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GER No. 9861
CODE 15800
DATE 1-10-61
REP. NO. 397-3

GOODYEAR

GOODYEAR AIRCRAFT CORPORATION

AKRON, OHIO

STRESS ANALYSIS OF INFLATOPLANE

MODEL GA-468

GER-9861
397-3

Contract No.
NOnr 2360(00)

January 10, 1961

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REFERENCES

1. MIL-HDBK-5, Strength of Metal Aircraft Elements, March 1959.

2. Goodyear Aircraft Corporation, Structures Design Manual.

3. GER-0060, Effect of Cross-Section Shape on Stress Distribution in Inflated Airmat Panels, dated June 10, 1960.

4. GER-2942, Summary Aerodynamics Report of One-Man Inflatoplane, Model GA-460, dated 10 August 1960.

5. GER-10012, Preliminary Results of the Wind-tunnel Investigation of the Aerodynamic and Structural Deflection Characteristi : of Model GA-460 Inflatoplane, dated 14 October 1960.

6. Timoshenko and Woinowsky-Krieger, Theory of Plates and Shells, Second Edition, 1959, McGraw-Hill Book Company, Inc., New York.

7. Civil Aeronautics Manual 3, "Airplane Worthiness; Normal, Utility, and Aerobatic Categories." September 1954.

8. GER-7759, "Stress Concentrations and Deflections in Axi-symmetrically Loaded Spherical Envelopes", by A. D. Topping and J. D. Markette, February 8, 1957.

9. MIL-A-0629, Airplane Strength and Rigidity, 20 August, 1953.

REFERENCE DRAWINGS

447A-001	Assembly One Place Inflatoplane
447A-002	Wing-Assembly of
447A-003	Fuselage - Assembly of
447A-004	Empennage - Assembly of
447A-005	Cockpit - Assembly of
447A-007	Engine and Mount - Assembly of
447A-011	Engine Mount
447A-013	Universal Gear - Assembly of
447A-025	Bonded Assembly One Place Inflatoplane

INTRODUCTION

This report is submitted in partial fulfillment of paragraph 10 of Amendment 8 to Office of Naval Research Contract NOnr 2368(00), "Inflatoplanes".

The Stress Analysis for GA-468 Model Inflatoplane is divided into six sections. These sections are listed below.

Section 1	General
Section 2	Load Analysis
Section 3	Wing Analysis
Section 4	Fuselage Analysis
Section 5	Engine Mount Analysis
Section 6	Summary of Landing Gear, Cockpit, and Empennage Analyses

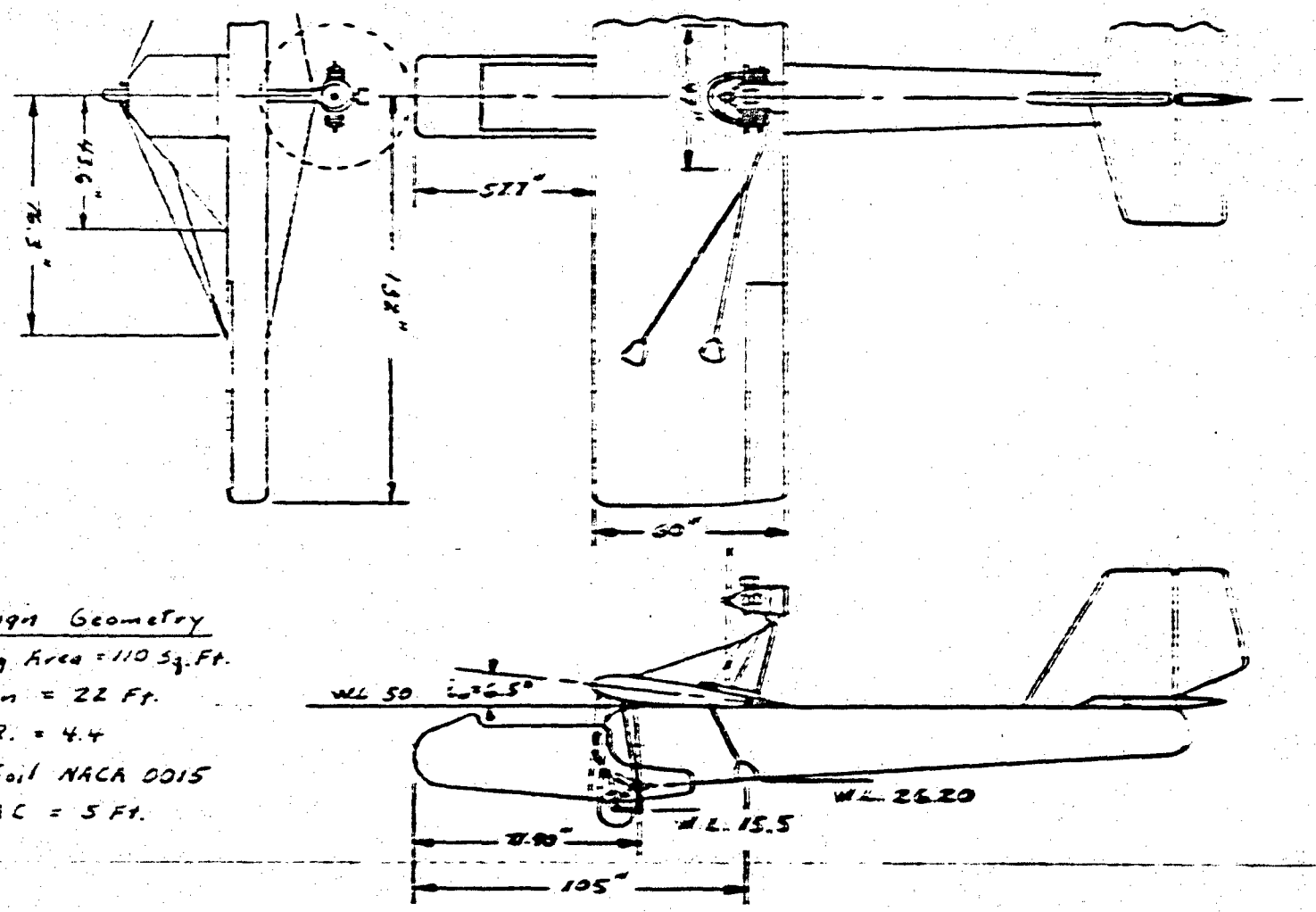
DISCUSSION

The Inflatoplane is a high wing monoplane with a Nelson H63A engine mounted on a pedestal above the wing and fuselage near the trailing edge of the wing. Each wing panel is restrained by two guy cables on the upper surface and three on the lower surface. The two upper cables are anchored to the engine pylon. The forward outboard and inboard cables are tied to the landing gear, while the aft outboard cable is attached to the bottom of fuselage at fuselage station 105.00. A single-seat cockpit is located forward of the conical shaped fuselage. A single wheel landing gear is mounted on the front hemispherical end of the fuselage; this together with the wing tip skids and a tail skid make up the landing gear system. Conventional tail surfaces are restrained by guy cables attached to the fuselage. A general layout with pertinent geometric data is shown in Figure 1.

The OA-460 Inflatoplane was static tested at Goodyear Aircraft to determine the buckling strength of the wing. An ultimate load factor of 5.6 was obtained.

A wind tunnel test was conducted in the NASA Langley full-scale wind tunnel to corroborate the static test results. This investigation resulted in ultimate load factors up to 5.13, reference 5.

An endurance load test was conducted at Goodyear Aircraft to determine the time effect under limit load on the Inflatoplane fabric. The Inflatoplane was inverted and a 2.5g limit load was applied to the wing and fuselage using shot bags. The test was run for 336 hours without appreciable creep.



Design Geometry

Wing Area = 110 Sq. Ft.

Span = 22 Ft.

A.R. = 4.4

Airfoil NACA 0015

MAC = 5 Ft.

Figure 1 Geometric Characteristics Of The Inflatoplane

GOODYEAR AIRCRAFT

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TABLE 1

CLOTH SPECIFICATIONS, INFLATOPLANE

1.	Goodyear Code	477	484	3503N	3511N	3514N	35[illegible]
2.	Material	Dacron	Dacron	Nylon	Nylon	Nylon	Nyl[illegible]
3.	Nominal Weight oz/sq.yd.	4.00	3.50	3.10	2.03	8.60	1.[illegible]
4.	Weight Tolerance oz/sq.yd.	± .13	± .20	.6	± .10	± .25	± .[illegible]
5.	Tensile-Min-lbs/inch-Warp	200	273	132	90	160	[illegible]
6.	-Fill	180	250	132	90	125	[illegible]
7.	Tensile Test Method (Quick Break)	1" Strip	1" Strip	1" Strip	1" Strip	1" Grab	2" [illegible]
8.	Denier-Warp	220/1	220/1	210/1	70/2	70/2	40/[illegible]
	-Fill	220/1	222/1	210/1	70/2	210/1	70/[illegible]
	-Pile	---	---	---	---	70/2	---
9.	Count-Warp, Minimum Ends/inch	62	84	49	96	73	11[illegible]
10.	Count-Fill, Minimum Ends/inch	53	78	49	96	40	[illegible]
11.	Count-Pile-Nominal, Tolerance (yarns/sq.in.)	--	--	--	--	30, ±5	--

1

TABLE I

CLOTH SPECIFICATIONS, INFLATOPLANE

484	3503N	3511N	3514N	3523N	2044	3364	7008B	8936	8937
Dacron	Nylon	Nylon	Nylon	Nylon	Dacron	Dacron	Nylon	Nylon	Nylon
3.50	3.10	2.03	0.60	1.20	6.50	5.00	4.25	15.00	9.25
± .20	.6	± .10	± .25	± .10	± .20	± .15	± .15	± .40	± .30
275	152	90	160	50	312	257	231	160	160
250	152	90	125	50	293	255	210	125	125
1" Strip	1" Strip	1" Strip	1" Grab	2" Strip	1" Strip	1" Strip	Grab	Grab	Grab
220/1	210/1	70/2	70/2	40/1	220/2	220/1	210/1	70/2	70/2
222/1	210/1	70/2	210/1	70/2	220/2	220/1	210/1	210/1	210/1
---	---	---	70/2	---	---	---	---	70/2	70/2
84	49	96	73	117	50	79	71	73	73
78	49	96	40	52	49	73	71	40	40
--	--	--	30, ±5	--	--	--	--	30, ±5	30, ±5

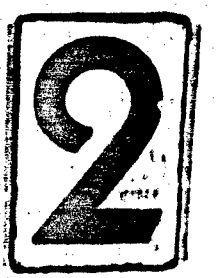

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TABLE II

FABRIC SPECIFICATIONS, LIFTATOPLANE

Processed Fabric Specifications

		1	2	3	4	5
		(Present)	(Proposed)		Empennage Aileron	
1.	Classification	Wing	Wing	Cockpit	& Flap	Fuselage(2)
2.	Goodyear Code	A350	A350	A351	A349	N313A105
3.	Outside Color	Plain	Plain	Plain	Plain	Plain
4.	Number of Plies	3	3	2	2	2
5.	Construction (outside to inside)	(1)	(1)	(1)	(1)	
	a.) Spread (oz/sq yd)	1.25	1.35	---	---	1.20
	b.) Cloth "	1.40BL	2.05BL	---	---	4.00B
	c.) Spread "	2.50	2.70	1.25	1.25	4.50
	d.) Cloth "	1.40BR	2.05BR	2.05S	2.05S	4.00S
	e.) Spread "	3.00	3.10	5.50	5.00	1.00
	f.) Airmat Cloth "	15.00S	15.00S	8.60S	9.25S	----
	g.) Spread "	----	----	----	----	----
6.	Nominal Weight - oz/sq yd	34.00	39.50	26.20	26.00	14.70
7.	Weight Tolerance - oz/sq yd	1.70	1.95	1.25	1.50	.50
8.	Tensile-Min-lbs/inch-Warp	180	180	150	140	300
9.	-Min-lbs/inch-Fill	174	174	150	140	260
10.	Min-lbs/sq inch-Pile	28	28	28	28	---
11.	Tensile Test Method					Cyl Burst
12.	Material	Nylon	Nylon	Nylon	Nylon	Dacron
13.	Cloth-Outside to Inside	3523N	3511N	3511N	3511N	
		3523N (1)	3511N (1)	3514N (1)	8937 (1)	477-477
		8936	8936			

(1) For Airmat construction each side is symmetrical.
(2) Also used for cockpit assembly straps.
(3) Used for reinforcement of fuselage fabric N313A105 on aircraft 4106-4113.
(4) Used for replacement fuselages on aircraft 4114 and 4115.
(5) To be used as replacement fuselages on aircraft 4106, 4108-4113.
(6) Used for hinge straps, D-ring, pulleys and fan patches, and reinforcement in lacing p
(7) Used for instrument brackets, lacing patches, scuff patches, and seam tape on wings.

TABLE II

FABRIC SPECIFICATIONS, INFLATOPLANE

	3	4	5	6	7	8	9	10
ed)	Cockpit A351 Plain 2 (1)	Empennage Aileron & Flap A349 Plain 2 (1)	Fuselage(2) N313A105 Plain 2	(3) N368A10 Plain 2	(4) ZX321 Plain 2	(5) ZX 356 Plain 2	Strap (6) ZX300 Plain 2	(7) Accessory A330 Plain 1
	---	---	1.20	1.10	1.20	2.00	1.25	4.50
	---	---	4.00B	4.00S	5.00B	5.50S	4.25S	3.10
	1.25	1.25	4.50	4.50	4.00	4.00	4.50	4.50
	2.05S	2.05S	4.00S	4.00S	6.50S	5.50S	4.25S	----
	5.50	5.00	1.00	1.10	1.00	1.00	1.25	----
	8.60S	9.25S	----	----	----	----	----	----
	----	----	----	----	----	----	----	----
	26.20	26.00	14.70	14.70	17.70	18.00	15.50	12.1
	1.25	1.50	.50	.50	.50	.50	1.00	.5
	150	140	300	300	435	410	425	190
	150	140	260	300	425	410	350	190
	28	28	---	---	---	---	---	---
	Nylon 3511N	Nylon 3511N	Cyl Burst Dacron	Cyl Burst Dacron	Cyl Burst Dacron	Cyl Burst Dacron	Strip Nylon	Grab Nylon
(1)	3514N (1)	8937 (1)	477-477	477-477	3364-2044	484-484	7008B-7008B	3503N

metrical.

c N313A105 on aircraft 4106-4113.
ft 4114 and 4115.
ircraft 4106, 4108-4113.
nd fan patches, and reinforcement in lacing patches.
tches, scuff patches, and seam tape on wings.

2

Summary of Minimum Margins of Safety

Page No.	Part Name	Drawing No.	Critical Condition	Type of Loading or Stress	M.S.
3.01.240	Wing, Patches & Cables	117A-002	Wind Tunnel Test	Ultimate	+0.17
4.01.050	Fuselage	117A-003	Condition F-11 & F-13	Hoop Tension with Longitudinal Stress = 0	+0.03
5.01.030	Engine Mount -11 Strut	117A-001	Ult. Side Load	Combined Axial, Bending & Torque	+0.01
6.01.050	Empennage	117A-004	A-11 Rudder Kick	Limit Bending	+1.51
6.01.150	Cockpit	117A-005	Condition 5	Ultimate Bending	-0.02
6.01.170	Landing Gear -13 Ring & -17 Tee Brace	117A-013	Condition 3	Ultimate Bending	+0.14

Design Criteria

The Design Criteria covers the design conditions, factors of safety, and allowable stresses for the Inflatoplane.

Design Conditions

The Design Conditions were selected using reference (9) as the specification. The maneuvering design conditions are in accordance with paragraph 3.4.2 of reference (9) for the VW classification (Special Search) except that the maximum limit load factor is 2.5 instead of 3.0. Two gross weights are investigated, namely, 550 lbs with most forward c.g. and 404 lbs with the most aft c.g. Maximum flight limit speed is 74.5 knots. Under gusting conditions the limit speed is 44 to 47 knots, based on maneuvering load factor of -1.0. The critical loads are summarized on page 2.00.030.

Ground handling loads are not critical. The landing conditions are based upon a sinking speed of 5 ft/sec. The critical landing conditions are level landing, tail down landing, and side load specified in reference (7).

Factors of Safety

The limit loads are multiplied by the following factors to obtain the structural design loads.

Metal Structure	Yield	1.15
Metal Structure	Ultimate	1.50
Fabric Structure	Wrinkling*	1.00
Fabric Structure	Ultimate	1.75

* No principal stress shall be less than zero at limit load.

Allowable Strength

For the metal structures, the requirements of paragraphs 3.2.1.2 through 3.2.1.8 of reference (9) are used.

The following reduction factors shall be applied to the Quick Break Strength of the fabric structures:

Inflation Only	4
Limit Load	3
Ultimate Load	1.5

The reduction factor for inflation and limit load are based on past experience and account for the fact that fabric under load for a period of time has a reduction in strength.

The reduction factor for ultimate load was chosen arbitrarily.

LOADS

SECTION 2

The critical flight loads shown in Table III, Summary of Critical Flight Loading Conditions were developed using Table 1 (Summary of Airloads), Figure 6 (V-n_z Diagram M.F.C.G. 550 lbs), and Figure 7 (V-n_z Diagram M.A.C.G. 404 lbs) of reference (4). The letters refer to points on the V-n_z diagram and subscripts to the different conditions applicable to the particular point. The loading conditions specified produced critical design loads for all the primary structure.

The weight distribution, C.G. locations, and moment of inertia calculations are found on pages 2.01.010 through 2.01.090.

Each part of the Loads Section has its own discussion and sketches.

Summary of Critical Loading Conditions

Table III

Description	Designation	Velocity	Angle of Attack	Aileron Deflection	Load Factor	Angular Accel. Pitch	Angular Accel. Roll	Angular Accel. Yaw	Wing Loads Port Lift	Wing Loads Port Drag	Wing Loads Stbd. Lift	Wing Loads Stbd. Drag	Empennage Loads Horizontal Port	Empennage Loads Horizontal Stbd.	Empennage Loads Vert.	Engine
		Knots	Deg.	Deg.		rad/sec²	rad/sec²	rad/sec²	Lb.	Lb.	Lb.	Lb.	Lb.	Lb.	Lb.	Lb.
Symmetrical Maneuver	A2	V	13.6	0	2.5	-1.61	0	0	632	79	632	79	30	30	0	0
	A3				2.5	+3.49	0	0	632	79	632	79	-31.5	-31.5	0	120
Unsymmetrical Maneuver	A4		11	20 up 15 dn	2.0	-1.61		0	652	79	448	40	36	29	0	0
	A5			--	2.0	+3.49		0	652	79	448	40	-42	-32	0	100
Gust — Symmetrical	A6	515	5.5	0	1.0	0	0	0	275	20	275	20	38.5	38.5	0	100
Gust — Unsymmetrical	A7		5.5	0	1.0	0	0	0	275	20	275	20	37	24	0	100
Steady Sideslip	A8		5.5	0	1.0	0	0	0	275	20	275	20			69	100
Rolling Pullout — Zero Slip	A9		11	25 up 15 dn	2.0	0	0	0	652	79	448	40			18	100
Rolling Pullout — Zero Rudder	A10		11	-	2.0	0	0	0	652	79	448	40			142	100
Rudder Kick	A11	395	4.85	0	1.0	0	0	0	275	22	275	22			181	120
Gust - Vertical Tail	A12	525	5.5	0	1.0	0	0	0	275	20	275	20			46	100
Combined Loads	A13				2.5 - 2.12	0							1	-6	46 - 142	
Symmetrical Maneuver	B2	▲	8	0	2.5	-2	0	0	688	64	688	64	23.5	23.5	0	121
	B3				2.5	2.65	0	0	688	64	688	64	-28.5	-28.5	0	121
Unsymmetrical Maneuver	B4		6.4	20 up 15 dn	2.0	-2		0	768	66	488	36	28	19	0	121
	B5		6.4	-	2.0	2.65		0	768	66	488	36	-35	-24		121
Gust — Symmetrical	B6	520	3.2	0	1.0	0	0	0	275	225	275	225	40	40	0	121
Gust — Unsymmetrical	B7		3.2	0	1.0	0	0	0	275	205	275	205	42	32	0	121
Steady Sideslip	B8		3.2	0	1.0	0	0	0	275	205	275	205			63	121
Rolling Pullout — Zero Slip	B9		6.4	25 up 15 dn	2.0	0									18	121
Rolling Pullout — Zero Rudder	B10	▼	6.4	-	2.0	0									142	
Rudder Kick	B11	395				SAME AS DESIGNATION A11										
Gust - Vertical Tail	B12	520				SPEED RESTRICTED IN GUSTS										
Combined Loads	B13				2.5 - 2.12								1	-17.5	46 - 142	
Symmetrical Maneuver	C2	▲	3.4	0	2.5	-1	0	0	688	53	688	53	16	16	0	120
	C3		3.4	0	2.5	1.52	0	0	688	53	688	53	-20	-20	0	120
Unsymmetrical Maneuver	C4		2.7	25 up 15 dn	2.0	-1		0	768	59	332	32	19	13		120
	C5		2.7	-	2.0	1.52		0	768	59	332	32	-24	-16		120
Gust — Symmetrical	C6	465	1.9	0	1.0	0	0	0	275	23	275	23	35	35		120
Gust — Unsymmetrical	C7		1.9	0	1.0	0	0	0	275	23	275	23	42	28		120
Steady Sideslip	C8		1.9	0	1.0	0	0	0	275	23	275	23			130	120
Rolling Pullout — Zero Slip	C9		2.7	25 up 15 dn	2.0	0									18	120
Rolling Pullout — Zero Rudder	C10	▼	2.7	-	2.0	0									142	120
	C11	395				SAME AS DESIGNATION A11										

Prepared by: [illegible]

Checked by: [illegible]

Date: [illegible]

Revised: [illegible]

UNSYMMETRICAL	C1		3.2	0	1.0	0	0	0	275	20.5	275	20.5	42	32	0	121				
STEADY SIDESLIP	B8		3.2	0	1.0	0	0	0	275	20.2	275	20.5			53	121				
ROLLING PULLOUT ZERO SLIP	B9		6.4	25 up 25 dn	2.0	0									118	121				
ROLLING PULLOUT ZERO RUDDER	B10	V	6.4	~	2.0	0									112					
RUDDER KICK	B11	395				SAME	AS	DESIGNATION	A10											
GUST - VERTICAL TAIL	B12	62				SPEED	RESTRICTED	IN	GUSTS											
COMBINED LOADS	B13				EQ-UAL									I	-17.4	EQ-UAL				
SYMMETRICAL MANEUVER	C2		3.4	0	2.5	-1	0	0	688	53	688	53	16	16	0	120				
	C3		3.4	0	2.5	1.92	0	0	688	53	688	53	-20	-23	0	120				
UNSYMMETRICAL MANEUVER	C4		2.7	25 up 25 dn	2.0	-1		0	768	59	332	32	19	13		120				
	C5		2.7	~	2.0	1.92		0	768	59	332	32	-24	-16		120				
GUST SYMMETRICAL	C6	345	1.4	0	1.0	0	0	0	275	23	275	23	35	35		121				
GUST UNSYMMETRICAL	C7		1.4	0	1.0	0	0	0	275	23	275	23	42	28		120				
STEADY SIDESLIP	C8		1.4	0	1.0	0	0	0	275	23	275	23			130	120				
ROLLING PULLOUT ZERO SLIP	C9		2.7	25 up 25 dn	2.0	0									118	120				
ROLLING PULLOUT ZERO RUDDER	C10		2.7	~	2.0	0									112	120				
RUDDER KICK	C11	345				SAME	AS	DESIGNATION	A11											
GUST - VERTICAL TAIL	C12					SPEED	RESTRICTED	IN	GUSTS											
COMBINED LOADS	C13				EQ-UAL C12									I	-9	EQ-UAL C11				
SYMMETRICAL MANEUVER	E2		-3.2	0	-1.0	-3.53	0	0	-275	20	-275	20	325	325		124				
	E3		-3.2	0	-1.0	2.0	0	0	-275	20	-275	20	-325	-325		124				
UNSYMMETRICAL MANEUVER	E4		0	25 up 25 dn	0	-3.53		0					46	31		121				
	E5		0	~	0	2.00		0					-41	-29		121				
GUST SYMMETRICAL	E6	62	3.2	0	1.0	0	0	0	275	225	275	205	-455	-455		124				
GUST UNSYMMETRICAL	E7		3.2	0	1.0	0	0	0					-55	-36		121				
STEADY SIDE SLIP	E8		3.2	0	1.0	0	0	0	275	20.2	275	21.2			93	121				
ROLLING PULLOUT ZERO SLIP	E9		0	25 up 25 dn	0	0									0	121				
ROLLING PULLOUT ZERO RUDDER	E10		0	~	0	0									0	121				
RUDDER KICK	E11	345				SAME	AS	DESIGNATION	A11											
GUST - VERTICAL TAIL	E12	62				SPEED	RESTRICTED	IN	GUSTS											
COMBINED LOADS	E13				EQ-UAL E12									5	-9	EQ-UAL E11				
SYMMETRICAL MANEUVER	F2		-17.6	0	-1.0	-2.0	0	0	-275	31	-275	31	57	57						
	F3		-17.6	0	-1.0	2.0	0	0	-275	31	-275	31	-57	-57						
UNSYMMETRICAL MANEUVER	F4		0	25 up 25 dn	0	-2.0							42	42						
	F5		0	~	0	2.0							-46	-46						
GUST SYMMETRICAL	F6	34	17.6	0	1.0	0	0	0	275	31	275	31	35	35						
GUST UNSYMMETRICAL	F7		17.6	0	1.0	0	0	0	275	31	275	31	-40	-28						
STEADY SIDESLIP	F8		17.6		1.0	0	0	0	275	31	275	31			28					
ROLLING PULLOUT ZERO SLIP	F9																			
ROLLING PULLOUT ZERO RUDDER	F10																			
RUDDER KICK	F11	345				SAME	AS	DESIGNATION	A11											
GUST - VERTICAL TAIL	F12	34				SPEED	RESTRICTED	IN	GUSTS											
COMBINED LOADS	F13				EQ-UAL F12									.5	-39	EQ-UAL F11				

PREPARED BY M.E.C.
CHKD AND BY
DATE
REVISED

GOODYEAR AIRCRAFT

PAGE 8.00.00
MODEL GA-468
SER
REP NO GER-

Discussion

Weight and Balance Section

This section of this report presents the weight and balance data and the moments of inertia of the aircraft.

There are two parts. The first is the Group Weight Statement (AN-9103-D) and the second portion incorporates the moments of inertia.

In the Group Weight Statement the weight of the pilot is assumed to be 200 lbs. For the most forward C.G. condition, 240 lbs. was chosen as the pilot's weight and for the most aft C.G. condition, 160 lbs. was used as the weight of the pilot. It is quite obvious that for an aircraft of such low empty weight, the C.G. is affected considerably by the weight of the pilot.

The horizontal datum (from which the X distances are measured) is the nose of the aircraft (Station 0). The vertical datum (from which the Z distances are measured) is the ground line. The lateral datum (from which the Y distances are measured) is the Centerline of the aircraft.

AN-9103-D
SUPERSEDING
AN-9103-C

NAME [illegible]
DATE December 12, 1940

PAGE 2.01.030
MODEL GA-460
REPORT GER-2061

GROUP WEIGHT STATEMENT

ESTIMATED · CALCULATED · ACTUAL
(Cross out those not applicable)

CONTRACT NO. NO nr 2360(00)
AIRPLANE, GOVERNMENT NO. YAO-3(C)
AIRPLANE, CONTRACTOR NO. GA-460
MANUFACTURED BY GOODYEAR AIRCRAFT CORPORATION

		MAIN	AUXILIARY
ENGINE	MANUFACTURED BY	Nelson	
	MODEL	H-63A Modified	
	NO.		
PROPELLER	MANUFACTURED BY	U.S. Propellers	
	DESIGN NO.	Model 380-31 47 In. Dia., Wood	
	NO.		

AN-9101-D
NAMK (illegible)
DATE December 17, 1960

GROUP WEIGHT STATEMENT
WEIGHT EMPTY

PAGE 2.01.010
MODEL CA-400
REPORT CRL-2001

1	WING GROUP					46.2
2	CENTER SECTION - BASIC STRUCTURE					
3	INTERMEDIATE PANEL - BASIC STRUCTURE				25.5	
4	OUTER PANEL - BASIC STRUCTURE (INCL. TIPS LBS.)					
5						
6	SECONDARY STRUCTURE (INCL. WINGFOLD MECHANISM LBS.)				11.0	
7	AILERONS (INCL. BALANCE WEIGHT LBS.)				3.3	
8	FLAPS - TRAILING EDGE				3.4	
9	- LEADING EDGE					
10	SLATS					
11	SPOILERS					
12	SPEED BRAKES					
13	INFLATION AIR				3.7	
14						
15	TAIL GROUP					10.6
16	STABILIZER - BASIC STRUCTURE				3.0	
17	FINS - BASIC STRUCTURE (INCL. DORSAL LBS.)				3.3	
18	SECONDARY STRUCTURE (STAB. & FINS)				1.4	
19	ELEVATOR (INCL. BALANCE WEIGHT LBS.)				2.5	
20	RUDDERS (INCL. BALANCE WEIGHT 4.5 LBS.)				6.0	
21	INFLATION AIR				.4	
22						
23	BODY GROUP					41.2
24	FUSELAGE OR HULL - BASIC STRUCTURE (ENVELOPE)				20.4	
25	BOOMS - BASIC STRUCTURE					
26	SECONDARY STRUCTURE - FUSELAGE OR HULL (CKPIT & FURNISHINGS)				10.2	
27	- BOOMS					
28	- SPEEDBRAKES					
29	- DOORS, PANELS & MISC.					
30	INFLATION AIR				1.2	
31	ALIGHTING GEAR GROUP - LAND (TYPE: UNICYCLE)					23.0
32	LOCATION	WHEELS, BRAKES TIRES, TUBES, AIR	STRUCTURE	CONTROLS		
33						
34	MAIN	5.3	16.1		21.4	
35	TAIL SKID				.7	
36	WING TIP SKIDS				.9	
37						
38						
39						
40	ALIGHTING GEAR GROUP - WATER					
41	LOCATION	FLOATS	STRUTS	CONTROLS		
42						
43						
44						
45						
46	SURFACE CONTROLS GROUP					2.7
47	COCKPIT CONTROLS - STICK				.7	
48	AUTOMATIC PILOT					
49	SYSTEM CONTROLS (INCL. POWER & FEEL CONTROLS LBS.)				2.0	
50						
51	ENGINE SECTION OR NACELLE GROUP					9.5
52	INBOARD					
53	CENTER					
54	OUTBOARD					
55	DOORS, PANELS & MISC.					
56						
57	TOTAL (TO BE BROUGHT FORWARD)					141.9

AN-9103-D
NAME
DATE December 19, 1960

GROUP WEIGHT STATEMENT
WEIGHT EMPTY

PAGE 2.01.010
MODEL OA-460
REPORT 9861

		AUXILIARY	MAIN		
1	PROPULSION GROUP				68.5
2					
3	ENGINE INSTALLATION			56.2	
4	AFTERBURNERS (IF FURN. SEPARATELY)				
5	ACCESSORY GEAR BOXES & DRIVES				
6	SUPERCHARGERS (FOR TURBO TYPES)				
7	AIR INDUCTION SYSTEM				
8	EXHAUST SYSTEM				
9	COOLING SYSTEM				
10	LUBRICATING SYSTEM				
11	TANKS				
12	COOLING INSTALLATION				
13	DUCTS, PLUMBING, ETC.				
14	FUEL SYSTEM			5.6	
15	TANKS - PROTECTED		3.1		
16	- UNPROTECTED				
17	PLUMBING, ETC.		2.5		
18	WATER INJECTION SYSTEM				
19	ENGINE CONTROLS			.8	
20	STARTING SYSTEM				
21	PROPELLER INSTALLATION			5.9	
22					
23					
24	AUXILIARY POWER PLANT GROUP				
25	INSTRUMENTS & NAVIGATIONAL EQUIPMENT GROUP				7.6
26	[redacted] PNEUMATIC GROUP				11.9
27					
28					
29	ELECTRICAL GROUP				2.0
30	BATTERY			1.4	
31	COCKPIT LIGHT			.6	
32	ELECTRONICS GROUP				
33	EQUIPMENT				
34	INSTALLATION				
35					
36	ARMAMENT GROUP (INCL. GUNFIRE PROTECTION LBS.)				
37	FURNISHINGS & EQUIPMENT GROUP				
38	ACCOMMODATIONS FOR PERSONNEL				
39	MISCELLANEOUS EQUIPMENT				
40	FURNISHINGS				
41	EMERGENCY EQUIPMENT				
42					
43	AIR CONDITIONING & ANTI-ICING EQUIPMENT GROUP				
44	AIR CONDITIONING				
45	ANTI-ICING				
46					
47	PHOTOGRAPHIC GROUP				
48	AUXILIARY GEAR GROUP				
49	HANDLING GEAR				
50	ARRESTING GEAR				
51	CATAPULTING GEAR				
52	ATO GEAR				
53					
54					
55	MANUFACTURING VARIATION				
56	TOTAL FROM PG. 2				141.9
57	WEIGHT EMPTY				231.9

AN-9101-D
NAME [illegible]
DATE December 19, 1960

GROUP WEIGHT STATEMENT
USEFUL LOAD & GROSS WEIGHT

PAGE 2.01.050
MODEL OA-460
REPORT OER-9061

1	LOAD CONDITION						
2							
3	CREW (NO. 1)						200.0
4	PASSENGERS (NO.)						
** 5	FUEL & OIL	Type **	Gals.				
6	UNUSABLE						
7	INTERNAL	at 6.09#/Gal.	19.4				118.1
8							
9							
10	EXTERNAL						
11							
12	BOMB BAY						
13							
** 14	OIL						**
15	TRAPPED						
16	ENGINE						
17							
18	FUEL TANKS (LOCATION)						
19	WATER INJECTION FLUID (GALS)						
20							
21	BAGGAGE						
22	CARGO						
23							
24	ARMAMENT						
25	GUNS (Location)	Pin. or Pinn.	Cap.	Cal.			
26							
27							
28							
29							
30							
31							
32	AMMUNITION						
33							
34							
35							
36							
37							
38							
39	INSTALLATIONS (BOMB, TORPEDO, ROCKET, ETC.)						
*40	BOMB OR TORPEDO RACKS						
41							
42							
43							
44							
45							
46	EQUIPMENT						
47	PYROTECHNICS						
48	PHOTOGRAPHIC						
49							
*50	OXYGEN						
51							
52	MISCELLANEOUS						
53							
54							
55	USEFUL LOAD						318.1
56	WEIGHT EMPTY						231.9
57	GROSS WEIGHT						550.0

*If not specified as weight empty.

** Fuel & oil are pre-mixed (by volume) in ratio 16 fuel/1 oil.

AL-9103-D
NAME [illegible]
DATE December 17, 1960

GROUP WEIGHT STATEMENT
DIMENSIONAL & STRUCTURAL DATA

PAGE 2.01.060
MODEL OA-460
REPORT GER-9061

1 LENGTH - OVERALL (FT.)	21.09			HEIGHT - OVERALL - STATIC (FT.)			8.1
2	Main Floats	Aux. Floats	Beams	Fuse or Hull	Inboard	Nacelles Center	Outboard
3 LENGTH - MAX. (FT.)				20.17			
4 DEPTH - MAX. (FT.)				2.96			
5 WIDTH - MAX. (FT.)				2.25			
6 WETTED AREA (SQ. FT.)							
*7 FLOAT OR HULL DISPL. - MAX. (LBS.)							
8 FUSELAGE VOLUME (CU. FT.)		PRESSURIZED			TOTAL		
9					Wing	H. Tail	V. Tail
10 GROSS AREA (SQ. FT.)					110.00	20.0	15.31
11 WEIGHT/GROSS AREA (LBS./SQ. FT.)					5.00		
12 SPAN (FT.)					22.0	6.67	
13 FOLDED SPAN (FT.)							
14							
15 SWEEPBACK - AT 25% CHORD LINE (DEGREES)					0		
16 - AT % CHORD LINE (DEGREES)					0		
**17 THEORETICAL ROOT CHORD - LENGTH (INCHES)					60		
18 - MAX. THICKNESS (INCHES)					9		
***19 CHORD AT PLANFORM BREAK - LENGTH (INCHES)					-		
20 - MAX. THICKNESS (INCHES)					-		
***21 THEORETICAL TIP CHORD - LENGTH (INCHES)					60		
22 - MAX. THICKNESS (INCHES)					9		
23 DORSAL AREA, INCLUDED IN (FUSE.) (HULL) (V. TAIL) AREA (SQ. FT.)							
24 TAIL LENGTH - 25% MAC WING TO 25% MAC H. TAIL (FT.)						14.23	
25 AREAS (SQ. FT.) Flaps	L.E.		T.E. 10.00				
26 Lateral Controls	Slots		Spoilers		Ailerons 12.00		
27 Speed Brakes	Wing		Fuse. or Hull				
28							
29							
30 ALIGHTING GEAR UNICYCLE (LOCATION)							
31 LENGTH - OLEO EXTENDED - ℄ AXLE TO ℄ TRUNNION (INCHES)							
32 OLEO TRAVEL - FULL EXTENDED TO FULL COLLAPSED (INCHES)							
33 FLOAT OR SKI STRUT LENGTH (INCHES)							
34 ARRESTING HOOK LENGTH - ℄ HOOK TRUNNION TO ℄ HOOK POINT (INCHES)							
35 HYDRAULIC SYSTEM CAPACITY (GALS.)							
36 FUEL & LUBE SYSTEMS	Location	No. Tanks	****Gals. Protected		No. Tanks	****Gals. Unprotected	
37 Fuel - Internal	Wing						
38	Fuse. or Hull				1	20	
39 - External							
40 - Bomb Bay							
41							
42 Oil							
43							
44							
45 STRUCTURAL DATA - CONDITION			Fuel in Wings (Lbs.)		Stress Gross Weight	Ult. L.F.	
46 FLIGHT			0		550	4.375	
47 LANDING							
48							
49 MAX. GROSS WEIGHT WITH ZERO WING FUEL							
50 CATAPULTING							
51 MIN. FLYING WEIGHT							
52 LIMIT AIRPLANE LANDING SINKING SPEED (FT./SEC.)							
53 WING LIFT ASSUMED FOR LANDING DESIGN CONDITION (%W)							
54 STALL SPEED - LANDING CONFIGURATION - POWER OFF (KNOTS)							
55 PRESSURIZED CABIN - ULT. DESIGN PRESSURE DIFFERENTIAL - FLIGHT (P.S.I.)							
56							
57 AIRFRAME WEIGHT (AS DEFINED IN AN-W-11) (LBS.)							

*Lbs. of sea water @ 64 lbs./cu. ft.
**Parallel to ℄ at ℄ airplane.
***Parallel to ℄ airplane.
****Total usable capacity.

PREPARED
CHECKED
DATE December 19
REV DATE

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TABLE IV

WEIGHT OF INERTIA CALCULATIONS

ITEM	LBS.	IN.			IN.-LBS.			
	W	X	Y	Z	WX	WY	WZ	
Wing Group	(46.9)	(87.8)	0	(31.3)	(4120)	0	(2569)	(
Envelope	23.5	88.5	0	36.2	2053	0	1133	
Ailerons (2)	3.5	108.8	±96.0	50.7	381	0	177	
Flap	2.4	100.8	0	50.7	241	0	122	
Brace Wire	2.5	94.4	0	50.0	236	0	123	
Hinge	1.5	103.0	±96.0	51.8	158	0	78	
Main Brace Patch	3.5	82.0	±60.0	54.0	287	0	189	
Strap & Patches	2.3	105.0	±50.0	50.0	242	0	115	
Bungee Cord	.5	97.0	±96.0	56.3	49	0	28	
Paint	1.5	82.0	0	56.2	123	0	84	
Inflation Air 7 psi	3.7	89.2	0	53.5	330	0	198	
Tail Group	10.6	(233.1)	(0)	(66.9)	(4935)	(0)	(1243)	(
Stabilizer	3.0	228.6	0	51.7	686	0	155	
Elevator	2.5	245.5	0	51.7	614	0	129	
Hinge, Control Horn, Patches	1.7	239.4	0	51.7	407	0	88	
Pin	3.3	222.1	0	70.7	733	0	233	
Rudder Structure	1.5	248.0	0	76.2	372	0	114	
Rudder Balance Weight	4.5	233.8	±6.0	86.5	1052	0	389	
Hinge, Control Horn, Patches	1.0	239.4	0	72.0	239	0	72	
Brace Wires	.3	160.0	0	57.0	48	0	17	
Paint	.4	229.5	0	60.0	92	0	24	
Inflation Air	.4	229.5	0	60.0	92	0	24	
Body Group	(41.2)	(90.8)	0	(34.2)	(3741)	0	(1409)	(
Envelope	16.0	130.1	0	39.2	2082	0	627	
Paint & Inflation	5.8	130.1	0	39.2	755	0	227	
Cockpit (Incl. Air, Paint, Patches)	19.4	46.6	0	28.6	904	0	555	
Alighting Gear Group	(23.0)	(72.6)		(19.7)	(1669)	0	(454)	
Main Gear	15.3	63.5	0	17.7	972	0	271	
Tail Skid	.7	231.3	0	34.0	162	0	24	
Wing Tip Skids	.9	98.0	±125.0	42.0	88	0	38	
Reinforcement	6.1	73.3	0	19.8	447	0	121	
Surface Controls Group	2.7	(58.9)	0	(34.1)	(159)	0	(92)	
Control Stick	.7	24.0	0	32.0	17	0	22	
Cables	2.0	71.0	0	35.0	142	0	70	

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TABLE IV

MENT OF INERTIA CALCULATIONS

	In.-Lbs.			Lb.-In.^{2}/1000					
Z	WX	WY	WZ	WX^2	WY^2	WZ^2	I_x	I_y	I_z
(31.3)	(4120)	0	(2549)	(367.3)	(69.1)	(130.9)	(202.7)	(9.8)	(210.2)
36.2	2053	0	1433	165.3	0	80.5	162.0	5.8	167.4
30.7	301	0	177	41.5	32.3	9.9	1.5	.1	1.6
30.7	261	0	122	20.4	0	6.2	2.8		2.8
30.0	236	0	125	22.3	0	6.3	3.0	.5	3.0
31.8	159	0	78	16.6	13.8	4.0	--	.7	.7
31.0	207	0	189	23.5	12.6	10.2	.7	.7	.7
30.0	242	0	115	23.4	5.8	5.8	.4	.4	.4
36.3	49	0	28	4.8	4.6	1.6	.1	.1	.1
36.2	123	0	84	10.1	0	4.7	9.5	.4	9.8
33.5	330	0	193	29.4	0	10.6	22.7	1.1	23.7
(66.9)	(4335)	(0)	(1245)	(1013.0)	(.2)	(87.0)	(4.2)	(3.1)	(6.1)
51.7	606	0	153	156.8	0	8.0	1.6	.4	1.8
51.7	614	0	129	150.7	0	6.7	1.4	.3	1.5
51.7	407	0	88	97.4	0	4.5	.9	.1	1.1
70.7	733	0	233	162.8	0	16.5	.5	1.2	.4
76.2	372	0	114	92.3	0	8.7	.2	.5	.7
86.5	1052	0	309	245.9	.2	33.6	.1	.1	.1
72.0	239	0	72	57.2	0	5.2	.1	.4	.3
57.0	40	0	17	7.7	0	1.0			
60.0	92	0	24	21.1	0	1.4	.1	.1	.2
60.0	92	0	24	21.1	0	1.4			
(34.2)	(3741)	0	(1409)	(411.2)	0	(49.4)	(6.5)	(71.1)	(69.7)
39.2	2082	0	627	270.9	0	24.6	2.6	55.5	55.5
39.2	755	0	227	98.2	0	8.9			
28.6	904	0	555	42.1	0	15.9	3.9	15.6	14.2
(19.7)	(1669)	0	(454)	(140.5)	(14.1)	(9.6)	(2.5)	(2.4)	(2.5)
17.7	972	0	271	61.7	0	4.8	2.0	2.0	2.0
34.0	162	0	24	37.5	0	.8	-	-	-
42.0	88	0	38	8.6	14.1	1.6	-	-	-
19.8	447	0	121	32.7	0	2.4	.5	.4	.5
(34.1)	(159)	0	(92)	(10.5)	0	(3.2)	(2.7)	(8.8)	(8.8)
32.0	17	0	22	.4	0	.7	-	-	-
35.0	142	0	70	10.1	0	2.5	2.7	8.8	8.8

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TABLE IV (Continued)

MOMENT OF INERTIA CALCULATIONS

	LBS.	IN.			IN.-LBS.		
ITEM	W	X	Y	Z	WX	WY	
Engine Section	(9.5)	(107.3)	0	(64.0)	(1019)	0	(
Engine Mount	8.8	107.8	0	64.7	949	0	
Fuselage Patch	.7	100.0	0	56.0	70	0	
Propulsion Group	(68.3)	(107.3)	0	(79.0)	(7352)	0	(7
Magneto	6.9	119.5	0	81.5	825	0	
Engine	45.0	100.8	0	82.2	4983	0	3
Carburetor	2.2	107.0	0	87.5	235	0	
Spark Plugs	1.3	107.5	0	82.3	140	0	
Propeller	4.1	99.4	0	82.7	408	0	
Hub, Flange, Bolts	1.8	99.4	0	82.7	179	0	
Fuel Cell	3.1	74.5	0	30.2	231	0	
Hoses	1.5	103.5	0	58.0	155	0	
Shut-off Valve, Press. Reg., Fuel Coupling & Fttg.	1.0	106.0	0	63.0	106	0	
Engine Controls	.8	112.5	0	65.5	90	0	
Instruments	(7.6)	(23.8)	(-1.8)	(43.0)	(181)	(-14)	(
Air Pressure Cage	.4	19.0	-3.5	43.0	8	-1	
Airspeed Indicator	.7	19.0	+7.5	42.0	13	+5	
Pitot Tube	.3	0	+13.7	34.0	0	+4	
Compass	.5	19.0	+3.0	42.5	10	+2	
Tachometer	1.9	19.0	-8.8	42.0	36	-17	
Altimeter	1.5	19.0	+5.0	42.0	29	+8	
Cylinder Head Temp.	1.5	19.0	-6.0	42.0	29	-9	
Thermocouple	.8	70.0	-8.0	55.0	56	-6	
Pneumatic Group	(11.9)	(117.6)	0	(80.9)	(1400)	0	(
Compressor	5.4	122.6	0	85.8	662	0	
Compressor Valve	4.9	119.0	0	85.8	583	0	
Relief Valve	.5	107.6	0	50.5	54	0	
Check Valve	.3	107.6	0	50.5	32	0	
Hose & Fitting	.5	85.0	0	50.0	43	0	
Control Cables	.3	85.0	0	50.0	26	0	
Electrical Group	(2.0)	(32.0)	(+6.5)	(28.5)	(64)	(+13)	
Battery	1.4	30.0	+5.0	25.0	42	+7	
Cockpit Light	.6	37.0	+10.0	36.0	22	+6	
Total Weight Empty	231.9	103.67	.004	56.56	24040	-1	13

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TABLE IV (Continued)

OMENT OF INERTIA CALCULATIONS

	IN.-LBS.			LB.-IN.^{2}/1000					
Z	WX	WY	WZ	WX^2	WY^2	WZ^2	I_x	I_y	I_z
(64.0)	(1019)	0	(608)	(109.3)	0	(39.0)	(1.0)	(2.6)	(1.6)
64.7	949	0	569	102.3	0	36.8	1.0	2.6	1.6
56.0	70	0	39	7.0	0	2.2			
(79.0)	(7352)	0	(5411)	(793.9)	0	(436.5)	(3.9)	(3.4)	(5.2)
81.5	825	0	562	98.6	0	45.8			
82.2	4983	0	3765	542.2	0	309.5	1.7	2.1	3.6
87.5	235	0	193	25.1	0	16.9			
82.3	140	0	107	15.1	0	8.8			
82.7	408	0	339	40.6	0	28.0	1.8	.9	.9
82.7	179	0	149	17.8	0	12.3			
30.2	231	0	94	17.2	0	2.8			
58.0	155	0	87	16.0	0	5.0	.4	.4	.7
63.0	106	0	63	11.2	0	4.0			
65.5	90	0	52	10.1	0	3.4	-	-	-
(43.0)	(181)	(-14)	(327)	(6.4)	(0.3)	(14.1)	(..1)	(.1)	(.1)
43.0	8	-1	17	.2	-	.7			
42.0	13	+5	29	.2	-	1.2			
34.0	0	+4	10	0	.1	.3			
42.5	10	+2	21	.2	-	.9	.1	.1	.1
42.0	36	-17	80	.7	.1	3.4			
42.0	29	+8	63	.6	-	2.6			
42.0	29	-9	63	.6	.1	2.6			
55.0	56	-6	44	3.9	-	2.4			
(80.9)	(1400)	0	(963)	(165.7)	0	(79.9)	(1.2)	(1.7)	(1.7)
85.8	662	0	463	81.2	0	39.7	.4	.6	.6
85.8	583	0	420	69.4	0	36.0	.4	.5	.5
50.5	54	0	25	5.8	0	1.3	.1	.2	.2
50.5	32	0	15	3.4	0	.8	.1	.1	.1
50.0	43	0	25	3.7	0	1.3	.1	.2	.2
50.0	26	0	15	2.2	0	.8	.1	.1	.1
(28.5)	(64)	(+13)	(57)	(2.1)	(.1)	(1.7)	-	-	-
25.0	42	+7	35	1.3	-	.9	-	-	-
36.0	22	+6	22	.8	.1	.8	-	-	-
56.56	24040	-1	13115	3019.9	83.8	859.3	225.5	103.0	305.9

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TABLE IV (Continued)

MOMENT OF INERTIA CALCULATIONS

ITEM	LBS.	IN.			IN.-LBS.			
	W	X	Y	Z	WX	WY	WZ	W
					MOST AFT C.G.			
Weight Empty	231.9	103.67	-.004	56.55	24014.0	-1	13113	
Pilot	160.0	40.0	0	37.2	6400	0	5952	
Fuel	12.4	74.6	0	25.0	925	0	310	
Gross Weight	404.3	77.50	-.0025	47.93	31365	-1	19377	

Most Aft C.G. Location in Percent MAC = $\left(\frac{77.6 - 57.7}{60.0}\right) 100 = 33.2$

					MOST FWD C.G.			
Weight Empty	231.9	103.67	-.004	56.55	24014.0	-1	13113	
Pilot	240.0	40.00	0	37.20	9600	0	8928	
Fuel	78.1	74.60	0	28.20	5826	0	2202	
Gross Weight	550.0	71.76	-.002	44.00	39466	-1	24243	

Most FWD C.G. Location in Percent MAC = $\left(\frac{71.8 - 57.7}{60.0}\right) 100 = 23.5$

Length of MAC = 60.0 inches.
L.E. of MAC is located at Station 57.7.

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TABLE IV (Continued)

ENT OF INERTIA CALCULATIONS

	IN.-LBS.			LB.-IN.^{2}/1000					
Z	WX	WY	WZ	WX^2	WY^2	WZ^2	I_x	I_y	I_z
	MOST AFT C.G.								
56.55	24040	-1	13115	3019.9	83.9	859.3	225.5	103.0	305.9
37.2	6400	0	5952	256.0	--	221.4	16.9	23.2	14.2
25.0	925	0	310	69.0	--	7.8	1.7	.4	2.0
47.93	31365	-1	19377	3344.9	83.8	1088.5	244.1	128.6	322.1
				-2433.3	-0	-928.7	83.8	911.6	911.6
				911.6	83.9	159.8	159.8	159.9	83.8
				Lb. In.^{2}/1000			487.7	1200.0	1317.5
00 = 33.2				Slug Ft.^{2}			105.3	259.0	284.4
	MOST FWD C.G.								
56.55	24040	-1	13115	3019.9	83.8	859.3	225.5	103.0	305.9
37.20	9600	0	8928	384.0	0	332.1	30.7	42.9	24.4
28.20	5826	0	2202	434.6	0	62.1	10.9	3.1	3.1
44.08	39466	-1	24245	3838.5	83.8	1253.5	267.1	149.0	333.4
				-2832.1	0	-1068.7	83.8	1006.4	1006.4
				1006.4	83.8	184.8	184.8	184.8	83.8
				Lb. In.^{2}/1000			535.7	1340.2	1423.6
				Slug Ft.^{2}			115.6	289.3	307.3
00 = 23.5									

WING LOADS

The wing is of Airmat construction and is a single-piece structural unit forward of the aileron hinge line. External support for the wing is provided by brace cables connected between the upper wing surface and engine mount pylon. The lower wing surface brace cables are connected to the landing gear and fuselage. The wing is attached to the fuselage by fabric straps at the rear cockpit bulkhead and through the engine mount attachment cables at the trailing edge of the wing.

The sign convention used for the wing load calculations is shown on page 2.02.020.

The shears, moments, and torques for the critical wing symmetrical conditions A_2 and C_2 are found on page 2.02.170 through 2.02.260. Unsymmetrical conditions were investigated and found not to be critical.
No torque results from the chordwise force.

WING LOADS

Sign Convention

Loads Positive As Shown
Moments L.H. Rule

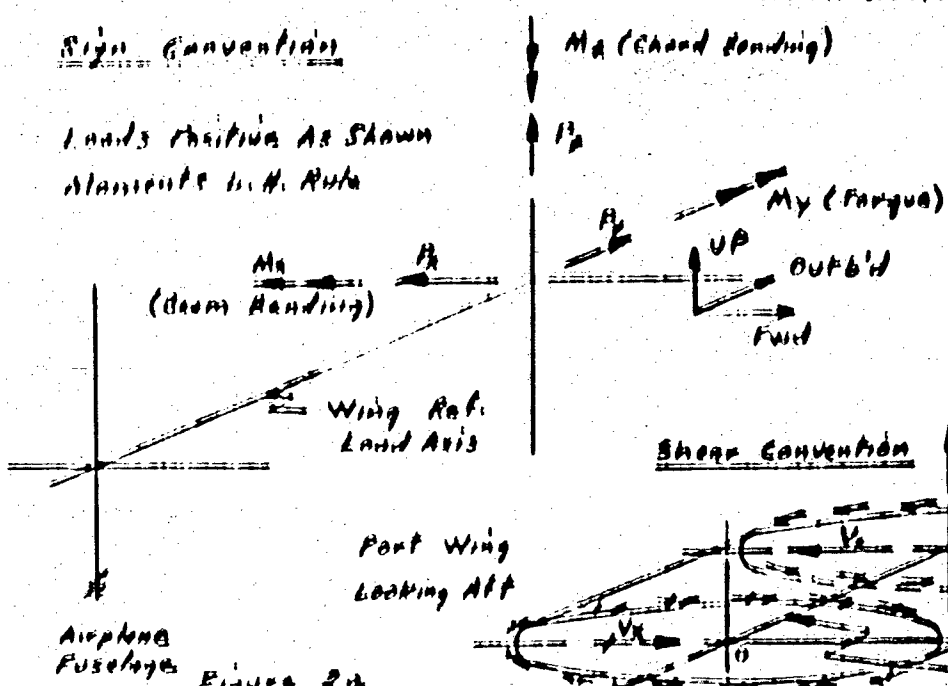

Figure 2a

Shear Convention

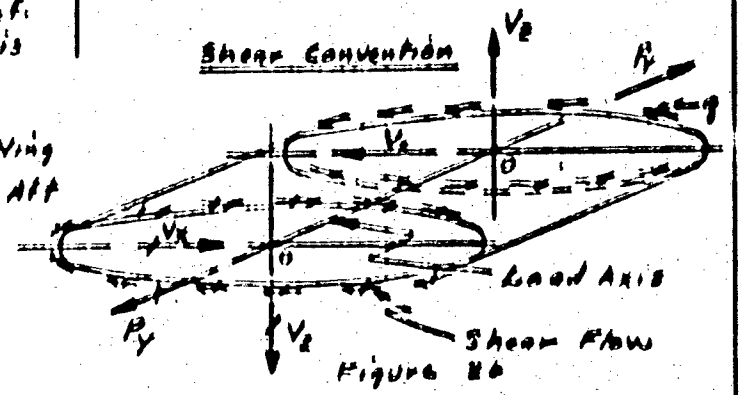

Figure 2b

P_y = Axial Tension
V_x = Chord Shear
V_z = Beam Shear

Outb'd Brace Wire Attachments
WING STA. 16.3

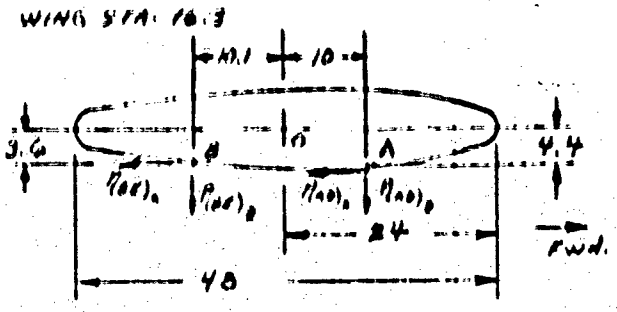

Figure 2c

Note:
Point O - Assumed Load Axis
Tension In Cables Positive

Inb'd Brace Wire Attachment
WING STA. 43.6

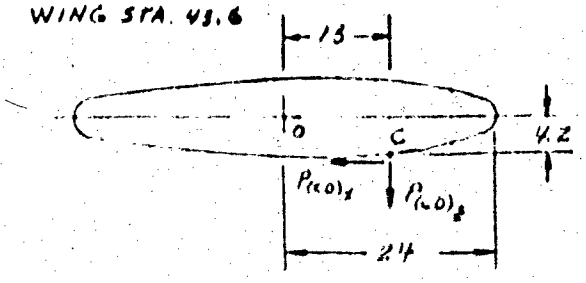

Figure 2d

WING LOADS

Calculation Of Shears, Moments And Torques From The Airloads

Using The Spanwise Lift Distribution Curve (C_{L_e} ~ Semi-Span) And The Spanwise Drag Distribution Curve ($C_{D_e}/b/2$ ~ Semi-Span) of Reference (4) The Normal Force Coefficient (C_N) And The Chord Force Coefficient (C_C) Are Calculated In Table V Using The Equations Listed Below:

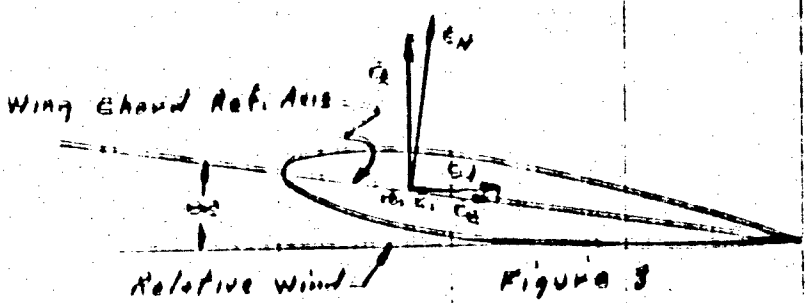

Figure 3

$$C_N = C_l \cos\alpha + C_d \sin\alpha$$
$$C_c = C_d \cos\alpha - C_l \sin\alpha$$

Where: C_l = Section Lift Coefficient
C_d = Section Drag Coefficient

The General Equation For Airfoil Lift Coefficient Is

$$C_L = \frac{L}{qS}$$

Where: L = Total Lift of The Airfoil - Lbs.
S = Airfoil Area - Ft^2
$q = \rho/2 V^2$ Dynamic Pressure - Lbs/Ft^2
ρ = .002378 Slugs / $Ft.^3$
V = Velocity In Ft/sec.

WING LOADS

Calculation of Shears, Moments, And Torques From The Airloads

For Section Lift Coefficient over The Elementary Area

$$c_l = \frac{dL}{q\,ds}$$

Where $ds = c\,dy = c\,\Delta b$

c = Airfoil Chord

Δb = An Elementary Spanwise Distance

Then

$$dL = c_l\, c\, q\, \Delta b$$

Similarly

$$dD = c_d\, c\, q\, \Delta b$$

Resolving These Into Normal And Chordwise Force Distribution Curves, dN And dC, And Using A Semi-graphical Method Of Intergration The Shears And Moments were Calculated. These Calculations Are Shown In Tables VI, VII, VIII, & IX.

The Torques About The Reference Axis Was Computed By Transferring Normal Force From The Quarter-Chord To The Reference Axis. The Calculation For The Torques Appears In Tables VI & VII.

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Calculation of Shears, Moments, And Torques From The Airloads

Table V

Col.	①	②	③	④	⑤	⑥	⑦	⑧	⑨
Item	Sta.	α Degrees	Cos α	Sin α	C_{l_c}	C_l	$C_{bt}/t_{1/2}$	C_d	$C_l \cos\alpha$
Ref.						⑤/5		⑦×11	③×⑥
Condition-A_2 HAA Symmetrical Maneuver	132	13.8	.97113	.23853	0	—	.0015	.0165	0
	120				3.70	0.740	.0056	.0616	0.717
	108				5.05	1.010	.0088	.0968	0.981
	96				5.90	1.180	.0112	.1232	1.146
	84				6.50	1.300	.0133	.1463	1.262
	76.3				6.79	1.358	.0144	.1584	1.319
	72				6.95	1.390	.0150	.1650	1.350
	60				7.25	1.450	.0163	.1793	1.408
	48				7.49	1.498	.0172	.1892	1.455
	43.6				7.54	1.508	.0175	.1925	1.464
	36				7.64	1.528	.0179	.1969	1.484
	24				7.76	1.552	.0184	.2024	1.507
	12				7.83	1.566	.0186	.2046	1.521
	0	13.8	.97113	.23853	7.85	1.570	.0188	.2068	1.525
Condition-C_2 LAA Symmetrical Maneuver	132	3.4	.99824	.05931	0	—	.0015	.0165	—
	120				1.95	0.390	.0025	.0275	0.389
	108				2.62	0.524	.0034	.0374	0.523
	96				3.05	0.610	.0041	.0451	0.609
	84				3.35	0.670	.0047	.0517	0.669
	76.3				3.50	0.700	.0050	.0550	0.699
	72				3.58	0.716	.0052	.0572	0.715
	60				3.72	0.744	.0055	.0605	0.743
	48				3.85	0.770	.00575	.0633	0.769
	43.6				3.90	0.780	.00585	.0644	0.779
	36				3.93	0.786	.0059	.0649	0.785
	24				3.98	0.796	.0060	.0660	0.795
	12				3.99	0.798	.0060	.0660	0.797
	0	3.4	.99824	.05931	4.00	0.800	.0060	.0660	0.799

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REP NO 207-3

WING LOADS

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Table IX

②	⑦	⑧	⑨	⑩	⑪	⑫	⑭	⑮
α_A	C_{DT}/C_{L^2}	C_d	$C_L \cos\alpha$	$C_L \sin\alpha$	C_N	$C_L \sin\alpha$	$C_d \cos\alpha$	C_a
①/5		⑦ × ⑪	③ × ⑥	④ × ⑧	⑨ + ⑩	④ × ②	③ × ⑤	⑬ − ⑫
—	.0015	.0165	0	.0039	0.0039	—	.0160	+.0160
190	.0056	.0616	0.717	.0147	0.7317	.1165	.3518	=.1167
210	.0088	.0968	0.981	.0231	1.0041	.2499	.3440	=.1467
230	.0112	.1232	1.146	.0294	1.1754	.2815	.1175	=.1817
250	.0133	.1463	1.262	.0349	1.2769	.3131	.1421	=.1630
258	.0144	.1584	1.319	.0378	1.3568	.3237	.1538	=.1701
270	.0150	.1650	1.350	.0394	1.3894	.3315	.1602	=.1714
290	.0163	.1793	1.408	.0428	1.4508	.3457	.1741	=.1713
298	.0172	.1892	1.455	.0451	1.5001	.3573	.1837	=.1736
308	.0175	.1925	1.464	.0457	1.5017	.3577	.1367	=.1723
328	.0177	.1969	1.484	.0470	1.5310	.3645	.1412	=.1733
352	.0184	.2024	1.507	.0483	1.5553	.3702	.1756	=.1786
366	.0188	.2046	1.521	.0488	1.5698	.3735	.1737	=.1749
370	.0188	.2068	1.525	.0493	1.5743	.3745	.2003	=.1737

—	.0015	.0165	—	.0010	.0010	—	.0165	.0165
590	.0025	.0276	0.389	.0016	.3906	.0231	.0275	.0044
624	.0034	.0374	0.523	.0022	.5252	.0311	.0373	.0062
610	.0041	.0451	0.609	.0027	.6117	.0362	.0450	.0088
670	.0047	.0517	0.669	.0031	.6721	.0397	.0516	.0119
700	.0050	.0550	0.699	.0033	.7023	.0415	.0549	.0134
.716	.0052	.0572	0.715	.0034	.7184	.0425	.0571	.0146
.744	.0055	.0605	0.743	.0036	.7466	.0441	.0604	.0163
.770	.00575	.0633	0.769	.0038	.7728	.0457	.0632	.0175
.780	.00585	.0644	0.779	.0038	.7828	.0463	.0643	.0180
.786	.0059	.0649	0.785	.0038	.7890	.0466	.0648	.0182
.796	.0060	.0660	0.795	.0039	.7989	.0472	.0659	.0187
.798	.0060	.0660	0.797	.0039	.8009	.0473	.0659	.0186
.800	.0060	.0660	0.799	.0039	.8029	.0474	.0659	.0185

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Calculation of Shears, Moments, and Torques For Normal From the Airloads

Table VI

Col.	(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	
Item	Sta	C_N	$C_N \times c$	Sum	Δb Ft	$\Delta(C_N c \Delta b)$	$C_N c \Delta b$	Shear	S
Ref.	In		(1) × 5	(3) + (3)		½ (4) (5)	Σ (6)	9.65 (7)	(8)
Condition - A_2 HAA Symmetrical Maneuver $q = 9.65$ #/Ft²	132	≅ 0	0				0	0	
				3.669	1.0	1.835			1
	120	.7337	3.669				1.835	17.71	
				8.690	1.0	4.345			9
	108	1.0041	5.021				6.180	59.64	
				10.795	1.0	5.449			17
	96	1.1754	5.877				11.639	112.22	
				12.362	1.0	6.181			2
	84	1.2969	6.485				17.810	171.87	
				13.269	.64	4.246			3
	76.3	1.3568	6.784				22.056	212.84	
				13.731	.36	2.472			4
	72	1.3894	6.947				24.528	236.70	
				14.201	1.0	7.101			54
	60	1.4508	7.254				31.629	305.22	
				14.755	1.0	7.378			6
	48	1.5001	7.501				39.007	376.42	
				15.051	.36	2.709			77
	43.6	1.5099	7.550				41.716	402.56	
				15.205	.64	4.866			85
	36	1.5310	7.655				46.582	449.52	
				15.432	1.0	7.716			97
	24	1.5553	7.777				54.298	523.98	
				15.626	1.0	7.813			112
	12	1.5698	7.849				62.111	599.37	
				15.721	1.0	7.861			11
	0	1.5743	7.872				69.972	675.03	

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PAGE 8.08.260
MODEL GA-448
SER- 4961
REP NO 5070-1

and Torques For Normal Forces — Wing Loads

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Table VI

(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)
Δb Ft.	Δ(CnCdb)	CnCdb	Shear	Sum	ΔM	Moment Ft-Lb	Moment In-Lb	Torque In-Lb
		Σ (4)(5)	9.65 (7)	(8)$_i$ + (8)$_{i+1}$	½ (5)(9)	Σ (10)	12 (11)	12 ½ (12)
		0	0			0	0	0
1.0	1.835			17.71	8.86			
		1.835	17.71			8.86	106.3	159.4
1.0	4.343			77.35	38.68			
		6.180	59.64			47.54	570.5	536.8
1.0	5.449			171.96	85.98			
		11.629	113.32			133.47	1601.6	1010.0
1.0	6.181			284.09	142.05			
		17.810	171.37			275.52	3306.2	1546.8
.64	4.246			384.71	123.11			
		23.056	222.54			398.63	4783.6	1915.6
.36	2.472			447.54	80.92			
		24.928	240.70			479.55	5754.6	2130.8
1.0	7.101			511.96	270.96			
		31.629	305.22			750.51	9006.1	2747.0
1.0	7.378			641.64	340.22			
		39.007	376.12			1091.13	13096.1	3387.8
.36	2.704			778.98	140.22			
		41.716	402.56			1231.55	14778.6	3623.0
.64	4.866			832.03	272.67			
		46.582	449.52			1504.22	18050.6	4045.7
1.0	7.716			971.50	486.75			
		54.298	523.915			1990.97	23891.6	4715.8
1.0	7.813			1173.35	586.68			
		62.111	599.37			2577.65	30931.8	5394.3
1.0	7.861			1374.50	687.30			
		69.972	675.23			3264.95	39179.4	6077.1

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Calculations of Shears and Moments For Drag Forces From the Airloads

Table VII

Col.	(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)
Item	Sta.	C_e	C_t+C	Sum	Δb Ft	$\Delta(C_t+C\Delta A)$	$C_t C \Delta A$	Shear
Ref.			(2)×5	(3)$_n$+(3)$_{n+1}$		½(5)(4)	½(8)	9.65(7) (9)
	132	.0169	.0331				0	0
				= .5035	1.0	= .2517		=
	120	= .1167	= .5335				= .2517	= 2.429
				= 1.318	1.0	= .659		
	108	= .1464	= .7345				= .4107	= 3.788
				= 1.544	1.0	= .772		=
	96	= .1619	= .8095				= 1.6327	= 16.938
				= 1.6495	1.0	= .8247		=
	84	= .1683	= .8430				= 2.5073	= 24.117
				= 1.6935	.64	= .5410		=
	76.3	= .1701	= .8505				= 3.0435	= 29.418
				= 1.7375	.36	= .3014		
	72	= .1714	= .8570				= 3.3557	= 32.334
				= 1.716	1.0	= .8580		=
	60	= .1718	= .8590				= 4.2137	= 40.664
				= 1.727	1.0	= .8635		=
	48	= .1736	= .8680				= 5.0774	= 48.117
				= 1.712	.36	= .3118		=
	6	= .1728	= .8640				= 5.3892	= 52.006
				= 1.7305	.64	= .5533		=
	36	= .1733	= .8665				= 5.9430	= 57.350
				= 1.7345	1.0	= .8672		=
	24	= .1736	= .8680				= 6.8102	= 65.718
				= 1.742	1.0	= .8710		=
	12	= .1748	= .8740				= 7.6812	= 74.124
				= 1.7425	1.0	= .8712		=
	0	= .1737	= .8685				= 8.5524	= 82.531

Condition A_2 HAF Symmetrical Maneuver
$q_0 = 9.65\ \#/ft^2$

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PAGE 2-52.078
MODEL ZA 764
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REP NO 5-17-1

Wing Loads

Humanly For Drag Forces.

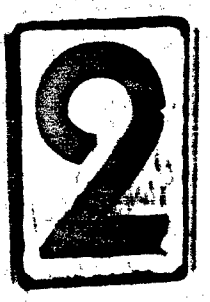

Table VII

(6) $\frac{1}{2}(5)(4)$	(7) Σ(6)	(8) Shear	(9) Sum $\Sigma(3)_{n+1}(8)_{n+1}+\frac{1}{2}(5)(7)$	(10) ΔM	(11) Moment Ft-Lb Σ(9)	(12) Normal In-Lb $12\times(11)$
	0	0			0	0
= .2511			= 0.437 = 1.814			
	= .2511	= 2.414			= 1.314	= 14.57
= .669			= 11.217 = 5.606			
	= .9180	= 3.716			= 5.922 = 71.13	
= .713			= 25.553 = 13.513			
	= 1.6327	= 14.936			= 19.335 = 232.15	
= .8218			= 43.415 = 29.311			
	= 2.5315	= 24.117			= 39.553 = 474.64	
= .5416			= 53.615 = 19.57			
	= 3.0435	= 29.413			= 56.719 = 693.63	
= .3314			= 61.603 = 11.124			
	= 3.3547	= 34.537			= 67.443 = 814.11	
= .2540			= 73.016 = 35.620			
	= 4.2137	= 40.644			= 104.367 = 1252.41	
= .2635			= 89.161 = 41.923			
	= 5.0774	= 48.117			= 151.417 = 1817.01	
= .3118			= 101.093 = 18.181			
	= 5.9892	= 57.096			= 172.372 = 2068.5	
= .5533			= 109.256 = 31.914			
	= 5.9430	= 67.350			= 207.373 = 2488.5	
= .5672			= 123.067 = 61.534			
	= 6.8102	= 65.718			= 268.906 = 3226.9	
= .3710			= 139.342 = 67.921			
	= 7.6817	= 74.124			= 338.317 = 4059.9	
= .8712			= 156.635 = 78.327			
	= 8.5529	= 82.581			= 417.154 = 5005.8	

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Calculation of Shears, Moments and Torques From the Airloads

Table VIII

Col.	(1)	(2)	(3)	(4)	(5)	(6)	(7)	
Item	Sta.	C_N	$C_N C$	Sum	$\frac{\Delta b}{FS}$	$\Delta(\Sigma C_N C \Delta b)$	$C_N C \Delta b$	
Ref	In		(2)x5	$(3)_n + (3)_{n-1}$		$\frac{1}{2}$ (4)(5)	Σ (6)	1
	132	≅0	0				0	
				1.953	1.0	.977		
	120	.9986	1.953				.977	
				4.579	1.0	2.289		
	108	.8351	2.626				3.267	
				5.685	1.0	2.843		
	96	.6117	3.059				6.110	1
				6.420	1.0	3.210		
	84	.6721	3.361				9.320	1
				6.873	.64	2.199		
	76.3	.7033	3.513				11.519	2
				7.104	.36	1.278		
	72	.7184	3.592				12.798	2
				7.325	1.0	3.663		
	60	.7466	3.733				16.461	3
				7.917	1.0	3.799		
	48	.7721	3.864				20.260	3
				7.778	.36	1.400		
	43.6	.7128	3.914				21.66	4
				7.854	.64	2.513		
	36.	.7125	3.940				24.173	4
				7.935	1.0	3.968		
	24	.7989	3.995				28.141	5
				8.000	1.0	4.000		
	12	.8009	4.005				32.141	6
				8.020	1.0	4.010		
	0	.8029	4.015				36.151	6

Condition - C_L LAA Symmetrical Maneuver
$q = 18.85$ #/in²

ments, and Torques for Normal Forces — Wing Loads

Table VIII

(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)
Δb Ft.	Δ(C_N Cdb)	C_N Cdb	Shear	Sum	ΔM	Moment Ft-Lb	Moment In-Lb	Torque In-Lb
	½ (4) (5)	Σ (6)	18.95 (7)	(8)n + (8)n+1	½ (5) (7)	Σ (10)	12 × (11)	12 × ¾ (8)
		0	0			0	0	0
1.0	.977			13.43	4.22			
		.977	18.43			4.22	110.6	165.9
1.0	2.893			80.01	40.01			
		3.267	61.52			49.23	590.8	554.2
1.0	2.843			176.75	88.38			
		6.110	115.17			137.61	1651.3	1036.5
1.0	3.210			290.35	145.43			
		9.320	175.68			283.04	3396.5	1581.1
.64	2.199			392.81	125.70			
		11.519	217.13			408.74	4904.9	1954.2
.36	1.279			458.37	82.51			
		12.798	241.24			441.25	5895.0	2171.2
1.0	3.663			551.58	275.77			
		16.461	310.29			757.02	8042.0	2792.6
1.0	3.799			692.19	346.10			
		20.260	381.90			1113.12	13357.4	3437.1
.36	1.400			790.19	142.23			
		21.66	408.29			1255.35	15064.2	3674.6
.64	2.513			863.95	276.46			
		24.173	455.66			1531.81	18381.7	4100.9
1.0	3.968			986.12	493.06			
		28.141	530.46			2024.87	24298.4	4774.1
1.0	4.000			1136.32	568.16			
		32.141	605.86			2593.03	31116.4	5452.7
1.0	4.010			1287.31	643.66			
		36.151	681.45			3236.70	38840.4	6133.1

2

32

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Calculations of Shears and Moments For Drag F
From the Airloads

Table II

Col.	(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)
Item	Sta.	C_c	$C_c C$	Sum	Δb Ft	$\Delta(C_c C \Delta b)$	$C_c C \Delta b$	Shear
Ref.			(2) x 5	$(3)_n + (3)_{n+1}$		½(4)(5)	Σ(6)	10.95(7)
	132	.0165	.0825					0
				.1045	1.0	.05224		
	120	.0044	.0220				.05224	.984
				.053	1.0	.0265		
	108	.0062	.031				.0788	1.489
				.075	1.0	.0375		
	96	.0088	.044				.1162	2.190
				.1035	1.0	.05175		
	84	.0119	.0595				.1680	3.167
				.1265	.64	.04048		
	76.3	.0134	.067				.2085	3.930
				.140	.36	.0252		
	72	.0146	.073				.2337	4.405
				.1545	1.0	.07725		
	60	.0163	.0815				.3109	5.860
				.1690	1.0	.0845		
	48	.0175	.0875				.3954	7.453
				.1775	.36	.03195		
	43.6	.0180	.090				.4274	8.056
				.181	.64	.05792		
	36	.0182	.091				.4853	9.148
				.1845	1.0	.09225		
	24	.0187	.0935				.5775	10.88
				.1865	1.0	.09325		
	12	.0186	.093				.6707	12.64
				.1855	1.0	.09275		
	0	.0185	.0925				.7635	14.39

Condition C_2 LAA Symmetrical Maneuver

...d Moments For Drag Forces

Wing Loads

2

Table II

(5) Δb Ft	(6) Δ(C_e C Δb) ½(4)(5)	(7) C_e C Δb Σ(6)	(8) Shear 18.85(7) 0	(9) Sum (8)_n + (8)_n+1	(10) ΔM ½(5)(9)	(11) Moment Ft-Lb Σ(10) 0	(12) Moment In-Lb 12×(11) 0
1.0	.05224			.9849	.4924		
		.05224	.9849			.492	5.90
1.0	.0265			2.470	1.235		
		.0788	1.485			1.727	20.72
1.0	.0375			3.675	1.837		
		.1162	2.190			3.564	42.77
1.0	.05175			5.357	2.678		
		.1680	3.167			6.242	74.90
.64	.04048			7.097	2.271		
		.2085	3.930			8.513	102.16
.36	.0252			8.335	1.500		
		.2337	4.405			10.013	120.16
1.0	.07725			10.365	5.182		
		.3109	5.860			15.145	181.74
1.0	.0845			13.313	6.656		
		.3954	7.453			21.801	261.61
.36	.03195			15.509	2.792		
		.4274	8.056			24.593	295.12
.64	.05792			17.204	5.505		
		.4853	9.148			30.098	361.18
1.0	.09225			20.034	10.017		
		.5775	10.886			40.115	481.38
1.0	.09325			23.527	11.763		
		.6707	12.643			51.878	622.54
1.0	.09275			27.035	13.517		
		.7635	14.392			65.395	784.74

WING LOADS

Calculation of Shears, Moments and Torques from the Inertia Loads

The condition solution for $n_z = 2.5$ was obtained using the wing unit solution, $n_z = 1$, shears and moments and the appropriate load factor. Since the angle of wing incidence is small, resolution of inertia loads in the chordwise direction is small and is conservatively neglected. The centroid of the inertia load was assumed to be located at reference axis. Therefore, no torque results from inertia loads.

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WING LOADS

Calculation Sheets and Nomograph
From FAA Inertia Loads

Limit Loads
Table X

Assuming Uniform Distribution
of Inertia Weight $w = \frac{45.8}{264} = .1775$ #/in

$n_z = 2.5$

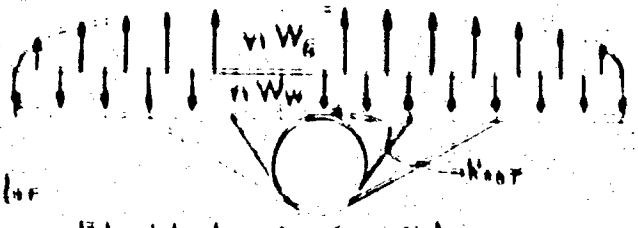
Cond.	Item	Running Load	Distance From Tip		Unit Solution		Shear Lbs	Moment In-Lbs	Equiv Sheer Lbs	Equiv Moment In-Lbs
	Col.	(1)	(2)		(3)		(4)	(5)	(6)	(7)
	Ref.						$w \times$	$\frac{w x^2}{2}$	2.5 x (4)	2.5 x (5)
Sta.		w	x		x =					
132		.1775	0		0		0	0	0	0
120			12		.122		-2.13	-13	-5.3	-32.5
108			24		.576		-4.26	-51	-10.7	-128
96			36		.1296		-6.39	-115	-16.0	-288
84			48		.2304		-8.52	-204	-21.3	-510
72.3			59.7		.3.052		-9.27	-275	-24.7	-685
72			60		.3600		-10.65	-319	-26.6	-798
60			72		.5184		-12.78	-460	-31.9	-1150
48			84		.7056		-14.91	-626	-37.2	-1567
43.4			88.4		.7814		-15.70	-697	-39.2	-1742
36			96		.9216		-17.04	-818	-42.5	-2045
24			108		11,664		-19.17	-1035	-47.8	-2590
12			120		14,400		-21.30	-1280	-53.2	-3200
0		.1775	132		17,424		-23.43	-1546	-58.6	-3865

WING LOADS

Calculation of Shears, Moments, And Torques From The Brace Cable Loads

The cable loads under conditions of flight are determined from values measured under static test conditions. The figure below shows that the wing inertia load when compared to the airload is:

(a) opposite in direction but multiplied by n in flight
(b) in same direction and constant in a static test.



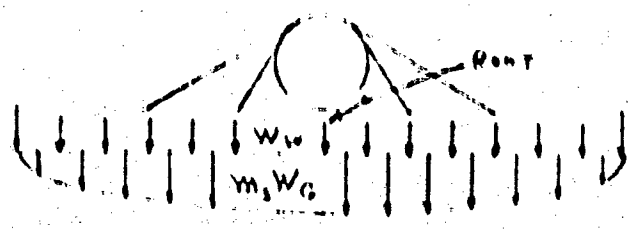
n = Flight Load Factor

(a) Flight Load Condition

W_G = Gross Wt. = 500 LB

W_W = Wing Wt. ≤ 50 LB

m_s = Static Test Load Factor



(b) Static Test Load Condition
Aircraft is Inverted

Figure Relation Between Airload and Wing Inertia Load For Flight and Static Test Conditions

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Wing Geometry

Lower Wing Brace Cable Coordinates in Wing Reference System

Points	X	Y	Z
A	14.0	76.3	4.4
B	34.1	76.3	3.6
C	11.0	43.6	4.2
D	19.0	8.0	40.0
E	50.8	4.0	25.9

Calculation of Direction Cosines For Cables

Member	X	Y	Z	L	$\frac{X}{L}$	$\frac{Y}{L}$	$\frac{Z}{L}$	Check
AD	5.0	68.3	35.6	77.18	.06478	.88494	.46125	1.0001
CD	8.0	35.6	35.6	51.12	.15650	.69643	.70455	.9999
BE	16.7	72.8	22.3	77.48	.21553	.93914	.28784	1.0000

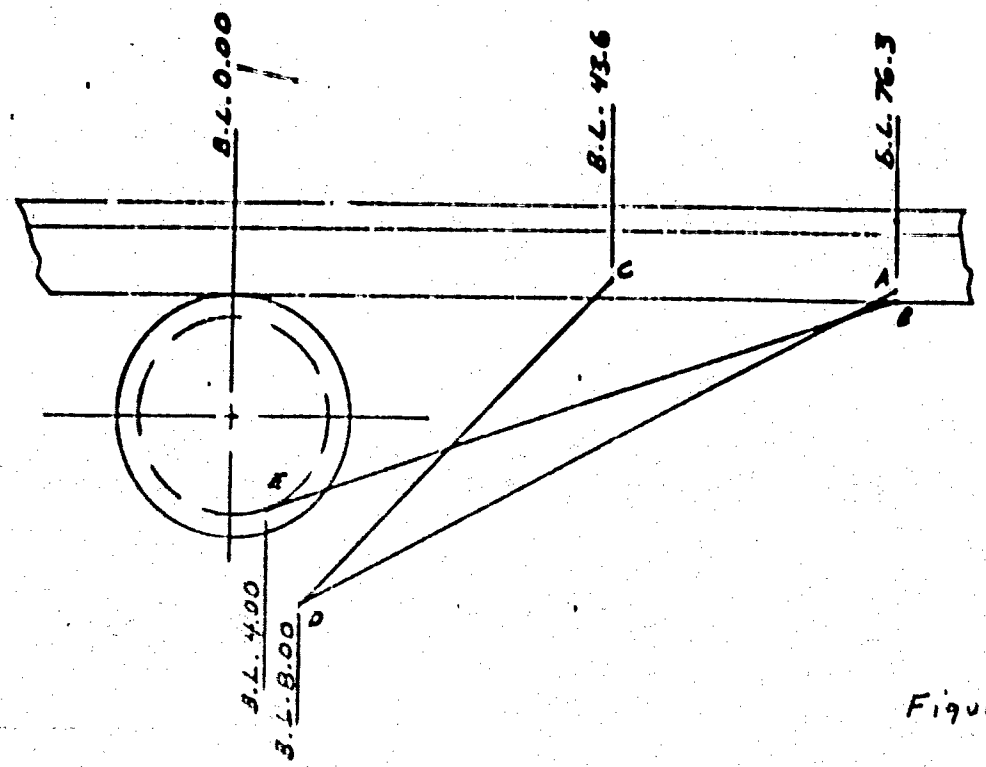


Figure 5

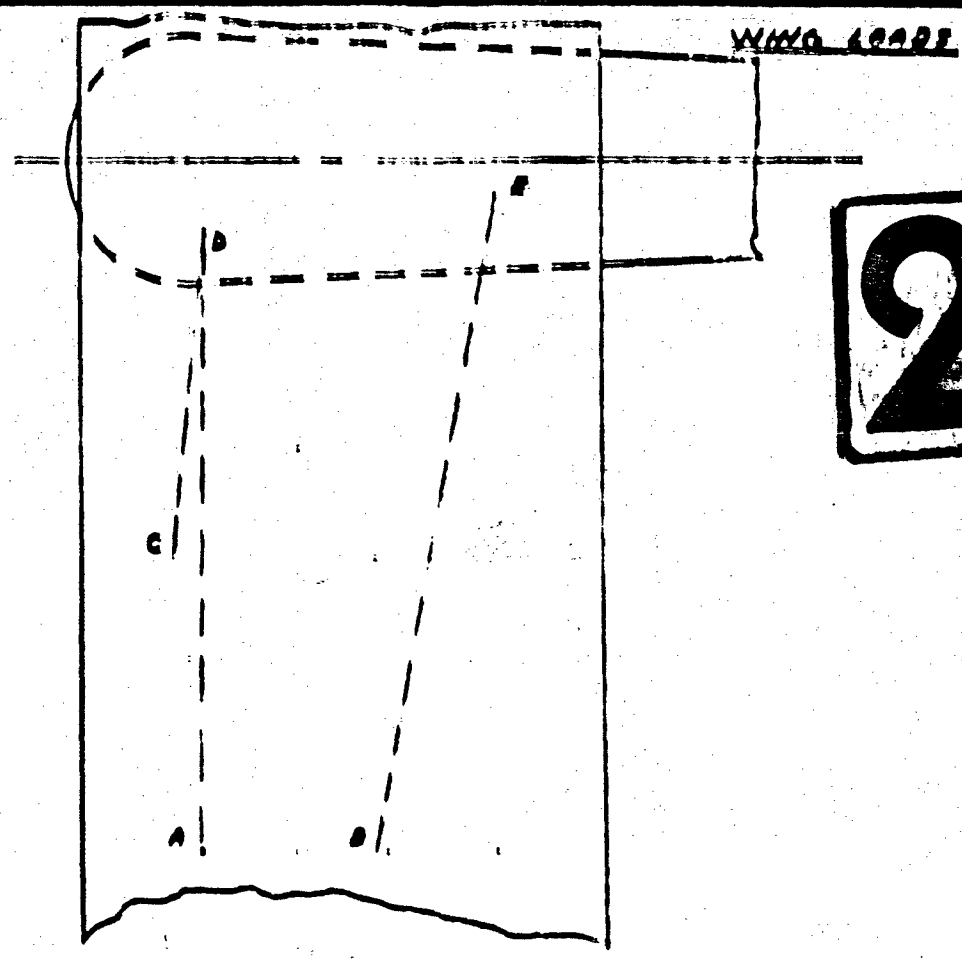
2

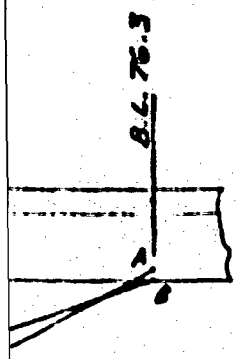
...tes

Z
4.4
3.6
4.2
40.0
25.9

For Cables

Y/L	Z/L	Check
.88444	.46125	1.0001
.69643	.70025	.9999
.93314	.28781	1.0000





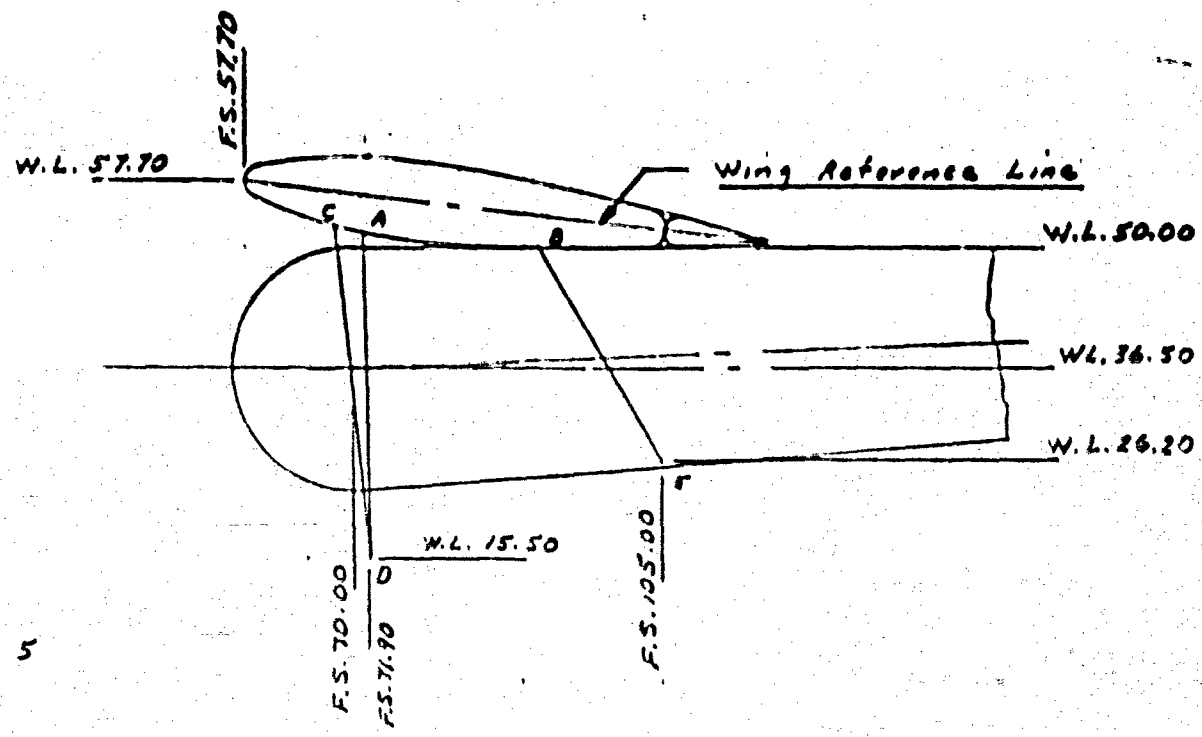


Figure 5

WING LOADS

The first table below gives the root moments due to airloads and wing inertia loads for various values of load factor. The next table gives cable loads measured under static test conditions for various load factors.

Table XI

①	②	③	④	⑤	⑥
Load Factor n or M_s	Root Moments In-Lb.				
	Airload Distribution (a)	Wing Weight (a)	Wing Weight (b)	Net Root Moment (a)	Net Root Moment (b)
Ref				② − ③	② + ④
1	15,500	1545	1545	13,955	17,045
2	31,000	3090	1545	27,910	32,545
2.5	38,800	3860	1545	34,940	40,345
3	46,500	4635	1545	41,815	48,045
4	62,000	6180	1545	55,820	63,545
5	77,500	7725	1545	69,775	79,045

Table XII

①	⑥	⑦	⑧	⑨
Load Factor M_s Ref.	Net Root Moment (b)	Inboard Cable Load CD Lb.	Forward Outboard Cable Load AD Lb.	Aft Outboard Cable Load BE Lb.
1	17,045	140	218	85
2	32,545	280	415	214
3	48,045	325	600	352
4	63,545	400	820	500

WING LOADS

Cable loads under flight conditions are determined from Figure 6 in which the load factors n and n_s are plotted vs. net root moment. On the same figure the static test cable loads are plotted vs static test net root moments. By assuming that the cable loads under flight conditions are dependent upon the net root moment it is possible to read flight cable loads from flight load factors n. The table below gives the cable loads corrected to flight load factors n.

Table XIII

Flight Load Factor n	Cable Loads, Lb		
	CD Inboard	AD Fwd Outbd.	BE Aft Outbd.
1	120	140	60
2	200	350	170
2.5	240	440	210
3	280	540	300
4	360	720	430
4.375	340	790	475

← Used as an example

Table XIV

Cable	Load Lb	Direction Cosines			Components, Lb.		
		x	y	z	x	y	z
CD	240	.1565	.69643	.70035	37.5	167.1	168
AD	440	.06478	.88494	.46125	28.5	389.4	203
BE	220	.21553	.92314	.28781	49.6	214.6	66

Component	Sum of Components	
	at Sta 43.6	at Sta 76.3
	CD	AD + BE
x	37.5	78.1
y	167.1	604.0
z	168	269

The cable loads, components, and sum of components at Stations 43.6 and 76.3 are given to left and above.

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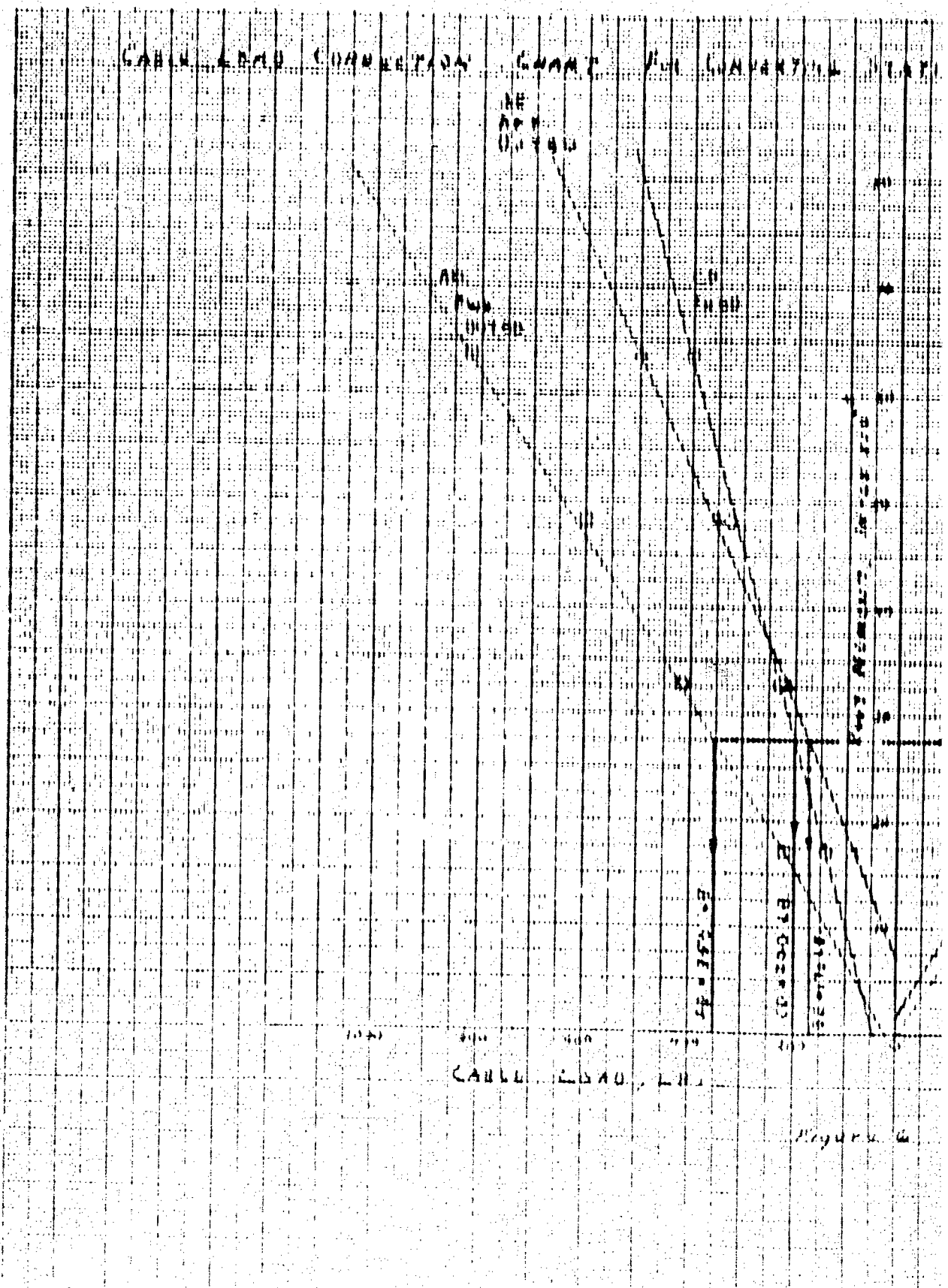

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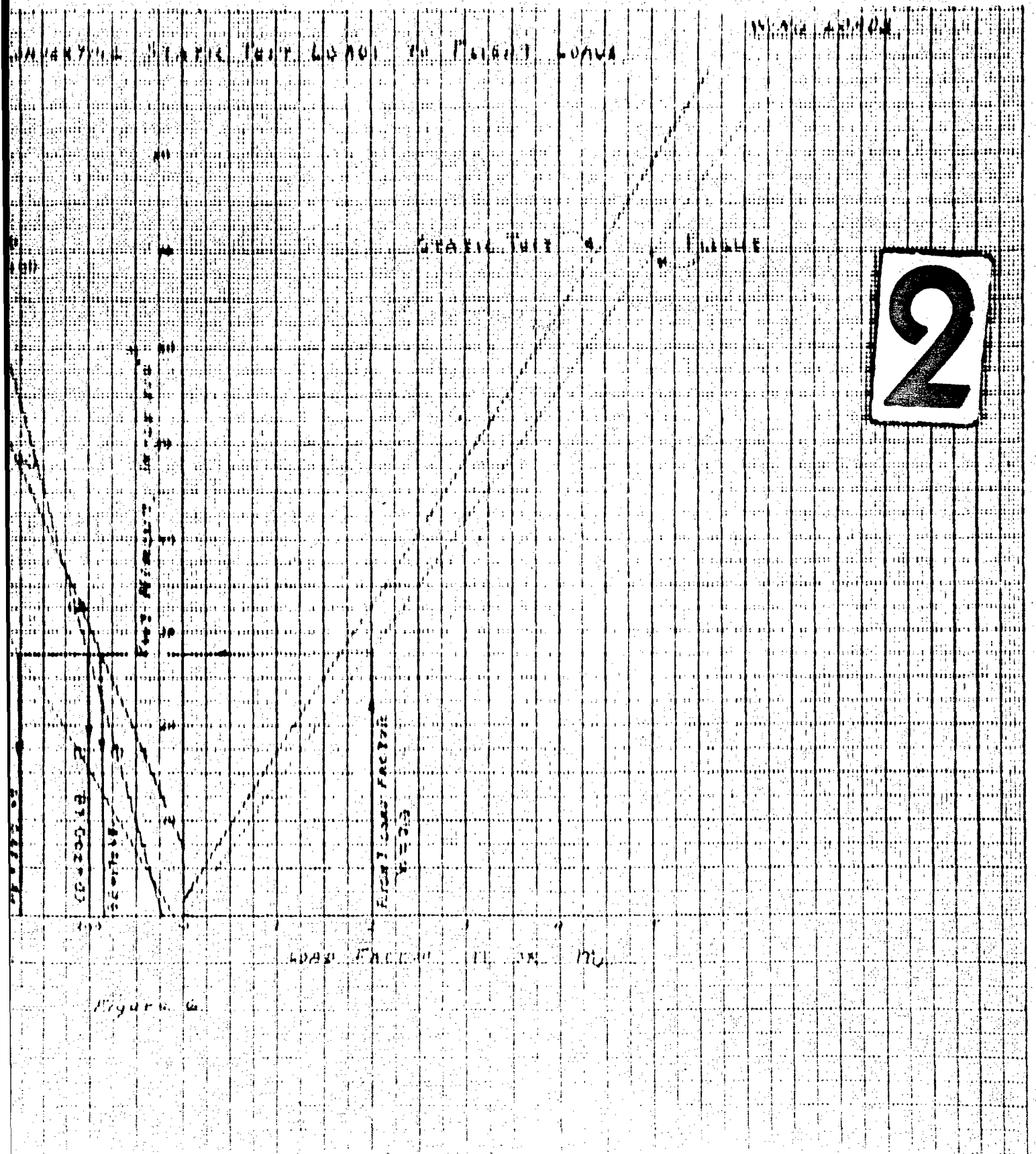

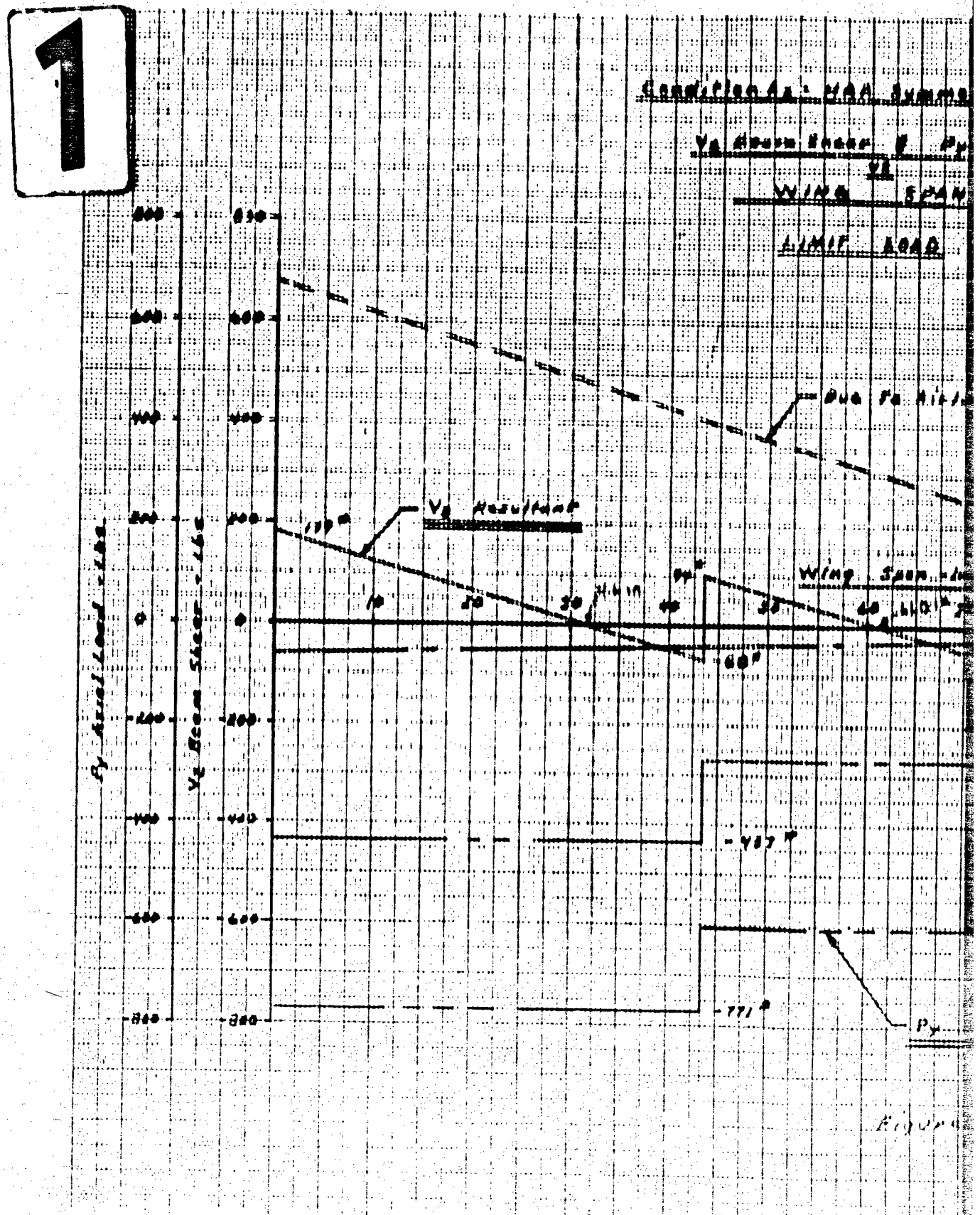

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Condition A2: NAA Symme
Vz Beam Shear & Py
VS
WING SPAN
LIMIT LOAD
Due To Airl
Vz Resultant
Wing Span
Py Axial Load - Lbs
Vz Beam Shear - Lbs
800
600
400
200
0
-200
-400
-600
-800
10
20
30
40
50
60
-437#
-771#
Py

Figure

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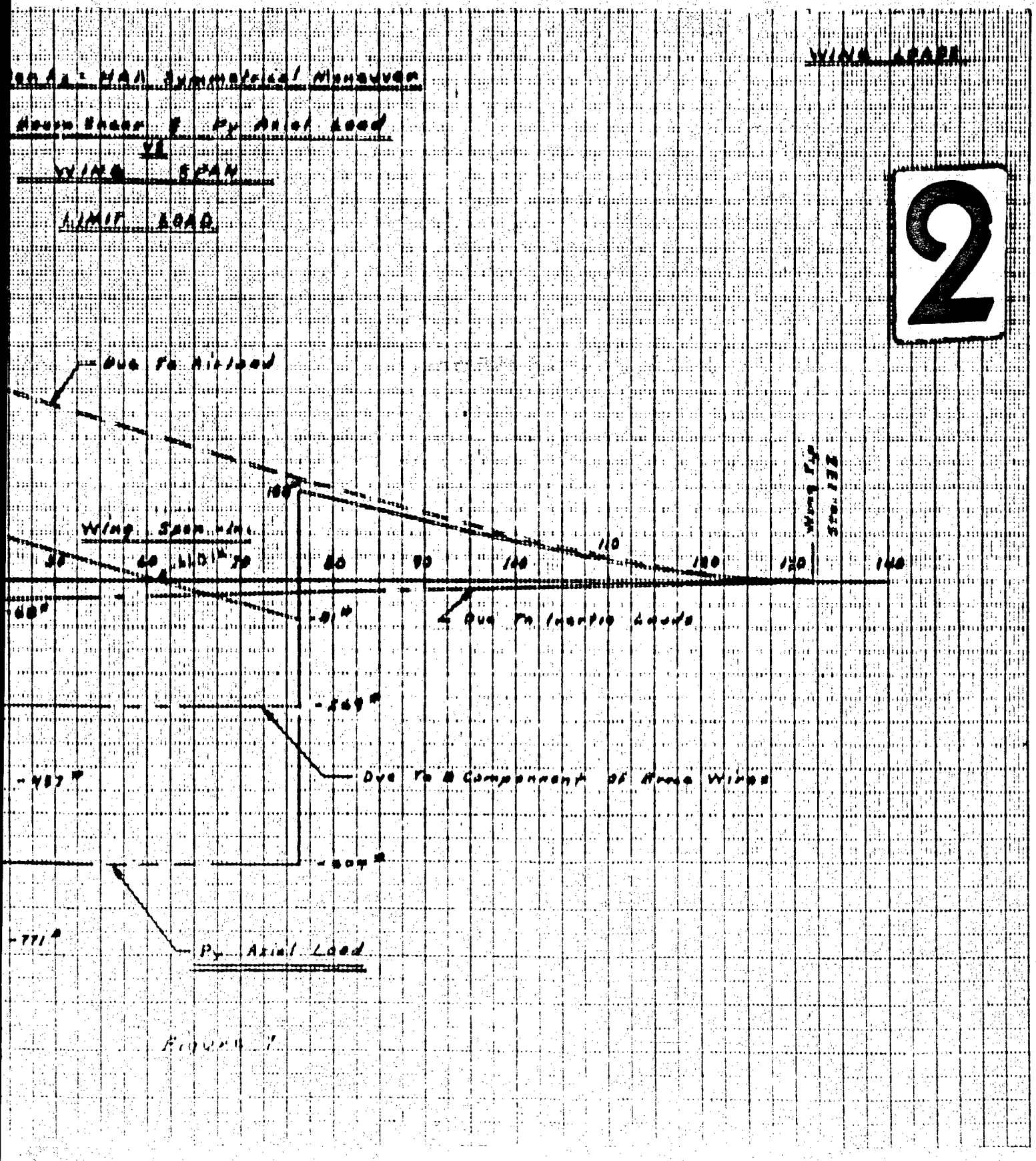

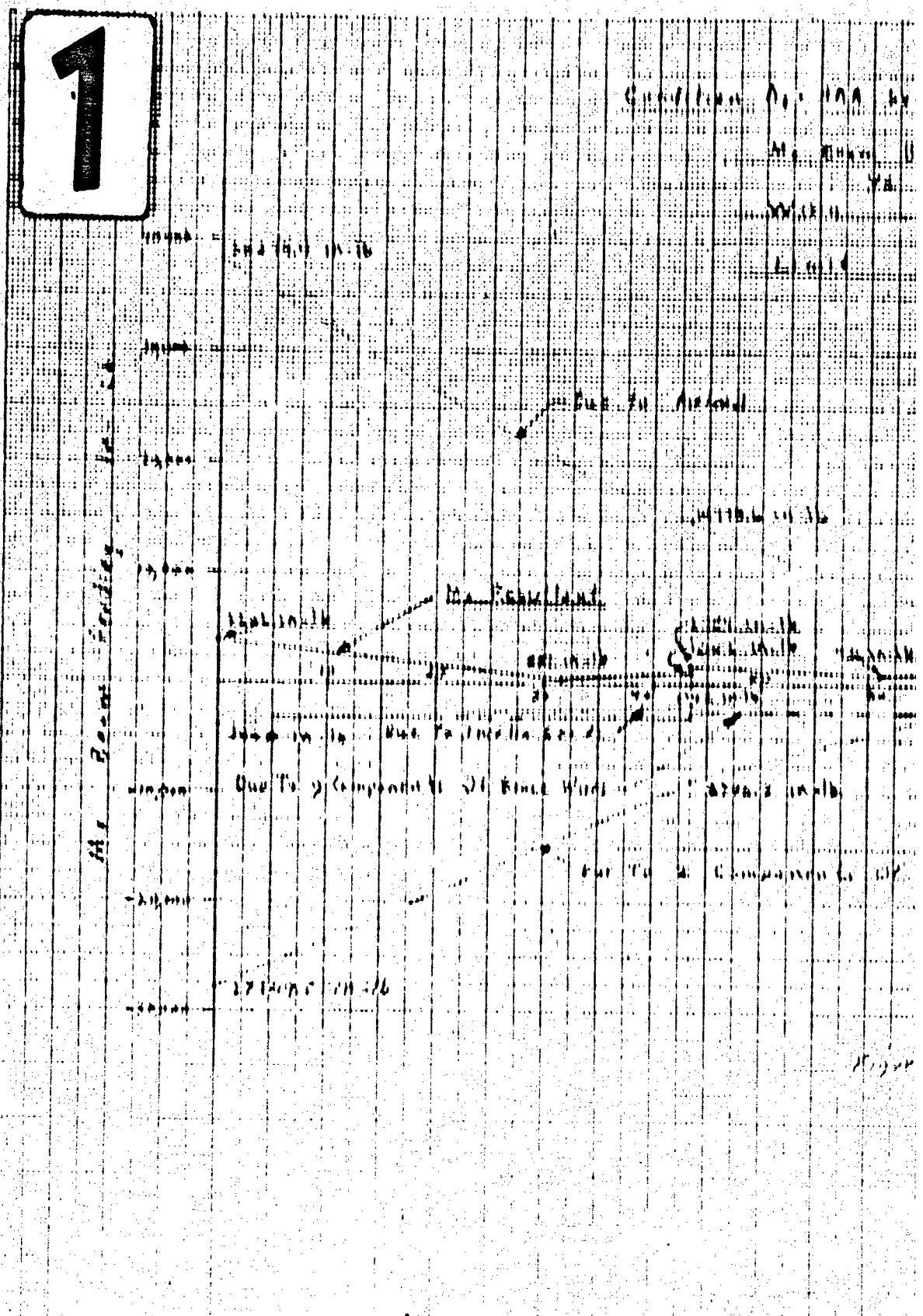

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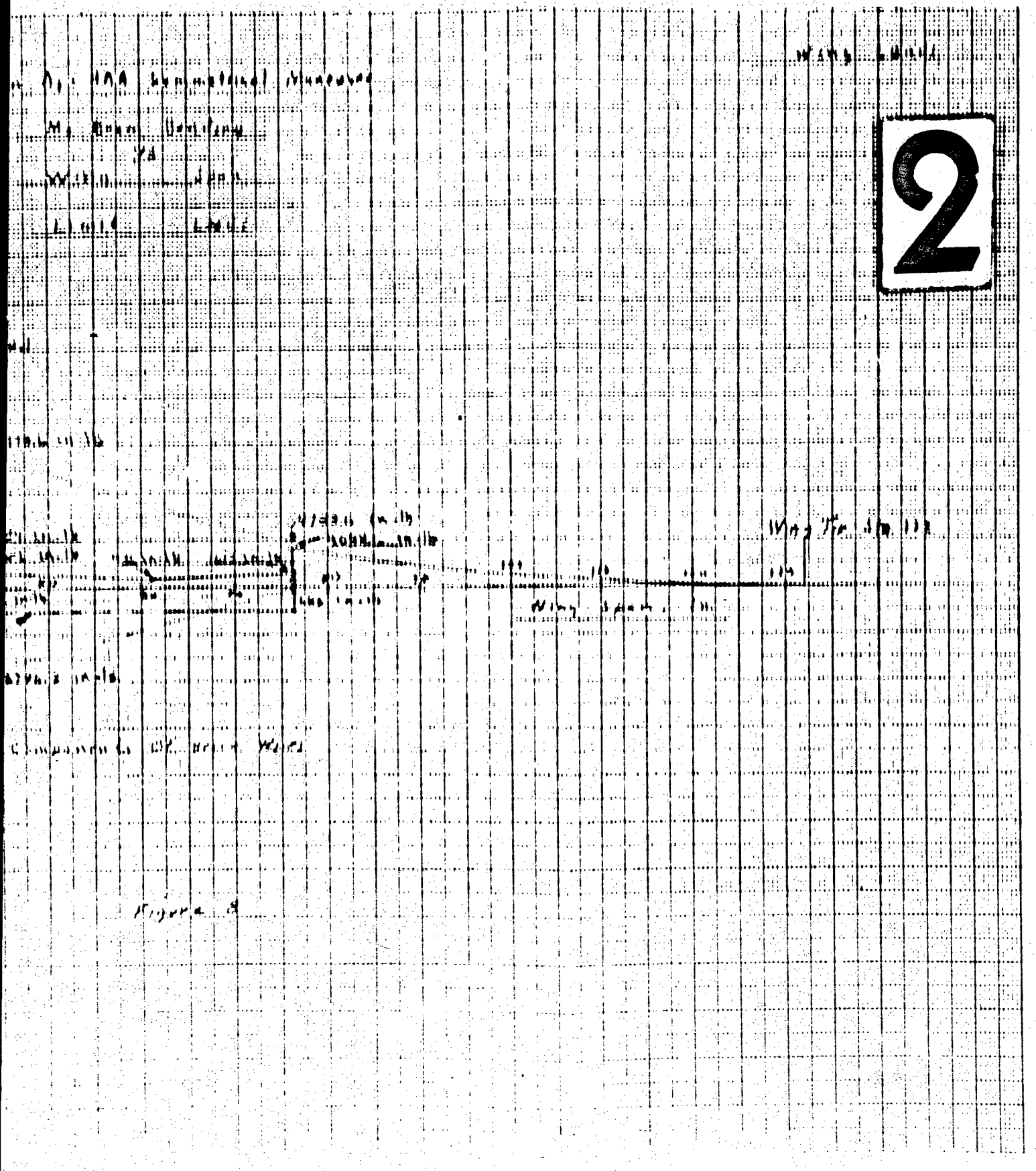

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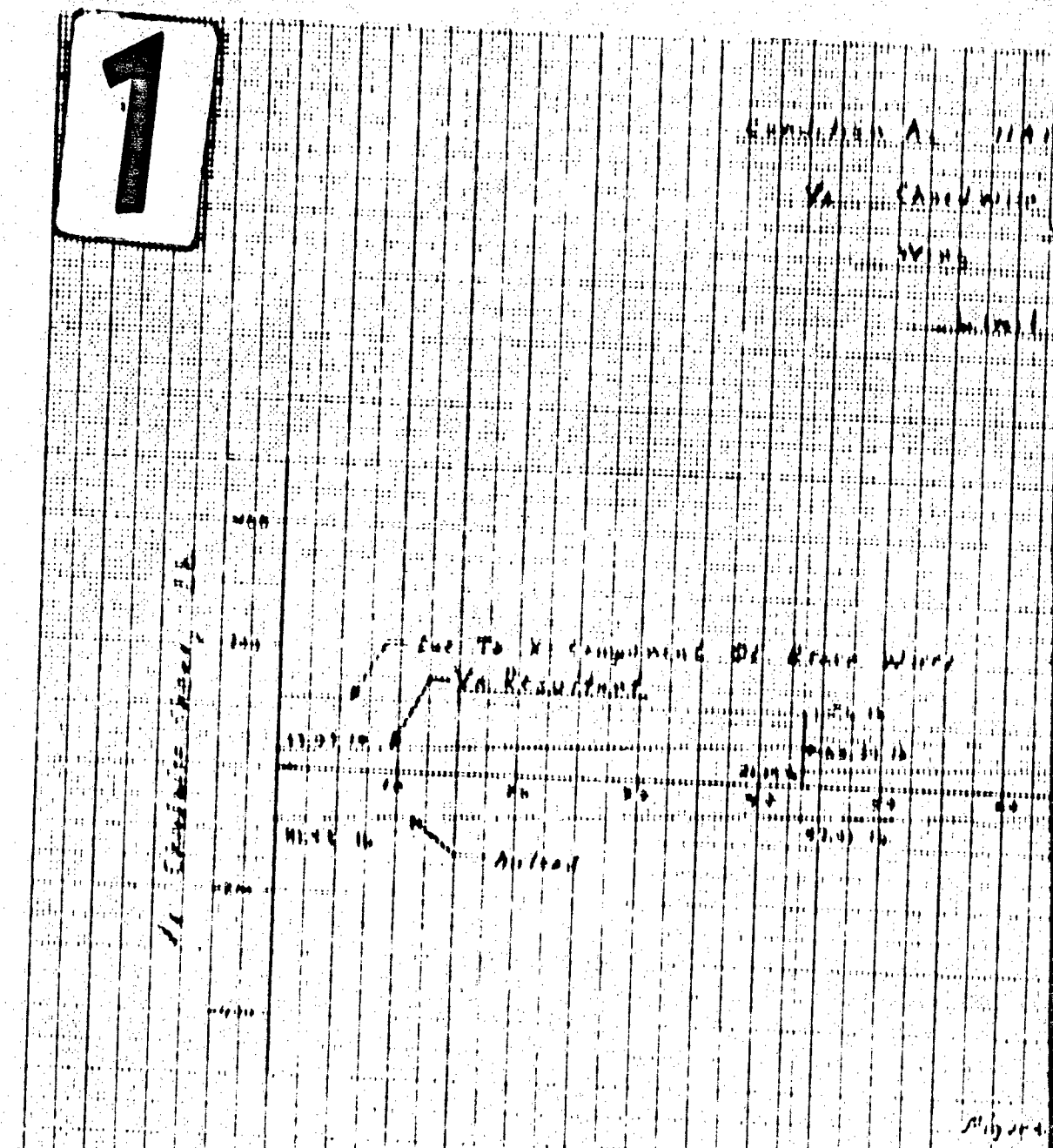

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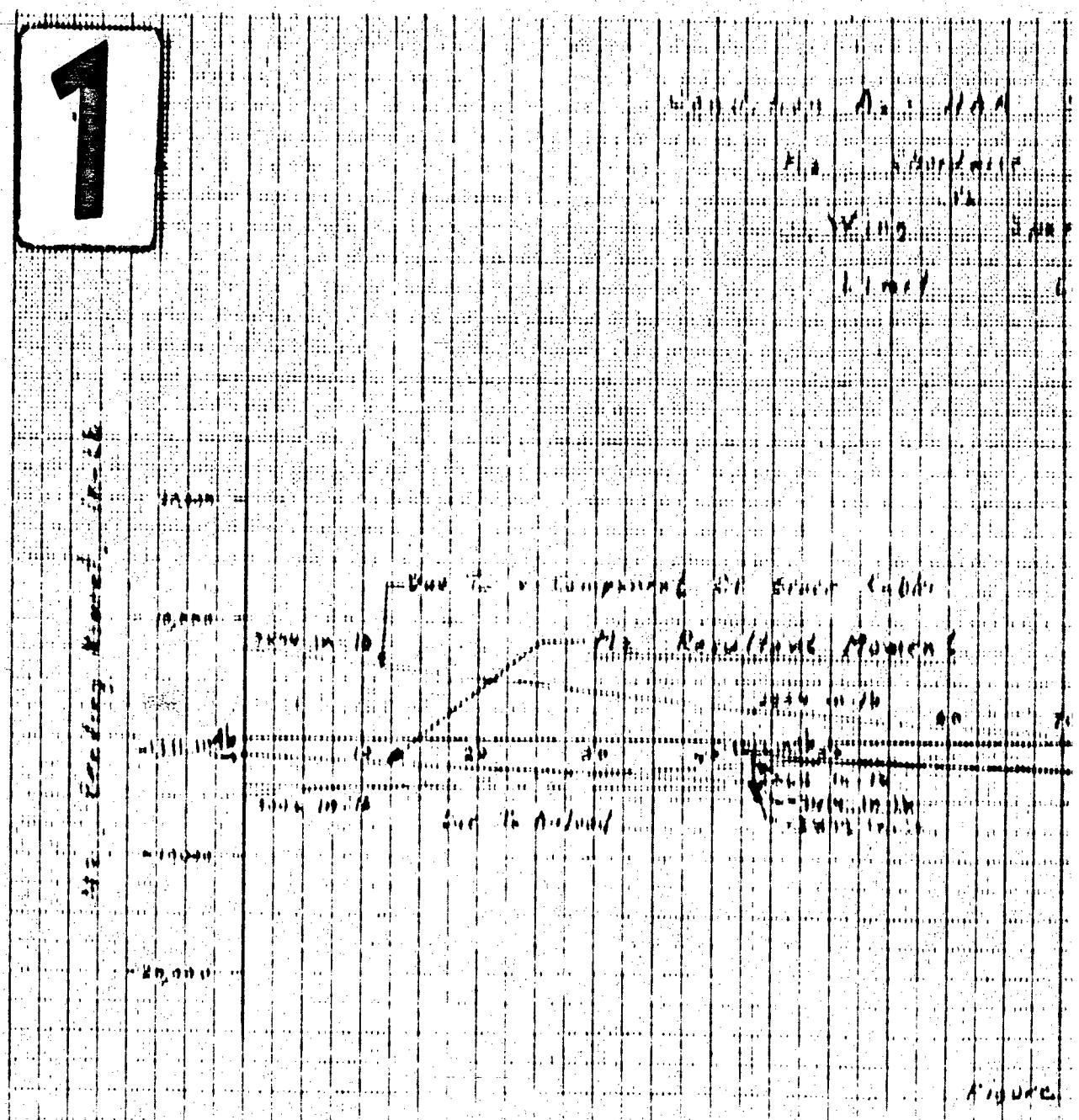

1
Resultant Moment
Figure

2

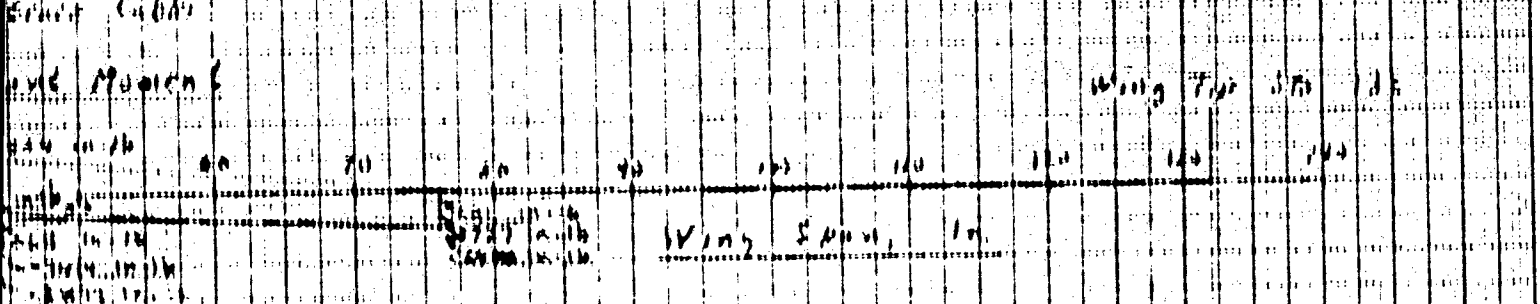
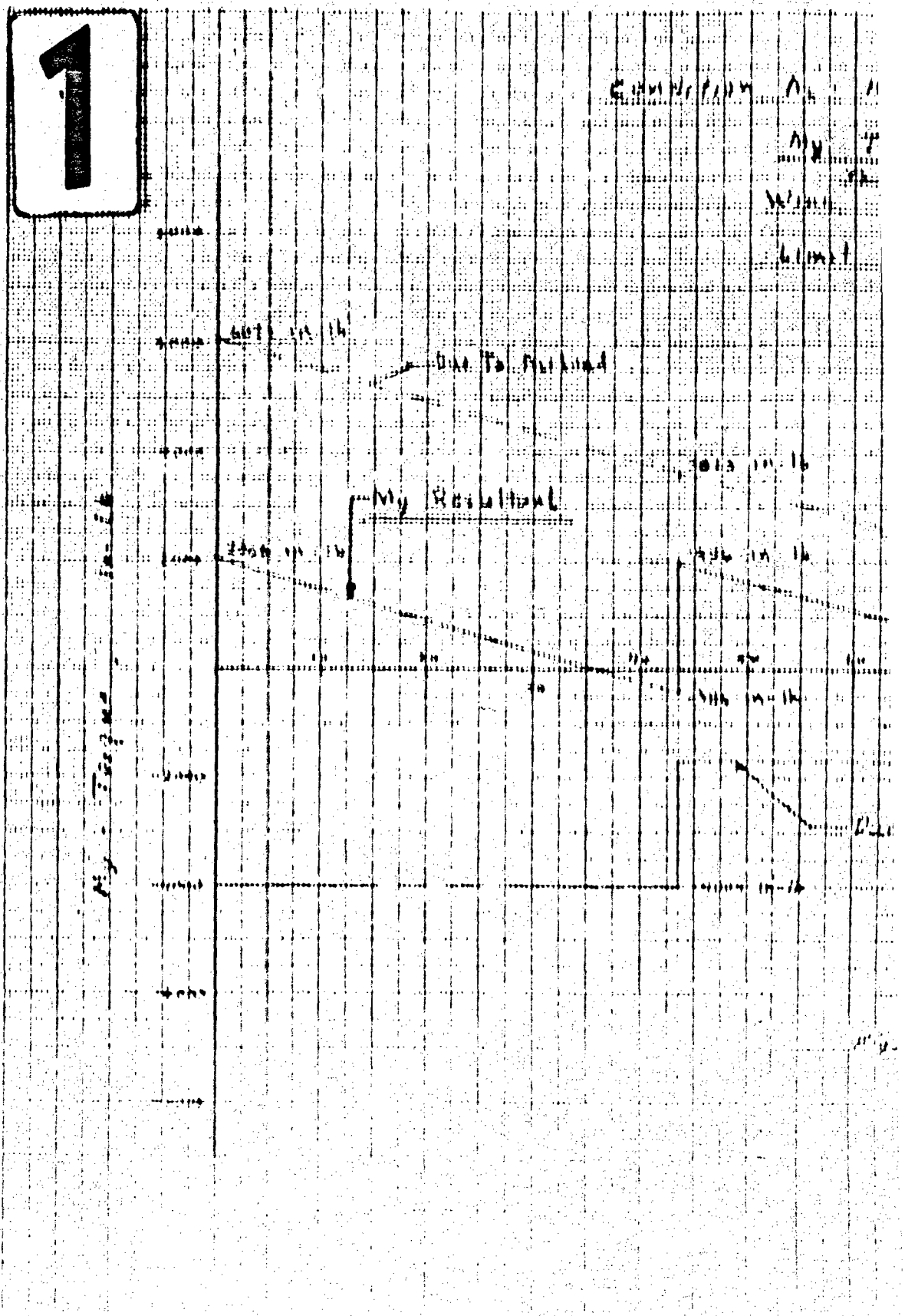
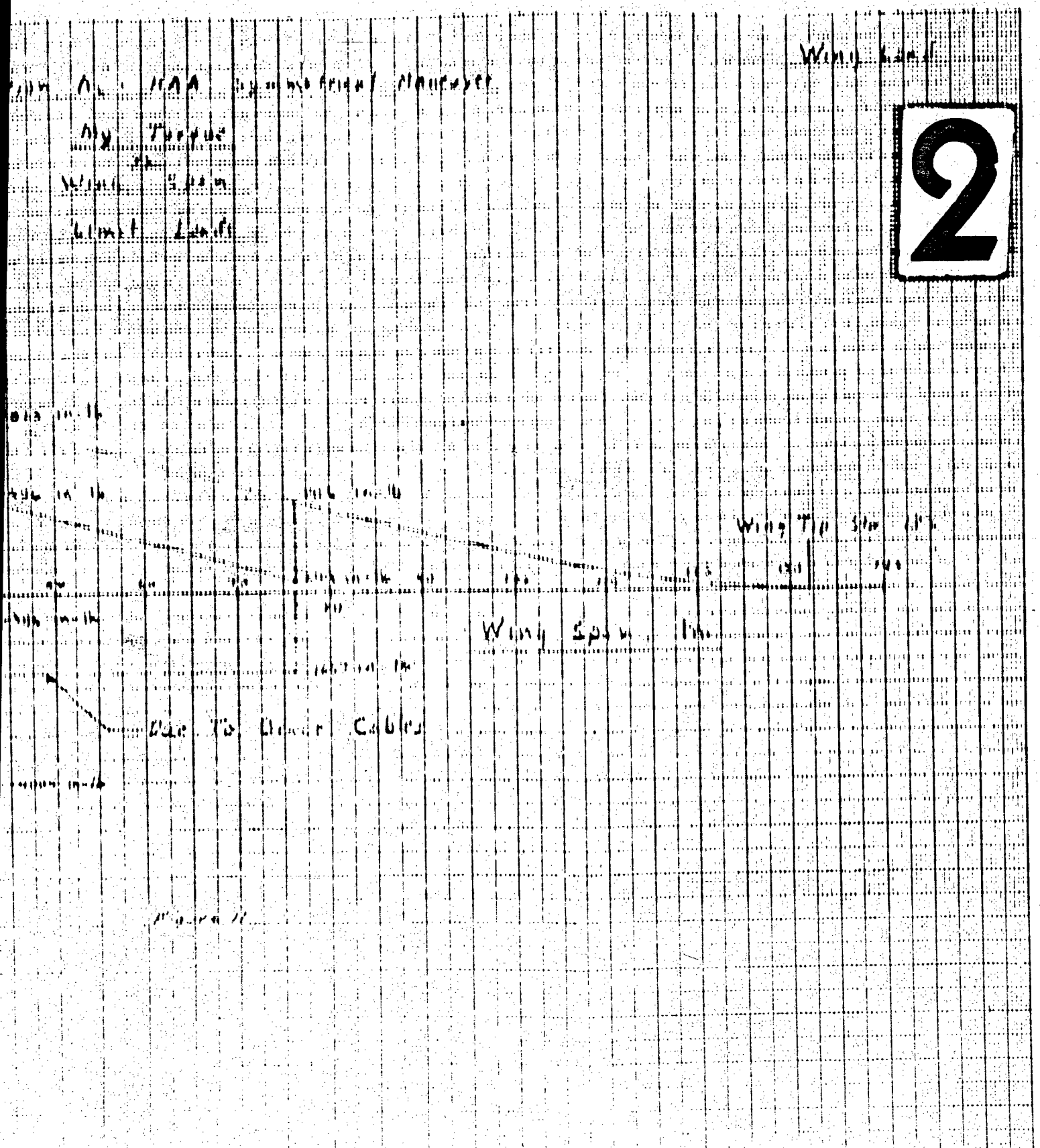


Figure 10





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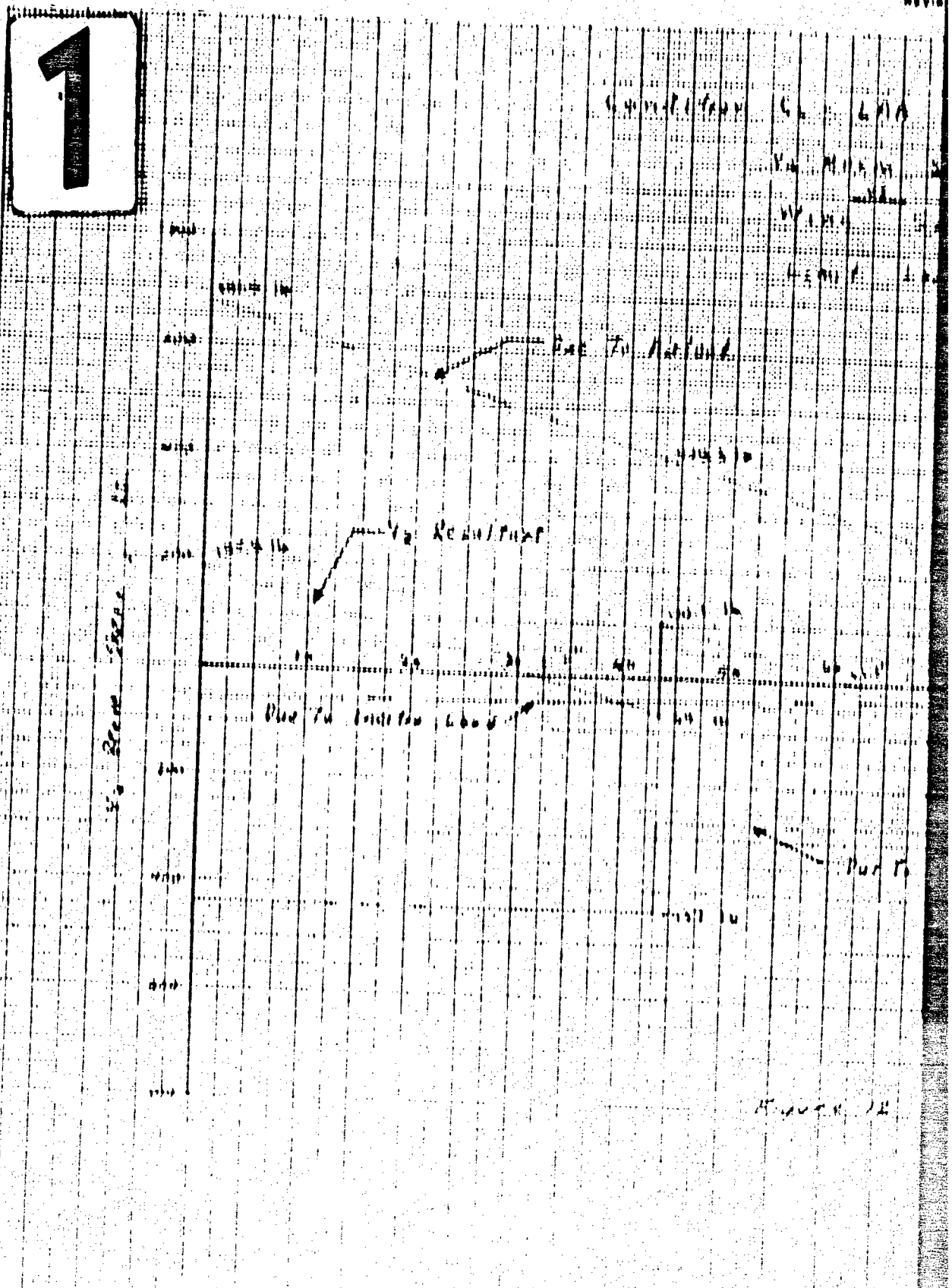

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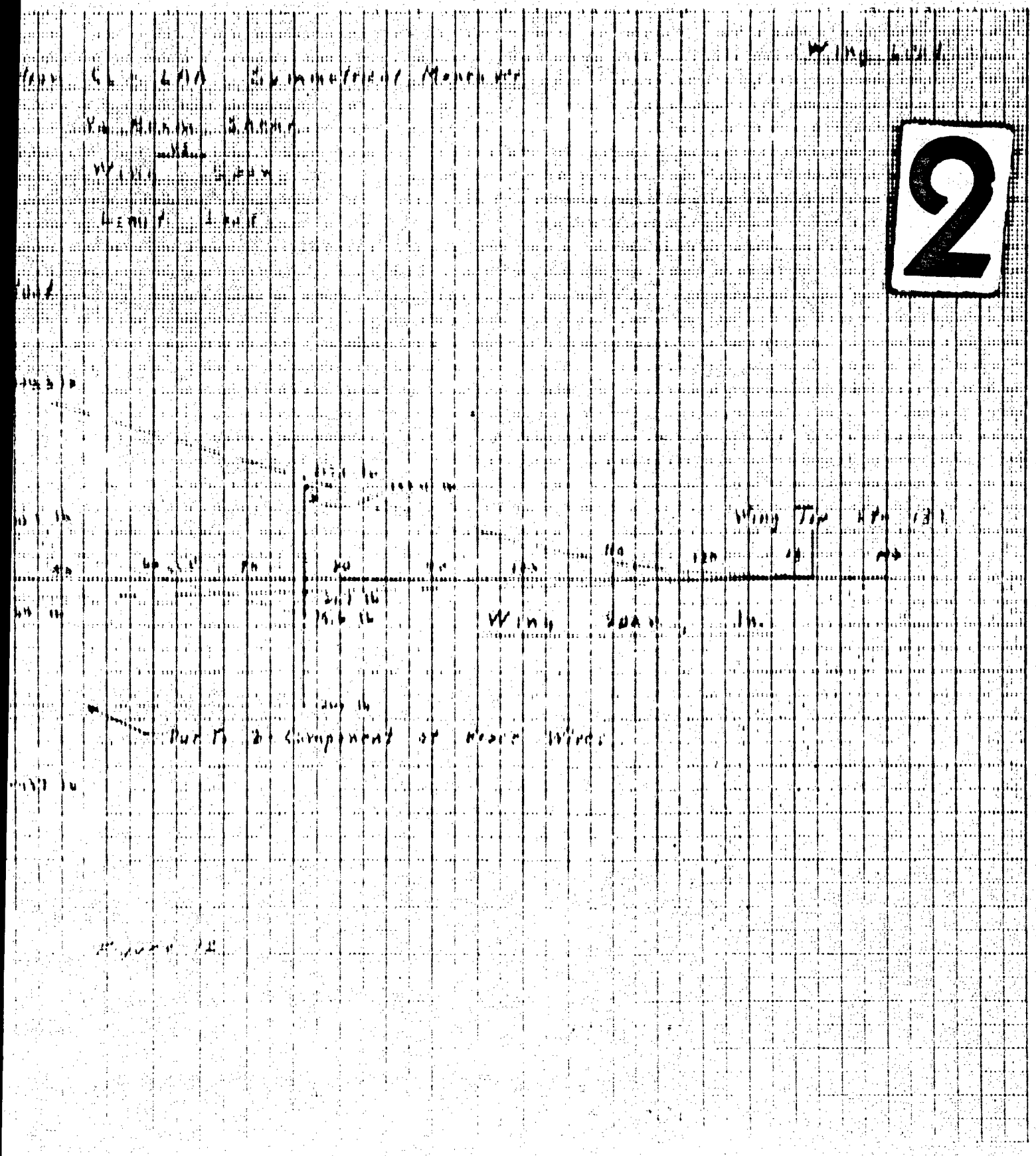

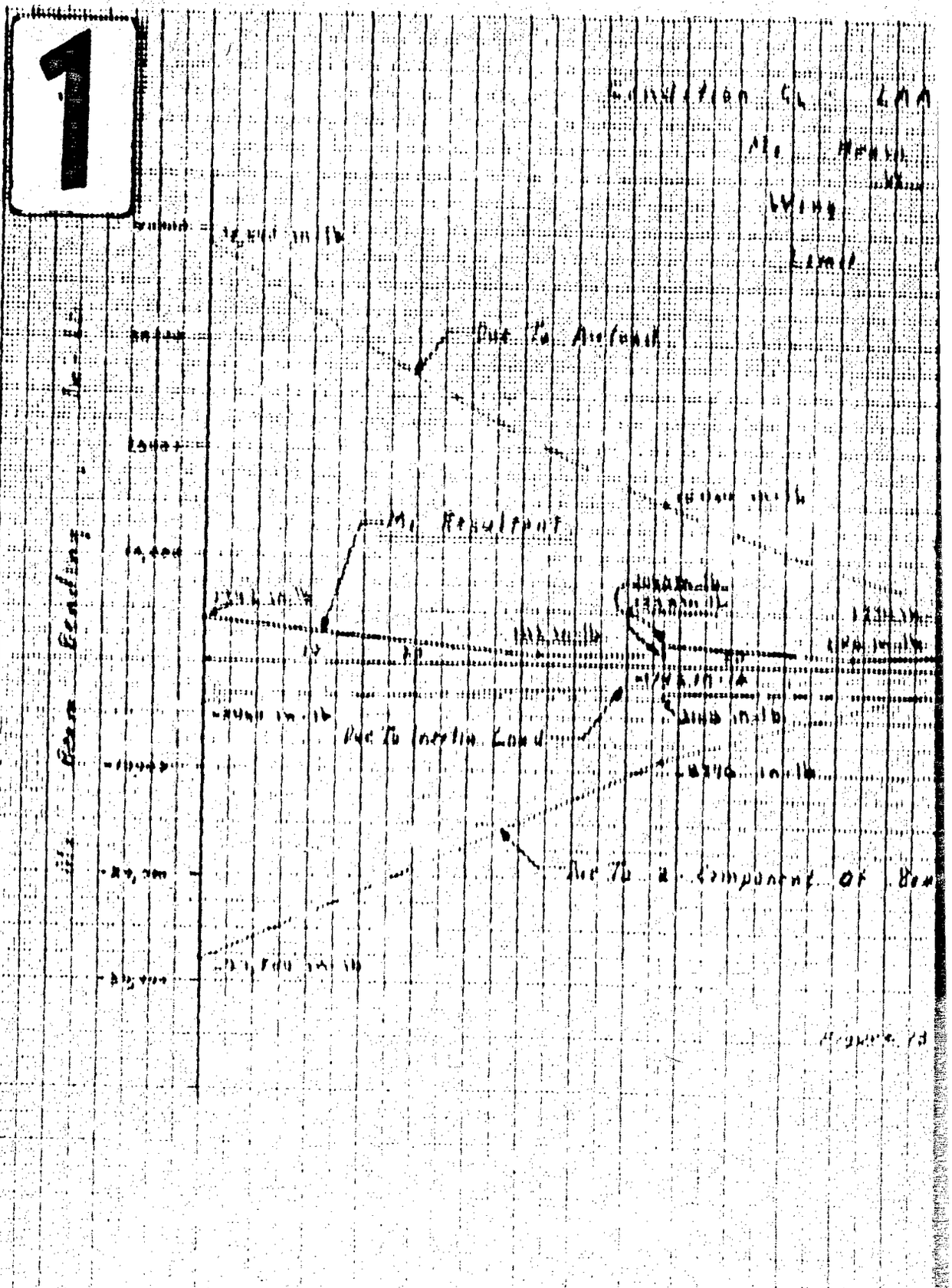
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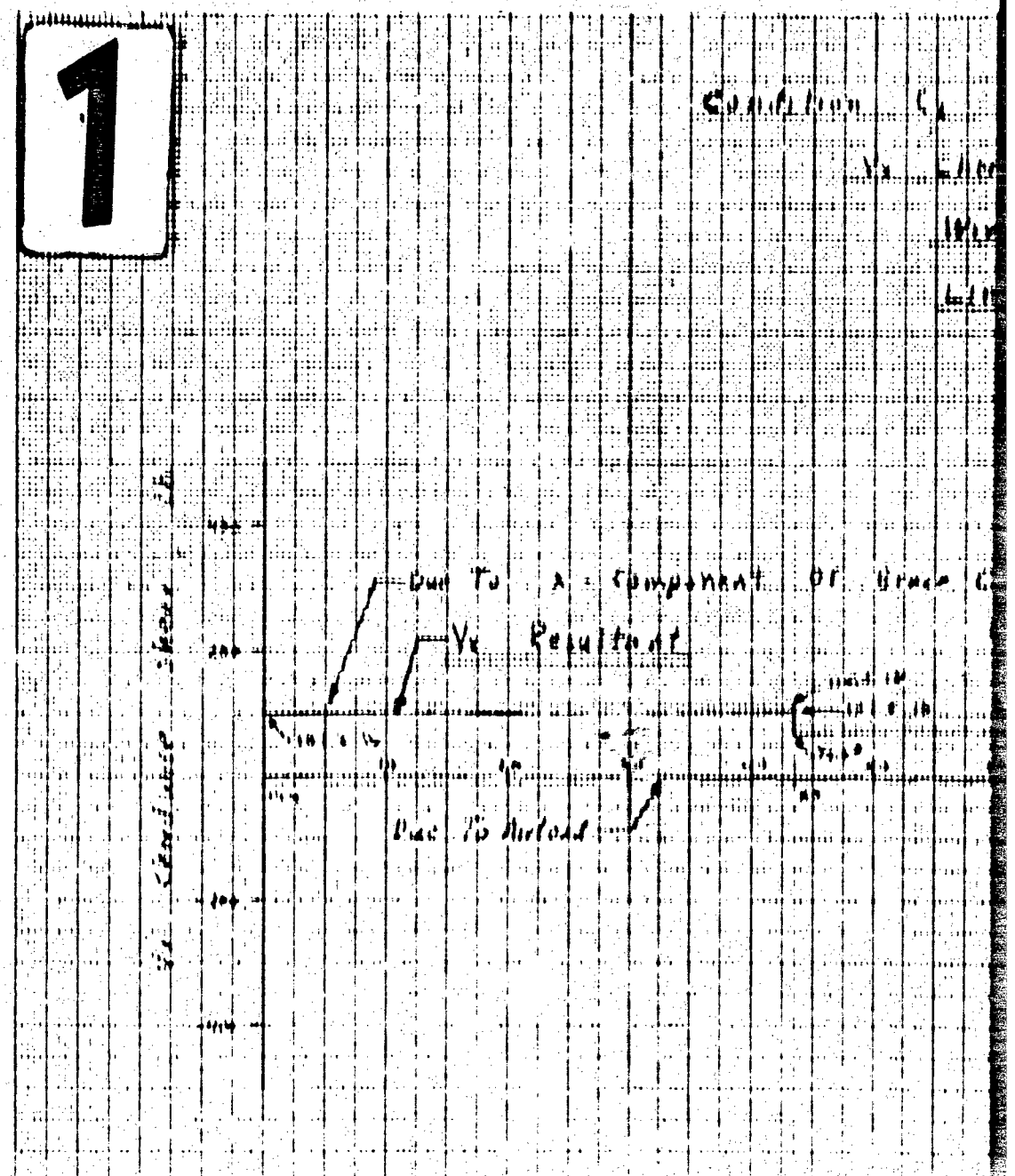
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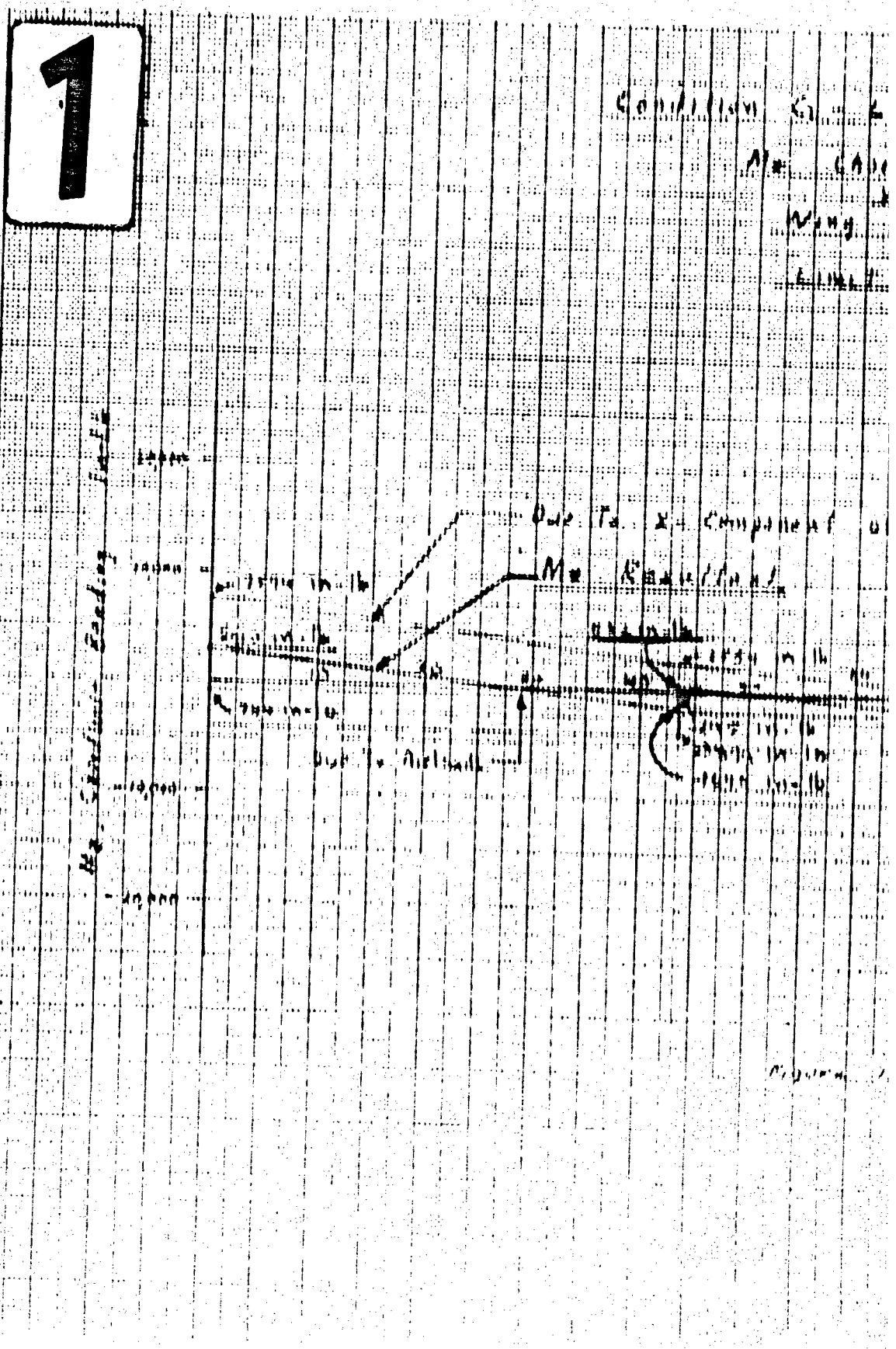
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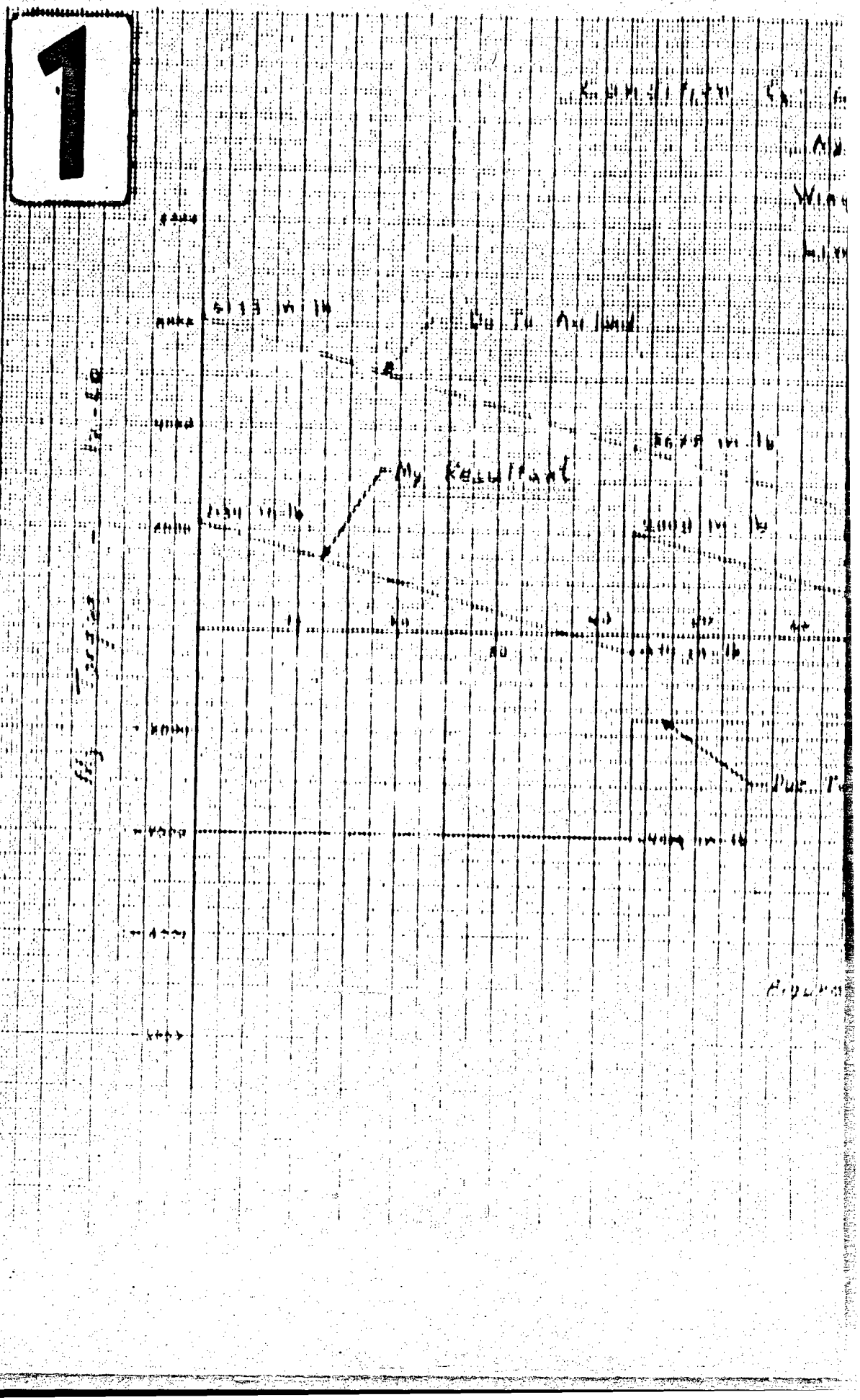

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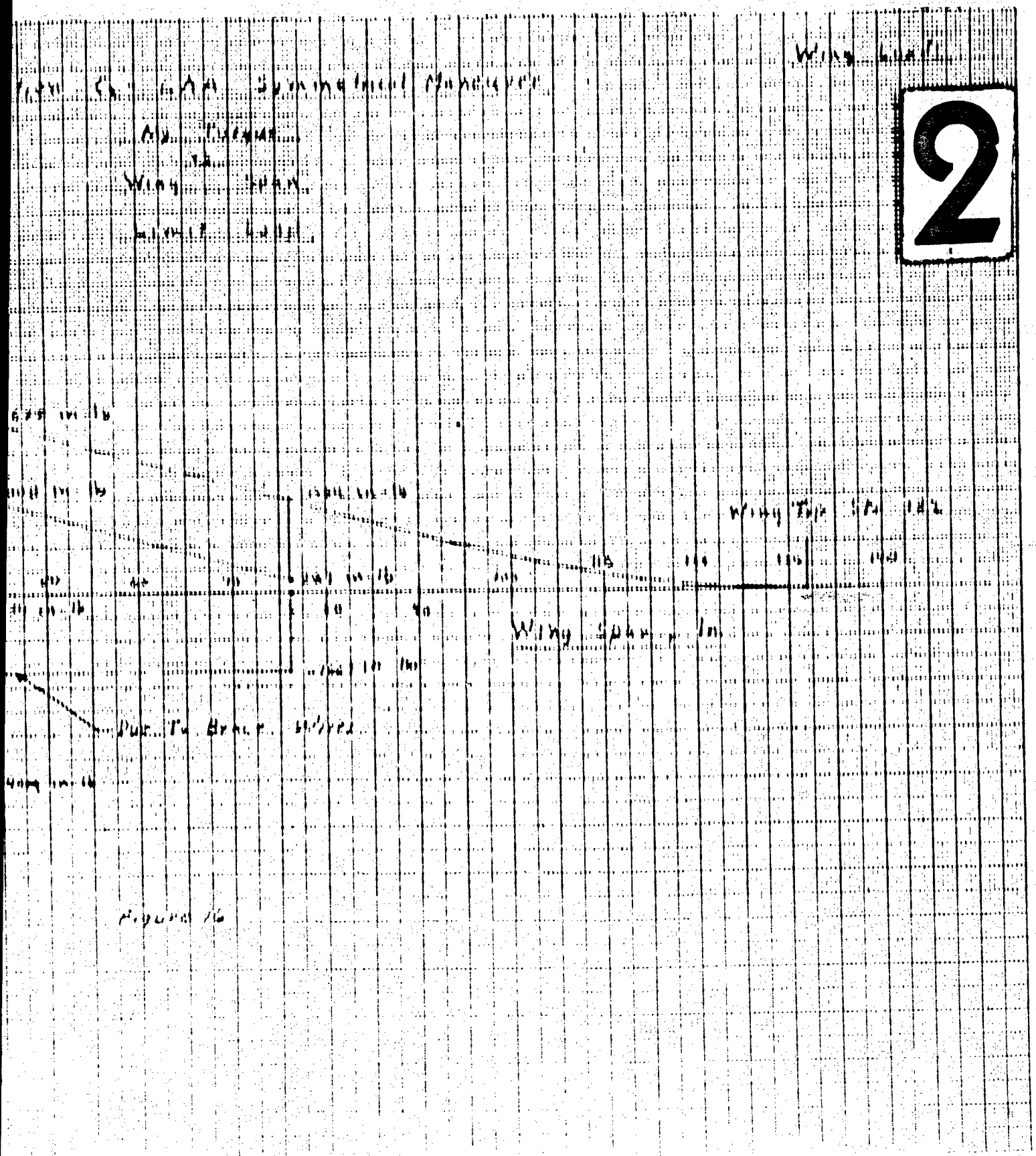

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FUSELAGE LOADS

The most important loads are those due to pitching and yawing accelerations and airloads on the horizontal and vertical surfaces. A cable from the engine mount supports the tail when down loads act.

Figure 17 shows the geometry of the fuselage. All the inertia loads are reduced to concentrated forces at points which are numbered 0 to 11. The point 0 is just beneath the cables that fasten the engine mount to the fuselage while point 11 is at the center of the aft spherical end cap. Forces on the horizontal and vertical tail are resolved at point 11 as forces and couples.

The calculations that follow give the rotational inertia loads for pitching about the most forward and aft centroids. The coordinates used for the centroids are early values and do not agree with the final values, however the effect is negligible.

COORDINATES OF CENTROID

		x	z
Early Values	Most Fwd	68.26	47.30
	Most Aft	78.40	47.50
Final Values	Most Fwd	71.76	44.08
	Most Aft	77.58	47.93

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Fuselage Geometry

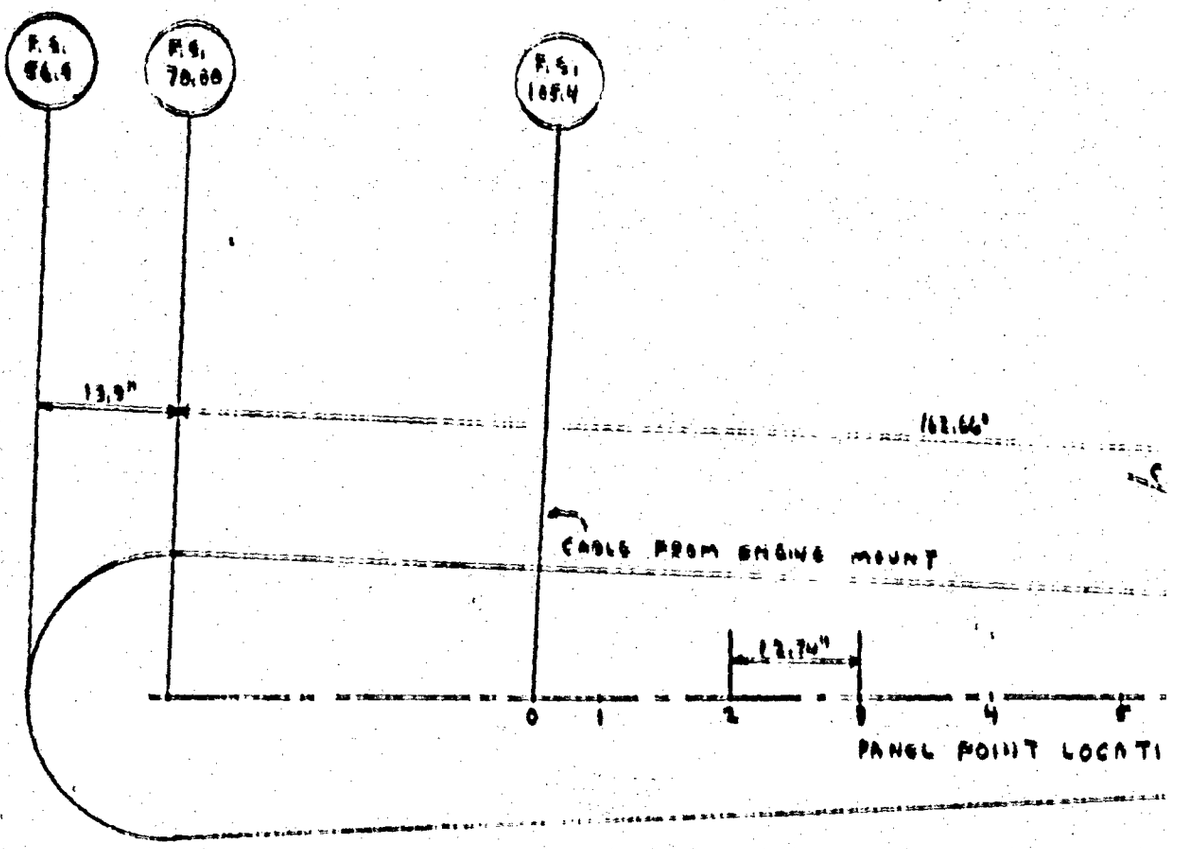

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Figure 17

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FUSELAGE LOADS

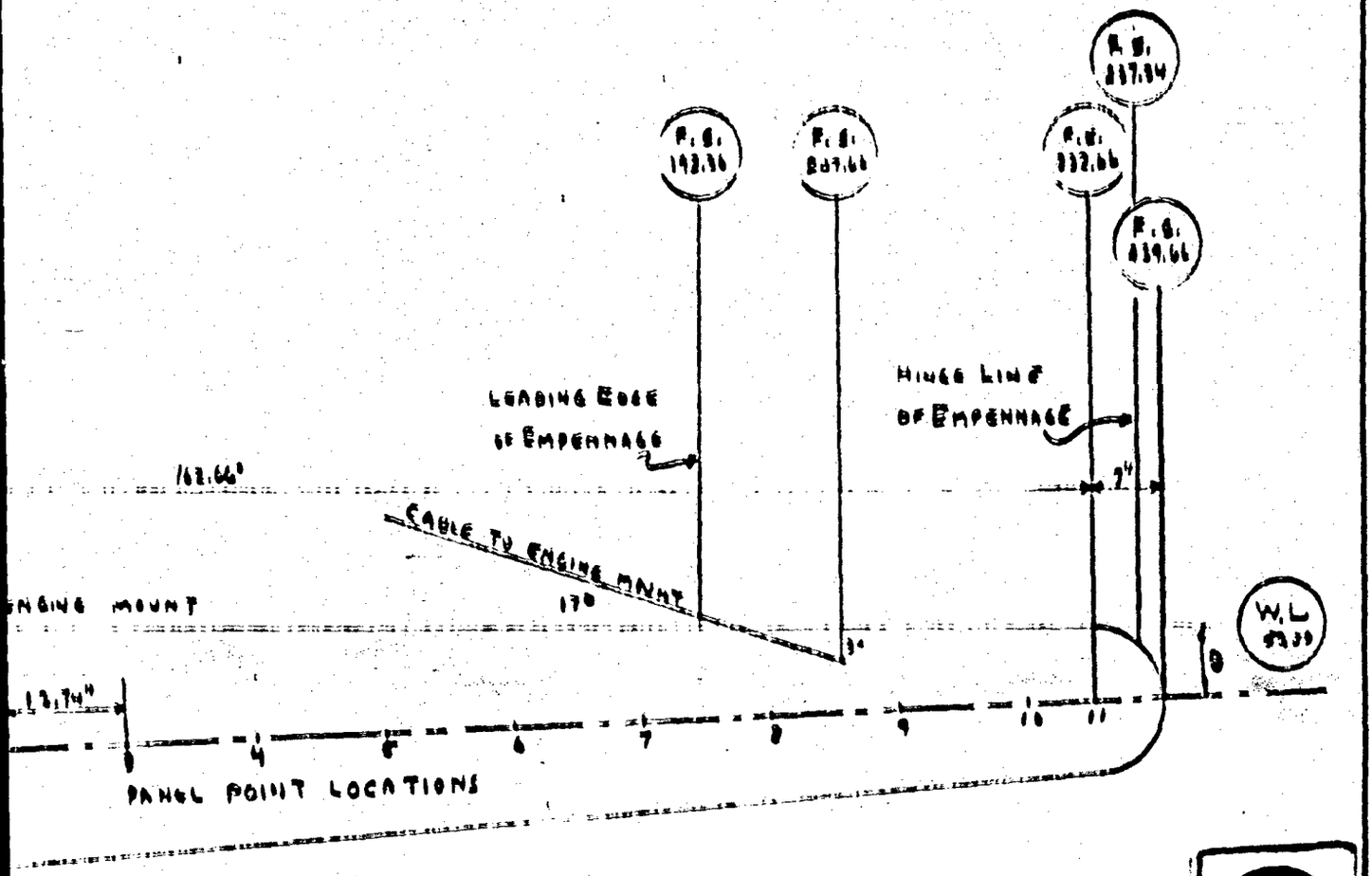

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Figure 17

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FUNCTIONAL INERTIA LOADS OF FUSELAGE — **FUSELAGE LOADS**

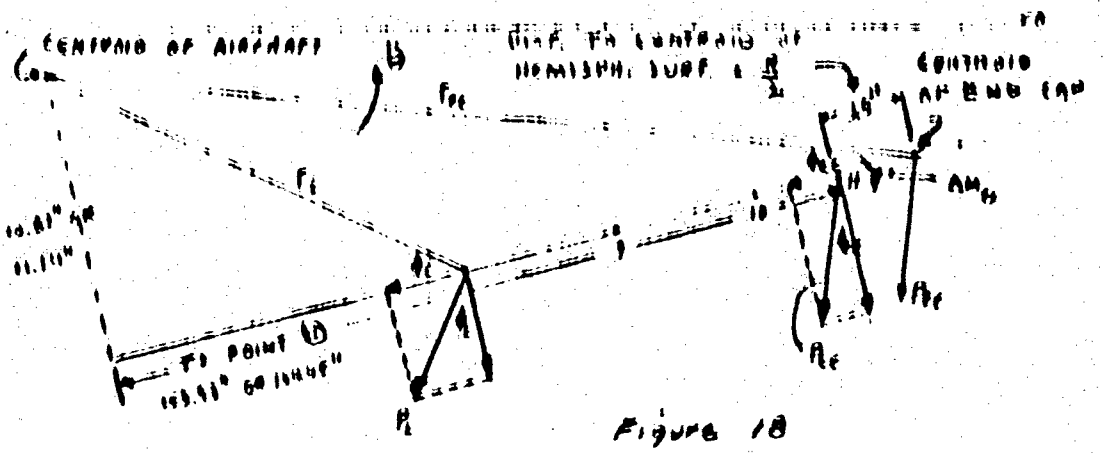

Figure 18

For ANY PANEL WEIGHT, W_t

$$P_t = r_t \ddot{\theta} \frac{W_t}{g}$$

$$P_t \cos\phi_t = \ddot{\theta} \frac{W_t}{g} (r_t \cos\phi_t) \quad \text{BENDING COMPONENT}$$

$$P_t \sin\phi_t = \ddot{\theta} \frac{W_t}{g} (r_t \sin\phi_t) \quad \text{THRUST COMPONENT}$$

For TRANSFER OF END CAP LOAD, P_{ec}, TO POINT (B)

MOST AFT POSITION OF C.G. — MOST FWD POSITION OF C.G.

$\phi_{ec} = \tan^{-1} \frac{10.81}{165.91 + 8.1} = 3.16°$

$r_{ec} = \frac{165.91}{\cos 3.16} = 166.1"$

$P_{ec} = \frac{W_{ec}}{g} r_{ec} \ddot{\theta} = \frac{.061}{32.2} \frac{173.1}{12} \ddot{\theta} = .2718 \ddot{\theta}$ LB

Bend Comp: $P_{ec} \cos\phi_{ec}$

$= \ddot{\theta} \frac{W_{ec}}{g} (r_{ec} \cos\phi_{ec}) = .2714 \ddot{\theta}$ LB

Thrust Comp: $\frac{\ddot{\theta} W_{ec}}{g} (r_{ec} \sin\phi_{ec}) = .01711 \ddot{\theta}$ LB (Compression)

$\Delta M_{ec} = 3.3 P_{ec} \cos\phi_{ec}$ IN-LBS

$= .8964 \ddot{\theta}$ IN LB

$\phi_{ec} = \tan^{-1} \frac{11.11}{167.96 + 8.1} = 3.32°$

$r_{ec} = \frac{167.96}{\cos 3.32} = 167.9"$

$P_{ec} = \frac{.61}{32.2} \frac{167.9}{12} \ddot{\theta} = .265 \ddot{\theta}$ LB

Bend. Comp = $.265 \ddot{\theta}$ LB

Thrust comp = $.01765 \ddot{\theta}$ LB (Compression)

$\Delta M_{ec} = .873 \ddot{\theta}$ IN. LB

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FUSELAGE LOADS

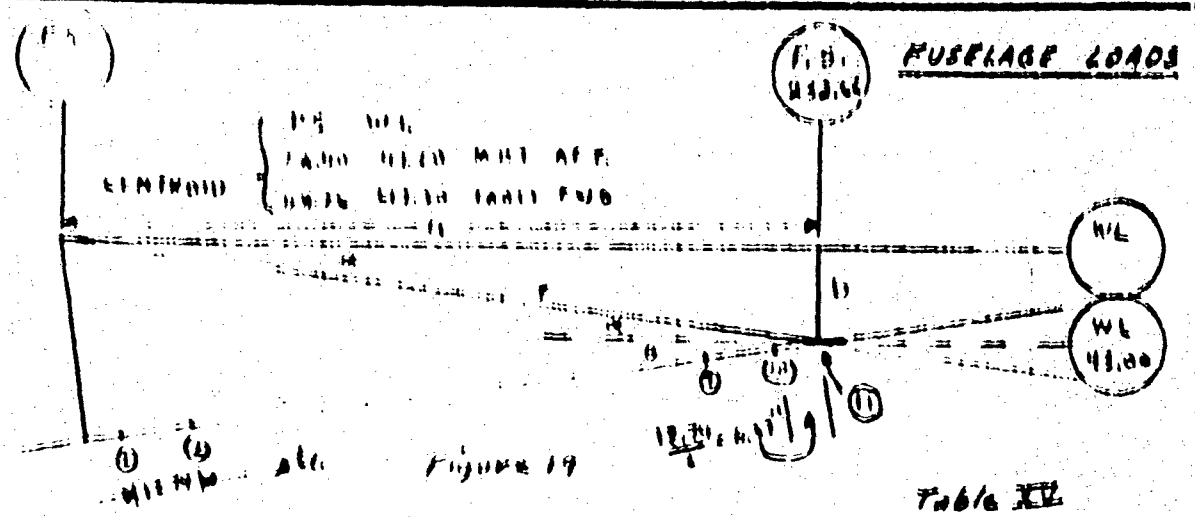

Figure 19

For MOST AFT POSITION

$a = 232.66 - 78.40 = 154.76''$

$b = 47.50 - 43.00 = 4.50''$

$\alpha = \tan^{-1} \frac{b}{a} = 1.67°$

$\theta = 2.29°$

$\phi = \alpha + \theta = 3.96°$

$r = a \sec \alpha = 154.83$

$r \sin \phi = 10.65''$

$r \cos \phi = 153.93''$

For MOST FWD POSITION

$a = 232.66 - 68.76 = 164.40''$

$b = 47.30 - 43.00 = 4.30''$

$\alpha = \tan^{-1} \frac{b}{a} = 1.50°$

$\theta = 2.29°$

$\phi = \alpha + \theta = 3.79$

$r = a \sec \alpha = 164.45''$

$r \sin \phi = 10.88''$

$r \cos \phi = 163.96''$

Table XV

Point	$r \cos \phi$	
	Most FWD	Most AFT
	IN.	IN
1	42.93	32.90
2	35.67	45.64
3	68.41	58.38
4	81.15	71.12
5	93.89	83.86
6	106.63	96.60
7	119.37	109.34
8	132.11	122.08
9	144.85	134.82
10	157.59	147.55
11	163.96	153.93

FUSELAGE LOADS

Example of Tail Inertia Load for Point (11)

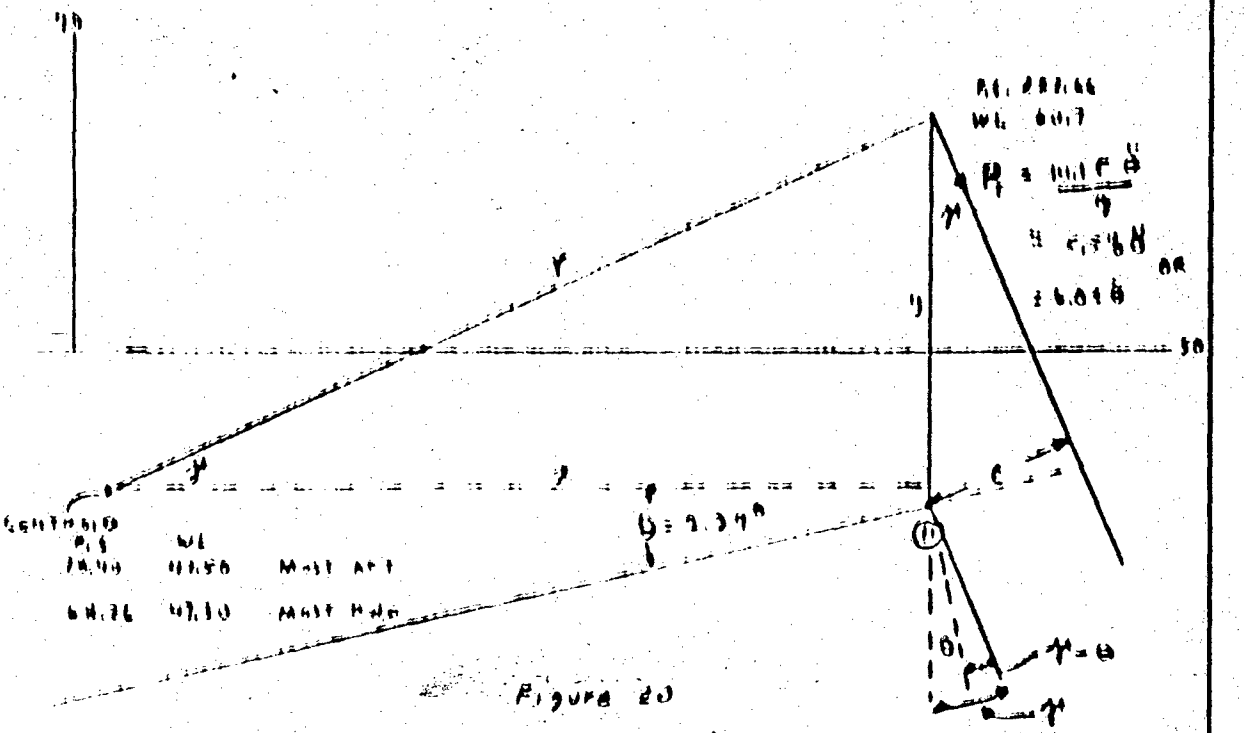

Figure 20

Most Aft Position

$y = 60.70 - 47.30 = 13.40''$

$x = 132.66 - 78.40 = 154.26''$

$\gamma = \tan^{-1} \frac{13.40}{154.26} = 4.91°$

$r = \frac{154.26}{\cos \gamma} = 155.20''$

$c = (60.70 - 47.30) \sin \gamma = 1.52''$

Moment = $Pc = 5.58\ddot{\theta} \times 1.52 = 8.50\ddot{\theta}$ in.lb

Bend. Comp = $P \cos(\gamma - \theta) = P \cos 2.64°$

$= .9990 P$

$= 5.58\ddot{\theta}$ LB

Thrust comp = $P \sin(\gamma - \theta)$
(Tension) $= .0461 P$

$= .257\ddot{\theta}$ LB

Most Fwd Position

$y = 60.70 - 47.30 = 13.40''$

$x = 132.66 - 68.26 = 164.40$

$\gamma = \tan^{-1} \frac{13.40}{164.40} = 4.68°$

$r = \frac{164.40}{\cos \gamma} = 165.2''$

$c = 13.40 \sin \gamma = 1.44''$

Moment = $Pc = 6.03\ddot{\theta} \times 1.44$

$= 8.68\ddot{\theta}$ in.lb

Bend Comp = $P \cos 2.37$

$= .9993 P$

$= 6.03\ddot{\theta}$ LB

Thrust Comp = $.0413 P$
(Tension)

$= .249\ddot{\theta}$ LB.

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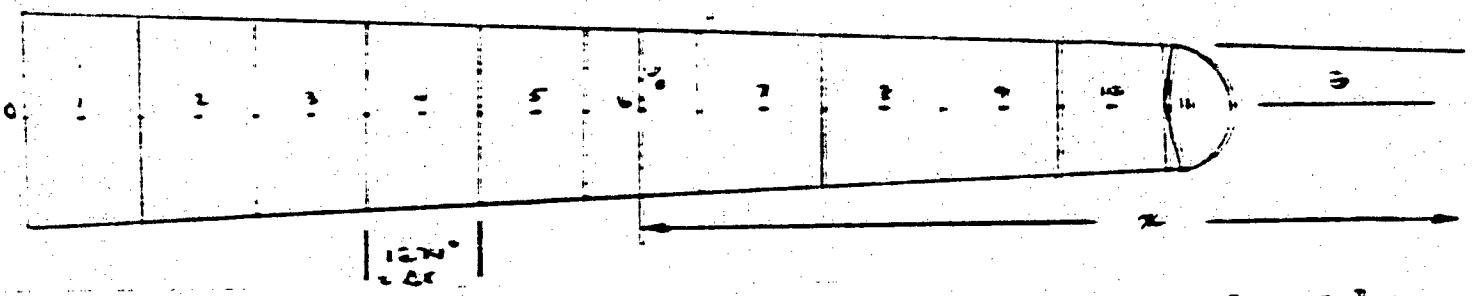
TABLE XVI RETATIONAL AND TRANSLATIONAL INERTIA LOADS OF FUSELAGE



$y = x \tan\theta$ $\quad y_{10} = 181.50 \tan 22.9^\circ = 7.27''$ $\quad A_{rea} = 2\pi y\,\Delta x$ $\quad A_{11} = \pi(7)^2 = 308\ in^2$

		y in	A in²	Wt lb	$r_c \cos\phi_c$ in	Bend Comp. lb	Thrust Comp lb	$r_c \cos\phi_c$ in.	Bend Comp in-lb	Thrust Comp lb	my^2 in²
	0	12.28									475
	1	12.04	978	1.937	42.93	.215	-.0558	32.9	.265	.0845	467
	2	11.51	935	1.852	55.67	.267	-.0695	45.64	.319	.0822	417
	3	10.98	892	1.765	68.41	.313	-.0556	58.35	.267	.0487	330
	4	10.45	849	1.682	81.15	.353	-.0465	71.12	.309	.0473	344
	5	9.93	806	1.594	93.89	.388	-.0360	83.86	.346	.0447	310
	6	9.40	763	1.510	106.63	.417	-.0435	96.60	.377	.0425	275
	7	8.87	720	1.425	119.37	.440	-.0411	109.34	.403	.0402	242
	8	8.34	677	1.340	132.11	.458	-.0386	122.08	.423	.0377	209
	9	7.80	634	1.255	144.85	.472	-.0362	134.82	.438	.0354	1911.5
	10	7.27	591	1.170	157.59	.477	-.0322	147.52	.443	.0325	1661.3
END CAP	R_c		302	.610	163.96	.265	-.0785	153.93	.248	.0783	
TAIL	P_T					.503	+.249		.507	.287	
END CAP	ΔM					.926			.868		
TAIL	ΔM					8.68			9.53		

$\Sigma = -.2$

NET THRUST AT POINT 6 IS 0.2# COMP.

NET THRUST BETWEEN POINTS 7 AND 8 IS 0.1# TENSION

THESE THRUSTS ARE NEGLIGIBLE

PUSHBAR LOADS

GOODYEAR AIRCRAFT

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SHEET 0A1W4
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FUSELAGE LOADS

HORIZONTAL AND VERTICAL TAIL
PROPULSION TAIL LOADS

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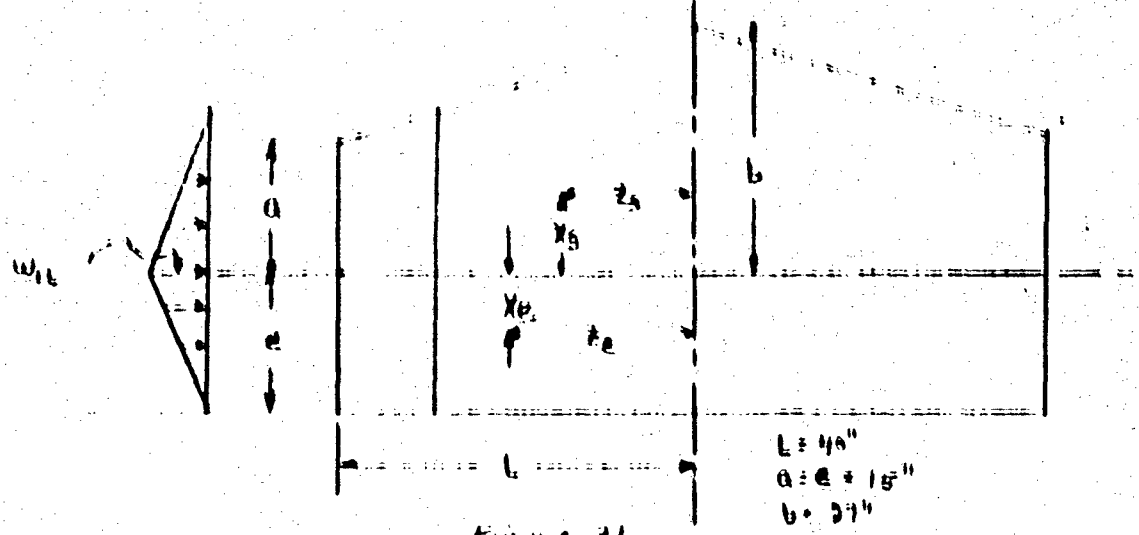

Figure 21

$P_s = \frac{a+b}{4} w_{1L} L$ = LOAD ON STABILIZER HALF SPAN

$P_e = \frac{a}{2} w_{1L} L$ " " " ELEVATOR " "

FOR HORIZ TAIL

$$x_s = \frac{2}{4}\,\frac{b^2+ab+a^2}{b+a} \qquad x_e = \frac{e}{2} \qquad w_{1L} = \frac{4W}{L(a+b+2e)}$$

$$z_s = \frac{2a+b}{2(a+b)} L \qquad z_e = \frac{L}{2}$$

W = LOAD ON HALF SPAN.

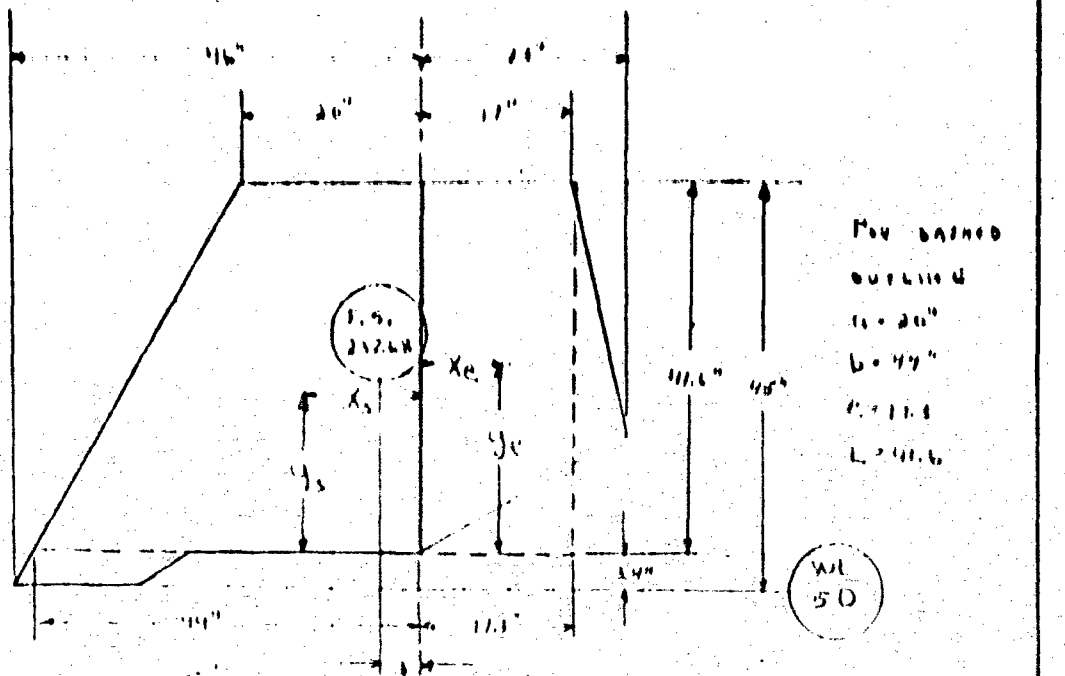

Figure 22

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FUSELAGE LOADS

FOR 100# LOAD ON HORIZONTAL OR VERTICAL TAIL SURFACES

	P_s	X_s	z_s HORIZ y_s VERT	P_e	X_e	z_e HORIZ y_e VERT
HORIZONTAL TAIL	29.7#	7.77"	18.07"	20.8#	5"	20"
VERTICAL TAIL	64.8#	11.76"	18.77"	35.2#	5.77"	20.8"

W = 50 LB FOR HORIZONTAL TAIL
= 100 LB " VERTICAL "

TORQUE CAUSED BY UNEQUAL LOADS ON ELEVATOR

LET P_{SL} = LOAD ON LEFT STABILIZER, P_{SR} = " RIGHT " $\Big\}$ $P_{SL} + P_{SR} = 2P_S$

FROM AERODYNAMICS, $\frac{P_{SL}}{P_{SR}} = \frac{1.7}{.8} = \frac{3}{2}$

$T_s = (P_{SL} - P_{SR})\, z_s$ = TORQUE FROM STABILIZER LOADS
$T_e = (P_{eL} - P_{eR})\, z_e$ = " " ELEVATOR "

THEN $T_s = \left[\frac{3}{5}(2P_s) - \frac{2}{5}(2P_s)\right] z_s = \frac{2}{5} P_s z_s$

$T_e = \frac{2}{5} P_e z_e$

NET TORQUE = $T_s + T_e = \frac{2}{5}(P_s z_s + P_e z_e)$

$\frac{2}{5}(29.7 \times 18.07 + 20.8 \times 20) = 378$ IN-LB

OF TORQUE FOR 100# ON THE HORIZONTAL TAIL

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FUSELAGE LOADS

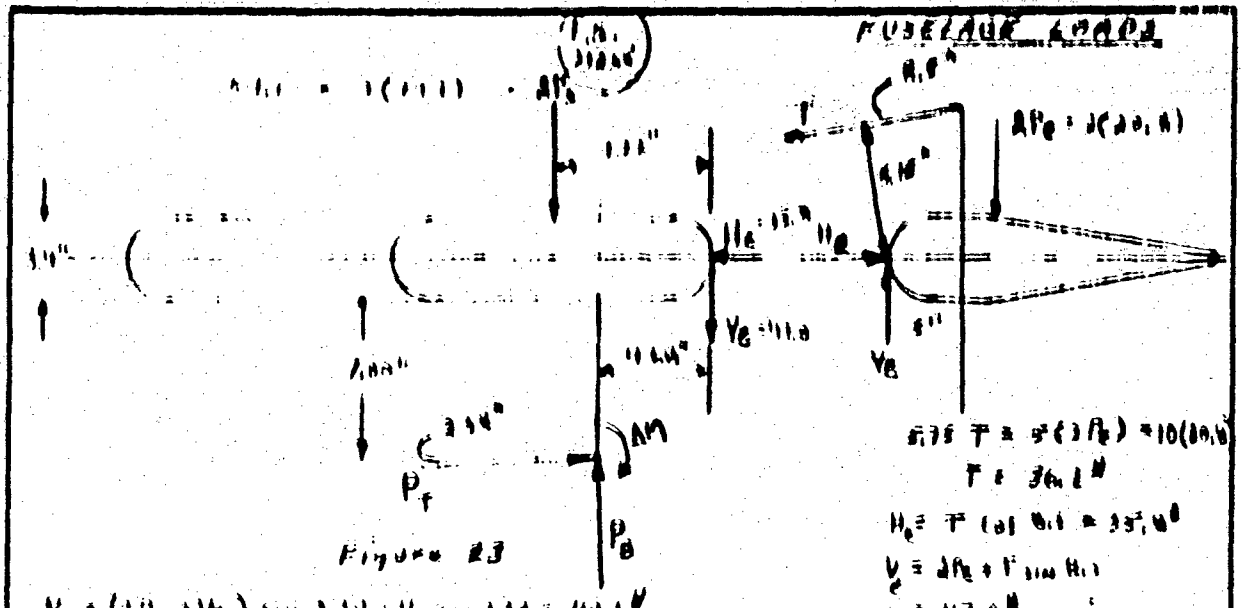

Figure 23

$5.75\,T = 5(2P_e) = 10(20.8)$

$T = 36.2^{\#}$

$H_e = T \cos 8.1 = 35.8^{\#}$

$V_e = 2P_e + T \sin 8.1$

$= 47.0^{\#}$

$P_T = (2P_1 + V_e)\sin 2.38 + H_e \cos 2.38 = 40.1^{\#}$

$P_B = (2P_1 + V_e)\cos 2.38 - H_e \sin 2.38 = 103.9^{\#}$

$\Delta M = (7.22 - 4.68)(2P_1) + (7.02 + 1.7)H_e - 4.68\,V_e = 239$ in.#

CALCULATIONS FOR $100^{\#}$ DOWN

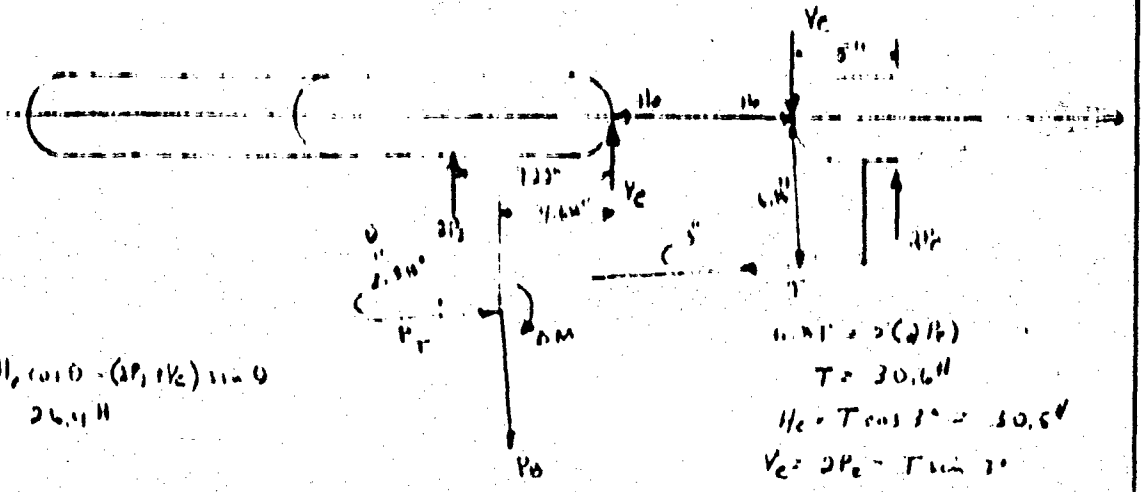

Figure 24

$6.8\,T = 5(2P_e)$

$T = 30.6^{\#}$

$H_e = T \cos 3° = 30.5^{\#}$

$V_e = 2P_e - T \sin 3°$

$= 40^{\#}$

$P_T = H_e \cos\theta - (2P_1 + V_e)\sin\theta$

$= 26.4^{\#}$

$P_B = (2P_1 + V_e)\cos\theta + H_e \sin\theta$

$= 100.8^{\#}$

$\Delta M = -2P_1(7.22 - 4.68) + (7.11 + 1.7)H_e + 4.68\,V_e = 302$ in-lb

REACTIONS AT (FS 232.8, WL 43.00) DUE TO UP AND DOWN LOADS ON HORIZONTAL TAIL. LOADS ARE FROM PREVIOUS PAGE

CALCULATIONS FOR $100^{\#}$ UP

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FUSELAGE LOADS

CALCULATIONS FOR 100# ON VERTICAL TAIL

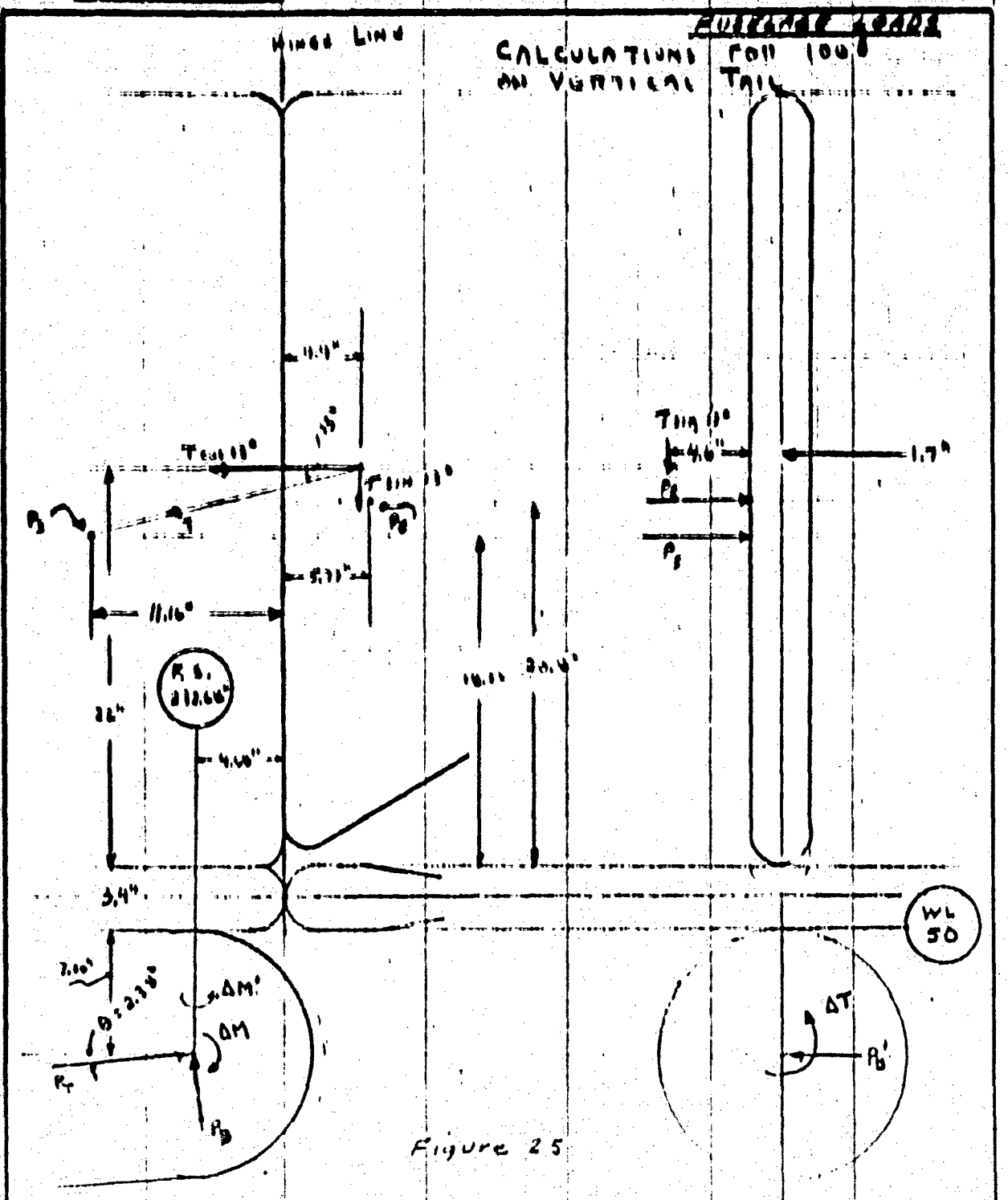

Figure 25

FOR CONTROL FORCE CABLE TENSION

$\Sigma M_{HINGE\ LINE} = 0$ $\quad (T \cos 13)(4.6+1.7) = 5.77\ Pc$

$T \cos 13° = 37.2\#$

$T \sin 13° = 7.45\#$

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FUSELAGE LOADS

CALCULATIONS FOR 100# ON VERTICAL TAIL

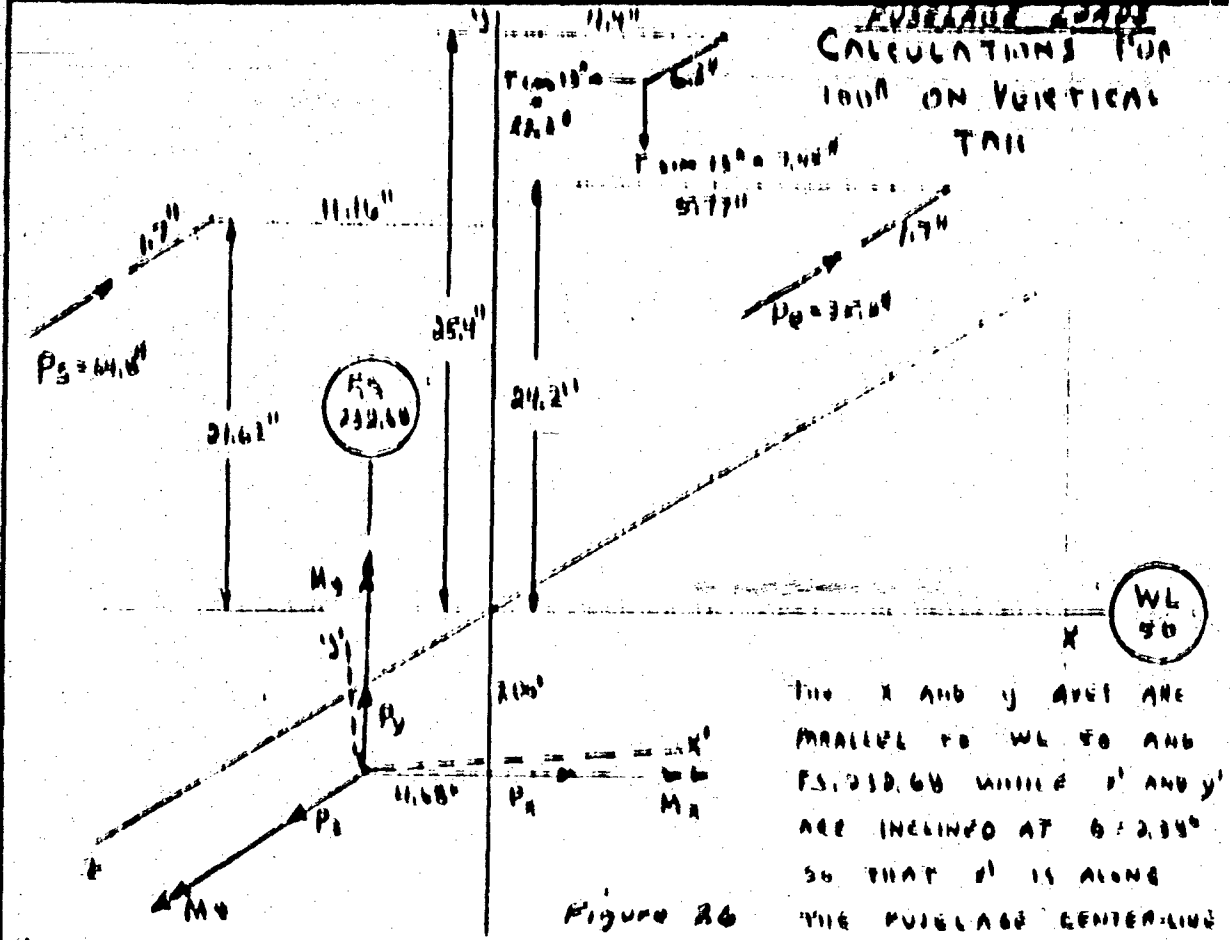

Figure 26

THE x AND y AXES ARE PARALLEL TO WL 50 AND FS. 232.68 WHILE x' AND y' ARE INCLINED AT θ = 2.38° SO THAT x' IS ALONG THE FUSELAGE CENTER-LINE

REACTIONS AT WL 43.00 F.S. 232.68

$P_x = T \cos 13° = 32.7\#$

$M_x = -6.3\, T \sin 13 + 31.2\, P_e + 24.22\, P_s = 2910$ IN-#

$P_y = T \sin 13° = 7.45\#$

$M_y = 6.3\, T \cos 13 - (5.77 + 4.68)\, P_e + (11.16 - 4.68)\, P_s = 255$ IN-#

$P_z = P_e + P_s = 100\#$ SHEAR

$M_z = -(7.00 + 25.4)(T \cos 13) + (4.68 + 4.4)\, T \sin 13° = -977$ IN-# BEND

WITH RESPECT TO THE INCLINED AXES

$P_{x'} = P_x \cos\theta + P_y \sin\theta = 32.4\#$ CAUSES COMP

$M_{x'} = M_x \cos\theta + M_y \sin\theta = 2911$ IN-# " TORSION

$P_{y'} = P_y \cos\theta - P_x \sin\theta = 6.1\#$ SHEAR

$M_{y'} = M_y \cos\theta - M_x \sin\theta = 137$ IN-# BEND

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FUSELAGE LOADS.

VERTICAL SHEAR FLOW CAUSED BY BENDING STRESSES

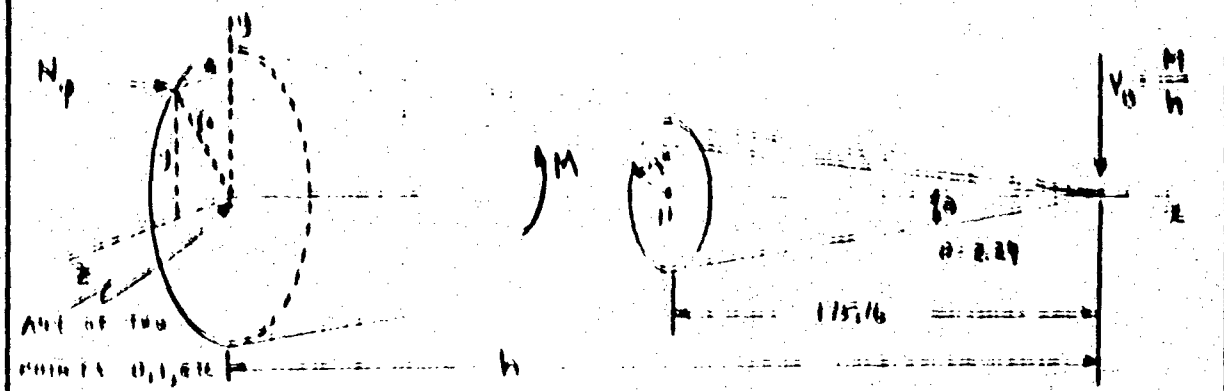

FIGURE 27

LET N_ϕ = BENDING STRESS, LB./IN., CAUSED BY A MOMENT M. IF THE BENDING STRESSES ARE ASSUMED TO ACT ALONG THE GENERATRI OF THE CONE, THEN THESE BENDING STRESSES ACT THROUGH A COMMON POINT, THE APEX OF THE CONE. THE STRESSES N_ϕ, WHOSE RESULTANT AT THE Y-Z PLANE IS A FORCE V_0 AND A COUPLE M, CAN BE RESOLVED AT THE APEX OF THE CONE INTO A SINGLE FORCE V_0 WHOSE VALUE MUST BE

$$V_0 = \frac{M}{h}$$

AND WHOSE DIRECTION IS DOWN WHEN THE MOMENT M CAUSES COMPRESSION ON THE UPPER SURFACE.

THE VALUES OF h AND $1/h$ ARE GIVEN BELOW.

Table XXII

POINT	0	1	2	3	4	5	6	7	8	9	10	11
h, IN.	302.6	296.2	291.5	279.7	258.0	245.2	233.5	219.8	207.0	194.3	184.5	175.7
$1/h$, IN.	.0033	.00341	.00343	.00359	.00387	.00407	.00430	.00455	.00483	.00514	.00551	.00570

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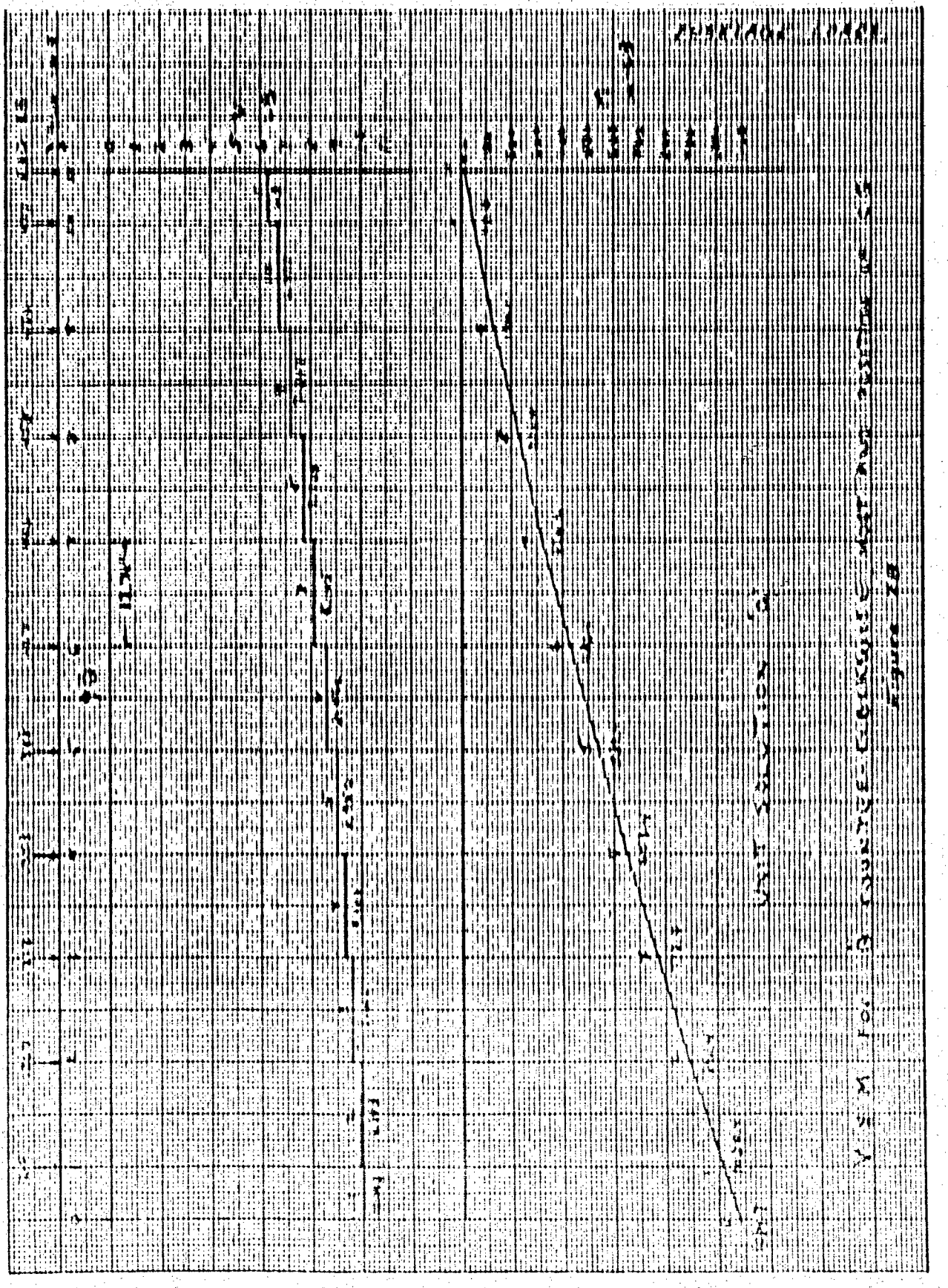

Figure 28

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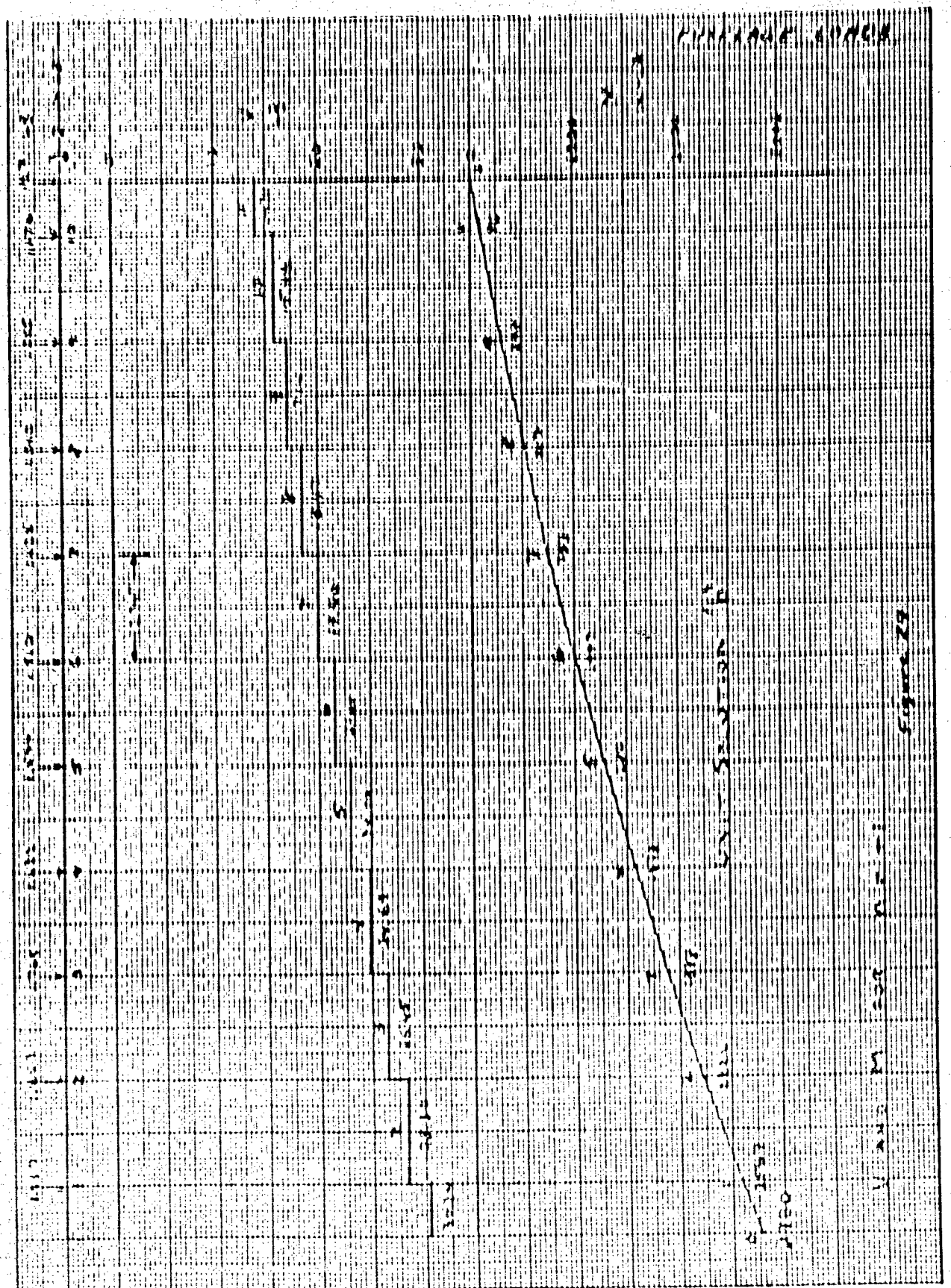

Figure 29

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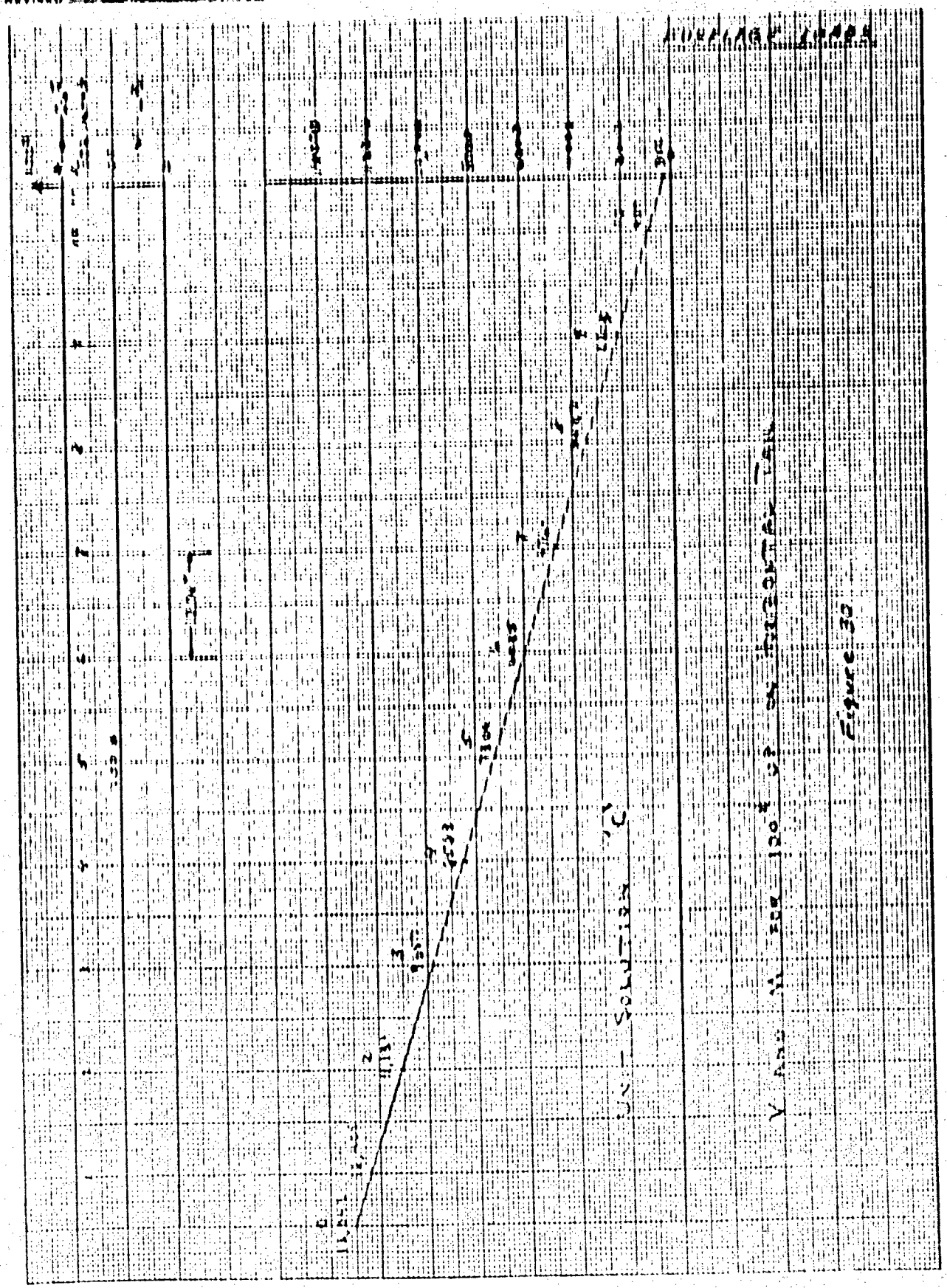

Figure 32

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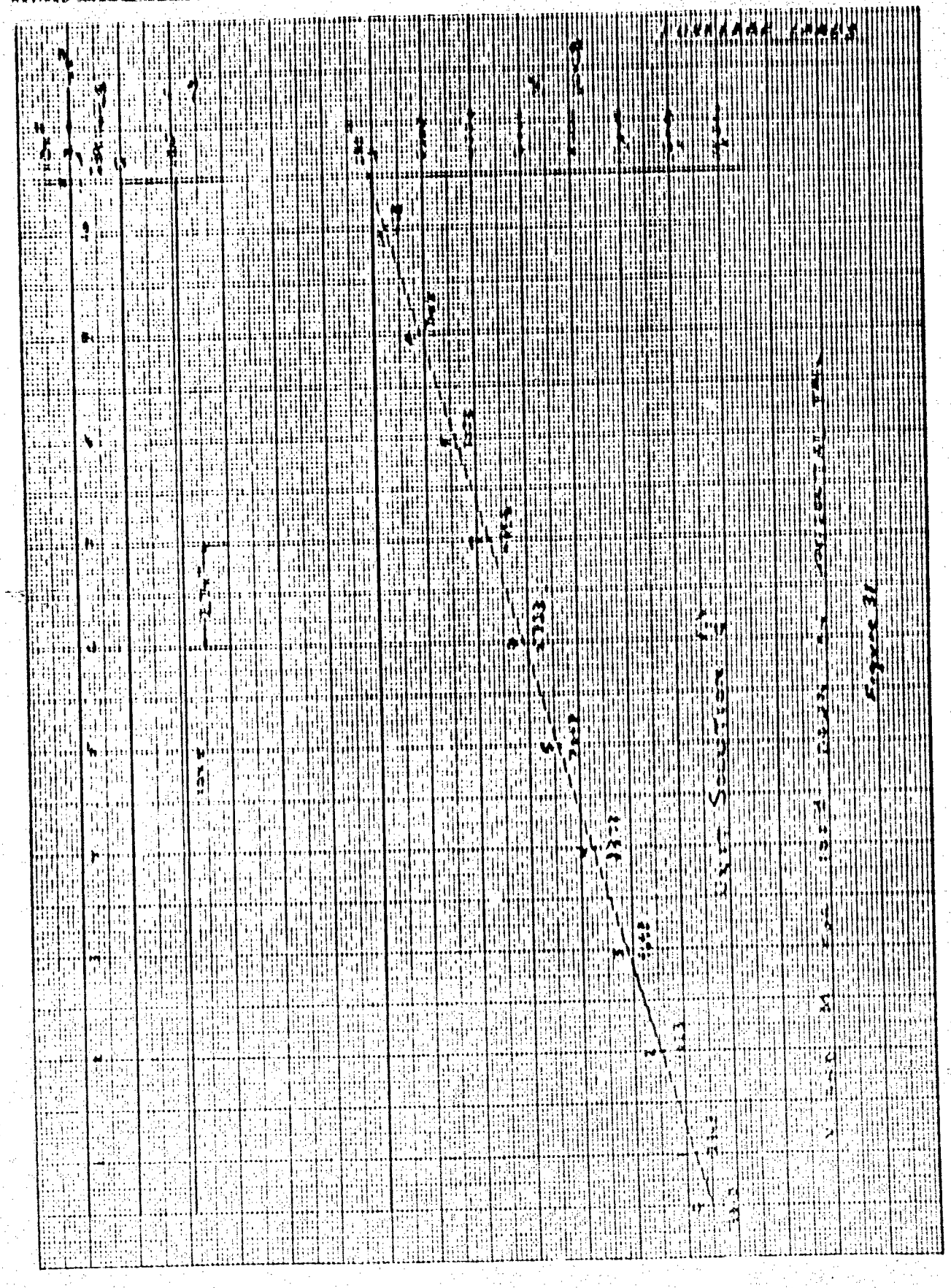

Figure 31

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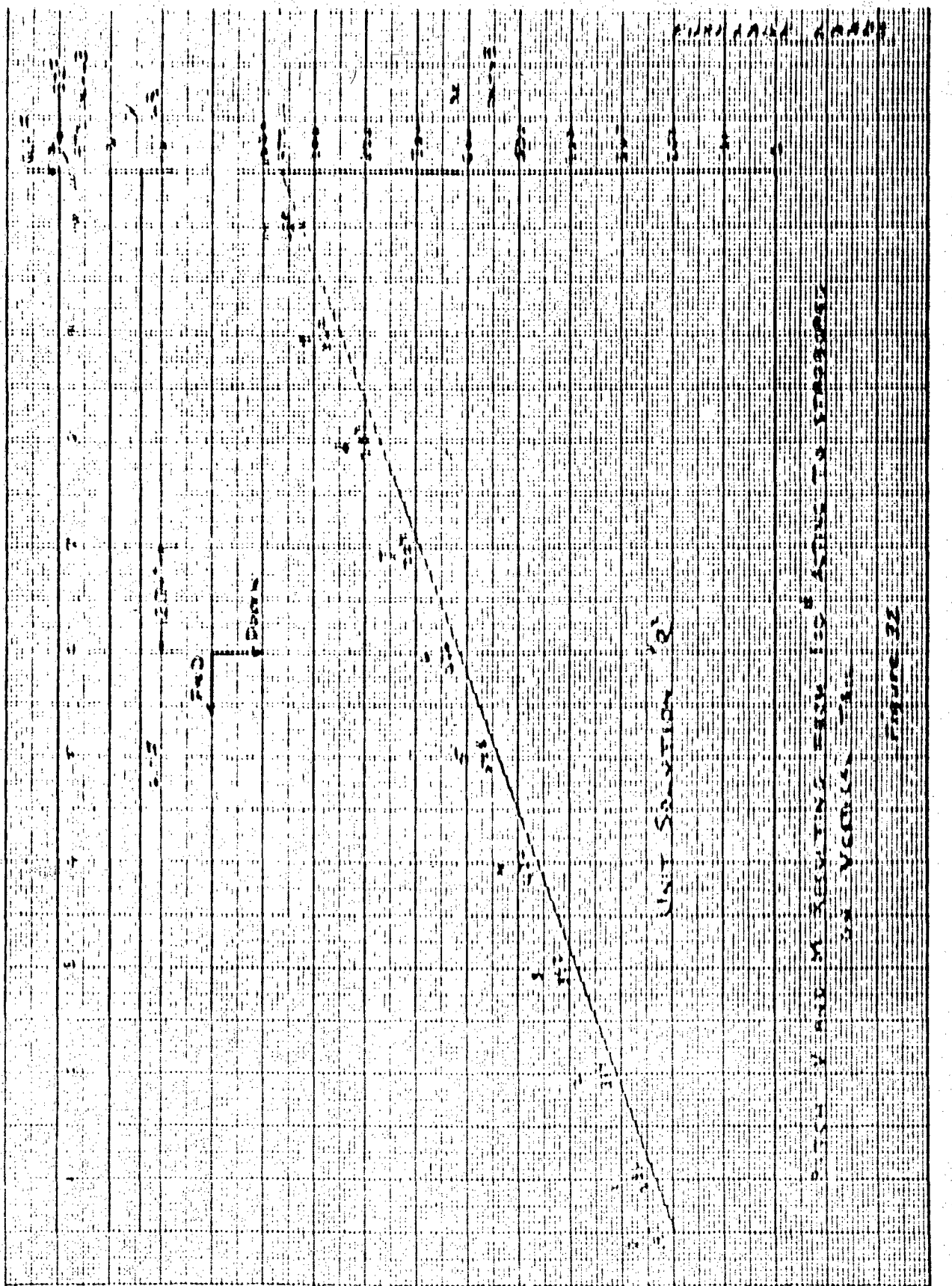

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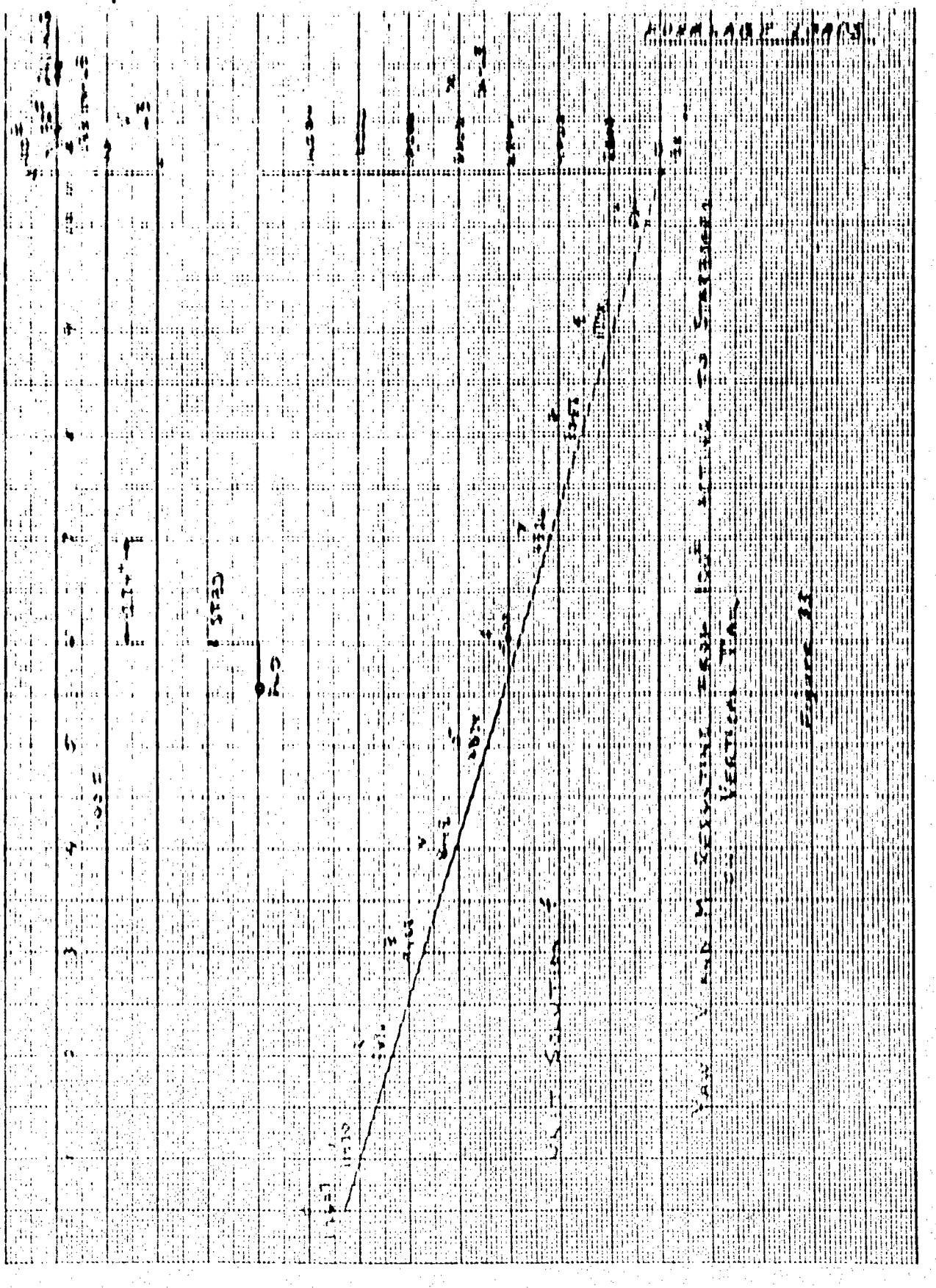

FUSELAGE LOADS

CRITICAL CONDITIONS

From the summary of airloads the most critical conditions are A2, A4, A6, and A7; B6 and B7; C7; E2; F2, F4, F6, F7, and F11 + horizontal tail load of F13. Conditions 3 and 5 are not critical because engine thrust loads up the tail cable.

As the unit solutions are labeled (a) through (f) then these solutions may be combined according to the equations

$$V \text{ or } M = -a\theta + nb + P/100\ c \qquad \text{For conditions 2, 4, 6, and 7}$$

$$V \text{ or } M = -a\theta + nb - P/100\ d \qquad \text{For conditions 3, 5, and 7}$$

The combined condition is given by the vector sum of pitching and yawing shears and moments and the superposition of the torque.

$$\left[V \text{ or } M\right]_{pitch} = nb - \frac{P}{100}\,d + \frac{P_{yaw}}{100}\,e$$

$$\left[V \text{ or } M\right]_{yaw} = -a\theta + \frac{P_{yaw}}{100}\,f$$

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SHEET 6A 4/68
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FUSELAGE LOADS

Table XVIII

Reference Point	Values from Unit Solutions a	b	c or d	Loading Quantities $\dot{\theta}$	γ_1	P	V LB	M_1 in -LB	Condition	
11	-6.3	-147	100				+11.8			
0	-10.1	-30.2	100	-1.83	+2.5	+60	-34		A2	
11	-9.0	-2	302					+159		
0	-10.85	-2760	+130.2					-853		
11							30.6			
0	"	"	"	0	2	60	-14		A7	
11								177		
0								2365		
11							+34.2			
0	"	"	"	0	2.5	71	-4.2		A6	
11								+209		
0								+2300		
11							+40.6			
0	"	"	"	0	2	71	+11.4		A7	PLUS TORQUE
11								+210		
0								-3690		
11							42.9			
0	"	"	"	0	2.5	80	4.3		B6	
11								236		
0								3490		
11							52.6			
0	"	"	"	0	2	80	19.6		B7	PLUS TORQUE
11								238		
0								4580		

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FUSELAGE LOADS

Reference Point	Values from Unit Solutions: a	b	c or c	Loading Quantities: b	n	P	V	M	Condition	
							LB	IN-LB		
1	-[illegible]	-14.7	+[illegible]				+[illegible]			
0	-10.1	-30.2	+100	0	2	70	-9.6		C7	
1	-9.6	-[illegible]	+302					+[illegible]		
0	-[illegible]	-[illegible]	+[illegible]					-[illegible]		
1							+[illegible]			
0	.	.	.	-2	-1	114	+124.2		F2	-[illegible]
1								+[illegible]		[illegible]
0								+[illegible]		
1							+[illegible]			
0	.	.	.	-2	0	114	+93.8		F4	-[illegible]
1								+[illegible]		[illegible]
0								+[illegible]		
1							+34.7			
0	.	.	.	0	-1	70	+[illegible]		F6	
1								-[illegible]		
0								+[illegible]		
1			-104				-[illegible]			
0	.	.	-104			-114	-98.5		F5	+[illegible]
1			+[illegible]					+[illegible]		[illegible]
0			-[illegible]					-[illegible]		
1							-[illegible]			
0	.	.	.	0	0	-70	-[illegible]		F7	-[illegible]
1								+[illegible]		[illegible]
0								-[illegible]		

FUSELAGE LOADS

COMBINED CONDITION FH, USING CONDITION FH FOR VERTICAL FAIL LOAD

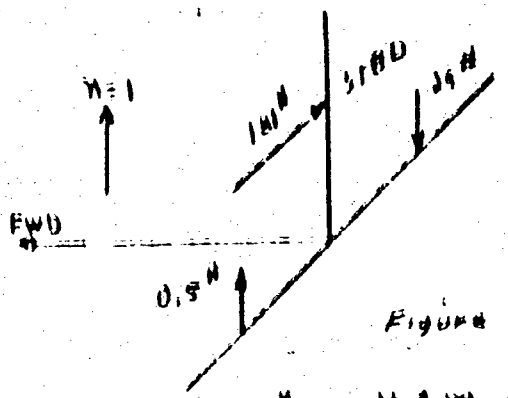

Figure 34

$$\text{Pitch V \& M} = nb = \frac{P_{UP}}{100}d + \frac{P_{SIDE}}{100}e$$

$$= b = \frac{(24-.5)}{100}d + 1.81e$$

$$= b = .285d + 1.81e$$

$$\text{Yaw V \& M} = -a\ddot{\gamma} + \frac{P_{SIDE}}{100}f$$

$$= -a\ddot{\gamma} + 1.81f$$

ACTUALLY THE VALUES 'a' ARE FOR $\ddot{\theta}$ AND NOT $\ddot{\gamma}$ BUT THE ERROR MADE IN USING 'a' IS INSIGNIFICANT

γ MAY BE CALCULATED FROM THE YAWING MOMENT OF THE 181# SIDE LOAD AND THE YAWING MOMENT OF INERTIA BOTH ABOUT AN AXIS OF YAW THROUGH THE CENTROID OF THE AIRCRAFT. THE PRINCIPAL AXES OF PITCH AND YAW ARE INCLINED AT ABOUT 20° WITH THE GEOMETRIC AXES SO THAT A YAWING MOMENT ABOUT THE GEOMETRICAL AXIS ACTUALLY CAUSES THE AIRCRAFT TO PITCH AND YAW BOTH ABOUT THE PRINCIPAL AXES.

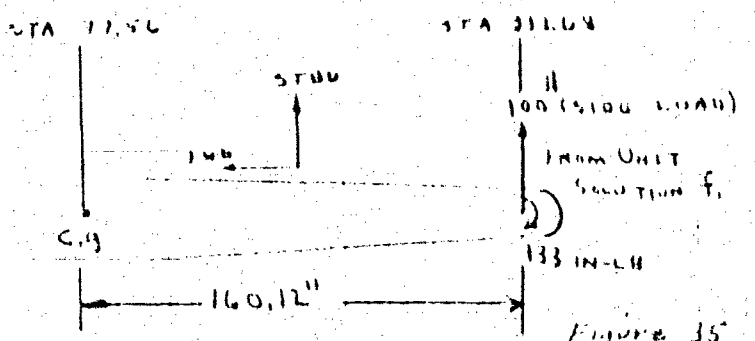

Figure 35

$$M_{YAW} = \left(100\,\frac{160.12}{12} - \frac{133}{12}\right)\frac{181}{100}$$

$$= 2390 \text{ FT-LB}$$

$$\ddot{\gamma} = \frac{M_{YAW}}{I_{YAW}} = \frac{2390}{300.8} = 8 \text{ rad/sec}^2$$

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PITCH V AND M

1	2	3	4	5	6	7	8	9	10	11	
-30.24	-28.3	-26.41	-24.64	-23.00	-21.44	-19.92	-18.47	-17.14	-15.88	-14.71	= wb = b
-29.7	-29.7	-29.7	-29.7	-29.7	-29.7	-29.7	-29.7	-29.7	-29.7	-29.7	= $-\frac{P_{cr}}{100}d = -29.7$
-11.1	-11.1	-11.1	-11.1	-11.1	-11.1	-11.1	-11.1	-11.1	-11.1	-11.1	= $-\frac{P_{cr}}{100}e = 1.3e = -11.1$
-71.0	-69.1	-67.2	-65.0	-63.3	-62.2	-60.7	-59.2	-57.9	-56.7	-55.5	= V LB

0	1	2	3	4	5	6	7	8	9	10	11	
-2780	-2540	-2330	-1890	-1570	-1280	-1010	-753	-520	-300	-96	-2	= wb = b
-3740	-3520	-3140	-2760	-2390	-2010	-1630	-1257	-880	-500	-121	-63	= $-.297d$
+1360	+1430	+1570	+710	+850	+1000	+1130	+1270	+1420	+1560	+1700	+1770	= 1.81e
-6160	-5680	-4800	-3940	-3110	-2290	-1510	-740	+20	+760	+1517	+1700	= M_p IN-LB

YAW V AND M

1	2	3	4	5	6	7	8	9	10	11	
-80.6	-79.0	-76.8	-74.3	-71.5	-68.7	-65.1	-61.5	-58.0	-54.2	-52.4	= $-a\ddot{y} = -8a$
+181	181	181	181	181	181	181	181	181	181	181	= 1.81 f
100.4	102	104.2	106.7	109.5	112.3	115.3	119.5	123	126.8	130.6	= V

0	1	2	3	4	5	6	7	8	9	10	11	
-8620	-8100	-7100	-6130	-5190	-4280	-3620	-2570	-1810	-1080	-391	-76	= $-a\ddot{r} = -8a$
22800	21600	19330	17030	14730	12430	10130	7820	5520	3220	912	-240	= 1.81 f
14180	13500	12230	10,900	9540	8150	6710	5150	3710	2140	516	-316	= M_r IN-LB
15400	14620	13130	11570	10050	8400	6880	5340	3710	2270	1600	1732	= $M = \sqrt{M_p^2 + M_r^2}$ IN-LB

FUSELAGE LOADS

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DATE 2.06.61
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HUBBARD AGARD

CORRECTION OF BEAM SHEAR FOR TAPER - PITCH

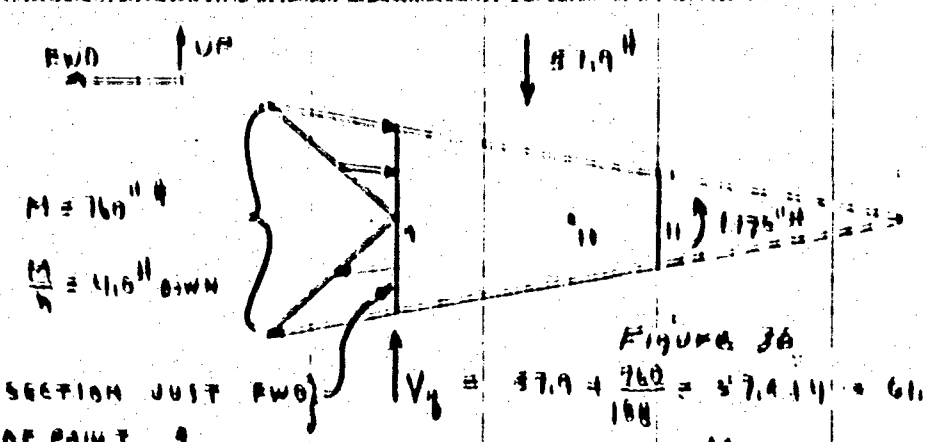

Figure 36

SECTION JUST FWD OF POINT 9

$V_g = 57.9 + \frac{760}{18H} = 57.9 + 4 = 61.9^{\#}$ UP

Thus $V_g = V_{DIAG} - \frac{M}{h} = 57.9 - 4 = 61.9^{\#}$ UP

For a section just fwd of point 6

$$V_g = -62.2 - \frac{1510}{23.6} = -62.2 + 6.7 = -55.4\ ^{\#}\ \text{UP.}$$

	0	1	2	3	4	5	6	7	8	9	10	11	
	-11.0	-11.0	-64.1	-67.1	-65.1	-64.9	-61.2	-60.9	-59.1	-57.9	-56.7	-55.4	V
	30.8	19.6	17.3	14.9	13.4	9.6	6.7	3.9	-.1	-4.0	-8.1	-10.1	$-\frac{M}{h}$
	-30.3	-31.4	-81.8	-82.2	-78.1	-84.2	-55.5	-57.2	-59.3	-81.9	-64.2	-65.6	V_g

CORRECTION OF BEAM SHEAR FOR TAPER - YAW

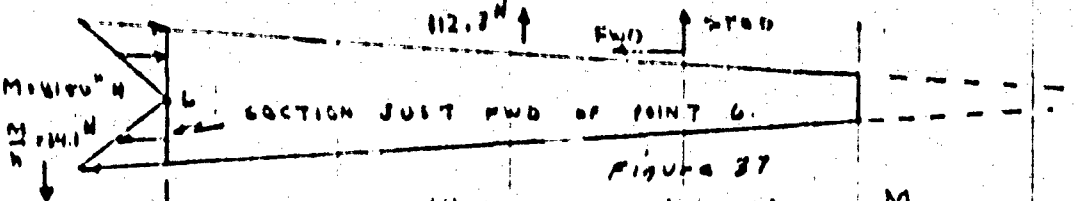

Figure 37

$V_g = 112.3 - \frac{6480}{236} = 81.8\ ^{\#} = V_{DIAG} - \frac{M}{h}$

	0	1	2	3	4	5	6	7	8	9	10	11	
	100.4	100.4	102.3	104.2	106.7	109.5	112.3	115.9	119.5	123	126.8	130.6	V
	-46.0	-46.7	-44.2	-41.3	-38.0	-34.1	-30.5	-24.1	-18.6	-11.4	-7.9	+1.9	$-\frac{M}{h}$
	52.4	53.7	57.5	62.9	68.7	75.4	81.8	91.8	100.9	111.6	123.1	132.5	V_g
$V = \sqrt{(V_g)^2_{PITCH} + (V_g)^2_{YAW}}$													
	72.4	74.5	77.1	81.8	86.3	93.0	99.0	108.2	117.0	127.5	140.0	141.0	

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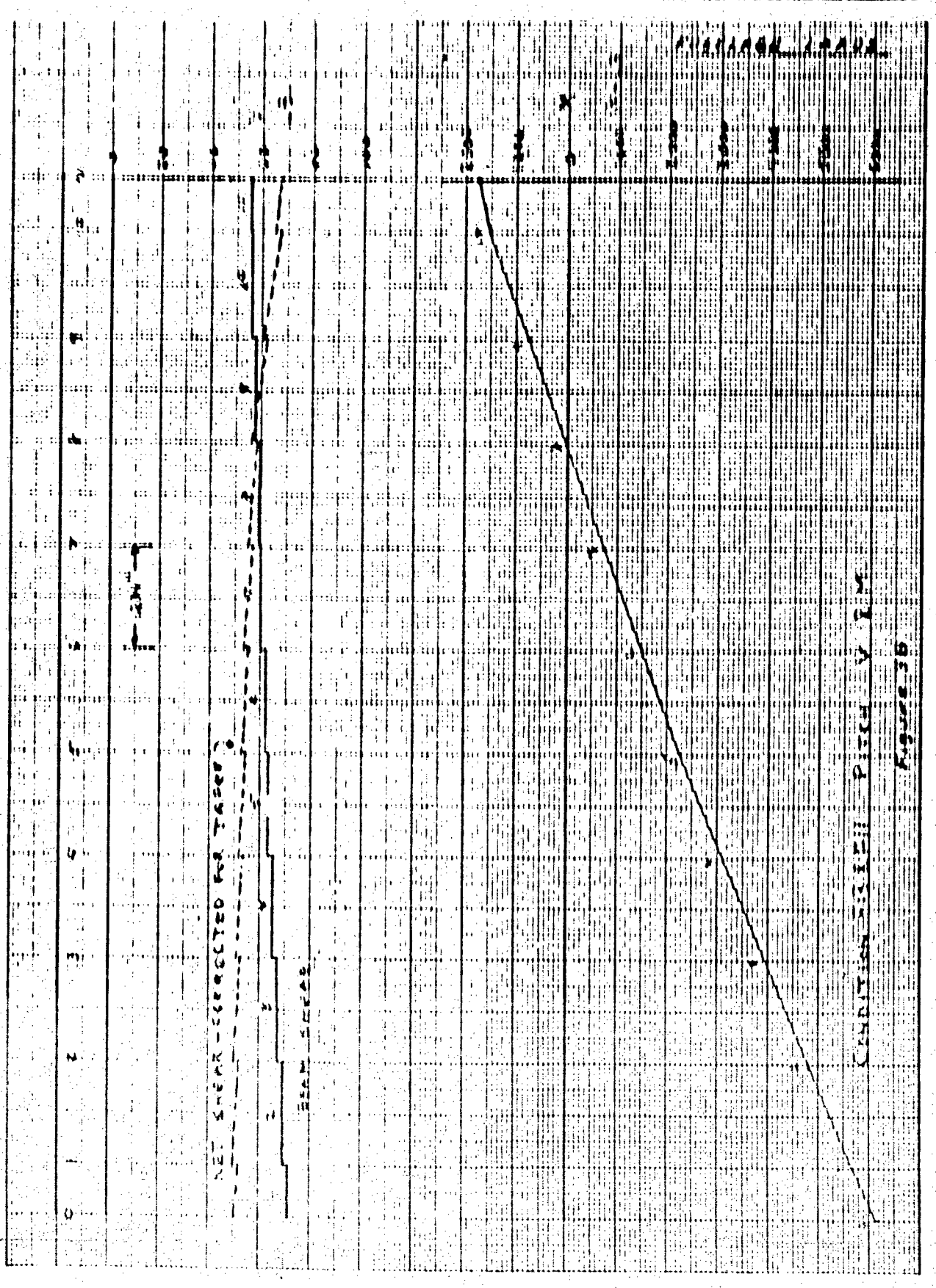

Figure 3B

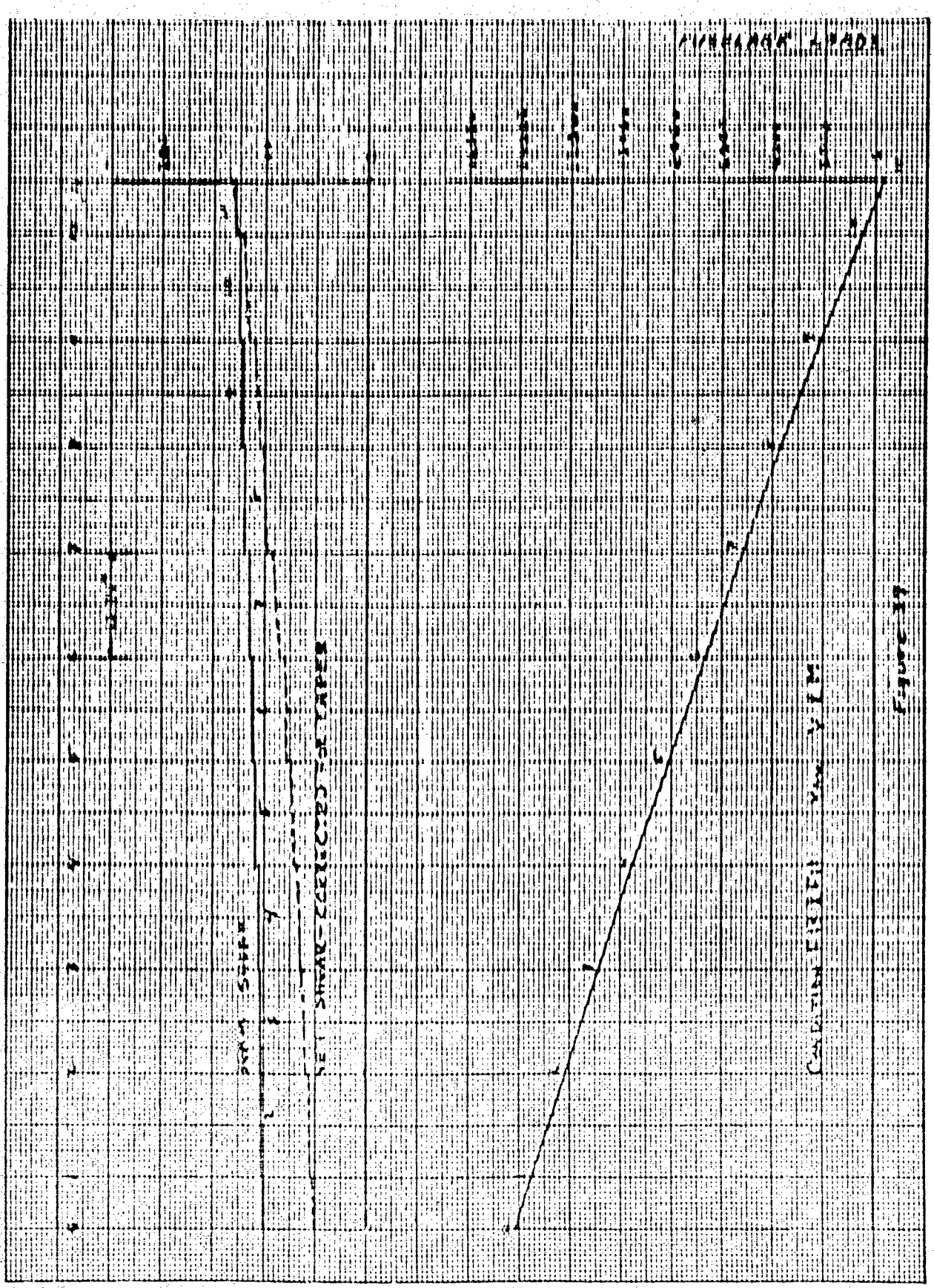

Figure 3

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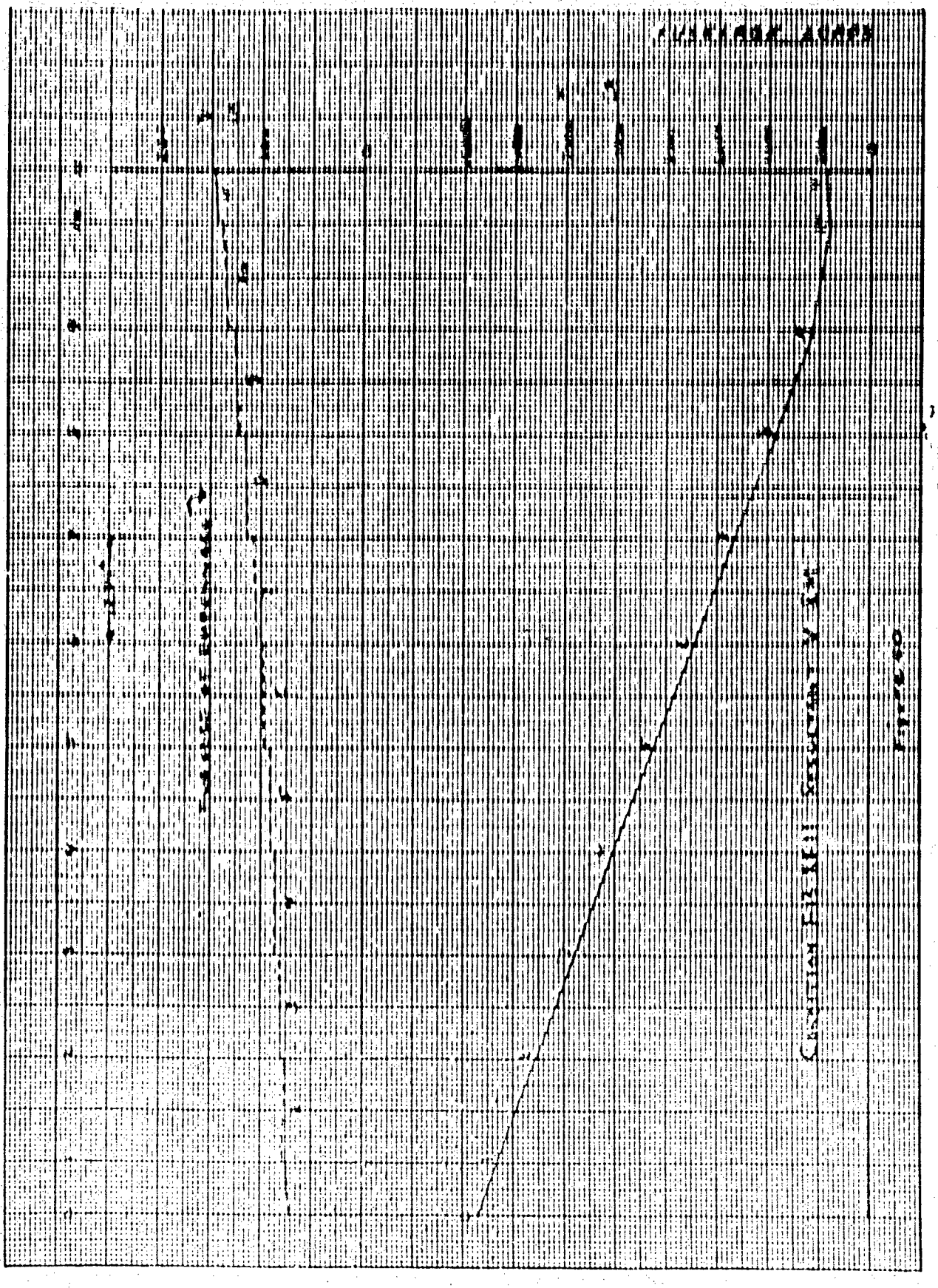


Figure 60

ENGINE MOUNT LOADS

REF. DWG 447A-007

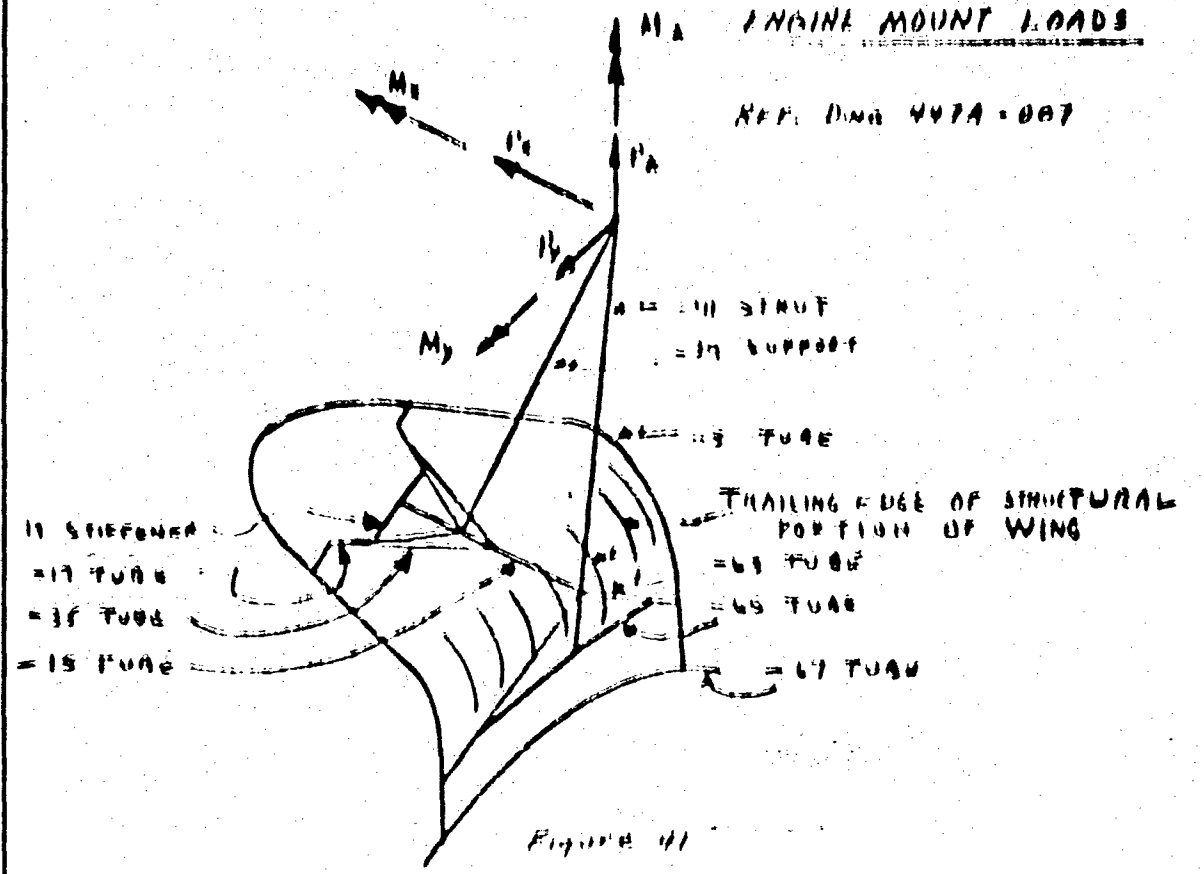


Figure 41

CRITICAL LOADS RESOLVED AT TOP OF TRUSS

Table XX

CONDITION	REF.	P_x LB	P_y LB	P_z LB	M_x IN-LB	M_y IN-LB	M_z IN-LB
$n_z = -1$ FLIGHT	PG. 2.00.010	915	0	-187	1512	-2625	0
$n_z = -3$ FLIGHT	PG. 2.00.030	230	0	-223	1512	-1856	0
$n_y = 1.33$ SIDE LOAD	REF. 7	230	-99	-74.5	1180	-1170	-456

CRITICAL ENGINE MOUNT LOADS

EMPENNAGE LOADS

The critical loading condition alone is presented as the stress analysis is a summary one. This load is 101 lb acting on the vertical tail. The distribution, using a 100 lb unit load, is that shown on page 2.05.070.

The hinge line reactions and geometry for the rudder and vertical stabilizer are given on pages 2.05.020 and 2.05.030. As the rudder attachment is statically determinate, the calculations are given on page 2.05.020. The support of the vertical stabilizer is statically indeterminate as is apparent from the equations below. (See page 2.05.030).

$$P_x = P_x - A_x - 32.2 + R_x = 0$$
$$P_y = -P_y - A_y - 7.45 + R_y = 0$$
$$P_z = P_z + A_z - 100 + R_z = 0$$

$$M_x = 24.1\ P_z + 30.6\ A_z - 32.2 \times 19.5 - 64.8 \times 10.22 = 0$$
$$M_y = 29\ P_z + R_z\ X_{Rz} - 64.8 \times 11.16 = 0$$
$$M_z = -24.1 P_x + 29 P_y + 30.6 A_x + 32.2 \times 22 - R_y\ X_{Ry} = 0$$

With direction cosines of page 2.05.030,

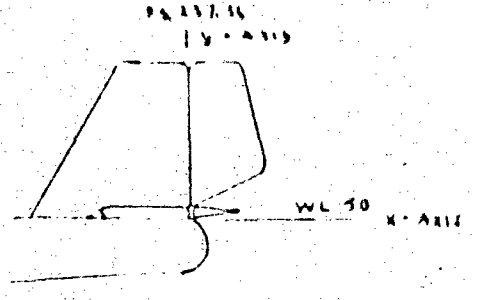
$.264P - .102A + R_x = 32.2$
$-.690P - .612A + R_y = 7.45$
$.675P + .782A + R_z = 100$
$16.23P + 23.7A = 1070$
$19.37P + X_{Rz} = 723$
$13.64P + 3.12A - X_{Ry} = -708.4$

These are plotted on Pg. 2.05.040

Solution when $X_{Rz} = X_{Ry}$

$P = .0308A = 2.92$ lb
$A = 76.7$ lb
$R_x = 32.3$ lb
$R_y = 55.9$ lb
$R_z = 38.5$ lb
$X_{Ry} = X_{Rz} = 17.6$ In.
$P_z = 1.97$ lb $\quad A_y = 47.0$ lb
$P_y = 2.01$ lb $\quad A_z = 60.0$ lb

The shear and moment diagrams are on page 2.05.050.



SKETCH SHOWING ORIENTATION OF AXES

EMPENNAGE LOADS

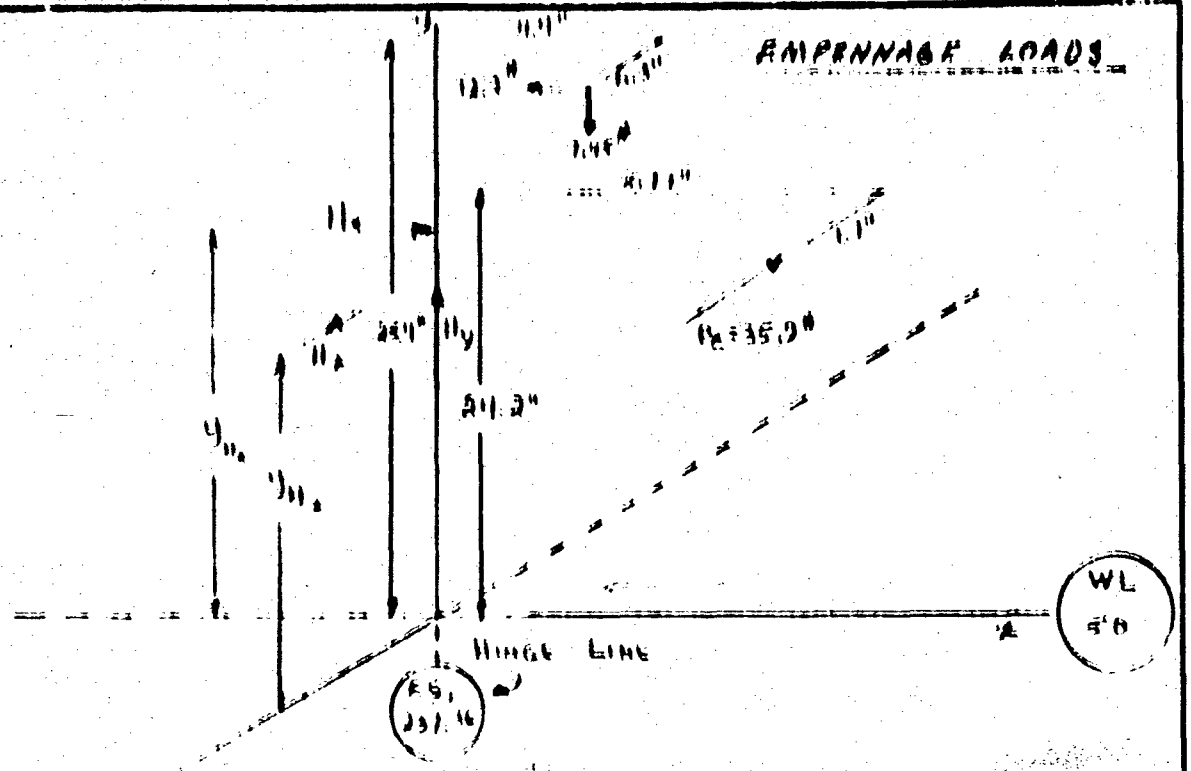


Figure 42

$\Sigma M_x = 35.2\, y_{H_x} - 35.2(24.2) + 7.45(6.3) = 0 \qquad y_{H_x} = 22.4"$

$\Sigma M_y = -32.2 \times 6.3 + 35.2 \times 5.77 = 0$ [illegible]

$\Sigma M_z = -32.2\, y_{H_z} + 32.2(25.4) - 7.45(4.4) = 0 \qquad y_{H_z} = 24.4"$

HINGE LINE REACTION FOR 100 LB ON VERTICAL TAIL

EMPENNAGE LOADS

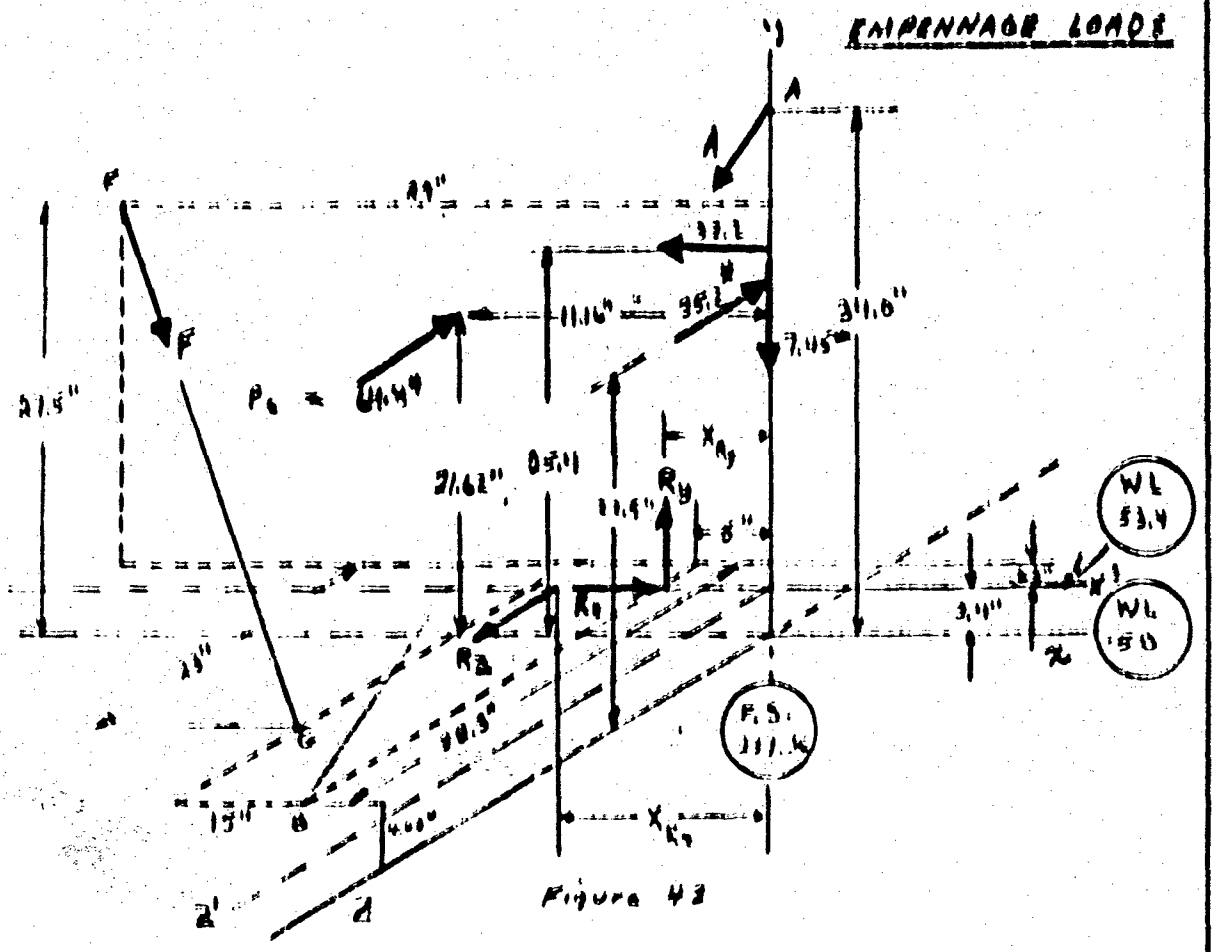


Figure 43

LOADS AND REACTIONS ON VERTICAL STABILIZER FOR 100 LB. SIDE LOAD.

DIRECTION NUMBERS AND COSINES FOR CABLE TENSIONS F AND A.

$X_F = 29 - 5 - 15 = 9"$ $\qquad$ $X_A = 5"$

$y_F = 27 - 3.4 - .63 = 23.5"$ $\qquad$ $y_A = 34 - 4.03 = 30"$

$z_F = 23"$ $\qquad$ $z_A = 38.3"$

$FG = \sqrt{9^2 + 23.5^2 + 23^2} = 34.1"$ $\qquad$ $AB = \sqrt{5^2 + 30^2 + 38.3^2} = 49.0"$

$F_x = \frac{9}{34.1} F = .274 F$ $\qquad$ $A_x = \frac{5}{49} A = .102 A$

$F_y = \frac{23.5}{34.1} F = .690 F$ $\qquad$ $A_y = \frac{30}{49} A = .612 A$

$F_z = \frac{23}{34.1} F = .675 F$ $\qquad$ $A_z = \frac{38.3}{49} A = .782 A$

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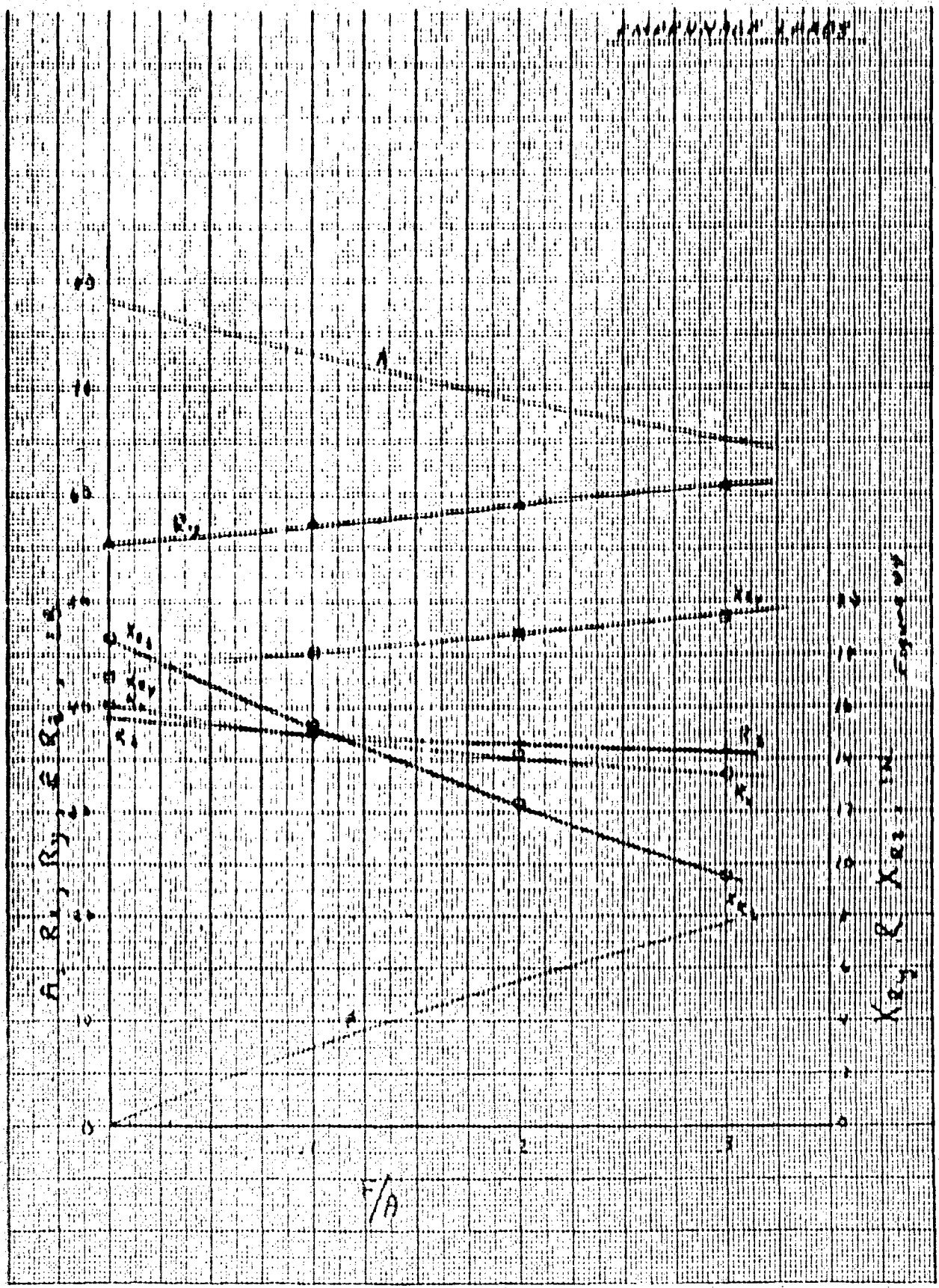

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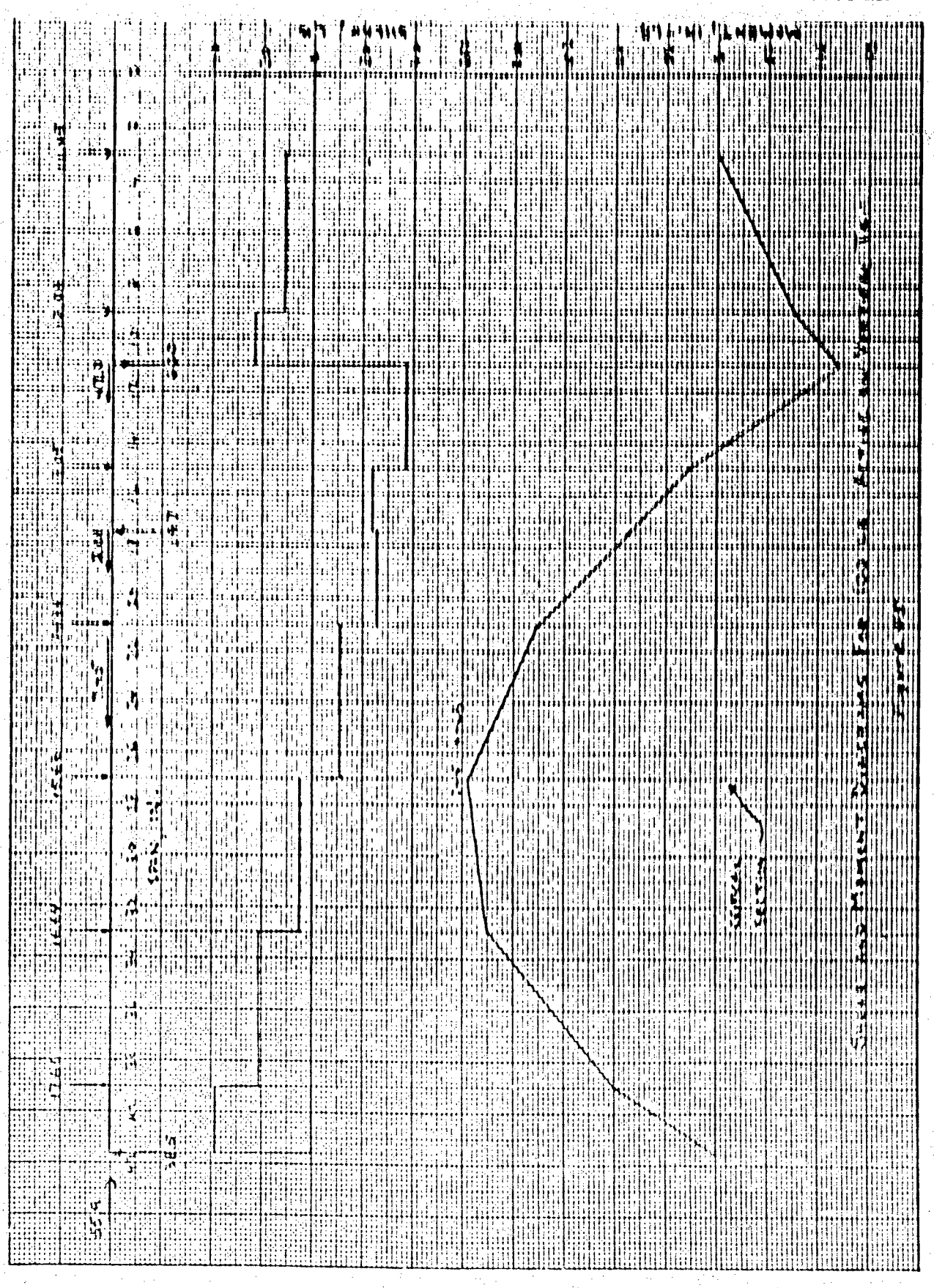


EMPENNAGE LOADS

TORQUE AT THE CRITICAL SECTION IS CALCULATED BELOW

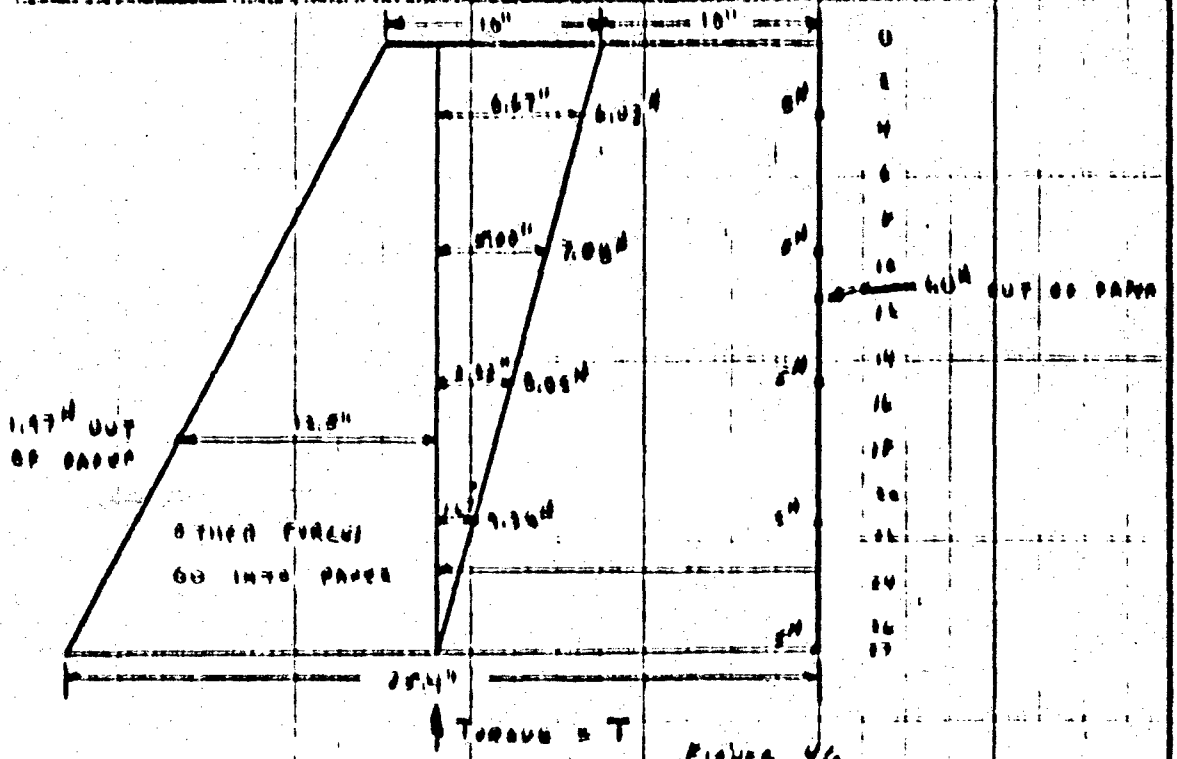


Figure 46

$T = 60 \times 17.7 - 1.97 \times 12.5 - 6.67 \times 6.03 - 5.00 \times 7.08 - 3.33 \times 8.05 - 1.67 \times 9.38 - 17.7 \times 25$
$= 477$ IN-LB

CORRECTION FOR INERTIA OF VERTICAL TAIL

THE VERTICAL TAIL WEIGHS 5.8 LB AND ITS CENTROID IS AT WL 72.3, STA 230.2 (PG 2.01.070). THE INERTIA LOAD DUE YAWING ACCELERATION IS:

$$P = \frac{W}{g} \times \ddot{\gamma} = \frac{5.8}{72.2}(13.13)(8) = 18.9 \text{ LB}$$

WHERE $x = (230.2 - 72.56) \div 12 = 13.13'$ } SEE PG 2.03.220
$\ddot{\gamma} = 8$ RAD/SEC^2

THUS THE ACTUAL TAIL LOAD OF 181# MAY BE REDUCED BY 18.9 LB TO 162 LB.

Cockpit Loads

The cockpit loads are:

(1) Pullout $n_z = 2.5$

(2) Pushover $n_z = 1.0$

(3) Yaw $n_z = 1$, $n_y = .675$

(4) Roll $n_z = 1$, $M_{roll} = 730$ in-lb, $M_{yaw} = 1537$ in-lb.

(5) Level Landing, inclined reactions

$$n_z = \frac{1347}{550} - \frac{1}{3} + 1 = 3.12$$

$$n_x = -\frac{371}{550} = -.675$$

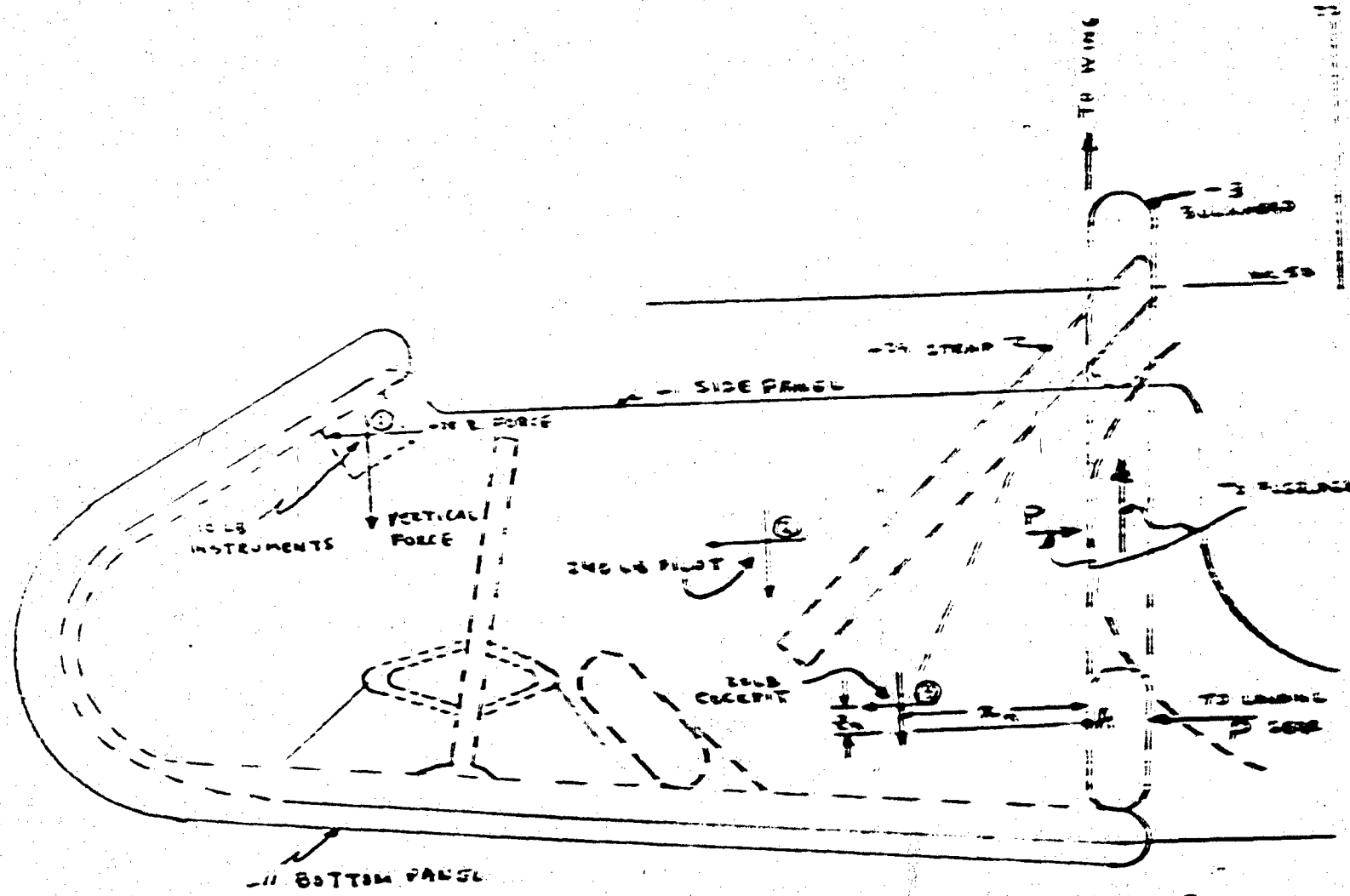
(6) Tail Down Landing $n_z = 3.12$

(7) Ground Load $n_z = 1.33$, $n_y = .83$

Conditions (1) - (4) are flight conditions and are partially derived from Table III, page 2.00.030.

Conditions (5) - (7) are derived from CAM -03, Art. 3.243 to 3.249.

The most critical loads are (1), (5), and (6).



COCKPIT AND BULKHEAD GAC DWG 447A-005

Figure #7

COCKPIT LOADS

POSITION OF LOADS (1), (2), AND (3) SEE COCKPIT AND BULKHEAD SKETCH. THE POSITIONS ARE MEASURED FROM A POINT 'A' ON THE LINE OF ACTION OF THE REACTION BETWEEN THE LANDING GEAR AND THE BOTTOM AND SIDE PANELS OF THE COCKPIT. ACTUALLY POINT 'A' IS DEPENDENT UPON THE LOAD FACTOR, HOWEVER AN UPPERMOST POSITION WAS DETERMINED BY CALCULATIONS USING STATIC TEST RESULTS

POSITION OF LOADS

COORDINATE	INSTRUMENTS (1)	PILOT (2)	COCKPIT (3)
X	37"	16.2"	9.8"
Z	17"	10.6"	1.6"

Table XXI

INERTIA LOADS

CONDITION	LOAD (1); 10# INSTRU.		LOAD (2); 240# PILOT		LOAD (3); 20# COCKPIT	
	HORIZONTAL 10 n_x, LB	VERTICAL 10 n_z, LB	HORIZONTAL 240 n_x, LB	VERTICAL 240 n_z, LB	HORIZONTAL 20 n_x, LB	VERTICAL 20 n_z, LB
1	0	25	0	600	0	50
5	6.75	31.2	16.2	748	13.5	62.4
6	0	31.2	0	748	0	62.4

Table XXII

MOMENTS AT LIMIT LOADS

CONDITION	(1)		(2)		(3)		SUM
	HORIZ IN-LB	VERT. IN-LB	HORIZ IN-LB	VERT. IN-LB	HORIZ IN-LB	VERT IN-LB	IN-LB
1	0	925	0	9720	0	480	11,125
5	115	1155	1720	12160	22	611	15,773
6	0	1155	0	12150	0	611	13,916

MOMENT OF HORIZ. FORCE IS HORIZ FORCE TIMES Z COORDINATE,
" " VERT " " VERT " " X "

218-83 (3-59)M

Landing Gear Loads

The CAM 3, reference 7, was used as a guide in the determination of landing loads for this aircraft. Article 3.243 of reference 7 gives a minimum descending velocity of 7 ft/sec. However, preliminary investigation showed that such a minimum could not be achieved without an elaborate and heavy landing gear. It was found that the kinetic energy of a 5 ft/sec descending velocity could be absorbed by a single wheel attached to a piston deflecting into the fuselage. The load and energy vs. deflection characteristics of the piston-fuselage system are given in figures 48 and 49, while figures 50 and 51 are for the tire. See reference 8 for the piston-fuselage deflection characteristics.

The conditions examined with the limit loads are given below. One flight condition is included as the lower wing brace cables are attached to the landing gear.

TABLE XXIII

Condition	Limit Loads, Lb.		
	Vertical	Aft	Side
(1) Level Landing, Inclined Reactions	1347	371	0
(2) Tail-Down Landing	1347	0	0
(3) Side Load	732	0	457
(4) Flight Load, $n_z = 3$	1353/1.5 = 902		

LANDING GEAR LOADS

CALCULATED
LOAD - DEFLECTION CURVE
FOR GA-468
INFLATABLE
LANDING GEAR

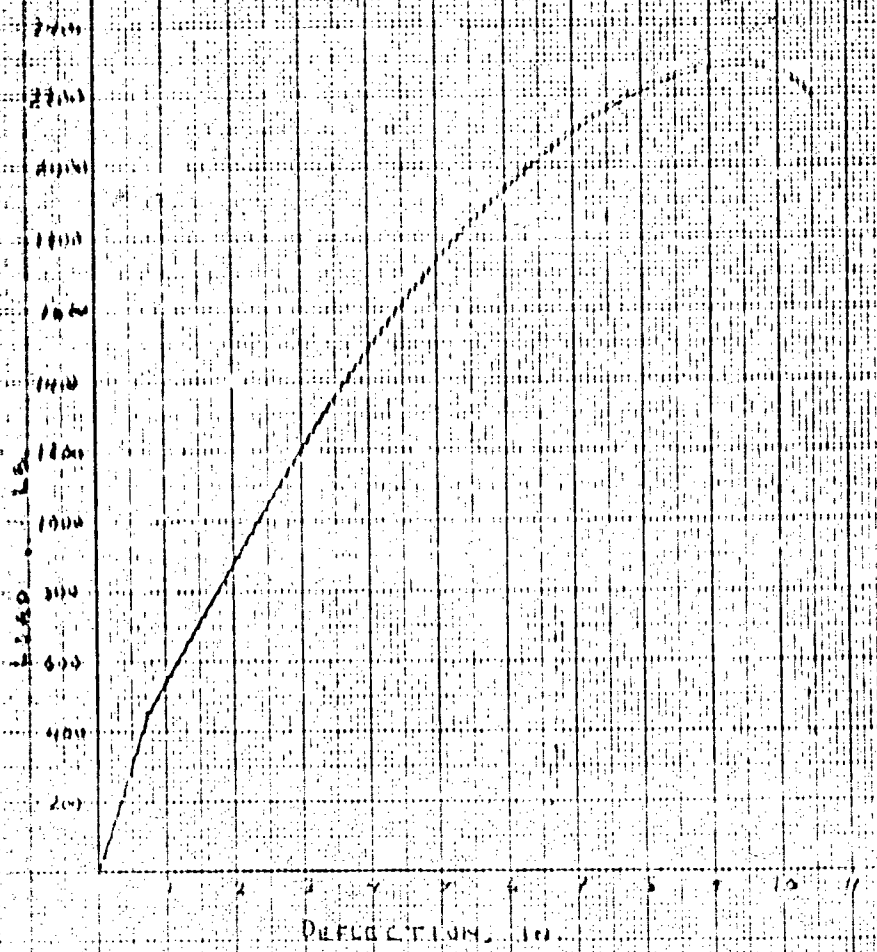


Figure 4B

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PAGE 2.02.030
MODEL GA 468
SER. 9861
REP. NO. G 9[illegible]

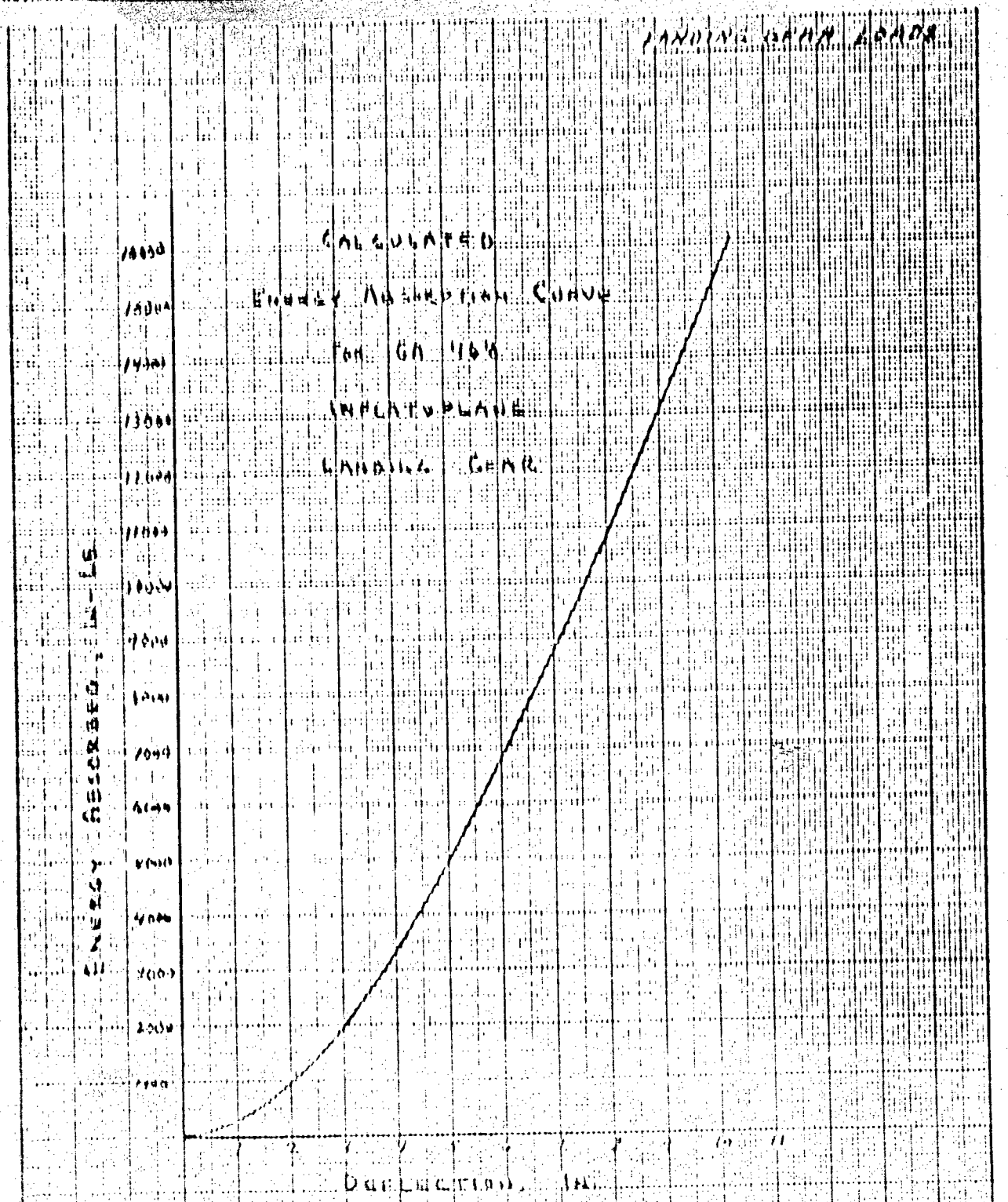

Figure [illegible]

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AKRON, OHIO

PAGE 2.02.040
MODEL GA-468
GER # 7861
REF NO.

LANDING GEAR LOADS

GOODYEAR 10.00 TYPE I PLAIN TREAD 8 PLY RATING NYLON

6 PLY AIRPLANE TIRE

INTIAL INFLATION 60 lbs/in²

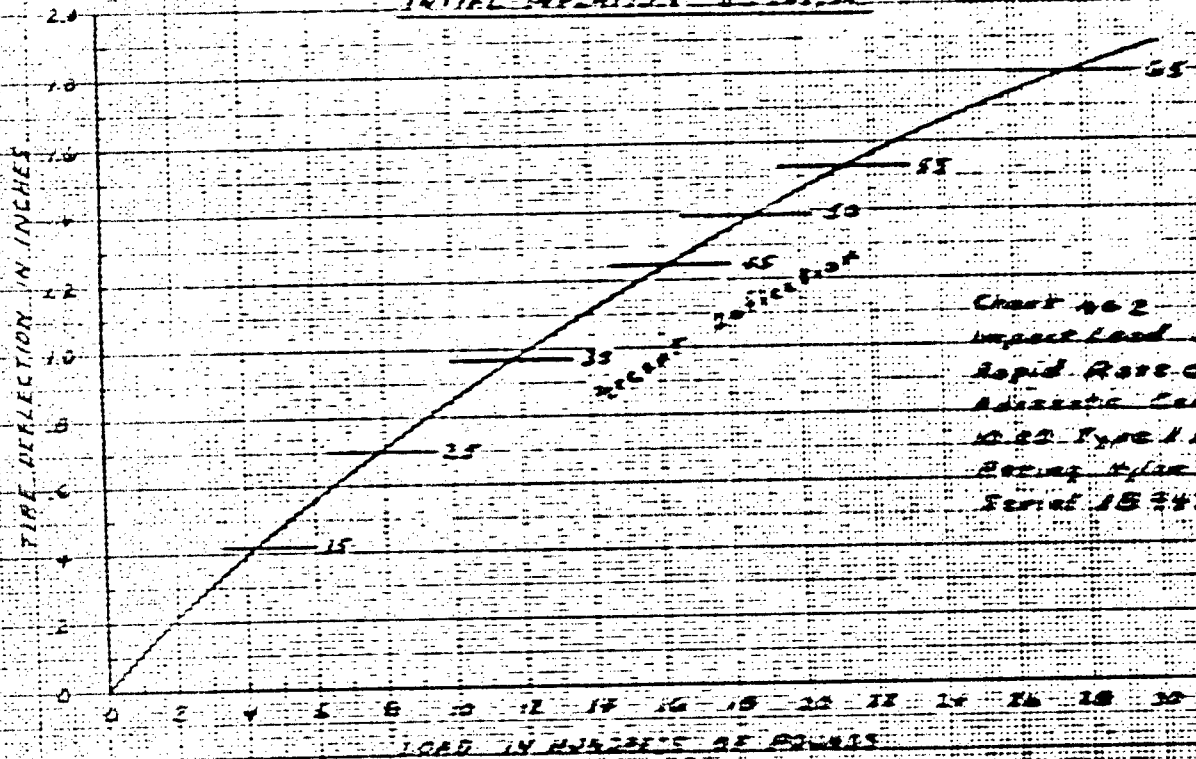

Figure 50

PREPARED BY JD

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DATE 1-18-61

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GOODYEAR AIRCRAFT CORPORATION

AKRON, OHIO

PAGE 2 OF 2

MODEL GA-748

SEQ 7261

REF NO.

LANDING LOAD GRAPH

GOODYEAR 19.00 TYPE I PLAIN TREAD, 22 PLY RATING NYLON

6.25" FLANGE TIRE

INITIAL INFLATION 60 PSIG

Cord Angle 34°

Impact Energy Definition Curve

Rapid Rate of Loading

Airborne Compression

19.00 Type I Plain Tread 22 Ply

Rating Nylon Cord

Serial 182172

FIGURE 21

TIRE DEFLECTION IN INCHES

2.0 1.8 1.6 1.4 1.2 1.0 .8 .6 .4 .2 0

0 2 4 6 8 10 12 14 16 18 20 22 24 26 28 30 32 34 36 38 40 42 44 46 48 50 52 54 56 58 60 62 64 66 68 70 72 74 76 78

WING ANALYSIS

Section 3

WING ANALYSIS

The wing is of Airmat construction and supports the external loads by virtue of tensile inflation stress. If one of the principal stresses at a point on the wing cross-section becomes zero due to the applied compressive stresses a wrinkle starts to form there. As more load is applied the wrinkle will enlarge to a point where collapse will occur. The pressure in the wing should be large enough to prevent a wrinkle from forming under limit loads and collapse at ultimate loads. For this analysis the ultimate load is 1.75 times limit load.

While the minimum principal stress at the point where a wrinkle first forms is zero, both principal stresses at a point on the opposite surface of the wing are tensile stresses. The allowable strength value on the tension side is the quick break value derived from a cylinder burst test divided by a creep rupture factor of 3. The factor of 3 accounts for the fact that fabric under load for a period of time has a reduction in strength.

The allowable hoop tension strength value on the compression side is the quick break value derived from the cylinder burst test times 0.65 divided by 3. The factor of 0.65 accounts for the fact that the fabric in a cylinder burst test is loaded at a 2-1 stress ratio, while on the compression side of the wing is loaded in only one direction and consequently gets little or no help from the bias plies. The factor is based on a comparison of cylinder burst tests and strip tensile tests.

Since the negative margins shown in Tables XXXII and XXXIII, pages 3.01.220 and 3.01.230 respectively, cover only a small percentage of the periphery of the wing cross-section shown on page 3.01.040, these negative margins are not serious. This is further proved in the wind tunnel test, which gave an ultimate margin of safety of +0.17, reference page 3.01.240.

WING ANALYSIS

Calculation of Bending Stress

Method of Analysis

For a cross-section in bending, the general equation for stresses due to bending may be determined from the following expression:

$$f_b = (C_1 M_z - C_2 M_x) z + (C_1 M_x - C_3 M_z) x$$

Where

$$C_1 = \frac{I_{xz}}{I_x I_z - I_{xz}^2}$$

$$C_2 = \frac{I_z}{I_x I_z - I_{xz}^2}$$

$$C_3 = \frac{I_x}{I_x I_z - I_{xz}^2}$$

I_x = Moment of inertia of cross-section about x-x axis
I_z = Moment of inertia of cross-section about z-z axis
M_x = Bending moment about x-x axis, positive when it causes compression above x-x axis
M_z = Bending moment about z-z axis, positive when it causes compression to right of z-z axis

x and z are the ordinates of the elements of the cross-section. x is positive above the x-x axis and z is positive to the right of the z-z axis.

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Section Properties

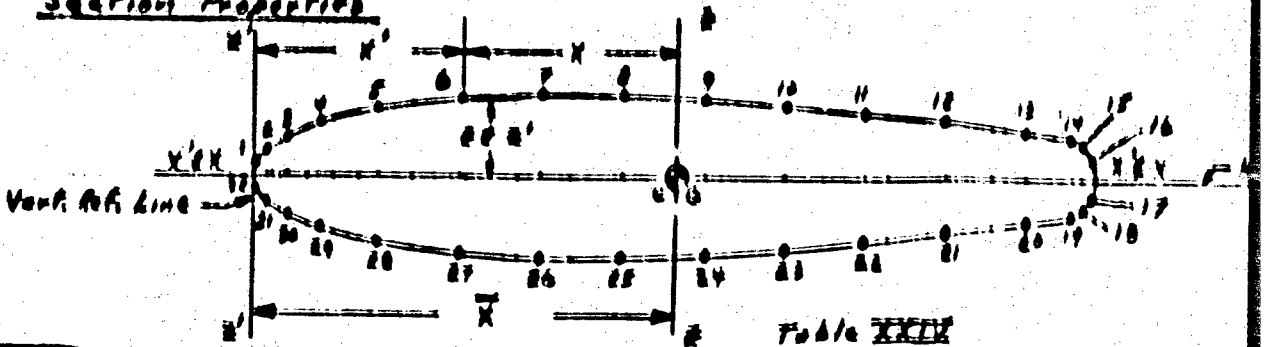

Table XXIV

Col.	①	②	③	④	⑤	⑥	⑦	⑧
Element	Area	z'	Az'	Az'^2	x'	Ax'	Ax'^2	$Ax'z'$
Ref.			①×②	②×③		①×⑤	⑤×⑥	⑥×⑤
1	0.85	0.53	0.4505	0.2388	0.11	0.0935	0.0103	0.0496
2	1.55	1.48	2.294	3.3951	0.82	1.271	1.342	1.8811
3	1.35	2.20	2.970	6.534	1.96	2.646	5.186	5.8212
4	2.40	2.90	6.960	20.184	3.68	8.832	32.502	25.6128
5	4.70	3.71	17.437	64.691	7.13	33.511	238.933	124.326
6	4.56	4.28	19.517	83.533	11.72	53.443	626.352	228.739
7	4.61	4.50	20.745	93.353	16.31	75.197	1226.33	338.351
8	4.61	4.47	20.607	92.113	20.90	96.347	2013.69	430.686
9	4.59	4.25	19.508	82.909	25.50	117.045	2984.65	497.454
10	4.58	3.96	18.137	71.823	30.09	137.812	4146.76	545.742
11	4.62	3.58	16.540	59.213	34.68	160.222	5556.50	573.607
12	4.61	3.07	14.153	43.450	39.27	181.035	7109.24	555.793
13	4.65	2.49	11.579	28.832	43.86	203.949	8945.20	507.855
14	1.04	2.04	2.122	4.329	46.59	48.453	2741.96	98.864
15	0.62	1.57	0.9734	1.528	47.26	29.301	1384.77	46.008
16	1.40	0.90	1.260	1.134	47.71	66.774	3186.74	60.115
Σ Upper	50.74		175.25	657.26		1215.95	40199.87	4040.9
Σ Lower	50.74		-175.25	657.26		1215.95	40199.87	-4040.9
Total	101.48		0	1314.52		2431.90	80399.74	0

$$\bar{x} = \frac{\Sigma Ax'}{\Sigma A} = \frac{2431.9}{101.48} = 23.95 \text{ in.}$$

$$I_{z'} = \Sigma Ax'^2 + I_{o_z} = 80$$

$$\bar{z} = \frac{\Sigma Az'}{\Sigma A} = \frac{0}{101.48} = 0$$

$$I_z = I_{z'} - \bar{x}^2 \Sigma A = 80{,}550 - (23.95^2 \times 101.48) = 22{,}34[illegible] \text{ in}^3$$

$$I_x = \Sigma Az'^2 - \bar{z}\Sigma A = 1314.52 - 0 \times 101.48 = [illegible] \text{ in}^3$$

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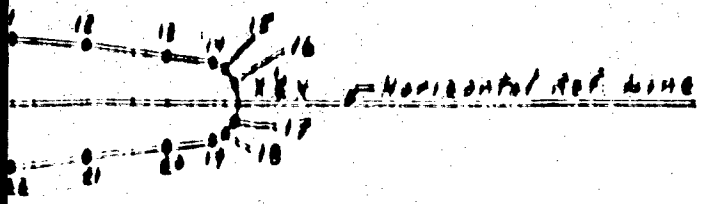
PAGE 3.01.240
MODEL GA-38B
SER 2281
CODE 81000 3.17.3

WING ANALYSIS



XXIV

(6)	(7)	(8)	(9)	(10)	(11)
Ax'	Ax'^2	$Ax'z'$	I_{o_z}	Distance From N.A. To Outer Fiber	
				x	z
(1) x (5)	(5) x (6)	(8) x (5)	$\frac{1}{12}bh^3$	(5) = $\bar{x}$	(8) = $\bar{z}$
0.0135	0.0102	0.0410	—	-23.84	3.53
1.271	1.342	1.8911	—	-22.13	1.48
2.646	5.186	5.8212	0.205	-21.77	2.23
8.882	33.582	25.6128	1.15	-20.27	2.73
38.511	213.913	124.886	8.65	-16.82	3.71
58.448	426.852	228.787	7.72	-12.28	4.23
75.137	1226.33	338.851	9.16	-7.64	4.50
96.641	2013.69	430.684	8.16	-3.06	4.47
117.046	2784.65	497.454	8.07	1.55	4.25
187.812	4148.76	845.742	9.04	6.14	3.96
169.222	5556.50	578.637	9.21	10.78	3.59
131.085	7139.24	555.713	8.16	15.32	3.07
203.777	8945.20	507.855	8.41	19.91	2.47
49.455	2741.96	98.864	—	22.64	2.04
27.301	1394.77	46.008	—	23.31	1.57
66.714	3186.14	60.115	—	25.16	0.93
1215.15	40191.57	4040.9	75.135		
1215.15	39177.81	-4043.1	75.135		
2431.90	79311.14	0	150.270		

$I_{x'} = \Sigma Ax'^2 + I_{o_z} = 80311.14 + 150.27 = 80,550\ in^3$

$N = \Sigma_x \Sigma_z = \Sigma_{xz}^2 = 1815\,(22,341) = 0$

$= 27.3784\,(1.3^3)\,in^6$

$c_1 = \Sigma_{xz}/N = \frac{0}{27.3784\,(1.3^3)} = 0$

$c_2 = \Sigma_z/N = \frac{22,341}{27.3784\,(1.3^3)} = 763.753\,(1.3^3)\,in^3$

$c_3 = \Sigma_x/N = \frac{1815}{27.3784\,(1.3^3)} = 44.73\,(1.3^3)\,in^3$

Note:
For Lower Surface Change Signs in Z Column (11) For Elements 11 thru 22

2

WING ANALYSIS

Calculation of Bending Stress

Condition A_2 Symmetrical Maneuver

Wing Sta. 0.00 (Wing Root)

$$f_b = (C_1 M_z - C_2 M_x)z + (C_1 M_x - C_3 M_z)x$$

$C_1 = 0$ $\qquad M_x = 3400$ in.-lbs. Ref. Pg. 2.02.180

$C_2 = 760.456 \times 10^{-6}$ $\qquad M_z = -1310$ in.-lbs. Ref. Pg. 2.02.180

$C_3 = 44.761 \times 10^{-6}$

$$(C_1 M_z - C_2 M_x) = -(760.456 \times 10^{-6})(3400)$$

$$= -2.59$$

$$(C_1 M_x - C_3 M_z) = (44.761 \times 10^{-6})(1310)$$

$$= 0.0586$$

$\therefore$

$$f_b = 0.0586\,x - 2.59\,z$$

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DATE 1-19-81
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GOODYEAR AIRCRAFT

PAGE 1.01.060
MODEL [illegible]
SERIAL [illegible]
REF NO. [illegible]

[illegible] Wing Sta 0.00 Table XXV WING ANALYSIS

Cal Element	(10) X	(11) Z	(12) = 2.59 Z	(13) 0.0544 X	(14) P_b = (12) + (13)	(15) (1) X (14)
1	= 23.84	0.53	= 1.38	= 1.40	= 2.78	= 2.38
2	= 23.13	1.48	= 3.83	= 1.36	= 5.19	= 8.05
3	= 21.99	2.20	= 5.70	= 1.29	= 6.99	= 9.44
4	= 20.27	2.90	= 7.52	= 1.19	= 8.71	= 20.90
5	= 16.82	3.71	= 9.60	= 0.99	= 10.59	= 49.73
6	= 12.23	4.28	= 11.10	= 0.72	= 11.82	= 54.03
7	= 7.64	4.50	= 11.68	= 0.45	= 12.13	= 56.63
8	= 3.05	4.47	= 11.60	= 0.18	= 11.78	= 54.23
9	1.55	4.25	= 11.00	0.09	= 10.91	= 50.03
10	6.14	3.96	= 10.25	0.36	= 9.89	= 45.33
11	10.73	3.58	= 9.29	0.63	= 8.66	= 40.03
12	15.32	3.07	= 7.95	0.90	= 7.05	= 32.58
13	19.91	2.47	= 6.45	1.17	= 5.28	= 24.60
14	22.64	2.04	= 5.30	1.33	= 3.97	= 4.13
15	23.31	1.57	= 4.07	1.37	= 2.7	= 1.68
16	23.76	0.90	= 2.33	1.39	= 0.94	= 1.32
17	23.76	0.90	2.33	1.39	3.72	5.22
18	23.31	= 1.57	4.07	1.37	5.44	3.37
19	22.64	= 2.04	5.30	1.33	6.63	6.90
20	19.91	= 2.47	6.45	1.17	7.62	35.47
21	15.32	= 3.07	7.95	0.90	8.85	40.27
22	10.73	= 3.58	9.29	0.63	9.92	45.86
23	6.14	3.96	10.25	0.36	10.61	48.57
24	1.55	= 4.25	11.00	0.09	11.09	50.87
25	= 3.05	= 4.47	11.60	= 0.18	11.42	52.64
26	= 7.64	= 4.50	11.68	= 0.45	11.23	51.87
27	= 12.23	= 4.28	11.10	= 0.72	10.38	47.37
28	= 16.82	= 3.71	9.60	= 0.99	8.61	40.50
29	= 20.27	= 2.90	7.52	= 1.19	6.33	15.20
30	= 21.99	= 2.20	5.70	= 1.29	4.41	5.95
31	= 23.13	= 1.48	3.83	= 1.36	2.47	3.83
32	= 23.84	= 0.53	1.38	= 1.40	= 0.02	= 0.02
Σ					Stress check	0.00

219-83 (3-5)

WING ANALYSIS

Calculation of Bending Stress

Condition C_2 Symmetrical Maneuver

Wing Sta. 0.00 (Wing Root)

$$f_b = (C_1 M_z - C_2 M_x) z + (C_1 M_x - C_3 M_z) x$$

$C_1 = 0$ $\qquad M_x = 3945$ in.-lbs.

$C_2 = 760.456 \times 10^{-6}$ $\qquad M_z = 2910$ in.-lbs.

$C_3 = 44.761 \times 10^{-6}$ $\qquad$ Loads Ref. Pg. E.02.230 & E.02.250

$$(C_1 M_z - C_2 M_x) = -(760.456 \times 10^{-6})(3945)$$

$$= -3$$

$$(C_1 M_x - C_3 M_z) = -(44.761 \times 10^{-6})(2910)$$

$$= -0.13$$

$\therefore$

$$f_b = -0.13\, x - 3\, z$$

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GOODYEAR
AIRCRAFT

PAGE 5.01.000
MODEL GA-460
SIZE 3861
DET NO S 07-3

Condition GA-1 Wing Sta. 0.00 Table XXVI WING ANALYSIS

Col.	(11)	(12)	(13)	(13)	(14)	(15)
Increment	Y	N	= 3 N	= 0.13 Y	(16) = (12) + (13)	(17) = (1) × (16)
1	= 23.84	0.53	= 1.59	3.10	1.51	1.29
2	= 23.13	1.48	= 4.44	3.01	= 1.43	= 2.22
3	= 21.99	2.20	= 6.60	2.86	= 3.74	= 5.05
4	= 20.27	2.90	= 8.70	2.64	= 6.06	= 14.55
5	= 16.82	3.71	= 11.13	2.19	= 8.94	= 42.00
6	= 12.23	4.28	= 12.84	1.59	= 11.25	= 51.10
7	= 7.64	4.50	= 13.50	1.00	= 12.50	= 57.60
8	= 3.05	4.47	= 13.41	0.40	= 13.01	= 60.00
9	1.55	4.25	= 12.75	= 0.20	= 12.95	= 59.50
10	6.14	3.96	= 11.88	= 0.80	= 12.68	= 58.00
11	10.73	3.58	= 10.74	= 1.40	= 12.14	= 55.90
12	15.32	3.07	= 9.21	= 1.99	= 11.20	= 51.70
13	19.91	2.49	= 7.47	= 2.59	= 10.06	= 46.80
14	22.64	2.04	= 6.12	= 2.95	= 9.07	= 9.45
15	23.31	1.57	= 4.71	= 3.03	= 7.74	= 4.80
16	23.76	0.90	= 2.70	= 3.09	= 5.79	= 8.10
17	23.76	- 0.90	2.70	= 3.09	- 0.39	= 0.55
18	23.31	- 1.57	4.71	- 3.03	1.68	1.00
19	22.64	- 2.04	6.12	- 2.95	3.17	3.30
20	19.91	- 2.49	7.47	- 2.59	4.88	22.30
21	15.32	- 3.07	9.21	- 1.99	7.22	33.30
22	10.73	- 3.58	10.74	- 1.40	9.34	43.10
23	6.14	- 3.96	11.88	- 0.80	11.08	50.70
24	1.55	- 4.25	12.75	- 0.20	12.55	57.60
25	- 3.05	- 4.47	13.41	0.40	13.81	63.90
26	- 7.64	- 4.50	13.50	1.00	14.50	67.00
27	- 12.23	- 4.28	12.84	1.59	14.43	66.00
28	- 16.82	- 3.71	11.13	2.19	13.32	62.60
29	- 20.27	- 2.90	8.70	2.64	11.34	27.19
30	- 21.99	- 2.20	6.60	2.86	9.46	12.80
31	- 23.13	- 1.48	4.44	3.01	7.45	11.55
32	- 23.84	- 0.53	1.59	3.10	4.69	3.99
Σ					Stress Check →	0.00

218-83 (3-5)

WING ANALYSIS

Calculation of Shear Stress

The shear flow at any point on the cross-section is given by:

$$q = (C_1V_x - C_2V_z)(Q_x - Q_{x_0}) + (C_1V_z - C_3V_x)(Q_z - Q_{z_0}) + q_m M_y$$

where

V_x, V_z = components of shearing force along the x-x and z-z axes, respectively

Q_x = ΣA_z, the summation of the product of the element of area and its coordinate from the x-x axis through the centroid of the section

Q_x = ΣA_x $\quad Q_{x_0} = \dfrac{\Sigma Q_x \; a}{\Sigma \Delta a}$ $\quad Q_{z_0} = \dfrac{\Sigma Q_z \; a}{\Sigma \Delta a}$

Δa = twice the area enclosed by lines joining the ends of an element of skin with the centroid of the entire cross section

q_m = $1/\Sigma \Delta a$ = shear flow for a unit torque

M_y = torsional moment about the longitudinal axis through centroid of section, positive if clockwise.

Convention of signs for positive direction of the coordinate axes and the shearing forces and moments applied to the cross section are shown below:

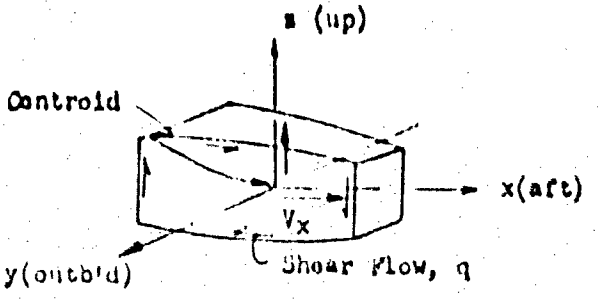
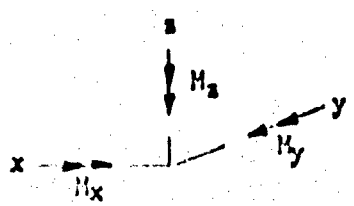


Figure 52a
Port Wing View Looking Inboard



Moments L.H. Rule
Figure 52b

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Shear Flow Factors X_0 & Z_0 Are words at beginning of section ;

Col.	①	①	①	④	⑤	⑥	⑦	⑧	⑨
Item	Ele	Area	x	Ax	z	Az	$Q_x = \Sigma Ax$	$Q_z = \Sigma Az$	$X_z Z_0$
Ref.				② x ③		② x ⑤	Σ ④	Σ ⑥	
	32	—	= 23.84	—	= .53	—	0	0	12.64
	1	.85	= 23.84	= 20.26	.53	.45	= 20.26	.45	= 12.26
	2	1.55	= 23.13	= 35.85	1.48	2.29	= 56.11	2.74	= 32.85
	3	1.35	= 21.99	= 29.69	2.20	2.97	= 85.83	5.71	= 44.59
	4	2.40	= 20.27	= 48.65	2.90	6.96	= 134.45	12.67	= 48.18
	5	4.70	= 16.82	= 79.05	3.71	17.44	= 213.50	30.11	= 45.37
	6	4.56	= 12.23	= 55.77	4.28	19.52	= 269.27	49.63	= 32.70
	7	4.61	= 7.64	= 35.22	4.50	20.75	= 304.49	70.38	= 13.73
	8	4.61	3.05	= 14.06	4.47	20.61	= 318.55	90.99	6.93
	9	4.59	1.55	7.11	4.25	19.51	= 311.44	110.50	26.10
	10	4.58	6.14	28.12	3.96	18.14	= 283.32	128.64	42.49
	11	4.62	10.73	49.57	3.58	16.59	= 233.75	145.18	54.85
	12	4.61	15.32	70.63	3.07	14.15	= 163.12	159.33	61.12
	13	4.65	19.91	92.58	2.49	11.58	= 70.54	170.91	56.37
	14	1.04	22.64	23.55	2.04	2.12	= 46.99	173.03	47.55
	15	.62	23.31	14.45	1.57	.97	= 32.54	174.00	37.30
	16	1.40	23.76	33.26	.90	1.26	.72	175.26	21.38
	17	1.40	23.76	33.26	- .90	- 1.26	33.98	174	- 20.99
	18	.62	23.31	14.45	- 1.57	- .97	48.43	173.03	- 35.51
	19	1.04	22.64	23.55	- 2.04	- 2.12	71.98	170.91	- 40.62
	20	4.65	19.91	92.58	- 2.49	- 11.58	164.56	159.33	- 38.15
	21	4.61	15.32	70.63	- 3.07	- 14.15	235.19	145.18	- 32.94
	22	4.62	10.73	49.57	- 3.58	- 16.54	284.76	128.64	- 21.98
	23	4.58	6.14	28.12	- 3.96	- 18.14	312.88	110.50	- 6.14
	24	4.59	1.55	7.11	- 4.25	- 19.51	319.99	90.99	12.96
	25	4.61	- 3.05	- 14.06	- 4.47	- 20.61	305.93	70.38	34.15
	26	4.61	- 7.64	- 35.22	- 4.50	- 20.75	270.71	49.63	55.04
	27	4.56	- 12.23	- 55.77	- 4.28	- 19.52	214.94	30.11	71.99
	28	4.70	- 16.82	- 79.05	- 3.71	- 17.44	135.89	12.67	75.20
	29	2.40	- 20.27	- 48.65	- 2.90	- 6.96	87.24	5.71	63.77
	30	1.35	- 21.99	- 29.69	- 2.20	- 2.97	57.55	2.74	50.84
	31	1.55	- 23.13	- 35.85	- 1.48	- 2.29	21.70	.45	35.78
	32	.85	- 23.84	- 20.26	- .53	- .45	0	0	

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ning of section; X_e & Z_e are coord. at end of section

(1) $Q_1 = \frac{\lambda}{\Sigma\lambda}$ (3)	(2) $Q_2 = \frac{\lambda}{\Sigma\lambda}$ (4)	(3) X_e Z_e	(4) X_o Z_o	(5) Δa (1)-(2)	(6) $Q_1 \Delta a$ (1)x(5)	(7) $Q_2 \Delta a$ (2)x(5)	(8) $Q_1 - Q_{1o}$ (1)-.69	(9) $Q_2 - Q_{2o}$ (2)-.8490
0	0	12.64	= 13.61	25.27	0	0	= .87	= 14.50
0.16	.115	= 13.26	= 35.11	23.32	= 4163.31	10.36	= 1.015	= 14.05
56.11	1.14	= 33.55	= 59.17	18.61	= 1033.46	53.17	= 58.80	= 11.76
5.83	5.71	= 41.57	= 63.77	14.11	= 1648.04	139.53	= 86.49	= 78.11
34.15	13.67	= 41.18	= 75.20	26.42	= 2552.17	511.74	= 135.4	= 71.13
11.69	33.11	= 45.31	= 71.11	26.62	= 5661.57	801.53	= 214.19	= 54.19
61.57	49.63	= 12.79	= 55.04	22.34	= 5315.11	1101.73	= 261.76	= 34.87
84.49	70.38	= 13.73	= 34.15	20.42	= 6217.67	1437.16	= 305.18	= 14.12
12.55	90.99	6.93	= 13.96	11.27	= 6355.76	1839.79	= 319.24	6.41
11.44	113.50	21.10	6.14	19.96	= 6216.34	2305.58	= 312.13	26.00
63.32	137.34	43.41	21.98	20.51	= 5713.61	2838.41	= 334.01	64.14
43.75	145.18	54.65	33.14	21.91	= 5.21.46	3183.19	= 334.44	60.67
63.12	159.33	61.12	37.15	22.97	= 5746.37	3553.41	= 161.21	74.33
7.54	170.31	56.17	43.62	15.75	= 1111.01	2671.23	= 71.23	20.41
46.17	173.03	47.55	35.54	13.01	= 564.35	2257.01	= 47.63	12.43
32.54	174.09	37.10	30.98	15.12	= 831.05	2157.62	= 33.23	81.50
.72	175.26	21.32	- 21.32	40.76	30.79	7411.12	.03	90.76
8.96	174	- 20.99	- 37.30	16.32	554.55	2831.62	33.29	81.50
2.43	137.08	- 35.31	- 47.55	12.01	531.64	2078.09	47.74	82.53
1.96	170.91	- 40.62	- 56.37	15.75	1133.61	2611.33	71.29	86.41
41.56	159.33	- 38.15	- 61.12	22.97	5717.31	3654.21	167.32	74.13
35.19	145.18	- 32.91	- 54.25	21.91	5153.31	3183.31	334.50	60.63
44.76	124.64	- 21.98	- 42.49	20.51	5840.43	2636.41	394.07	414.14
12.26	110.50	- 6.11	- 26.10	19.96	6245.02	2205.58	312.19	26.00
19.19	90.99	12.96	- 6.93	19.29	6304.60	1201.79	319.30	6.41
25.73	70.38	34.15	13.73	20.42	6247.09	1437.16	305.74	- 14.13
10.71	49.63	55.04	32.70	22.34	6047.66	1108.73	270.02	- 34.27
14.94	30.11	71.99	45.37	26.62	5724.70	801.53	214.25	- 54.39
35.89	12.67	75.70	42.75	26.42	3590.2	334.74	135.20	- 71.83
47.24	5.71	63.77	44.57	19.12	1673.76	101.51	86.55	- 78.79
57.55	0.74	50.24	32.55	18.34	1055.47	50.25	56.86	- 31.74
21.70	.115	35.28	12.26	23.00	499.53	10.36	21.01	- 84.05

WING ANALYSIS

Table XXVII

$$Q_{1o} = \frac{\Sigma Q_1 \Delta a}{\Sigma \Delta a} = \frac{470.98}{678.85} = .69$$

$$Q_{2o} = \frac{\Sigma Q_2 \Delta a}{\Sigma \Delta a} = \frac{5746.88}{677.16} = 8.50$$

$$\beta_m = \frac{1}{\Sigma \Delta a} = \frac{1}{677.16} = 0.00147$$

2

Σ (11) = 671.36
Σ (12) = 470.98
Σ (13) = 5746.88

Calculation of Shear Stress — WING ANALYSIS

Condition A_2 Symmetrical Maneuver

Wing Sta. 0.00 (Wing Root)

$$q = (C_1 V_x - C_2 V_z)(Q_x - Q_{x_0}) + (C_1 V_z - C_3 V_x)(Q_z - Q_{z_0}) + g_m M_y$$

$C_1 = 0$ $\quad V_x = 33$ lbs. Ref. Pg. 2.02.190

$C_2 = 760.456 \times 10^{-6}$ $\quad V_z = 179$ lbs. Ref. Pg. 2.02.170

$C_3 = 44.761 \times 10^{-6}$ $\quad M_y = 2070$ In-Lbs. Ref. Pg. 2.02.210

$g_m = 1470 \times 10^{-6}$ $\quad g_m M_y = 3.05$

$$(C_1 V_x - C_2 V_z) = -(760.456 \times 10^{-6})(179)$$
$$= -0.136$$

$$(C_1 V_z - C_3 V_x) = -(44.761 \times 10^{-6})(33)$$
$$= -0.00148$$

$\therefore$

$$q = 3.05 - 0.136\,(Q_x - Q_{x_0}) - 0.00148\,(Q_z - Q_{z_0})$$

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CONTINUED AT, WING STA. 0.00 TWIN KEVLIR WING ANALYSIS

CHK POINT	(14) $Q_A = \sum Q_D$	(15) $Q_D = \sum q_D$	(16) $Q/136.4$ (13)	(17) $= (16)/(12)$	(18) $\frac{1}{2}(15)*(12)$	(19) (11) x (18)	(20) STA.	(21) γ in STA.
1-2	= 20.95	= 84.05	11.43	0.031	14.51	334	2	14.39
2-3	= 56.80	= 81.76	11.13	0.084	14.26	261	3	14.08
3-4	= 86.49	= 78.79	10.71	0.128	13.80	266	4	13.46
4-5	= 135.14	= 71.83	9.78	0.200	13.03	344	5	11.90
5-6	= 214.19	= 54.39	7.40	0.317	10.77	287	6	9.49
6-7	= 269.96	= 34.87	4.75	0.400	8.20	183	7	6.81
7-8	= 305.18	= 14.12	1.92	0.451	5.42	111	8	4.03
8-9	= 319.24	6.49	= 0.88	0.473	2.64	53	9	1.31
9-10	= 312.13	26.00	= 3.54	0.463	= 0.03	= 1	10	= 1.28
10-11	= 284.01	44.14	= 6.00	0.420	= 2.53	= 52	11	= 3.69
11-12	= 234.44	60.68	= 8.25	0.347	= 4.85	= 106	12	= 5.88
12-13	= 163.81	74.93	= 10.19	0.243	= 6.90	= 158	13	= 7.75
13-14	= 71.23	86.41	= 11.75	0.106	= 8.59	= 135	14	= 8.76
14-15	= 47.68	88.53	= 12.05	0.071	= 8.93	= 107	15	= 9.02
15-16	= 33.23	89.50	= 12.20	0.049	= 9.10	= 149	16	= 9.20
16-17	0.03	90.76	= 12.35	0.000	= 9.30	= 300	17	= 9.25
17-18	33.23	89.50	= 12.20	- 0.049	= 9.20	= 150	18	= 9.14
18-19	47.68	88.53	= 12.05	- 0.071	= 9.07	= 100	19	= 8.94
19-20	71.23	86.41	= 11.75	- 0.106	= 8.81	= 130	20	= 8.10
20-21	163.81	74.93	= 10.19	- 0.243	= 7.38	= 169	21	= 6.47
21-22	234.44	60.68	= 8.25	- 0.347	= 5.55	= 121	22	= 4.46
22-23	284.01	44.14	= 6.00	- 0.420	= 3.37	= 69	23	= 2.16
23-24	312.13	26.00	= 3.54	- 0.463	= 0.05	= 19	24	0.37
24-25	319.24	6.49	= 0.88	- 0.473	1.69	34	25	3.11
25-26	305.18	= 14.12	1.92	- 0.451	4.52	92	26	5.96
26-27	269.96	= 34.87	4.75	- 0.400	7.40	165	27	8.77
27-28	214.19	= 54.39	7.40	- 0.317	10.13	270	28	11.38
28-29	135.14	= 71.83	9.78	- 0.200	12.63	334	29	13.13
29-30	86.49	= 78.79	10.71	- 0.128	13.63	262	30	13.81
30-31	56.80	= 81.76	11.13	- 0.084	14.10	258	31	14.28
31-32	20.95	= 84.05	11.43	- 0.031	14.45	332	32	14.50
32-1	- 0.69	= 84.50	11.50	0.000	14.55	368	1	14.53

Stress Check → M_y = 2070 Vs. M_D = 2071

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Calculation of Shear Stress WING Analysis

Condition C_2 Symmetrical Maneuver

Wing Sta. 0.00 (Wing Root)

$$q = (C_1 V_x - C_2 V_z)(Q_x - Q_{x_0}) + (C_1 V_z - C_3 V_x)(Q_z - Q_{z_0}) + q_m M_y$$

$C_1 = 0$ $V_x = 101$ lbs. Ref. Pg. 2.02.240

$C_2 = 760.456 \times 10^{-6}$ $V_z = 186$ lbs. Ref. Pg. 2.02.220

$C_3 = 44.761 \times 10^{-6}$ $M_y = 2135$ in-lbs. Ref. Pg. 2.01.260

$q_m = 1470 \times 10^{-6}$ $q_m M_y = 3.14$

$$(C_1 V_x - C_2 V_z) = -(760.456 \times 10^{-6})(186)$$
$$= -0.1416$$

$$(C_1 V_z - C_3 V_x) = -(44.761 \times 10^{-6})(101)$$
$$= -0.00452$$

$$q = 3.14 - 0.1416(Q_x - Q_{x_0}) - 0.00452(Q_z - Q_{z_0})$$

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PART 3.01.140
MODEL GA-420
SIZE DHC
REF NO N 37-3

Condition C.L. Wing STA. 0.00 TO STA XXIX WING ANALYSIS

Col. Element	(14) $C_{n_2} - C_{n_1}$	(15) $C_{R_2} - C_{R_1}$	(16) $-0.416 \times (15)$	(17) $0.00452 (14)$	(18) $3.14 \times (17)$	(19) $(17) \times (18)$	(3) Σ (19) STA
1-2	= 20.95	= 84.05	= 11.90	0.095	15.14	349	1 15.04
2-3	= 56.80	= 81.76	11.58	0.257	14.08	275	2 14.83
3-4	= 86.49	= 78.19	11.15	0.391	14.68	282	3 14.80
4-5	= 135.14	= 71.83	10.17	0.610	13.92	368	4 14.87
5-6	= 214.19	= 54.39	7.70	0.970	11.81	315	5 12.86
6-7	= 269.96	= 34.87	4.94	1.220	9.30	208	6 10.56
7-8	= 305.18	= 14.12	2.00	1.380	6.52	133	7 7.01
8-9	= 319.24	6.49	= 0.92	1.440	3.66	73	8 5.09
9-10	= 312.13	26.00	= 3.68	1.410	0.87	17.4	9 2.27
10-11	= 284.01	44.14	= 6.25	1.285	- 1.82	- 37.4	10 - 0.48
11-12	= 234.44	60.68	= 8.60	1.060	- 4.40	- 96.6	11 - 3.11
12-13	= 163.81	74.83	= 10.60	0.740	- 6.72	-154	12 - 5.56
13-14	= 71.23	86.41	= 12.20	0.322	- 8.74	- 138	13 - 7.73
14-15	= 42.68	88.53	= 12.55	0.216	- 9.19	- 110	14 - 8.97
15-16	= 33.23	89.50	= 12.70	0.150	- 9.41	-152	15 - 9.30
16-17	0.03	90.76	= 12.85	0.000	- 9.71	- 415	16 - 9.56
17-18	33.23	89.50	- 12.70	- 0.150	- 9.71	- 159	17 - 9.71
18-19	42.68	88.53	- 12.55	- 0.216	- 9.63	- 115.5	18 - 9.67
19-20	71.23	86.41	- 12.20	- 0.322	- 9.38	- 148	19 - 9.51
20-21	163.81	74.83	- 10.60	- 0.740	- 8.20	- 189	20 - 8.79
21-22	234.44	60.68	- 8.60	- 1.060	- 6.52	- 143	21 - 7.36
22-23	284.01	44.14	- 6.25	- 1.285	- 4.40	- 00.4	22 - 5.46
23-24	312.13	26.00	- 3.68	- 1.410	- 1.95	- 39	23 - 3.18
24-25	319.24	6.49	- 0.92	- 1.440	0.78	15.5	24 - 0.59
25-26	305.18	- 14.12	2.00	- 1.380	3.76	71	25 2.27
26-27	269.96	- 34.87	4.94	- 1.220	6.86	153	26 5.31
27-28	214.19	- 54.39	7.70	- 0.970	9.87	263	27 8.37
28-29	135.14	- 71.83	10.17	- 0.610	12.10	336	28 11.29
29-30	86.49	- 78.19	11.15	- 0.391	13.90	267	29 13.30
30-31	56.80	- 81.76	11.58	- 0.257	14.46	265	30 14.18
31-32	20.95	- 84.05	11.90	- 0.095	14.95	345	31 14.71
32-1	- 0.60	- 84.50	11.95	0.000	15.09	382	32 15.02
							1 15.12

Stress Check → M_y = 2135 vs. Σ (19) = 2135

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WING ANALYSIS

Calculation of Axial Stress

The axial stress (f_a) is simply the axial load (P_y) divided by the perimeter of the wing cross-section (Total of column ①, page 3.01.040).

The axial load (P_y) is taken as the same for conditions A_2 and C_2.

Conditions A_2 and C_2, Wing Sta. 0.00

$P_y = -771$ lbs Ref. P_y. 2.02.170

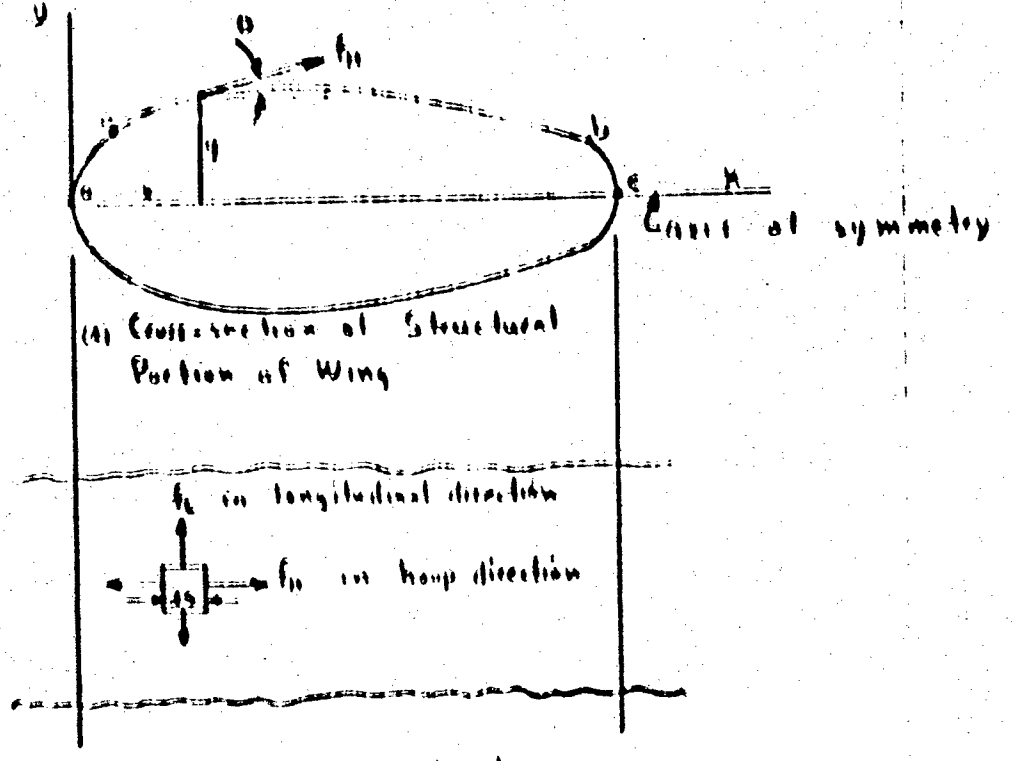
Perimeter = 101.48 in.

$$\therefore \quad f_a = -\frac{771}{101.48} = -7.56 \text{ lbs./in.}$$

Inflation Stresses in an Airmat Wing

WING ANALYSIS

(See Ref. 3)



(a) Cross-section of Structural Portion of Wing

(b) Sketch Showing Orientation of Membrane Stresses

Figure 53 Longitudinal and Hoop Inflation Stresses

From summation of forces in x-direction on part of an elemental chordwise strip

$f_H = \frac{py}{\cos\theta}$ = hoop tension, lb/in, y and θ are in (a) and

p = internal pressure, psi. From summation of forces in the longitudinal direction:

$pA = \int_0^c f_L \, ds$, lb. in which A = enclosed

area above x-axis (only half needed because of symmetry),

WING ANALYSIS

f_L = longitudinal stress, lb/in and ds is an element of arc. The longitudinal unit strain ϵ_L is assumed to be constant (i.e. independent of x) and is given by

$$\epsilon_L = \frac{1}{E}\left[f_L - \mu f_n \right]$$ in which E = elastic modulus

of the fabric, lb/in and μ = Poisson's Ratio

From the above three equations and $dx = ds \cos\theta$

$$f_L = \frac{pA}{{}_aS_c} + \frac{\mu p y}{\cos\theta} - \frac{\mu}{{}_aS_c}\int_0^{x_c} \frac{y\,dx}{\cos^2\theta}$$

The integral is evaluated in three parts

$$I = \int_0^{x_c} \frac{y\,dx}{\cos^2\theta} = {}_aI_a + {}_aI_b + {}_bI_c$$

${}_aI_a$ and ${}_bI_c$ are for circular arcs $\widehat{oa}$ and $\widehat{bc}$ while ${}_aI_b$ is evaluated numerically, using NACA 0015 airfoil coordinates, in the table below the arc $\widehat{ab}$ is divided into 10 parts referred to as 'points' and $\mu = 0.2$ is assumed

Table XXX

Point	y	θ	$\frac{1}{\cos\theta}$	$\frac{y}{\cos^2\theta}$	$\frac{f_n}{p}$	$\frac{f_L}{p}$
	in	Deg			in	in
1	2.52	23.3	1.0888	2.98	2.74	3.254
2	3.71	4.6	1.0148	3.82	3.77	3.460
3	4.28	4.3	1.0022	4.30	4.30	3.566
4	4.50	0	1.0000	4.50	4.50	3.606
5	4.47	0.8	1.0001	4.47	4.47	3.600
6	4.25	3.0	1.0014	4.25	4.25	3.556
7	3.96	4.7	1.0034	3.97	3.98	3.502
8	3.58	5.2	1.0041	3.62	3.59	3.424
9	3.07	6.5	1.0065	3.12	3.09	3.324
10	2.49	6.8	1.0071	2.53	2.51	3.208

The inflation stresses are plotted vs chord positions on the next page. The resultant of f_L does not pass through the centroid of 'A' and hence gives a small chordwise moment that is neglected.

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GOODYEAR
GOODYEAR AIRCRAFT CORPORATION
(Arizona Inc.)

PAGE 2.01.100
MODEL GA 468
GER 1561
REP NO 891-3

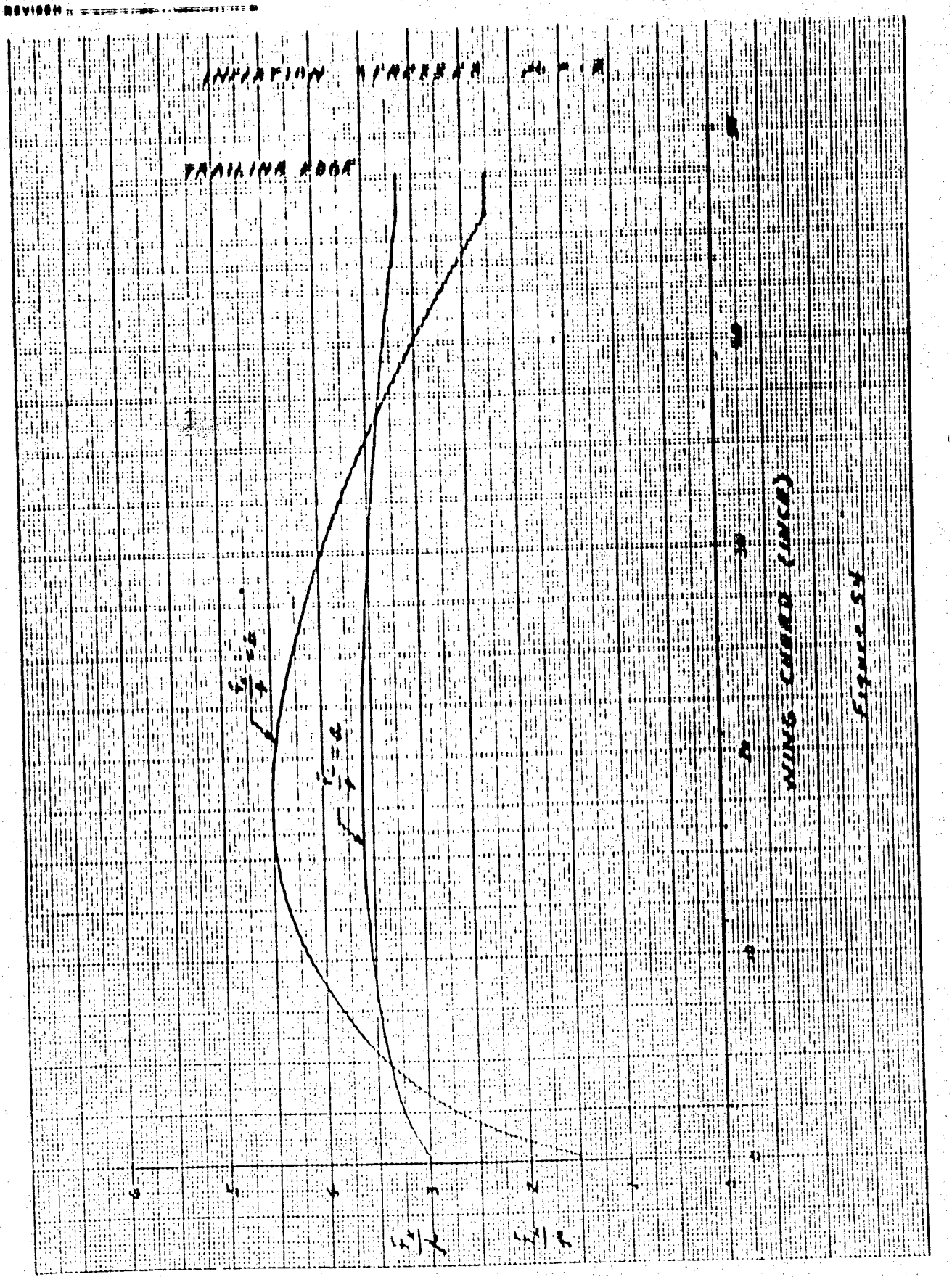


Figure 54

JR 120 (2-60)

WING ANALYSIS

Calculation of Inflation Stresses

Conditions A_2 and C_L, Any Wing Station

These stresses are taken from the figure of page 3.01.180

Table XXXV

Column	①	②
Element Pg. 3.01.040	$a = \frac{f_L}{p}$	$b = \frac{f_N}{p}$
1 & 32	3.00	1.50
2 & 31	3.07	1.92
3 & 30	3.20	2.50
4 & 29	3.33	3.08
5 & 28	3.45	3.79
6 & 27	3.51	4.30
7 & 26	3.60	4.50
8 & 25	3.60	4.45
9 & 24	3.56	4.26
10 & 23	3.50	3.98
11 & 22	3.43	3.60
12 & 21	3.33	3.10
13 & 20	3.22	2.55
14 & 19	3.15	2.27
15 & 18	3.15	2.27
16 & 17	3.15	2.27

Limit Margin of Safety for Wrinkling — WING ANALYSIS

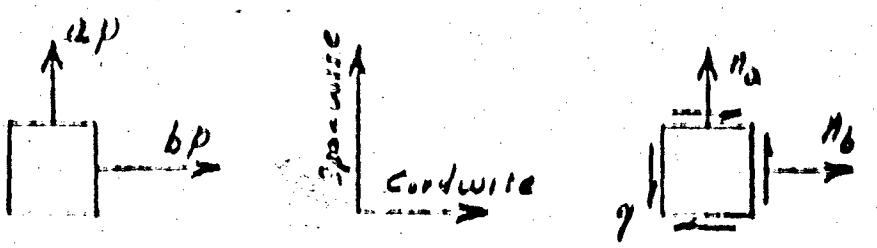
The limit margin of safety for the wing is based on the pressure required to prevent wrinkling of the wing, i.e.

$$M.S. = \frac{P_{inflation}}{P_{req'd}} - 1$$

For $P_{inflation} = 7.0$ psi,

$$M.S. = \frac{7.0}{P_{req'd}} - 1$$

Determination of $P_{req'd}$



Inflation Stresses — Applied Stresses

where:

$$a = \frac{f_L}{P} \qquad b = \frac{f_H}{P}$$

$$n_a = f_a + f_b$$

$$n_b = 0$$

Wrinkling occurs when one of the principal stresses is zero. It may then be seen from a Mohr's stress circle that the shear stress required to cause wrinkling may be expressed as,

WING ANALYSIS

$$q = \sqrt{n_x n_y}$$

where: $n_x = ap + n_a$
$n_y = bp + n_b$

then,

$$q^2 = (ap + n_a)(bp + n_b)$$

$$abp^2 + (bn_a + an_b)p + n_a n_b - q^2 = 0$$

$$p^2 + \left(\frac{n_a}{a} + \frac{n_b}{b}\right)p + \frac{n_a n_b}{ab} - \frac{q^2}{ab} = 0$$

therefore, the pressure required to prevent wrinkling is given by,

$$p = -\frac{1}{2}\left(\frac{n_a}{a} + \frac{n_b}{b}\right) \pm \frac{1}{2}\left[\left(\frac{n_a}{a} + \frac{n_b}{b}\right)^2 + 4\left(\frac{q^2}{ab} - \frac{n_a n_b}{ab}\right)\right]^{\frac{1}{2}}$$

since, $n_b = 0$,

$$p_{Rigid} = -\frac{n_a}{2a} \pm \frac{1}{2}\left[\left(\frac{n_a}{a}\right)^2 + \frac{4q^2}{ab}\right]^{\frac{1}{2}}$$

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Condition A_2, Wing Sta. 0.00

Calculation of Margin of Safety for Wrinkling

Col. / Item / Ref. Pg. / Elem.	① f_a	② f_b	③ $\eta_2 =$ ①+②	④ q	⑤ q^2	⑥ a	⑦ b	⑧ ab
	3.01.159	3.01.060		3.01.160		3.01.170	3.01.170	
1	-7.56	-2.78	-10.34	14.53	211	3.50	1.50	4.50
2	-7.56	-5.19	-12.75	14.39	207	3.57	1.92	5.99
3	-7.56	-6.99	-14.55	14.08	198	3.29	2.50	8.00
4	-7.56	-8.71	-16.27	13.46	180.5	3.33	3.08	10.25
5	-7.56	-10.59	-18.15	11.99	141.5	3.45	3.79	13.10
6	-7.56	-11.82	-19.38	9.49	90	3.57	4.39	15.40
7	-7.56	-12.13	-19.69	6.81	46.3	3.60	4.59	16.50
8	-7.56	-11.78	-19.34	4.03	16.2	3.60	4.45	16.00
9	-7.56	-10.91	-18.47	1.31	1.71	3.56	4.26	15.20
10	-7.56	-9.89	-17.45	-1.29	1.64	3.50	3.98	13.90
11	-7.56	-8.66	-16.22	-3.69	13.6	3.43	3.60	12.35
12	-7.56	-7.05	-14.61	-5.88	34.5	3.33	3.10	10.33
13	-7.56	-5.28	-12.84	-7.75	60	3.22	2.55	8.22
14	-7.56	-3.97	-11.53	-8.76	76.5	3.15	2.27	7.15
15	-7.56	-2.70	-10.26	-9.02	81	3.15	2.27	7.15
16	-7.56	-0.94	-8.50	-9.20	82.5	3.15	2.27	7.15
17	-7.56	3.72	-3.84	-9.25	85.3	3.15	2.27	7.15
18	-7.56	5.44	-2.12	-9.14	83.2	3.15	2.27	7.15
19	-7.56	6.63	-0.93	-8.94	80	3.15	2.27	7.15
20	-7.56	7.62	0.06	-8.10	65.5	3.22	2.55	8.22
21	-7.56	8.85	1.29	-6.47	41.8	3.33	3.10	10.33
22	-7.56	9.92	2.36	-4.46	19.8	3.43	3.60	12.35
23	-7.56	10.61	3.05	-2.16	4.66	3.50	3.98	13.90
24	-7.56	11.09	3.53	0.37	0.137	3.56	4.26	15.20
25	-7.56	11.42	3.86	3.11	9.65	3.60	4.45	16.00
26	-7.56	11.23	3.67	5.96	35.5	3.60	4.50	16.20
27	-7.56	10.38	2.82	8.77	77	3.57	4.30	15.40
28	-7.56	8.61	1.05	11.38	129	3.45	3.79	13.10
29	-7.56	6.33	-1.23	13.43	172	3.33	3.08	10.25
30	-7.56	4.41	-3.15	13.87	192	3.29	2.50	8.00
31	-7.56	2.47	-5.09	14.28	203	3.57	1.92	6.90
32	-7.56	-0.02	-7.58	14.50	210	[illegible]	[illegible]	[illegible]

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GOODYEAR AIRCRAFT

PAGE X-31-220
MODEL 31-X-18
SER- 3631
REP NO 5-37-3

$P_{rly} = -\frac{11_b}{2a} \pm \frac{1}{2}\left[\left(\frac{11_b}{a}\right)^2 + \frac{4\,\Delta^2}{10}\right]^{1/2}$ WING AIRFOILS

for Wrinkling $M.S. = \frac{20}{11_b} - 1$

Table XXXIC

(a) a	(b) b	(c) ab	(d) $\frac{4I^2}{\Delta b}$	(e) $\frac{11_b}{b}$	(f) $\left(\frac{11_b}{b}\right)^2$	(g)+(f)	$\sqrt{(g)}$	-(h)+(e)	$P_{rly} = \frac{1}{2}$(i)	M.S.
0.120	0.01.120									
3.00	1.50	4.50	1345	= 3.45	11.90	13.340	14.13	17.58	8.79	= 0.20
3.07	1.34	5.93	140	= 4.15	17.20	157.20	12.54	16.69	8.35	= 0.16
2.43	2.33	8.00	33	= 4.55	20.70	113.70	10.33	15.48	7.74	= 0.10
3.33	3.32	10.45	70.5	= 4.30	24.00	34.59	3.74	14.54	7.32	= 0.04
3.45	3.79	13.10	43.5	= 5.25	27.50	79.70	8.41	13.66	6.93	0.03
3.47	4.10	15.40	23.4	= 5.44	29.60	53.00	7.28	12.72	6.36	0.10
3.43	4.73	16.20	11.4	= 5.47	29.90	41.20	6.44	11.91	5.96	0.17
3.60	4.45	16.00	8.05	= 5.31	28.20	32.25	5.75	11.23	5.62	0.24
3.56	4.23	15.20	6.45	= 5.20	27.00	27.45	5.25	10.45	5.23	0.34
3.50	3.33	13.00	5.47	= 4.90	24.00	25.47	5.03	10.02	5.01	0.40
3.43	3.60	12.35	4.40	= 4.73	22.30	26.70	5.17	9.99	4.95	0.41
3.33	3.10	10.33	13.40	= 4.40	19.30	32.70	5.72	10.12	5.05	0.33
3.22	2.55	8.22	23.20	= 4.00	16.00	45.20	6.72	10.72	5.37	0.30
3.15	2.27	7.15	42.80	= 3.67	13.40	56.20	7.50	11.17	5.59	0.25
3.15	2.27	7.15	45.40	= 3.56	10.60	56.00	7.49	10.75	5.38	0.30
3.15	2.27	7.15	45.20	= 2.70	7.29	53.49	7.32	10.02	5.01	0.40
3.15	2.27	7.15	47.10	= 1.22	1.49	49.19	7.02	8.24	4.12	0.70
3.15	2.27	7.15	46.50	= 0.675	0.456	46.96	6.86	7.54	3.77	0.85
3.15	2.27	7.15	44.90	= 0.295	0.087	44.99	6.70	7.00	3.50	1.00
3.22	2.55	8.22	31.85	0.019	0.00	31.85	5.65	5.63	2.82	1.43
3.33	3.10	10.33	16.20	0.388	0.15	16.35	4.05	3.66	1.83	2.83
3.43	3.60	12.35	6.41	0.699	0.474	6.88	2.62	1.93	0.97	6.20
3.50	3.93	13.90	1.34	0.912	0.757	2.10	1.45	0.58	0.29	19.41
3.56	4.26	15.20	0.04	0.993	0.985	1.03	1.015	0.33	0.17	40.20
3.60	4.45	16.00	2.41	1.07	1.14	3.55	1.89	0.82	0.41	16.10
3.60	4.50	16.20	9.11	1.02	1.04	9.81	3.12	2.10	1.05	5.66
3.57	4.30	15.40	20.00	0.764	0.58	20.58	4.55	3.89	1.95	2.59
3.45	3.79	13.10	39.40	0.304	0.092	39.49	6.29	5.99	3.00	1.33
3.22	3.03	10.45	61.10	-0.31	0.137	61.24	8.21	8.58	4.29	0.63
3.10	2.80	8.68	96.00	-0.985	0.97	96.97	9.85	10.84	5.42	0.29
3.07	1.00	3.00	137.00	-1.66	2.75	140.25	11.85	13.51	6.76	0.03
2.[illegible]	1.[illegible]	[illegible]	[illegible]	-2.[illegible]	6.40	102.90	13.30	16.43	8.22	-0.15

2

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Condition C_2, Wing Sta. 0.00

1

Calculation of Margin of Safety for Wrinkling

Col. / Item / Ref. Pg. / Elem.	① f_a 3.01.158	② f_b 3.01.183	③ n_a = ①+②	④ q 3.01.143	⑤ q^2	⑥ a 3.01.143	⑦ b 3.01.150	⑧ ab
1	-7.56	1.51	-6.05	15.14	229	3.00	1.50	4.50
2	-7.56	-1.43	-8.99	15.06	226	3.37	1.92	5.99
3	-7.56	-3.74	-11.30	14.83	220	3.20	2.50	8.00
4	-7.56	-6.06	-13.62	14.30	204	3.33	3.08	10.25
5	-7.56	-8.94	-16.50	12.87	165	3.45	3.79	13.10
6	-7.56	-11.25	-18.81	10.56	111	3.57	4.39	15.48
7	-7.56	-12.59	-20.06	7.91	62.5	3.60	4.50	16.20
8	-7.56	-13.81	-20.[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	13.00
9	-7.56	-12.95	-20.51	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
10	-7.56	-12.69	-20.24	-0.48	0.23	3.50	3.98	13.90
11	-7.56	-12.14	-19.70	-3.11	9.65	3.43	3.60	12.35
12	-7.56	-11.20	-18.76	-5.56	30.8	3.33	3.10	10.33
13	-7.56	-10.06	-17.62	-7.73	59.7	3.22	2.55	8.22
14	-7.56	-9.07	-16.63	-8.97	80	3.15	2.27	7.15
15	-7.56	-7.74	-15.30	-9.30	86.2	3.15	2.27	7.15
16	-7.56	-5.79	-13.35	-9.56	91.2	3.15	2.27	7.15
17	-7.56	-0.39	-7.95	-9.71	94	3.15	2.27	7.15
18	-7.56	1.68	-5.89	-9.67	93	3.15	2.27	7.15
19	-7.56	3.17	-4.39	-9.51	90	3.15	2.27	7.15
20	-7.56	4.88	-2.68	-8.79	77	3.22	2.55	8.22
21	-7.56	7.22	-0.34	-7.36	54	3.33	3.10	10.33
22	-7.56	9.34	1.78	-5.46	29.8	3.43	3.60	12.35
23	-7.56	11.08	3.52	-3.18	10.1	3.50	3.98	13.90
24	-7.56	12.55	4.99	-0.59	0.35	3.56	4.26	15.20
25	-7.56	13.81	6.25	2.27	5.14	3.60	4.45	16.00
26	-7.56	14.50	6.94	5.31	28.1	3.60	4.50	16.20
27	-7.56	14.43	6.87	8.31	70	3.57	4.30	15.40
28	-7.56	13.32	5.76	11.29	127	3.45	3.79	13.10
29	-7.56	11.34	3.78	13.30	176	3.33	3.08	10.25
30	-7.56	9.46	1.90	14.18	200	3.20	2.50	8.00
31	-7.56	7.45	-0.11	14.71	216	3.07	1.92	5.90
32	-7.56	4.69	-2.87	15.[illegible]	225	3.00	1.50	4.50

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GOODYEAR AIRCRAFT

PAGE 3.01.220
MODEL GA-468
SER- 3861
REP NO. 3.02.3

$$P_{req'd} = -\frac{n_z}{2a} \pm \frac{1}{2}\left[\left(\frac{n_z}{a}\right)^2 + \frac{4qA}{ab}\right]^{\frac{1}{2}}$$

WING ANALYSIS

ty for Wrinkling M.S. = $\frac{P_{allow}}{P_{req'd}} - 1$

Table XXIII

⑦ a.	⑧ b	⑨ ab	⑩ $\frac{4\gamma^4}{ab}$	⑪ $\frac{n_z}{a}$	⑫ $\left(\frac{n_z}{a}\right)^2$	⑬ ⑨ + ⑫	⑭ $\sqrt{⑬}$	⑮ -⑩ + ⑭	⑯ $\frac{P_{req'd}}{2}$ ⑮	⑰ M.S.
3.01.112	3.01.490									
3.00	1.50	4.50	202.5	-2.04	4.16	206.6	14.33	16.41	8.21	-0.15
3.37	1.92	5.90	153	-2.93	8.60	161.6	12.73	15.71	7.86	-0.11
3.20	2.50	8.00	110	-3.54	12.50	122.5	11.03	14.63	7.32	-0.04
3.33	3.03	10.65	72.7	-4.10	16.80	96.5	9.84	13.94	6.97	0.00
3.45	3.73	13.10	50.4	-4.70	22.00	73.3	8.57	13.26	6.63	0.05
3.71	4.33	15.40	36.4	-5.27	27.70	54.5	7.52	12.79	6.40	0.03
3.60	4.50	16.20	33.4	-5.59	31.00	46.4	6.81	12.33	6.20	0.13
3.50	4.45	15.60	6.45	-5.70	32.40	38.9	6.24	11.94	5.97	0.17
3.56	4.26	15.20	1.55	-5.45	33.00	34.4	5.87	11.62	5.81	0.23
3.59	3.93	13.90	0.066	-5.25	34.20	34.3	5.86	11.71	5.86	0.13
3.45	3.60	12.35	3.12	-5.75	33.00	36.1	6.02	11.77	5.89	0.17
3.35	3.10	10.33	11.04	-5.64	31.70	43.4	6.61	12.25	6.13	0.14
3.22	2.65	8.44	23.10	-5.19	30.00	53.1	7.30	13.19	6.59	0.02
3.15	2.41	7.15	44.3	-5.30	28.00	72.8	8.54	13.84	6.92	0.01
3.15	2.41	7.15	49.3	-4.45	23.50	71.9	8.49	13.33	6.67	0.05
3.15	2.27	7.15	51	-4.25	18.00	69	8.32	12.57	6.29	0.11
3.15	2.27	7.15	52.5	-2.52	6.34	58.8	7.67	10.19	5.20	0.34
3.15	2.41	7.15	52	-1.91	3.50	55.5	7.45	9.32	4.66	0.50
3.15	2.41	7.15	50.4	-1.40	1.96	52.4	7.25	8.65	4.33	0.62
2.28	2.55	6.22	37.5	-0.84	0.71	38.2	6.19	7.03	3.52	0.23
3.33	3.10	10.33	20.3	-0.10	0.01	20.3	4.51	4.61	2.34	1.33
3.13	3.00	12.00	2.55	0.52	0.27	9.0	3.15	2.63	1.32	4.30
3.00	3.09	13.00	2.01	1.01	1.02	3.9	1.93	0.07	0.40	13.30
3.16	2.22	11.20	0.001	1.40	1.96	2.1	1.45	0.05	0.03	232.0
3.00	1.17	10.50	1.295	1.74	3.02	4.3	2.08	0.34	0.17	40.20
3.00	2.50	14.20	5.05	1.33	3.71	10.7	3.29	1.35	0.68	9.30
3.57	2.00	14.40	19.4	1.33	3.71	21.0	4.69	2.76	1.38	4.07
3.00	3.13	19.18	33.8	1.67	2.79	41.6	6.46	4.79	2.40	1.32
3.33	3.05	16.45	62.6	1.14	1.30	69.9	8.36	7.22	3.61	0.34
2.00	2.00	8.00	100	0.59	0.35	100.4	10	9.41	4.71	0.43
3.01	1.92	5.90	146.5	-0.04	0.00	146.5	12.1	12.14	6.07	0.15
3.00	1.50	4.50	200	-0.96	0.92	200.0	14.15	15.11	7.56	-0.08

2

WING ANALYSIS

Maximum Stress - Combined Loading

$n_x = a_p + n_a = a_p + f_a + f_b$

$n_y = b_p + n_b = b_p + 0 = b_p$

Maximum Tensile Stress

$$f_{max} = \frac{n_x + n_y}{2} \pm \sqrt{\left(\frac{n_x - n_y}{2}\right)^2 + g^2}$$

$$M.S. = \frac{\text{Allowable Limit Strength}}{f_{max}} - 1$$

Inflation Stress

$f_{inflation} = b_p$

$$M.S. = \frac{\text{Allowable Inflation Strength}}{b_p} - 1$$

Tensile Stress Check

Condition C_2 Element 20 Critical

$$f_{max} = \frac{29.91 + 26.50}{2} \pm \sqrt{\left(\frac{29.91 + 26.50}{2}\right)^2 + 11.29^2} = 39.60 \text{ \#/in.}$$

$$M.S._{limit} = \frac{174 \div 3}{39.60} - 1 = +0.46$$

Ref. Pg. 1.00.080 For 174 #/in Allowable

Inflation Stress Check

Element 26 Critical

$b_p = 4.50 \times 7 = 31.50$ #/in.

$$M.S._{inflation} = \frac{174 \div 4}{31.50} - 1 = +0.39$$

Static Test And Wind Tunnel M.S. Comparison

Ref. Pg. 1.00.050 For g's.

Static Test M.S.

$$M.S._{ult.} = \frac{5.60}{2.5 \times 1.75} - 1 = +0.28$$

Wind Tunnel M.S.

$$M.S._{ult} = \frac{5.13}{2.5 \times 1.75} - 1 = \underline{+0.17}$$

218-63 (3-59)M

FUSELAGE ANALYSIS

Section 4

FUSELAGE ANALYSIS

The fuselage is an inflated conical envelope with nearly hemispherical ends. It supports external loads by virtue of tensile inflation stresses. If one of the principal stresses at a point in the fuselage becomes zero due to applied compression stresses, a wrinkle starts to form there. As more load is applied the wrinkle will enlarge to a point where collapse will occur. The pressure in the fuselage should be large enough to prevent a wrinkle from forming under limit loads and collapse at ultimate loads. For this analysis the ultimate load is 1.75 times limit load.

While the minimum principal stress at the point where a wrinkle first forms is zero, both principal stresses at a diametrically opposite point are tensile stresses. The material properties should be such that the maximum tensile stress does not exceed some allowable value. The allowable strength value on the tension side is the quick break value derived from a cylinder burst test divided by a creep rupture factor of 3. The factor of 3 accounts for the fact that fabric under load for a period of time has a reduction in strength.

The allowable hoop tension strength value on the compression side is the quick break value derived from the cylinder burst test times 0.65 divided by 3. The factor of 0.65 accounts for the fact that the fabric in a cylinder burst test is loaded at a 2-1 stress ratio, while on the compression side of the fuselage is loaded in only one direction and consequently gets little or no help from the bias ply. The factor is based on a comparison of cylinder burst tests and strip tensile tests.

MEMBRANE STRESSES IN A CONE — FUSELAGE ANALYSIS

INFLATION STRESSES

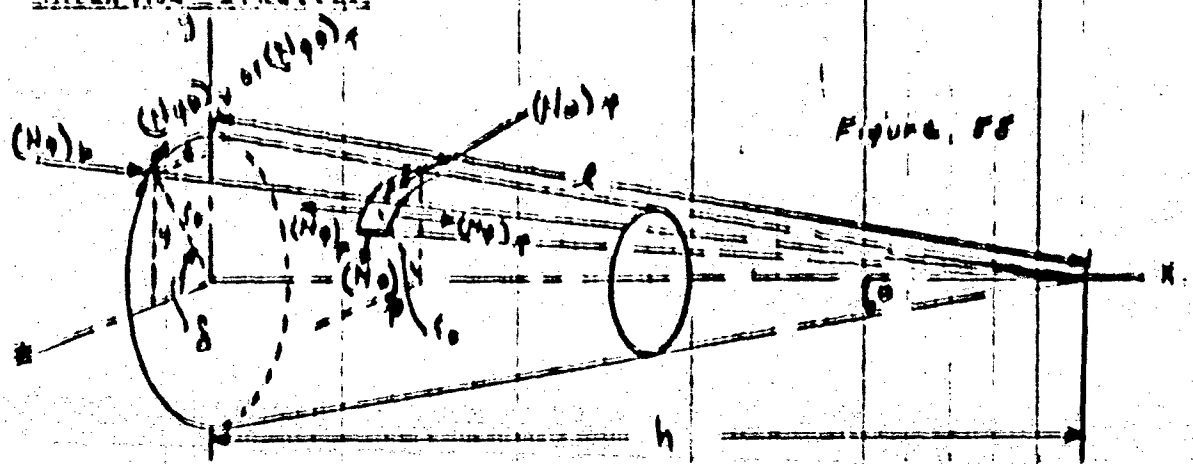

FIGURE 58

THE INFLATION STRESSES $(N_\varphi)_p$ AND $(N_\theta)_p$ ARE GIVEN IN REF. 6 PAGE 437.

$(N_\varphi)_p = \frac{p r_0}{2\cos\theta}$, LB/IN = LONGITUDINAL INFLATION STRESS

$(N_\theta)_p = \frac{p r_0}{\cos\theta}$, LB/IN = HOOP TENSION

WHERE p = INFLATION PRESSURE, PSI.
r_0 = RADIUS OF NORMAL SECTION, IN.
θ = ONE HALF OF ANGLE OF CONE.

BENDING STRESSES

IF THE BENDING STRESS $(N_\varphi)_b$ IS ASSUMED TO ACT ALONG A GENERATOR OF THE CONE THEN ITS X-COMPONENT MUST SATISFY THE FLEXURE FORMULA

$$N_x = \frac{h}{\ell} N_\varphi = \frac{M_z y}{I}$$ WHERE M_z IS THE MOMENT ABOUT THE Z-AXIS.

OR $$(N_\varphi)_b = \frac{M_z y}{I}\frac{\ell}{h} = \frac{M_z y}{I\cos\theta}$$

THE MAXIMUM VALUE OF $(N_\varphi)_b$ IS

$$(N_\varphi)_{bm} = \frac{M_z}{\pi r_0^2 \cos\theta}$$

218-83 (3-5)

BEAM SHEAR STRESSES. FUSELAGE ANALYSIS

IF THE NET SHEAR AT A SECTION IS ALONG THE Y-AXIS, THEN THE SHEAR STRESS IS GIVEN BY THE ELEMENTARY SHEAR STRESS FORMULA TO BE

$$(N_{\varphi\theta})_V = \frac{V \cos\delta}{\pi r_0} \quad \text{LB/IN}$$

TORSIONAL SHEAR STRESSES

THE TORSION OF THIN WALLED SHELLS GIVES

$$(N_{\varphi\theta})_T = \frac{T}{2\pi r_0^2} \quad \text{LB/IN}$$

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Table XXXIV

(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)
CONDITION AND SECTION		LOCATION ON CIRCUM.	M	V	T	T_h (Thrust)	r_o	$(N_\theta)_p$ $\frac{7(4)}{\cos\theta}$	(N_p) $\frac{1}{2}(9)$
Ref. Pg. 2.03.200 & 2.03.210									
			IN-LB	LB	IN-LB	LB	IN		
F-2	SECTION AT POINT O	$\frac{a}{b}$	15480	124	0	-30	12.28	86.0	43.0
F4	SECTION AT POINT O	$\frac{a}{b}$	12700	94	432	-30	12.28	86.0	43.0
F11 & F13	SECTION BETWEEN 7 & 8	$\frac{a'}{b'}$	4520	111	5270	-58	8.62	60.3	30.1
	SECTION AT POINT O	$\frac{a'}{b'}$	15400	72	5270	-58	12.28	86.0	43.0

F-2 OR F4 AT SECTION AT POINT O

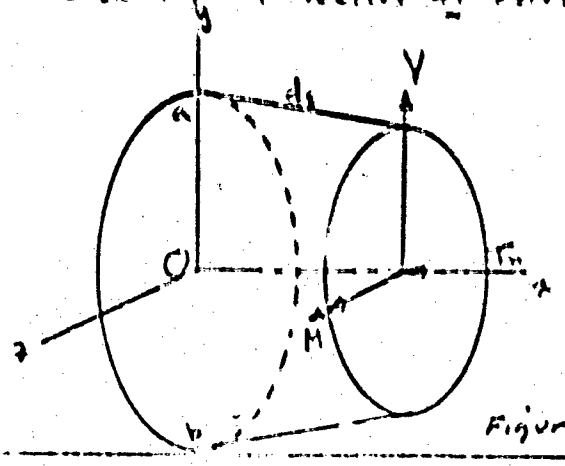
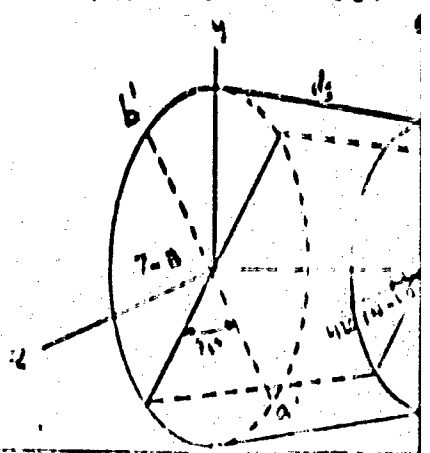


Figure 56a

F11 & F13 SECTION BETWE



$$P_{req} = -\frac{1}{2}\left(\frac{n_a}{a} + \frac{n_b}{b}\right) + \frac{1}{2}\sqrt{\left(\frac{n_a}{a} + \frac{n_b}{b}\right)^2 + 4\left(\frac{q^2}{ab} - \frac{n_a n_b}{ab}\right)}$$

$a \cdot p$ = (10)

$b \cdot p$ = (9)

n_a = (11) + (12)

n_b = 0

q = (17)

F11 & F13 AT SECTION BETWEEN POINTS 7 & 8

$$\frac{n_a}{a} = -\frac{19.4 + 1.1}{4.31} = -4.76$$

$$\frac{q^2}{ab} = \frac{\overline{15.4}^2}{4.31 \times 8.62} = 6.46$$

NOTE: IN COLUMN (17) IT IS CONSERVATIVE TO ADD (13) TO (14)

$$\therefore P_{req} = \frac{4.76}{2}$$

$$= 5.85$$

$$M.S. = \frac{7}{5.85}$$

FROM A MOHR'S

$$N_{max} = \frac{N_o + \ldots}{2}$$

PREPARED BY M.G.G.
CHECKED BY
DATE 1-18-61
REVISED

GOODYEAR AIRCRAFT

PAGE 4.01.050
MODEL GA 468
GER- 7401
REF NO. 377-3

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Table XXXIV

(8)	(9)	(10)	(11)	(12)	(13)	(14)	(15)	(16)	(17)	(18)	(19)
r_0	$(N_\theta)_p$ $\frac{7(11)}{\cos\theta}$	$(N_\phi)_p$ $\frac{1}{2}(9)$	$(N_\phi)_{bin}$ $\frac{(11)}{(8)\pi\cos\theta}$	$(N_\phi)_c$ $\frac{(7)}{(5)2\pi}$	$(N_{\phi\theta})_V$ $\frac{(5)}{(8)\pi}$	$(N_{\phi\theta})_T$ $\frac{(6)}{(5)2\pi}$	N_θ (9)	N_ϕ (10)+(11)+(12)	$N_{\phi\theta}$ (13)+(14) OR (13)	N_{max} PRINCIPAL STRESS	P_{req}
IN	← LB/IN										→ PSI
12.28	86.0	43.0	-32.9 / +32.9	-.4	3.2	0	86.0	+9.7 / 75.5	3.2	86.0	5.42
12.28	86.0	43.0	-26.9 / +26.9	-.4	2.4	.5	86.0	15.9 / 69.5	2.4	86.4	4.49
8.62	60.3	30.1	-19.4 / +19.4	-1.1	4.1	11.3	60.3	9.6 / 47.4	15.4	70.9	5.35
12.28	86.0	43.0	-32.7 / +32.7	-.8	1.9	5.6	86.0	9.5 / 74.9	7.5	89.8	5.60

F13 SECTION BETWEEN POINTS 7 & 8 - - - - - - - - SECTION AT POINT O

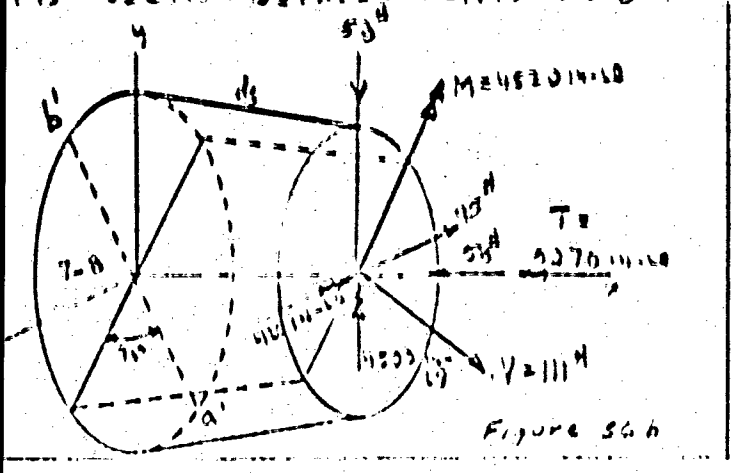

Figure 56b

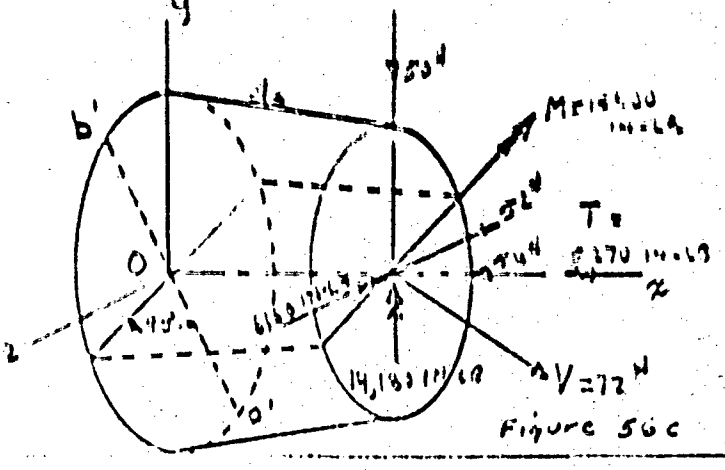

Figure 56c

$$\therefore P_{req} = \frac{4.76}{2} + \frac{1}{2}\sqrt{4.76^2 + 4(6.46)}$$

$$= 5.85 \text{ psi}$$

$$M.S. = \frac{7}{5.85} - 1 = .20$$

FROM A MOHR'S CIRCLE

$$N_{max} = \frac{N_\theta + N_\phi}{2} + \sqrt{\left(\frac{N_\theta - N_\phi}{2}\right)^2 + N_{\phi\theta}^2}$$

FOR F11 & F13 AT SECTION AT POINT O

$$N_{max} = \frac{86 + 74.9}{2} + \sqrt{\left(\frac{86 - 74.9}{2}\right)^2 + 7.5^2}$$

$N_{max} = 89.8$ LB/IN

Inflation stress at Max. Radius

$N_{max} = 7(13.5) = 94.5$ $\quad M.S. = \frac{410}{94.5 \times 4} - 1 = +0.08$

HOOP STRESS MARGIN

$$M.S. = \frac{410 \times .65}{86 \times 3} - 1 = +.03$$

ENGINE MOUNT ANALYSIS

Section 5

ENGINE MOUNT ANALYSIS

An engine saddle mount is attached to the top of the wing by lacing to patch attachments on the top surface of the wing. The saddle mount is also used to fasten the wing to the fuselage, at the trailing edge, by means of two short cable-patch attachments. The engine is attached to the pylon of the saddle structure through the four engine mount fittings. The mount is a weldment of 6061-T6 aluminum tubing heat treated after welding.

218-83 13-5

Summary Loads and Stresses of Engine Mount

Table XXXI

Member and Loading Condition		Section Properties: A (in²)	I_x (in⁴)	I_y (in⁴)	Limit Loads: M_x (in-lb)	M_y (in-lb)	Torque (in-lb)	Axial (lb)	Ultimate Stresses: F_b (psi)	F_s (psi)	f_{axial} (psi)	Allowables: F_b (psi)	F_s (psi)	F_{axial} (psi)	M.S. = Safety
L.H.A. Support (Angle 4130 Sheet) Overall Gusset	n_z=-1 Flight	.3701	.0267	.3690	478	2206	215	-2722	22[illegible]	20[illegible]	-[illegible]	37[illegible]	24[illegible]	3[illegible]	.35
-41 Strut 2.09"x1.19"x.065" Streamline	Side Load	.3951	.0652	.2[illegible]	434	4553	412	118	50[illegible]	2[illegible]	2020	28200	20150	[illegible]	.28
-15, -17, -35, -37 1⅝"x.049" Tubes Together	n_z=-1 Flight	.609		.454		6[illegible]	705		-[illegible]	57[illegible]		50[illegible]	3[illegible]		.80
-53 Tubes 1⅝"x.049"	n_z=-1 Flight	.1764	.84[illegible]	.0106				2303			2[illegible]			[illegible]	.75
-55 Tubes 1"x.049"	n_z=-1 Flight	.1764	.0122	.0[illegible]				-2134			82[illegible]			[illegible]	.[illegible]
-29 Tube Ring 1"x.065"	n_z=+3.0	.[illegible]	.02[illegible]	.02[illegible]	1361	443	1[illegible]		[illegible]	[illegible]		37[illegible]	2[illegible]		.22
-57 Tube 1⅜"x.049"	n_z=3	.1079	.0063	.0063	28	405	405		3[illegible]	[illegible]		32[illegible]	25[illegible]		.28
-5 Tube 1"x.049"	Side Load	.1764	.0[illegible]	.02[illegible]	767	[illegible]	352		21[illegible]	20[illegible]		5[illegible]	3[illegible]		.48

Note: Changes in some members have not been accounted for as the section properties as larger margins would result.

Ref. Dwg. 447A-011. For loads see pg. 2.04.000.

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Checked by [illegible]
Date [illegible]
Revised [illegible]

GOODYEAR AIRCRAFT

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Date [illegible]
Rev No. [illegible]

SUMMARY OF

EMPENNAGE, COCKPIT, AND LANDING GEAR ANALYSES

Section 6

Summary of Empennage, Cockpit, and Landing Gear

Description of Empennage

The horizontal tail is supported along its center line on the fuselage and by cables attached to outboard edges. The main support cables are the aft cables which are attached near the hinge line of the elevator while forward cables are added for stability. Thus half of the horizontal stabilizer acts as a cantilever beam supported at the end.

The vertical tail is hinged to the horizontal tail and to the fuselage and also supported by cables; the main cable being attached near the rudder hinge line and a forward cable added for stability is attached to the leading edge. As the largest tail loads are applied to the vertical tail and as the vertical tail is not supported as well as the horizontal tail, the vertical tail is the most critically loaded part of the empennage.

Description of Cockpit

The cockpit is made up of flat sections of Airmat, three inches thick, consisting of two side panels, a bottom panel, a seat bulkhead, and a rear bulkhead. These panels are joined to form the cockpit which attaches to the fuselage section. A hammock type seat is provided for the pilot, which is cemented to the top of the rear bulkhead and seat bulkhead.

Description of Landing Gear

A single wheel landing gear is used on the aircraft. The structure supporting the wheel is a weldment of 6061-T6 aluminum tubing heat treated after welding, and is attached by lacing to patch attachments on the forward end of the fuselage. This structure is designed to provide a shock absorber action during landing. The wing lower brace cables are attached to the channel support structure at the rear of the landing structure.

Stress Analysis of Empennage

The empennage is an Airmat structure and supports the external loads by making use of the tensile inflation stress. The pressure in the empennage should be large enough to prevent collapsing due to the applied compression stresses under ultimate loads. On the tension side the allowable strength value is the quick break strength derived from a cylinder burst divided by the reduction factors shown on page 1.00.110.

STRESS ANALYSIS OF EMPENNAGE

FROM PG 2.05.060 THE UNIT SOLUTION GIVES

SHEAR $V = 114$ LB
MOMENT $M = 245$ IN-LB
TORQUE $T = 477$ IN-LB
SECTION WIDTH = 35.4 IN

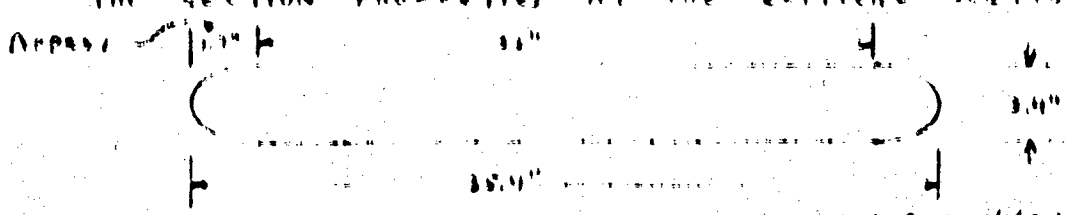
} FOR THE CRITICAL SECTION

FROM PG 2.05.060 THE CRITICAL LOAD, CORRECTED FOR VERTICAL TAIL INERTIA, IS 162 LB; HENCE, THE UNIT VALUES ABOVE BECOME:

CRITICAL CONDITION A 11 RUDDER KICK

$V = 1.62 \times 114 = 18.5$ LB
$M = 1.62 \times 245 = 397$ IN-LB
$T = 1.62 \times 477 = 772$ IN-LB
$P_c = 1.62 \times 55.4 = 90.5$ LB

THE SECTION PROPERTIES AT THE CRITICAL SECTION ARE:



REF DWG 4471A-004

$I = \frac{\pi}{2}(1.7)^2 + 32(1.7)^2(2) = 185.9 \text{ IN}^2$

$A = \pi(1.7)^2 + 32(3.4) = 117.9 \text{ IN}^2$

$S = 3.4\pi + (2)(32) = 74.7$ IN

THE APPLIED STRESSES ARE:

$n_a = \frac{Mc}{I} + \frac{P_c}{S} = \frac{397(1.7)}{185.9} + \frac{90.5}{74.7} = 4.77$ LB/IN

$q = \frac{T}{2A} = \frac{772}{2(117.9)} = 3.28$ LB/IN

EMPENNAGE ANALYSIS

THE INFLATION STRESSES ARE:

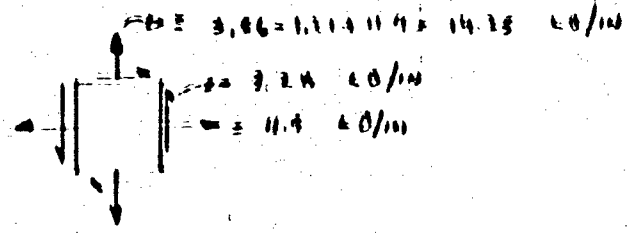
$$a\,p = b\,p = 1.7(7) = 11.9 \text{ LB/IN}$$

$$p_{reg} = +\frac{1}{2}\left(\frac{n_a}{a} + \frac{n_b}{b}\right) \pm \frac{1}{2}\sqrt{\left(\frac{n_a}{a} + \frac{n_b}{b}\right)^2 + 4\left(\frac{g^2}{ab} - \frac{n_a n_b}{ab}\right)}$$

$$= +\frac{1}{2}\left(\frac{4.77}{1.7}\right) \pm \frac{1}{2}\sqrt{\left(\frac{4.77}{1.7}\right)^2 + 4\left(\frac{3.28}{1.7}\right)^2} = 3.80 \text{ PSI}$$

$$M.S. = \frac{7}{3.80} - 1 = .84$$

THE MAXIMUM FABRIC STRESS IS: (ON OPPOSITE SIDE)



$$f_{max} = \frac{14.25 + 11.9}{2} + \sqrt{\left(\frac{14.25 - 11.9}{2}\right)^2 + (3.28)^2} = 16.60 \text{ LB/IN}$$

THE FABRIC FACTORS OF SAFETY ARE:

REF PG. 1.00.080

$$M.S. = \frac{140}{4 \times 11.9} - 1 = 1.96 \quad \text{ON INFLATION STRESS}$$

$$M.S. = \frac{140}{3 \times 16.6} - 1 = 1.81 \quad \text{ON MAXIMUM LIMIT STRESSES}$$

$$M.S. = \frac{140}{1.5 \times 16.6 \times 1.75} - 1 = 2.20 \quad \text{ON MAXIMUM ULT. STRESSES}$$

COCKPIT STRESS ANALYSIS

REF DWG 447A-008

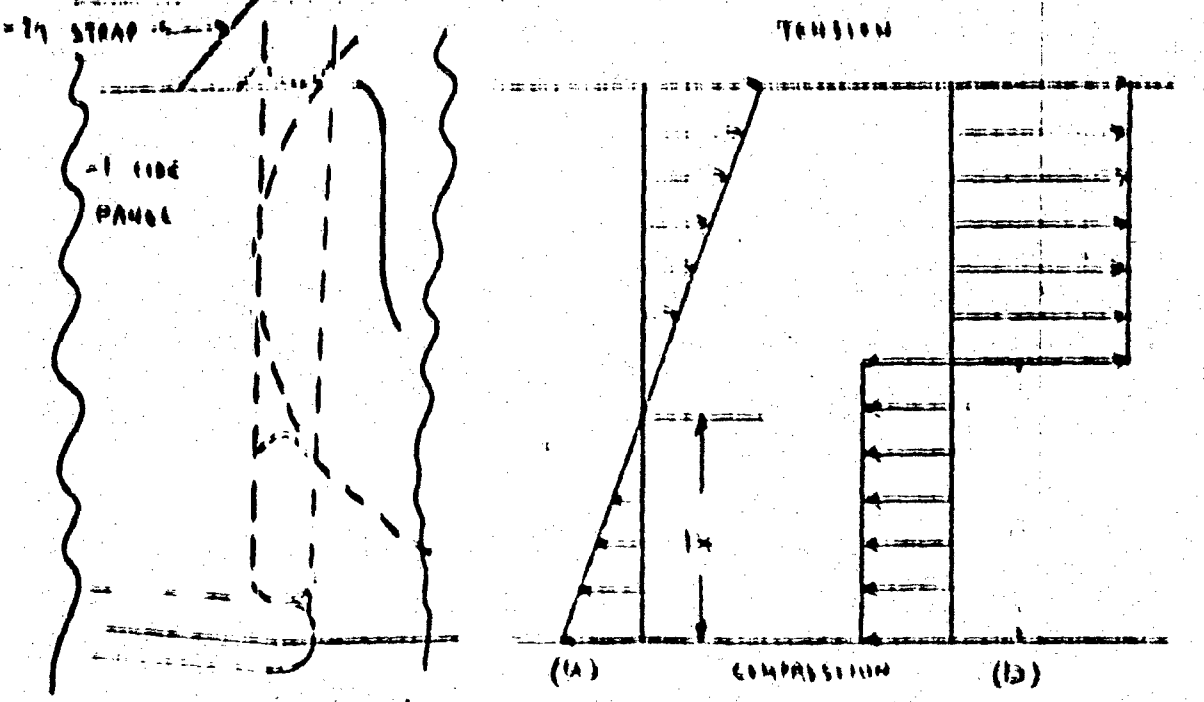


Figure 57

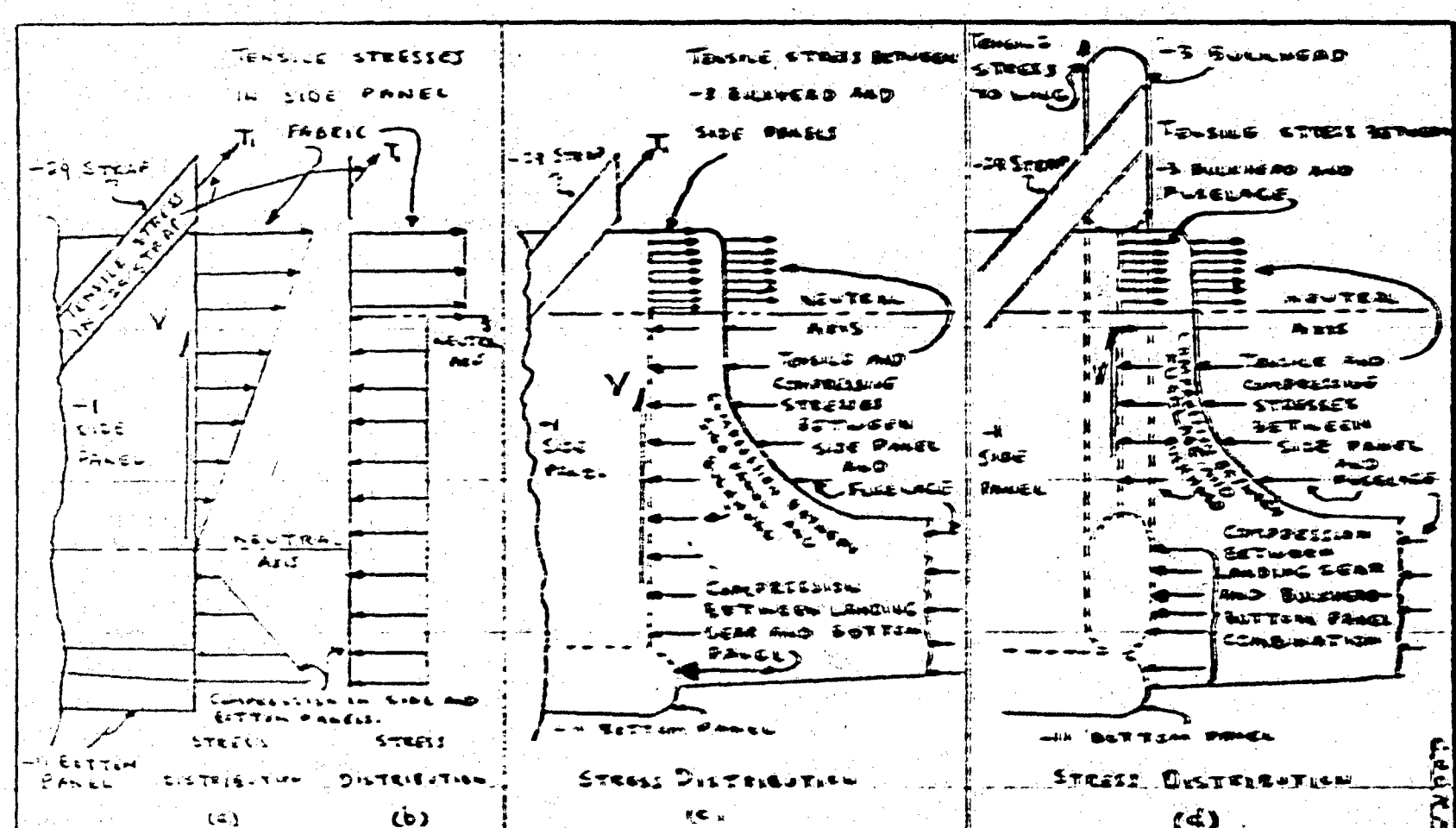
STRESS DISTRIBUTION UP TO WRINKLING OF BOTTOM PANEL

STRESS DISTRIBUTION UP TO COLLAPSE OF COCKPIT

THE ORDINARY BEAM BENDING STRESS DISTRIBUTION IS ASSUMED UP UNTIL THE BENDING STRESSES IN THE BOTTOM PANEL EQUAL THE INFLATION STRESSES. AFTER THE BOTTOM PANEL WRINKLES IT CANNOT CARRY ANY ADDITIONAL COMPRESSIVE LOAD, SO THE SIDE PANELS BEGIN TO WRINKLE PROGRESSIVELY UPWARD. THIS SHIFTS THE NEUTRAL AXIS UP AS SHOWN IN THE SKETCH ABOVE. NEAR THE COLLAPSE LOAD OF THE COCKPIT IT IS ASSUMED THAT THE TENSILE STRESS DISTRIBUTION ABOVE THE NEUTRAL AXIS IS UNIFORM AS SHOWN IN FIGURE (b) ABOVE.

ON THE NEXT PAGE ARE MORE DETAILED SKETCHES SHOWING THE TRANSFER OF LOADS FROM THE COCKPIT SIDE AND BOTTOM PANELS TO THE BULKHEAD, WING, LANDING GEAR, AND FUSELAGE. AS LONG AS THE COCKPIT DOES NOT COLLAPSE, THE LOAD CAPABILITIES ARE DEPENDENT UPON THE FABRIC TENSILE STRENGTH AND THE PRESSURE.

218-83 (3-59)M



THE FREE-BODY CONSISTS OF THE COCKPIT JUST FORWARD OF THE -3 BULKHEAD

THE FREE-BODY CONSISTS OF THE COCKPIT SIDE PANELS, BOTTOM PANEL, THE -29 STRAP; BUT NOT THE BULKHEAD

THE FREE-BODY CONSISTS OF THE COCKPIT SIDE PANELS, BOTTOM PANEL, AND BULKHEAD. THE -29 STRAP FORCES ARE INTERNAL.

Figure 5B

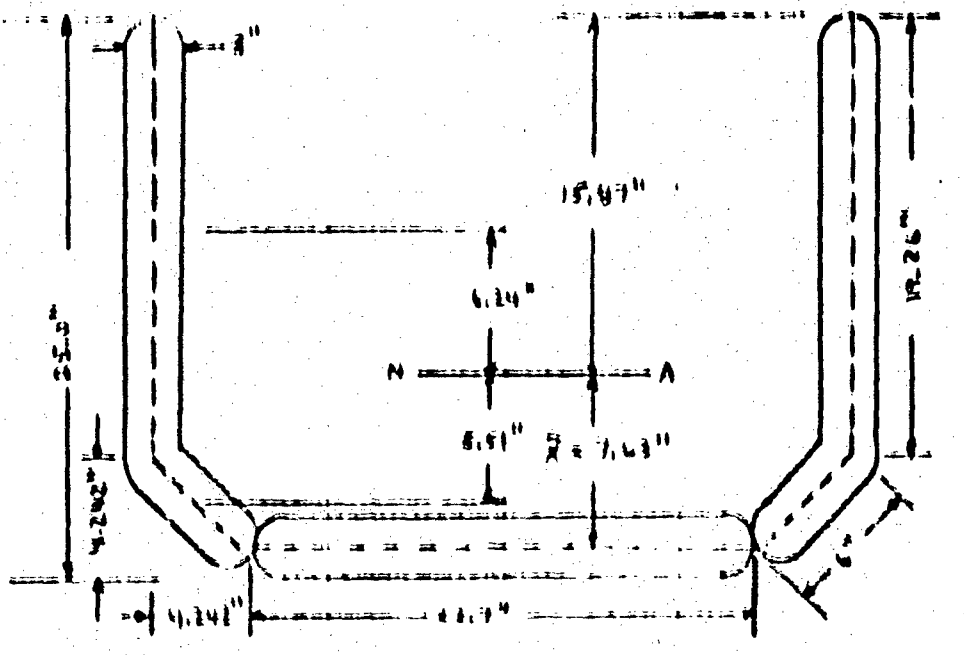
GOODYEAR AIRCRAFT

PREPARED BY N.C.C.
CHECKED BY
DATE 1-18-61
REVISED

PLAN 4.01.070
SHEET G.A.46H
SERIAL 4861
REF NO 597-3

COCKPIT ANALYSIS

STRESS DISTRIBUTION (A) — COCKPIT ANALYSIS



SECTION JUST FORWARD OF BULKHEAD

FIGURE 54

SECTION PROPERTIES USING CENTERLINE DIMENSIONS

$$\left[(19.26+6)2 + 22.7\right]\bar{Y} = 19.26\left(4.24 + \frac{19.26}{2}\right)2 + 6(2.12)(2)$$

$$\bar{X} = 7.63''$$

$$_{N}I_{A} = \left[\frac{\overline{19.26}^{3}}{12} + 19.26(6.24)^2 + \frac{6^3}{12}(.707)^2 + 6(5.51)^2\right]2 + 22.7(7.63)^2$$

$$= 4395 \text{ IN.}^2$$

SECTION MODULUS FOR BOTTOM PANEL = $\frac{4395}{7.63}$ = 577 IN^2

" " " TOP OF SIDE PANEL = $\frac{4395}{15.87}$ = 277 IN.^{2}

WRINKLING STRESS OF BOTTOM PANEL = (3)(1)(7) = 21 #/IN.

WRINKLING MOMENT = (21)(577) = 12,120 IN-LB.

THIS NEGLECTS THE CONTRIBUTION T_1 OF THE -29 STRAP

STRESS DISTRIBUTIONS (M) AND (C) — COCKPIT ANALYSIS

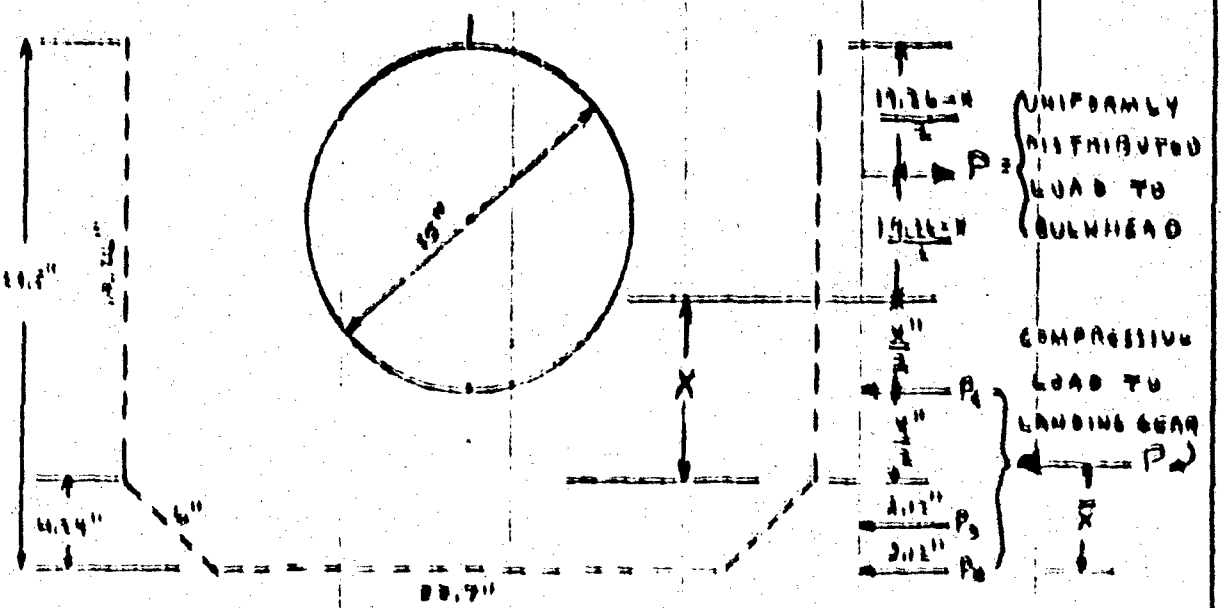


Figure 60

LET p = INFLATION PRESSURE

$P_B = 3(33.7)p = 68.1p$, LB = WRINKLING LOAD OF BOTTOM PANEL

$P_s = 2[3(6)]p = 36.0p$, LB = " " " SLOPING PORTION

$P_c = 2[3(x)]p = 6xp$, LB = " " " X" OF SIDE PANEL

$P = (68.1 + 36.0 + 6x)p = (104.1 + 6x)p$ = COMPRESSIVE LOAD

$$\bar{X} = \frac{2.12(36p) + (4.24 + \frac{x}{2})(P_c)}{P} = \frac{76.36 + 25.45x + 3x^2}{104.1 + 6x}$$

$$M = \left(23.5 - \frac{19.26 - x}{2} - \bar{x}\right)P$$

or $$\frac{M}{p} = 1368 + 109.8x$$

THE UNIFORMLY DISTRIBUTED TENSION IS $T_u = \dfrac{P}{(19.26 - x)(2)}$ LB/SIDE PANEL

OR $$\frac{T_u}{p} = \frac{104.1 + 6x}{2(19.26 - x)}$$ LB/IN OF SIDE PANEL

THESE SAME EQUATIONS DESCRIBE STRESS DISTRIBUTION (C) AT LONG AS THE FORCES P, P_c, P_s, & P_B ARE THE FORCES IN THE CONNECTIONS

STRESS DISTRIBUTION (d) COCKPIT ANALYSIS

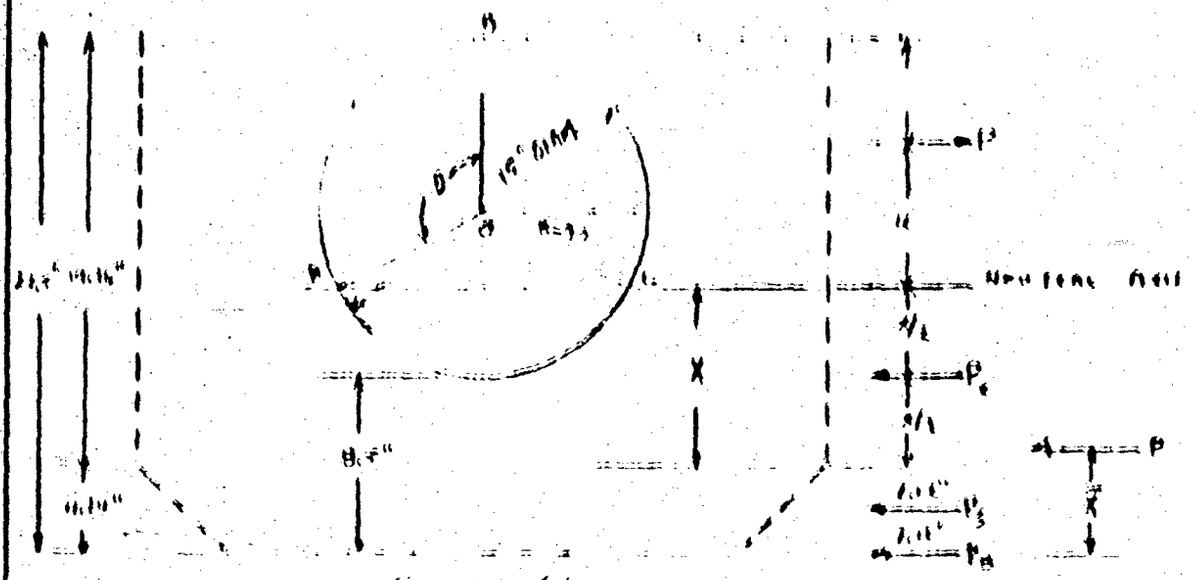


FIGURE 61

$P_B = 104.1\,p$, LB
$P_S = 36.0\,p$, LB
$P_E = 6.4\,p$, LB

THESE FORCES ARE NOW THE COMPRESSIVE FORCES BETWEEN FUSELAGE AND BULKHEAD, SIDE PANEL AND FUSELAGE, AND BULKHEAD AND LANDING GEAR. THEIR NUMERICAL VALUES ARE NOT ASSUMED TO EXCEED THE WHIRLING LOAD OF THE BOTTOM AND SIDE PANELS.

THE UPPER TENSILE LOAD P IS ACTUALLY DISTRIBUTED ALONG THE ARC $\widehat{ABC}$ AND BETWEEN THE SIDE PANELS AND FUSELAGE. THE PORTION ALONG THE ARC $\widehat{ABC}$ ALSO CONTAINS THE EFFECT OF THE STRAP LOAD T_1. IN THE COMPUTATIONS BELOW ALL OF THE TENSILE LOAD IS ASSUMED TO BE ALONG THE ARC $\widehat{ABC}$

u = DISTANCE TO CENTROID OF $\widehat{ABC}$ = $7.5\left(\frac{\sin\beta}{\beta} - \cos\beta\right)$

$x = 8.5 + 7.5 - 4.24 - 7.5\sin(90-\beta) = 11.76 + 7.5\cos\beta$

$\bar{X}$ = SAME VALUE AS FOR (b) & (c)

$M = (u + x + 4.24 - \bar{x})P$

$P = (104.1 + 6.4)p$ AS BEFORE

$T_u = \frac{P}{15\beta}$ LB/IN

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SUMMARY OF EQUATIONS AND SAMPLE CALCULATIONS

STRESS DISTRIBUTION (a)

SEE PG. 6.01.060

CENTERLINE STRESS = $\frac{MC}{I_a} = S$ LB/IN

FABRIC STRESS = $\frac{S}{2}$, LB/IN

INFLATION STRESS = $\frac{p \times \text{PANEL THICK}}{2}$

$= 1.5p$ LB/IN

NET FABRIC TENSION = $\frac{S}{2} + 1.5p$ LB/IN

THE LIMIT MOMENT FOR CONDITION 1 PG 2.06.030 IS 11,125 IN-LB

$S = \frac{11{,}125 \times 15.87}{4395} = 40.2$ LB/IN

NET TENSION IN COCKPIT FABRIC $= \frac{40.2}{2} + 1.5(7) = 30.6$ LB/IN

TENSION IN STRAPS BETWEEN BULKHEAD AND SIDE PANELS

$= 40.2$ LB/IN

STRESS DISTRIBUTIONS (b) AND (c)

SEE PG 6.01.070

$P = (104.1 + 6x)p$ LB

$\bar{x} = \frac{76.36 + 25.45x + 3x^2}{104.1 + 6x}$ IN

$M = (1368 + 100.5x)p$ IN-LB

$T_u = \frac{P}{2(9.2 - x)}$ LB/IN

= STRESS IN CONNECTIONS, I.E. STRESSES (c) ARE DIVIDED BETWEEN BULKHEAD AND FUSELAGE CONNECTIONS

FABRIC TENSION IN SIDE PANEL $= \frac{T_u}{2}$ LB/IN

NET FABRIC TENSION $= \frac{T_u}{2} + 1.5p$

FOR $x = 10''$, $p = 7$ PSI

$P = 1149$ LB

$\bar{x} = 3.85$ IN.

$M = 17{,}260$ IN-LB

$T_u = 62.0$ LB/IN

$\frac{T_u}{2} = 31.0$ LB/IN

NET TENSION = $31 + 10.5 = 41.5$ LB/IN

STRESS DISTRIBUTION (d)

SEE PG. 6.01.070

P AND $\bar{x}$ AS IN (b) AND (c)

$u = 7.5\left(\frac{\sin\beta}{\beta} - \cos\beta\right)$

$x = 11.76 + 7.5\cos\beta$

$M = (u + x + 4.24 - \bar{x})P$

$T_u = \frac{P}{15\beta}$

FOR $x = 10''$, $p = 7$ PSI

$P = 1149$ LB

$\bar{x} = 3.85''$

$\cos\beta = -.5346$

$\beta = 180 - 70.4 = 103.6°$

$= \frac{103.6}{57.3} = 1.81$ RADIANS

$u = 5.80''$

$M = 18{,}600$ IN-LB

$T_u = 42.3$ LB/IN

THE EXPRESSIONS FOR M AND T_u ARE PLOTTED VS X ON THE NEXT PAGE

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PAGE: [illegible]
MODEL: GA 460
GER: [illegible]
REP NO.: [illegible]

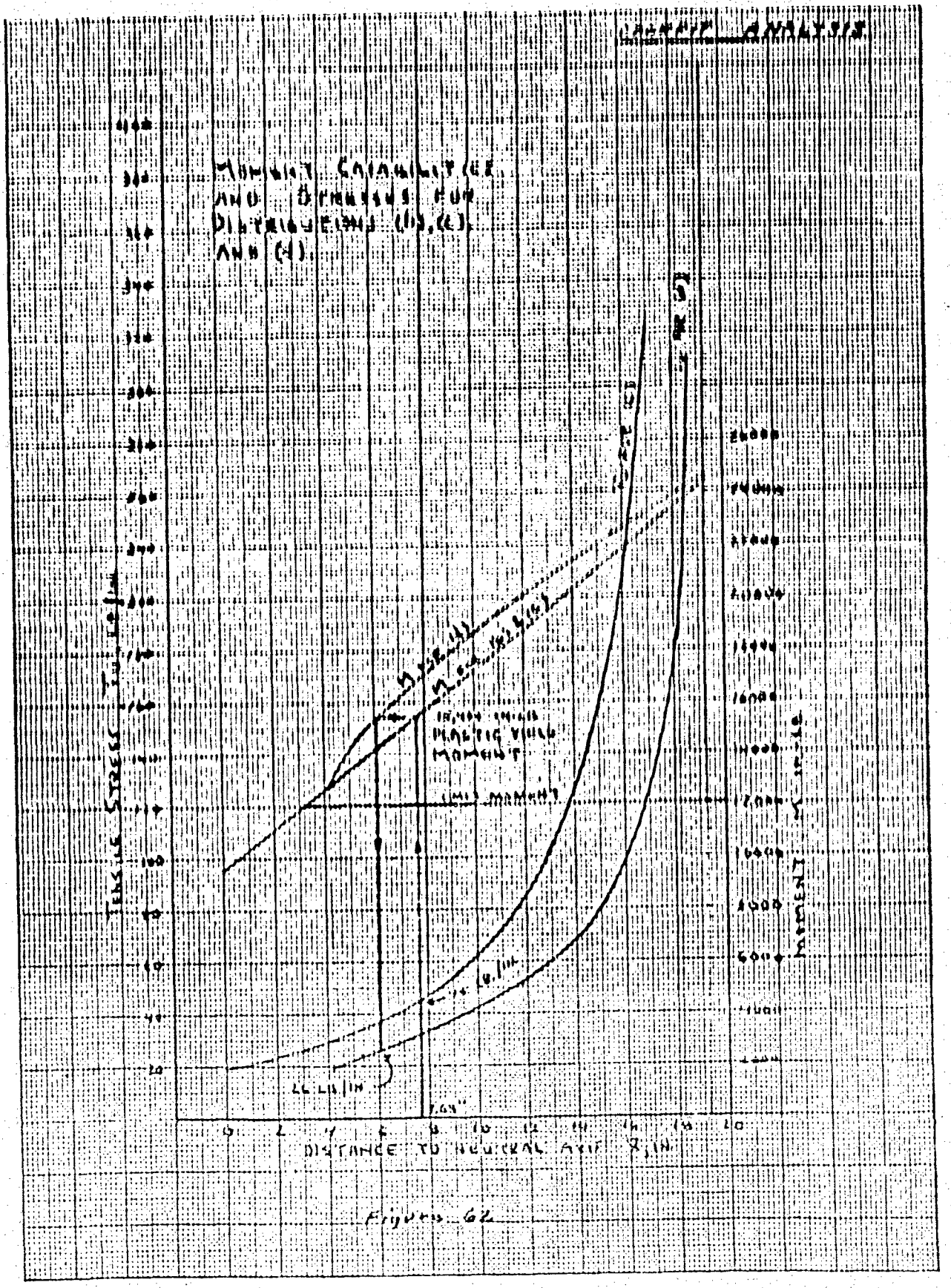

Figure 62

COCKPIT ANALYSIS

From page 6.01.080 the neutral axis for triangular stress distribution (a) is at $\bar{x}$ = 7.63 in. The curves on page 6.01.130 give the moments and stresses vs. neutral axis location for plastic yield or uniform stress distributions (b), (c), and (d). Changing from stresses (a) to (b) or (c) at $\bar{x}$ = 7.63 changes the limit moment 12,120 in-lb (page 6.01.120 to an initial plastic yield moment of 15400 in-lb (page 6.01.120). Thus:

For $M \leq 12120$ in-lb - - - - - - - - - stresses (a)
For $12120 \leq M \leq 15400$ in-lb - - - - transition
For $M \geq 15400$ - - - - - - - - - - - - - stresses (b) and (c)
} stresses (d)

The curves of page 6.01.140 are derived from those of page 6.01.120 by simultaneous values of M and T_u for $15400 \leq M \leq 23000$. The stresses $(T_u)_b$ are calculated from

$$(T_u)_b = \frac{1}{2}(T_u)_c + \frac{3}{2}(7) = \frac{1}{2}(T_u)_c + 10.5 \text{ lb/in.}$$

in which the stress 10.5 lb/in is the cockpit panels inflation stresses.

The fabric factors of safety are three on limit stress, 1.5 on stress at ultimate load, and four on inflation stress. The cockpit panels are made from Airmat fabric A351 with cylinder burst values of 150 x 150 lb/in warp and fill, the bulkhead is fastened to the side and bottom panels with N313A105 fabric straps with strip tensiles of 200 x 180 lb/in warp and fill; and the bulkhead and side panels are fastened to the fuselage with ZX-300 fabric straps with 425 x 350 lb/in warp and fill strip tensiles.

The limit and ultimate moments are given in the table below.

Condition	1	5	6
Limit Moment	11,125	15,773	13,916
Ultimate Moment	19,440*	20,800**	18,400**

* 11,125 x 1.75 = 19,440 In-lb - - - - Using a factor of safety 1.75
** $15,773\sqrt{1.75}$ = 20,800 In-lb - - - - Using a factor of safety of 1.75 on energy of absorption for landing, and noting that loads are proportional to $\sqrt{\text{energy}}$.

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REP. NO. 5[illegible]

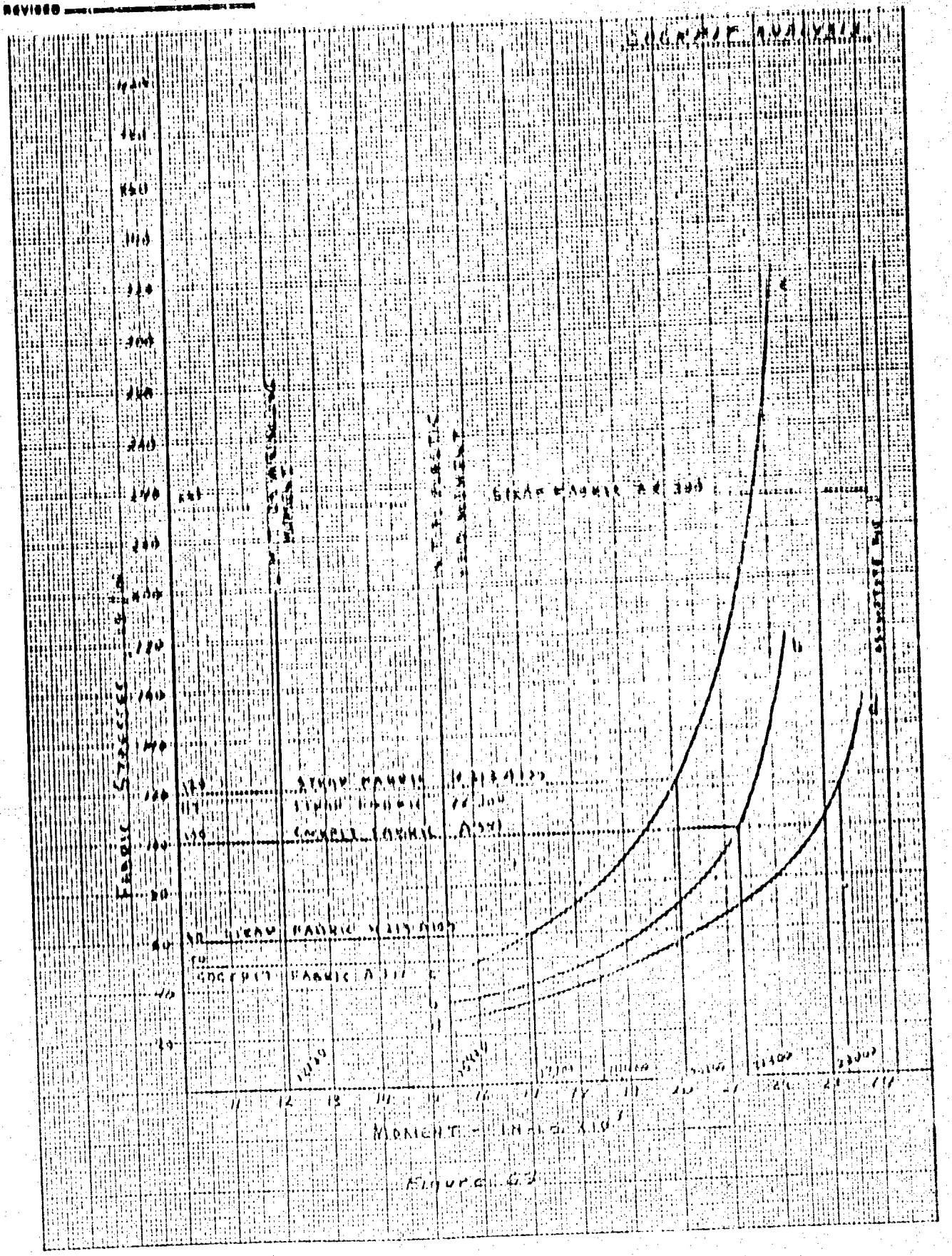

Figure 63

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PAGE 0.01.150
SHEET GA 448
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FABRIC ALLOWABLE STRESSES AND MOMENTS

	F_{tu}	$\frac{F_{tu}}{1.5}$	M_{ult}	$\frac{F_{tu}}{3}$	M_{lim}	INITIAL PLASTIC MOMENT	WRINKLING MOMENT
	LB/IN	LB/IN	IN-LB	LB/IN	IN-LB	IN-LB	IN-LB
COCKPIT FABRIC A351	150	100	21300	50	13400	15400	12120
STRAP FABRIC N313 A105	180	120	20100	60	17100	15400	—
STRAP FABRIC ZX 300	350	233	>23000	117	>23000	—	—

Table XXXVI

FABRIC MARGINS OF SAFETY

	ON ULTIMATE MOMENT			ON LIMIT MOMENT			ON INITIAL PLASTIC MOMENT		
CONDITION	1	5	6	1	5	6	1	5	6
COCKPIT FABRIC A351	.1	.02	.16	.66	.17	.32	.38	-.02	.1
STRAP FABRIC N313 A105	.03	-.02	.09	.54	.07	.23	.38	-.02	.1
STRAP FABRIC ZX 300	>.18	>.1	>.25	>1.07	>.46	>.65	—	—	—

CONDITION	M.S. ON WRINKLING
1	.09
5	-.23
6	-.13

COCKPIT ANALYSIS

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LANDING GEAR ANALYSIS

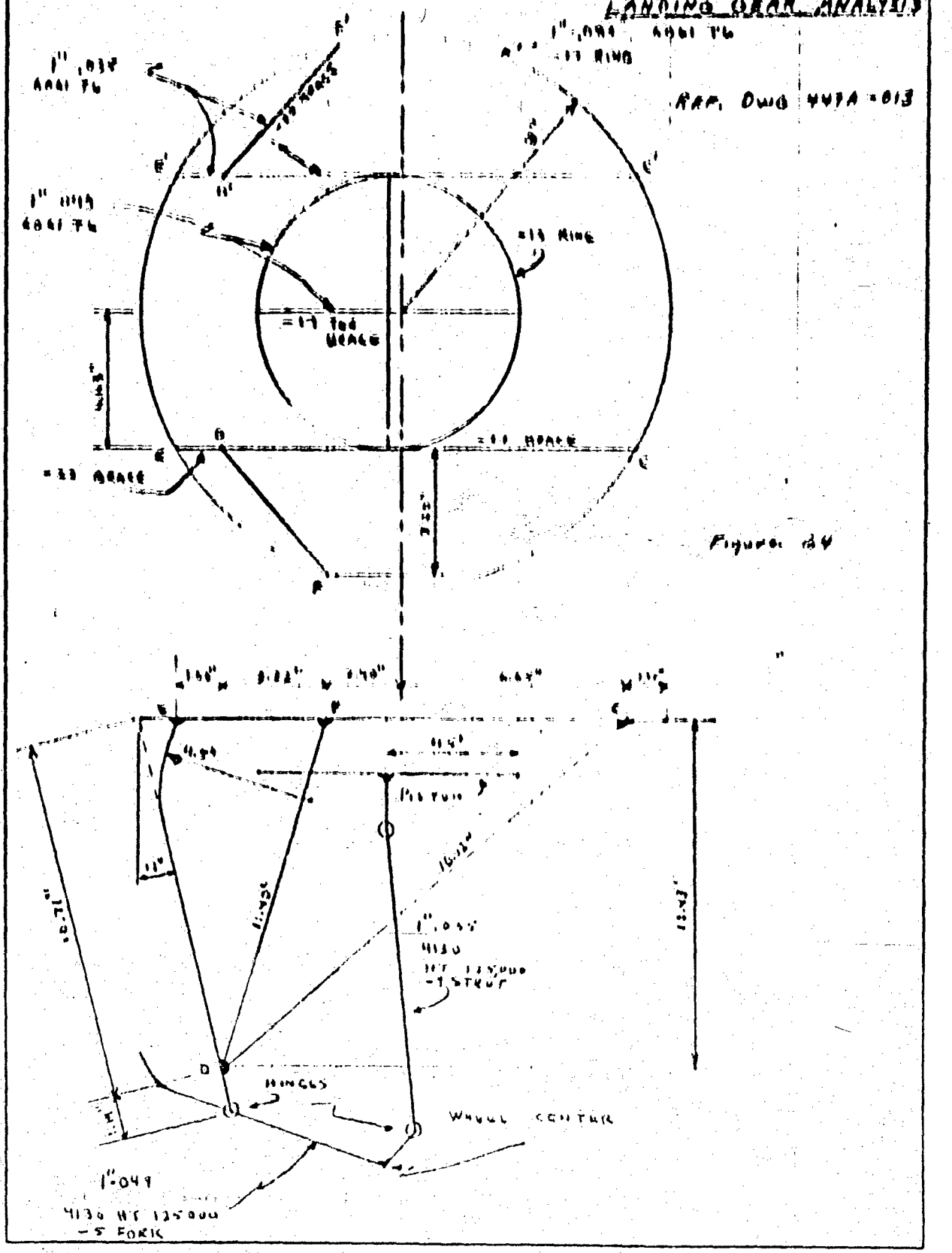

Figure 24

218-83 (3-

218-83 (3-5) A

STRESS ANALYSIS: SUMMARY OF LANDING GEAR - GAC DWG 447A-013

Table XXXVI

MEMBER	SIZE AND MATERIAL	CONDITION	ULTIMATE LOADS			ALLOWABLES			MARGIN OF SAFETY
			THRUST LB	MOMENT IN-LB	TORQUE IN-LB	F_{tu} PSI	F_b PSI	F_c OR F_{ct} PSI	
-23 FWD BRACE	1" .035" 6061-T6	(3)	-537	537	0	42,000	48,700		.64
-30 MIDDLE BRACE		(3)	-1470	0	0	42,000	—		AMPLE
-27 AFT BRACE		(3)	+1480	0	0	42,000	—		AMPLE
-13 RING AND -17 TEE BRACE		(3)	0	4460	0	42,000	51,200		.14
-27 RING	1" .063" 6061-T6	(3)	0	1905	1249	42,000	54,500	26,000	.41
-9 STRUT	1" .035" SAE 4130	(1) & (2)	-990	0	0	125,000	—	102,000	HIGH
-5 FORK	1" .065" SAE 4130	(4)	+1353	3152	0	125,000	152,000	—	.74

LANDING GEAR ANALYSIS

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