAFT DWG 285-0517 E.O. #3

MATERIAL - 4340 STL. H.T. 160/180 KSI



LIMIT LOADS

COND	WEIGHTED FATIGUE	$2\frac{1}{2}$G MANEUVER
M_1	42390"#	52290"# *
M_2	39400"#	48750"# *
S_N	11831#	29175#
S_C	±5502#	±6640#
R_1	±2480	±2985
R_2	±7982	±9625

* REFERENCE 30 Pg. 5.3.4.10

SECTION AA

$D_O = 5.301$

$D_I = 5.000$

$$A = \frac{\pi(\overline{5.301}^2 - \overline{5.000}^2)}{4} = 2.40\ \text{IN}^2$$

$$t = \frac{5.301 - 5.000}{2} = .150 \qquad I = \pi R^3 t = \pi(2.575)^3 .150 = 8.07$$

WITH THE RETAINING NUT TORQUED TO PRELOAD THE SHAFT, ONLY THE ULTIMATE BENDING CONDITION IS ANALYZED.

$2\frac{1}{2}$ G MAN. COND - ULT.

$$f_t = \frac{(52290)(1.5)(2.65)}{8.07} + \frac{(29175)(1.5)}{2.40} = 43950\ \text{PSI}$$

$$M.S. = \frac{160000}{43950} - 1 = \underline{\underline{+2.64}}$$

WEIGHTED FATIGUE COND.-

NO FATIGUE CHECK IS MADE SINCE THE RETAINING NUT IS TORQUED TO PRELOAD THE SHAFT TO 60000#

MAIN ROTOR SHAFT DWG 285-0517 E.O. #3

SECTION BB

$D_o = 5.490$ $\quad A = \frac{\pi(5.490^2 - 5.000^2)}{4} = 4.05\ IN^2$

$D_i = 5.000$

$t = .245$ $\quad I = \pi r^3 t = \pi(2.622)^3 .245 = 13.90\ IN^4$

WITH THE RETAINING NUT TORQUED TO PRELOAD THE SHAFT ONLY THE ULTIMATE BENDING CONDITION IS ANALYSED.

$2\frac{1}{2}$ G MAN. COND. ULT.

$M_{BB} = 52290 + 6640^{\#}(10.4) = 121290\ ''\#$

$f_t = \frac{(121290)1.5(2.75)}{13.09} + \frac{29175(1.5)}{4.05} = 39000\ PSI$

$M.S. = \frac{160000}{39000} - 1 = +3.1$

WEIGHTED FATIGUE CONDITION

NO FATIGUE CHECK IS MADE SINCE THE RETAINING NUT IS TORQUED TO PRELOAD THE SHAFT TO 60000 #

SECTION CC.

$D_o = 5.750$ $\quad A = \frac{\pi(5.750^2 - 5.000^2)}{4} = 6.28\ IN^2$

$D_i = 5.000$

$t = .375$ $\quad I = \pi r^3 t = \pi(2.68)^3 .375 = 22.7\ IN^4$

$M_{CC} = 42390 + 5502^{\#}(12.9) = 113390\ ''\#$ WEIGHTED FATIGUE CONDITION

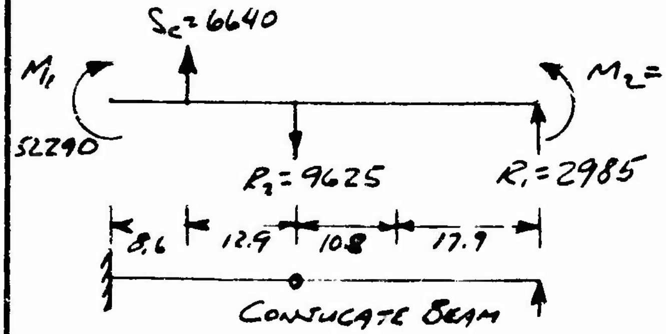
$f_t = \frac{(113390)(2.875)}{22.7} = 14400\ PSI$

$F_a = 35000\ PSI$ (REF FIG 2.8.1 REF. 20)

$M.S. = \frac{35000}{14400} - 1 = +1.43$

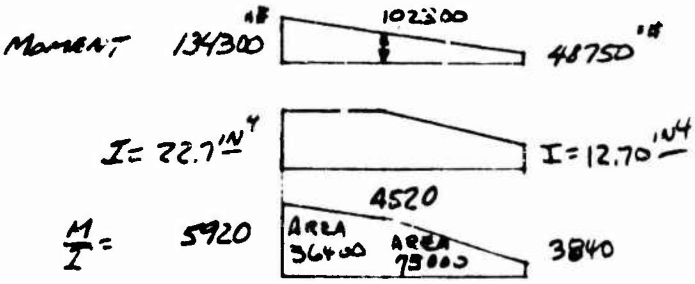
MAIN ROTOR SHAFT DWG 285-0517 E.O. #3

MAIN ROTOR SHAFT DEFLECTIONS



$2\frac{1}{2}$ G MANEUVER COND.

$$\theta_{R_2} = \frac{(\text{AREA } \frac{M}{I} \text{ LOADING})\bar{X}}{EI}$$



(AREA OF $\frac{M}{I}$ LOADING)X

$56400(23.5) = 1325000$
$75000(9.2) = 690000$
2015000

$$\frac{2015000}{29 \times 10^6 \times 28.7} = .00242 \text{ RADIANS}$$

M @ 3.75" ABOVE $R_2 = 109650$"

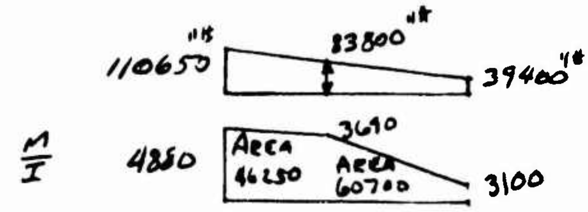
$\Delta\theta$ DUE TO ECCENTRICITY OF UPPER BEARING

$\Delta L = 3.75$" $M_{AV} = \frac{134300+109650}{2} = 122000$

$$\Delta\theta = \frac{122000}{(22.7)} \times \frac{3.75}{29 \times 10^6} = .00070 \text{ RADIANS}$$

TOTAL ROTATION $= .00312$ RADIANS
$= .18°$

WEIGHTED FATIGUE



$46250(23.5) = 1085000$
$60700(9.2) = 558000$
1643000

$$\frac{1643000}{29 \times 10^6 \times 28.7} = .00197 \text{ RADIANS}$$

$R_1 = 2480$

$R_2 = 7982$

M @ 3.75" ABOVE $R_2 = 87900$

$\Delta\theta$ DUE TO ECCENTRICITY OF UPPER BEARING

$\Delta L = 3.75$

$M_{AV} = \frac{110650+89900}{2} = 100300$ "#

$$\Delta\theta = \frac{100300}{(22.7)} \times \frac{3.75}{29 \times 10^6} = .00057 \text{ RADIANS}$$

TOTAL ROTATION $= .00254$ RADIANS $= .15°$

STRESS ANALYSIS - POWER MODULE

The power module is a unit that is attached to the fuselage by four bolts. It is composed of the supporting structure for the rotor shaft and lateral pylons that support the right and left nacelles and the engines. The landing gear oleo strut attaches to a fitting on the nacelle main frame located in the plane of the front spar of the lateral pylon.

The tubular structure supporting the rotor is 4130 steel tubing; the lateral pylon is made up of 2024 aluminum; the nacelle structure is A-286 heat-resistant steel, except for the lower stressed cover between the front and rear spar, which is 2024 aluminum; and the engine mounts, which are 4130 steel.

The lower tube members supporting the rotor thrust bearing are aligned to carry the thrust loads directly into the fittings attaching to the fuselage. Additional upper members have been added to the lower thrust bearing support. These members have been added to increase the rigidity of this support.

The upper rotor shaft bearing resists only lateral load. The structure supporting this bearing is mounted on top of the front and rear pylon spars. Loads applied here must travel down the center power module truss work to be reacted at the fuselage attachments.

The lateral pylon is composed of a front and rear spar with web stiffeners and extruded angles for caps and a top and bottom stressed skin between spars. A rigid bulkhead closes off the pylon at BL 22. 0. This is also the station at which the attachments to the fuselage are located. There is also a bulkhead closing off the pylon where it joins the nacelle.

The lateral pylon carries the shears, moments, and torques from the nacelles to the fuselage attachments, where they are reacted.

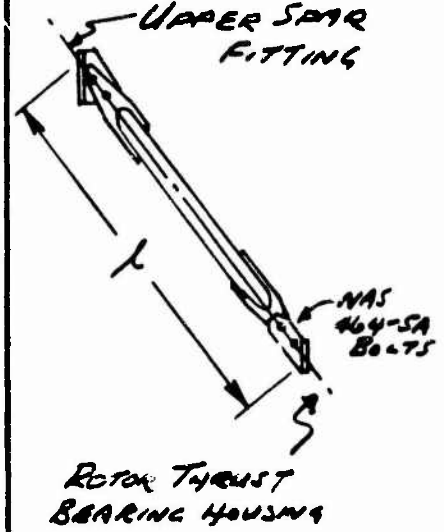
The engine is supported by a tubular engine mount cantilevered ahead of the front lateral pylon spar.

There are two forward and one aft support points for mounting each engine. The forward inboard support is capable of taking loads in all three directions. The forward outboard support is capable of taking only vertical load. The aft engine support is capable of taking vertical and side loads.

The engine and engine mount are covered by nonstructural aluminum cowling.

LOWER ROTOR BEARING SUPPORT LOADS

DWG 385-5033 TO 385-5037 & 39



MEMBER	LENGTH IN.	LOAD IN MEMBER # CASE 10a	CASE 10b
I'E	21.60	-12660	-13307
I'F	21.60	-12660	-13307
I'G	18.70	-11760	-10863
I'H	18.70	-11760	-10863

REF PAGE 210 & 215

I'E & I'F ARE CRITICAL MEMBERS
TUBE IS $1\frac{3}{8}$ DIA, .120 4130 STL, TUBES COND. N.

$A = .567\,in^2$ $\rho = .534\,in$

$\frac{L}{\rho} = \frac{21.60}{.534} = 40.5$ $F_c = 67000$ PSI $\frac{P}{A} = \frac{13307 \times 1.5}{.567} = 35200$ PSI

$M.S. = \frac{67000}{35200} - 1 = \underline{\underline{+.90}}$

BOLTS - NAS 464-5A (DOUBLE SHEAR)

ALLOWABLE SHEAR = 4(7300) = 29200 #

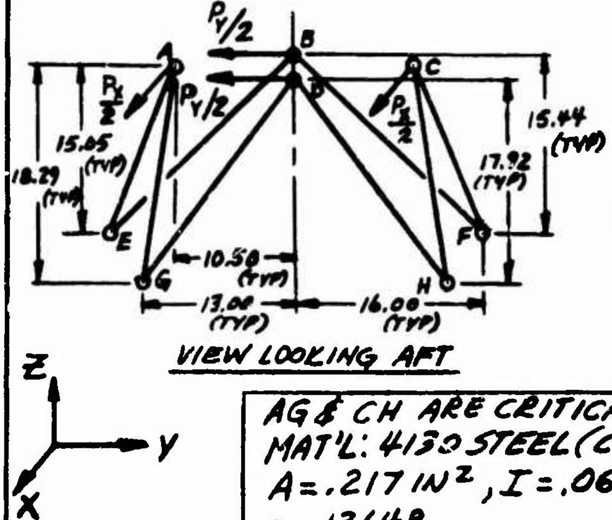
$M.S. = \frac{29200\#}{13307 \times 1.5} = \underline{\underline{+.46}}$

NOTE - THE ANALYSIS FOR IJ, IK, IL, AND IN ON THE NEXT PAGE IS SHOWN FOR THE LOADS AS CALCULATED BEFORE THE NEW MEMBERS WHERE ADDED. THE ACTUAL LOADS ARE NOW MUCH LOWER REF. p 215

UPPER ROTOR BEARING SUPPORT LOADS (ULT.) (DWG. No. 385-5025)

CONDITION 8	APPLIED LOADS P_x #	APPLIED LOADS P_y #
10 G FWD CRASH	41,524	—
4 G SIDE CRASH	—	±16,610

MEMBER	LENGTH IN.	LOAD IN MEMBER # 10 G FWD CRASH	LOAD IN MEMBER # 4 G SIDE CRASH
AE	25.63	15,368 t	—
CF	25.63	15,368 t	—
AG	25.64	12,648 c	—
CH	25.64	12,648 c	—
DG	23.31	—	7,442 c t
DH	23.31	—	7,442 t c
BE	24.19	—	6,282 c t
BF	24.19	—	6,282 t c
			$+P_y$ $-P_y$



REF. PG. 215

AG & CH ARE CRITICAL MEMBERS

MAT'L: 4130 STEEL (COND. N), 1.625 O.D. X .049 W.T. TUBE

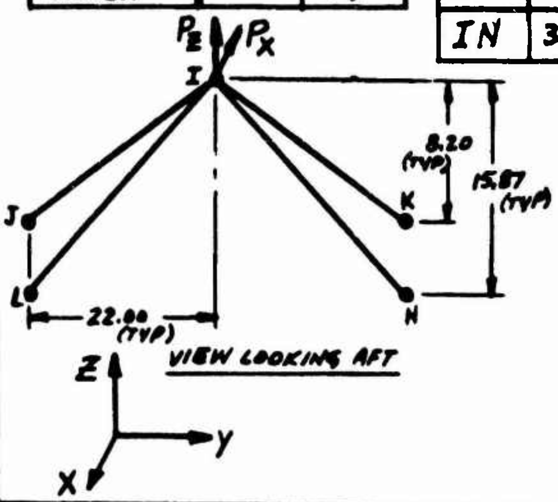
$A = .217\ IN^2$, $I = .068\ IN^4$, $L/\rho = 45$, $F_c = 65{,}000$ PSI

$f_c = \frac{12648}{.217} = 58{,}286$ PSI, $M.S. = \frac{65000}{58286} - 1 = \underline{\underline{+.11}}$

LOWER ROTOR BEARING SUPPORT LOADS (ULT) (DWG No. 385-5018)

CONDITION	APPLIED LOADS P_x #	APPLIED LOADS P_z #
2.5 G MAN. THRUST FWD	-4,478	61,764
2.5 G MAN. THRUST AFT	4,478	61,764
10 G DOWN CRASH	—	-33,690

MEMBER	LENGTH IN.	LOAD IN MEMBER # 2.5 G THRUST FWD	LOAD IN MEMBER # 2.5 G THRUST AFT	LOAD IN MEMBER # 10 G DWN CRASH
IJ	30.15	40,671 t	35,939 t	20,895 c
IK	30.15	40,671 t	35,939 t	20,895 c
IL	32.92	41,085 t	43,755 t	23,139 c
IN	32.92	41,085 t	43,755 t	23,139 c



IJ & IL ARE CRITICAL MEMBERS

MAT'L: 4130 STEEL (COND. N)

1.625 O.D X .120 W.T.

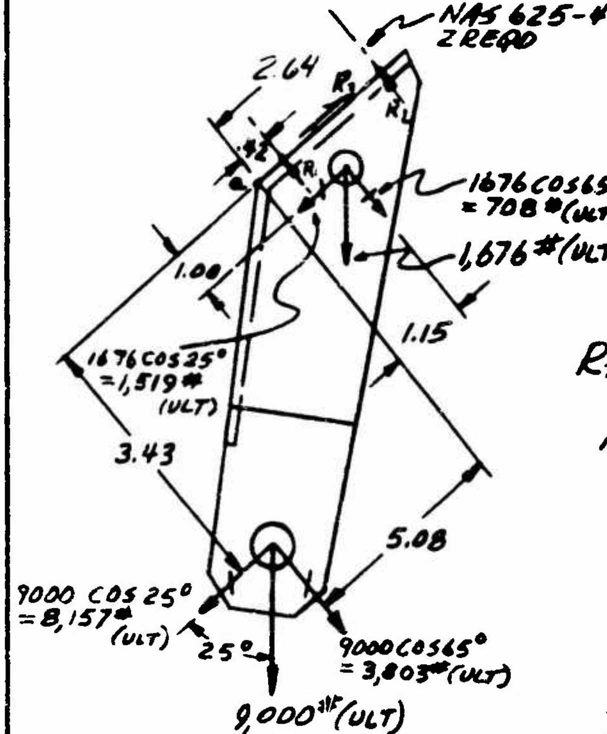
$A = .567\ IN^2$, $F_{tu} = 90{,}000$ PSI

$f_t = \frac{P}{A} = \frac{43755}{.567} = 77{,}169$ PSI

$M.S. = \frac{90000}{77169} - 1 = \underline{\underline{+.16}}$

385-5025 UPPER ROTOR BEARING SUPPORT (CONT)

-27 CHANNEL MAT'L: 4130 ST'L (COND. N)



$R_3 = 1519 + 8157 = 9{,}676\#(ULT)$

$\Sigma M_1 = 0 = 708(.73) + 1519(1.00) - 3803(5.50) + 8157(3.43) + 2.22 R_2$

$R_2 = \frac{9135}{2.22} = 4120\# ULT$

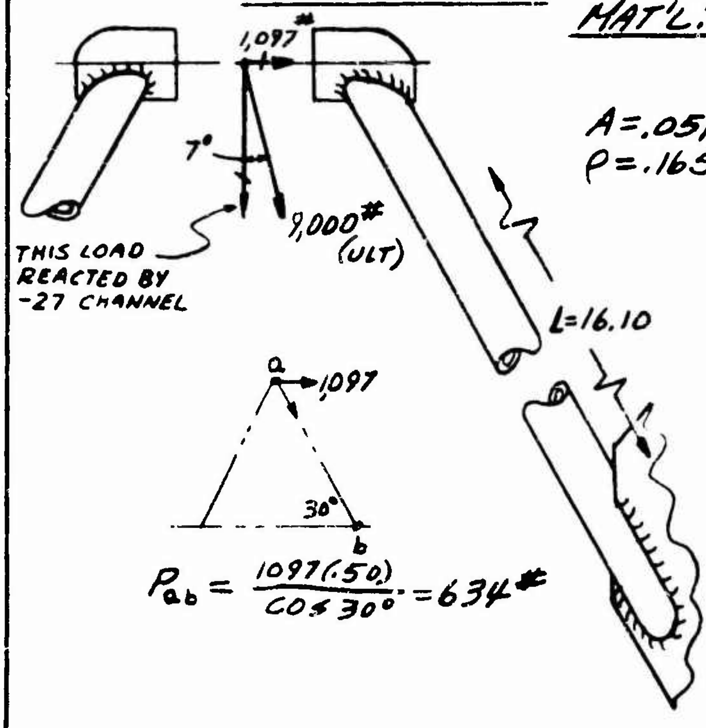
$R_1 = 708 + 3803 - 4120 = +391\#(ULT)$

M.S. = + HIGH

9000# ULT EQUALS RELIEF SETTING OF ACTUATOR CYLINDER × 1.5

-25 & -26 TUBE

MAT'L: 4130 ST'L TUBE (COND. N) .50 O.D × .035 W.T.



$A = .051\ IN^2,\ I = .0014\ IN^4$

$\rho = .165\ IN$

$L'/\rho = \frac{16.10}{.165} = 97.58$

$F_c = 30{,}000\ PSI$

$f_c = \frac{P_{ab}}{A} = \frac{634}{.051} = 12{,}431\ PSI$

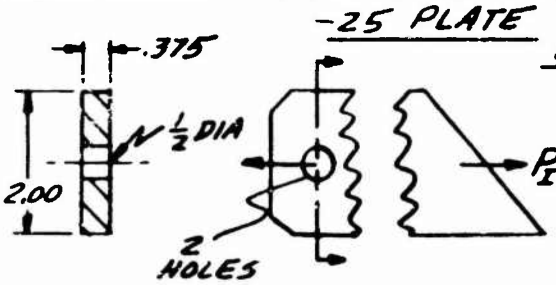
M.S. = + HIGH

$P_{ab} = \frac{1097(.50)}{\cos 30°} = 634\#$

9000#(ULT) EQUALS RELIEF SETTING OF ACTUATOR CYLINDER TIMES 1.5

385-501B LOWER ROTOR BEARING SUPPORT (CONT.)

-25 PLATE



MAT'L: 4130 STEEL (COND. N.)

$F_{tu} = 90{,}000$ PSI

$F_{bru} = 140{,}000$ PSI ($e/D = 2.00$)

$P_{IK} = 40{,}671\#$ (ULT)

$$f_t = \frac{P_{IK}}{A} = \frac{40671}{.375(1.5)} = 72{,}304 \text{ PSI}$$

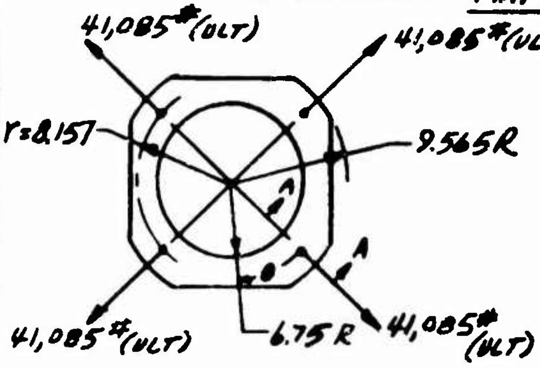
$$f_{br} = \frac{.50(40671)}{.50(.375)} = 108{,}456 \text{ PSI}$$

$$M.S. = \frac{90000}{72304} - 1 = \underline{\underline{+.24}}$$

$$M.S. = \frac{140000}{108456} - 1 = \underline{\underline{+.29}}$$

-3 FITTING

MAT'L: 4130 STEEL, H.T. TO 140-160 KSI

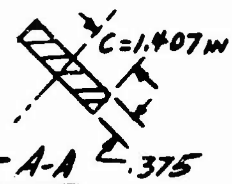


$$M_{A\text{-}A} = Pr\left[\frac{\sin\theta + \cos\theta}{2} - .6366\right]$$ {REF. PROD. ENG. JAN. 7, 1963 P73, EQ. 45 REF. 31}

@ A-A: $\theta = 0°$

$$M_{A\text{-}A} = 41085(8.157)(.50 - .6366)$$
$$= -45{,}779 \text{ IN}\#$$

ASSUMING THE CENTROID & NEUTRAL AXIS COINCIDE.



SECT A-A

$$f_b' = \frac{Mc}{I} = \frac{45779(1.407)(12)}{.375(2.815)^3} = 92{,}401 \text{ PSI}$$

TO ADJUST FOR ABOVE ASSUMPTION

{REF: ROARK PG 148 CASE I REF. 32}

$$\frac{R}{C} = \frac{8.157}{1.407} = 5.80$$

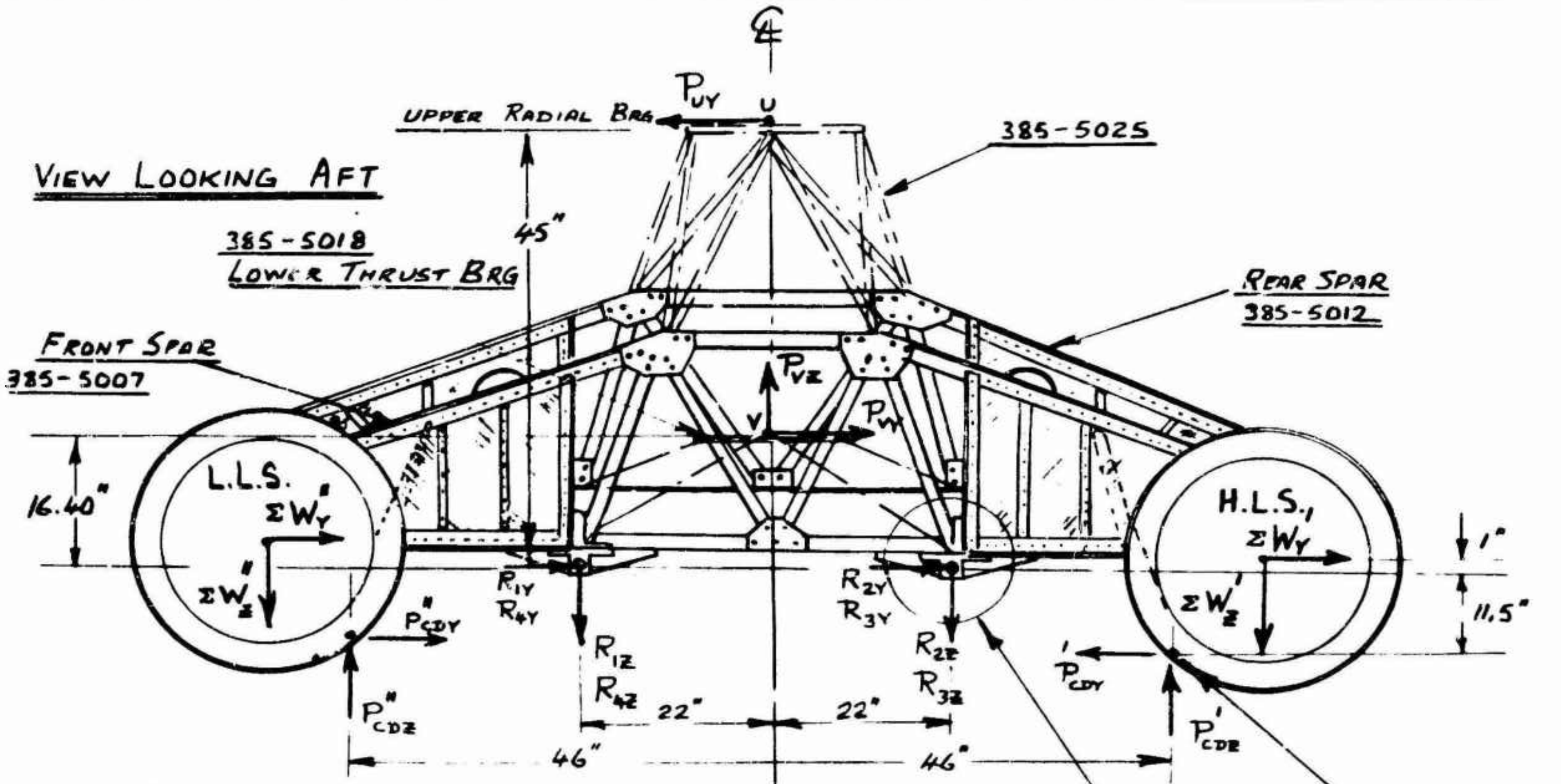
$$k_i \approx 1.11 = \frac{f_b}{f_b'}$$

(INNER SURFACE)

$$f_b = 1.11(92401) = 102{,}565 \text{ PSI}$$

$$M.S. = \frac{140{,}000}{102{,}565} - 1 = \underline{\underline{+.36}}$$

ALL OTHER PARTS & BOLTS HAVE A MARGIN OF SAFETY GREATER THAN +.50.



POWER MODULE INTERNAL STRUCTURE

(SKINS OMMITTED FOR CLARITY)

REF DWG Nº 385-5000

LOADS POSITIVE WHEN ACTING AS SHOWN

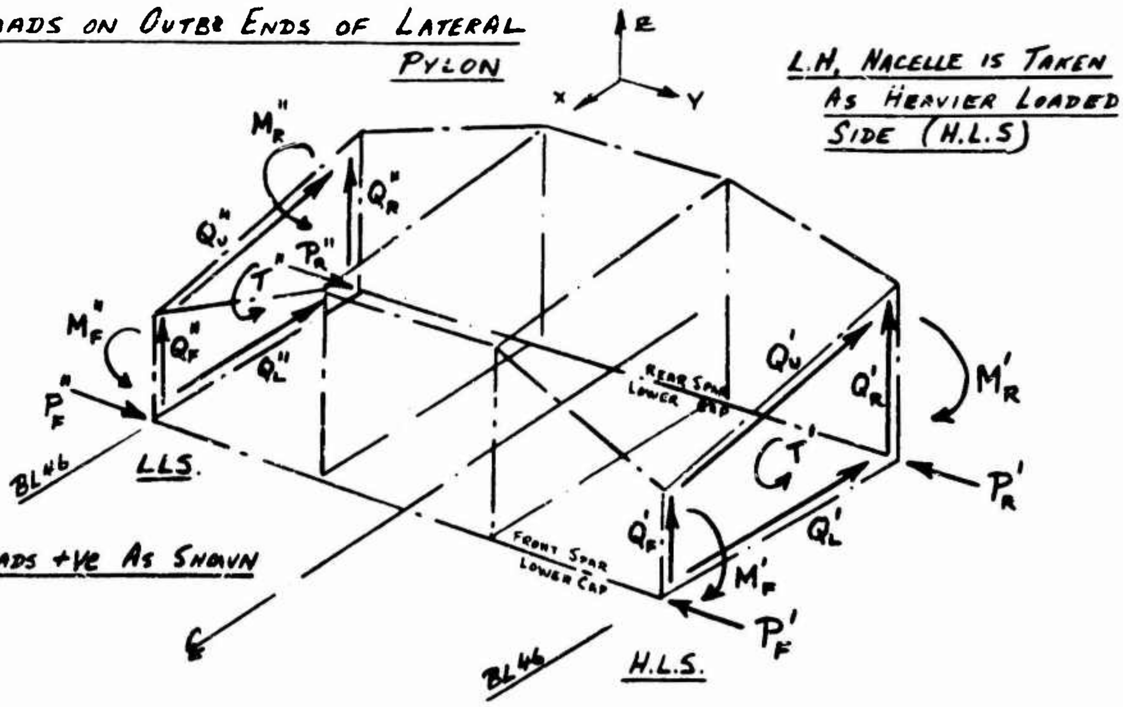
LEFT HAND NACELLE = HEAVILY LOADED SIDE (HLS)

(NOTE:— $P_{CD}' = HLS$. $P_{CD}'' = LLS$ ETC ETC)

LATERAL PYLON

LOADS ON OUTBD ENDS OF LATERAL PYLON

L.H. NACELLE IS TAKEN AS HEAVIER LOADED SIDE (H.L.S)



LOADS +VE AS SHOWN

SUMMARY OF CRITICAL LOADS (LIMIT) (DUE TO OLEO, INERTIA RELIEF & ENGINE LOADS)

1) HLS

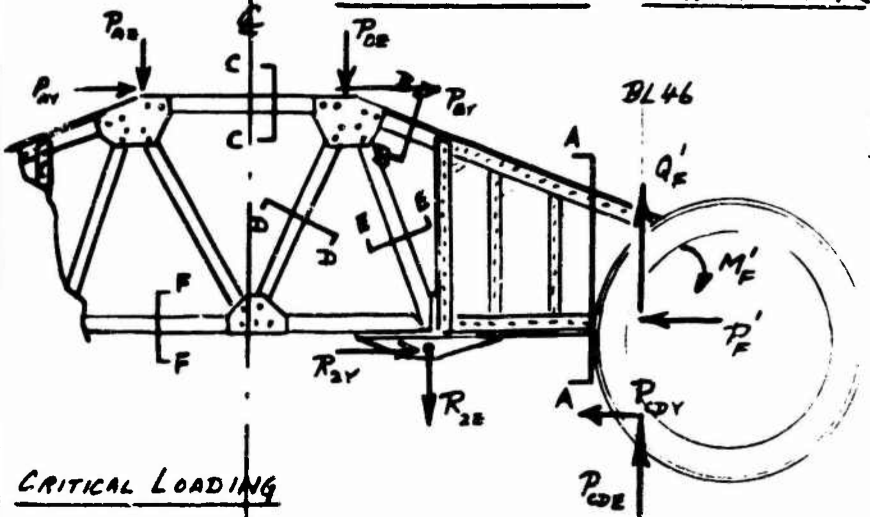
CASE	Q_F' lbs	Q_R' lbs	Q_V' lbs	Q_L' lbs	P_F' lbs	P_R' lbs	M_F' lbs ins	M_R' lbs ins	T' lbs ins
3 - POWER OFF	+12,221	+451	+2996	+2996	+5144	−198	+130,429	−28,804	+195,220
4' - POWER ON	+6917	−1743	+2318	+2318	+3311	−1819	+110,020	+2297	+271,451
5 - POWER OFF	+7751	+103	+1960	+1960	+3355	−122	+87,785	−16,555	+185,654

2) LLS

CASE	Q_F'' lbs	Q_R'' lbs	Q_V'' lbs	Q_L'' lbs	P_F'' lbs	P_R'' lbs	M_F'' lbs ins	M_R'' lbs ins	T'' lbs ins
3 - POWER OFF	+3156	−230	+2292	+2292	+4036	−122	+96,680	−14,420	+188,829
4' - POWER ON	+8005	−414	+2510	+2510	+1703	−997	+97,950	−12,455	+160,441
5 - POWER OFF	+231	+282	−98	−98	+214	−82	−2484	−3027	−25,038

LATERAL PYLON

POWER MODULE – FRONT SPAR – DWG Nº 385-5007



P_{AZ} P_{AY} P_{BZ} P_{BY} :– LOADS FROM UPPER TRUSS

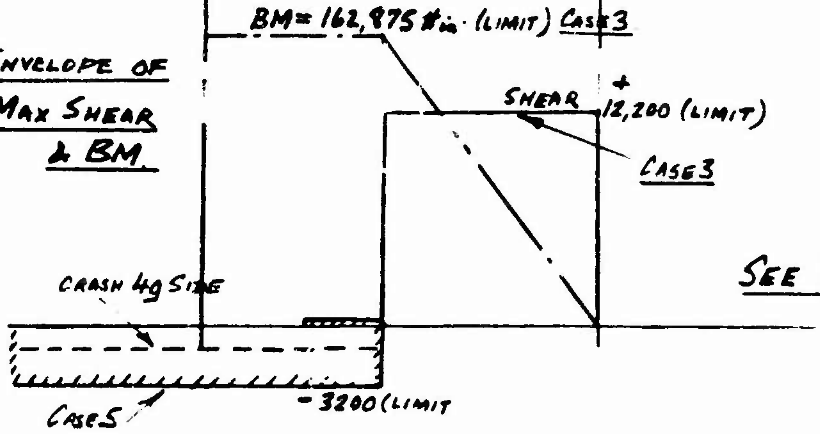
P_{CDY} P_{CDZ} = OLEO

R_{2Z} R_{2Y} = FUSELAGE REACTIONS

Q_F M_F P_Y = NACELLE LOADS

CRITICAL LOADING

ENVELOPE OF MAX SHEAR & BM



WEB = ·071 7075-T6

CAPS:– 2024-T3

TUBES:– 2024-T3

SEE PAGE 215 OF LOADS SECTION

CRITICAL SECTIONS (+Ve LOAD = COMPRESSION) ULTIMATE VALUES

SECTION A-A $q = 1220\,\#/in$ $t = \cdot 071''$ $f_s = 17{,}180\,\#/in^2$ $f_{ALLOW} = 31{,}000\,\#/in^2$ M.S. +·80

SECTION B-B $P = +12{,}216\,\#$ $A = \cdot 602\,in^2$ f_c $20{,}300\,\#/in^2$ $f_{ALLOW} = 40{,}000$ M.S. +·97

SECTION C-C (CASE 10b) $P = +20{,}853\,\#$ $A = \cdot 977\,in^2$ $f_c = 21{,}340\,\#/in^2$ $L/\rho = 52$ $f_{cr} = 27{,}300\,\#/in^2$ M.S. +·28

SECTION D-D (CASE 4') $P = +6{,}352\,\#$ $A = \cdot 470\,in^2$ $L/\rho = 58$ $f_c = 13{,}514\,\#/in^2$ $f_{cr} = 24{,}000$ M.S. +.78

DUE TO CONTROLS MOUNTED ON TUBE $f = 19{,}600\,\#/in^2$ (BENDING) $f_{ALLOW.} = 64{,}000$ M.S. HIGH

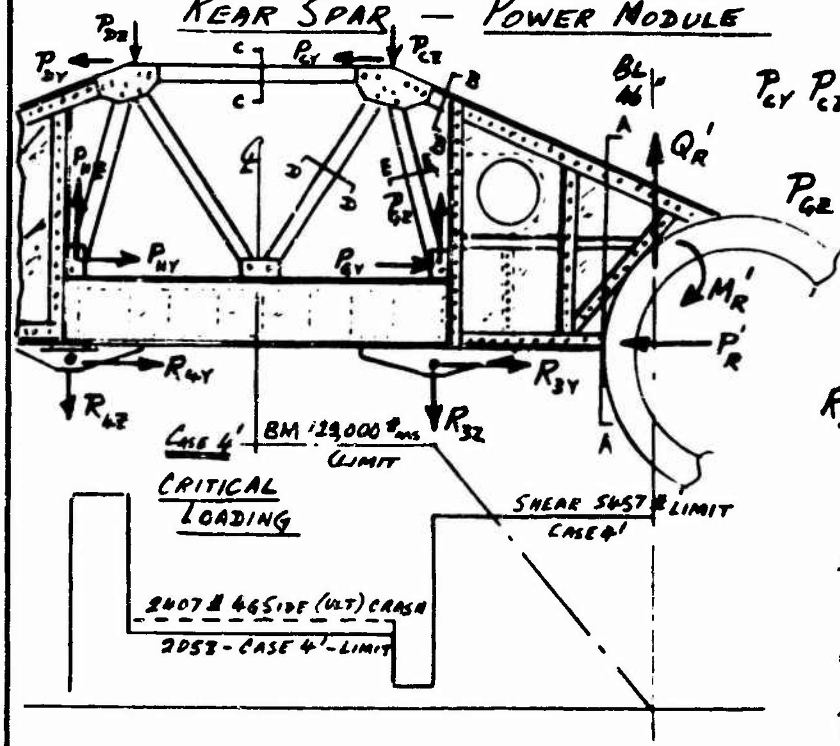
SECTION EE – CRASH 4G SIDE $P = +10{,}985\,\#$ $f = 23{,}400$ psi. $f_{cr} = 26{,}000$ psi. M.S. +.11

SECTION FF – $2\tfrac{1}{2}$G MAN. $P = +13{,}572\,\#$ $A = 1.65$ $f = 11{,}545\,\#/in^2$ M.S. +1.51
$M = 33{,}200\,\#in$ $Z = 10$ $f_{cr} = 29{,}000\,\#/in^2$

LATERAL PYLON

REAR SPAR — POWER MODULE

DWG № 385 — 5012



P_{CY} P_{CZ} P_{DY} P_{DZ} = LOADS FROM UPPER TRUSS

P_{GZ} P_{GY} P_{HZ} P_{HY} = LOADS FROM LOWER TRUSS

Q_R' & M_R' & P_R' = LOADS FROM NACELLE

R_3 R_4 = FUSELAGE REACTIONS

WEB :- ·025 — A.286 & ·040" — 2024 T3

CAPS :- 2024 - T.3

TUBES :- 2024 — T.3

CRITICAL SECTIONS (+ve LOAD = COMPRESSION) ULTIMATE VALUES

SECTION A·A (i) ·025" — A·286 WEB. $q = 454$ #/in, $f_s = 18,260$ #/in², $f_{s\,ALLOW} = 54,000$ — M.S. +2.00

(ii) ·040" — 2024·T3 WEB $q = 390$ #/in, $f_s = 9750$ #/in², $f_{ALLOW} = 21,000$ — M.S. +1.15

SECTION B-B $P = +7720$# $A = ·53$ in² $f_c = 14,600$ #/in², $f_{ALL} = 40,000$ #/in² — M.S. HIGH

SECTION C-C (CASE 10a) $P = +17,751$# $A = ·977$ in² $f_c = 18,170$ #/in², $f_{cr} = 27,000$ #/in² — M.S. +·48

SECTION D-D — CASE 8c $P = 5456$# $f = 11,600$ #/in², $f_{cr} = 24,000$ — M.S. + 1.07

SECTION E-E — CASE 8(c) (COMP^{n}) $P = +8,639$# $A = ·43$ in²

$f_c = 20,090$ #/in², $f_{cr} = 26,000$ #/in² — M.S. +·24

SKIN ASSY — LATERAL PYLON — DWG Nº 385-5015

CASE 4' - POWER-ON — ULTIMATE LOADS   TORQUE BOX AREA = 622 in²

$Q'_L = 3477$ # $T' = 407,176$ # ins. q (LOWER SKIN) = 417 #/in.

$t = .025$ (2024 T3) $f_s = 16,700$ #/in² $f_{ALLOWABLE} = 21,000$ M.S. +.26

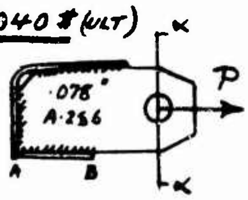
CRASH — 10g FWD. — (ULT)

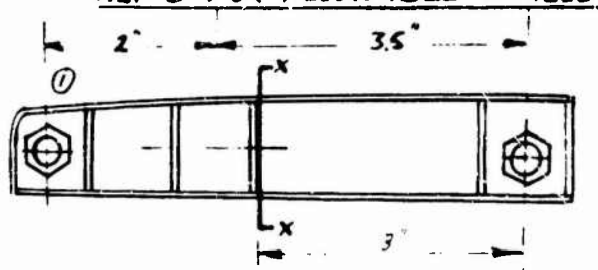
$T' = 512,830$ # ins $q = 412$ #/in — COVERED BY ABOVE

MOUNTINGS FOR DIVERTER VALVE   REFER TO PAGE 222 FOR LOADS

i) DWG Nº 385-5014 — 27   REF 3 FOR ALLOWABLE STRESSES

$P = 1.5 \times 1960 = 2940$ # (ULT)





SECTION αα $P = 2940$ # $A = .136$ in² $f_t = 21,600$ #/in² $f_{tu} = 117,000$ #/in² @ 800°F   M.S. HIGH

WELD AB $q = 1470$ #/in $f_s = 27,000$ $f_{su} = 81,000$ @ 800°F   M.S. HIGH

SECTION XX $M = 3207$ #.ins $Z = .0637$ in³ $f_t = 50,300$ #/in² $f_{tu} = 117,000$ #/in²   M.S. +1.32

SKIN ATT'NS AT ①   LOAD = 1870 #; 2-1/8 + 2 5/32 MONEL RIVETS = 2154 #   M.S. +.15

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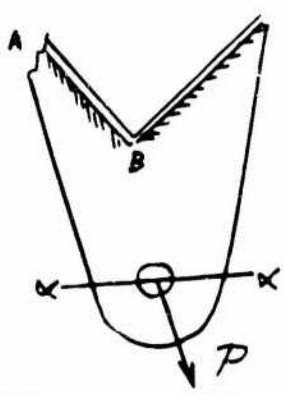
ii) DWG Nº 385-5028 $P = 9,350$ # (LIMIT)   14,000 # (ULT)



SECTION αα $A = .174$ in²

$f_t = 80,000$ #/in² $f_{tu} = 117,000$ M.S. +.45

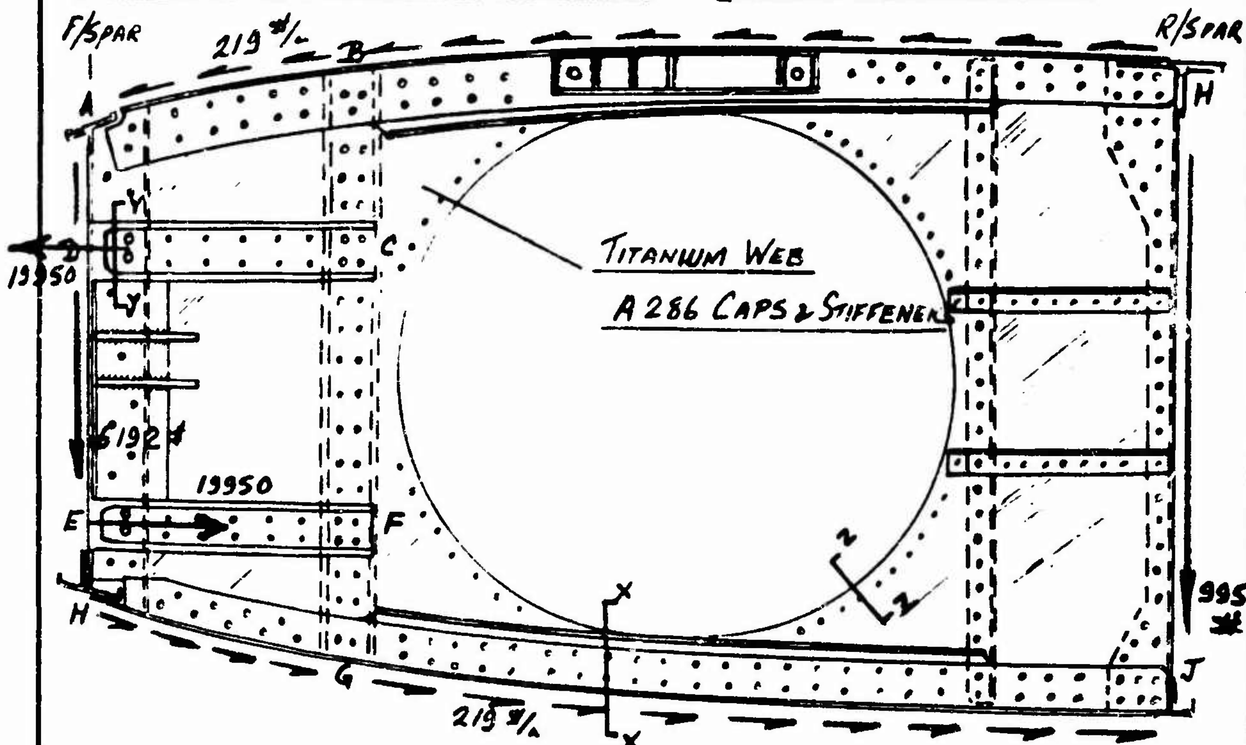
WELD AB $q = 1900$ #/in (per lug)

$f_s = 24,500$ #/in²

$f_{su} = 81,000$ #/in²   M.S. HIGH

LATERAL PYLON

CANTED RIB - DWG Nº 385-5014 (CASE 4'-ULT LOADS)



CRITICAL SECTIONS (+VE LOAD = COMPRESSION) ULTIMATE VALUES

SECTION X-X – CASE 4' $P = 3778$ # $M = 10,212$ # INS

$A_c = \cdot 7166$ in² $Z_c = \cdot 2937$ THERMAL STRESS $f = 34,150$ #/in²

TOTAL STRESS $f_c = 74,200$ #/in²

FLANGE CRIPPLING $b/t = 11.6$ (ONE EDGE FREE) $F_{cc} = 75,000$ #/in² M.S. +.01

SECTION YY – CASE 4' $P = 19,950$ # $A = \cdot 296$ in² $f_c = 71,600$ #/in²

FLANGE CRIPPLING $b/t = 24$ (NO EDGE FREE) $F_{cc} = 98,000$ #/in² M.S. +.37

SECTION ZZ – CASE 4' $P = 6,700$ # $A = \cdot 153$ in² $f_c = 43,800$ #/in²

FLANGE CRIPPLING $b/t = 38$ (NO EDGE FREE) $F_{cc} = 70,000$ #/in² M.S. +.60

RIVETING D-E – CASE 4' $q = 1570$ #/in M.S. +.11

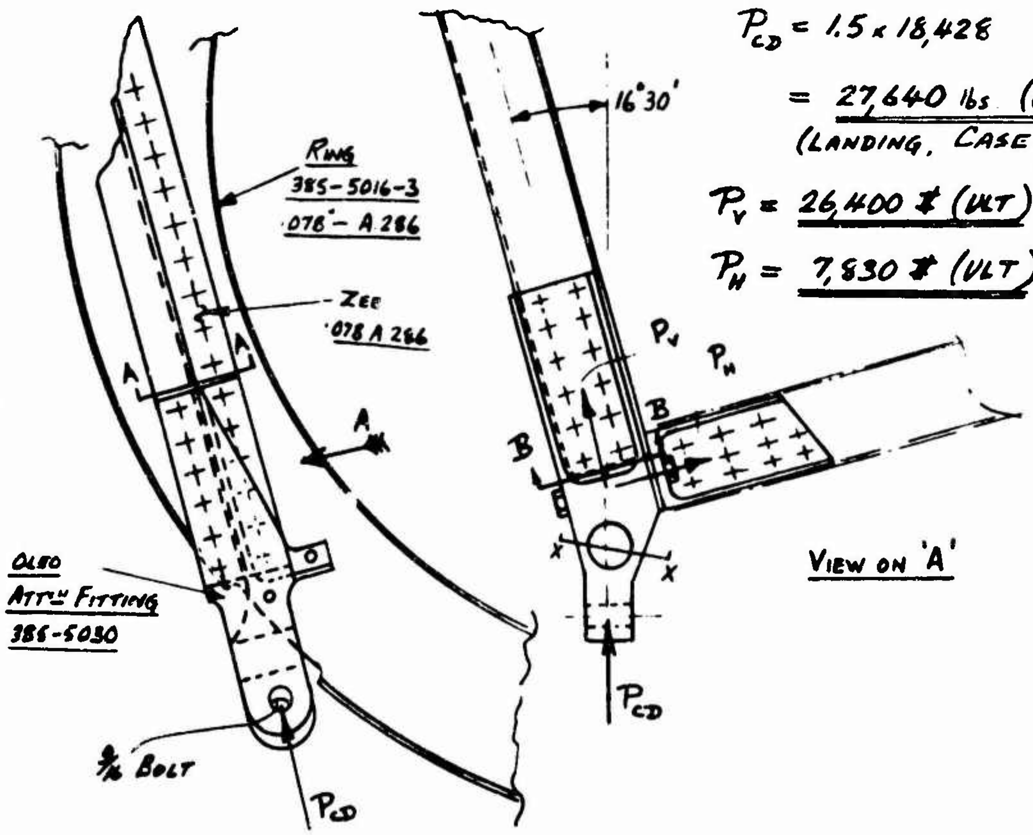
OLEO ATTCH TO POWER MODULE — DWG. No 385-5029

$P_{CD} = 1.5 \times 18{,}428$

$= 27{,}640$ lbs (ULTIMATE)
(LANDING, CASE 3)

$P_V = 26{,}400$ # (ULT)

$P_H = 7{,}830$ # (ULT)



CRITICAL SECTIONS :-

SECTION A-A $P = 26{,}400$ # $A = .4363$ in² (INCL. EFFECTIVE SHEET) $f_c = 60{,}550$ #/in²
LOCAL FLANGE CRIPPLING $F_{cc} = 78{,}800$ #/in²
M.S. +.30

SECTION BB $P = 26{,}400$ # $A = .60$ in² $f = 44{,}000$ #/in²
$f_{ALL} = 98{,}000$ "
M.S. HIGH

SECTION XX $P = 27{,}640$ # $A = .552$ in² $f_t = 50{,}100$ #/in²
$f_{t\,ALL} = 113{,}000$ #/in²
M.S. +1.22

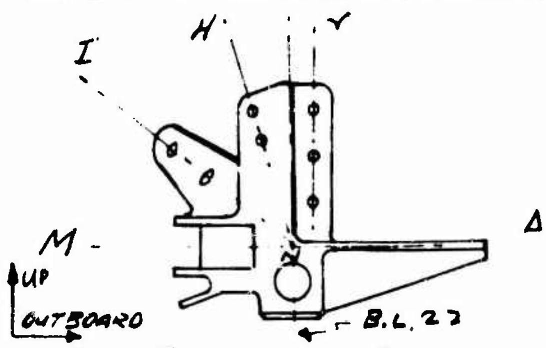
9/16 NAS BOLT LOAD = 27,640 # STRENGTH (DOUBLE SHEAR) = 47,200 lbs
M.S. +.70

ATTCH TO FRONT SPAR LOAD = 26,400 lbs.
19 - 3/16 MONELS STRENGTH = 27,322 # M.S. +.03

COLUMN STRENGTH OF ZEE STIFFENER :- L (EFFECTIVE) = 13.2" $\rho = .707$"
$L/\rho = 18.7$ $f_{cr} = 96{,}000$ #/in² $f_c = 84{,}600$ #/in²
M.S. +.13

SPAR FITTING ANALYSIS

FITTING ASSY. FRONT SPAR LOWER CAP DWG 385-5008



REF DWG 385-5015

MATL = 2014 AL ALLOY

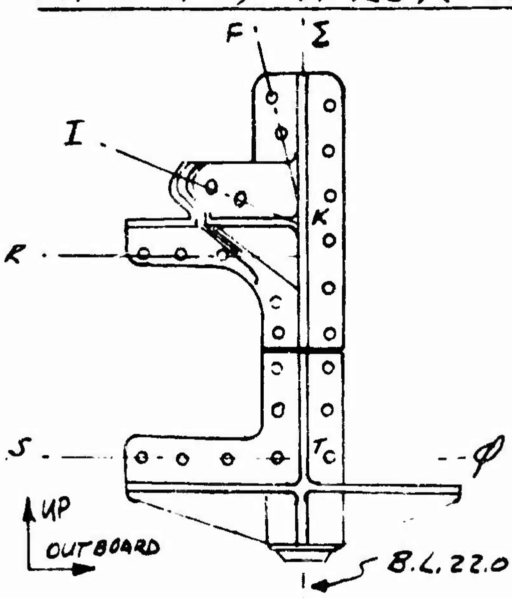
LOADS FROM PAGE 209, 215

NOMINAL WEB THICKNESS .25"

THIS FITTING IS ANALYSED FOR ATTACHMENTS

MEMBER	LIMIT LOAD	ATTACHMENT	CRITICAL COND.	M.S.
H.N.	-10401	NAS 1105 BOLT S.S 4 REQ'D	BEARING IN HN TUBE	+.47
I.N.	+7900	NAS 1308 BOLT D.S. 2 REQ'D	BOLT SHEAR	+5.30
M.N.	+13572	NAS 1105 BOLT D.S. 6 REQ'D	BOLT SHEAR	+1.87
N.Γ.	-13375*	NAS 1105 BOLT S.S 3 REQ'D; NAS 1106 BOLT S.S 2 REQ'D	BEARING & BOLT SHEAR	+1.96
N.Δ	-11494	HUCK BOLTS 1/4" S.S. 6 REQ'D	BOLT SHEAR	+.62
N (REACT.)	-29515*	NAS 632 BOLT 1 REQ'D	BEARING IN FITTING	+2.96

FITTING ASSY. REAR SPAR LOWER CAP DWG 385-5010



REF DWG 385-5012

MATL 2014 AL ALLOY HAND FORGING.

LOADS FROM PAGE 209·215

NOMINAL WEB THICKNESS .25"

THIS FITTING IS ANALYSED FOR ATTACHMENTS.

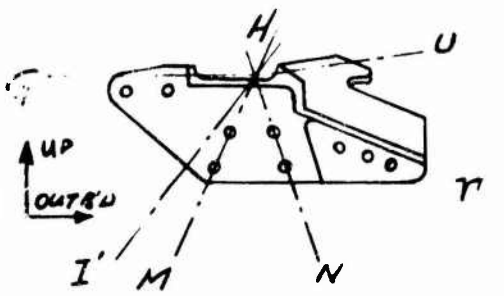
NOTE- ALL LOADS HAVING * ARE CRASH LOADS & ARE ULTIMATE LOADS

MEMBER	LIMIT LOAD	ATTACHMENTS	CRITICAL COND.	M.S.
KE	+12122	1/4 HUCK BOLTS S.S 5 REQ'D; 5/16 LOCK BOLTS S.S 4 REQ'D; 1/4 LOCK BOLTS S.S 4 REQ'D	BEARING & BOLT SHEAR	+1.80
FK	+10792	NAS 1105-6-SS 4 REQ'D	BEARING IN TUBE F.K	+.47
IK	+7323	NAS-1308 BOLT S.S. 2 REQ'D	BEARING IN ATTACHMENTS	+4.08
KR	-9853	5/16 LOCK BOLT S.S. 6 REQ'D	SHEAR ON LOCK BOLTS	+3.45
KT	+26280*	5/16 LOCK BOLT S.S.-3 REQ'D; 1/4 LOCK BOLT S.S.-3 REQ'D; NAS 1137 S.S.-3 REQ'D	BEARING IN ATTACHMENTS	+.37
ST	+4380	1/4 LOCK BOLT S.S.-8 REQ'D	BEARING IN ATTACHMENTS	+2.60
TΦ	+4670	1/4 LOCK BOLT S.S.-8 REQ'D	BEARING IN ATTACHMENTS	+2.03
T (REACT.)	+26280*	NAS 692 BOLT 1 REQ'D	BOLT TENSION	+1.70

SPAR FITTING ANALYSIS

FITTING ASSY. FRONT SPAR UPPER CAP DWG 385-5009

REF DWG 385-5007
DWG 385-5015



MATL = 2014 AL ALLOY

LOADS FROM PAGE 215

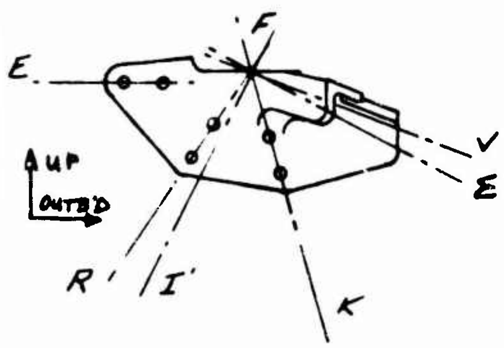
WEB = .25" THICK

THIS FITTING IS ANALYSED FOR CRITICAL ATTACHMENTS.

MEMBER	LIMIT LOAD	ATTACHMENT	CRITICAL COND.	M.S.
G.H	+20853	NAS 1106-7 SS 4 REQ.	BEARING IN TUBE GH	+.52
H.M	+6352	NAS 1104-6 SS 4 REQ.	BEARING IN TUBE HM	+.11
H.N.	-10401	NAS 1105-6 SS 4 REQ	BEARING IN TUBE HN	+.[illegible]7
H.T.	+8150	1/4" LOCK BOLT DS SS 9 REQ	BEARING IN SPAR & DOUBLER	+.08
HU	+11661	3/16 LOCK BOLTS SS. 2 REQ 5/16 LOCK BOLTS SS. 2 REQ	BEARING IN STIFFENER AND DOUBLER	+.03
H.I'	-11760	NAS 464 S.S 4 REQ	BOLT SHEAR	+.40

FITTING ASSY. REAR SPAR UPPER CAP DWG 385-5011

REF DWG 385-5012
DWG 385-5015



MATERIAL = 2014 AL ALLOY

LOADS FROM PAGE 215

WEB THICKNESS = .25"

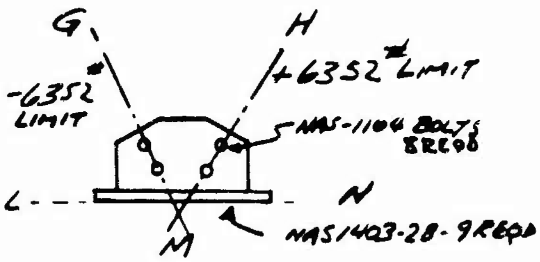
THIS FITTING IS ANALYSED FOR CRITICAL ATTACHMENTS

MEMBER	LIMIT LOAD	ATTACHMENT	CRITICAL CONDITION	M.S.
EF	+17751	NAS 1106 S.S 4 REQ	BEARING IN TUBE EF	+.79
FR	+5456*	NAS 1104 S.S. 4 REQ	BEARING IN TUBE FR	+.29
FK	+10792	NAS 1105 S.S 4 REQ 1/4" LOCKBOLT 4 REQ	BEARING IN TUBE FK	+.47
FE	-7190	1/4" LOCK BOLT 9 REQ	BEARING SPAR & DOUBLER	+2.04
FV	+11211	3/16 LOCK BOLT 2 REQ 5/16 LOCK BOLT 3 REQ	BEARING IN STIFFENER AND DOUBLER	+.30
FI'	-13307	NAS 464 S.S 4 REQ	BOLT SHEAR	+.19

*NOTE ALL LOADS HAVING * ARE CRASH LOADS AND ARE ULTIMATE LOADS

SPAR FITTING ANALYSIS

FITTING - FRONT SPAR LOWER CENTER DWG 385-5020



MATERIAL = 2014-T4 AL.

LOADS FROM PG 215

MEMBERS GM OR HM IN BOLT BEARING HAS LOWEST MARGIN. FITTING AND BOLTS OK.

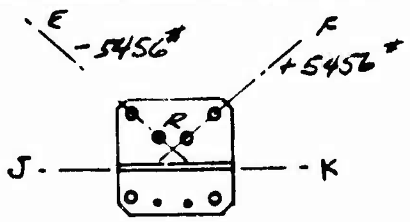
F_{BR} IN .083-2024 = 4(2650) = 10600#

$$M.S. = \frac{10600}{6352 \times 1.5} - 1 = +.11$$

LOAD TO MEMBER LMN = 2(6352).443 = 5620#

$\frac{5620}{9} = 625$ #/BOLT O.K. BY INSPECTION

FITTING - REAR SPAR LOWER CENTER DWG 385-5026



MATERIAL = 2014-T4 AL

LOADS FROM PG 215

VALUES SHOWN ARE FOR CRASH CONDITION & ARE ULTIMATE.

MEMBER ER OR FR HAS THE LOWEST MARGIN IN BOLT BEARING. FITTING AND BOLTS O.K.

F_{BR} IN .083" WALL 2024 WALL AL TUBE = 4(2650) = 10600#

$$M.S. = \frac{10600}{5456} = +.83$$

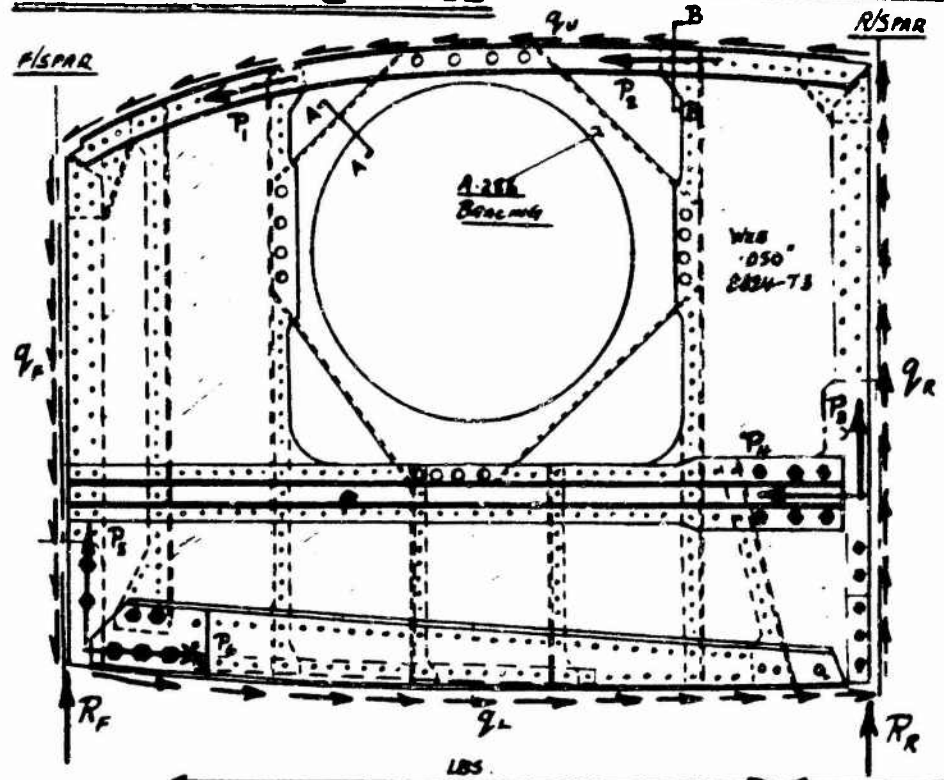
LOAD INTO MEMBER JRK

2(5456# x .592) = 6470# ULT

M.S. HIGH PASSED BY INSPECTION

LATERAL PYLON

RIB INSTALLATION @ B.L. 22" DWG No 385-5013



LOADS QUOTED ARE ULTIMATE
+VE AS SHOWN

CASE	P_1 (LBS.)	P_2	P_3	P_4	P_5	P_6	q_U (LBS/IN)	q_L	q_F	q_R
8(b) CRASH 10G FWD	8403	11070	-521	-1200	520	610	76	-212	-96	-40
10(a) 2½ MAN. THRUST AFT	13663	16816	2986	6887	5393	6333	105	105	190	20
10(b) 2½ MAN THRUST FWD	13870	8664	2585	5959	5712	6707	87	87	160	14

SECTION A-A (CASE 8(b)) $A = .079$ $P_c = 5332$ lbs

$\therefore f_c = 67{,}500$ #/in² **M.S. 0.00**

FLANGE CRIPPLING ALLOWABLE (AVGE) = 67,500 #/in²

SECTION BB

MAX TENSION (CASE 10a) = 16,182 # AREA = .190 in² (EQUIV. A-286)

$f_t = 85{,}168$ #/in²

$f_{tu} = 140{,}000$ **M.S. +.64**

MAX COMPS (CASE 8b) = 7350 lbs

TENSION FIELD EFFECTS $P = 2630$ # (COMPS) $A_c = .216$ in² $Z = .054$ in³ (EQUIV A-286)

$M = 1191$ # ins

$P/A + M/Z = 22{,}055 + 46{,}204 = 68{,}254$ #/in² **M.S. +.02**

FLANGE CRIPPLING $f_{cc} = 70{,}000$

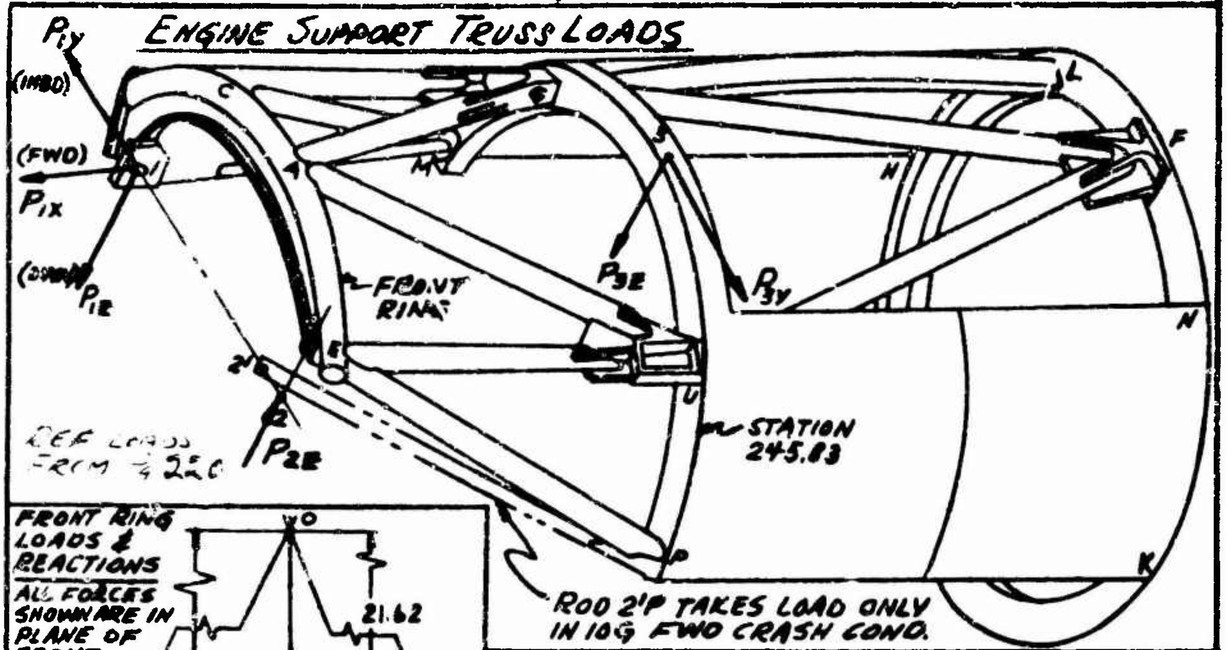
SHEAR WEBS

CASE 8(b) $q = 811$ #/in $f_s = 16{,}220$ #/in² @ 23,000 #/in² **M.S. +.42**

5/32 RIVETS (MONEL) @ 1" PITCH STRENGTH 811 #/in **M.S. 0.00**

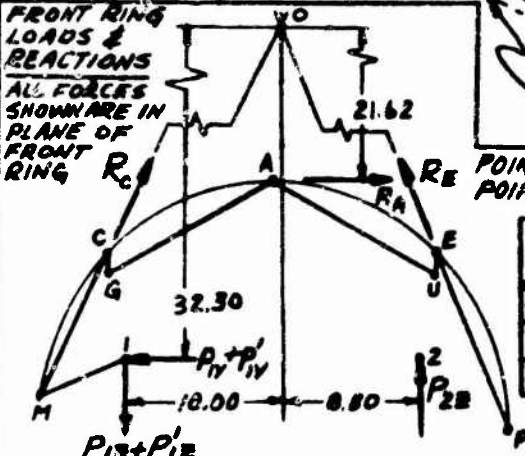
ENGINE MOUNT ANALYSIS

ENGINE SUPPORT TRUSS LOADS



FRONT RING LOADS & REACTIONS

ALL FORCES SHOWN ARE IN PLANE OF FRONT RING



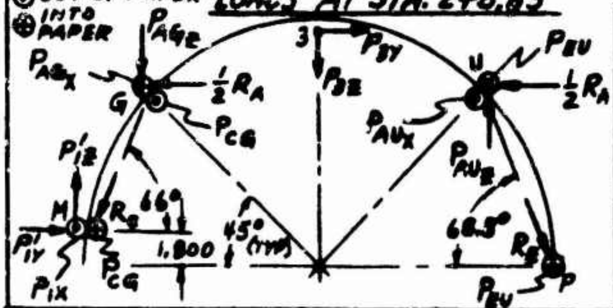
POINTS U & G ARE DRAWN DISPLACED FOR CLARITY
POINT U IS BEHIND E & POINT G IS BEHIND C.

CONDITION	P_{1X}	P_{1Y}	P_{1Z}	P_{2Z}	R_A $\Sigma M_C=0$	R_C $\Sigma M_E=0$	R_E $\Sigma M_C=0$
[illegible] ONLY (LIM)	7820	2695	419	1740	5106	-1194	819
FLIGHT COND. #13 (LIM)	6005	2661	1496	1986	3997	34	260
4G SIDE CRASH (ULT)	6300	1499	2133	3496	1550	1535	-1786

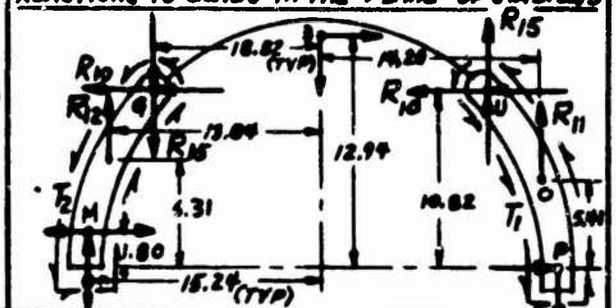
$$P'_{1Y}=.209\,P_{1X}\,,\quad P'_{1Z}=.127\,P_{1X}\,,\quad P_{1M}=1.03\,P_{1X}$$

LOADS AT STA. 246.83

⊙ OUT OF PAPER
⊕ INTO PAPER



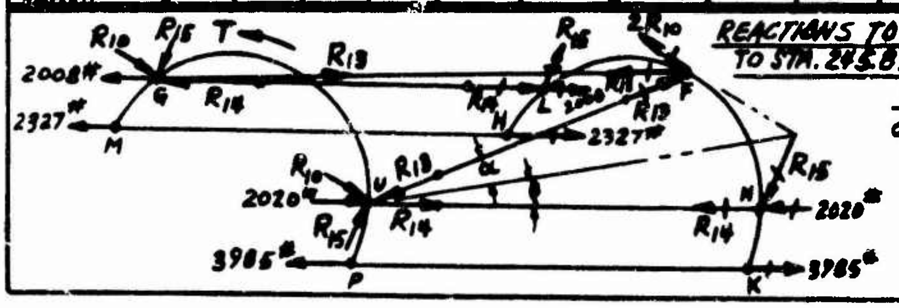
REACTIONS TO LOADS IN THE PLANE OF STA 245.83



CONDITION	P_{3Y}	P_{3Z}	P_{CG}	P_{EU}	$P_{AGX}=P_{AUX}$	$P_{AGZ}=P_{AUZ}$	R_{10} $\Sigma F_H=0$	R_{11} $\Sigma F_V=0$	R_{12} $\Sigma F_V=0$	R_{15} *	T ** $\Sigma M_o=0$
FLIGHT COND. #13 (LIM)	3212	3149	89	500	5046	824	274	858	1771	120	1336
4G SIDE CRASH (ULT)	4310	1063	3973	-3985	1965	321	1403	0	0	614	-44820 IN#

* SEE BELOW
** ASSUME $T_1=T_2=\frac{1}{2}T$

REACTIONS TO LOADS PERPENDICULAR TO STA. 245.83 & TO R_{10}



4G SIDE CRASH (ULT)

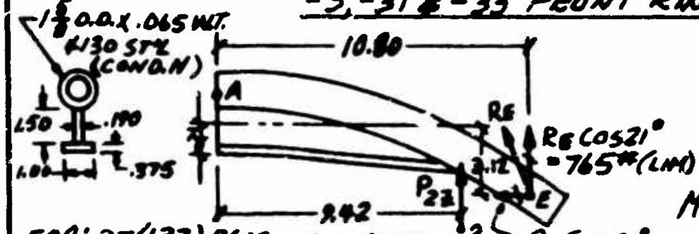
$\alpha=17°23'$, $\beta=7°52'$

$R_{13}=4{,}695\#$
$R_{14}=4{,}443\#$
$R_{15}=614\#$

ENGINE MOUNT ANALYSIS

ENGINE SUPPORT TRUSS ASS'Y (DWG No. 385-5005)

-3, -31 & -33 FRONT RING



FOR: 27(1.33) PSIG ONLY COND.
$R_E = 819^{\#}$ (LIM), $P_{2Z} = 1{,}740^{\#}$ (LIM)

$A = .974\ IN^2$
$\bar{y} = 1.268\ IN.$
$I = 1.229\ IN^4$

$M = 1.5\left[10.80(765) + 9.42(1740) - 3.12(293)\right]$
$= 35{,}608\ IN^{\#}$ (ULT)

MAX. f_c OCCURS IN -3 TUBE

$D/t = 25$; $F_b = 105{,}000$ PSI

$L'/\rho = \dfrac{10.80}{.552} = 19.6$

$F_c = 67{,}000$ PSI

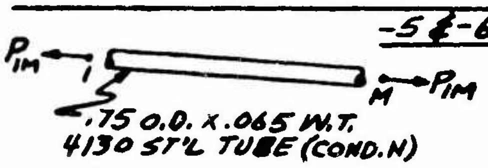
$f_{bc} = \dfrac{MC}{I} = \dfrac{35608(2.232)}{1.229} = 64{,}660$ PSI

$f_c = \dfrac{P}{A} = \dfrac{293(1.5)}{.974} = 451$ PSI (ULT)

$$\frac{f_{bc}}{F_b} + \frac{f_c}{F_c} = \frac{64660}{105000} + \frac{451}{67000} = .622 < 1$$

$$M.S. = \frac{1}{.622} - 1 = \underline{\underline{+.61}}$$

-5 & -6 TUBE

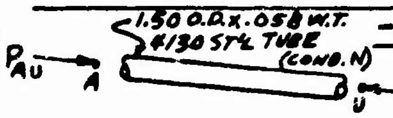


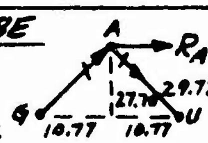
$P_{IX} = 7820(1.5) = 11{,}730^{\#}$ (ULT) (27 X 1.33 PSIG ONLY COND.)
$P_{IM} = 1.03 P_{IX} = 12{,}082^{\#}$
$A = .140\ IN^2$

$f_t = \dfrac{P_{IM}}{A} = \dfrac{12082}{.140} = 86{,}300$ PSI, $F_{tu} = 90{,}000$ PSI (IN WELD AREA)

$$M.S. = \frac{90000}{86300} - 1 = \underline{\underline{+.04}}$$

-11 & -12 TUBE





$R_A = 5{,}106^{\#}$ (LIM) (27 X 1.33 PSIG ONLY COND.)

$P_{AU} = \dfrac{29.72(5106)(1.5)}{10.77(2)}$
$= 10{,}568^{\#}$ (ULT)

$A = .263\ IN^2$, $\rho = .510$ IN, $I = .068\ IN^4$

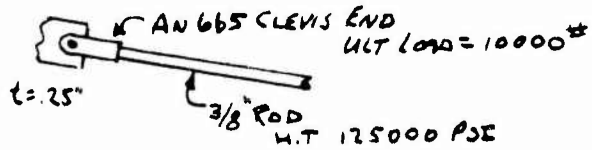
$L'/\rho = \dfrac{29.72}{.510} = 58.3$, $F_c = 57{,}000$ PSI

$f_c = \dfrac{P_{AU}}{A} = \dfrac{10568}{.263} = 40{,}183$ PSI

$$M.S. = \frac{57000}{40183} - 1 = \underline{\underline{+.42}}$$

$P_{2'P}$ ROD REF DWG 385-7109

CRASH COND. 8b



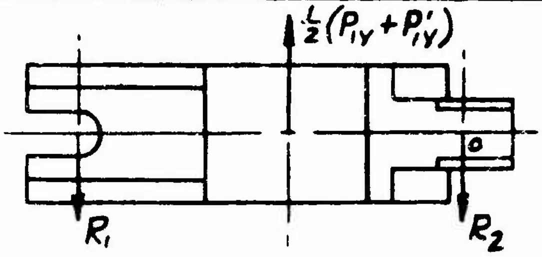
$P_{2'X} = +7362$
$P_{2'Y} = +2054$
$P_{2'Z} = +1421$

$P_{AXIAL} = 7770^{\#}$

TENSION THROUGH THREADS ON $\frac{3}{8}$" DIA ROD $A = .0824\ IN^2$

$\dfrac{P}{A} = \dfrac{7770}{.0824} = 94300$ PSI $\qquad M.S. = \dfrac{125000}{94300} - 1 = \underline{\underline{+.33}}$

FOWARD ENGINE MOUNT CLAMP (DWG No. 385-7313)



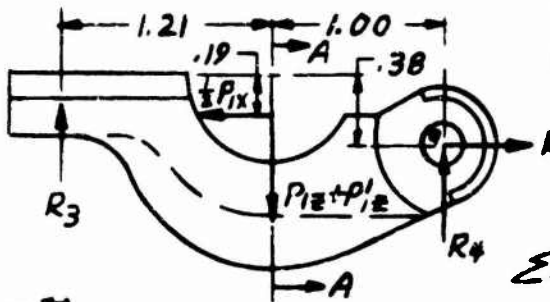
MAT'L: 410 CRES BAR (AMS 5612)
H.T TO 180-200 KSI

FLIGHT CASE #13 (LIM)

$P_{1X} = 6{,}005^{\#}$

$P_{1Y}+P'_{1Y} = 3{,}916^{\#}$

$P_{1Z}+P'_{1Z} = 2{,}259^{\#}$



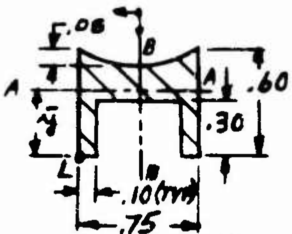
$\Sigma M_0 = 1.00\left(\frac{3916}{2}\right) - 2.21 R_1 = 0$

$R_1 = \frac{3916}{4.41} = 888^{\#}$

$\Sigma M_Q = 1.00(2259) + .19\left(\frac{6005}{2}\right) - 2.21 R_3 = 0$

$R_3 = \frac{2259 + .19(3003)}{2.21} = 1{,}280^{\#}$

$R_5 = \frac{6005}{2} = 3{,}003^{\#}$



SECTION A-A

$A = .255\ IN^2$

$\bar{y} = .364\ IN$, $I_{AA} = .005\ IN^4$, $I_{BB} = .015\ IN^4$

$M_{AA} = 1.21 R_3 (1.5) = 2{,}323\ IN^{\#}$ (ULT)

$M_{BB} = 1.21 R_1 (1.5) = 1{,}610\ IN^{\#}$ (ULT)

AT POINT L

$$f_{b_{AA}} = \frac{MC}{I}\Big|_{AA} = \frac{2323(.364)}{.005} = 169{,}114\ PSI$$

$$f_{b_{BB}} = \frac{MC}{I}\Big|_{BB} = \frac{1610(.375)}{015} = 40{,}250\ PSI$$

TOTAL $f_b = 209{,}364$ PSI

$$f_t = \frac{P_{1X}}{2A} = \frac{6005(1.5)}{2(.255)} = 17{,}662\ PSI\ (ULT)$$

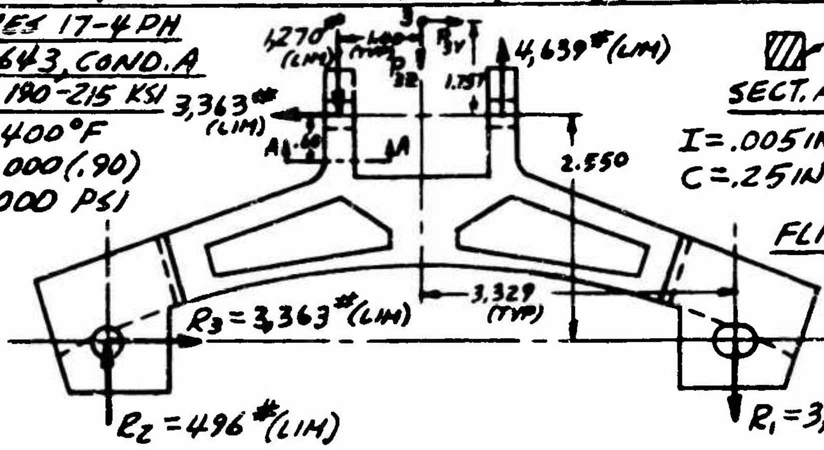
$F_b = 265{,}000$ PSI, $F_{tu} = 180{,}000$ PSI

$$\frac{f_b}{F_b} + \frac{f_t}{F_{tu}} = \frac{209364}{265000} + \frac{17662}{180000} = .88 < 1$$

$$M.S. = \frac{1}{.88} - 1 = \underline{\underline{+.14}}$$

AFT ENGINE MOUNT BRACE (DWG No. 385-7306)

MAT'L: CRES 17-4PH
AMS 5643, COND. A
H.T. TO 190-215 KSI
TEMP. ≈ 400°F
$F_{tu} = 190{,}000(.90)$
$= 171{,}000$ PSI



.50X.50
SECT. A-A
$I = .005\,IN^4$, $A = .25\,IN^2$
$C = .25\,IN$

FLIGHT CASE #13
$P_{3Y} = 3{,}363\#$ (LIM)
$P_{3Z} = 3{,}369\#$ (LIM)

@A-A; $M = 3363(1.5)(.60) = 3{,}027\ IN\#$ (ULT)

$$f_b = \frac{MC}{I} = \frac{3027(.25)}{.005} = 151{,}350\ PSI$$

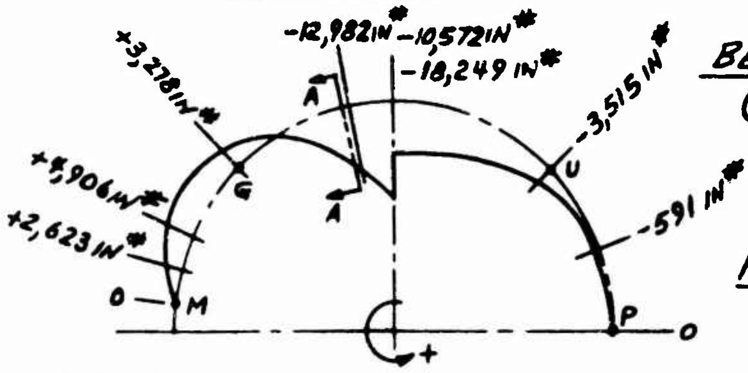
$$f_c' = \frac{P}{A} = \frac{1270(1.5)}{.25} = 7{,}620\ PSI\ (ULT)$$

TOTAL $f_c = f_b + f_c' = 158{,}970$ PSI, $M.S. = \frac{171{,}000}{158{,}970} - 1 = \underline{+.07}$

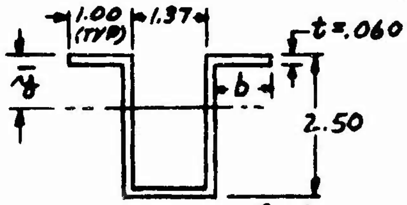
FRAME ASS'Y - NACELLE STATION 245.83 TO FUS. STA. 279.80 (DWG NO. 385-5006)

-17 FRAME

BENDING MOMENT DIAGRAM
(FLIGHT CASE #13)



MAT'L: A286 CRES
(AMS 5525B)



SECTION A-A

$\bar{y} = 1.17\,IN$, $I = .443\,IN^4$

$M = 12{,}480(1.5) = 18{,}720\ IN\#$ (ULT)

$$f_{bc} = \frac{M\bar{y}}{I} = \frac{18720(1.17)}{.443} = 49{,}441\ PSI$$

FOR; $E = 29(10)^6$ PSI, $E' = 26(10)^6$ PSI, $\bar{E} = \frac{EE'}{(\sqrt{E}+\sqrt{E'})^2} = 30.6(10)^6$ PSI

$$F_{CR} = .452\,\bar{E}\left(\frac{t}{b}\right)^2 = .452(30.6)(10)^6\left(\frac{.06}{.94}\right)^2 = 55{,}325\ PSI$$

$$M.S. = \frac{55325}{49441} - 1 = \underline{+.12}$$

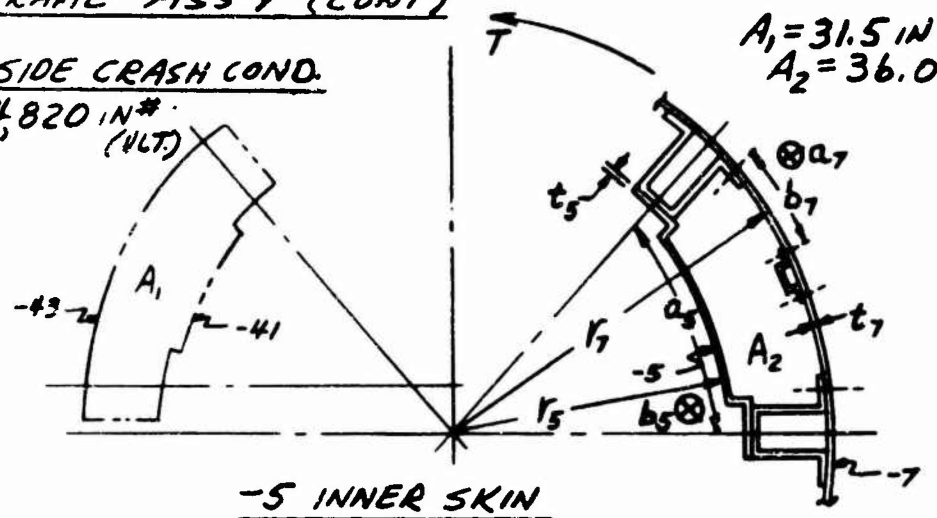
FRAME ASS'Y (CONT)

4G SIDE CRASH COND.

$T = 44{,}820$ IN# (ULT.)

$A_1 = 31.5\ IN^2$

$A_2 = 36.0\ IN^2$



-5 INNER SKIN

$r_5 = 13.5\ IN$, $a_5 = 9.20$, $b_5 = 3.4\ IN$

$t_5 = .025\ IN$

MAT'L: A286 CRES (AMS 5525B)

TEMP. = 450 °F

$E = .92(29)(10)^6 = 26.7(10)^6\ PSI$

$$F_{CR} = .10E\left(\frac{t_5}{r_5}\right) + 6.2E\left(\frac{t_5}{b_5}\right)^2 = 26.7(10)^6(.025)\left[\frac{.10}{13.5} + 6.2\left(\frac{.025}{3.4^2}\right)\right]$$

$$= 13{,}884\ PSI$$

$$f_s = \frac{T}{2t_5(A_1+A_2)} = \frac{44820}{2(.025)(67.5)} = 13{,}280\ PSI$$

$$M.S. = \frac{13884}{13280} - 1 = \underline{\underline{+.04}}$$

-7 PANEL

$r_7 = 16.5\ IN$, $a_7 = 14.0\ IN$, $b_7 = 3.6\ IN$

$t_7 = .040\ IN$

MAT'L: 2024-T3 ALUM ALLOY (QQ-A-362)

$E = 10.5(10)^6\ PSI$

$$F_{CR} = .10E\left(\frac{t_7}{r_7}\right) + 5.8E\left(\frac{t_7}{b_7}\right)^2 = 10.5(10)^6(.040)\left[\frac{.10}{16.5} + 5.8\frac{(.04)}{(3.6)^2}\right]$$

$$= 10{,}080\ PSI$$

$$f_s = \frac{T}{2t_7(A_1+A_2)} = \frac{44820}{2(.04)(67.5)} = 8{,}300\ PSI$$

$$M.S. = \frac{10080}{8300} - 1 = \underline{\underline{+.21}}$$

-31 & -32 TUBE

$R_{13} = 4{,}695$# (ULT) (4G SIDE CRASH COND.)

1.50 O.D. X .035 W.T. 4130 ST'L TUBE (COND. N)

$A = .161\ IN^2$, $\rho = .518\ IN$

$L/\rho = 35.94/.518 = 69.38$, $F_c = 48{,}000\ PSI$

$$f_c = \frac{R_{13}}{A} = \frac{4695}{.161} = 29{,}150\ PSI$$

$$M.S. = \frac{48000}{29150} - 1 = \underline{\underline{+.64}}$$

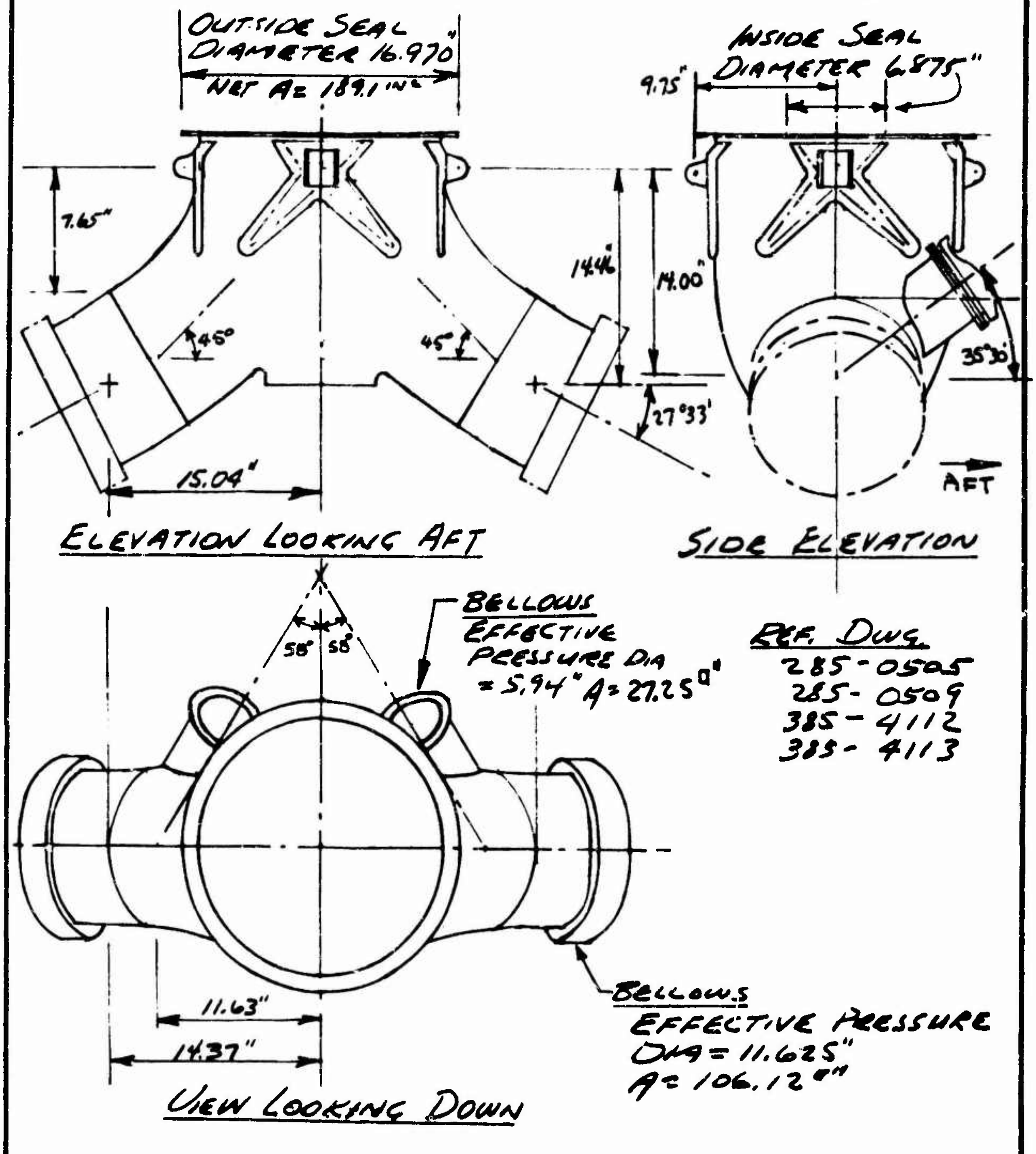
STRESS ANALYSIS - HOT GAS TRANSFER SYSTEM

The hot gas transfer system carries gas from the YT-64 gas generators to the rotor blades and to the yaw control valve. The system is fabricated from thin metal and is designed to carry the duct pressure by hoop tension. Stiffeners and straps are used where necessary because of a change in duct contour or where concentrated loads are applied to the duct.

The ducts are analyzed for a burst pressure of 54 psi, which is twice the operating pressure. This condition results in higher stresses than the operating pressure acting with the inertia loads. The allowable stresses are the 1,000-hour creep allowable stresses at the expected duct temperature.

DUCT ASSEM. LOWER STAT.

DUCT ASSEMBLY LOWER STATIONARY DWG 3&5-1603

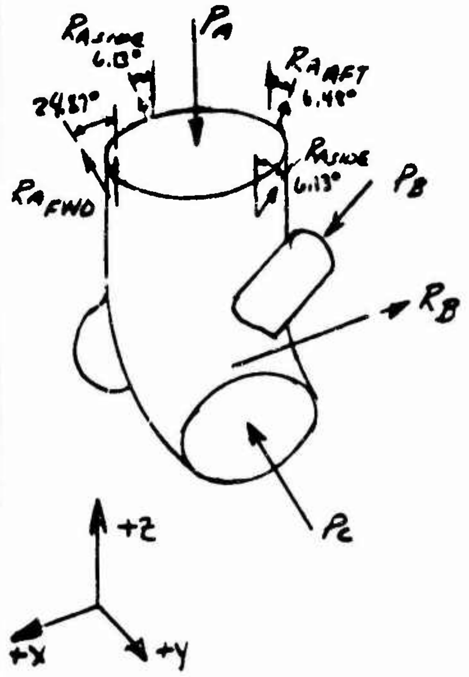


REF. DWG.
285-0505
285-0509
385-4112
385-4113

DUCT MATERIAL – INCONEL 718

DUCT BURST PRESSURE = 27 PSI X 1.33 x 1.5 = 54 PSI

DUCT ASSEMBLY LOWER STATIONARY DWG 385-1603

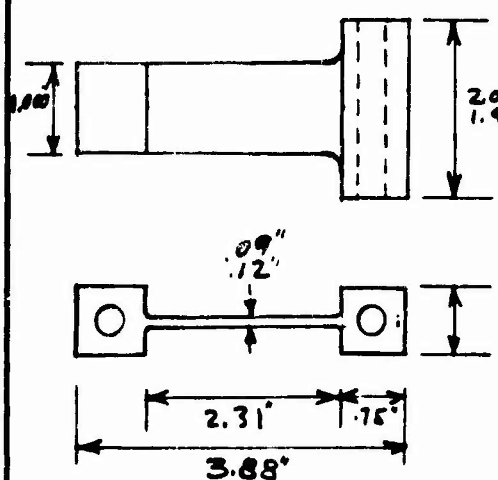


LOADS & REACTIONS - BURST CONDIT.

ITEM	LOAD	Z COMP	Y COMP	X COMP
P_A	10210	-10210	—	—
P_B*	1470	-739	±739	+1089
P_C	5730	+2650	±4800	—
RA_{AFT}	+2929	+2910	—	-330
RA_{FWD}	+2235	+2025	—	+940
RA_{SIDE}	+735	+730	±79	—
R_B	1395	—	—	+1395

* Z COMP = .5025 Y COMP = .5025 X COMP = .7390

-65 FWD FITTING



MATERIAL - TYPE 347 COR. RES. STL.

-67 & 69 FITTING LESS CRITICAL THAN -65

$$\frac{RA_{FWD}}{A} = \frac{2235^{\#}}{1.000 \times .09} = 24900 \text{ PSI ULT}$$

$$= 24900 \times \frac{1.33}{2} = 16550 \text{ PSI YIELD}$$

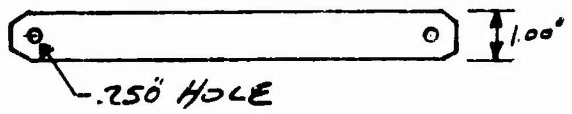
$F_{ty} = 30000 \text{ PSI} \times .7$ @ 700°F ASSUMED

$F_{ty} = 21000 \text{ PSI}$

REF. MIL-HDBK-5 USE AISI 301 VALUES

$$M.S. = \frac{21000}{16550} - 1 = +.27$$

-3 STRAP DWG 385-1200



MATERIAL - .063" A-286 t = 700°F

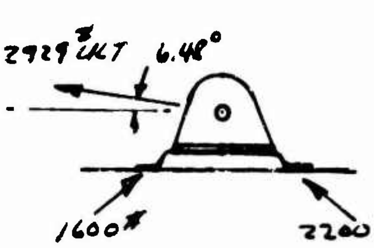
$R_B = 1395^{\#}$

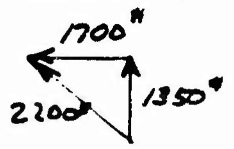
$$f_{br} = \frac{1395^{\#}}{.063 \times .25} = 88500 \text{ PSI}$$

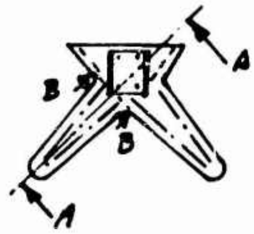
$$M.S. = \frac{294000 \times .9}{88500} = +1.95$$

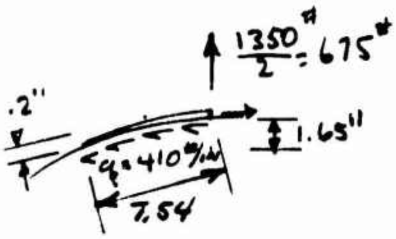
DUCT ASSEMBLY LOWER STATIONARY DWG 385-1603

-15 FITTING ASSEM & -11 STIFFENER

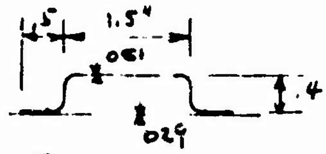








SECTION AA



SECTION B-B

$A = .241$

$P = 7.54 \times 410 = 3080^{\#}$

$M = .2 \times 3080 = 616.0''^{\#}$

$\frac{P}{A} = \frac{3080}{.241} = 12800 \text{ PSI}$

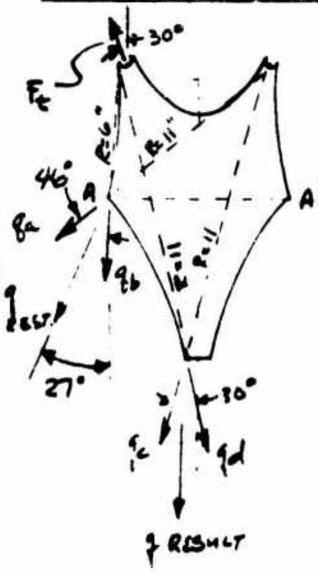
$\frac{M}{D} = \frac{616}{.4} = 1025^{\#}$ $\quad \frac{1025}{1.5 \times .051} = 13400 \text{ PSI}$

$P/A + \frac{M}{DA} = 26200 \text{ PSI}$

ASSUME $t = 1100°F$
$F_B = 50000$ PSI FOR
1000 HR CREEP

$M.S. = \frac{50000}{26200} - 1 = +.90$

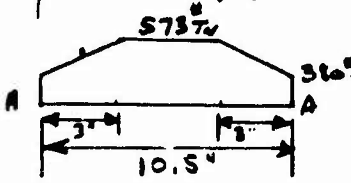
-27 BRACE —



$q_a = q_c = q_d = \frac{54 \text{ PSI} \times 11''}{2} = 297 \frac{\#}{IN}$ $\quad q_b = 162 \#/IN$

q RESULTANT OF $q_a, q_b = \sqrt{297^2 + 162^2 + 2(297)(162)\cos 46°} = 426 \frac{\#}{IN}$

q RESULTANT OF $q_c\, q_d = \sqrt{297^2 + 297^2 + 2(297)\cos 30°} = 573 \frac{\#}{IN}$



$426 \frac{\#}{IN} \times \cos 27° = 380 \frac{\#}{IN}$

$F_t = \left[\left(\frac{380 + 573}{2}\right) 6 + 573\,(4.5)\right] \frac{1}{2} \cdot \frac{1}{\cos 30°} = 3140^{\#}$

AREA RESISTING F_t

-27 PART	= .032 × 1.5 = .048	
-31 "	= .032 × 1.0 = .032	= .192 IN²
-3 "	= .032 × 1.0 = .032	
-29 "	= .080 × 1.0 = .080	

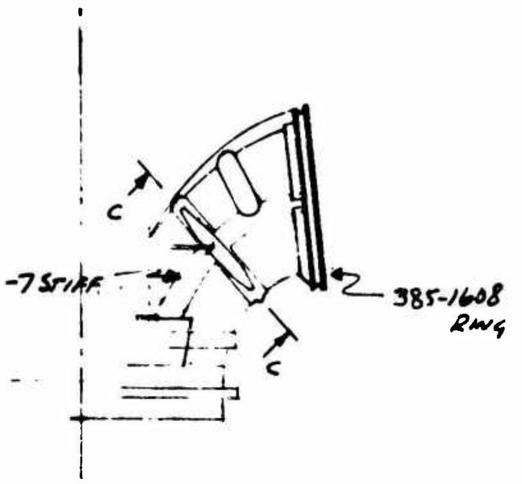
$\frac{F_t}{A} = 16350 \text{ PSI}$

F_t HOOP TENSION = 25000 PSI $\quad M.S. = \frac{25000}{16350} - 1 = +.54$

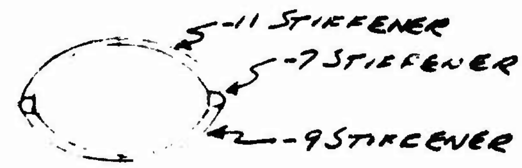
DUCT ASSEMBLY UPPER ROTATING

MATERIAL - INCONEL 718

DWG 385-1607
DWG 385-1605

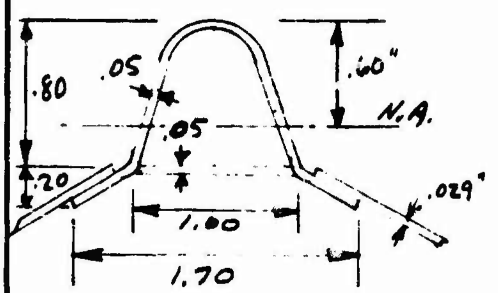


THIS ANALYSIS PERTAINS TO THE METAL DUCTING AND STIFFENERS WHICH ARE PRIMARILY INVOLVED IN THE REDESIGN OF THIS PART FOR WEIGHT SAVING



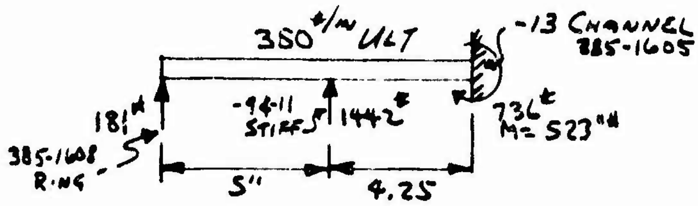
SECTION C.C.

CHECK -7 STIFFENER



$A = .10 \text{ IN}^2 \qquad I = .020 \text{ IN}^3$

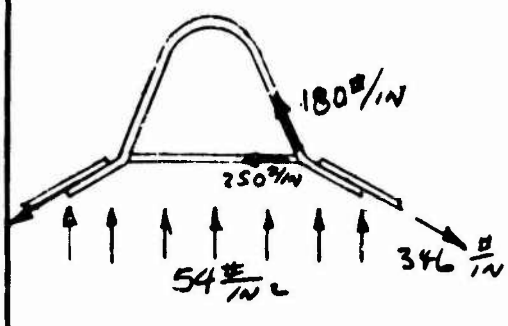
$R = 6.4" \qquad PR = 6.4 \times 54 = 346 \, \#/\text{IN}$



M @ 9-11 STIFF SUPPORT = 554"#

$\frac{MC}{I} = \frac{554(.60)}{.02} = 16600 \text{ PSI}$

$M.S. = \frac{50000}{16600} - 1 = \text{LARGE}$



TYPICAL SECTION AND LOADING OF -7 STIFFENER

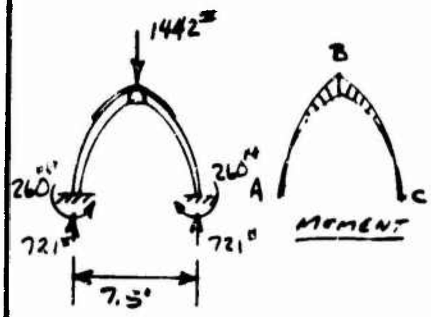
DUCT HOOP TENSION

$\frac{346 \, \#/\text{IN}}{.029 \text{ IN}} = 11900 \text{ PSI}$

$MS = \frac{25000}{11900} - 1 = +1.10$

DUCT ASSEMBLY UPPER ROTATING DWG 385-1607
DWG 385-1605

CHECK -9 & -11 STIFFENER



SECTION C.C.

$I_B = .0128\ IN^4$ $M_B = 2450$ "# $C = .287$

$\frac{Mc}{I} = \frac{2450(.287)}{.0128} = -55000$ PSI @ B AT STRAP

$\frac{Mc}{I} = \frac{2450(.287)}{.0128} = +53400$ PSI @ B @ DUCT SKIN.

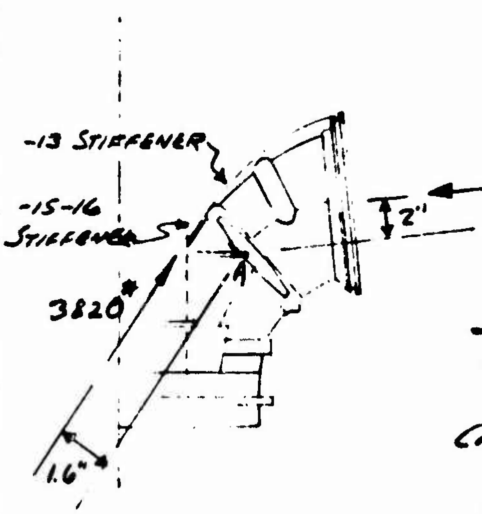
$F_B = 50{,}000$ PSI @ $t = 1100°F$

APPLY BENDING MODULUS FACTOR OF 1.33 FOR EQUIVALENT TUBE OF $D = .6$ & $t = .05$

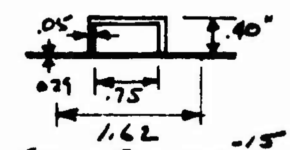
$M.S. = \frac{50000(1.33)}{53000} - 1 = +.25$

CHECK -15-16 STIFFENER

$\Sigma M_A = 0 = 3060(2") - 3820(1.6)$



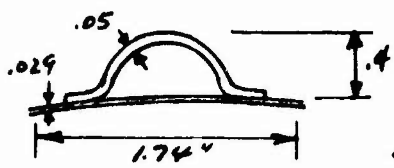
54 PSI (56.7 IN^2) = 3060 # ULT



CROSS SECTION -15 -16

AREA -15 -16 = .119 IN^2

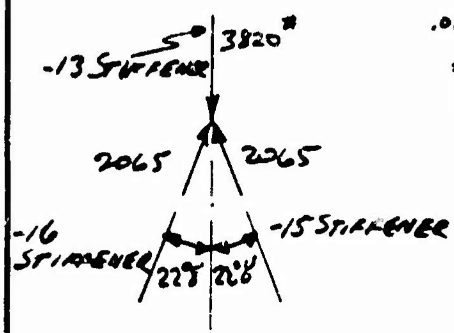
$\frac{P}{A} = \frac{2065}{.119} = 17400$ PSI



CROSS SECTION -13 STIFFENER

AREA -13 = .130 IN^2

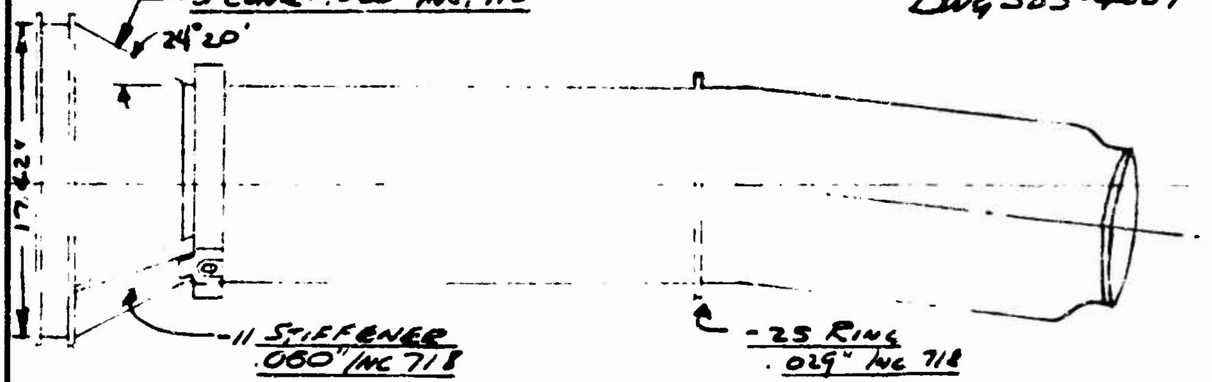
$\frac{P}{A} = \frac{3820}{.130} = 29400$ PSI



$M.S. = \frac{50000}{29400} - 1 = +.70$

ASSY - ENGINE EXHAUST TAIL PIPE WELDED DWG 385-4202

SHEETS 1/2, 2/2
DWG 385-4001



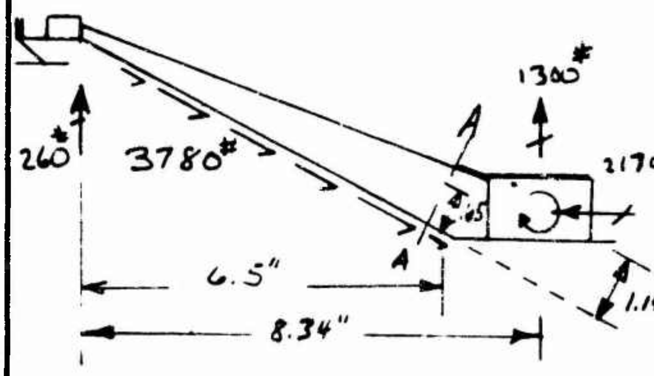
-5 CONE HOOP TENSION

$P = 2(27) = 54$ PSI BURST PRESSURE

$$f_t = \frac{PR}{t} = \frac{54(8.71)}{.020} = 23500 \text{ PSI}$$

$$M.S. = \frac{25000}{23500} - 1 = +.06$$

-11 STIFFENER



ASSUME 50% ECCENTRIC STIFFENER LOAD GOES OUT AS A COUPLE

$$M = \frac{1}{2}(3780^{\#})1.15'' = 2170''^{\#}$$

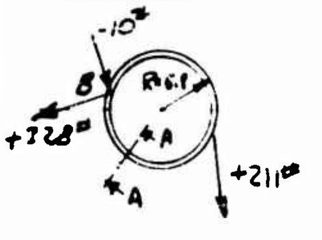
$M_{AA} = 450''^{\#}$ $P_{AA} = 3290$

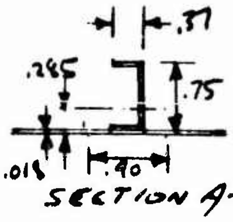
$I_{AA} = .0312$ $AREA = .2105$

$$\frac{MC}{I} + \frac{P}{A} = \frac{450(.65)}{.0312} + \frac{3290}{.2105} = 25000$$

$$M.S. = \frac{50000}{25000} - 1 = +1.00$$

-25 RING





$I = .0059$ $A = .057$ $C = .483$

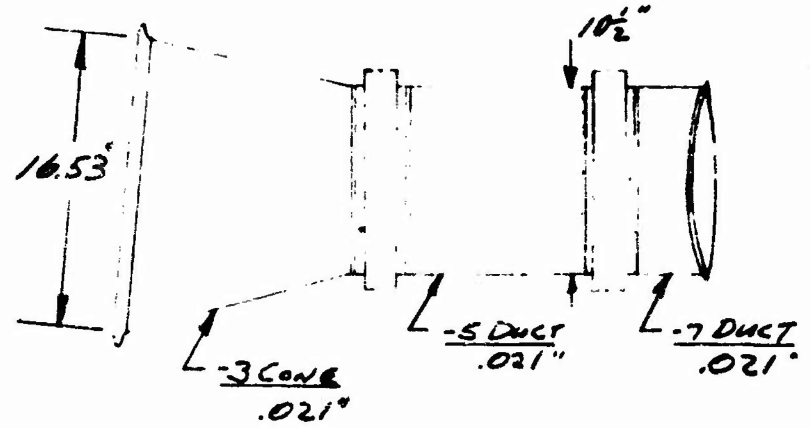
$P = 85^{\#}$ $M_{MAX} @ B = 420''^{\#}$

$$\frac{P}{A} + \frac{MC}{I} = \frac{85}{.057} + \frac{420(.483)}{.0059} = 36000 \text{ PSI}$$

$$M.S. = \frac{50000}{36000} - 1 = +.39$$

TRANSITION DUCT ASSEMBLY - HOT GAS SYSTEM

DWG 385-4112



MATERIAL
INCONEL 718

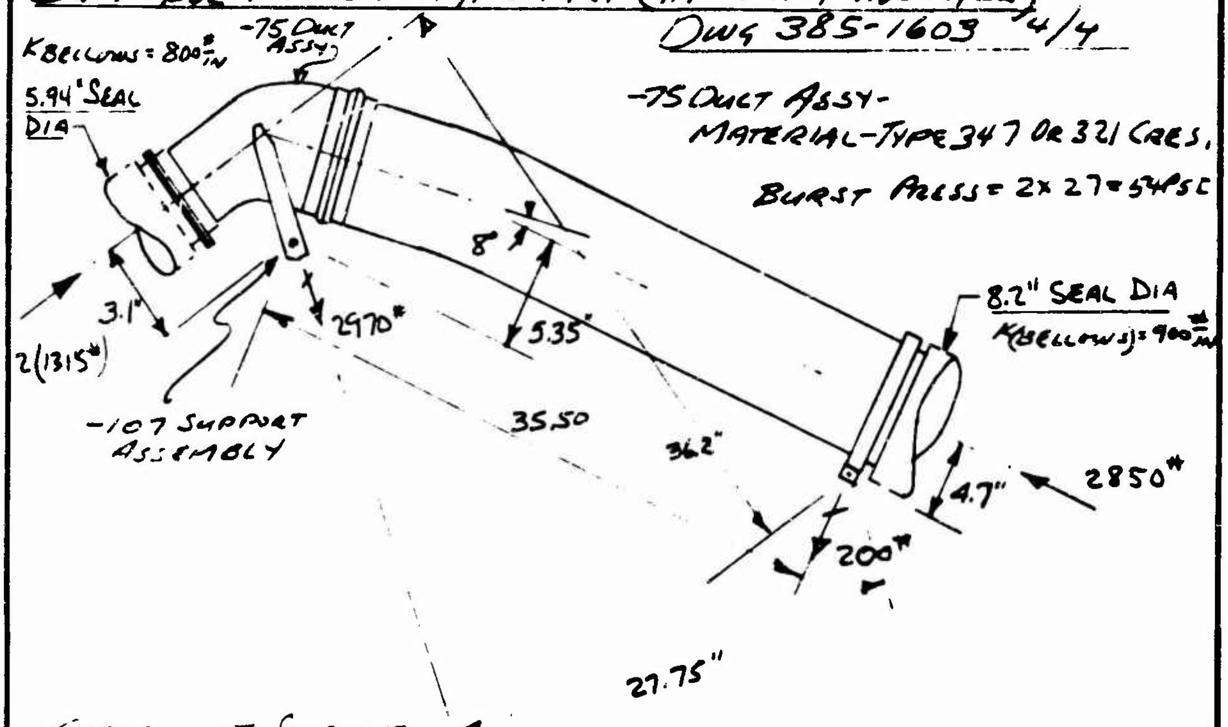
BURST PRESSURE
$= 2 \times 27\ PSI = 54\ PSI$

$f_t = \frac{PR}{t} = \frac{54(16.53)}{.021 \cdot 2}$

$= 21300\ PSI$

$M.S. = \frac{25000}{21300} - 1 = +.17$

DUCT ASSEM LOWER STATIONARY (YAW DUCT INSTALL)
DWG 385-1603 4/4



-75 DUCT ASSY -
MATERIAL - TYPE 347 OR 321 CRES.
BURST PRESS = 2 × 27 = 54 PSI

CHECK -107 SUPPORT

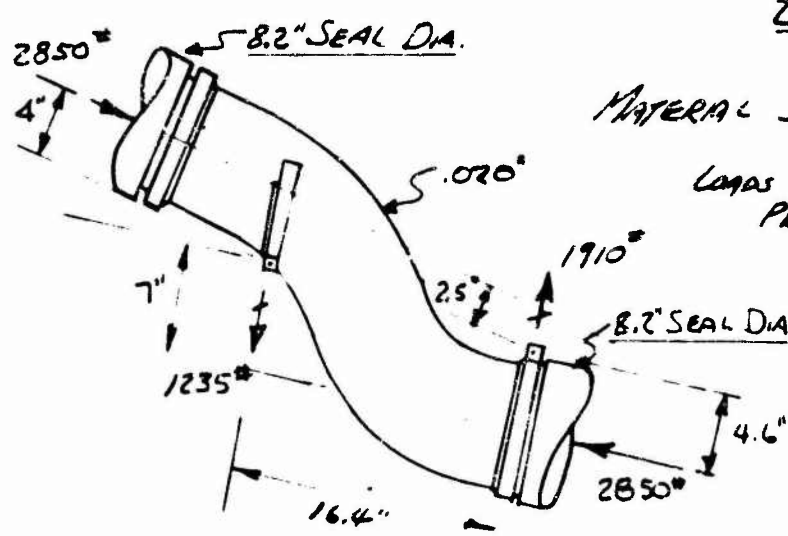
$P = \frac{2970}{2} = 1485\ \#/SIDE$

BOLT BEARING $A = .25(.032 + .050)$
$= .020\ IN^2$

$f_{brg} = \frac{1485}{.020} = 74250\ Psi$ $\quad 74250\ PSI \frac{1.33}{2} = 49500\ PSI = f_{brg}$

REF. MIL HDBK-5 USE AISI 301 A VALUES $\quad M.S. = \frac{50000}{49500} - 1 = +.01$

DUCT ASSEMBLY YAW CONTROL SUPPLY "S" SECTION
DWG 385-4323



MATERIAL 347 CRES. SHT.

LOADS BASED ON BURST PRESSURE 2 X 27 PSI = 54 PSI

DUCT DIA = 7"
t = 1000°F ASSUMED

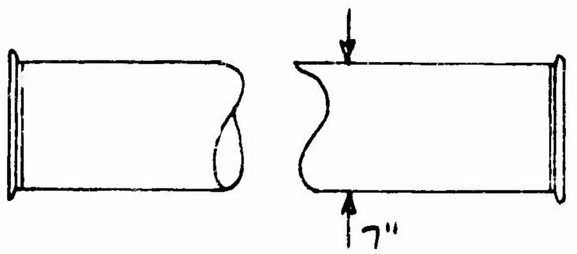
CHECK BENDING ON "S" SECTION   ECCENTRICITY = 2"

$P = (2850^2 + 1235^2)^{1/2} = 3110^{\#}$ $A = .44^{\square"}$ $I = \pi R^3 t = 2.68^{in^4}$ $M = 6220''^{\#}$

$\frac{MC}{I} + \frac{P}{A} = \frac{6220(3.5)}{2.68} + \frac{3110}{.44} = 15200$ PSI

$f_{CR} = K\bar{E}\frac{t}{R} = .3(14.7\times10^6)\frac{.010}{3.5} = 25200$ PSI $M.S. = \frac{25200}{15200} - 1 = +.66$

DUCT ASSEMBLY - YAW CONTROL SUPPLY DWG 385-4322



MATERIAL - 321 CRES - ST
t = .010"

$f_t = \frac{54 \text{ PSI } (3.5)''}{.010''} = 18900$ PSI

$M.S. = \frac{25000}{18900} - 1 = +.32$

STRESS ANALYSIS - FUSELAGE

The cockpit area at the forward end of the fuselage is a modified OH-6A cockpit enclosure. This cockpit extends to fuselage Station 200. Aft of Station 200 the fuselage is designed to the requirements of the XV-9A.

The main load-carrying elements of the fuselage are the two upper and two lower longerons, which are designed to resist all of the fuselage bending moments. The longerons are 7075-T6 aluminum extrusions. The fuselage is covered with stressed skin capable of taking direct and torque shears. The skin is supported by 0.032-inch 2024-T42 aluminum former rings spaced at approximately 8 inches in the forward structure and 10 inches in the aft structure. There are no stringers in the fuselage.

The maximum fuselage bending moments on the fuselage aft of the power module for both positive and negative bending are produced in the maximum autogyro level flight condition for symmetrical tail loading.

The maximum fuselage bending moments forward of the power module are produced by Case 4, two wheel landing with side load on one wheel, and by the maximum autogyro level flight condition for symmetrical down tail load.

The main landing gear fits into a steel tubular shaft located at fuselage Station 238.3. This tube spans across the fuselage and is capable of taking bending moments from the landing gear bearing points. Without this tube, these bending moments would have resulted in much heavier supporting frames. Two heavy frames, at Station 235.10 and Station 241.50, provide the support for this tube.

The top of the fuselage is cut out above the top longeron, between Station 271.50 and Station 321.00, to provide clearance for mounting the power module.

The attachments for the power module are at the upper longeron [illegible]d to the rigid fuselage bulkheads at Station 278.81 and Station 316 [illegible]. The attachment is such that forward and aft loads go directly into the upper longerons.

In this cutout region are two 6-inch-deep shelves that extend from bulkhead Station 271.50 to Station 321.00, one on each side of the fuselage. These shelves, acting with the main attachment bulkheads, distribute side loads to the top and bottom fuselage skins.

The vertical power module loads are distributed from the attachment to the side skins between the upper and lower longerons by the bulkheads at Station 278.81 and Station 316.51.

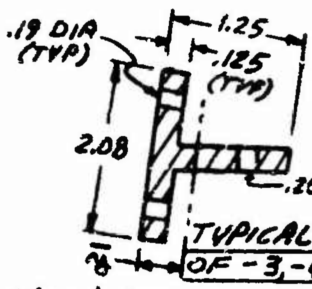
The tailwheel is mounted to the fuselage at the rigid bulkhead at fuselage Station 581.00. The oleo strut is attached to the bulkhead at Station 616.50. The loads from the tailwheel determine the design of these two bulkheads and of the side skin for vertical shear between these bulkheads.

The formers aft of fuselage Station 587.5 are 0.040 inch thick. The increased gage is required because the outside skin aft of this area is 0.040 inch thick, to resist torsional shear stresses.

In this area, the cross section is tapering down, resulting in high torsional shear stresses. The highest torque on the aft fuselage is from the maximum autogyro level flight asymmetrical loading condition.

LONGERONS (DWG No. 385-2001)

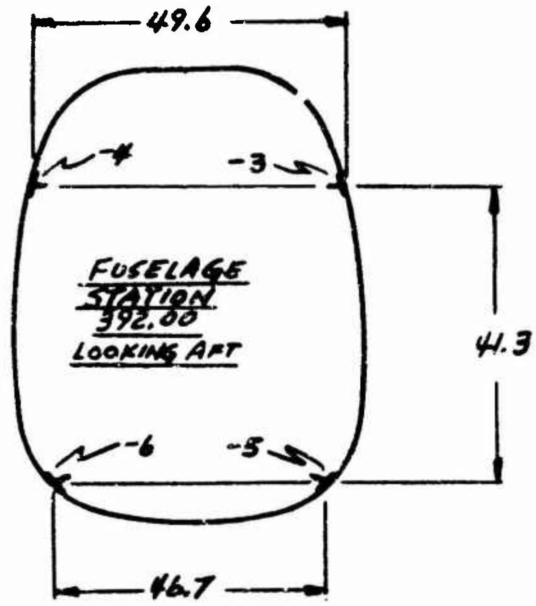
MAXIMUM VERTICAL FUSELAGE BENDING MOMENTS (LIMIT) OCCUR AT STATION 318.00 (−870,000 IN#) AND AT STATION 392.00 (+810,000 IN#). (REF. PG 213)
MAXIMUM SIDE FUSELAGE BENDING MOMENTS (LIMIT) OCCURES AT STATION 392.00 (+550,000 IN#). (REF. PG 217)



TYPICAL SECTION OF −3, −4, −5 & −6

MAT'L: 7075-T6 ALUM. ALLOY EXTRUSION (PIONEER PA10939)

$F_{tu} = 78{,}000$ PSI
$F_{cy} = 71{,}000$ PSI



VERTICAL BENDING (−3 & −4 IN COMP, −5 & −6 IN TEN)

$$P_c = P_t = \frac{81000(1.5)}{2(41.3)} = 14{,}700^{\#} \text{ (ULT)}$$

−3 & −4

$A_c = (2.08 + 1.125)(.125) = .40 \text{ IN}^2$, $\bar{y} = .282$ IN.

$I = .051 \text{ IN}^4$, $L = 7.75$ IN, $L'/\rho = \dfrac{7.75}{\left(\frac{.051}{.40}\right)^{1/2}} = 21.7$

$F_{co} = F_{cy}\left(1 + \dfrac{\sqrt{F_{cy}}}{2000}\right) = 80{,}400$ PSI, $E = 10.3(10)^6$ PSI

$$F_c = F_{co}\left[1 - \frac{.272(L'/\rho)}{\pi (E/F_{co})^{1/2}}\right] = 67{,}000 \text{ PSI}$$

$f_c = \dfrac{P_c}{A_c} = \dfrac{14700}{.40} = 36{,}800$ PSI

$M.S. = \dfrac{67000}{36800} - 1 = \underline{\underline{+.82}}$

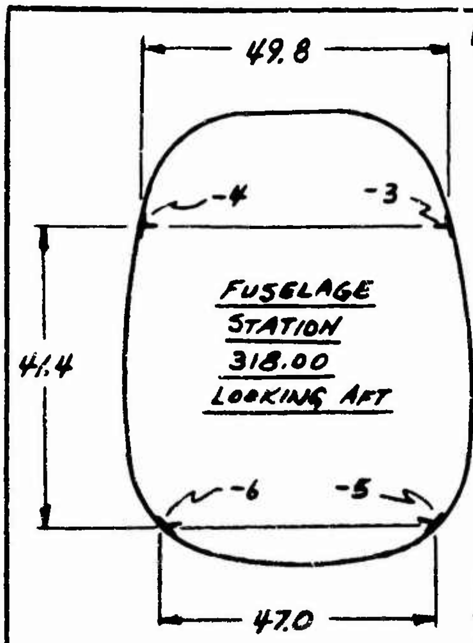
−5 & −6

$A_t = .32 \text{ IN}^2$

$f_t = \dfrac{P_t}{A_t} = \dfrac{14700}{.32} = 46{,}000$ PSI

$M.S. = \dfrac{78000}{46000} - 1 = \underline{\underline{+.70}}$

THE STRESS FROM SIDE BENDING IS LOWER THAN FROM VERTICAL BENDING.



VERTICAL BENDING (-3 & -4 IN TEN., -5 & -6 IN COMP.)

SECTION OF -3 & -4 SAME AS AT STATION 392.00

$$P_c = P_t = \frac{870000(1.5)}{41.4(2)} = 15{,}760^{\#}\text{(ULT)}$$

-3 & -4

$$A_t = .32\ IN^2$$

$$f_t = \frac{P_t}{A_t} = \frac{15760}{.32} = 49{,}300\ PSI$$

$$M.S. = \frac{78000}{49300} - 1 = +\underline{.58}$$

-5 & -6

SECTION OF -5 & -6 SAME AS AT STATION 392.00 EXCEPT 2.08 DIM. IS 2.22

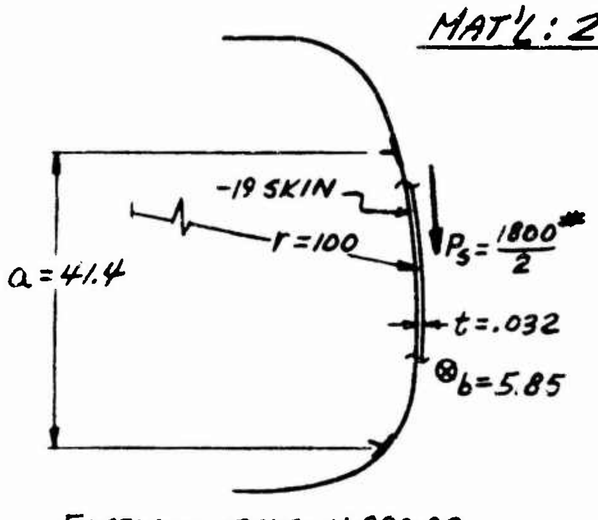
$L = 3.5\ IN$, $M.S. = \underline{+HIGH}$

FUSELAGE SKINS (DWG. NO. 385-2200)

MAXIMUM VERTICAL SHEAR OCCURS AT STATION 280.00 (18,000# LIMIT) (REF. PG 211)

-19 SKIN

MAT'L: 2024-T3 ALUM. ALLOY (QQ-A-362)



FUSELAGE STATION 280.00

$$E = 10.5(10)^6\ PSI$$

$$F_{CR} = .10E\left(\frac{t}{r}\right) + 5.8\left(\frac{t}{b}\right)^2 E$$

$$= 10.5(10)^6\left[.10\left(\frac{.032}{100}\right) + 5.8\left(\frac{.032}{5.85}\right)^2\right]$$

$$= 2{,}160\ PSI$$

$$f_s = \frac{P_s}{A} = \frac{900(1.5)}{41.4(.032)}$$

$$= 10{,}200\ PSI\ (ULT)$$

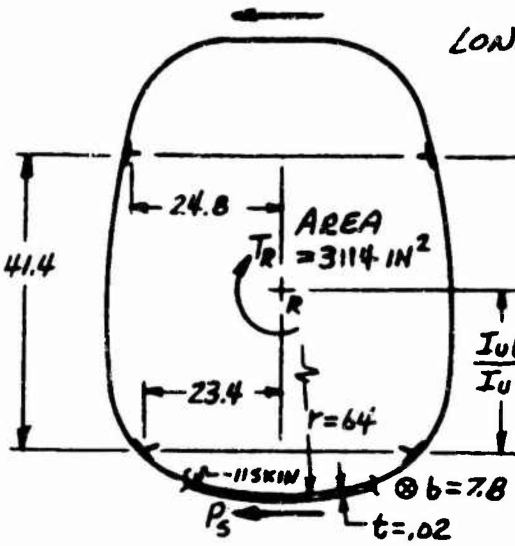
$$F_s = 20{,}000\ PSI$$

$$M.S. = \frac{20000}{10200} - 1 = +\underline{.96}$$

SIDE SHEAR & TORQUE

AT FUS. STA. 245.00 { SHEAR = 6,230 # (LIMIT)
TORQUE = 480,000 IN# (LIMIT) (ABOUT FRP 21.02) } REF. PG 216 & 218

LONGERON AREA (TYP) = .125(1.57 + .625)
= .27 IN²



$I_U = 2(.27)(24.8)^2 = 334\ IN^4$

FRP + 1.36

$\dfrac{I_U(41.4)}{I_U + I_L} = \dfrac{334(41.4)}{334 + 296} = 21.9\ IN$

$I_L = 2(.27)(23.4)^2 = 296\ IN^4$

FUSELAGE STATION 245.00

$T_R = 480000 - 6230(21.02 - 1.36) = 357{,}500\ IN\#$

$P_S = \dfrac{19.5(6230)}{41.4} = 2{,}935\#$

-11 SKIN

MAT'L: 2024-T3 ALUM. ALLOY (QQ-A-362)

$E = 10.5(10)^6\ PSI$

$F_{CR} = .10E\left(\frac{t}{r}\right) + 5.8\left(\frac{t}{b}\right)^2 E = .10(10.5)(10)^6\left(\frac{.02}{64}\right) + 5.8(10.5)(10)^6\left(\frac{.02}{7.8}\right)^2$

$= 729\ PSI$

$f_S = \dfrac{P_S}{A_S} + \dfrac{T_R}{2tA} = 1.5\left[\dfrac{2935}{.02(52)} + \dfrac{357500}{2(.02)(3114)}\right]$

$= 8{,}540\ PSI\ (ULT)$

$F_S = 20{,}000\ PSI$ M.S. = + HIGH

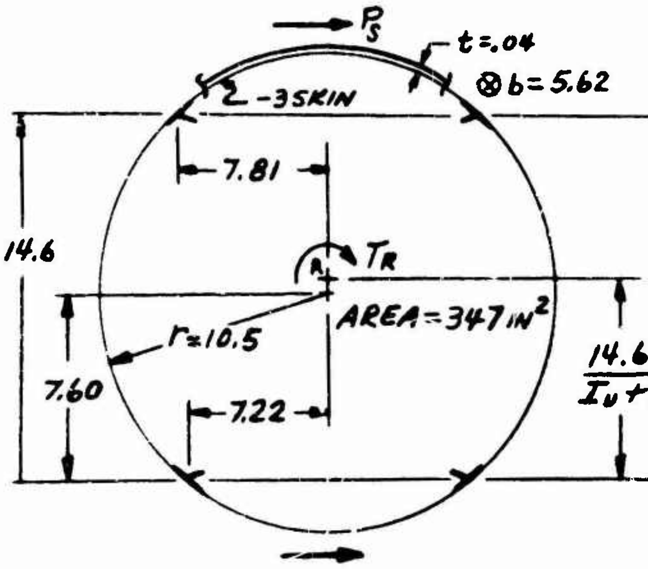
AFT FUSELAGE SKINS (DWG NO. 385-2300)

SIDE SHEAR & TORQUE

AT FUSELAGE STATION 632.00

SHEAR = −2,670# (LIM)
TORQUE = 215,000 IN.# (LIM) (ABOUT TAIL CONE AXIS).

REF. PG 216 & 218



LONGERON AREA (TYP) = .27 IN^2

$I_U = 2(.27)(7.81)^2 = 32.9\ IN^4$

$\frac{14.6 I_U}{I_U + I_L} = \frac{14.6(32.9)}{32.9+28.2} = 7.86$

$I_L = 2(.27)(7.22)^2 = 28.2\ IN^4$

FUSELAGE STATION 632.00

$T_R = 215000 - 2670(7.86 - 7.60) = 214{,}300\ IN\#$

$P_S = \frac{2670(7.86)}{14.6} = 1{,}438\#$

-3 SKIN

MAT'L: 2024-T3 ALUM. ALLOY (QQ-A-362)

$E = 10.5(10)^6\ PSI$

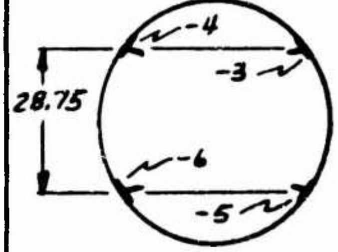
$F_{CR} = .10E\left(\frac{t}{r}\right) + 5.8E\left(\frac{t}{b}\right)^2 = 10.5(10)^6\left[\frac{.10(.04)}{10.5} + 5.8\left(\frac{.04}{5.62}\right)^2\right]$
$= 7{,}100\ PSI$

$f_S = \frac{P_S}{A_S} + \frac{T_R}{2tA} = 1.5\left[\frac{1438}{.04(15.95)} + \frac{214300}{2(.04)(347)}\right]$
$= 15{,}000\ PSI\ (ULT)$

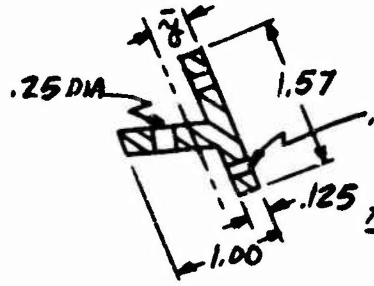
$F_S = 20{,}000\ PSI$

$M.S. = \frac{20000}{15000} - 1 = \underline{\underline{+.33}}$

AFT FUSELAGE LONGERONS (DWG. No. 385-2001)



FUSELAGE STATION 524.00
LOOKING AFT



TYPICAL SECTION OF -3, -4, -5 & -6

MAT'L: 7075-T6 ALUM. ALLOY EXTRUSION (PIONEER PA 10939)

$F_{tu} = 78{,}000$ PSI

$F_{cy} = 71{,}000$ PSI

VERTICAL BENDING

$M = \pm 470{,}000$ IN# (LIM) (REF PG. 217)

$$P_c = P_t = \frac{470000(1.5)}{2(28.75)} = 12{,}250\#\ (ULT)$$

$$A_c = .125(1.57 + .875) = .305\ IN^2$$

$\bar{y} = .24$ IN, $I = .025$ IN, $L = 10.00$ IN, $L/\rho = \dfrac{10}{\left(\frac{.025}{.305}\right)^{1/2}} = 34.9$

$F_{co} = 80{,}400$ PSI, $E = 10.3(10)^6$ PSI

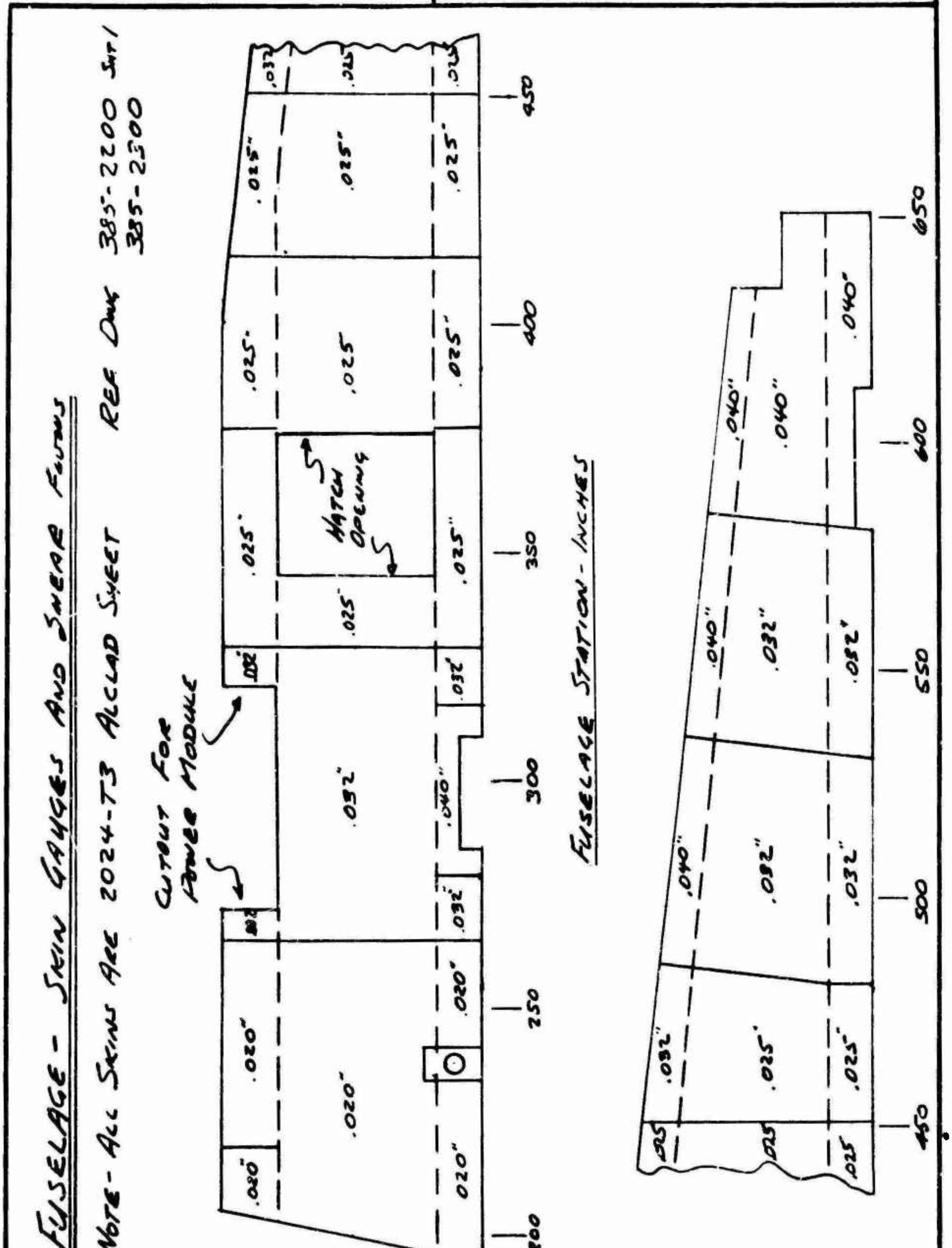
$$F_c = F_{co}\left[1 - \frac{.272(L/\rho)}{\pi (E/F_{co})^{1/2}}\right] = 59{,}000\ PSI$$

$$f_c = \frac{P_c}{A_c} = \frac{12250}{.305} = 40{,}200\ PSI$$

$$M.S. = \frac{59000}{40200} - 1 = \underline{\underline{+.47}}$$

$$A_t = .305 - .125(.25 + .375) = .227\ IN^2$$

$$f_t = \frac{P_t}{A_t} = \frac{12250}{.227} = 54{,}000\ PSI \qquad M.S. = \frac{78000}{54000} - 1 = \underline{\underline{+.44}}$$


FUSELAGE - SKIN GAUGES AND SHEAR FLOWS
NOTE - ALL SKINS ARE 2024-T3 ALCLAD SHEET
REF DWG 385-2200 SHT 1
385-2300
CUTOUT FOR POWER MODULE
HATCH OPENING
FUSELAGE STATION - INCHES
200
250
300
350
400
450
500
550
600
650

FUSELAGE - ANALYSIS OF SKIN SPLICES AND ATTACHMENTS

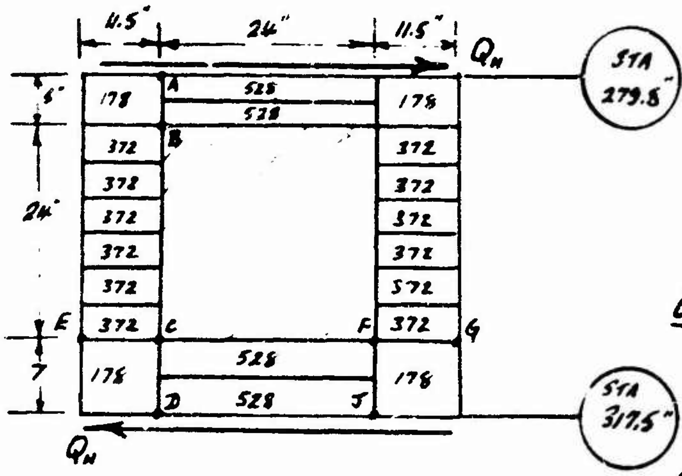
REF 385-2200 SHT 1, 385-2300 — F_S ALLOW = 20000 PSI (SKIN)

	PANEL STA	t"	CONDITION	$q_{MAX\,ULT}$	f_s	ATTACHMENT	ALLOW	M.S.
TOP SKIN ABOVE LONGERONS	200-264.93	.020	COND. 8C - CRASH 4g SIDE	227 #/IN	11350	AD4@3/4	303 #/IN	+.16
	264.93-329.13	.032	COND. 8C - CRASH 4g SIDE	227 #/IN	7100	AD4@3/4"	497 #/IN	+1.19
	329.13-451.	.025	COND 12 - ASYMMETRICAL TAIL LD	100 #/IN	4000	AD4@3/4	497 #/IN	+2.55
	451.0-483.	.032	COND. 12 - ASYMMETRICAL TAIL LD.	132 #/IN	4130	AD4@3/4	497 #/IN	+1.69
	483-633.12	.040	COND 12 - ASYMMETRICAL TAIL LD.	726 #/IN	18200	AD[illegible] 3/4	795 #/IN	+.02
SIDE SKIN BETWEEN LONGERONS	200-264.93	.020	COND. 4'-2WHEEL LANDING W/H SIDE LOAD	295 #/IN	13000	AD4@3/4	303 #/IN	+.08
	264.93-329.13	.032	COND. 8L - CRASH 10g FWD.	632 #/IN	19750	DOUBLE ROW AD4@3/4	954 #/IN	+.51
	329.13-483.0	.025	COND 12. ASYMMETRICAL TAIL LOAD	145 #/IN	5800	AD4@3/4	497 #/IN	+1.45
	483.0-581.0	.032	COND. 12. ASYMMETRICAL TAIL LOAD	210 #/IN	6560	AD4@3/4	497 #/IN	+.69
	581.0-650.85	.040	COND. 12.-ASYMMETRICAL TAIL LOAD	521 #/IN	13040	AD5@3/4	795 #/IN	+.52
BOTTOM SKIN BETWEEN LONGERONS	200-264.93	.020	COND. 4'-2WHEEL LANDING W/H SIDE LOAD	237 #/IN	11200	AD4@3/4	303 #/IN	+.12
	264.93-278.81	.032	COND. 4' - 2WHEEL LANDING W/H SIDE LD	224 #/IN	7000	AD4@3/4	497 #/IN	+.58
	316.51-329.13	.032	COND. 4' - 2WHEEL LANDING - SIDE LD	26 #/IN	812	AD4@3/4	497 #/IN	+12.65
	483.0-581.0	.032	COND 12 - ASYMMETRICAL TAIL LOAD	122 #/IN	3820	AD4@3/4	497 #/IN	+1.90
	329.13-483.0	.025	COND 12 ASYMMETRICAL TAIL LOAD	62 #/IN	2480	AD4@3/4	497 #/IN	+4.71
	278.81-316.51	.04	COND 4' 2WHEEL LANDING - SIDE LOAD	528 #/IN	13200	AD5@3/4	795 #/IN	+.50
	581.0-650.85	.04	COND 12. ASYMMETRICAL TAIL LOAD	369 #/IN	9230	AD5@3/4	795 #/IN	+1.15

FUSELAGE SKIN ANALYSIS

BOTTOM FUSELAGE CUT-OUT STA 279.8" TO 317.5"

DWG Nº 385-2200 SH. 2



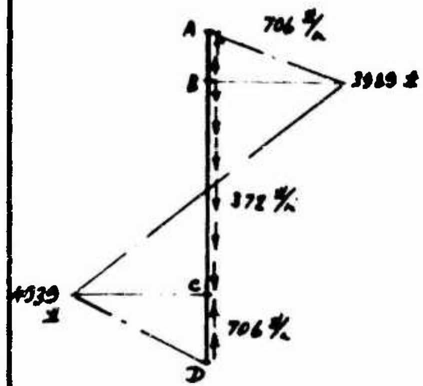
CUT-OUT SHOWN SHADED IN SKETCH

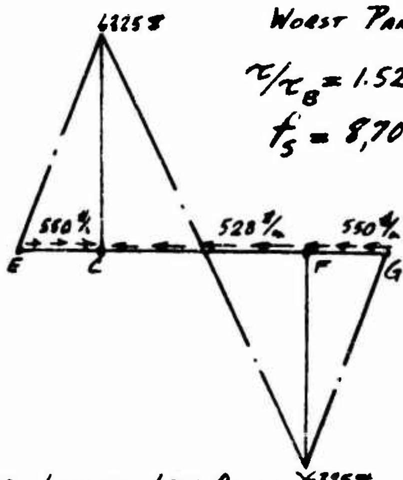
NUMBERS IN PANELS ARE SHEAR FLOWS IN LBS/INCH (ULTIMATE VALUES)

CRITICAL CASE:- MAX SIDE SHEAR LANDING CASE #4

$Q_H = 8550$ # ULT (SEE PAGE 218)

END LOADS IN MEMBERS



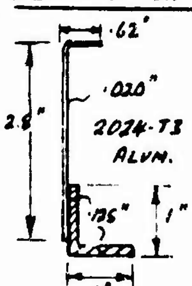


WORST PANEL $q = 528$ #/in

$\tau/\tau_B = 1.52$ $K = .09$

$f_s = 8{,}700$ #/in² M.S. > +1.00

SECTION ON ECFG (ABCD SIMILAR, LOADING LESS CRITICAL)



$A = .302$ in² $P = 6{,}325$ # $P/A = 20{,}943$ #/in²

TENSION FIELD BM. = 570 # INS • Z (SHADED PORTION) = .072 in³

$M/Z = \frac{.7950}{.072} = 7950$ #/in² TOTAL $f_c = 28{,}893$ #/in²

$\rho = .30$ $L = 12$" (EFFECTIVE) $L/\rho = 40$ $F_c = 32{,}000$ #/in² M.S. + .10

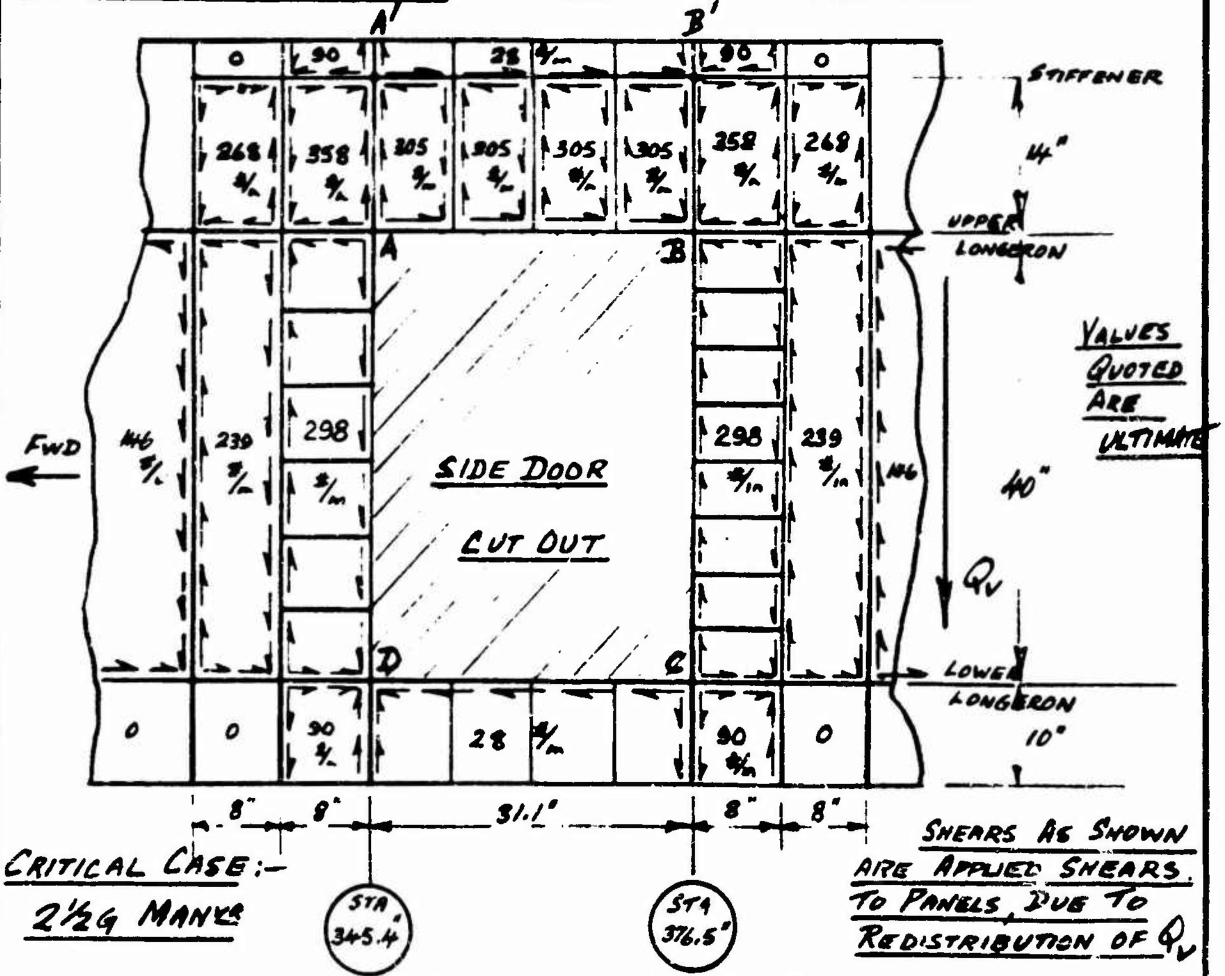
STEEL STRAP ON ABCD (.050" × 1" × 4130 - HT. 150,000)

$P = 4939$ # $A = .050$ $f_c = 98{,}800$ #/in²

$L/t = 22$ INTER-RIVET BUCKLING $F_c = 120{,}000$ M.S. + .21

RIVETS:- MAX $q = 528$ #/in 5/32 2117-T4 @ .75" IN .040" STRENGTH 794 #/in M.S. + .50

CUT OUT IN SIDE SKIN — DWG N° 385-2200 SHT. 1



CRITICAL CASE:— 2½G MANŒ

SHEARS AS SHOWN ARE APPLIED SHEARS TO PANELS, DUE TO REDISTRIBUTION OF Q_V

FROM PAGE 211 $\qquad Q_V = 5824$ #/SIDE (ULT.)

CORNER JOINTS A & B $\qquad P = 5107$ # (IN STEEL STRAP)
$A = .0422$ IN² (NET) $\quad f_t = 121{,}000$ #/in² @ 160,000 $\qquad$ M.S. +.24

WORST SKIN PANEL $\qquad q = 358$ #/in $\quad \tau = 14{,}320$ #/in² $\quad \tau_B = 2{,}900$ #/in²
$\tau_{ALLOW} = 19{,}600$ #/in² $\qquad$ M.S. + .37

UPPER LONGERON @ B $\qquad P = 14{,}930$ # $\quad A = .40$ in² $\quad P/A = 37{,}300$ #/in²

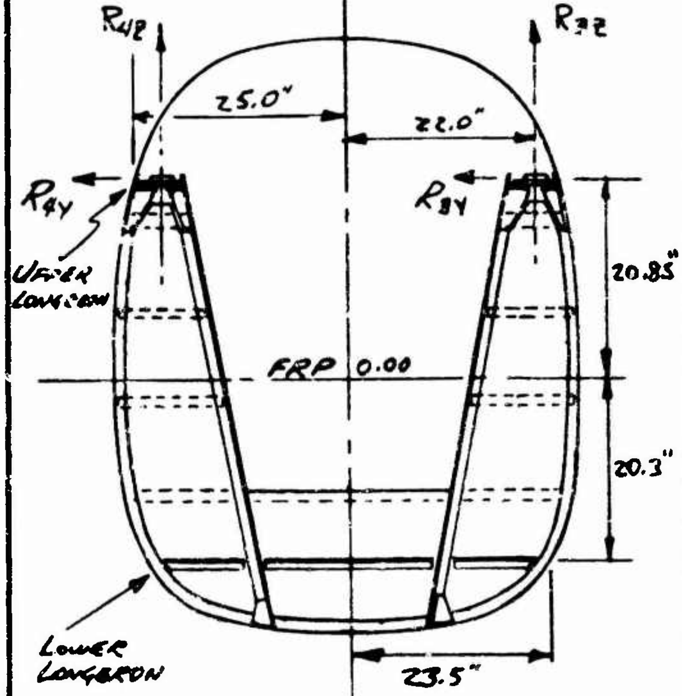
TENSION FIELD LOADING:— $\sigma_c = 14{,}250$ #/in $\quad M = 315$ #.INS. $\quad Z = .0833$ in³ $\quad M/Z = 3780$ #/in²

TOTAL $f_c = 14{,}250 + 3780 + 37{,}300 = 55{,}360$ #/in² @ 70,000 $\qquad$ M.S. +.26
GENERAL INSTABILITY NOT CRITICAL, USE f_{cy}.

RIVETING A.A' & B.B' LOAD = 663 #/in
1/8" @ 1" CC DOUBLE ROW :— IN .025" STRENGTH = 732 #/in $\qquad$ M.S. + .10

MAIN FRAME ASSY STA 279.80 AND STA 317.50

REF DRWG 385-2201



LOAD SUMMARY-REF P. 209

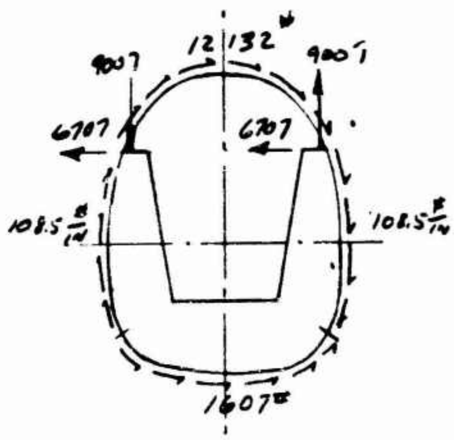
	CASE 8B 10g FWD CRASH	CASE 8C 4g SIDE CRASH
R_{3X}	−16788	−4369
R_{3Y}	0	+6707
R_{3Z}	+26280*	+9007
R_{4X}	−16788	+4369
R_{4Y}	0	+6707
R_{4Z}	+26280*	−9007

* NOTE - TENSION ON STA 317.5 COMP. ON STA 279.8

THE FRAME IS ANALYSED FOR CASE 8C-4g SIDE CRASH

$$\Sigma \text{MOMENTS} = 2 \times 20.85 \times 6707^{\#} + 9007^{\#} \times 22 \times 2 = 676000''^{\#}$$

$$q_T = \frac{T}{2A} = \frac{676000}{2(3114)} = 108.5 \ ^{\#}/_{IN}$$



UPPER SHEAR REACTION

$$= q_T(50') + \frac{2 \times 6707}{2} = 12132^{\#}$$

LOWER SHEAR REACTION

$$= q_T(47) - \frac{2 \times 6707}{2} = +1607^{\#}$$

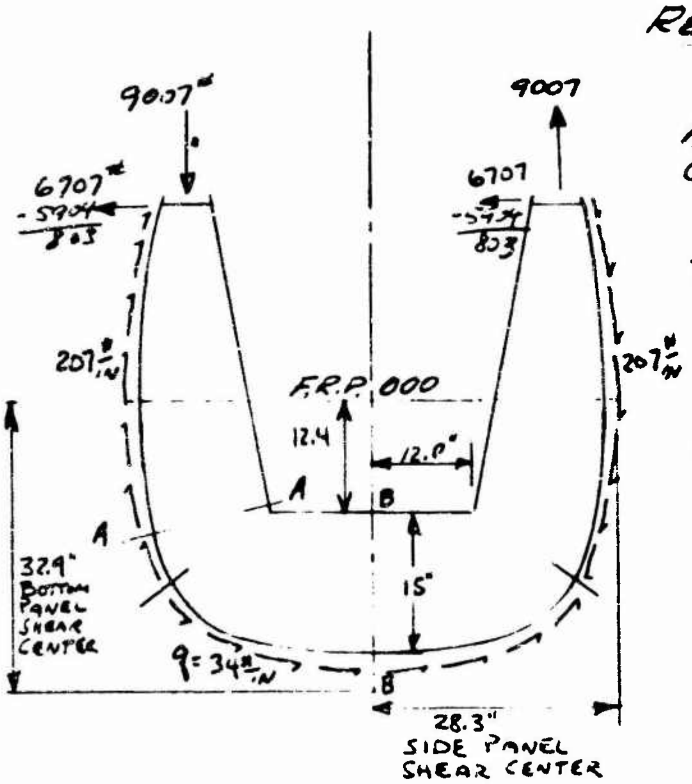
LOAD REACTED IN SHELF AT UPPER LONGERON

$$= R_{3Y} + R_{4Y} - \text{LOWER SHEAR REACTION}$$

$$= 6707 + 6707 - 1607 = 11807 \text{ OR } 5904 \ ^{\#}/\text{SHELF}$$

MAIN FRAME ASSY STA 279.80 AND STA 317.50

REF. DWG 385-2201



ANALYSE FOR CASE 8C 4g SIDE CRASH CONDITION

$T = 803^{\#}(2)(20.85+32.9) + 9007(2)22$

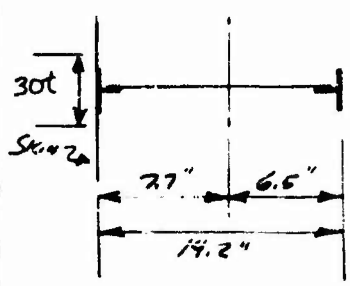
$= 86450 + 396000 = 482450''^{\#}$

$q_{\text{SIDE PANEL}} = \dfrac{T}{2e} \times \dfrac{1}{D} = \dfrac{482450}{2(28.3)41.15}$

$= 207\ ^{\#}/_{IN}$

$q_{\text{BOTTOM PANEL}} = \dfrac{2(803)}{47.0} = 34\ ^{\#}/_{IN}$

CHECK BENDING AND AXIAL STRESSES ON SECTION AA



CROSS SECTION AA

$A = 1.52\ IN^2 \qquad I = 56.3\ IN^3$ ALUMINUM MAT'L 2024-T4

$M_{NA} = \underbrace{9007(22-18.5)}_{-31500} + \underbrace{803(12.4+20.85)}_{-26700} - \underbrace{207(12.4+20.85)(28.3-18.5)}_{+67500}$

$= +9300\ ^{\prime\prime\#}$

$P = -9007 + 207\ ^{\#}/_{IN}(12.4+20.85) = -2107^{\#}$

$\dfrac{MC}{I} + \dfrac{P}{A} = \dfrac{9300(6.5)}{56.3} + \dfrac{2107}{1.52} = \pm 2462$ PSI @ SKIN

M.S. = LARGE

CHECK SHEAR IN WEB AT SECTION BB

$V = -9007^{\#} + 207\ ^{\#}/_{IN}(41.15^{\prime\prime}) - 34\ ^{\#}/_{IN}(8.2^{\prime\prime}) = 1756^{\#}$

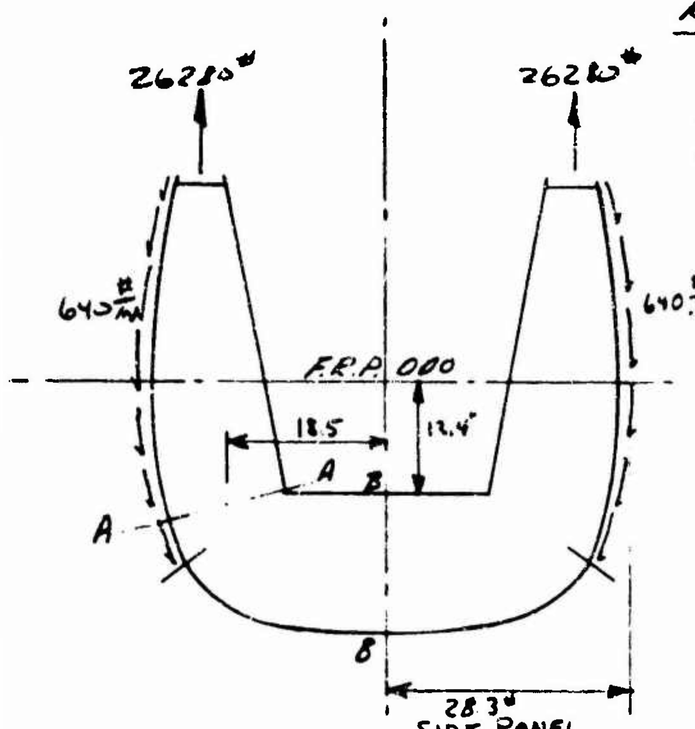
$t_{WEB} = .032''$ 2024-T-3 CLAD ALUMINUM $a/b = \dfrac{12}{7.25} = 1.6,\ K = 7$

$q = \dfrac{1756^{\#}}{15''} = 117\ ^{\#}/_{IN} = 3660$ PSI $\qquad f_{SCR} = KE\left(\dfrac{t}{b}\right)^2 = 7.0(10^7)\left(\dfrac{.032}{7.25}\right)^2 = 1360$ PSI

PANEL WILL BUCKLE FOR TENSION FIELD M.S. = LARGE.

MAIN FRAME ASSY STA. 279.80 AND STA. 317.50

REF. DWG 385-2201



ANALYSE BULKHEAD FOR CASE 88 10g FWD CRASH

$q = \frac{26280}{41.15} = 640 \text{ \#/IN}$

$M_{AA} = 26280(22.0-18.5) - 640(33.25)(28.3-18.5)$

$= 117000 \text{ "\#}$

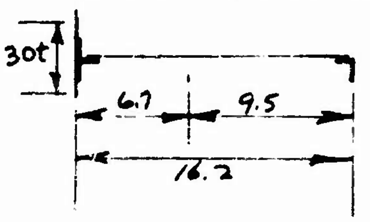
$P = 26280 - 640(33.25) = 4980\#$

$M_{BB} = 26280(22.0-28.3)$

$= 165500 \text{ "\#}$

NOTE - 26280# IS TENSION ON STA 317.5 & COMP. ON STA. 279.8

CHECK SECTION BB



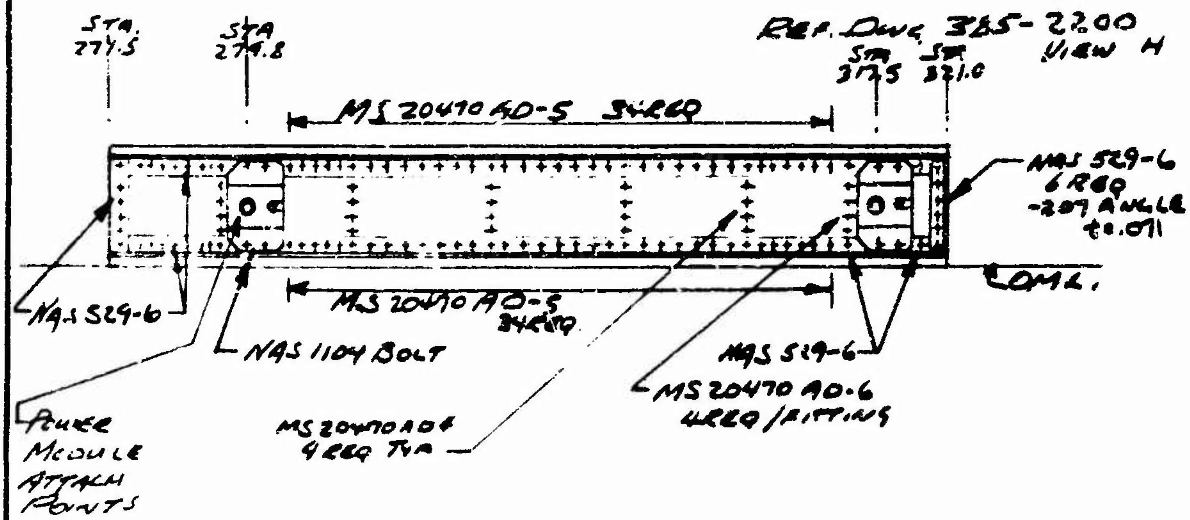
$A = 1.22 \text{ IN}^2$ $I = 51.2 \text{ IN}^3$ MAT'L: ALUMINUM 2024-T4

$\frac{Mc}{I} = \frac{165500(9.5)}{51.2} = 30700 \text{ PSI}$

$MS = \frac{37000}{30700} - 1 = +.20$

POWER MODULE TO FUSELAGE ATTACH.

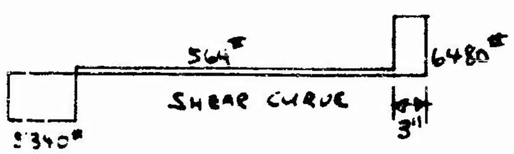
SIDE BEAM FUSELAGE STA 271.5 TO STA 321



CHECK FOR POWER MODULE ATTACH LOADS FOR 4g SIDE CRASH

LOAD FROM FITTING TO .071 WEB
5 NAS 1104 BOLTS = 5(1825) = 9125 BEARING

$M.S. = \frac{9125}{5904} - 1 = +.54$



CHECK ATTACH OF WEB TO CAP BETWEEN STA. 317.5 TO STA 321.

$P = 6480^{\#} \times \frac{3.00}{4.25} = 4575^{\#}$ CHECK BEARING

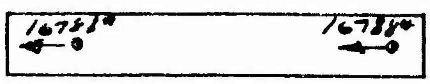
2 NAS 529-6 = 2(1356) = 2712
2 NAS 1104 BOLT = 2(1825) = 3650
6362

$M.S. = \frac{6362}{4575} - 1 = +.39$

CHECK RIVETS TO TAKE 6480 REACTION
6 NAS 529-6 RIVETS = 6(1356) = 8136#

$M.S. = \frac{8136}{6480} - 1 = +.25$

CHECK FOR POWER MODULE ATTACH LOADS FOR 10g FWD CRASH



LOAD FROM FITTING INTO SHELF
4 NAS 1104 BOLTS BEARING IN .071 WEB & .15 LONG.
= 4(5680) = 22700
1 NAS 1104 BOLT BEARING IN .071 WEB = 1825
24525

M.S = +.46

CHECK LOAD FROM WEB TO INNER LONGERON
LOAD ALREADY IN LONGERON BY BEARING OF 5/16" BOLT =

$2\frac{(3860)}{24525} \times 2 \times 16788^{\#} = 10580^{\#}$ LOAD IN WEB = 2(16788) - 10580 = 23000#

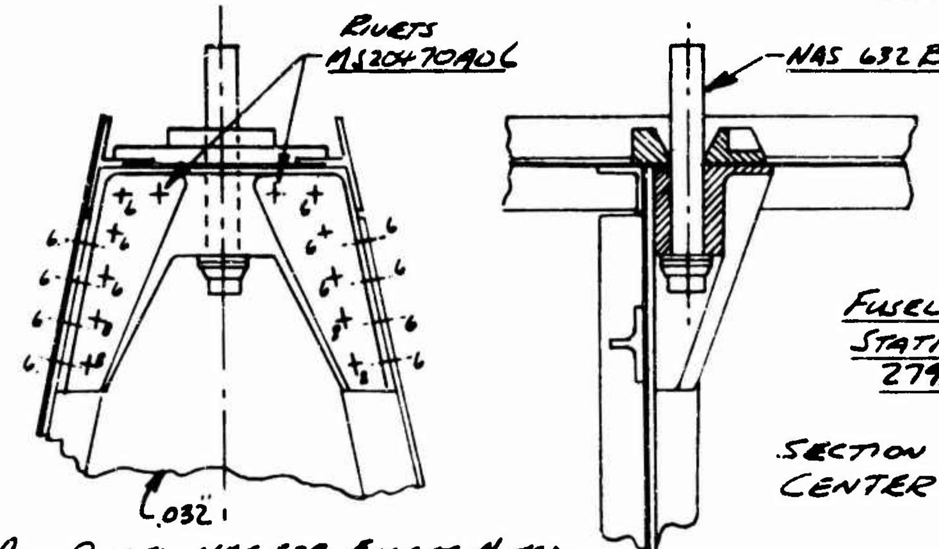
$\frac{23000^{\#}}{49.5 \text{ IN}} = 465 \frac{\#}{\text{IN}}$ AD6 @ 1" BEARING .071 RIVET SHEAR CRITICAL

$M.S. = \frac{862}{465} - 1 = +.85$

POWER MODULE TO FUSELAGE ATTACH

MAIN FRAME ASSY STA. 279.80 AND STA. 317.50

REF. DRWG 385-2202
385-2207



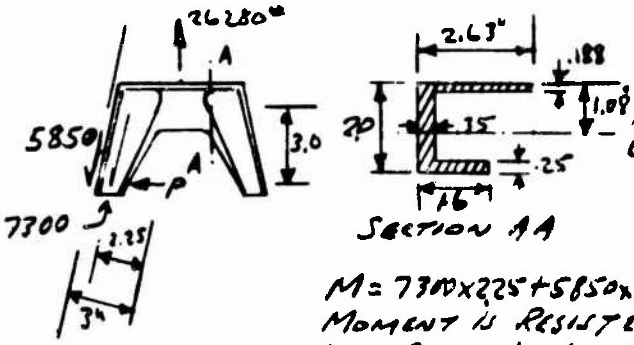
SECTION THROUGH CENTER OF FITTING

ALL RIVETS NAS 529 EXCEPT NOTED

NOTE - REF. LOADS FROM PAGE 209
R_{3Y} AND R_{3X} LOADS HAVE BEEN ANALYSED FOR LOADING UP THE SHELF. THE PORTION REMAINING IN THE BULKHEAD IS PASSED BY INSPECTION

CHECK FITTING FOR R_{3Z} LOAD TAKEN BY FITTING

$R_{3Z} = +26280^{\#}$ ULT. TEN. STRENGTH BOLT = $71100^{\#}$



ASSUME THE LOAD GOES OUT IN PROPORTION TO THE NUMBER OF RIVETS

$\frac{5}{9} \times 13140 = 7300$ $\frac{4}{9} \times 13140 = 5850$

$M = 7300 \times 2.25 + 5850 \times 3.0 = 24650''^{\#}$ ASSUME $\frac{1}{2}$ THIS MOMENT IS RESISTED IN BENDING ON SECTION AA AND OTHER $\frac{1}{2}$ IS RESISTED BY P.

$P = \frac{24650}{2} \times \frac{1}{3} = 4100^{\#}$

LOAD ON NAS 8 RIVETS $= \left[\left(\frac{4100}{2}\right)^2 + \left(\frac{7300}{5}\right)^2\right]^{1/2} = 2520$ $F_s = 3680$

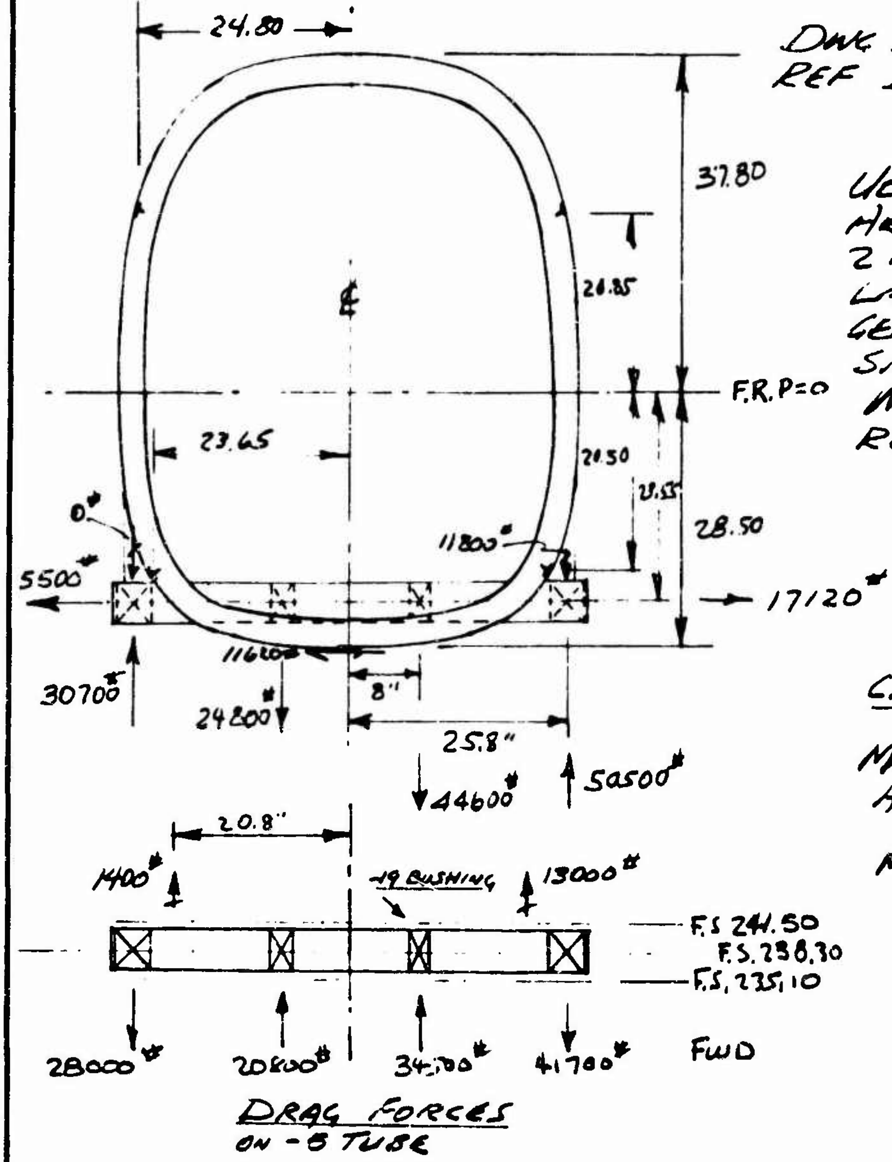
$M.S. = \frac{3680}{2520} - 1 = +.46$

SECTION AA.

$I = .82$ $A = 1.44 \text{ in}^2$ $\frac{Mc}{I} + \frac{P}{A} = \frac{24650(1.08)}{.82} + \frac{4100}{1.44} = 19100 \text{ psi}$ M.S. = HIGH

LANDING GEAR SUPPORT INST. MAIN GEAR

DWG 385-2209
REF DWG 385-2208



ULTIMATE LOADS ARE SHOWN FOR 2 POINT LEVEL LANDING ON MAIN GEAR WITH OUTBOARD SIDE LOAD ON ONE WHEEL. CONDITION 4
REF PG 200 & 204

CHECK -5 TUBE

MAT'L 4135 STL TUBE
H.T. 180-200000 PSI

$M_{MAX} = (684^2 + 576^2)^{1/2} \times 10^3$

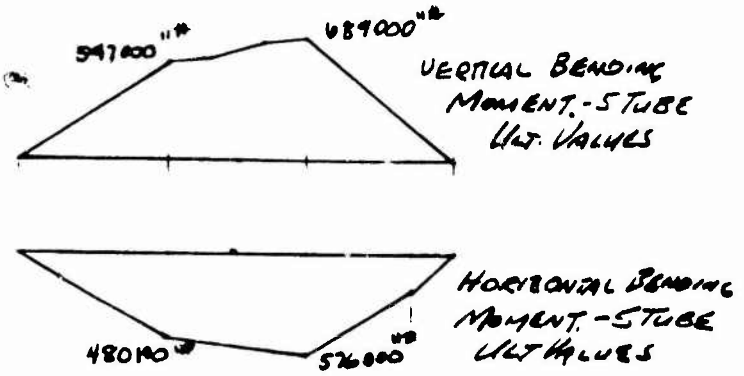
$= 898000$

O.D = 4.750"
I.D. = 4.120"
t = .315" A = 4.4 in²

$I = 10.89 \quad D/t = 15.06$

$\frac{Mc}{I} = 196000$ PSI

$F_B = 238000$ PSI

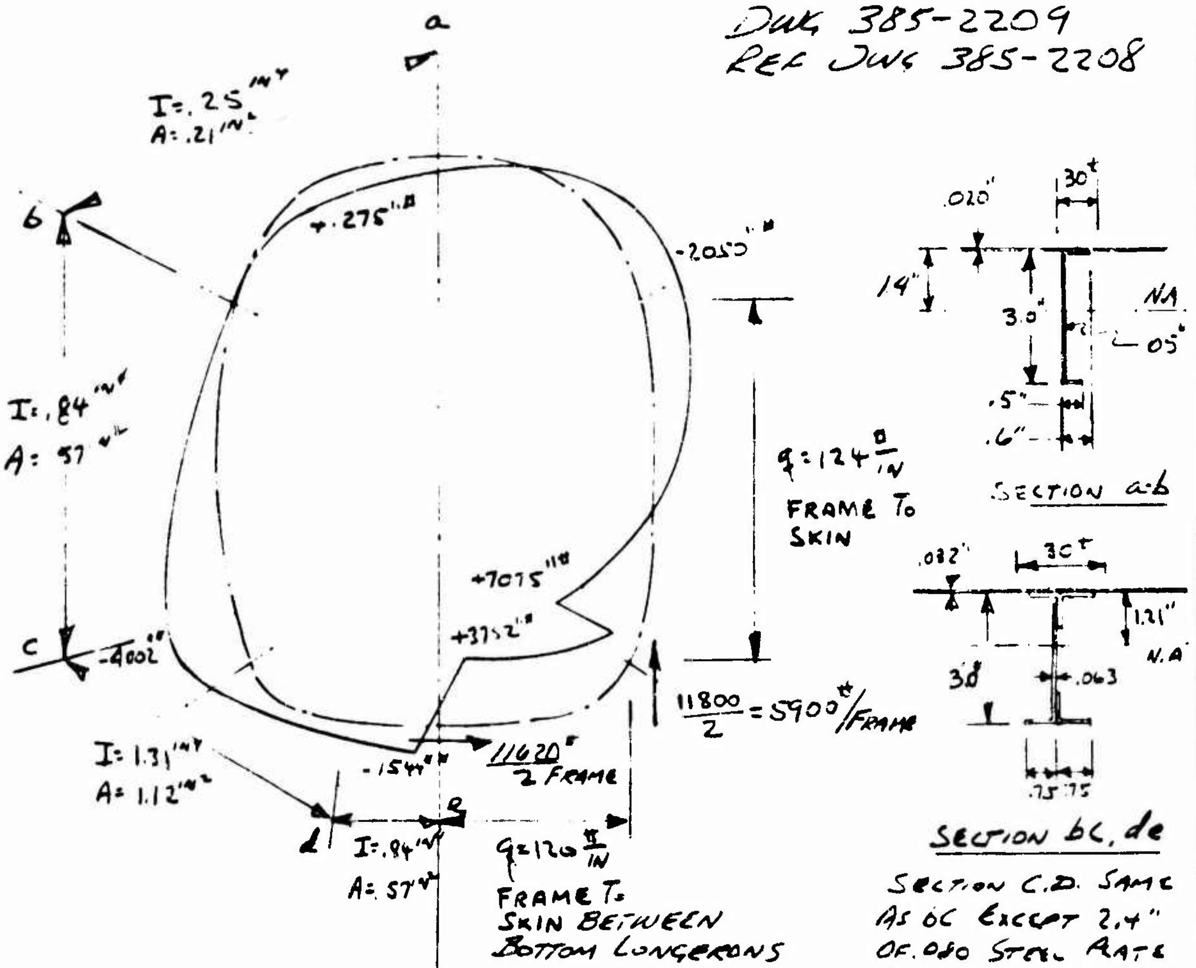


$M.S. = \frac{238000}{196000} - 1$

$M.S. = +.21$

LANDING GEAR SUPPORT INST. MAIN GEAR

DWG 385-2209
REF DWG 385-2208



SECTION C.D. SAME AS bc EXCEPT 2.4" OF .080 STEEL PLATE ACTS WITH THE SECTION

IT IS ASSUMED THAT THE LANDING GEAR LOADS DISTRIBUTE EQUALLY TO THE FRAMES AT STA. 235.0 & STA 241.50. THE MOMENT CURVE SHOWN IS TYPICAL FOR EITHER FRAME.

$$\frac{MC}{I} + \frac{P}{A} = \frac{7075(1.21)}{.84} + \frac{5900^{\#}}{.57} = -20550\ \#/in \qquad M.S. = \frac{41000}{20550} - 1 = +1.00$$

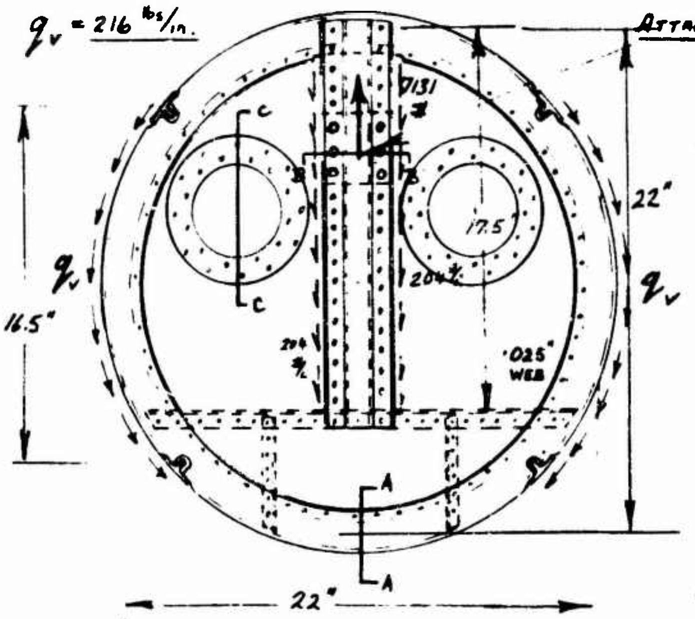
CHECK SHEAR OUT OF -19 BUSHING — MAT'L. ALUMINUM BRONZE

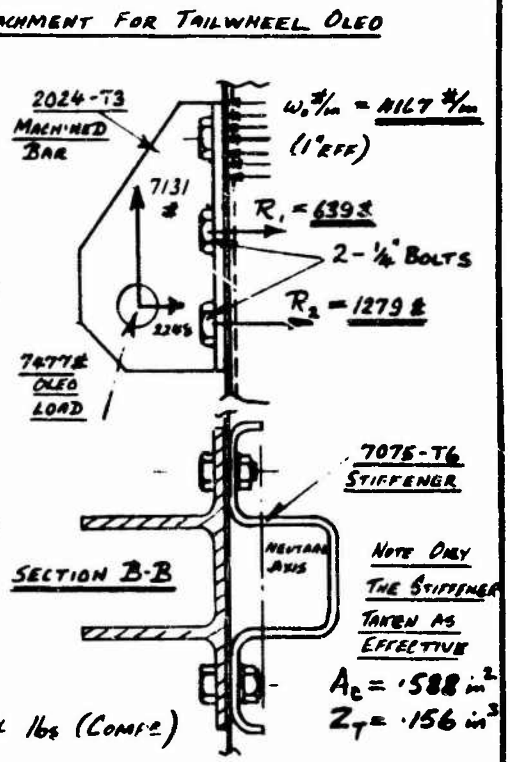
$$A_s = 2\left[2 \times .25 \times .35 + .25 \times .375\right] = .54\ in^2$$

$$f_s = \frac{17120}{.54} = 32000\ PSI \qquad M.S. = \frac{41000}{32000} - 1 = +.28$$

BULKHEAD @ F/S STA 616.50 DWG. Nº 385-2304 SHT. 1

ULTIMATE LOADS CRITICAL CASE TOWING b(i) (PAGE 205)





SECTION BB $M = 12,479$ lbs. ins. $P = 2574$ lbs (COMPR.)

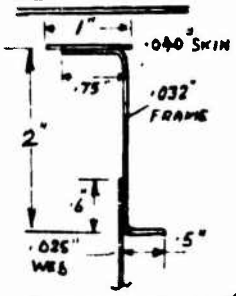
MAX TENSION $M/Z + P/A = 80,108 - 4378 = 75,730$ #/in² @ 77,000 #/in² M.S. +.01

STRENGTH OF STIFFENER FLANGE IN COMPRESSION = 2125 # (EACH)
LOAD = 4167 # TOTAL = 2084 # PER FLANGE M.S. +.02

STRENGTH OF STIFFENER FLANGE IN TENSION = 749 #/inch (EACH)
LOAD = 1279 #/in TOTAL = 639 #/inch (EACH) M.S. +.16

ATTCH OF STIFFENER TO WEB :- 5/32 - 2117-T3 RIVETS @ .75" STRENGTH = 533 #/in
LOADING = 204 #/in M.S. >+1.0

SECTION A-A



AREA = .040 + .104 + .015 = .159 in²

$$P_{AA} = \frac{7131}{2} \times \frac{11}{22} = 1783 \text{ \# (COMPR)} \qquad f = \frac{1783}{.159} = 11,200 \text{ \#/in}^2$$

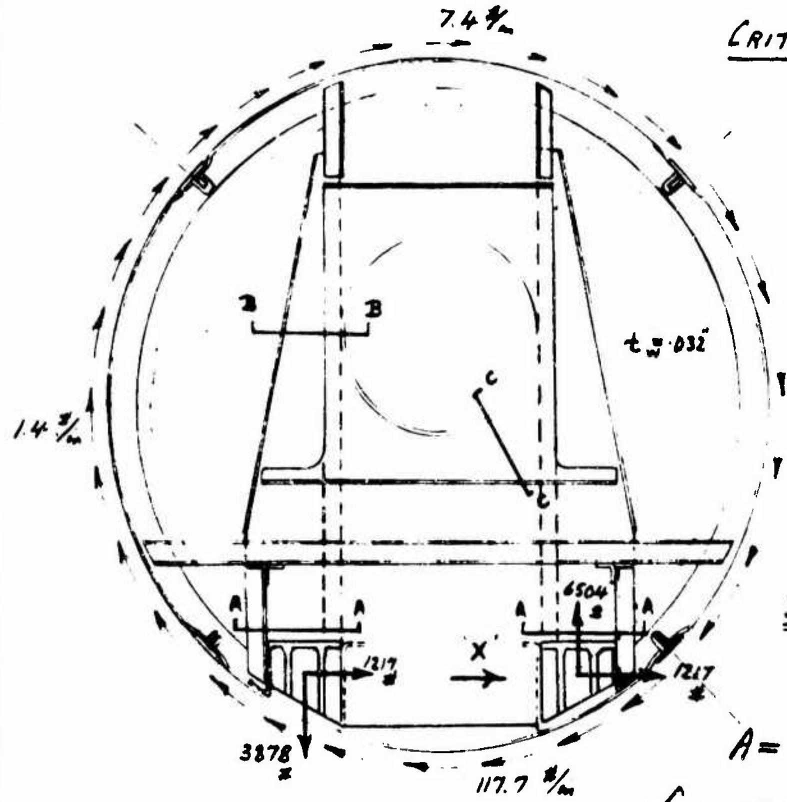
FLANGE CRIPPLING $b/t = \frac{.75}{.032} = 23.4$ $f_{cc} = 15,000$ #/in² M.S. +.34

$b/t = \frac{2.0}{.032} = 62.5$ $f_{cc} = 15,000$ "

SECTION C-C (ACROSS HOLES) $f_s = 9,850$ #/in²
ALLOWABLE = 10,000 #/in² M.S. +.07

TAIL GEAR ATTACHMENT

ATTACHMENT FOR TAILWHEEL YOKE



BULKHEAD @ 581"

CRITICAL CASE :- TOWING C(i)

SECTION A·A

P = 6504 lbs (COMPS)

A = .274 in²

f_c = 23,700 #/in²

FLANGE CRIPPLING f_{cc} = 23,700 #/in²

M.S. 0.00

SECTION BB

P = 2120 # (LANDING 6(iii))

A = .118 (EQUIVALENT) in²

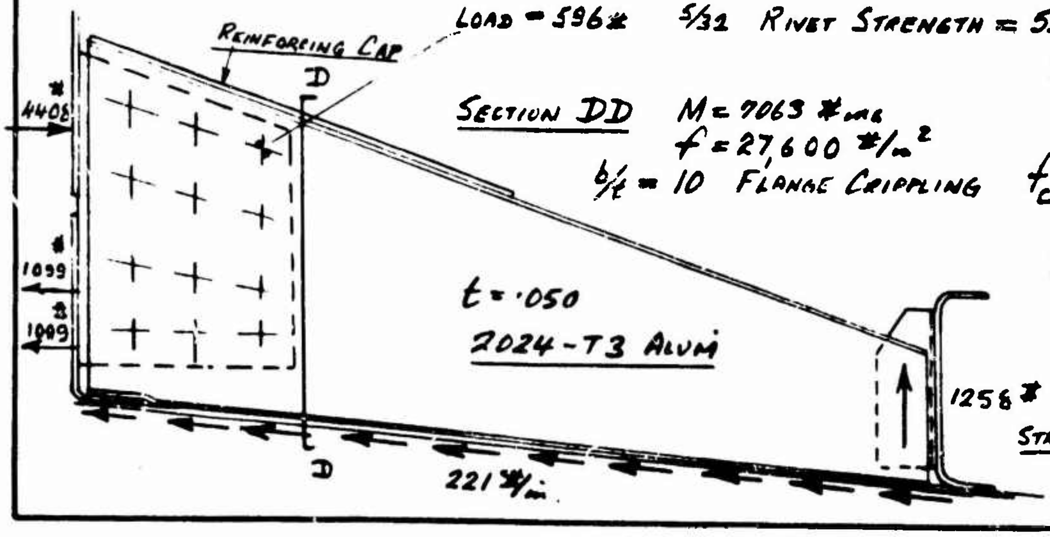
f = 17,905 #/in² @ 32,000 M.S.+.70

SECTION C-C (CRITICAL IN LANDING CASE 6(iii) TEMP 400°F ON FLANGE)

M_{cc} = 1625 # ins f = 25,375 FLANGE CRIPPLING = 32,000 #/in² (@ 400°F)

M.S. + .26

VIEW ON ARROW X



LOAD = 596 # 5/32 RIVET STRENGTH = 596 #

M.S. 0.00

SECTION DD M = 7063 # ins

f = 27,600 #/in²

b/t = 10 FLANGE CRIPPLING f_{cc} = 37,500

M.S. + .34

ATTCH. STRENGTH 1400 #

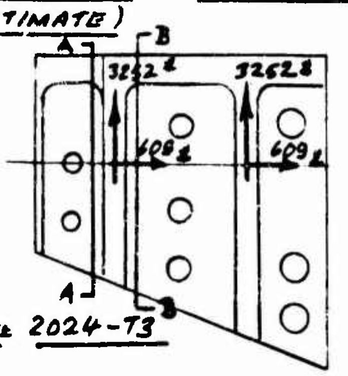
MS+.11

ATTACHMENT FOR TAILWHEEL YOKE – CONTINUED

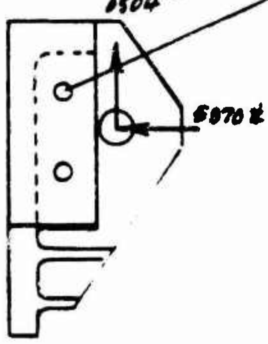
THE MARGINS OF SAFETY QUOTED ON THE PR[illegible]OUS PAGE ARE BASED ON TOWING LOADS DERIVED FROM REF. 26. THE WORST LANDING CONDITION WOULD BE TAIL FIRST WITH SIDE LOAD (#6 iii) WHICH IS LESS CRITICAL THAN THE WORST TOWING CASE, WHICH IS TOWING WITH TAIL WHEEL SWIVELLED 45° FROM FWD. POSITION WITH HELICOPTER AT DESIGN GROSS WEIGHT (15,300 LBS)

FITTING FOR YOKE ATT^CH

MAX LOADS – TOWING CASE C(i)
(ULTIMATE)

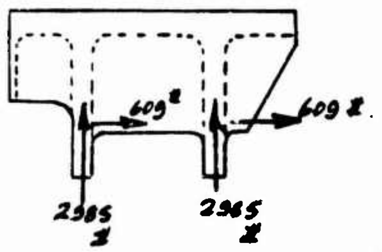


MAT'L 2024-T3



10.32 SCREW (H.T. 160,000)
LOAD = 2720 lbs
STRENGTH = 2720 "

M.S. 0.00



SECTION A.A

HORZ. M_{AA} = 2445 #ins $Z = .035\ in^3$

VERT. M_{AA} = 2220 # ins $Z = .570\ in^3$

TORSION T_0 = 2537 #ins $Z_T = .073\ in^3$

MAX BENDING f = 47,000 #/in² @ 64,000 $R_A = .350$

" TORSION f_s = 34,600 " " 37,000 $R_s = .934$

INTERACTION $R_A^2 + R_s^2 = 1$

M.S. 0.00

SECTION B-B

HORZ. M_{BB} = 3677 #ins $Z = .440$

VERT M_{BB} = 3554 #ins $Z = .857$

TORSION T_0 = 2286 #ins $Z_T = .072$

MAX BENDING f = 12,500 #/in² @ 64,000 $R_A = .195$

" TORSION f_s = 31,700 " @ 37,000 $R_s = .857$

INTERACTION $R_A^2 + R_s^2 = 1$

M.S. +.12

STRESS ANALYSIS - EMPENNAGE

The stabilizer portion of the empennage consists of two cantilever spars and two torsion boxes (nose and aft). The spar caps are 7075-T6 aluminum extrusions and are riveted to 0.040-inch-thick 2024-T3 aluminum webs. The spar shears are carried by partial diagonal tension fields, with stiffeners provided by the rib attachment flanges and, where necessary, at midpoints between ribs. The torsion on the structure is carried wholly by the skins as shear (partial diagonal tension fields), except toward the root, where differential bending becomes significant as a result of the axial constraint of the spar caps. The torsion box carries on through the center section, with the root ribs taking out the main attachment loads to the fuselage. There are four hinge ribs upon which the rudder is mounted, and these are made up of two standard ribs riveted back to back. The nose ribs are all identical, except for those closing off the ends.

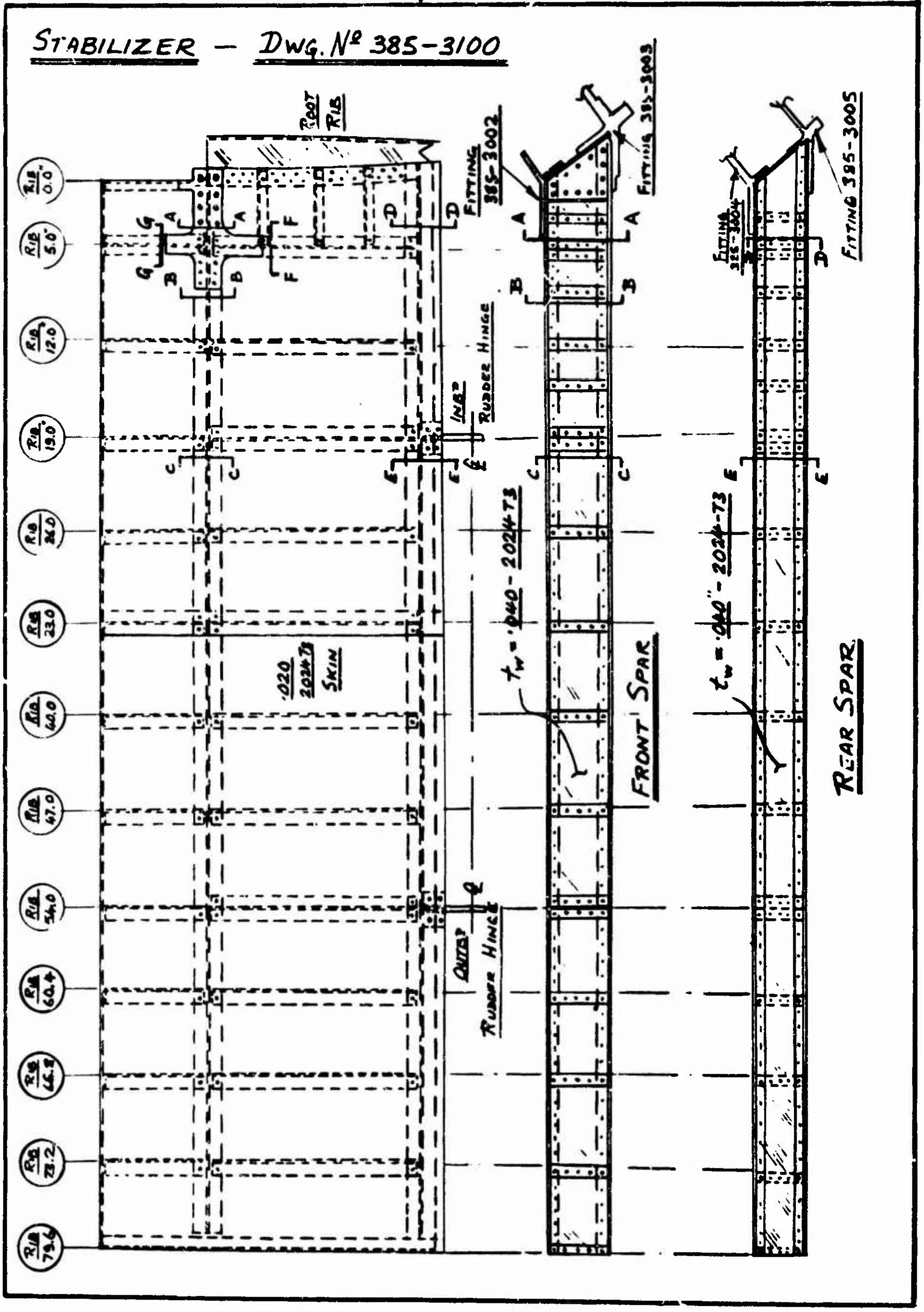
Each rudder is a single-spar two-cell torsion box, supported at four hinge points and with the control torque tube taken off the root rib. The spar is made up of 0.040-inch 2024-T3 aluminum channel, with doublers riveted to the flanges to form the caps. The nose torsion box is interrupted at each hinge, but the aft torsion box is continuous. The skins forming these boxes are 0.016-inch-thick 2024-T3 aluminum, and the torsion shears are carried as partial diagonal tension field.

The design cases for which the tailplane has been analyzed are:

A	(i)	symmetric with center pressure at 0.25 chord
A	(ii)	symmetric with center pressure at 0.50 chord
B	(i)	unsymmetric with center pressure at 0.25 chord
B	(ii)	unsymmetric with center pressure at 0.50 chord

The airloads for these conditions are found under Structural Design Criteria, Section 7 of this report. The spanwise distribution for all cases was assumed to be uniform (that is, center pressure at midspan). The chordwise distribution was taken as parabolic for center pressure at 0.25 chord and trapezoidal for center pressure at 0.50 chord. The effects of inertia have been neglected, as they would be small compared with airloading. A further case, parked tail-to-wind at 40 knots, was investigated, but did not prove to be more critical than the cases above, except locally at the trailing edge of the rudder.

STABILIZER


STABILIZER — DWG. No 385-3100
ROOT RIB
FITTING 385-3002
FITTING 385-3003
FITTING 385-3004
FITTING 385-3005
INBD RUDDER HINGE
OUTBD RUDDER HINGE
.020 2024-T3 SKIN
t_w = .040 - 2024-T3
t_w = .040" - 2024-T3
FRONT SPAR
REAR SPAR
RIB 0.0"
RIB 5.0
RIB 12.0
RIB 19.0
RIB 26.0
RIB 33.0
RIB 40.0
RIB 47.0
RIB 54.0
RIB 60.4
RIB 66.8
RIB 73.2
RIB 79.6

STABILIZER — DWG Nº 385-3100

FRONT SPAR — CRITICAL SECTIONS —(TENSION FIELD EFFECTS INC.)

SECTION A-A:— UPPER CAP $P = 32,680$ lbs. (COMPR) $f_c = 55,500$ #/in²
CASE A(i) MAT'L 7075-T6(EXT'D) ALLOWABLE $f_c = 65,000$ #/in²
M.S. +.18

LOWER CAP $P = 32,318$ lbs (TENS.) $f_t = 74,550$ #/in²
MAT'L 7075-T6 (EXT'D) ALLOWABLE $f_{tu} = 78,000$ #/in²
M.S. +.05

SECTION B-B:— UPPER CAP $P = 28,188$ lbs (COMPR) $f_c = 55,900$ #/in²
CASE A(i) MAT'L 7075-T6(EXT'D) ALLOWABLE $f_c = 65,000$ #/in²
M.S. +.16

LOWER CAP $P = 27,812$ lbs. (TENS.) $f_t = 69,550$ #/in²
MAT'L 7075-T6(EXT'D) ALLOWABLE $f_{tu} = 78,000$ #/in²
M.S. +.12

SECTION C-C:— UPPER CAP $P = 19,009$ lbs. (COMPR) $f_c = 43,750$ #/in²
CASE A(i) MAT'L 7075-T6 (EXT'D) ALLOWABLE $f_{tu} = 65,000$ #/in²
M.S. +.48

WEB (0" TO 5") CASE A(i) $q = 970$ #/in $\tau = 24,250$ lbs/in² $\tau_B = 14,400$ #/in²
$\tau/\tau_B = 1.68$ $K = .110$ f_s ALLOWABLE $= 25,500$ lbs/in² M.S. + .05
RIVETS (WEB TO FLANGE) 3/16 (2117-T3) RIVETS IN .040" 2024-T3
STRENGTH = 999 #/in M.S. + .03

REAR SPAR — CRITICAL SECTIONS —(TENSION FIELD EFFECTS INC.)

SECTION D-D:— UPPER CAP $P = 18,300$ lbs (COMPR) $f_c = 71,170$ #/in²
CASE A(ii) MAT'L 7075-T6(EXT'D) ALLOWABLE $f_c = 77,000$ #/in² *
M.S. +.08

LOWER CAP $P = 16,500$ # (TENS) $f_t = 84,600$ #/in²
MAT'L 7075-T6(EXT'D) ALLOWABLE $f_{tu} = 85,800$ #/in² * MS +.010

* THESE ALLOWABLES INCLUDE 1.10 FACTOR (PLASTIC BENDING)

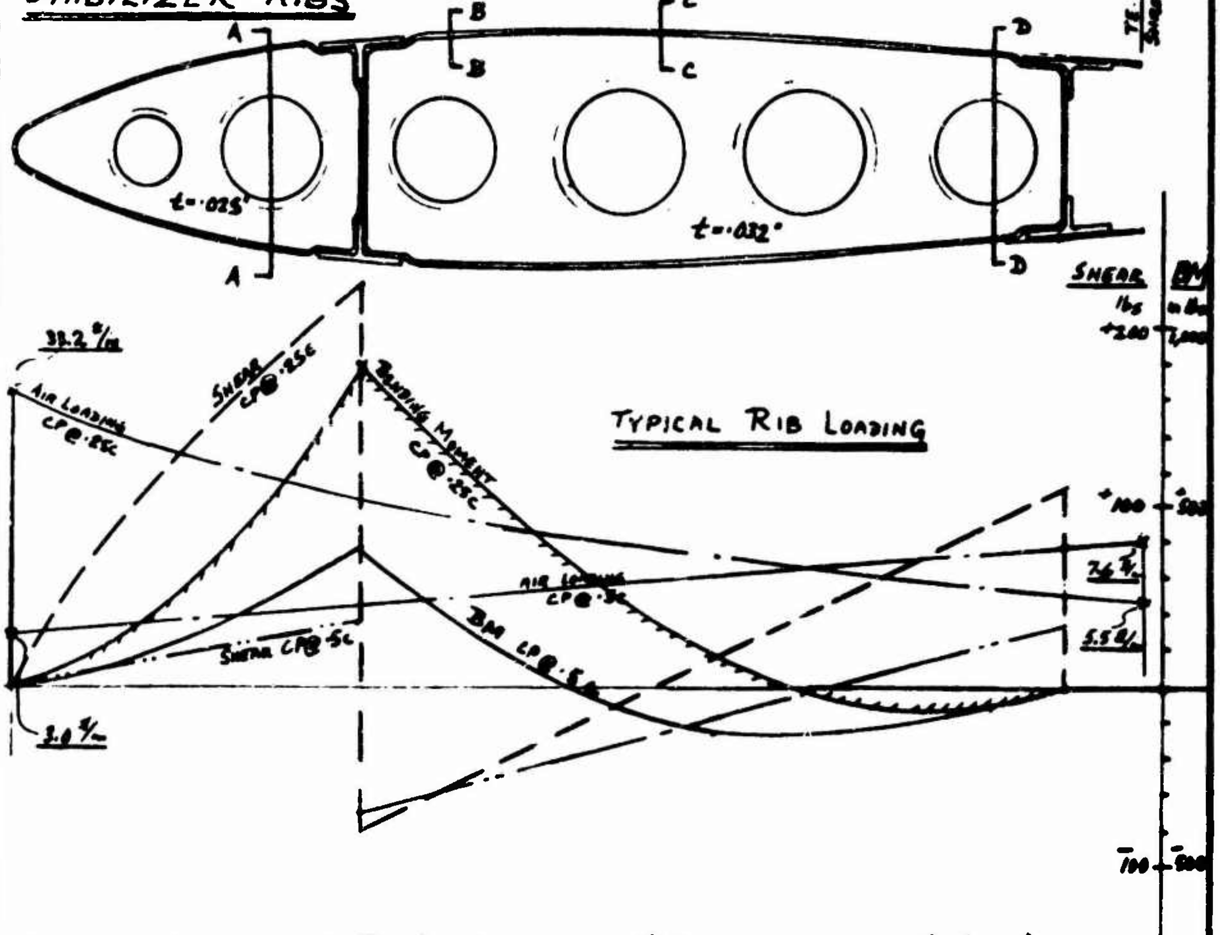
WEB (STA 0" TO 5") CASE A(ii) $q = 790$ #/in $\tau = 19,750$ #/in² $\tau_B = 22,900$ #/in²
$\tau/\tau_B < 1.0$ f_s ALLOWABLE $= 27,500$ #/in²
M.S. +.39

RIVETS (WEB TO FLANGE) 3/16 (2117-T3) RIVETS IN .040" 2024-T3
STRENGTH 999 #/in M.S. +.26

SKIN (0" TO 5") BETWEEN SPARS $q_{MAX} = 278$ #/in (ULT) $t = .020"$
$\tau = 14,000$ #/in² $\tau_B = 457$ #/in² $\tau/\tau_B = 30$ $K = .63$
ALLOWABLE $f_s = 21,000$ #/in² M.S + .50
RIVETS (TO REAR SPAR) 5/32 IN .020 STRENGTH = 432 #/in M.S. +.31
RIVET LOAD = 330 #/in

STABILIZER

STABILIZER RIBS



SECTION A-A $q = 54\ \#/in$ (NOM.) $q_b = 47\ \#/in^2$ (NOM.) $q_{ALL} = 95\ \#/in$ (NOM.)

M.S. + .76

SECTION BB $M = 900$ # INS. $d = 4.5''$ $M/d = 200\ \#$
$A = .050$ $\frac{P}{A} = 4{,}000$ $f_{TF} = 11{,}400\ \#/in^2$ TOTAL $f_c = 15{,}400\ \#/in^2$
(f_{TF} = STRESS DUE TO SKIN TENSION FIELD) $b/t = 20$ FLANGE CRIPPLING = 18,500

M.S. + .17

SECTION CC $M = 350$ # INS. $d = 4.2''$ $M/d = 83\ \#$
$A = .050''$ $\frac{P}{A} = 1660\ \#/in^2$ $f_{T.F.} = 11{,}400\ \#/in^2$ EFF. $L = 12.8''$ $\rho = .21$ $L/\rho = 61$

GENERAL INSTABILITY $f_{cr} = 23{,}000\ \#/in^2$ THIS IS LESS CRITICAL THAN LOCAL INST.

SECTION DD (ON A HINGE RIB) $M/d = 886\ \#$ $A = .111\ in^2$ $\frac{P}{A} = 8{,}000\ \#/in^2$

$f_{TF} = 2/3 \times 11{,}400 = 7{,}600$ TOTAL $f_c = 15{,}600\ \#/in^2$

FLANGE CRIPPLING, $b/t = 20$ $f_{cc} = 18{,}500\ \#/in^2$

M.S. + .18

SHEAR ACROSS HOLE, $q = 156\ \#/in$ $q_{cr} = 195\ \#/in$ $q_{ALLOW} = 414\ \#/in$

(THESE SHEAR FLOWS ARE NOMINAL, BASED ON WHOLE SECTION)

M.S. + 1.65

RIB @ 5.0' TRANSFER OF NOSEBOX TORSION

SECTION GG (PAGE 309) $P = 1246\#$ $A = .086\ in^2$ $P/A = 14{,}500\ \#/in^2$
$b/t = 26$ (ONE EDGE FREE) FLANGE CRIPPLING $f_{cc} = 14{,}500\ \#/in^2$
M.S. 0.00

SECTION F.F (PAGE 309) $P = 1563\#$ $A = .103\ in^2$ $P/A = 15{,}170\ \#/in^2$
SKIN T.F STRESS = 8400 TOTAL $f_c = 23{,}600\ \#/in^2$
FLANGE INSTABILITY $f_{cc} = 27{,}700\ \#/in^2$ M.S. +.17

SECTION ACROSS FRONT SPAR — NOT AS CRITICAL AS ABOVE.

RIB @ 0.0" — ROOT RIB. MAX SHEAR STRESS = $7{,}100\ \#/in^2$ (NOM)
ALLOWABLE $f_s = 12{,}200\ \#/in^2$ (NOM.)
M.S. +.72

MAX CAP END LOAD (INC: T.F) = $565\#$ $A = .114\ in^2$
TF. BENDING STRESS (FROM SKIN) = $8{,}300\ \#/in^2$ $f_{TOTAL} = 13{,}200\ \#/in^2$
FLANGE CRIPPLING STRESS $f_{cc} = 22{,}000$ M.S. +.66

ATTACHMENTS RIVETS ALONG SPAR :— STRENGTH = $432\ \#/in$
(SKIN TO SPAR) LOADING = 330 " M.S. +.31

CENTER SECTION OF TAILPLANE — DWG Nº 385-3006

FRONT SPAR :— $q_{MAX} = 4119\ \#/in$ — CASE B(i)

WEB = .125" (2024-T3) $f_s = 32{,}952\ \#/in^2$ $f_{SU} = 40{,}000$ M.S. +.21
1/4 HI-SHEAR PINS IN BRG @ .92" PITCH $q_{ALLOW} = 4201\ \#/in$ M.S. +.02

ATTACH. CLIPS TO FITTING STRENGTH = $19{,}190\#$ (IN BRG) M.S. +.01
LOAD = $19{,}000\#$

REAR SPAR - $q_{MAX} = 1915\ \#/in$ — CASE B(ii)

WEB = .063 (2024-T3) $f_s = 30{,}400\ \#/in^2$ $f_{SU} = 40{,}000$ M.S. +.31

3/16 HI.SHEAR PINS IN BRG. STRENGTH = $2072\ \#/in$ M.S. +.08

CAP LOADS :— MAX CAP LOADS ARE COVERED BY SYMM. CASE ON OUTBD PORTION

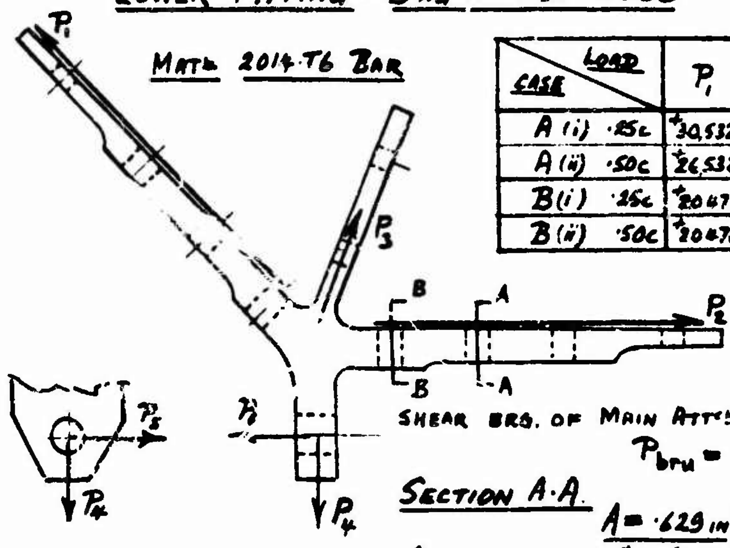
'KINK' LOADS :—
(i) F/SPAR FITTING f_{brg} OF 5/16 BOLTS IN .071 ANGLE M.S. 0.00
$P = 24{,}484\#$.

ii) R/SPAR FITTING OK BY COMPARISON WITH F/S FITTING
$P = 11{,}520\#$

TAILPLANE ATTCH FITTINGS — FWD.

LOWER FITTING - DWG № 385-3003

MATL 2014-T6 BAR



(ULTIMATE LOADS)

CASE \ LOAD	P_1	P_2	P_3	P_4	P_5	P_6
A (i) ·25c	+30,532	+28674	+20795	+2072	−16	0
A (ii) ·50c	+26,532	+26598	+29,500	−153	−16	0
B (i) ·25c	+20476	+18468	−2,929	+16828	−628	+4085
B (ii) ·50c	+20476	+20388	−171	+14646	−3769	+4085

SHEAR BRG. OF MAIN ATTCH LUG:– $P_4 = 16,828$ #

$P_{bru} = 19,000$ # M.S. + ·12

SECTION A·A. $A = ·629$ in² $P = 15,300$ # $M = 1715$ #-in

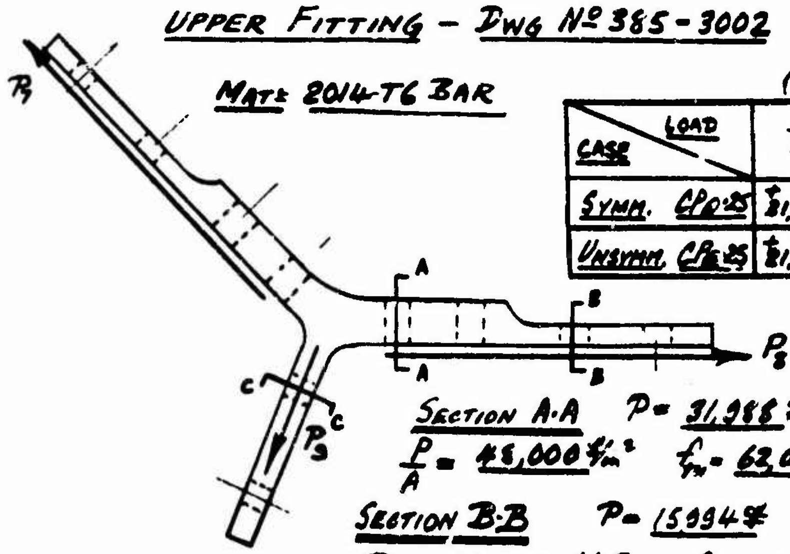
(*INCL PLASTIC BENDING FACTOR) $f_{TOTAL} = 68,500$ #/in²* $f_{ALLOW} = 77,500$ #/in² M.S. + ·13

SECTION B·B $P = 20,388$ # $M = 2,500$ #-ins $A = ·98$ in² $Z = ·0867$ in³

$f_{TOTAL} = 49,600$ #/in² $f_{ALLOW} = 77,500$ #/in² M.S + ·56

UPPER FITTING - DWG № 385-3002

MATL 2014-T6 BAR



(ULTIMATE LOADS)

CASE \ LOAD	P_7	P_8	P_9
SYMM. CP@·25	+31,988	+31,988	+24,484
UNSYMM. CP@·25	+31,513	+22,400	+16,840

SECTION A·A $P = 31,988$ # $A = ·667$ in²

$\frac{P}{A} = 48,000$ #/in² $f_{tu} = 62,000$ #/in² M.S. + ·19

SECTION B·B $P = 15,994$ # $A = ·357$ in²

$P/A = 44,800$ #/in² $f_{tu} = 62,000$ M.S. + ·38

SECTION C·C $P_9 = 24,484$ # $A = ·52$ in² $P/A = 47,200$ #/in²

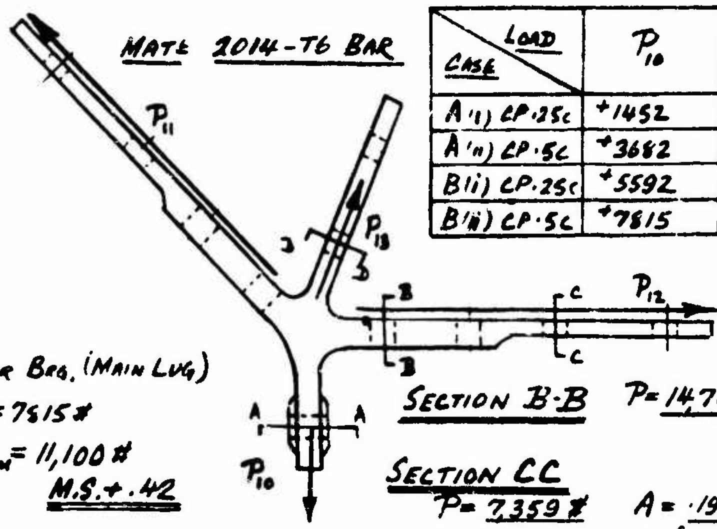
· @ $f_{tu} = 62,000$ #/in² M.S. + ·42

TAILPLANE ATT^CH FITTINGS – AFT.

LOWER FITTING - DWG Nº 385-3005

MAT: 2014-T6 BAR

LOAD / CASE	P_{10}	P_{11}	P_{12}	P_{13}
A (i) CP·25c	+1452	+9682	+9081	+5609
A (ii) CP·5c	+3682	+14718	+13194	+6697
B (i) CP·25c	+5592	+8490	+6174	-441
B (ii) CP·5c	+7815	+8480	+5253	-3200



SECTION A·A. $P_T = 7815^{\#}$
$f_{tu} = 18{,}200\#$
M.S.+1.33

SHEAR BRG. (MAIN LUG)
$P = 7815\#$
$P_{bru} = 11{,}100\#$
M.S.+.42

SECTION B·B $P = 14718\#$ $A = .504\,in^2$ $P/A = 29{,}150\,\#/in^2$
$f_{tu} = 62{,}000\,\#/in^2$ M.S. 1.12

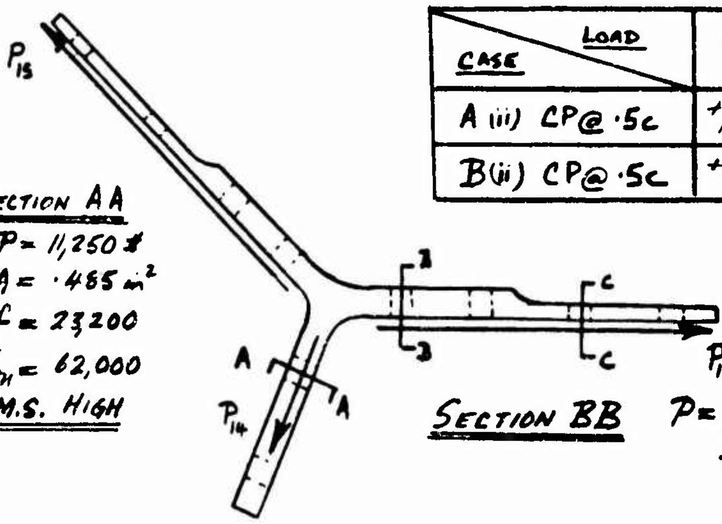
SECTION CC
$P = 7{,}359\#$ $A = .195\,in^2$ $P/A = 37{,}700\,\#/in^2$
$f_{tu} = 62{,}000\,\#/in^2$ M.S +.64

SECTION D·D $P = 6697\#$ $A = .485$ ∴ M.S. HIGH

UPPER FITTING - DWG Nº 385-3004

MAT: 2014-T6 BAR

LOAD / CASE	P_{14}	P_{15}	P_{16}
A (ii) CP@·5c	+11,520	+15,052	+15052
B (ii) CP@·5c	+7293	+8920	+10,136



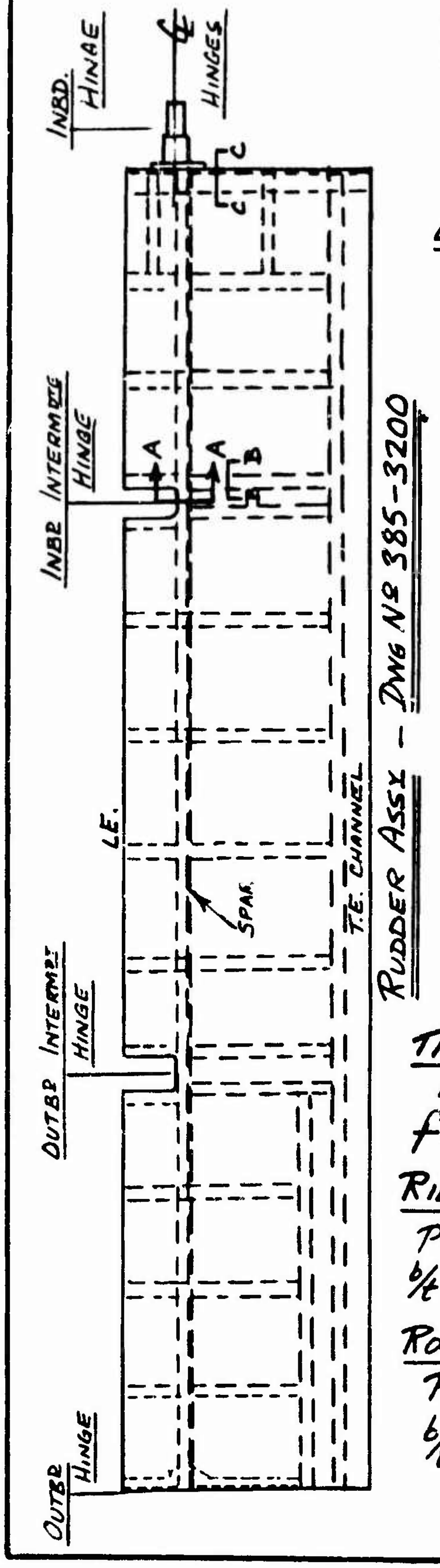
SECTION AA
$P = 11{,}250\#$
$A = .485\,in^2$
$f = 23{,}200$
$f_{tu} = 62{,}000$
M.S. HIGH

SECTION BB $P = 15{,}052\#$ $A = .543\,in^2$ $f = 27700\,\#/in^2$
$f_{tu} = 62{,}000\,\#/in^2$
M.S.+1.24

SECTION C-C $P = 7526\#$ $A = .214\,in^2$ $f = 35{,}200\,\#/in^2$
$f_{tu} = 62{,}000$
M.S+.76

8 BOLTS ($\frac{1}{4}''$ DIA) 1880 LBS/BOLT $f_{bry} = 58{,}000\,\#/in^2$
$f_{bru} = 98{,}000$
M.S.+.69

RUDDER



REFERENCE APPENDIX I, RUDDER LOADS

RUDDER SPAR — .040" 2024-T3

CRITICAL SECTION AA   DEPTH BETWEEN CENTROIDS = 2.75"

$M = 4815$ #ins (LIMIT) 7222 #ins (ULT)

CAP E.L. = 2626 lbs (ULT) $A = .2548 \text{ in}^2$

$f_c = 10{,}300$ #/in² $f_{ALLOW} = 11{,}020$ #/in²
(LOCAL INSTABILITY)

M.S. +.07

MAX. SHEAR $Q_v = 350 \times 1.5$
$= 525$ # ULT.

$q = 191$ #/in $f_s = 4{,}775$ #/in²   M.S. HIGH

SKINS — CRITICAL @ INBD END

(a) NOSE $q_{MAX} = 60$ #/in $f_s = 3750$ #/in²

$\tau_B = 7{,}100$ #/in²   PANEL DOES NOT BUCKLE
M.S. HIGH

(b) AFT $q_{MAX} = 48$ #/in $f_s = 3{,}000$ #/in²

$\tau_B = 475$ #/in² $\tau/\tau_B = 6.32$ $K = .38$

$f_{SH} = 16{,}500$ #/in²

M.S. HIGH

TF. PULL ON CHANNEL (T.E.)

$M = 54$ #ins $P = 86$ $A = .047 \text{ in}^2$ $Z = .0034 \text{ in}^3$

$f = 17{,}700$ #/in² $f_{cc} = 16{,}500$ ($b/t = 20$)   M.S. +.04

RIB CAP — SECTION B-B

$P = 508$ # (ULT) $A = .036 \text{ in}^2$ $f = 14{,}100$ #/in²

$b/t = 26$ FLANGE CRIPPLING $f_{cc} = 14{,}500$ #/in²   M.S +.03

ROOT RIB — SECTION C-C

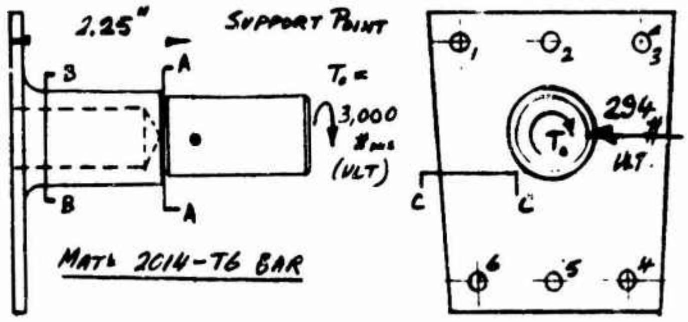
$P = 632$ # (ULT) $A = .060 \text{ in}^2$ $f = 10{,}600$ #/in²

$b/t = 28$ FLANGE CRIPPLING $f_{cc} = 13{,}500$ #/in²

M.S. +.27

RUDDER – (CONT'D)

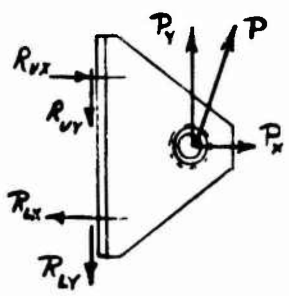
FITTING – TORQUE TUBE – DWG Nº 385-3202



MAX BOLT LOAD:-
BOLT 4 OR 6 P = 383#
f_{brg} = 51,200 #/in² in .040"
f_{bru} = 100,000 M.S + .91

SECTION AA T_0 = 3,000 #ins. Z_T = .115 f_s = 26,086 #/in² @ f_{su} = 38,000 M.S. + .46
SECTION BB T_0 = 3,000 " Z_T = .166 – COVERED BY AA
*SECTION CC M_{cc} = 133 #ins Z = .0026 in³ f = 51,153 #/in² M.S. + .56
(* BENDING DUE TO 294# RUDDER SUPPORT LOAD) f_{ULT} = 80,000 (PLASTIC BENDING)

FITTING – RUDDER HINGE



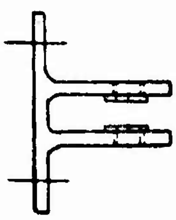
DWG. Nº 385-3201
MAT: 2014-T6 BAR

P = 1036 lbs (ULT)
P_Y = 974# P_X = 268#

R_{UX} = 265#/BOLT R_{UY} = 244#/BOLT
R_{LX} = 456#/BOLT R_{LY} = 244#/BOLT

BASE IN BENDING :— M.S. HIGH

LUG TRANSVERSE ANALYSIS:— M.S. HIGH



ATT'N TO CLIPS ON RUDDER:-
STRENGTH OF .078 ANGLE, .45 ECCENT.
= 460#/BOLT

LOAD (R_{LX}) = 456 LBS/BOLT M.S. + .01

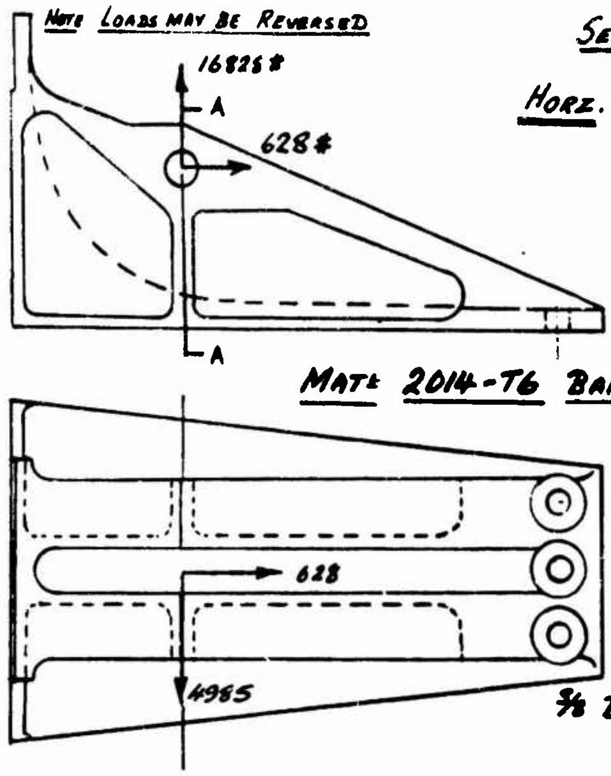
RIVETS ATT'N CLIP TO RIB :- LOAD = 388#
STRENGTH 1/8" RIVET IN .025 = 388# M.S 0.00

MASS BALANCE INSTALLATION — DWG. Nº 385-3203

STRENGTH OF THIS PART NOT CRITICAL M.S. HIGH

TAILPLANE ATTCH. FITTINGS (ULTIMATE LOADING)

FWD. FITTING DWG. Nº 385- 2305



MAT! 2014-T6 BAR

SECTION AA. (CASE B(i))

HORZ. $M_H = 2756$ #ins. PER BEAM
$Z = \cdot 043$ in³

$M/Z = \underline{64,100\ \#/in^2}$

VERT $M_V = 23,621$ # INS.

$Z = \cdot 8750\ in^3$

$M/Z = \underline{27,000\ \#/in^2}$

TOTAL $f_b = \underline{91,100\ \#/in^2}$

$f_{ALLOWABLE}$ (PLASTIC BENDING)
$= \underline{91,300\ \#/in^2}$

M.S + 0.00

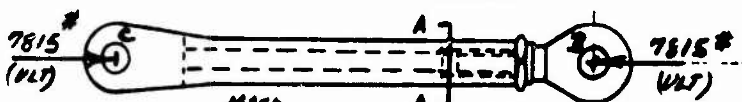
3/8 BOLT IN DOUBLE SHEAR = 17,000
@ 21,000 #

MS +.23

ANGLE OF INCIDENCE ADJUSTMENT ROD

DWG. Nº 385-2003 CASE B(ii)

LOADS MAY BE REVERSED



MAT! 4130 STEEL COND F4

ROD END AT B:- STRENGTH = 8300 # **M.S. +.06**

BENDING OF BOLT AT B. $\frac{M}{Z} = 227,400\ \#/in^2$ **M.S. +.10**

160,000 #/in² H.T. BOLT K=1.7 $f_{ALLOWABLE}$ (PLASTIC BENDING) 252,000 #/in²

SECTION AA $P_c = 7815$ NET $A = \cdot 086\ in^2$ $\rho = \cdot 156$ $L = 5.68"$

GENERAL INSTABILITY :- $\frac{L}{\rho} = 36$ $f_{cr} = \underline{95,000\ \#/in^2}$

$\frac{P}{A} = \frac{7815}{\cdot 086} = \underline{90,900\ \#/in^2}$ **M.S. +.04**

STRESS ANALYSIS - CONTROL SYSTEM

The XV-9A has incorporated in its design a number of the OH-6A control system components in the cockpit area and all of the Model 285 upper rotor control system parts. No analysis of these parts is included in this report.

A summary of the minimum margins of safety for the OH-6A control system parts used is included here. These margins of safety are taken from Reference 33.

The Model 285 upper rotor control system has been analyzed in Reference 30.

The portion of the flight control system between the cockpit and the stationary swashplate is designed to the design criteria presented in Section 7.

STATIONARY SWASHPLATE

DWG Nº 285-0313

LOADING CASES

a) ULTIMATE:-

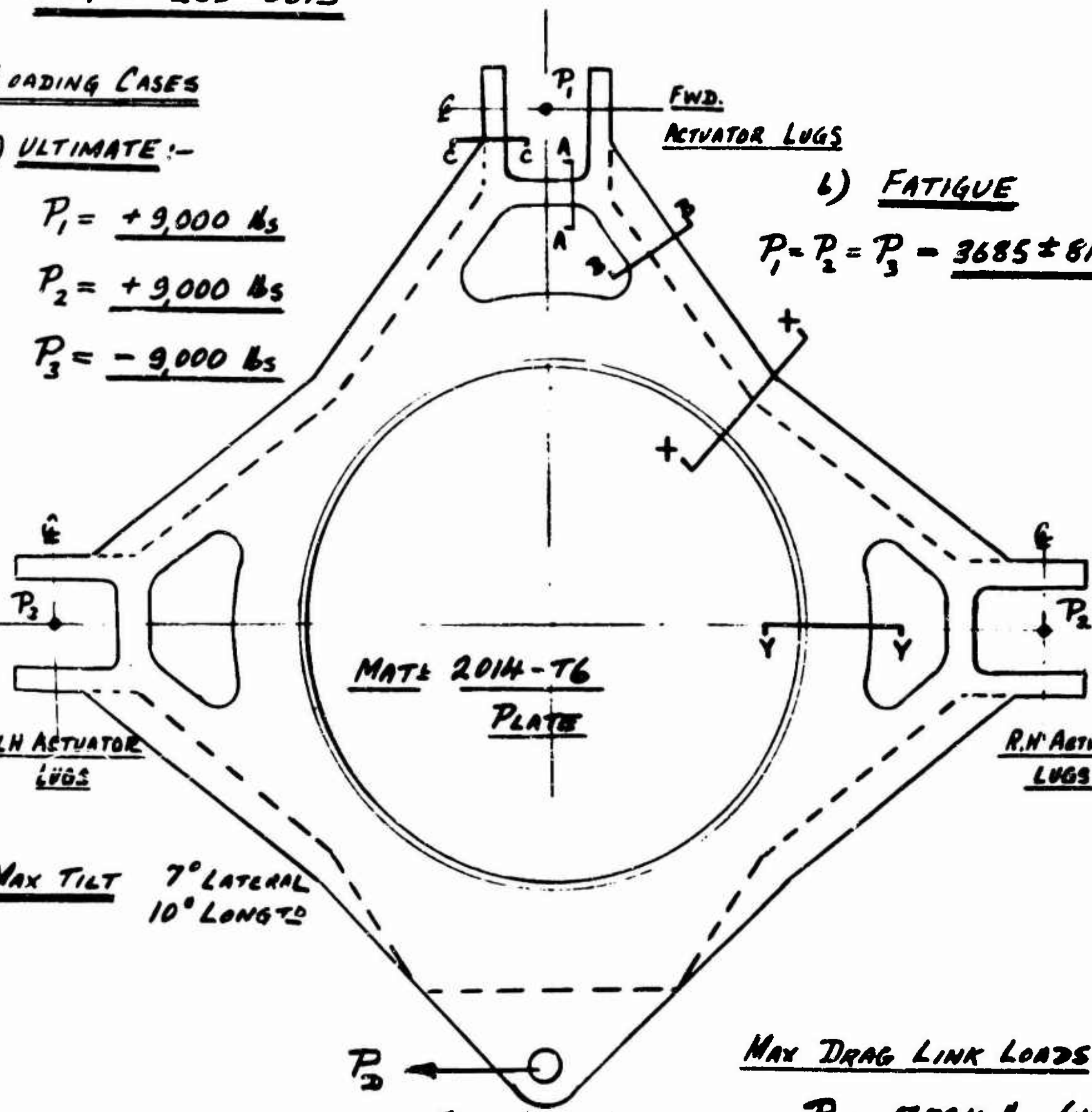
$P_1 = +9{,}000$ lbs

$P_2 = +9{,}000$ lbs

$P_3 = -9{,}000$ lbs

b) FATIGUE

$P_1 = P_2 = P_3 = 3685 \pm 815$ #



MAX TILT 7° LATERAL
10° LONG^TD

MAX DRAG LINK LOADS:-

$P_D = 7724$ lb. (ULT)

SECTION A-A	$f_{AA} = 7{,}350$ #/in² (ULT.) $= 3{,}045 \pm 674$ #/in² (FATIGUE)	M.S. HIGH
SECTION BB	$f_{BB} = 22{,}645$ #/in² (ULT.) @ 54,000 #/in² $= 9{,}300 \pm 2051$ #/in² (FATIGUE)	M.S. + 1.38
SECTION CC	$f_{CC} = 20{,}000$ #/in² (ULT) @ 54,000 #/in² $= 8{,}200 \pm 1805$ #/in² (FATIGUE)	M.S. + 1.70

STATIONARY SWASHPLATE – CONT^D

TRANSVERSE STRENGTH OF LUG $P_{TRU} = 7{,}000$ lbs M.S. +.23

$P_{TR} = 5{,}700$ # (ULT. LOAD)

SECTION XX

ULTIMATE LOADING $\begin{cases} f_{AXIAL} = 36{,}800\ \#/in^2 @ 54{,}000\ \#/in^2 & R_A = .681 \\ f_{SHEAR} = 10{,}943\ \#/in^2 @ 34{,}000\ \#/in^2 & R_S = .322 \end{cases}$

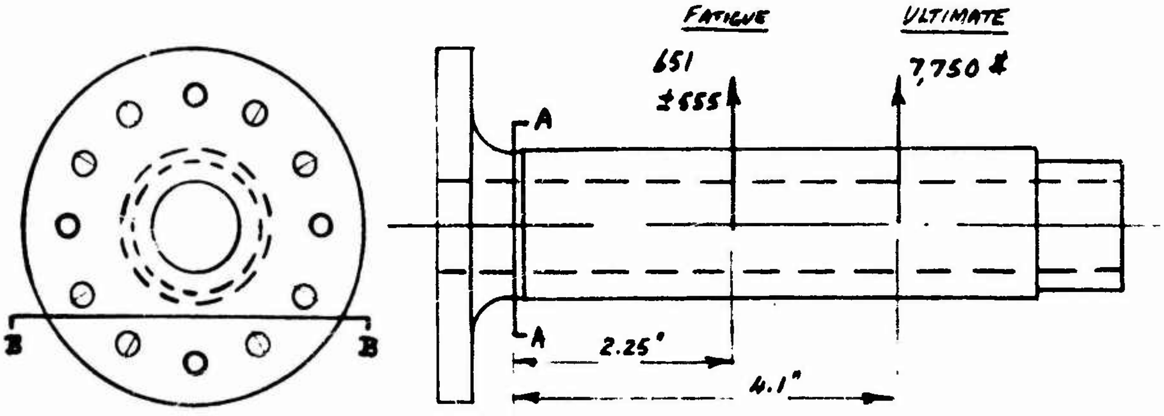
INTERACTION $R_A^2 + R_S^2 = 1$ M.S. + .33

SECTION YY

ULTIMATE LOADING $\begin{cases} f_{AXIAL} = 9{,}000\ \#/in^2 @ 54{,}000 & R_1 = .167 \\ f_{SHEAR} = 25{,}400\ \#/in^2 @ 34{,}000 & R_2 = .747 \\ f_{BENDING} = 18{,}700\ \#/in^2 @ 54{,}000 & R_3 = .346 \end{cases}$

INERACTION $M.S. = \dfrac{1}{R_1 + \sqrt{R_2^2 + R_3^2}} - 1$ M.S. + .01

SPINDLE – DWG. Nº 385-1008



SECTION A·A (a) FATIGUE $\dfrac{M}{Z} = 3250\ \#/in^2 \pm 2{,}770\ \#/in^2$

$f_{ALLOW.} = \pm 10{,}000$ M.S. HIGH

(b) ULTIMATE $\dfrac{M}{Z} = 60{,}800\ \#/in^2$ @ $140{,}000\ \#/in^2$ M.S. + 1.30

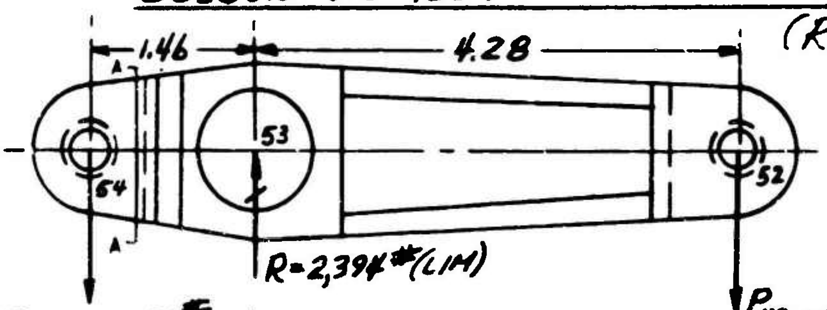
SECTION BB ULT. $\dfrac{M}{Z} = 127{,}890\ \#/in^2$ @ $140{,}000\ \#/in^2$ M.S. + .09

IN THIS ANALYSIS ONLY THE PARTS AND SECTIONS WITH A MARGIN OF SAFETY LESS THAN +.50 ARE SHOWN. ALL OTHER PARTS AND SECTIONS HAVE A MARGIN OF SAFETY GREATER THAN +.50.

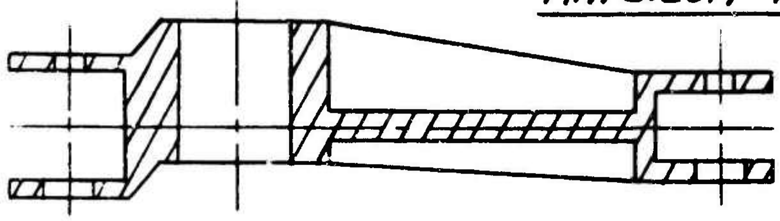
ALL COCKPIT CONTROL PARTS NOT SHOWN HERE ARE THE SAME AS PARTS USED ON THE OH-6A HELICOPTER AND WERE DESIGNED IN ACCORD WITH F.A.A. REGULATIONS.

BELLCRANK ASS'Y (DWG NO. 385-6132)

(REF PG 226)



MAT'L: 2014-T6 ALUM ALLOY



SECTION A-A

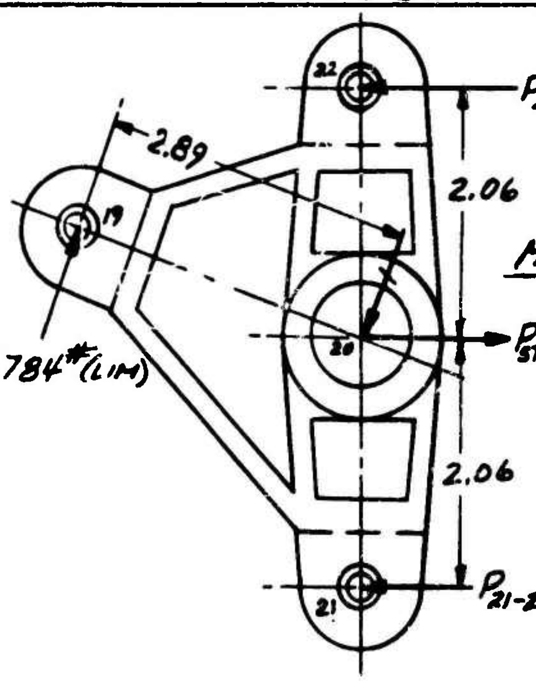
$M = 1660$ #INS ULT.

$f_b = 31{,}663$ #/in²

$f_{tu} = 62{,}000$ "

M.S. +.96

LATERAL FOLLOWER BELLCRANK (DWG NO. 385-6125)



MAT'L: 2014-T6 ALUM ALLOY (QQ-A-266)

M.S. = + HIGH

MIXER ASS'Y & INSTALLATION (DWG. NO. 385-6120)

LOADS (LIM) CASE II a

(REF PP 225 & 226)

$P_a = P_{14\text{-}16} = 1{,}258^{\#}$

$P_b = R_{a1} = R_{b1} = P_{17\text{-}19} = 784^{\#}$

$R_{a2} = P_a - R_{a1} = 474^{\#}$

$R_{b2} = R_{b3} = P_{22LA} = 550^{\#}$

$P_c = R_{b1}(1.23)/4.33 = 223^{\#}$

$R_{c1} = P_c = 223^{\#}$

$P_d = R_{d1} = R_{b1} - P_c = 561^{\#}$

$P_e = R_{e1} = P_{54\text{-}55} = 1{,}785^{\#}$

$P_g = P_{57\text{-}59} = 1{,}428^{\#}$

$R_{e2} = R_{e3} = P_g(1.23)/4.33 = 406^{\#}$

$R_{f1} = R_{f2} = P_g - R_{e2} = 1{,}022^{\#}$

$M_f = M_{e4} = 3.50 R_{f1} = 3{,}577\ \text{IN}^{\#}$

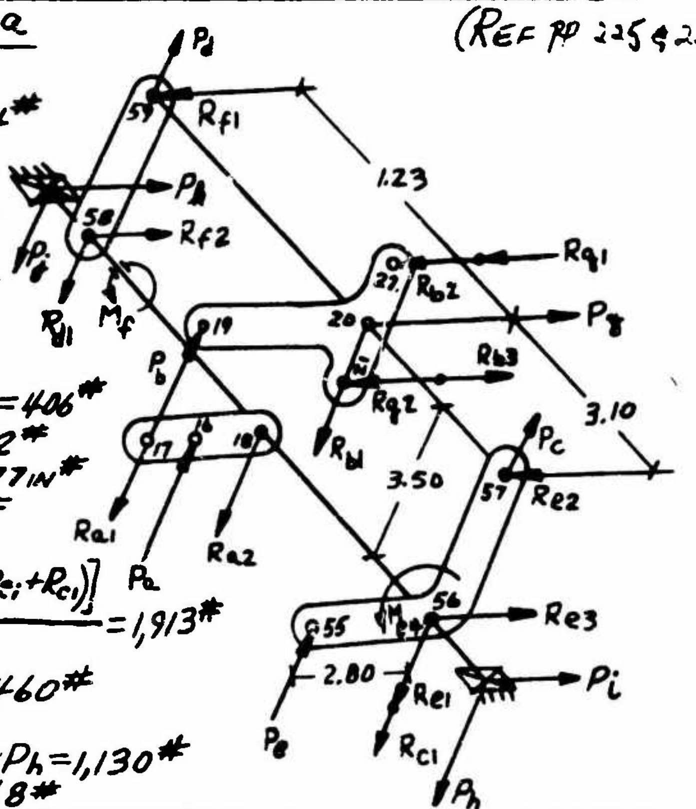
$R_{g1} = R_{g2} = P_g/2 = 714^{\#}$

$$P_h = \frac{[.68R_{d1} + 1.91R_{a2} + 5.01(R_{e1} + R_{c1})]}{5.93} = 1{,}913^{\#}$$

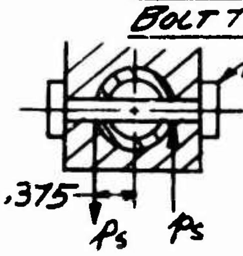
$$P_i = \frac{.68R_{f2} + 5.01R_{e3}}{5.93} = 460^{\#}$$

$P_j = R_{d1} + R_{a2} + R_{e1} + R_{c1} - P_h = 1{,}130^{\#}$

$P_k = R_{f2} + R_{e3} - P_i = 968^{\#}$



BOLT THRU 385-6121 COLL. DRIVER & 385-6128-1 TUBE

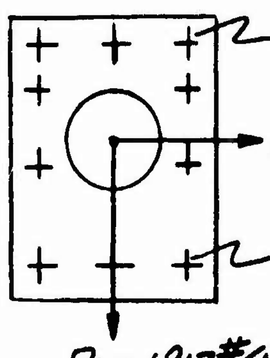


NAS464P5A19 BOLT ($P_S = 7{,}300^{\#}$)

$$P_S = \frac{M_{e4}(1.5)}{2(.375)} = 7{,}154^{\#}(\text{ULT})$$

$$M.S. = \frac{7300}{7154} = +.02$$

RIVETS THRU 385-6129 FLANGE SUPPORT



MS20470AD4, 7 RIVETS ($P_S = 386^{\#}$)

$P_i = 460^{\#}(\text{LIM})$

MS20600AD4, 3 RIVETS ($P_S = 321^{\#}$)

$P_h = 1{,}913^{\#}(\text{LIM})$

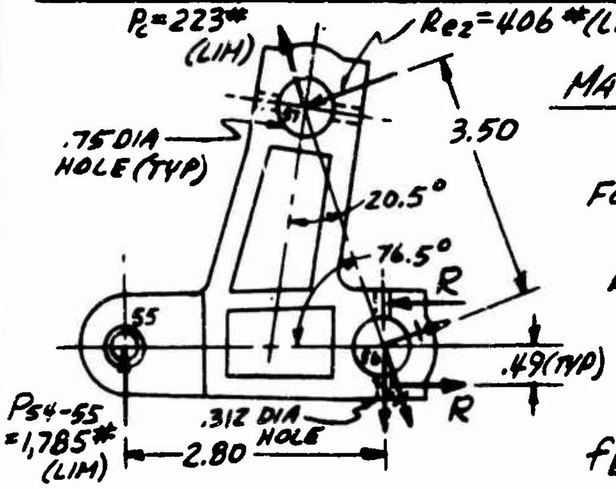
$$M.S. = \frac{386}{296} - 1 = +.30$$

$$M.S. = \frac{321}{296} - 1 = +.08$$

$$P_S/\text{RIVET} = \left[\left(\frac{460}{10}\right)^2 + \left(\frac{1913}{10}\right)^2\right]^{1/2}(1.5) = 296^{\#}(\text{ULT})$$

COLLECTIVE DRIVER BELLCRANK (DWG. NO. 385-6121)

(REF PP225 & 226)



MAT'L: 2014-T6 ALUM ALLOY (QQ-A-266)

FOR $e/D = 1.5$ $F_{br} = 98{,}000$ PSI

$$R = \frac{M_{e4}(1.5)}{2(.49)} = 5{,}475\#(ULT)$$

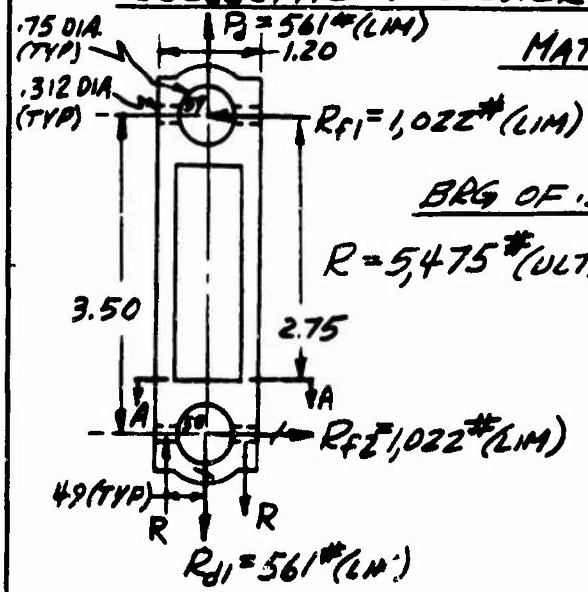
$(M_{e4} = 3{,}577 \text{ IN}\#\text{LIM})$

BRG OF .312 DIA BOLT @ 56

$$f_{br} = \frac{R}{A} = \frac{5475}{.312(.225)} = 77{,}991 \text{ PSI}$$

$$M.S. = \frac{98000}{77991} - 1 = \underline{\underline{+.25}}$$

COLLECTIVE FOLLOWER (DWG NO. 385-6127)

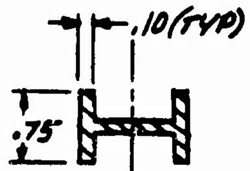


MAT'L: 2014-T6 ALUM ALLOY (QQ-A-266)

$F_{tu} = 65{,}000$ PSI

BRG OF .312 DIA. BOLT @ 58

$R = 5{,}475\#(ULT)$, $f_{br} = 77{,}991$ PSI
$M.S. = \underline{\underline{+.25}}$ } SEE ABOVE



SECTION A-A

$$I = \frac{.75(1.20)^3 - .65(1.00)^3}{12} = .053 \text{ IN}^4$$

$C = .60$ IN

$M = 1022(2.75)(1.5) = 4{,}216$ IN#(ULT)

$$f_b = \frac{Mc}{I} = \frac{4216(.60)}{.053} = 47{,}728 \text{ PSI}$$

$$f_t = \frac{P_d}{A} = \frac{561(1.5)}{.10(2.50)} = 3{,}366 \text{ PSI (ULT)}$$

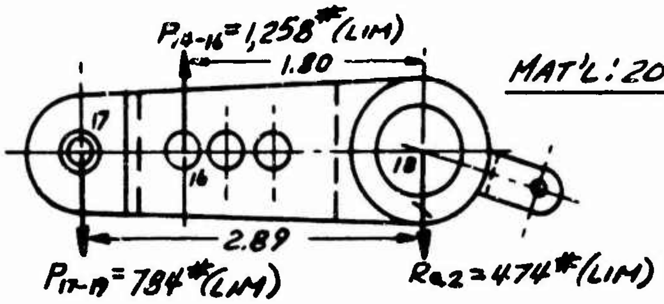
TOT $f_t = 47728 + 3366 = 51{,}094$ PSI

$$M.S. = \frac{65{,}000}{51{,}094} - 1 = \underline{\underline{+.27}}$$

ROTOR CONTROLS

LATERAL DRIVER LEVER (DWG. NO. 385-6122)

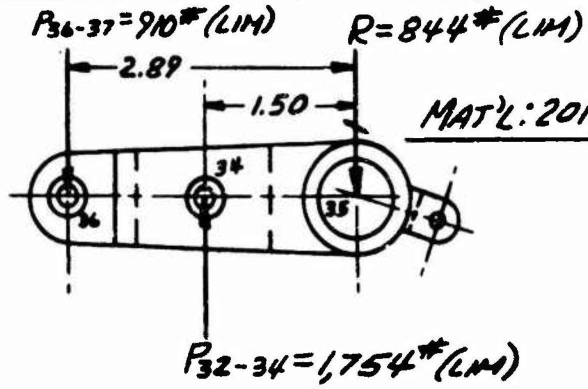
(REF PP 225 & 226)



MAT'L: 2014-T6 ALUM ALLOY (QQ-A-266)

M.S. = + HIGH

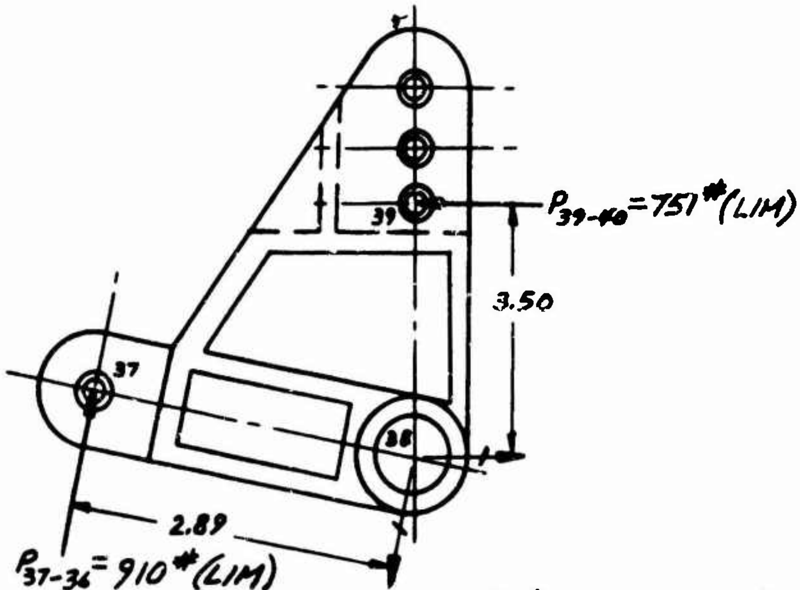
LONGITUDINAL DRIVER LEVER (DWG NO. 385-6123)



MAT'L: 2014-T6 ALUM. ALLOY (QQ-A-266)

M.S. = + HIGH

LONGITUDINAL FOLLOWER BELLCRANK (DWG NO. 385-6126)



MAT'L: 2014-T6 ALUM ALLOY (QQ-A-266)

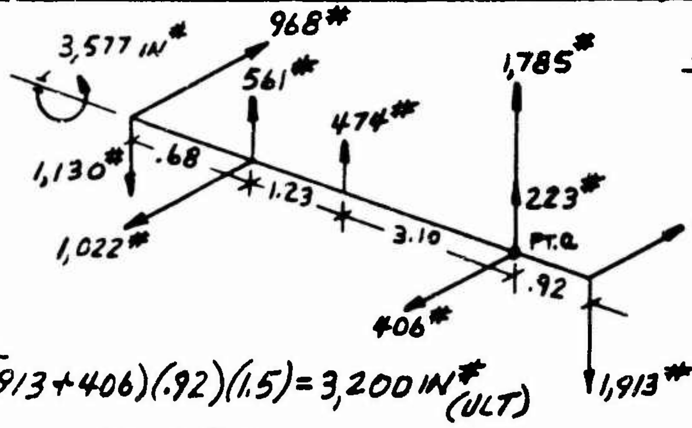
M.S. = + HIGH

INTERCONNECT TUBE (DWG NO. 385-6128)

-1 TUBE

MAT'L: 4130 STL TUBE (MIL-T-6730, COND. N), H.T. 160-180 KSI

.75 O.D X .125 W.T.



② PT. a

$M_{MAX} = (1913+406)(.92)(1.5) = 3{,}200\ IN\#_{(ULT)}$

$I = .012\ IN^4 \quad C = .375\ IN \qquad f_b = \frac{MC}{I} = \frac{3200(.375)}{.012} = 100{,}000\ PSI$

$MAX\ f_s = \frac{2T r_1}{\pi(r_1^4 - r_0^4)} = \frac{2(3577)(1.5)(.375)}{\pi[(.375)^4 - (.25)^4]} = 80{,}764\ PSI\ (ULT)$

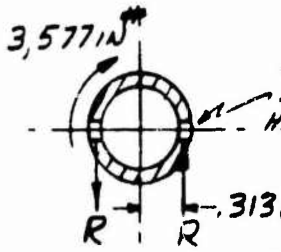
$D/t = \frac{.75}{.125} = 6\ ;\ \ell/D = \frac{5.93}{.75} = 7.91$

FOR $F_{tu} = 160{,}000\ PSI$; $F_{st} = 110{,}000\ PSI$, $F_b = 220{,}000\ PSI$

$\left(\frac{f_b}{F_b}\right)^2 + \left(\frac{f_s}{F_{st}}\right)^2 = \left(\frac{100}{220}\right)^2 + \left(\frac{80{,}764}{110{,}000}\right)^2 = .75 < 1.00$

$M.S. = \frac{1}{[R_b^2 + R_s^2]^{1/2}} - 1 = \frac{1}{\sqrt{.75}} - 1 = \underline{\underline{+.15}}$

TUBE IN BEARING



$R = \frac{3577(1.5)}{2(.313)} = 8{,}571\#(ULT)$

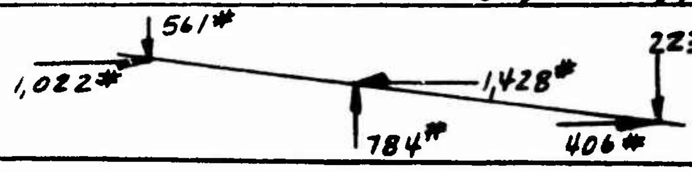
$f_{br} = \frac{R}{A} = \frac{8571}{.125(.312)} = 219{,}769\#(ULT)$

FOR ($e/D = 2.00$) $F_{br} = 287{,}000\ PSI$

$M.S. = \frac{287{,}000}{219{,}769} - 1 = \underline{\underline{+.31}}$

-3 TUBE

MAT'L: 4130 ST'L TUBE (MIL-T-6730, COND. N), .75 O.D. X .125 W.T.



M.S. = + HIGH

ROD ASS'Y (DWG NO 385-6105-1 & -3)

(REF PP 225 & 226)

P ⟶ ⟵ P

ℓ

FOR -1; $P_{39-40} = 751^{\#}$ (LIM), $\ell = 71.04$ IN
FOR -3; $P_{22-23} = P_{21-27} = 1,264^{\#}$ (LIM), $\ell = 69.06$ IN

MAT'L: 1.75 O.D. X .058 W.T., 2024-T3 ALUM ALLOY (WW-T-785)

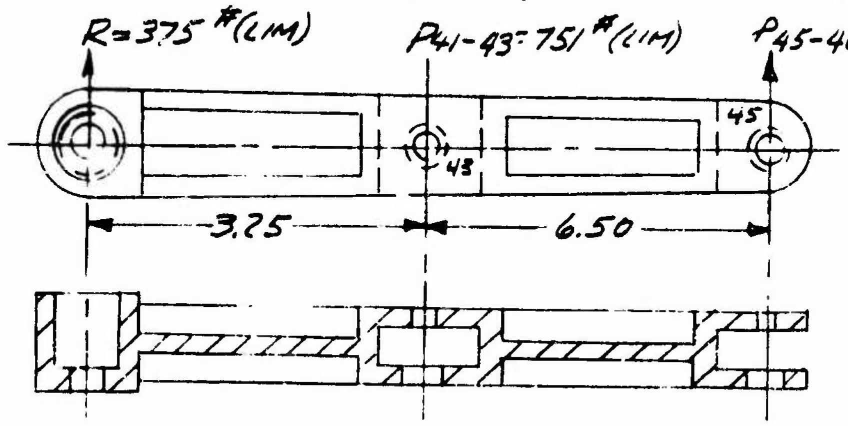
$$f_c = \frac{P_{22-23}}{A} = \frac{1264(1.5)}{.371} = 5111 \text{ PSI}$$

$A = .371 \text{ IN}^2$; $\rho = .545$ IN

$$\text{FOR } \frac{L'}{\rho} = 126.7;\quad F_c = \frac{\pi^2 E}{(L'/\rho)^2} = 6454 \text{ PSI}$$

$$M.S. = \frac{6454}{5111} - 1 = +.26$$

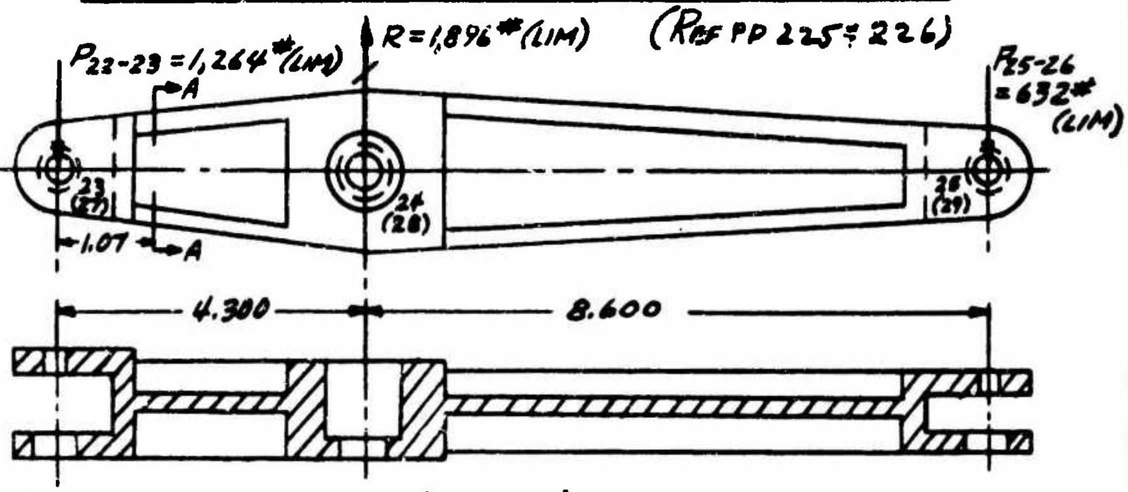
LONGITUDINAL LEVER ASSY (DWG NO. 385-6116)



MAT'L: 2014-T6 ALUM ALLOY (QQ-A-266)

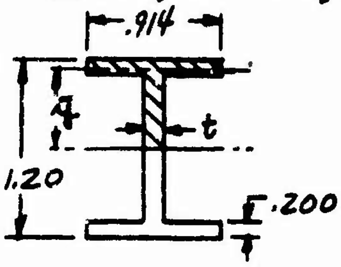
M.S. = +HIGH

BELLCRANK ASS'Y (DWG No. 385-6115)



MAT'L: 2014-T6 ALUM. ALLOY (QQ-A-266)

F_{tu} = 62,000 PSI, F_{su} = 38,000 PSI



SECTION A-A

SHEAR AT SECTION A-A

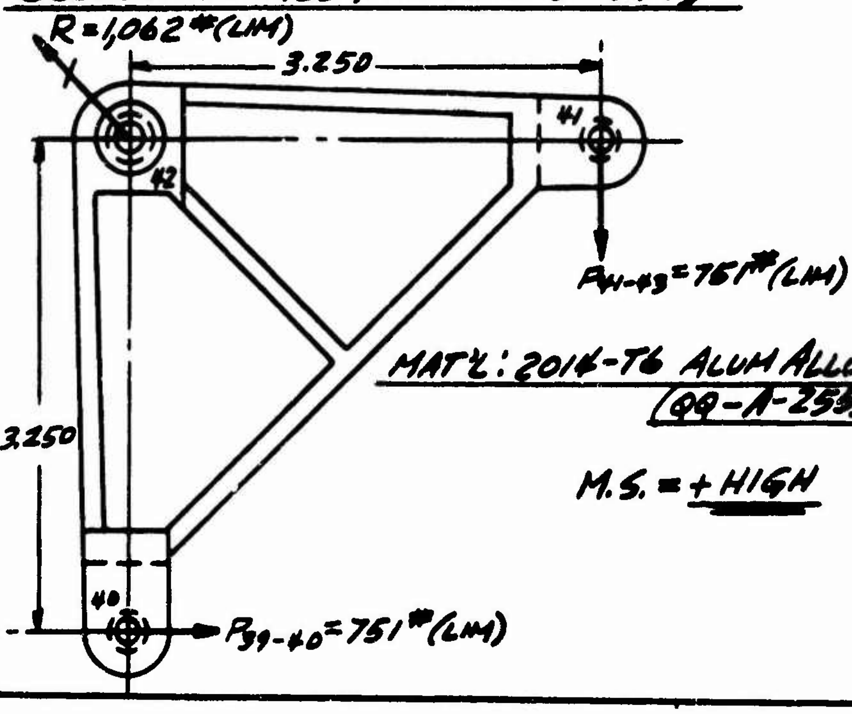
$\bar{y} = \frac{.097}{A}$; $t = .07$ MIN ; $I = .095\ \text{IN}^4$

$$\text{MAX.}\ f_s = \frac{VA\bar{y}}{It} = \frac{1264(1.5)A\left(\frac{.097}{A}\right)}{.095(.07)}$$

$$= 27{,}643\ \text{PSI (ULT)}$$

$$M.S. = \frac{38000}{27643} - 1 = \underline{+.37}$$

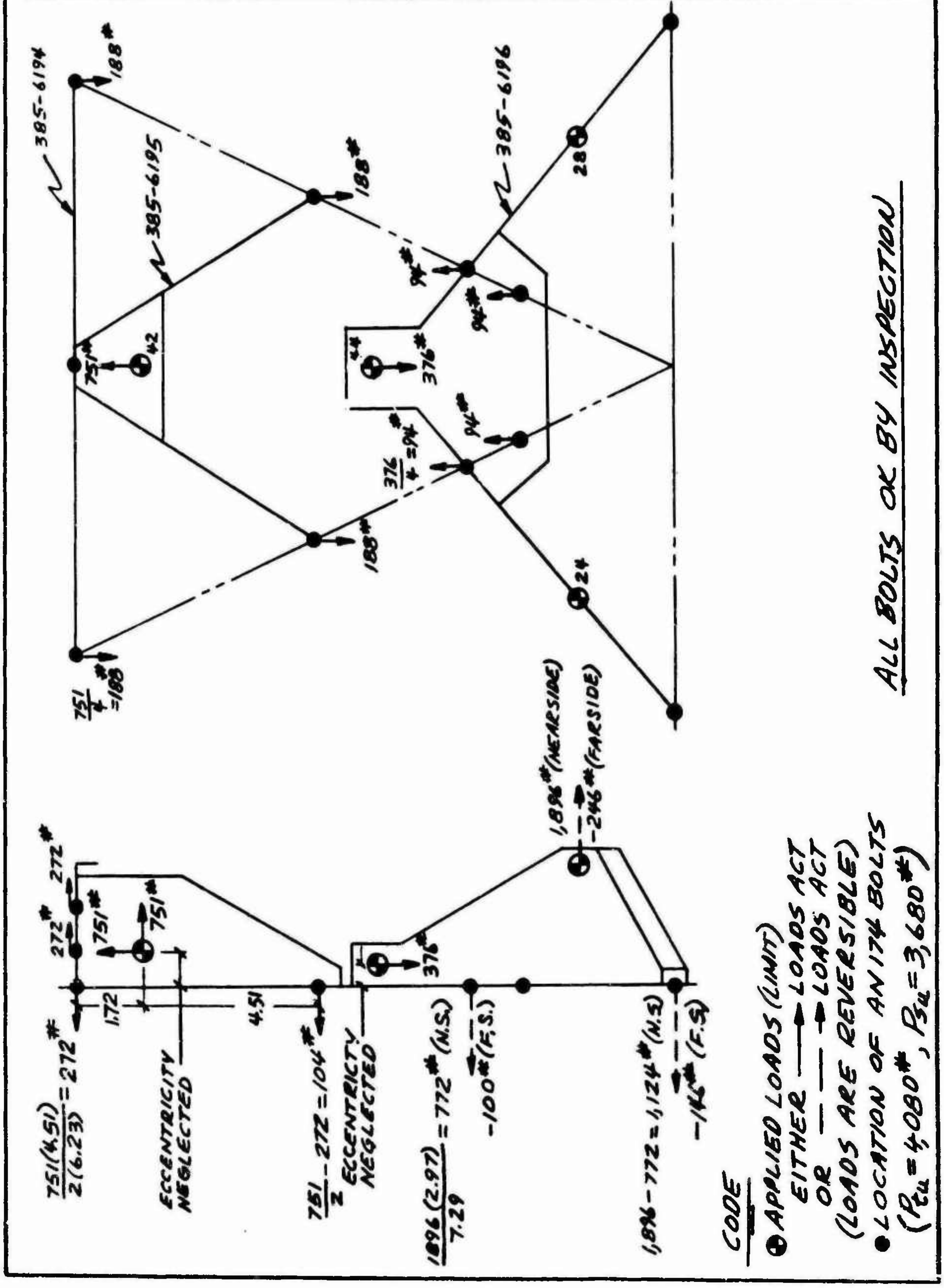
BELLCRANK ASS'Y (DWG 385-6117)



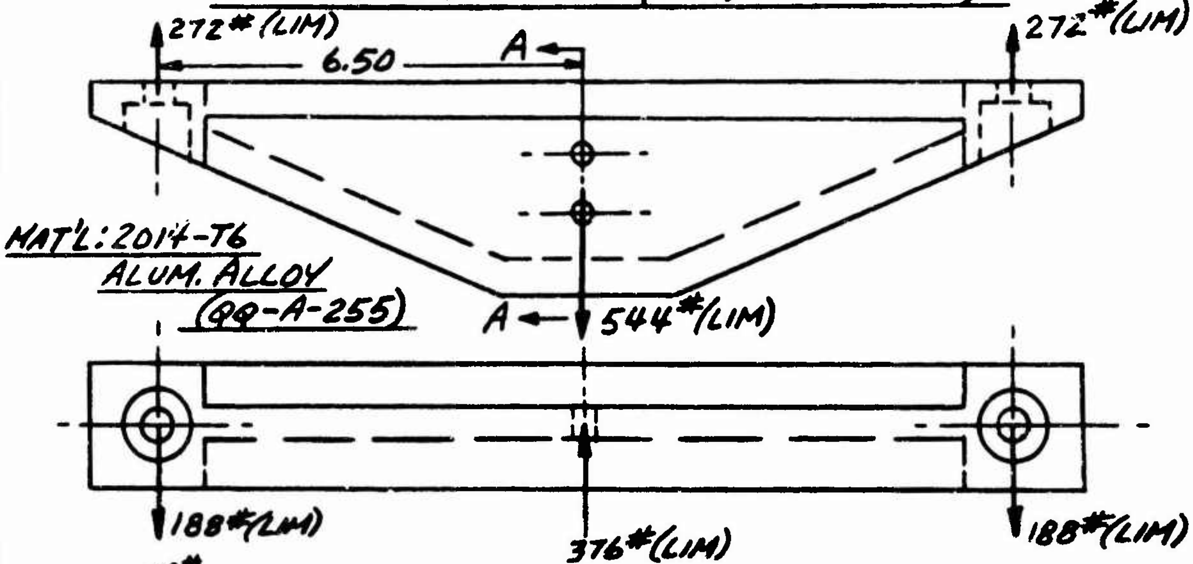
MAT'L: 2014-T6 ALUM ALLOY (QQ-A-255)

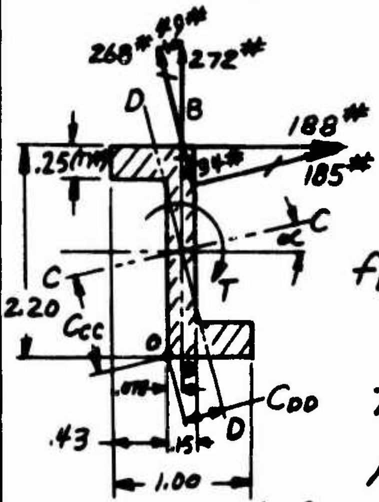
M.S. = $\underline{+\text{HIGH}}$

LEVER SUPPORT BRACKET INSTALLATION (DWG No. 385-6193)



MODULE & BRACKET (DWG 385-6194)





SECTION A-A

$I_{CC} = .343\ IN^4$; $I_{DD} = .015\ IN^4$

$\alpha = 10°24'$; $C_{CC} = 1.07\ IN$; $C_{DD} = .272\ IN$

@PT. O ; $f_b = \dfrac{M_{CC}C_{CC}}{I_{CC}} + \dfrac{M_{DD}C_{DD}}{I_{DD}}$

$$f_b = \frac{(268-34)(1.5)(6.50)(1.07)}{.343} + \frac{(185+49)(1.50)(6.50)(.272)}{.015}$$

$$= 7117 + 41371 = 48{,}488 \text{ PSI (ULT)}$$

$$T = (185+49)(1.5)(1.08) - (268-34)(1.5)(.198) = 310 \text{ IN\# (ULT)}$$

$$M_{MAX} = \frac{T}{h}\, a \tanh\left(\frac{l}{a}\right) \quad \text{(REF: ROARK PG 178 \& 182)}$$

$h = 1.95\ IN$, $a = 3.044$, $l = 6.50\ IN.$

$M_{MAX} = 470\ IN\#$

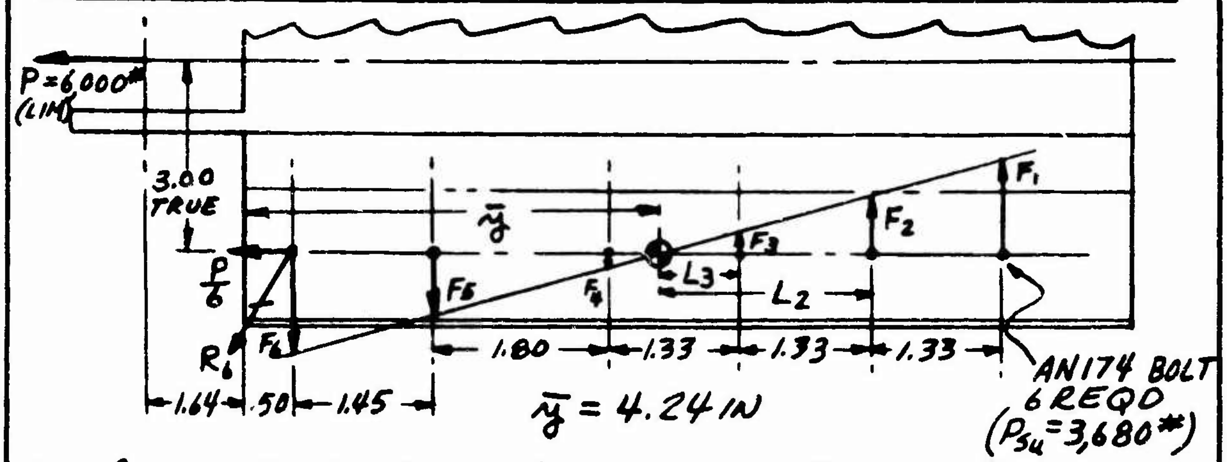
$$f_b' = \frac{M_{MAX}\, b}{I_{BB}} = \frac{470(1.00)}{.026} = 18{,}077 \text{ PSI}$$

TOTAL $f_b = 48{,}488 + 18{,}077 = 66{,}565$ PSI

$F_{tu} = 67{,}000$ PSI

$$M.S. = \frac{67000}{66565} - 1 = \underline{\underline{0}}$$

ACTUATOR SUPPORT BKT INSTALLATION (DWG No. 385-6191)



$\bar{y} = 4.24$ IN

$$3.00P = 3.74F_6 + 2.29F_5 + .49F_4 + .84F_3 + 2.17F_2 + 3.50F_1$$

$$F_n = \frac{F_6 L_n}{3.74};\ n = 1,2,3,4,5 \quad ;\ 3P = 3.74F_6 + \frac{F_6}{3.74}\sum_1^5 L_n^2$$

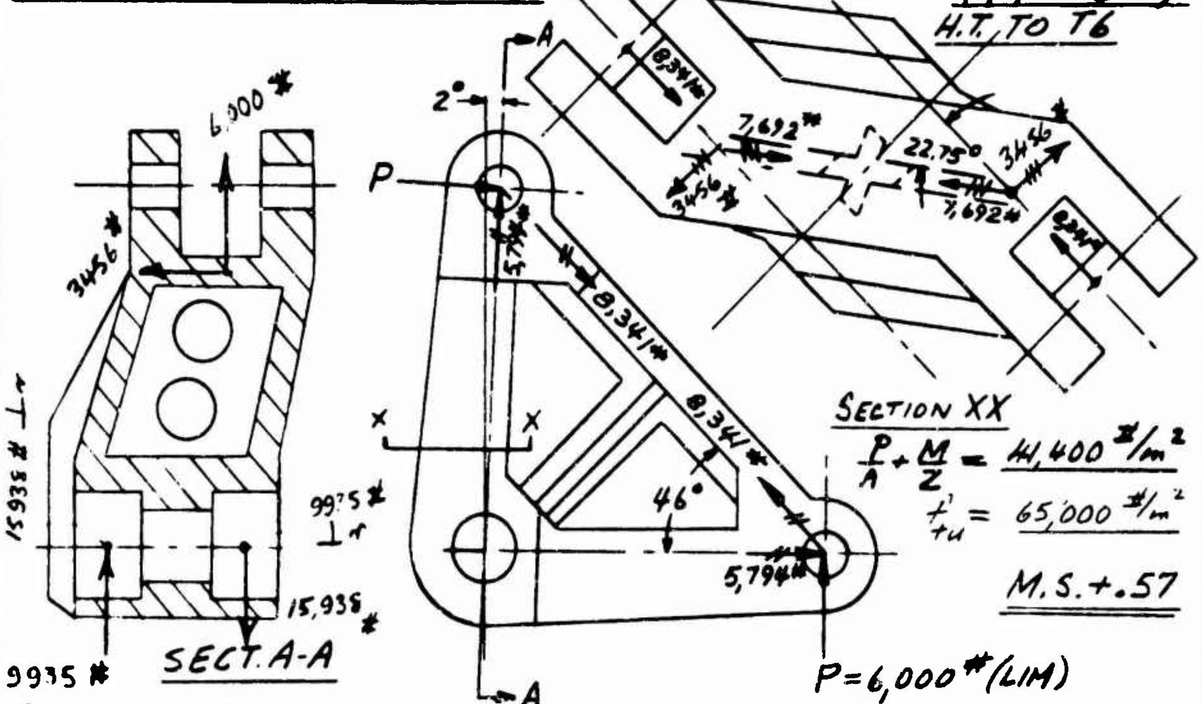
$$F_6 = \frac{3(3.74)P}{\sum^6 L_n^2} = .302\,P$$

$$R_6 = \left[\left(\frac{P}{6}\right)^2 + (.302)^2 P^2\right]^{1/2} = .344(6000)(1.5) = 3{,}096\#\ (\text{ULT})$$

$$M.S. = \frac{3680}{3096} - 1 = +.18$$

LATERAL CONTROL BELLCRANK (DWG NO. 385-6111)

MAT'L: 2014 ALUM. ALLOY (QQ-A-367) H.T. TO T6



SECTION XX

$$\frac{P}{A} + \frac{M}{Z} = 41{,}400\ \#/\text{in}^2$$

$$f_{tu} = 65{,}000\ \#/\text{in}^2$$

M.S. +.57

RESULTANT ON BEARINGS = 18,900 lbs (LIMIT) STRENGTH = 28,800# (LIM) M.S.+.52

BELLCRANK SUPPORT BKT ASS'Y (DWG. NO. 385-6119) & BELLCRANK SUPPORT BKT INSTALL. (DW'G NO. 385-6197)

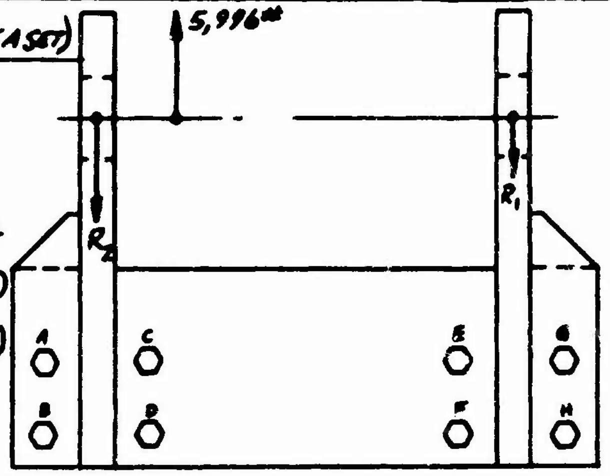
LOADS (ALL LOADS ARE REVERSIBLE AS A SET)

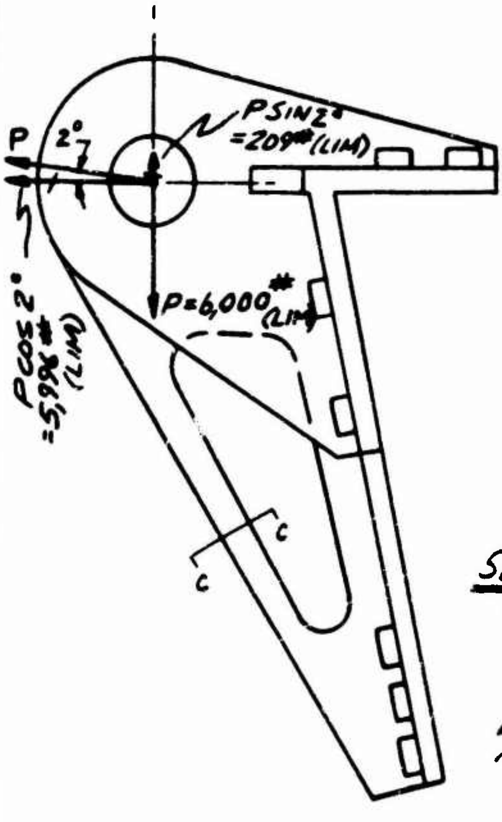
$$R_1 = \frac{5996(1.00)}{5.20} = 1{,}153^{\#}\,(LIM)$$

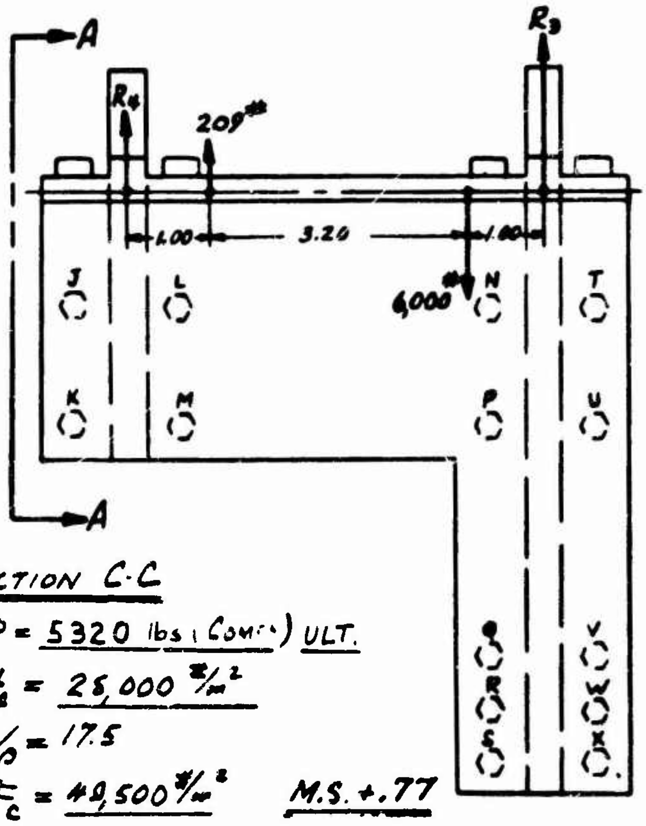
$$R_2 = 5996 - 1153 = 4843^{\#}\,(LIM)$$

$$R_3 = \frac{6000(4.20) - 209(1.00)}{5.20} = 4{,}806^{\#}\,(LIM)$$

$$R_4 = 6000 - 209 - 4806 = 985^{\#}\,(LIM)$$







SECTION C-C

$P = 5320$ lbs (COMP) ULT.

$f_c = 25{,}000\ \#/in^2$

$L/\rho = 17.5$

$F_c = 49{,}500\ \#/in^2$ M.S. +.77

MAT'L: 2014-T6 ALUM ALLOY (QQ-A-367)

$F_{tu} = 62{,}000$ PSI

$F_{cy} = 50{,}000$ PSI

$F_{su} = 38{,}000$ PSI

M.S. = + HIGH

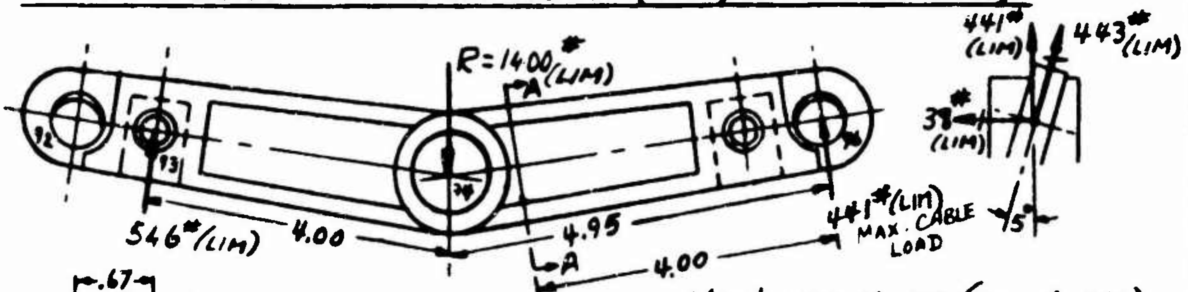
BOLTS: AN174 (1/4-28)

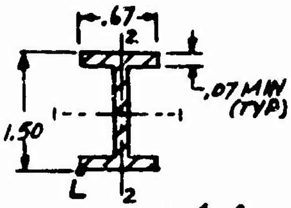
$P_{tu} = 4{,}080^{\#}$

$P_{su} = 3{,}680^{\#}$

M.S. = + HIGH

RUDDER CONTROL LEVER ASS'Y (DWG No. 385-6171)





SECTION A-A

MAT'L: 2014-T6 (QQ-A-266)

$F_{tu} = 65{,}000$ PSI

$F_{su} = 38{,}000$ PSI

$I_{11} = .063\ IN^4,\ C_{11} = .75\ IN$

$I_{22} = .0035\ IN^4,\ C_{22} = .335\ IN$

BENDING IS CRITICAL SECTION A-A

$M_{11} = 443(1.5)(4.00) = 2{,}658\ IN^{\#}$ (ULT)

$M_{22} = 38(1.5)(4.00) = 228\ IN^{\#}$ (ULT)

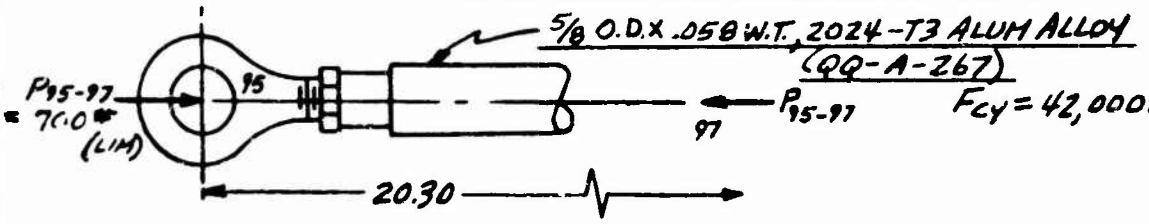
@ POINT L

$$f_t = \frac{Mc}{I}\Big|_{11} + \frac{Mc}{I}\Big|_{22} = \frac{2658(.75)}{.063} + \frac{228(.335)}{.0035}$$

$$= 31643 + 21823 = 53{,}466\ PSI$$

$$M.S. = \frac{65000}{53466} - 1 = \underline{\underline{+.21}}$$

RUDDER CONTROL TUBE ASS'Y (DWG No. 385-6173)



$F_{cy} = 42{,}000$ PSI

$A = .1033\ IN^2$

$I = .0042\ IN^4$

$\rho = .2015\ IN$

$$L'/\rho = \frac{20.30}{.2015} = 100.7$$

$F_c = 10{,}200$ PSI

$$f_c = \frac{P_{95\text{-}97}}{A} = \frac{700(1.5)}{.1033} = 10{,}164\ PSI\ (ULT)$$

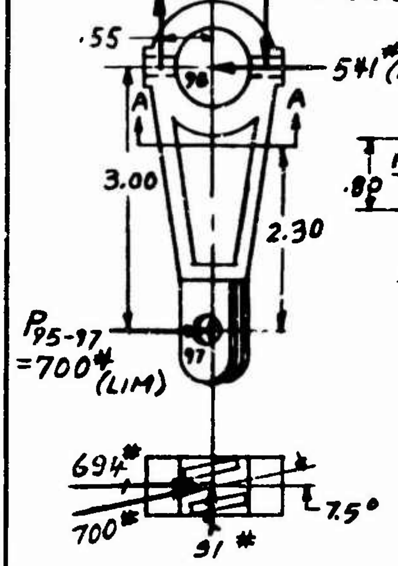
$$M.S. = \frac{10200}{10{,}164} - 1 = \underline{\underline{0.00}}$$

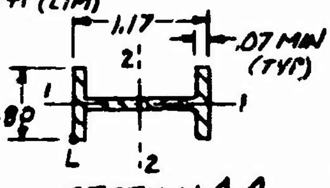
RUDDER OPERATING BELLCRANK (DWG No. 385-6174)

$R = \dfrac{541(3.00)}{2(.55)} = 1{,}475^{\#}_{(LIM)}$

MAT'L: 2014-T6 ALUM ALLOY (QQ-A-266)

$F_{tu} = 65{,}000\ PSI$





SECTION A-A

$I_{11} = .006\ IN^4,\ C_{11} = .40\ IN$

$I_{22} = .040\ IN^4;\ C_{22} = .585\ IN$

$M_{11} = 91(1.5)(2.30) = 314\ IN^{\#}(ULT)$

$M_{22} = 694(1.5)(2.30) = 2394\ IN^{\#}(ULT)$

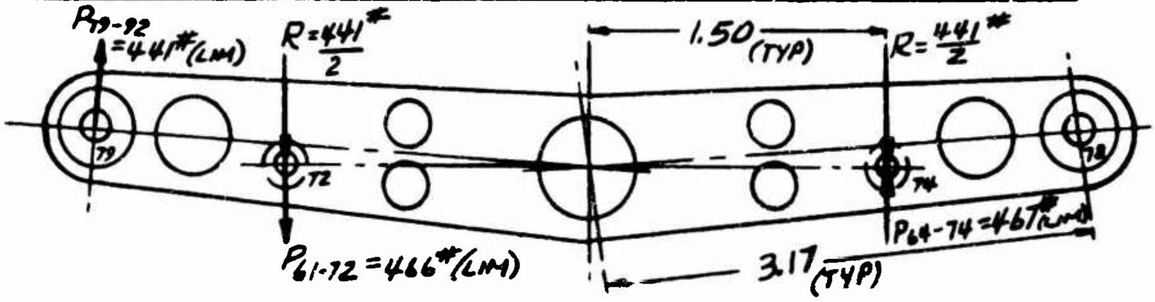
@POINT: L

$$f_t = \frac{Mc}{I}\bigg|_{11} + \frac{Mc}{I}\bigg|_{22} = \frac{314(.40)}{.006} + \frac{2394(.585)}{.040}$$

$$= 20{,}930 + 35{,}000 = 55{,}930\ PSI$$

$$M.S. = \frac{65000}{55{,}900} - 1 = \underline{+.16}$$

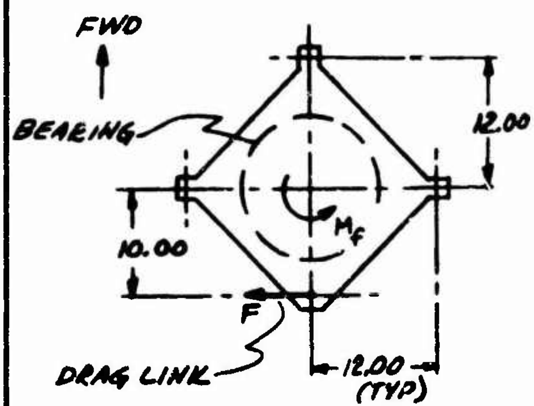
RUDDER CONTROL DRIVE PLATE (DWG No. 385-6151)

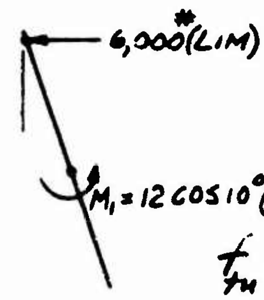


MAT'L: 2024-T3 ALUM. ALLOY (QQ-A-355)

$M.S. = +\underline{HIGH}$

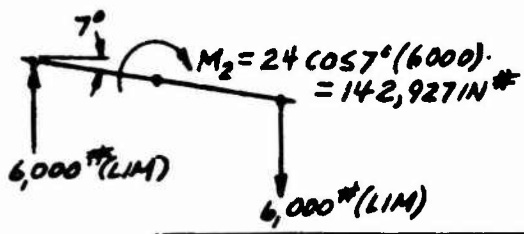
STATIONARY SWASHPLATE DRAG LINK (DWG. No. 385-6106)





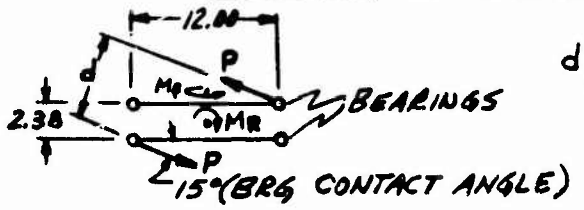
$M_1 = 12 \cos 10°(6000) = 70,906 \text{ IN}^{\#}$

f_{tu} OF WELDED TUBE $= 81,000 \#/\text{in}^2$



$M_2 = 24 \cos 7°(6000) = 142,927 \text{ IN}^{\#}$

$M_R = (M_1^2 + M_2^2)^{1/2}$
$= [(14.293)^2 + (7.091)^2]^{1/2}(10)^4$
$= 159,500 \text{ IN}^{\#}$

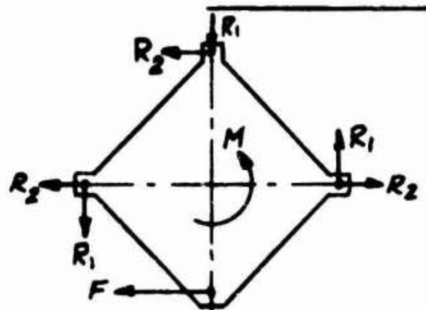


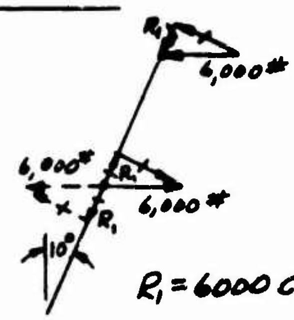
$d = 12 \sin 15° + 2.38 \cos 15°$
$= 5.405 \text{ IN.}$

$Pd = M_R$

$P = \dfrac{159500}{5.405} = 29,510^{\#}$

$M_f = P\mu(12)$; $\mu = .05$
$= 29510(.05)(12) = 17,706 \text{ IN}^{\#}$





$R_1 = 6000 \cos 80° = 1,042^{\#}$

$M = 24R_1 + 12R_2 = 24(1042) + 12(731)$
$= 33,780 \text{ IN}^{\#}$

$R_2 = 6000 \cos 83° = 731^{\#}$

$A = .174$

$P/A = \dfrac{7724}{.174} = 44,400 \#/\text{in}^2$

$F = \dfrac{M + M_f}{10.00} = \dfrac{33780 + 17706}{10}$
$= 5,149^{\#}$ (LIM)
$= 7,724^{\#}$ (ULT)

M.S +.82

APPENDIX III
CALCULATED ROTOR BLADE LIFE

The calculated service life of 107-1/2 hours for the XV-9A rotor blades is based on a typical flight load spectrum and on an S-N curve established by fatigue testing of full-scale rotor blade specimens.

FLIGHT LOAD SPECTRUM

The load spectrum of flight 13 is taken as a typical flight, as it includes speeds to 103 knots calibrated, turns, climbs, descents, and hovering turns (Reference 34).

S-N CURVE

The S-N curve used in this analysis is based on data from the fatigue testing of two full-scale specimens of the root end section of the rotor blade. They were tested for loads based on a weighted fatigue condition, which includes flapwise and chordwise bending, centrifugal load, and blade torsion (see Section 12 of Reference 9).

The data points used in developing the S-N curve are based on the cyclic axial stress in the spar and the corresponding number of cycles at which a crack devloped in the spar of the specimen. No reduction was made from test points to account for scatter in establishing the S-N curve. The spar, however, still sustained the test loads and had additional life before ultimate failure of the spar would have occurred.

ROTOR BLADE LIFE

A review of the load spectrum from the whirl tests and the tie-down tests shows that an insignificant amount of fatigue damage occurred during these tests. Therefore, the 15-hour flight test program is the only testing to date that has used up any of the rotor blade life.

DISCUSSION

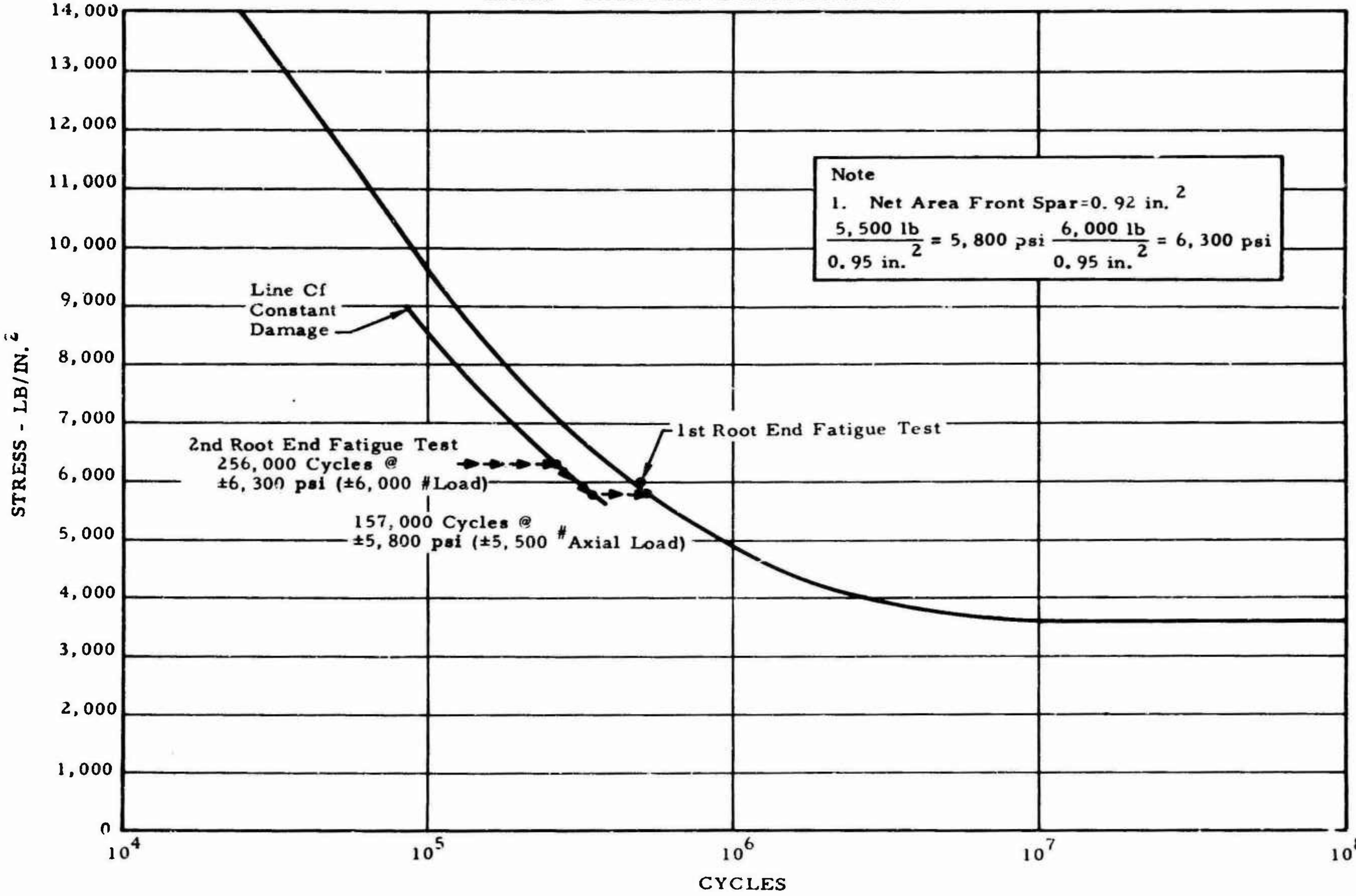
In both blade root and fatigue test specimens, cracks occurred in the spar at the bolt hole of the spar to segment attachment. These cracks developed between 400, 000 to 500, 000 load cycles. The

S-N curve is based on this data, which accounts for the low calculated service life of the XV-9A rotor blade.

The blade life could be improved by reducing the notch effect of the bolt hole in the spar. Some investigation of this problem was made by reduced-scale spar to segment attachment fatigue tests (see Section 14 of Reference 9). Various configurations for bushing the hole in the spar were tested.

Tests showed that a countersunk clamped up bushing installation through a clearance hole in the spar resulted in a large improvement in fatigue life.

S-N CURVE XV-9A HELICOPTER
SPARS - CHORDWISE BENDING STRESS



XV-9A ROTOR BLADE LIFE

FLIGHT 13 LOAD SPECTRUM (REF. 34)

LOAD SPEC. – FLT-13				
CYCLIC AXIAL LOAD STA. 90.75 LBS	CYCLIC AXIAL STRESS STA. 90.75 LBS/IN2	n CYCLES PER 100 HOURS	"N" LIFE CYCLES BASED ON 2ND FAT. TEST	n/N
4500	4890	445,000	950,000	.4680
5250	5700	65,000	560,000	.1160
5750	6250	14,000	400,000	.0350
6250	6800	7,000	300,000	.0233
6750	7340	4,900	235,000	.0208
7250	7875	3,700	185,000	.0200
7750	8425	1,100	150,000	.0073
8250	8970	400	125,000	.0032
8750	9500	300	105,000	.0030
				.6966

$$\text{SERVICE LIFE} = \frac{100 \text{ HOURS}}{.6966} \times .75 = 107\tfrac{1}{2} \text{ HOURS}$$

BASED ON S-N CURVE INCLUDING DATA FROM THE SECOND ROOT END FATIGUE SPECIMEN.

REFERENCES

1. Preliminary Design Study, Hot Cycle Research Aircraft, Summary Report, Hughes Tool Company, Aircraft Division Report 62-31, TREC CRD 62-102, U. S. Army Transportation Research Command,* Fort Eustis, Virginia, March 1963.

2. Hot Cycle Rotor Duct Closing Valve System, Hughes Tool Company, Aircraft Division Report 62-32, TREC Report 62-103, U. S. Army Transportation Research Command,* Fort Eustis, Virginia, March 1963.

3. Model Specification, XV-9A Hot Cycle Research Aircraft, Hughes Tool Company, Aircraft Division Report 62-22 (385-X-01), Revision E, October 1963.

4. Fernberger, J. M., T64 Gas Generator Data, General Electric Company Memorandum, T64 Applications and Installation Application, Lynn, Massachusetts, 15 March 1961.

5. Stevens, F. R., Jr., T64 Gas Generator Generalized Performance, Small Aircraft Engine Department Technical Memorandum TM SE 1570, General Electric Company, Lynn, Massachusetts, 1 July 1962.

6. Hot Cycle Rotor System Design Report, Hughes Tool Company, Aircraft Division Report 62-12, March 1962.

7. Installation Manual T-64 Engine, SEI-123, Small Aircraft Engine Division, General Electric Company, Lynn, Massachusetts, 31 March 1963.

8. Oil Cooler Tests, Report HE 6139, Harrison Radiator Division, General Motors Corporation, Lockport, New York, 24 March 1959.

9. Component Testing, XV-9A Hot Cycle Research Aircraft Summary Report, Hughes Tool Company, Aircraft Division Report 64-26, USAAML Technical Report 65-38, August 1965

10. Gessow, A., and Myers, G., Aerodynamics of the Helicopter, The Macmillan Company, New York, New York, 1952

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*In March 1965, the name of this Command was changed to U. S. Army Aviation Materiel Laboratories.

11. Hoerner, Sighard, F., Fluid Dynamic Drag (privately published), 1958.

12. Perkins, C. D., and Hage, R. E., Airplane Performance, Stability and Control, John Wiley Company, New York, New York, 1959.

13. Gessow, A., and Tapscott, R., Charts for Estimating Performance of High-Performance Helicopters, NACA Technical Note TN-3323, 1955.

14. Helicopter Flying and Ground Handling Qualities; General Requirements for, Military Specification, SPEC MIL-H-8501A, 7 September 1961.

15. Salmire, S., and Tapscott, R., The Effects of Various Combinations of Damping and Control Power on Helicopter Handling Qualities during Both Instrument and Visual Flight, NASA Technical Note TN D-58, October 1959.

16. Purser, Paul E., and Campbell, John P., Experimental Verification of a Simplified Vee-Tail Theory and Analysis of Available Data on Complete Models With Vee-Tails, NACA TR823, 1945.

17. Gerstenberger, W., and Wood, E., Analysis of Helicopter Aeroelastic Characteristics in High Speed Flight, Institute of Aeronautical Sciences Paper IAS 63-72, January 1963.

18. Dynamic Characteristics, Hot Cycle Rotor, Hughes Tool Company, Aircraft Division Report 62-14, March 1962.

19. Engine and Whirl Tests, XV-9A Hot Cycle Research Aircraft Summary Report, Hughes Tool Company, Aircraft Division Report 64-23, USATRECOM Technical Report 64-67, U. S. Army Transportation Research Command, Fort Eustis, Virginia, February 1965.

20. Hot Cycle Rotor System Structural Analysis, Volume I, Hughes Tool Company, Aircraft Division, Report 62-13, June 1962.

21. Structural Design Requirements, Helicopter (ASG), Military Specification, SPEC MIL-S-8698(1), 28 February 1958.

22. Hot Cycle Rotor System Preliminary Design, Hughes Tool Company, Aircraft Division Report 285-7, September 1956.

23. Graham, Donald J., Nitzberg, Gerald E., and Olson, Robert N., A Systematic Investigation of Pressure Distribution on Five Representative NACA Low-Drag and Conventional Airfoil Sections, NACA Technical Note TN832, 1945.

24. Buscher, R., On the Installation of Jet Engine Nacelles on a Wing, Fourth Partial Report, Pressure Distribution on a Sweptback Wing with Jet Engine Nacelles, NACA Technical Memorandum TM 1226, July 1949.

25. Request for Landing Gear Data, Letter SE-4810, to Hughes Tool Company, Aircraft Division, from Sikorsky Aircraft, Division - United Aircraft Corporation, Windsor Locks, Connecticut, Model CH-34A, 7 May 1962.

26. ANC-2 Ground Loads, Munitions Board Aircraft Committee, October 1952.

27. Asmus, J. F., Performance of T64 Gas Generators, General Electric Company, Company Memorandum, VTOL Operation, FPLD, Cincinnati, Ohio, 24 June 1959.

28. Hot Cycle Rotor Blade Structural Analysis, Volume II, Hughes Tool Company, Aircraft Division Report 62-13, June 1962.

29. Structural Principles and Data, Royal Aeronautic Society, New Era Publishing Company, LTD., London, England, 1953.

30. Hot Cycle Rotor System, Hub and Control System Structural Analysis, Volume III, Hughes Tool Company, Aircraft Division Report 62-13, June 1962.

31. Blake, Alexander, "How to Find Deflections and Moments of Rings in Arcuate Beams," Product Engineering, Volume 34, p 73, 7 January 1963.

32. Roark, Raymond J., Formulas for Stress and Strain, McGraw Hill Company, New York, New York, 1938.

33. Stress Analysis, Flight Controls, Model 369 Helicopter, Hughes Tool Company, Aircraft Division Report 369-S-5201, 24 July 1963.

34. Ground and Flight Tests, XV-9A Hot Cycle Research Aircraft Summary Report, Hughes Tool Company, Aircraft Division Report 65-13, May 1965.

UNCLASSIFIED

Security Classification

DOCUMENT CONTROL DATA - R&D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1 ORIGINATING ACTIVITY (Corporate author) Hughes Tool Company - Aircraft Division Culver City, California	2a REPORT SECURITY CLASSIFICATION Unclassified 2b GROUP

3 REPORT TITLE

AIRCRAFT DESIGN,
XV-9A HOT CYCLE RESEARCH AIRCRAFT
SUMMARY REPORT

4 DESCRIPTIVE NOTES (Type of report and inclusive dates)

Final Report, 28 September 1962 to 15 March 1965

5 AUTHOR(S) (Last name, first name, initial)

Hirsh, Norman B.

6 REPORT DATE	7a TOTAL NO OF PAGES	7b NO OF REFS
August 1965	343	34

8a CONTRACT OR GRANT NO DA 44-177-AMC-877(T) b PROJECT NO Task 1M121401A14403 c d	9a ORIGINATOR'S REPORT NUMBER(S) USAAVLABS Technical Report 65-29 9b OTHER REPORT NO(S) (Any other numbers that may be assigned this report) HTC-AD 64-11

10 AVAILABILITY LIMITATION NOTICES

Qualified requesters may obtain copies of this report from DDC.
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11 SUPPLEMENTARY NOTES	12 SPONSORING MILITARY ACTIVITY U. S. Army Aviation Materiel Laboratories Fort Eustis, Virginia

13 ABSTRACT

A summary of the design of the XV-9A Hot Cycle Research Aircraft is presented. A discussion of the concepts utilized in design and additional information relating to configuration, weight and balance, performance, stability and control, dynamics, and structural characteristics are presented.

DD FORM 1 JAN 64 1473

UNCLASSIFIED

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14. KEY WORDS	LINK A ROLE	LINK A WT	LINK B ROLE	LINK B WT	LINK C ROLE	LINK C WT
Hot Cycle Rotor System						
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