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FAIRCHILD ENGINE AND AIRPLANE CORP., FAIRCHILD AIRCRAFT DIV., HAGERSTOWN, MD. (ENGINEERING REPORT NO. R107-014)

BASIC FLIGHT CRITERIA - PACK ON - MODEL XC-120 (M-107)

A. ATHANASIAN; J. POOLE; A. GRENIS AND OTHERS 15 MARCH'49
300PP. GRAPHS, DRWGS

STRUCTURES (7) LOADS, CRITICAL
LOADS AND CRITERIA (1) AIRPLANES - AERODYNAMICS
C-120

RESTRICTED

REPORT NO. R107-014

TITLE BASIC FLIGHT CRITERIA - PACK ON

MODEL M-107

RESTRICTED

FAIRCHILD AIRCRAFT

HAGERSTOWN, MARYLAND

DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION

ENGINEERING REPORT NO.

R107-014

RESTRICTED

SUBJECT

BASIC FLIGHT CRITERIA - PACK OF

LEGIBILITY POOR

MODEL: XC-120 (M-107)

FAIRCHILD AIRCRAFT

Division of

FAIRCHILD ENGINE & AIRPLANE CORPORATION

HAGERSTOWN, MARYLAND

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REVISIONS

Date	Pages Affected	By	Remarks

TABLE OF CONTENTS

	Page No.
PART I. GENERAL	1
A. INTRODUCTION	1
B. NOMENCLATURE	3
C. SUMMARY OF CRITICAL LOADINGS FOR STRESS ANALYSIS	6
1. Symmetrical Flight - Pitch	7
a. Wing	7
b. Ailerons	8
c. Aileron Tabs	8
d. Flaps	8
e. Horizontal Tail	8
f. Elevator Tabs	9
g. Pack Nose Loading Door	9
2 Asymmetrical Flight - Yaw	10
a. Vertical Tail	10
b. Rudder Tabs	12
c. Nose Door	12
3. Asymmetrical Flight - Roll	13
a. Wing	13
b. Ailerons	13
c. Aileron Tabs	13
D. AIRPLANE [illegible] DIAGRAMS	14
E. LOADING CONDITIONS AND C.G. LOCATIONS	18
F. SPECIFICATION REQUIREMENTS FOR DESIGN CRITERIA	21
1. Symmetrical Flight	21
a. Complete Airplane	21
b. Wing and Aileron (Neutral)	22
c. Horizontal Tail and Elevator	22
d. Elevator Tabs	23
e. Wing Flaps	23
f. Fuselage, Booms, Cowling, Doors	23
2. Asymmetrical Flight	24
a. Complete Airplane	24
b. Wing and Aileron (Deflected)	24
c. Aileron Tabs	24
d. Horizontal Tail and Elevator	24
e. Vertical Tail and Rudder	25
f. Rudder Tabs	25

TABLE OF CONTENTS

	Page
G. AIRPLANE PHYSICAL CHARACTERISTICS	26
H. REFERENCES	31
PART II. SYMMETRICAL FLIGHT - PITCH	32
A. GENERAL	32
1. Summary of Aerodynamic Coefficients of Component Parts	32
2. Addition of Component Parts to Determine Airplane Characteristics - Pack On - Symmetrical Flight Only	33
3. Angular Relationships	42
4. Summary of Equations for Complete Airplane	45
B. PITCHING MOMENTS OF COMPONENT PARTS	48
1. Wing Unflapped - Power Off	49
2. Wing Flapped - Power Off $\delta_f = 40°$	51
3. Fuselage	53
4. Booms	56
5. Vertical Tails	59
6. Landing Gear	61
7. Propellers (2) - Military Power	63
8. Slipstream Effects	78
a. Wing Unflapped	78
b. Wing Flapped	89
C. PITCHING MOMENTS OF TAILLESS AIRPLANE	98
1. Flaps and Gear Up	99
a. Power Off	99
b. Military Power	103
2. Flaps and Gear Down	123
a. Power Off	123
b. Military Power	127
D. BALANCING TAIL LIFT	135
1. Flaps and Gear Up	137
a. Power Off	137
b. Power On	140
2. Flaps and Gear Down	151
a. Power Off	151
b. Power On	154

TABLE OF CONTENTS

Page No.

E. COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS 159

1. Flaps and Gear Up 160
   a. Power Off 160
   b. Power On 165
2. Flaps and Gear Down 181
   a. Power Off 181
   b. Power On 185

F. COMPLETE AIRPLANE POLAR DIAGRAM 192

G. V-n DIAGRAMS 197

1. Limit Speeds and Load Factors 197
2. Maximum Lift Lines 200
   a. Flaps and Gear Up - Power Off 200
   b. Flaps and Gear Up - Power On 201
3. Gust Load Factors 203
   a. Flaps and Gear Up - Power Off 203
   b. Flaps and Gear Up - Power On 204
   c. Flaps and Gear Down - Power Off 208
   d. Flaps and Gear Down - Power On 210

H. COMPLETE AIRPLANE COMPONENT LOADS - BALANCED CONDITION 212

1. Flaps and Gear Up 212
   a. Power Off 212
   b. Power On 219
2. Flaps and Gear Down, Power Off and On 228

I. WING LOADS 229

1. Flaps and Gear Up 229
   a. Balanced Conditions 229
   b. Special Balance Conditions 231
   c. Special Leading Edge Condition 231
2. Flaps and Gear Down 233

J. AILERON LOADS 234

K. AILERON TAB LOADS 234

L. FLAP LOADS 235

TABLE OF CONTENTS

Page No.

M. HORIZONTAL TAIL LOADS 244

1. Balance Conditions 244
2. Gust Loads - Horizontal Tail 245
   a. Increment Gust Loads 245
   b. Balance Loads 247
   c. Total Horizontal Tail Loads with Gust 256
3. Maneuvering Loads 259
4. Summary of Design Horizontal Tail Loads 263

N. ELEVATOR TAB LOADS 264

1. Elevator Trim Tab 265
2. Elevator Spring Tab 266

O. FUSELAGE, BOOMS, COWLING, DOORS, AND MISCELLANEOUS 267

1. Fuselage and Booms 267
2. Cowling 267
3. Doors 267
4. Front Cargo Doors 267
   a. Method of Analysis 267
   b. Nose Door Load and Moment Coefficients 275
   c. Nose Door Loads and Moments 295

PART III. ASYMMETRICAL FLIGHT - YAW 300

A. GENERAL 300

1. Summation of Yawing Moments About Any C.G. Location - Two Engine Operation 302
2. Summary of Equations for Complete Airplane - Two Engine Operation 307
3. Summation of Yawing Moments About Any C.G. Location - Single Engine Operation 308
4. Summary of Equations for Complete Airplane - Single Engine Operation 312

B. YAWING MOMENTS OF COMPONENT PARTS 313

1. Wing Unflapped 314
2. Wing Flapped 316
3. Fuselage 317
4. Booms 320

TABLE OF CONTENTS

Page No.

5. Horizontal Tail 321
6. Landing Gear 322
7. Propeller 323
8. Slipstream Effects 325
   a. Wing Unflapped 325
   b. Wing Flapped 329
9. Vertical Tail 330
   a. Single Engine Operation 330
   b. Two Engine Operation 339

C. YAWING MOMENT OF THE TAILLESS AIRPLANE 343

1. Zero Yaw - Single Engine Operation 343
2. 5° Yaw - VD = $V\sqrt{\sigma}$ = 313 mph 344

D. VERTICAL TAIL LOADS - TABS NEUTRAL 345

E. RUDDER TAB LOADS 352

1. Rudder Trim Tab 353
2. Rudder Spring Tab 354

PART IV. ASYMMETRICAL FLIGHT - ROLL 355

PART I-A

A. INTRODUCTION

It is the purpose of this report to determine the aerodynamic characteristics of the complete airplane (pack on) and to determine the critical loads on the component parts for design and stress analysis in accordance with the requirements of reference (2).

This report is divided into four parts: Part I consists of general information and includes a summary of the critical loads and design data; Part II covers the determination of the aerodynamic characteristics of the complete airplane (pack on) and the critical loads on the component parts for symmetrical flight, considering pitch only: Part III covers the determination of the aerodynamic characteristics of the complete airplane and the critical loads on the vertical tail surfaces for assymetrical flight, considering yaw only; and Part IV covers the determination of the aerodynamic characteristics of the airplane and the critical loads on the wings and ailerons in assymetrical flight, considering roll only. The four parts combined form one report, "Basic Flight Criteria", Volume I, Pack On. It should be appreciated that this report, Volume I, covers only pack-on configuration. Volume II of this same report will be issued at a later date and will cover only pack-off configuration.

Throughout this report all references to power conditions means military power. Flap down conditions are analyzed for 40° flap deflection only.

The XC-120 airplane is a modified version of the C-119B airplane and features a detachable cargo compartment with both front and rear loading provisions, which is called the "pack". The compartment provided for pilot and crew members is of course not detachable and is designated "crew nacelle". The fuselage consists of the "pack" plus the "crew nacelle". Due to the peculiar configuration required for the detachable "pack" a quadricycle landing gear is required. Wing incidence has been increased from 3° (C-119B) to 7° at the wing root chord.

PART I-A

INTRODUCTION (Cont.)

This report is one of a series of aerodynamic reports in which the characteristics of the component parts are summed and expanded in order to define the operating characteristics of the complete airplane. The complete series of reports are as follows:

Fairchild No.	Model	Title
R107B-001	XC-120	Estimation of Lateral Stability Derivatives.
R107-012	XC-120	Basic Aerodynamic Data
R107-013	XC-120	Maneuvering Horizontal Tail Loads Volume I - Pack On Volume II - Pack Off
R107-014	XC-120	Basic Flight Criteria Volume I - Pack On Volume II - Pack Off
R107-015	XC-120	Air Load Distribution Volume I - Pack On Volume II - Pack Off
R107-016	XC-120	Performance Calculations
R107-017	XC-120	Analysis of Wind Tunnel Data
R107-018	XC-120	Analysis of Stability and Control Volume I - Pack On Volume II - Pack Off

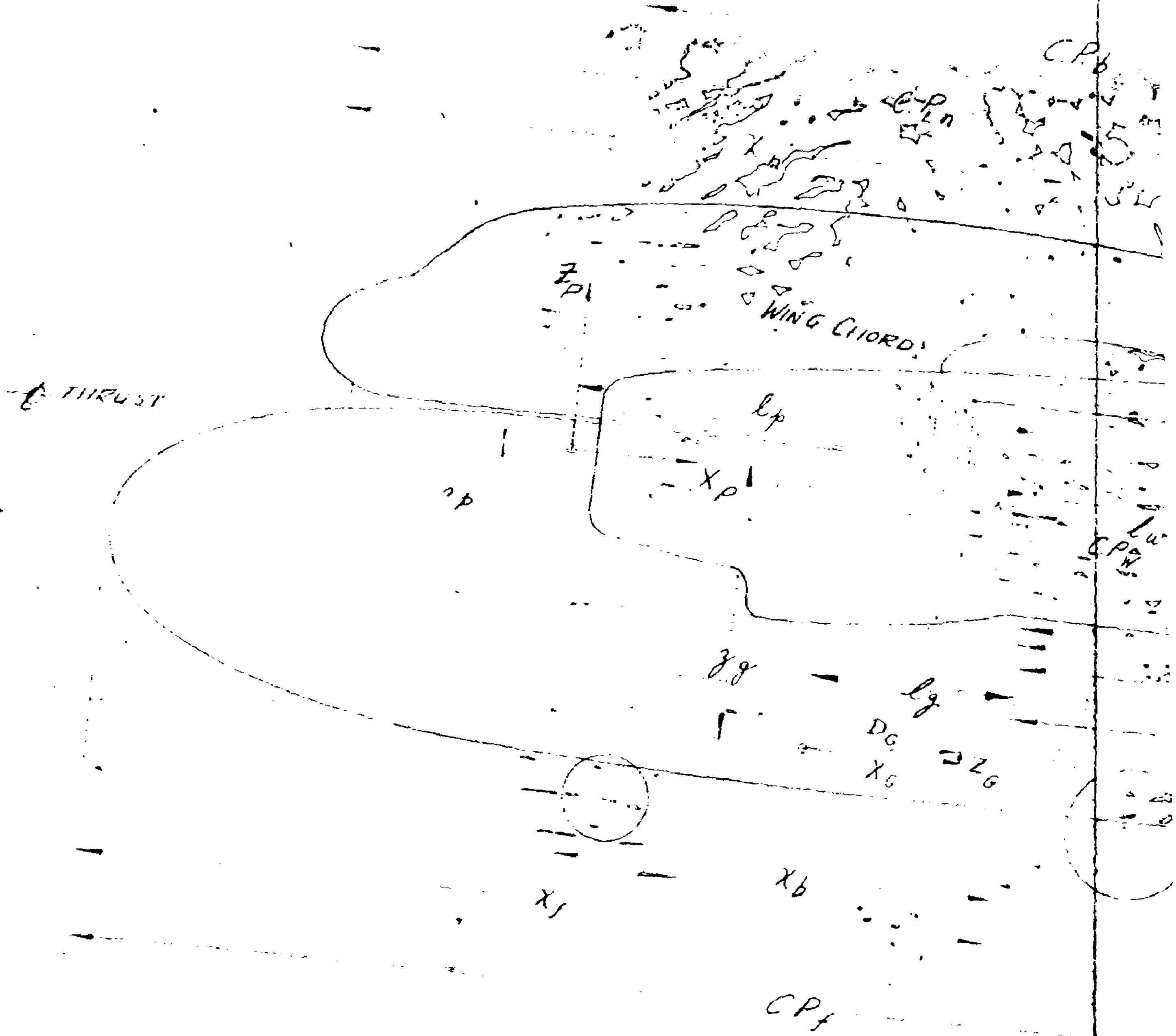
PART I-B

B. NOMENCLATURE

All nomenclature used herein conforms to the nomenclature as used in references (1) and (5).

Where pressure distributions are determined, the nomenclature of reference (4) applies in addition to the above.

Figures 1 and 2 present a physical picture of some of the more important dimensions and distances.


CP_b
Z_p
WING CHORD
℄ THRUST
l_p
X_p
l_w
CP_W
Z_g
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D_G
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X_b
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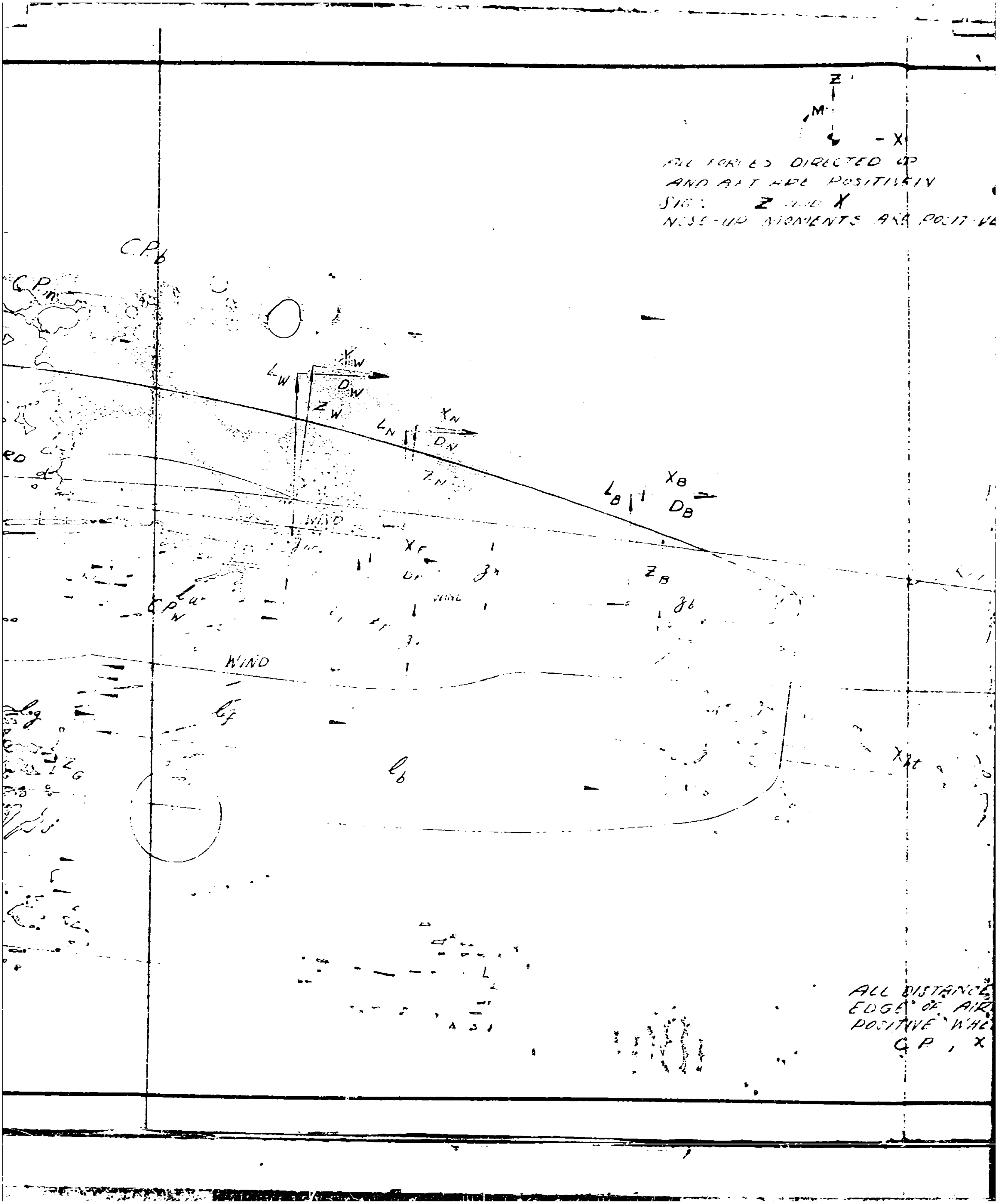
Z
M
X
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Z X
C.P.b
C.P.n
L_W
X_W
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Z_N
L_B
X_B
D_B
X_F
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WIND
ℓ_f
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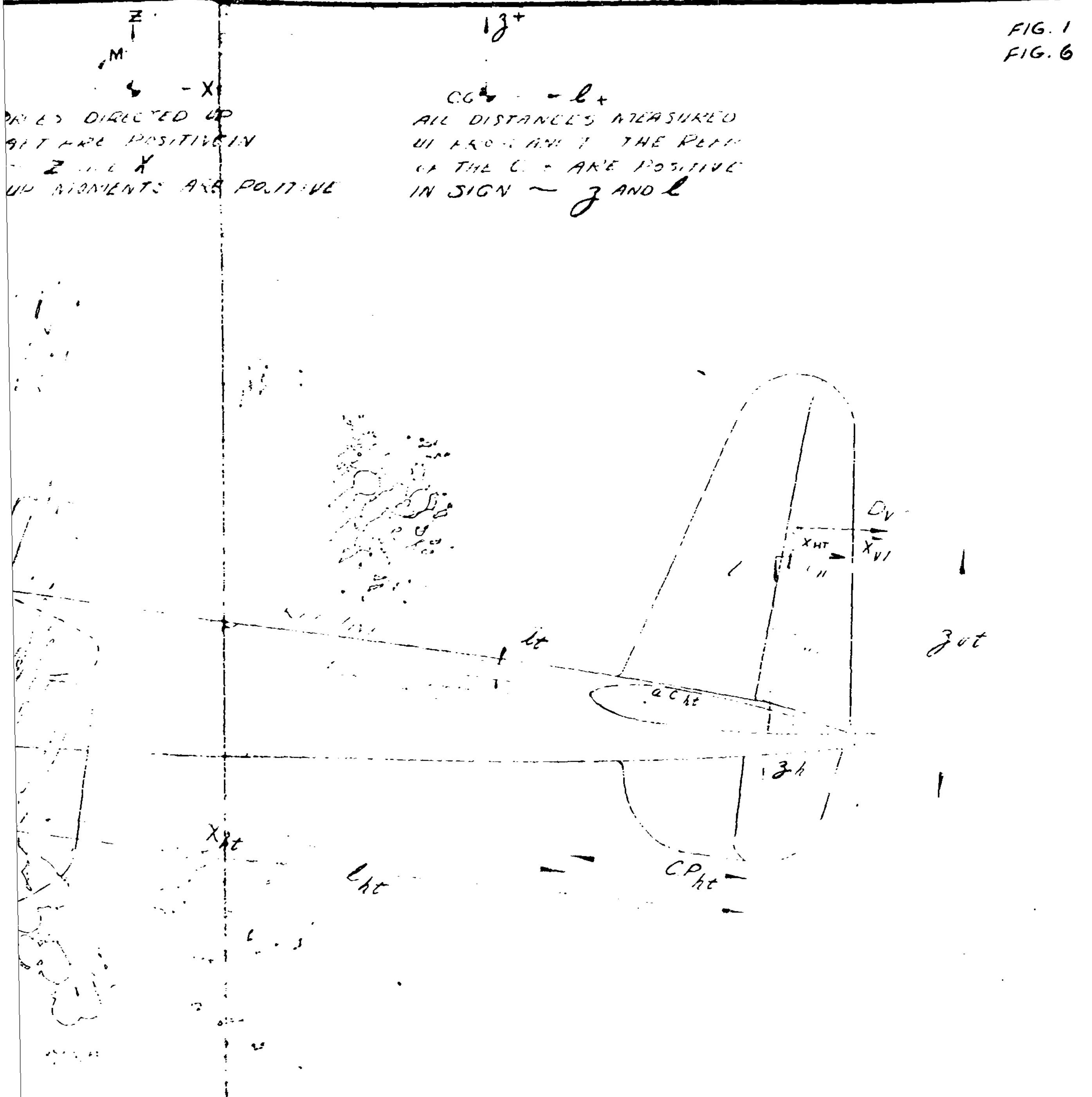
ALL DISTANC
EDGE OF AIR
POSITIVE WHE
C.P, X

FIG. 1
FIG. 6



ALL DISTANCES MEASURED FROM LEADING EDGE OF AIRFOILS OR BODIES ARE CONSIDERED POSITIVE WHEN AFT OF LEADING EDGE — C.P., X, AND a.c.

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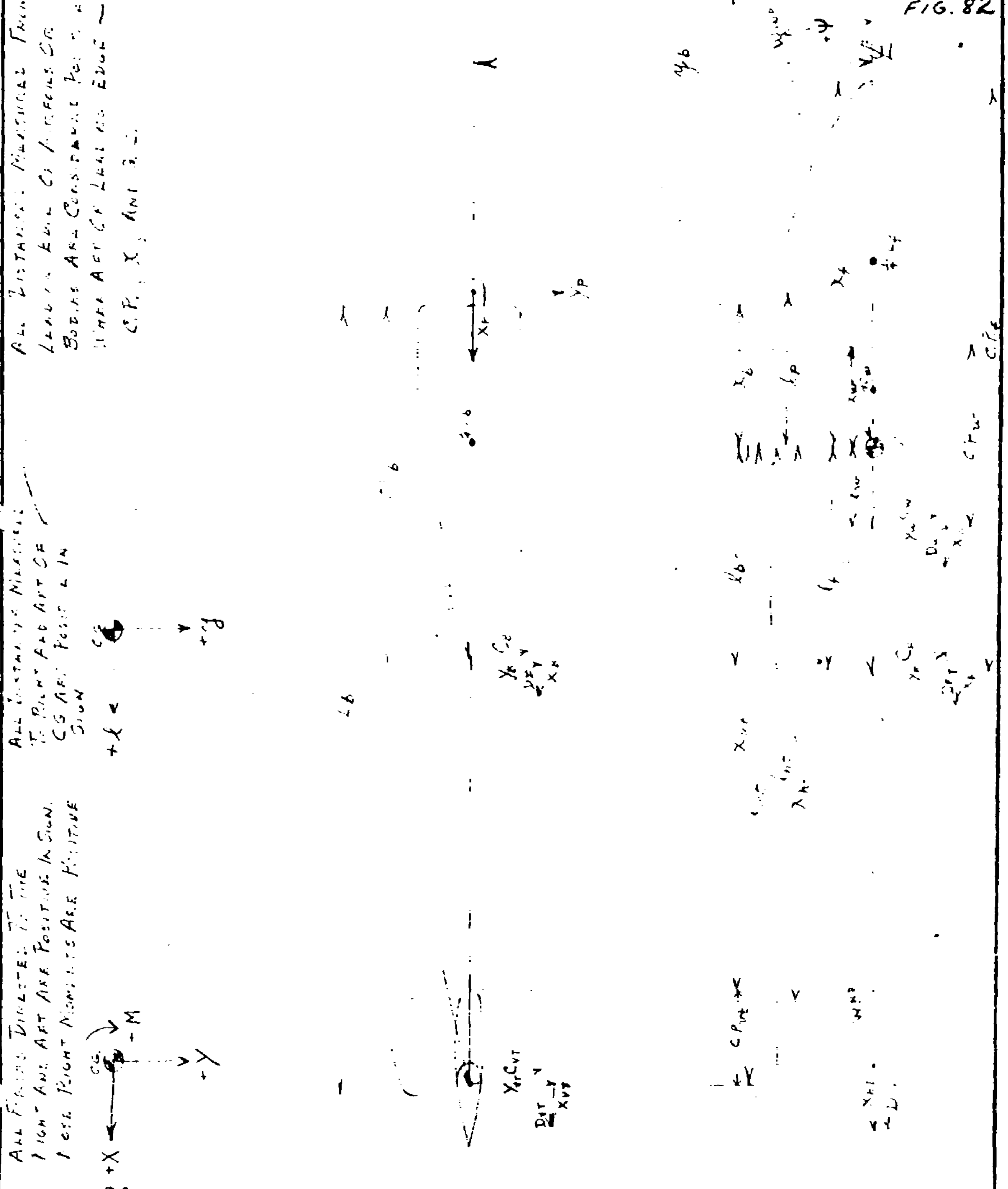
REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION	PAGES	PAGE 5a-301
MODEL XO-120	PREPARED BY	CHECKED BY	APPROVED BY

DATE February 18, 1949

REVISED

Subject:— BASIC FLIGHT CRITERIA

FIG. 2
FIG. 82



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PART I - C

C. SUMMARY OF CRITICAL LOADINGS FOR STRESS ANALYSIS

This section summarizes the critical loads on the component parts as determined in the main body of the report. Spanwise and chordwise pressure distributions for the critical conditions are presented in reference (7).

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

Subject:— BASIC FLIGHT CRITERIA - PACK ON — DATE 15 March 1949 — REVISED

PART I-C-1

1. Symmetrical Flight - Pitch

a. Wing

The wing loads as determined in Part II-I are tabulated below:

Condition	Power	C. G. Loc.	Gross Wt. lbs.	Airplane Load Factor	C_{L_W}	$V\sqrt{\sigma}$ mph	Z wing load lbs.	X wing load lbs.	Moment about A_{C_W} ft.lbs.
1. PHAA	off	3	64000	3.00	1.4010	191.2	189530	-13792	-119530
Possibly critical for 1 Upper surface in compression 2 Lower surface in tension, outer panel. 3 Front spar, outer panel 4 Rear spar, lower cap, center section. 5 Outer panel to center section splice, front spar.									
2. $PLAA_{V_H}$	off	1	64000	3.00	.8403	250	194637	2336	-204528
Possibly critical for 1 Upper surface in compression . 2 Outer panel to center section splice, front and rear spars. 3 Lower surface in tension, center section 4 Front spar, lower cap, center section.									
3. $PLAA_{V_D}$	off	1	64000	2.51	.4799	313	173126	13376	-319575
Possibly critical for 1 Rear spar, outer panel.									
4. NHAA	off	1	64000	-1.50	-.5670	198.9	-79405	-18266	-129358
Possibly critical for 1 Lower surface in compression. 2 Upper surface, outer panel to center section splice.									
5. $NLAA_{V_D}$	off	2	64000	-1.14	-.1204	313	-43354	-4557	-319575
Possibly critical for 1 Outer panel skin splices									
6. $PLAA_{V_H}$ Gust	on	5	42136	3.295	.6130	250	141876	7408	-204528
Possibly critical for 1) Local attachments of weight items									
7. PHAA	off		64000	3.00	1.751	170.2	181800	-19200	-94800
Possibly critical for 1 Special leading edge conditions (apply to forward 20% chord only.)									
8. NLAA	off		64000	-1.5	-.709	176.8	-73900	-17980	-102000
Possibly critical for 1 Special leading edge conditions (Apply to forward 20% chord only)									

PART I-C-1

a. (Cont.)

The positive lift coefficient giving the most forward center of pressure corresponds to PHAA conditions in the table on the previous page.

The lift coefficient giving the maximum positive limit load at V_D or V_H corresponds to the PLAA condition.

The distribution giving the maximum negative load on the forward 20% of the wing corresponds to conditions 1 and 4.

Conditions 7 and 8 correspond to the momentary increase in leading edge loadings occuring in accelerated flight on the forward 20% of the wing.

b. <u>Ailerons</u>

The aileron loads in symmetrical flight are those occuring when the ailerons are neutral and are not critical.

c. <u>Aileron Tabs</u>

The aileron trim and balance tab loads are considered under assymetrical flight - roll.

d. <u>Flaps</u>

The average load on the flaps as determine in Part II-L is 140.72 lbs./sq. ft. The load is 159.6 lbs./sq. ft. on the flap area in the slipstream and 107.4 lbs./sq. ft. on the flap area outside of the slipstream.

e. <u>Horizontal Tail</u>

The maximum horizontal tail balancing loads as determined in Part II-H are -13,789 lbs. with power off and -14,873 lbs. with power on under the following conditions.

Condition	Power	C.G. Loc.	Gross Wt. lbs.	Airplane Load Factor	α_{ht} deg.	δ_e deg.	$V\sqrt{\sigma}$	$\frac{q_{ht}}{q}$	Horiz. Tail Load - lbs.
$NLAA_{V_D}$	off	2	64000	-1.14	-7.99	8.13	313	1.00	-13789
$NLAA_{V_D}$	on	2	64000	-1.14	-7.94	7.54	313	1.00	-14873

PART I-C-1

e. Horizontal Tail (Cont.)

The maximum horizontal tail load occuring with a 50 fps gust as determined in Part II-M-2 is -12,660 lbs. with a negative gust and the following conditions.

Condition	Power	C.G. Loc.	Gross Wt. lbs.	Airplane Load Factor	α_{ht} deg.	δ_e deg.	$V\sqrt{\sigma}$	$\frac{q_{ht}}{q}$	Total Horiz. Tail Load-lbs. Gust & Balance
-50 fps gust	on	1	64000	1.0	-7.61	4.63	250	1.00	-12,660

The maximum maneuvering tail loads as determined in reference (8) are -11,131 lbs. and -9,774 lbs.

Condition	Power	C.G. Loc.	Gross Wt. lbs.	Airplane Load Factor	α_{ht} deg.	δ_e deg.	$V\sqrt{\sigma}$	$\frac{q_{ht}}{q}$	Horiz. Tail Load - lbs.
Maneuvering	on	35% MAC_w maximum upper	64000	3.0	4.19	6.28	184	1.081	11,131
Maneuvering	on	20% MAC_w maximum lower	64000	3.0	0.31	-12.85	184	1.081	-9,774

f. Elevator Tabs

The maximum load on the elevator trim tab as determined in Part II-N-1 is 83.6 lbs./sq. ft. of tab area. The maximum load on the elevator spring tab as shown in Part II-N-2 is 73.5 lbs./sq. ft. of tab area.

g. Pack Nose Loading Door

The critical nose door loads in pitch as determined in Part II-O-4 are as follows.

Condition	V mph	α_{TL} deg	ψ deg	F_X lbs.	F_Y lbs.	F_Z lbs.	M_X 10^{-3} in.lbs.	M_Y 10^{-3} in. lbs.	M_Z 10^{-3} in. lbs.
$MLAAV_D$	313	-9.5	0	502	-3410	7200	401	-296	-55.2
Special Latch Load	Apply 1 q load at 313 MPH (V_D) uniformly over door.								

PART I-C-2

2. **Assymetrical Flight - Yaw**

a. Vertical Tail

The critical vertical tail loads as determined in Part III-D are as follows:

The vertical tail load for a side gust of 50 fps with initial angle of attack of the vertical tail zero and zero rudder deflection at V_H is 5346 lbs. per vertical tail

Condition	Power	α_{vt} deg.	δ_r deg.	$V\sigma^{\frac{1}{2}}$ mph	$\frac{q_{vt}}{q}$	Vertical Tail Load - lbs.
50 fps Side Gust	On	7.8	0	250	1.024	5346

For zero yaw with one engine dead and the other engine delivering military power (same as take-off power) the maximum load on the vertical tail in the slipstream of the operating engine is - 1135 lbs. The maximum load on the other vertical tail is - 788 lbs.

Condition	Power	α_{vt} deg.	δ_r deg.	$V\sigma^{\frac{1}{2}}$ mph	$\frac{q_{vt}}{q}$	Vertical Tail Load - lbs.
Zero - Yaw	On Operating Engine Side	0	-15.35	100	1.44	-1135
Zero - Yaw	Off Dead Engine Side	0	-15.35	100	1.00	-788

PART I-C-2 (Cont.)

For zero yaw, single engine operation at maximum speed for this condition, plus a gust of 30 fps, the maximum load on the vertical tail in the slipstream of the operating engine is - 3240 lbs., with the gust load additive. The maximum load on the other vertical tail is - 3070 lbs., with the gust load additive.

Condition	Power	α_{vt} deg.	δ_r deg.	$V\sigma^{\frac{1}{2}}$ mph	$\frac{q_{vt}}{q}$	Vertical Tail Load - lbs.
Zero Yaw - -30 fps Gust	On - Oper. engine side	-6.1	-3.8	190	1.055	-3240
Zero Yaw - -30 fps Gust	Off -Dead engine side	-6.1	-3.8	190	1.00	-3070

For 5° yaw two engine operation with the resulting rudder deflection at V_D the maximum load on each vertical tail is 2040 lbs.

Condition	Power	α_{vt} deg.	δ_r deg.	$V\sigma^{\frac{1}{2}}$ mph	$\frac{q_{vt}}{q}$	Vertical Tail Load - lbs.
5° Yaw	On	4.334	-5.00	313	1.016	2040

For 5° yaw two engine operation with zero rudder deflection at V_D, the maximum load on each vertical tail is 4580 lbs.

Condition	Power	α_{vt} deg.	δ_r deg.	$V\sigma^{\frac{1}{2}}$ mph	$\frac{q_{vt}}{q}$	Vertical Tail Load - lbs.
5° Yaw	On	4.334	0	313	1.016	4580

PART I-C-2

b. Rudder Tabs

The maximum loads on the rudder trim and spring tabs as determined in Part III-E did not exceed the minimum design requirements of 50 lbs. per sq. ft. therefore, the rudder tab loads are the minimum of 50 lbs/sq. ft.

c. Nose Door

The critical nose door loads in yaw as determined in Part II-C-4 are as follows:

Condition	$V\sqrt{\sigma}$ mph	α_{TL} deg.	ψ deg.	F_X lbs.	F_Y lbs.	F_Z lbs.	M_X 10^{-3} in. lbs.	M_Y 10^{-3} in. lbs.	M_Z 10^{-3} in lbs
Level Flight	313	-5.5	+10	-2260	-7530	5525	324	-216	319
Yaw - V_D			-10	2510	753	5525	324	-216	-364

PART I-C-3

3. ASSYMETRICAL FLIGHT - ROLL

a. Wing

The wing must be checked for the moment coefficients of the deflected ailerons at the wing attitude corresponding to that point of the V-n diagram which gives the highest torsional moment on the wing. See Part IV-D.

This condition corresponds to

$$\frac{p\ B_w}{2\ V} = .08 \text{ at } V\sqrt{\sigma} = 200 \text{ mph,}$$

$$\eta = +2 \text{ where wing twist is a maximum}$$

Chordwise and spanwiew distributions for this condition are presented in reference (7).

In addition the wing must be checked for an unbalanced loading condition using 100% of design load on one wing and 85% on the other, considering angular inertia effects. Since this is an arbitrary loading condition not dependent upon true aerodynamic loading - no analysis is necessary in this report.

b. Ailerons

The maximum aileron load, determined in Part IV-D, obtained in meeting the required rates of roll is -1357.6 lbs. and occurs at $V\sqrt{\sigma} = 200$ mph on the right aileron with $\delta_a = -24^\circ$ (left aileron $\delta_a = 12^\circ$).

Condition	Aileron Deflection	Airplane Load Factor	Aileron Load lbs.	$V\sqrt{\sigma}$ mph
$\frac{p\ B_w}{2\ V} = .08$	-24°	-1.0	-1357.6	200

c. Aileron Tabs

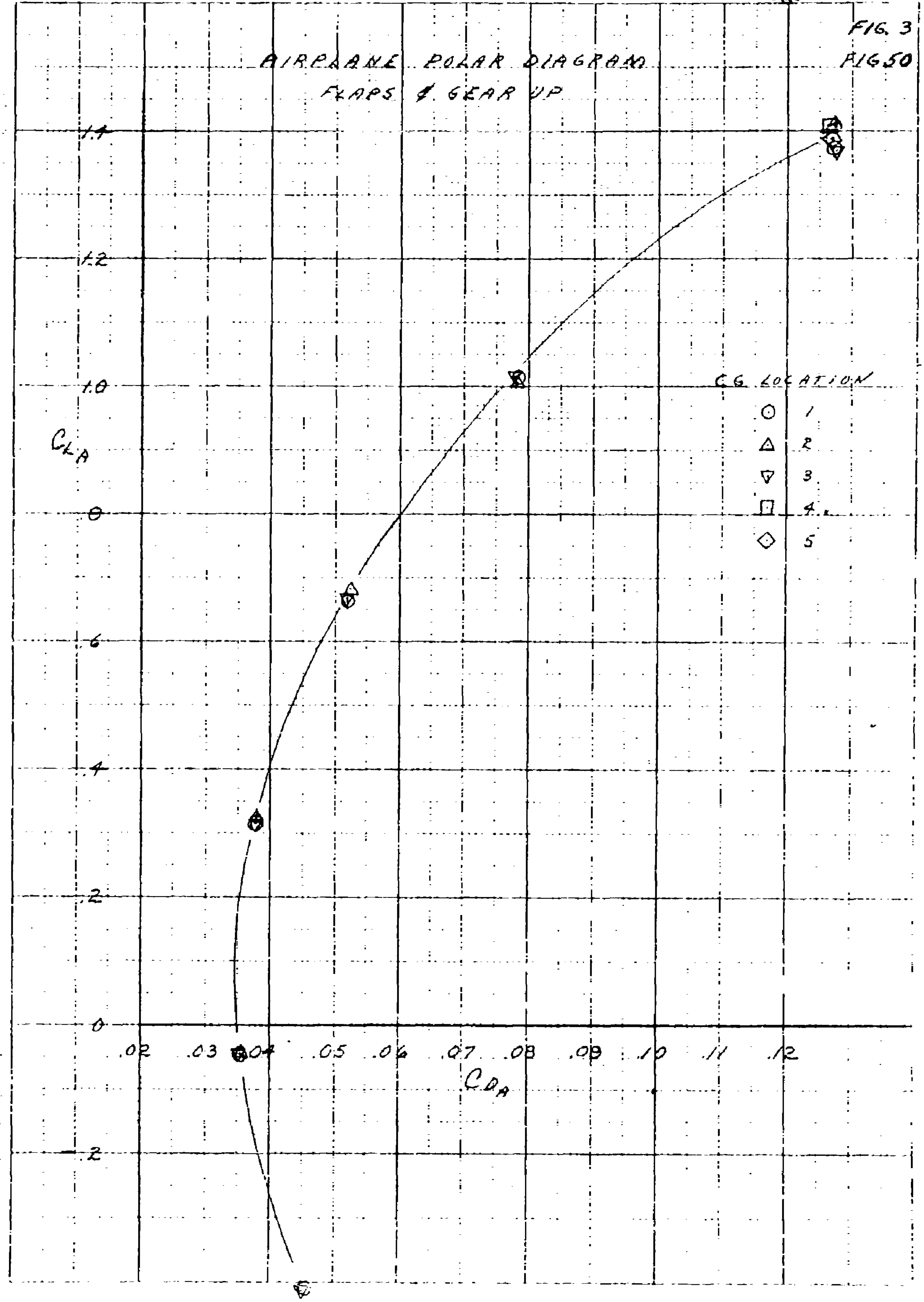
The maximum loads on the aileron trim and balance tabs as determined in Part IV-E of reference (5) did not exceed the minimum design requirements of 50 lbs/sq. ft. Therefore, the aileron tab loads are the minimum of 50 lbs/sq. ft.

PART I - D

D. AIRPLANE POLAR DIAGRAM

The airplane polar diagram for the complete airplane as determined by the addition of the lift and drag coefficient of the component parts is shown on Figure 3 for flaps and gear up.

FIG. 3
FIG 50

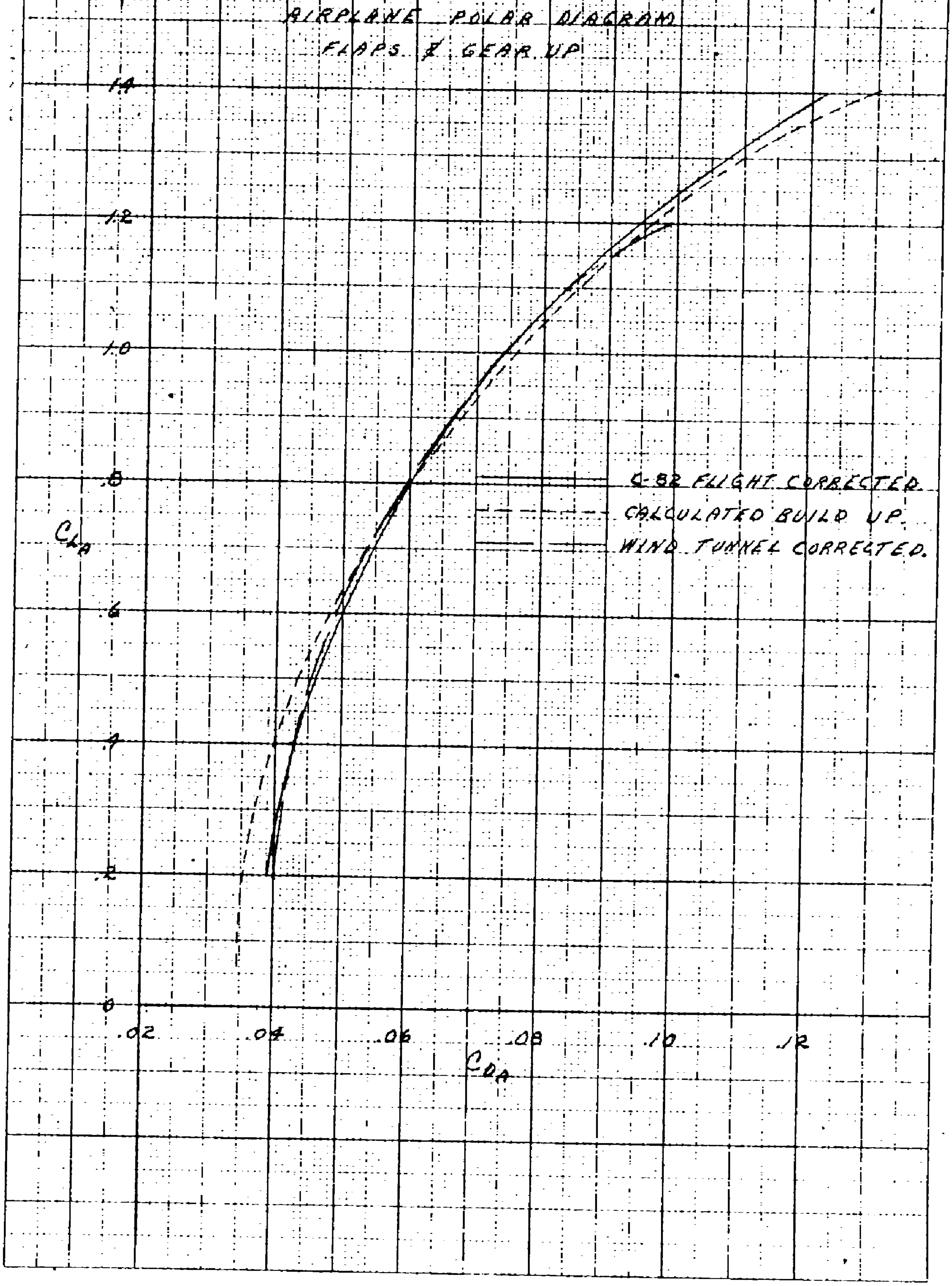

AIRPLANE POLAR DIAGRAM
FLAPS & GEAR UP
1.4
1.2
1.0
.8
.6
.4
.2
0
.2
C_{L_A}
.02
.03
.04
.05
.06
.07
.08
.09
.10
.11
.12
C_{D_A}
CG LOCATION
1
2
3
4
5

PART I-D (Cont.)

A comparison of the airplane polar diagram as determined by the addition of the lift and drag coefficients of the component parts with the airplane polar diagram obtained by correcting a C-82 flight test polar for the change in configuration between the C-82 and XC-120 airplane is shown on figure 4.

The polar diagram as determined from corrected available wind tunnel data is also shown on figure 4 for comparison.

FIG. 4


AIRPLANE POLAR DIAGRAM
FLAPS & GEAR UP
1.4
1.2
1.0
.8
.6
.4
.2
0
.02
.04
.06
.08
.10
.12
C_{L_A}
C_{D_A}
C-82 FLIGHT CORRECTED.
CALCULATED BUILD UP.
WIND TUNNEL CORRECTED.

PART I - E

E. LOADING CONDITIONS AND C.G. LOCATIONS

LOADING CONDITIONS

In accordance with reference (2) the loading conditions considered are presented in the table immediately following.

C. G. LOCATIONS

The basic center of gravity locations were selected as 20% and 30% of the mean aerodynamic chord of the wing for the horizontal locations. The vertical locations were selected as to envelope all possible c.g. locations with maximum fuel and no cargo being the prime determinant of the upper c.g. locations and minimum fuel and low cargo determining the lower c.g. locations. The envelope selected is presented in figure 5 and is used for all gross weights. Minimum flying weight conditions flaps up and flaps down fall inside the envelope and are considered as special points. The c.g. locations are designated as follows:

C. G. Location	Description
1	Most forward and highest
2	Most aft and highest
3	Most forward and lowest
4	Most aft and lowest
5	Minimum flying weight, flaps up
6	Minimum flying weight, flaps and gear down

C. G. locations 1 2 3 4 and 5 as shown on the diagram were considered as all flaps up conditions, while 1 2 3 4 and 6 were considered as flaps and gear down conditions.

RESTRICTED

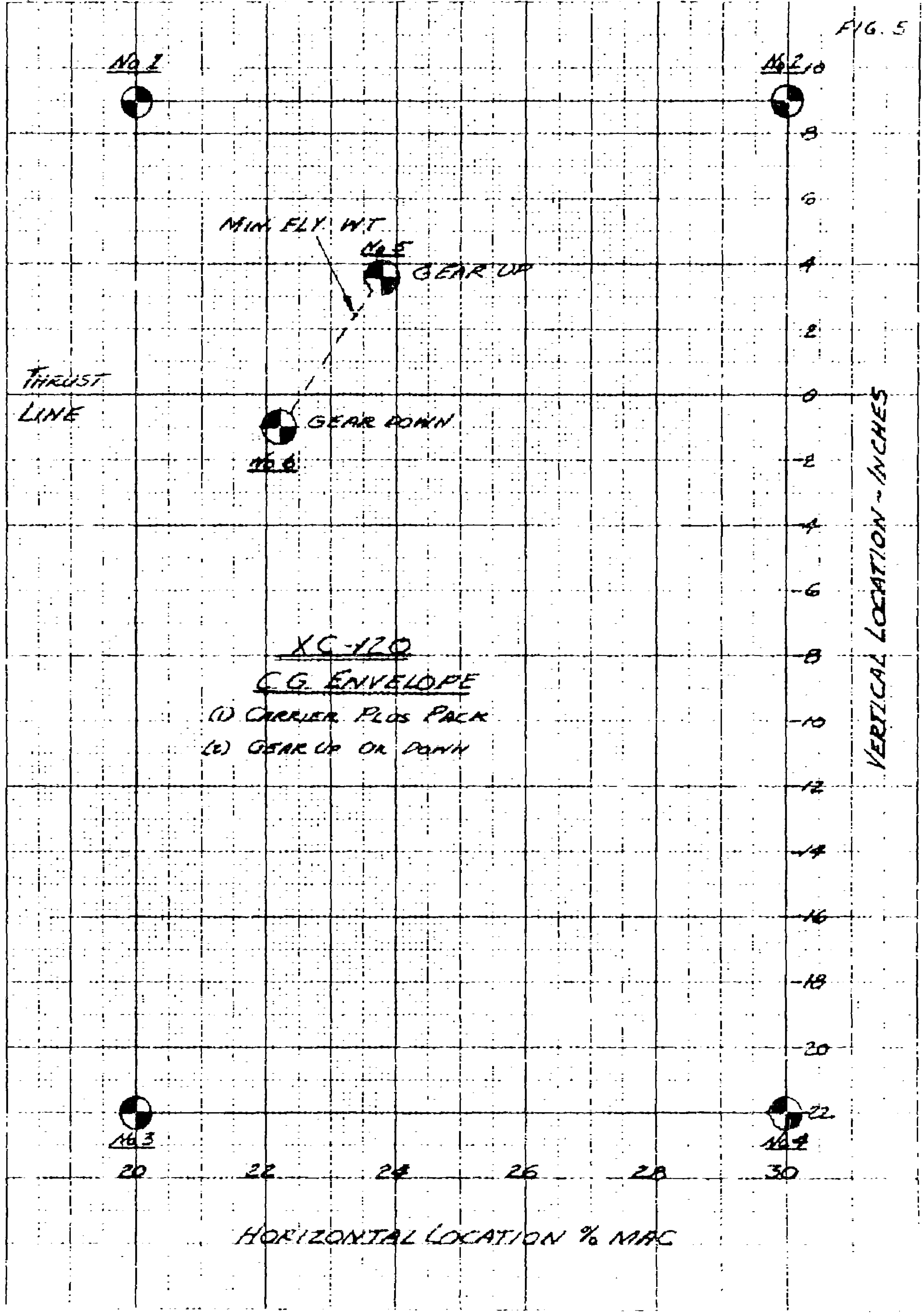
REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION		PAGES	PAGE 29
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY	
Subject: BASIC FLIGHT CRITERIA- PACK ON			DATE February 18,1949 REVISED	

PART I - E.

ANALYSIS OF LOADING CONDITIONS CONSIDERED

		Weight, Lbs.
1. Design weight (normal take-off weight) Specified by contractor in reference (6)		64,000
2. Maximum landing weight Specified by contractor in reference (6)		60,000
As specified by reference (2) this weight shall consist of not less than:		
weight empty	40465	
50% of internal fuel (2670 gal.)	8010 lbs.	
75% of maximum oil	675 lbs.	
trapped fuel and oil	245 lbs.	
crew	1000 lbs.	
cargo allowable	9605	
	60000 lbs.	
3. Light weight condition This condition selected for zero fuel load in the wing		48,000
design gross weight	64000	
fuel weight	16020	
	47980 lbs.	
4. Maximum alternate weight Specified by contractor in reference (6)		74,000
weight empty	40465	
trapped fuel and oil	245 lbs.	
miscellaneous equipment	27 lbs.	
crew	1000 lbs.	
full oil	900 lbs.	
cargo	20000	
fuel	11363	
Total	74000 lbs.	
5. Minimum flying weight As defined by reference (2)		42,136
weight empty	40465	
pilot and co-pilot	400 lbs.	
5% of fuel plus trapped fuel	901 lbs.	
25% of oil plus trapped oil	370 lbs.	
	42136	
6. Take-off Weight Considered as design weight for normal take-off or as maximum alternate weight		64,000 74,000

FIG. 5


No 1
No 2
MIN. FLY. WT.
No 5
GEAR UP
THRUST LINE
GEAR DOWN
No 6
XC-120
C.G. ENVELOPE
(1) CARRIER PLUS PACK
(2) GEAR UP OR DOWN
No 3
No 4
10
8
6
4
2
0
-2
-4
-6
-8
-10
-12
-14
-16
-18
-20
-22
20
22
24
26
28
30
VERTICAL LOCATION ~ INCHES
HORIZONTAL LOCATION % MAC

PART I-F-1

F. SPECIFICATION REQUIREMENTS FOR DESIGN CRITERIA

The following requirements for Design Criteria are obtained from reference (2).

1. SYMMETRICAL FLIGHT

a. Complete Airplane

Gross Weight	Load Factor	Flaps & Gear	Power (1)	C. G.	Gust Loads	Comments
Max. T.O.	+2, -1	Up	Off & On	envelope	All conditions	No
Max. Alt.	+2, -1	Up	Off & On	envelope	to be checked	pitching
Design					for gust load	acceleration
Gross	+3, -1.5	Up	Off & On	envelope	from minimum	required
Max.Land	+3, -1.5	Up	Off & On	envelope	flying speed	
Lt. Alt.	+3, -1.5	Up	Off & On	envelope	to high speed	
Minimum					level flight	
Flying	+3, -1.5	Up	Off & On	fixed	$U\sigma^{\frac{1}{2}} = 50K$ fps	
Max. T. O.	None	Down	Off & On	envelope	All conditions	No
Max Alt.	Specified	Down	Off & On	envelope	to be checked	pitching
Design	except				for gust load	acceleration
Gross	gust	Down	Off & On	envelope	from minimum	required
Max.Land		Down	Off & On	envelope	flying speed	
Lt. Alt		Down	Off & On	envelope	to 160 mph(3)	
Minimum					$U\sigma^{\frac{1}{2}} = 50K$ fps	
Flying		Down	Off & On	fixed		

Notes: (1) Power-on is defined as take-off power up to 2 V_S and military rated power above 2 V_S.

(2) V-n diagrams are required for all weight conditions at sea level for the clean condition. Flaps down do not require V-n diagrams since they are not intended as maneuvering aids. However, a V-n diagram is presented for gust conditions.

PART I-F-1 (Cont.)

Notes: (Cont.)

(3) Note that 1.75 V_S is specified for this condition but flap limit speed is 160 mph on present C-82 airplane. Flap limit speed shall remain the same for the XC-120 airplane.

It is noted that limit diving speed is the greatest speed considered in the stress analysis unless it is greater than terminal velocity. For maximum alternate gross weight, the true high speed in level flight shall be considered as true limit diving speed.

b. WING AND AILERONS (NEUTRAL)

1. Wings must be designed for all loadings under complete airplane in Section A.

2. Wing ribs and leading edge shall be investigated for pressure distributions corresponding to:

(a) positive lift coefficient giving most forward C. P.

(b) lift coefficient giving positive limit load factor at limit diving speed or at V_H if greater.

(c) distribution giving maximum negative load on forward 20%.

3. Wing leading edge designed for loads in 2(a) and 2(c) above modified for momentary increases in leading edge loading in accelerated flight (extend C_L versus α in straight line and other coefficients in a reasonable manner - use constant load factor and reduce speed).

c. HORIZONTAL TAIL AND ELEVATOR

1. Tail must be designed for all loadings under complete airplane in section A. - balancing loads

2. Maneuvering horizontal tail loads with pitching, C. G. at 20%, 30%, 35% MAC_w, one point on V-n diagram, (References (4) and (9), flaps and gear up.

PART I-F-1 (Cont.)

3. Add increment due to gust $U\sigma^{\frac{1}{2}} = 50$ K to balancing tail load at level-flight high speed (consider angular acceleration).

d. ELEVATOR TABS

1. Tabs designed for most severe combination of speed and tab normal force likely to be encountered. In no case is limit tab load less than 50 lbs/sq. ft. (use trapezoidal loading with L. E. twice trailing edge loading).

2. Note that the limit load over the fixed surface and the distribution of this load is assumed to be unaffected by tab deflection.

3. For balancing condition with speed is V_D and for maneuvering distribution speed is $\sqrt{n}\, V_S$ where n is positive limit load factor.

e. WING FLAPS

1. The limit loading is determined from the formula $W = 0.002558\, C_{L_{flap}} \times V^2 \times \sigma$

   where C_L is flap lift coefficient in fully deflected position with the gust condition and at the true speed corresponding to the most severe conditions given in Section A. (trapezoidal chord loading with L. E. twice trailing edge loading).

f. FUSELAGE, BOOMS, COWLING, DOORS

1. Determine pressure distributions corrected for compressibility @ V_D.

2. All loads for Section (A)

PART I-F-2

2. ASSYMETRICAL FLIGHT

a. COMPLETE AIRPLANE

	Load Factor
1. Design for $\frac{p\,B_W}{2\,V}$ = .08 at .8 V_H, flaps and gear up,	+2, -1
2. Design for $\frac{p\,B_W}{2\,V}$ = .015 at 1.25 V_H, flaps and gear up,	+2

b. WING AND AILERON (DEFLECTED)

1. Design for requirements under section (A) above (apply moment coefficients of deflected ailerons at the wing attitude corresponding to that point on V-n diagram which gives highest torsional moment on the wing).

2. Design for unbalanced loading using 100% of design load on one wing and 85% on the other considering angular inertia of the airplane.

3. In no case may average limit loads on ailerons be less than 20 lbs/sq. ft.

4. The design load shall be distributed over aileron chord as uniformly varying load 1/6 W at trailing edge to 11/6 W at L. E. where W = average load at any chord section unless wind tunnel data indicates a more rational distribution.

c. AILERON TABS

1. Trim tab design for worst condition encountered but not less than 50 lbs/sq. ft. trapezoidal load with L. E. load twice trailing edge load.

d. HORIZONTAL TAIL & ELEVATOR

1. Design for unsymmetrical loading where load on one-half of the tail is the maximum obtained for any condition and that on the other half is the maximum load multiplied by $(1 - n/7.33)$ where n is limit positive maneuvering load factor.

PART I-F-2

e. VERTICAL TAIL AND RUDDER

1. Design gust load $U\sigma^{\frac{1}{2}}$ = 50 fps normal to plane of tail at V_H with tail at zero angle of attack initially loading $W = 5 \times m \times V_H \sigma^{\frac{1}{2}}$. Where total limit load = average loading x total fin and rudder area.
2. Use triangular distribution of load with zero at trailing edge.

3. Design for zero yaw with one engine dead and the other delivering take-off power at speeds up to 2 V_S, pressure by reference (4) or flight test data.

4. Design for zero yaw with one engine dead and the other delivering normal rated power at V_{max} under this condition plus gust load added to vertical tail balancing load. Gust normal to surface effective velocity of 30 ft./sec. Gust increment may be obtained from

$$W = 3.0 \times m \times V\sigma^{\frac{1}{2}}$$

5. Design for five degree yaw with resulting rudder deflection and with zero rudder deflection at V_D.

f. RUDDER TABS

1. Design trim tab for most severe condition but not less than 50 lbs./sq. ft. trapezoidal loading with L. E. twice trailing edge load.

2. Design spring tabs for most severe condition to cause limit load on rudder. (Again never less than 50 lbs/sq. ft.).

3. Design control surface for distribution as unaffected by tab plus tab loading using V_D for balance conditions and $\sqrt{n}V_S$ for maneuvering distribution

(n = positive limit load factor)

PART I-G.

G. AIRPLANE PHYSICAL CHARACTERISTICS

This section covers the physical characteristics and geometric relationships of the component parts of the airplane with respect to each other and to the center of gravity points of the airplane.

These data are used in determining the symmetrical and asymmetrical aerodynamic characteristics of the component parts which are summed up to obtain the complete airplane aerodynamic characteristics.

The actual physical characteristics of each of the component parts are the same as shown in PART I-G of reference (1).

The tables immediately following present necessary dimensions for the component parts from the various C. G. Locations and dimensionless ratios of distances and areas used frequently in this analysis.

Subject: BASIC FLIGHT CRITERIA - PACK ON

PART I-G

DIMENSIONS FROM VARIOUS C.G. LOCATIONS

C. G. Locations Dimensions (Inches)	1 20%	2 30%	3 20%	4 30%	5 Gear Up — Minimum Flying Weight	6 Gear Down — Minimum Flying Weight
x_f	336.6	353.4	336.6	353.4	343.0	340.3
x_b	166.1	182.9	166.1	182.9	172.5	169.8
x_{ht}	-541.6	-524.7	-541.6	-524.7	-535.3	-537.9
x_w	33.66	50.48	33.66	50.48	40.01	37.32
x_{wi}	40.75	57.57	40.75	57.57	47.14	44.41
z_f	-27.02	-27.02	3.985	3.985	-21.54	-17.02
z_b	-9.00	-9.00	22.00	22.00	-3.52	+1.00
z_{vt}	48.65	48.65	79.65	79.65	54.13	58.65
z_{ht}	15.44	15.44	47.44	47.44	20.92	25.44
z_g	-87.00	-87.00	-56.00	-56.00	-81.52	-77.00
z_p	-9.00	-9.00	22.00	22.00	-3.52	+1.00
z_w	26.89	26.89	57.89	57.89	32.37	36.89
z_{wi}	13.87	13.87	44.87	44.87	19.35	23.87
l_g	31.5	14.7	31.5	14.7	25.1	27.8
l_{ht}	621.1	604.2	621.1	604.2	614.8	617.4
l_p	-174.5	-191.3	-174.5	-191.3	-180.9	-178.2
l_{vt}	621.1	604.2	621.1	604.2	614.8	617.4
y_b	175	175	175	175	175	175

PART I-G

Dimensionless Ratios

C. G. Locations	①	②	③	④	⑤	⑥
$\frac{x_w}{MAC_w}$	.20	.30	.20	.30	.238	.222
$\frac{z_w}{MAC_w}$	.1598	.1598	.3440	.3440	.1924	.2192
$\frac{A.C_w}{MAC_w}$	.256	.256	.256	.256	.256	.256
$\frac{x_{wi}}{MAC_w}$	.2422	.3421	.2422	.3421	.2801	.2639
$\frac{z_{wi}}{MAC_w}$	.0824	.0824	.2666	.2666	.1150	.1418
$\frac{AC_{wi}}{MAC_w}$	.2274	.2274	.2274	.2274	.2274	.2274
$\frac{P_f}{S_w}$	.4001	.4001	.4001	.4001	.4001	.4001
$\frac{L_f}{MAC_w}$	3.989	3.989	3.989	3.989	3.989	3.989
$\frac{x_f}{MAC_w}$	2.000	2.100	2.000	2.100	2.038	2.022
$\frac{.25\ L_f}{MAC_w}$	.9973	.9973	.9973	.9973	.9973	.9973
$\frac{F_f}{S_w}$	.1211	.1211	.1211	.1211	.1211	.1211
$\frac{z_f}{MAC_w}$	-.1605	-.1605	.0237	.0237	-.1280	-.1011
$\frac{P_b}{S_w}$	.2159	.2159	.2159	.2159	.2159	.2159
$\frac{L_b}{MAC_w}$	4.861	4.861	4.861	4.861	4.861	4.861
$\frac{x_b}{MAC_w}$	.987	1.087	.987	1.087	1.025	1.009

RESTRICTED

REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION		PAGES	PAGE 29
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY	
Subject:— BASIC FLIGHT CRITERIA – PACK ON			DATE 18 February 1949	REVISED

PART I-G

Dimensionless Ratios (Cont.)

C. G. Locations	①	②	③	④	⑤	⑥
$\frac{.25\ L_b}{MAC_w}$	1.215	1.215	1.215	1.215	1.215	1.215
$\frac{F_b}{S_w}$	.0282	.0282	.0282	.0282	.0282	.0282
$\frac{z_b}{MAC_w}$	-.0535	-.0535	.1307	.1307	-.0209	.0059
$\frac{z_{vt}}{MAC_w}$	.2891	.2891	.4733	.4733	.3217	.3485
$\frac{l_g}{MAC_w}$	.187	.0873	.187	.0873	.149	.165
$\frac{z_g}{MAC_w}$	-.5165	-.5165	-.3322	-.3322	-.4850	-.4575
$\frac{MAC_w}{l_{ht}}$	.2709	.2785	.2709	.2785	.2737	.2726
$\frac{x_f}{MAC_w} - \frac{.25L_f}{MAC_w}$	1.003	1.103	1.003	1.103	1.041	1.025
$\frac{x_b}{MAC_w} - \frac{.25L_b}{MAC_w}$	-.2285	-.1278	-.2285	-.1278	-.1900	-.2060
$\frac{x_w}{MAC_w} - \frac{AC_w}{MAC_w}$	-.056	.044	-.056	.044	-.018	-.034
$\frac{F_f}{S_w}\ \frac{L_f}{MAC_w}$	1.596	1.596	1.596	1.596	1.596	1.596
$\frac{F_f}{S_w}\left(\frac{x_f}{MAC_w} - \frac{.25L_f}{MAC_w}\right)$	.4013	.4413	.4013	.4413	.4165	.4101
$\frac{F_f}{S_w}\ \frac{z_f}{MAC_w}$	-.0194	-.0194	.0029	.0029	-.0155	-.0122
$\frac{F_b}{S_w}\ \frac{L_b}{MAC_w}$	1.0495	1.0495	1.0495	1.0495	1.0495	1.0495
$\frac{F_b}{S_w}\left(\frac{x_b}{MAC_w} - \frac{.25L_b}{MAC_w}\right)$	-.0492	-.0276	-.0492	-.0276	-.0410	-.0445

PART I-G

Dimensionless Ratios (Cont.)

C.G. Locations	①	②	③	④	⑤	⑥
$\frac{F_b}{S_w} \frac{z_b}{MAC_w}$	-.0015	-.0015	.0037	.0037	-.00056	.00017

PART I - H.

REFERENCES

(1) Fairchild Engineering Report R107-012, "Basic Aerodynamic Data - Model XC-120", dated 9 July 1948.

(2) Army Air Forces Specification No. C-1803-E including Specification R-1803-2B and amendment No. 1, R-1803-3A and amendment No. 3, R-1803-5B and amendment No. 1 and R-1803-6B, dated 2 April 1948.

(3) Fairchild Engineering Report R107-017, "Analysis of Wind Tunnel Data", dated 31 March 1949.

(4) Army-Navy-Civil Committee on Aircraft Design Criteria Report ANC-1(2), "Chordwise Airload Distribution", dated 28 October 1942.

(5) Fairchild Engineering Report R110-008, "Basic Flight Criteria - Model C-119B", dated 27 July 1948.

(6) Fairchild Engineering Report R107-000, "Model Specification for XC-120 Airplane", dated 21 October 1947.

(7) Fairchild Engineering Report R107-015, "Air Load Distribution - Model XC-120"

(8) Fairchild Engineering R107-013, "Maneuvering Horizontal Tail Loads - Model XC-120", Volume I, dated 4 March 1949.

(9) Army Air Forces Technical Report 5185, "Maneuvering Horizontal Tail Loads". Air Service Technical Command, dated 25 January 1945.

(10) Fairchild Engineering Report R107-016, "Performance Calculations - Model XC-120", dated 9 March 1949.

(11) NACA Technical Report 633, "Pressure Distribution over an NACA 23012 Airfoil with a Slotted and a Plain Flap", by C. J. Wenzinger and J. B. Delano, dated 1938.

(12) Fairchild Engineering Report R110-007, "Basic Aerodynamic Data - Model C-119B" - Appendix I, dated 21 February 1949.

RESTRICTED

PART II - Symmetrical Flight - Pitch

A. GENERAL

It is the purpose of this section to combine the previously calculated aerodynamic characteristics of the various component parts of the airplane in order to determine the characteristics of the complete airplane in pitch. In order to clarify the procedure for ease of separate checks, the method of analysis and method of combination is presented in detail.

1. SUMMARY OF AERODYNAMIC COEFFICIENTS OF COMPONENT PARTS

The analysis of component parts such as wing, fuselage, booms, horizontal tail, vertical tail, landing gear, and propeller have been previously presented in reference (1). However, each of the component parts were considered separately and therefore have their own particular aerodynamic coefficients. A summary of these coefficients is presented below for convenience in collecting and using data on the separate parts.

COMPONENT PART	LIFT COEFFICIENT	DRAG COEFFICIENT	PITCHING MOMENT COEFFICIENT
Wing	$C_{L_W} = L_W/qS_W$	$C_{D_W} = D_W/qS_W$	$C_{M_{W_{A.C.}}} = M_W/qS_W MAC_W$
Fuselage	$C_{L_F} = L_F/qP_f$	$C_{D_F} = D_F/qF_f$	$C_{M_{F_{.25L_f}}} = M_F/qP_fL_f$
Boom (one)	$C_{L_B} = L_B/qP_b$	$C_{D_B} = D_B/qF_b$	$C_{M_{B_{.25Lb.}}} = M_B/qP_bL_b$
(1) Tail	$C_{L_T} = L_T/q_tS_t$	$C_{D_T} = D_T/q_tS_t$	$C_{M_{T_{A.C.}}} = M_T/q_tS_tc_t$

Notes: (1) Horizontal and vertical tails are designated by the subscript H, h, V, or v, respectively, such as $C_{L_{HT}}$ or S_{vt}.

Part II - A-1

It is noted from reference (1) that all of the lifts and drags on the component parts have been resolved parallel and perpendicular to the airplane thrust line and are presented in form C_X and C_Z, respectively. The coefficients of C_X and C_Z are based on the same areas, lengths, etc. as the coefficients of C_D and C_L presented in the above table.

(2) ADDITION OF COMPONENT PARTS TO DETERMINE AIRPLANE CHARACTERISTICS PACK ON - SYMMETRICAL FLIGHT ONLY

(a) Summation of Z Force

$$Z_A = Z_W + Z_F + 2Z_B + Z_{HT} + 2Z_G + 2Z_P + 2\Delta Z_{WI}$$

Note: The factor of 2 is introduced by the twin boom arrangement; all characteristics previously calculated are for one unit. Note that the contribution of vertical tail is assumed negligible.

Now basing the total airplane coefficient on wing area, chord, and free flight velocity.

$$\frac{Z_A}{qS_W} = \frac{Z_W}{qS_W} + \frac{Z_F}{qS_W} + \frac{2Z_B}{qS_W} + \frac{Z_{HT}}{qS_W} + \frac{2Z_G}{qS_W} + \frac{2Z_P}{qS_W} + \frac{2\Delta Z_{WI}}{qS_W}$$

$$C_{Z_A} = C_{Z_W} + C_{Z_F}\frac{F_f}{S_W} + 2C_{Z_B}\frac{P_b}{S_W} + C_{Z_{HT}}\frac{q_{ht}S_{ht}}{q\,S_W} + 2C_{Z_G} + \frac{2Z_P}{q\,S_W} + 2\Delta C_{Z_{WI}}$$

(b) Summation of X Force

The same general procedure is followed here as for summation of Z Forces except the vertical tail contributes to the forces.

$$X_A = X_W + X_F + 2X_B + X_{HT} + 2X_{VT} + 2X_G + 2X_P + 2\Delta X_{WI}$$

$$\frac{X_A}{qS_W} = \frac{X_W}{qS_W} + \frac{X_F}{qS_W} + \frac{2X_B}{qS_W} + \frac{X_{HT}}{qS_W} + \frac{2X_{VT}}{qS_W} + \frac{2X_G}{qS_W} + \frac{2X_P}{qS_W} + \frac{2\Delta X_{WI}}{qS_W}$$

PART II - A-2 (Cont.)

(b). Summation of X Force (Cont.)

$$C_{X_A} = C_{X_W} + C_{X_F} \frac{F_f}{S_w} + 2C_{X_B} \frac{F_b}{S_w} + C_{X_{HT}} \frac{q_{ht}S_{ht}}{qS_w} + 2C_{X_{VT}} \frac{q_{vt}S_{vt}}{qS_w} + 2\,C_{X_G} + \frac{2\,X_P}{q\,S_w} + 2\Delta C_{X_{WI}}$$

(c). Summation of Moments About Any C. G. Location

Figure 6 presents a general picture of the airplane in symmetrical flight with the pertinent forces and arms noted. Taking moments about the C.G.

$$M_W = -l_wZ_W + z_wX_W$$

$$M_F = -l_fZ_F + z_fX_f$$

$$M_B = -(l_bZ_B + z_b X_B)\,2$$

$$M_{HT} = -l_{ht}Z_{HT} + z_{ht} X_{ht}$$

$$M_{VT} = (z_{vt}X_{VT})\,2$$

$$M_G = (-l_gZ_G + z_gX_G)\,2$$

$$M_S = (-l_{wi}\,\Delta Z_{WI} + z_{wi}\,\Delta X_{WI})\,2$$

$$M_P = (-l_pZ_P + z_pX_P)\,2$$

Now consider the contribution of each component part to the moment about the c.g. and reference the final moment coefficients to wing area, wing chord, and free flight velocity.

Wing

Consider two possible c.g. locations so that -

$$M_{W_1\,c.g.} = -l_{w_1} Z_W + z_{w_1} X_W$$

$$M_{W_2\,c.g.} = -l_{w_2} Z_W + z_{w_2} X_W$$

$$C_{M_{W_1}\,c.g.} = -C_{Z_W} \frac{l_{w_1}}{MAC_w} + C_{X_W} \frac{z_{w_1}}{MAC_w}$$

PART II - A-3

(c) Summation of Moments about any C.G. Location (Cont.)

Now from Figure 6 we see that $l_w = C.P_w - x_w$ where C.P. and x are both positive measured aft of the leading edge of airfoil -

so:

$$C_{M_{W_1}}{}_{c.g.} = -C_{Z_W}\left(\frac{C.P_w}{MAC_w} - \frac{x_{w_1}}{MAC_w}\right) + C_{X_W}\frac{z_{w_1}}{MAC_w}$$

$$C_{M_{W_2}}{}_{c.g.} = -C_{Z_W}\left(\frac{C.P_w}{MAC_w} - \frac{x_{w_2}}{MAC_w}\right) + C_{X_W}\frac{z_{w_2}}{MAC_w}$$

$$C_{M_{W_2}}{}_{c.g.} - C_{M_{W_1}}{}_{c.g.} = -C_{Z_W}\left(\frac{C.P._w}{MAC_w} - \frac{x_{w_2}}{MAC_w}\right) + C_{Z_W}\left(\frac{C.P._w}{MAC_w} - \frac{x_{w_1}}{MAC_w}\right)$$

$$+ C_{X_W}\frac{z_{w_2}}{MAC_w} - C_{X_W}\frac{z_{w_1}}{MAC_w}$$

$$C_{M_{W_2}}{}_{c.g.} = C_{M_{W_1}}{}_{c.g.} + C_{Z_W}\left(\frac{x_{w_2}}{MAC_w} - \frac{x_{w_1}}{MAC_w}\right) + C_{X_W}\left(\frac{z_{w_2}}{MAC_w} - \frac{z_{w_1}}{MAC_w}\right)$$

Now assuming that the first C.G. position lies at the A.C. of the wing and neglecting the small vertical displacement between the point of action of the forces and the A.C.

Then: $C_{M_{W_1}}{}_{c.g.} = C_{M_{W_{A.C.}}}$ and

$z_{w_1} = 0$

x_{w_1} = location of $A.C._w$ from leading edge of MAC_w

So: $$C_{M_W}{}_{c.g.} = C_{M_{W_{A.C.}}} + C_{Z_W}\left(\frac{x_w}{MAC_w} - \frac{A.C._w}{MAC_w}\right) + C_{X_W} \times \left(\frac{z_w}{MAC_w}\right)$$

Fuselage

Consider two possible c.g. locations so that -

$$M_{F_1}{}_{c.g.} = -l_{f_1} Z_F + z_{f_1} X_F$$

$$M_{F_2}{}_{c.g.} = -l_{f_2} Z_F + z_{f_2} X_F$$

PART II - A - 2 (Cont.)

Fuselage

transferring to coefficient form based on wing area, etc.

$$C_{M_{F_1}\,c.g.} = -C_{Z_F}\frac{P_f}{S_w}\frac{l_{f_1}}{MAC_w} + C_{X_F}\frac{F_f}{S_w}\frac{z_{f_1}}{MAC_w}$$

$$C_{M_{F_2}\,c.g.} = -C_{Z_F}\frac{P_f}{S_w}\frac{l_{f_2}}{MAC_w} + C_{X_F}\frac{F_f}{S_w}\frac{z_{f_2}}{MAC_w}$$

$$C_{M_{F_2}\,c.g.} - C_{M_{F_1}\,c.g.} = C_{Z_F}\frac{P_f}{S_w}\left(-\frac{l_{f_2}}{MAC_w} + \frac{l_{f_1}}{MAC_w}\right) + C_{X_F}\frac{F_f}{S_w}\left(\frac{z_{f_2}}{MAC_w} - \frac{z_{f_1}}{MAC_w}\right)$$

$$C_{M_{F_2}\,c.g.} = C_{M_{F_1}\,c.g.} + C_{Z_F}\frac{P_f}{S_w}\left(\frac{l_{f_1}}{MAC_w} - \frac{l_{f_2}}{MAC_w}\right) + C_{X_F}\frac{F_f}{S_w}\left(\frac{z_{f_2}}{MAC_w} - \frac{z_{f_1}}{MAC_w}\right)$$

now from Figure 6 we see that $l_f = C.P._f - x_f$

$$C_{M_{F_2}\,c.g.} = C_{M_{F_1}\,c.g.} + C_{Z_F}\frac{P_f}{S_w}\left(\frac{C.P._f}{MAC_w} - \frac{x_{f_1}}{MAC_w} - \frac{C.P._f}{MAC_w} + \frac{x_{f_2}}{MAC_w}\right) +$$

$$C_{X_F}\frac{F_f}{S_w}\left(\frac{z_{f_2}}{MAC_w} - \frac{z_{f_1}}{MAC_w}\right)$$

now assume that the first C.G. Location is at the $.25L_f$ on the fuselage reference line - then:

$$C_{M_{F_1}\,c.g.}' = C_{M_{F}\,.25L_f}\frac{P_f L_f}{S_w MAC_w}$$

$$z_{f_1} = 0$$

$$x_{f_1} = .25L_f$$

$$C_{M_F\,c.g.} = C_{M_F\,.25L_f}\frac{P_f L_f}{S_w MAC_w} + C_{Z_F}\frac{P_f}{S_w}\left(\frac{x_f}{MAC_w} - \frac{.25L_f}{MAC_w}\right) + C_{X_F}\frac{F_f}{S_w}\left(\frac{z_f}{MAC_w}\right)$$

PART II - A - 2 (Cont.)

Boom

The development of the transfer formulas for the boom is the same as for the fuselage:

$$C_{M_{B\,c.g.}} = 2\left[C_{M_{B\,.25L_b.}} \frac{P_b L_b}{S_w MAC_w} + C_{Z_B} \frac{F_b}{S_w}\left(\frac{x_b}{MAC_w} - \frac{.25L_b}{MAC_w}\right) + C_{X_B} \frac{F_b}{S_w}\left(\frac{z_b}{MAC_w}\right)\right]$$

PART II - A-2 (Cont.)

Horizontal Tail

Now considering the horizontal tail the situation is somewhat different because of the effects of change in dynamic pressure and downwash at the horizontal tail.

$$M_{HT_1 \, c.g.} = - l_{ht_1} Z_{HT} + z_{ht_1} X_{HT}$$

$$M_{HT_2 \, c.g.} = - l_{ht_2} Z_{HT} + z_{ht_2} X_{HT}$$

Now transferring to coefficient form based on wing area, etc.

$$C_{M_{HT_1} \, c.g.} = -C_{Z_{HT}} \frac{l_{ht_1}}{MAC_w} \frac{q_{ht}}{q} \frac{S_{ht}}{S_w} + C_{X_{HT}} \frac{z_{ht_1}}{MAC_w} \frac{q_{ht}}{q} \frac{S_{ht}}{S_w}$$

$$C_{M_{HT_2} \, c.g.} = - C_{Z_{HT}} \frac{l_{ht_2}}{MAC_w} \frac{q_{ht}}{q} \frac{S_{ht}}{S_w} + C_{X_{HT}} \frac{z_{ht_2}}{MAC_w} \frac{q_{ht}}{q} \frac{S_{ht}}{S_w}$$

now combining

$$C_{M_{HT_2} \, c.g.} = C_{M_{HT_1} \, c.g.} + C_{Z_{HT}} \frac{q_{ht} S_{ht}}{q \, S_w} \left(\frac{l_{ht_1}}{MAC_w} - \frac{l_{ht_2}}{MAC_w} \right) + C_{X_{HT}} \frac{q_{ht}}{q} \frac{S_{ht}}{S_w} \left(\frac{z_{ht_2}}{MAC_w} - \frac{z_{ht_1}}{MAC_w} \right)$$

now we know that $l_{ht} = C.P._{ht} - x_{ht}$

$$\text{so: } \frac{l_{ht_1}}{MAC_w} - \frac{l_{ht_2}}{MAC_w} = \frac{C.P._{ht}}{MAC_w} - \frac{x_{ht_1}}{MAC_w} - \frac{C.P._{ht}}{MAC_w} + \frac{x_{ht_2}}{MAC_w} = \left(\frac{x_{ht_2}}{MAC_w} - \frac{x_{ht_1}}{MAC_w} \right)$$

Now assume that the first C.G. position lies at the A.C. of the horizontal tail

$$\text{then: } C_{M_{HT_1} \, c.g.} = C_{M_{HT_{A.C.}}} \frac{q_{ht}}{q} \frac{S_{ht}}{S_w} \frac{c_{ht_c}}{MAC_w}$$

$$z_{ht_1} = 0$$

$$x_{ht_1} = A.C._{ht}$$

$$\text{so: } C_{M_{HT} \, c.g.} = C_{M_{HT_{A.C.}}} \frac{q_{ht}}{q} \frac{S_{ht}}{S_w} \frac{c_{ht_c}}{MAC_w} + C_{Z_{HT}} \frac{q_{ht}}{q} \frac{S_{ht}}{S_w} \left(\frac{x_{ht}}{MAC_w} - \frac{AC_{ht}}{MAC_w} \right) + C_{X_{HT}} \frac{q_{ht}}{q} \frac{S_{ht}}{S_w} \left(\frac{z_{ht}}{MAC_w} \right)$$

PART II - A-2 (Cont.)

Vertical Tail

The only force contributed by the vertical tail is assumed to be an X force which is assumed to act at the centroid of vertical tail area.

$$M_{VT_{c.g.}} = 2\,(z_{vt}\,X_{VT})$$

converting to coefficient form based on wing area, etc.

$$C_{M_{VT_{c.g.}}} = 2\left(C_{X_{VT}}\,\frac{q_{vt}}{q}\,\frac{S_{vt}}{S_w}\,\frac{z_{vt}}{MAC_w}\right)$$

Landing Gear Extended

Gear extended is a special condition. When gear is down the only force acting on the gear is assumed to be a drag force which is applied at the drag centroid of the landing gear and is constant regardless of angle of attack. The drag figures for the gear previously presented in reference (1) are for one gear, so the factor of two has to be considered in the drag forces.

so: $$M_{G_{c.g.}} = 2\,(-l_g\,Z_G + z_g\,X_G)$$

converting to coefficient form based on wing area, etc. It is noted that the previously presented forces on the gear were C_D , C_Z , and C_X based on the true wing area.

$$C_{M_{G_{c.g.}}} = 2\,C_{X_G}\,\frac{z_g}{MAC_w} - 2C_{Z_G}\,\frac{l_g}{MAC_w}$$

PART II - A - 2 (Cont.)

Slipstream Effects

The application of power will have two significant effects:

(a) The incremental changes in wing lift, drag, and moment due to the slipstream effects, and
(b) The thrust and normal force acting at the propeller itself.

This section considers item (a) only - item (b) will be discussed later.

$$M_S = 2\left(-l_{wi}\;\Delta Z_{WI} + z_{wi}\;\Delta X_{WI}\right)$$

Consider two possible C. G. locations so that

$$\underset{c.g.}{M_{S_1}} = 2(-l_{wi_1}\;\Delta Z_{WI} + z_{wi}\;\Delta X_{WI})$$

$$\underset{c.g.}{M_{S_2}} = 2(-l_{wi_2}\;\Delta Z_{WI} + z_{wi_2}\;\Delta X_{WI})$$

$$\underset{c.g.}{C_{M_{S_1}}} = 2\left(-\Delta C_{Z_{WI}}\;\frac{l_{wi_1}}{MAC_w} + \Delta C_{X_{WI}}\;\frac{z_{wi_1}}{MAC_w}\right)$$

$$\underset{c.g.}{C_{M_{S_2}}} = 2\left(-\Delta C_{Z_{WI}}\;\frac{l_{wi_2}}{MAC_w} + \Delta C_{X_{WI}}\;\frac{z_{wi_2}}{MAC_w}\right)$$

$$l_{wi} = C.P._{wi} - x_{wi}$$

$$\underset{c.g.}{C_{M_{S_1}}} = 2\left(-\Delta C_{Z_{WI}}\left(\frac{CP_{wi}}{MAC_w} - \frac{x_{wi_1}}{MAC_w}\right) + \Delta C_{X_{WI}}\;\frac{z_{wi_1}}{MAC_w}\right)$$

$$\underset{c.g.}{C_{M_{S_2}}} = 2\left(-\Delta C_{Z_{WI}}\left(\frac{CP_{wi}}{MAC_w} - \frac{x_{wi_2}}{MAC_w}\right) + \Delta C_{X_{WI}}\;\frac{z_{wi_2}}{MAC_w}\right)$$

$$\underset{c.g.}{C_{M_{S_2}}} - \underset{c.g.}{C_{M_{S_1}}} = 2\left[\Delta C_{Z_{WI}}\left(\frac{x_{wi_2}}{MAC_w} - \frac{x_{wi_1}}{MAC_w}\right) + \Delta C_{X_{WI}}\left(\frac{z_{wi_2}}{MAC_w} - \frac{z_{wi_1}}{MAC_w}\right)\right]$$

$$\underset{c.g.}{C_{M_{S_2}}} = \underset{c.g.}{C_{M_{S_1}}} + 2\left[\Delta C_{Z_{WI}}\left(\frac{x_{wi_2}}{MAC_w} - \frac{x_{wi_1}}{MAC_w}\right) + \Delta C_{X_{WI}}\left(\frac{z_{wi_2}}{MAC_w} - \frac{z_{wi_1}}{MAC_w}\right)\right]$$

PART II - A - 2 (Cont.)

Slipstream Effects (Cont.)

now assume the first c. g. position lies at A. C. of portion of wing immersed in slipstream.

$$C_{M_{S_1}\,c.g.} = 2\,\Delta C_{M_{WI}}$$

$$z_{wi_1} = 0$$

x_{wi_1} = location of A. C. of immersed portion from leading edge.

$$C_{M_S\,c.g.} = 2\,\Delta C_{M_{WI}} + 2\left[\Delta C_{Z_{WI}}\left(\frac{x_{wi}}{MAC_w} - \frac{A.C._{wi}}{MAC_w}\right) + \Delta C_{X_{WI}}\,\frac{z_{wi}}{MAC_w}\right]$$

PART II - A - 2 (Cont.)

Propeller Effects

In considering power effects it is noted that the thrust force usually considered positive when directed forward is herein considered negative in line with the sign convention adopted for aft forces being positive. All propeller forces are assumed to act at the center of the propeller hub:

$$M_{p_{c.g.}} = 2 \left(- l_p Z_p + z_p X_p \right)$$

converting to coefficient form based on wing area

$$C_{M_{p_{c.g.}}} = \frac{2}{q S_w MAC_w} \left(- l_p Z_p + z_p X_p \right)$$

3 Angular Relationships

In order to properly sum the component parts of the airplane it is necessary to know the relationship of each part to the thrust line since all of the characteristics terms are parallel and perpendicular to the thrust line. These relationships are explained in reference (1) but are summarized here for convenience and in Fig.

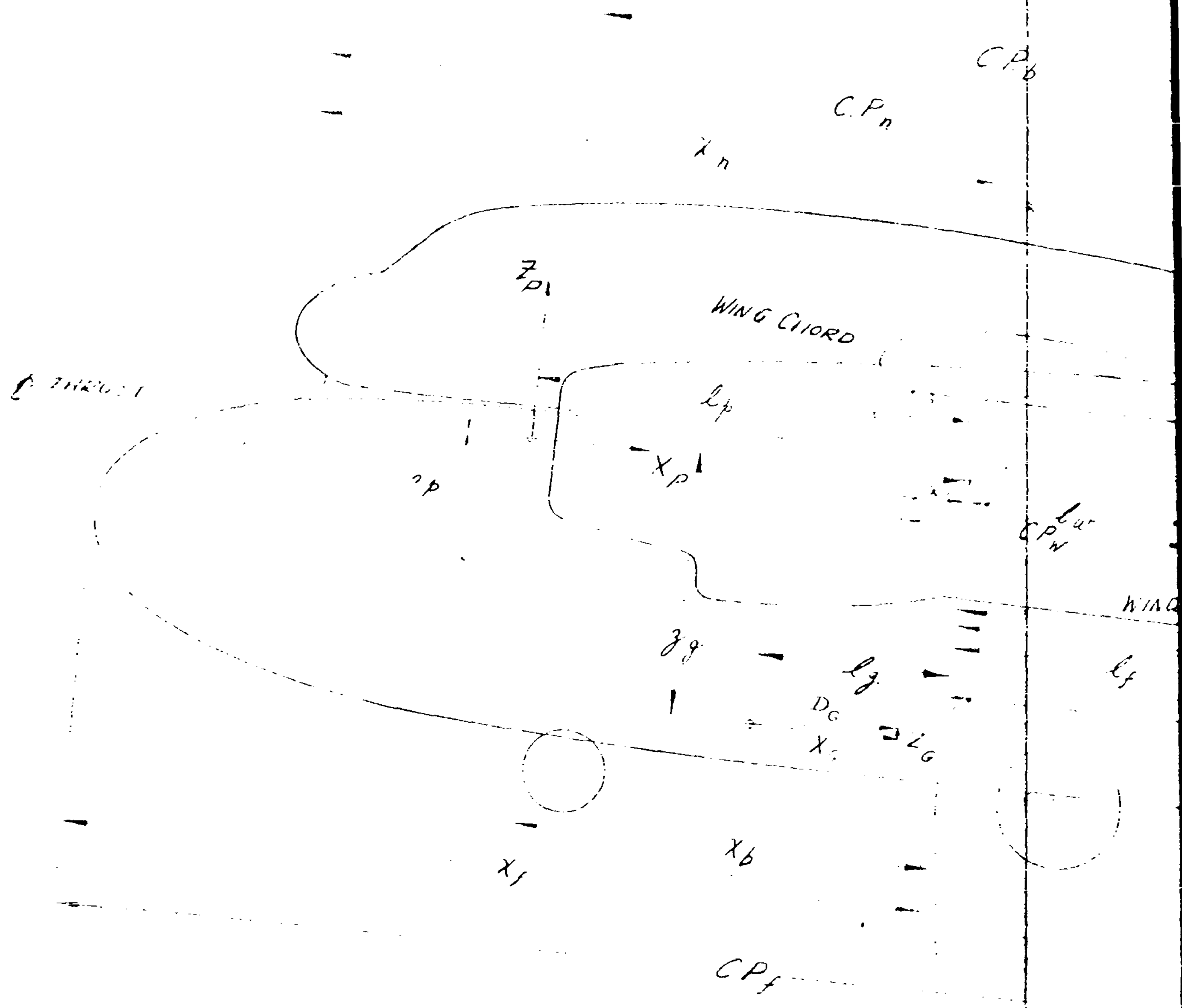
$$\alpha_c = \alpha_{T.L.} + 7^\circ$$

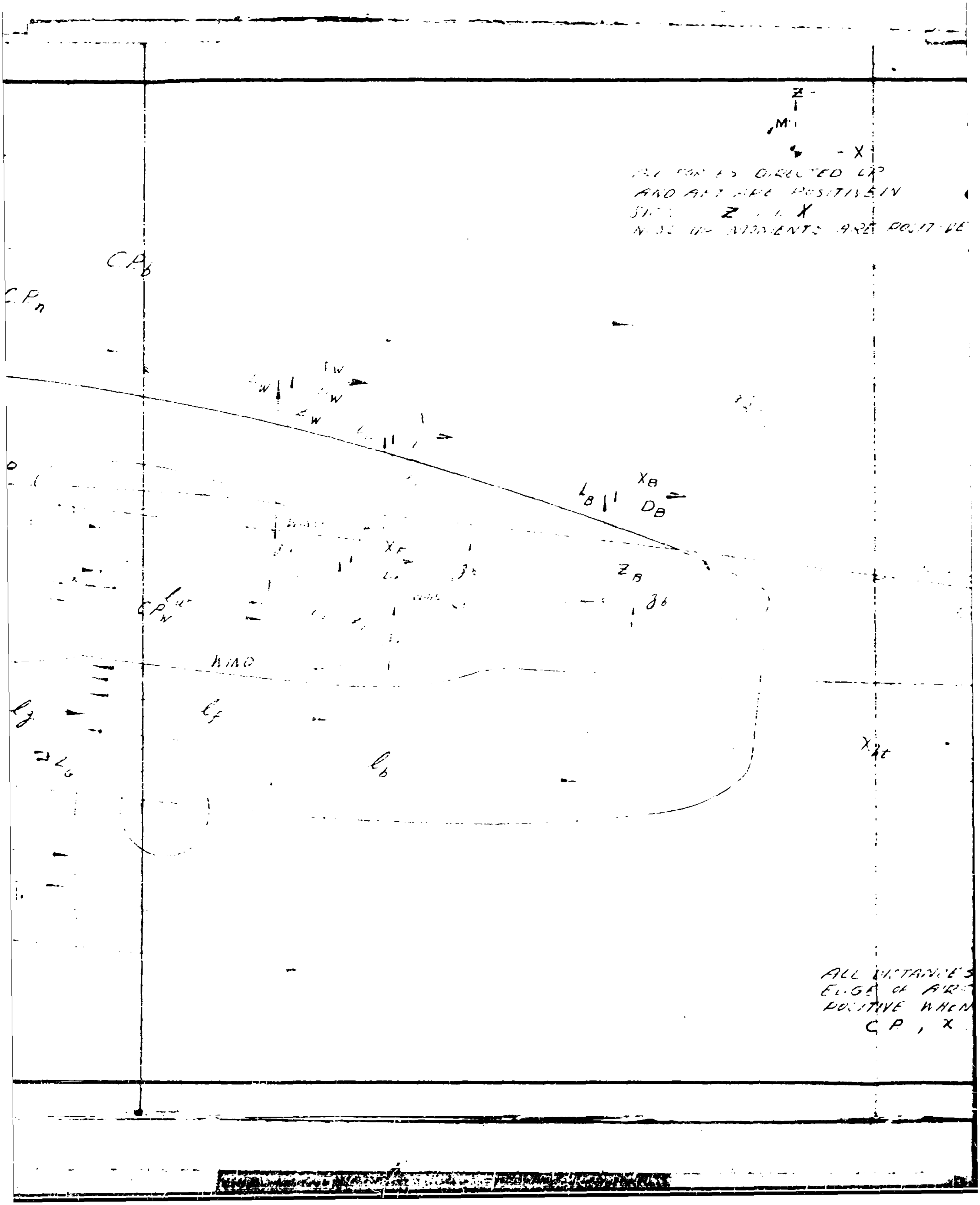
$$\alpha_f = \alpha_{T.L.}$$

$$\alpha_{ht} = \alpha_{T.L.} + 1.0^\circ - \epsilon_{ave} \text{ (explained in Fig. 7)}$$

No angle relationship is necessary for the gear and the vertical tail.

Report No. R107-014
Model XC-120





Z
X
CP_b
CP_n
X_B
L_B
D_B
Z_B
CP_W
C P , X

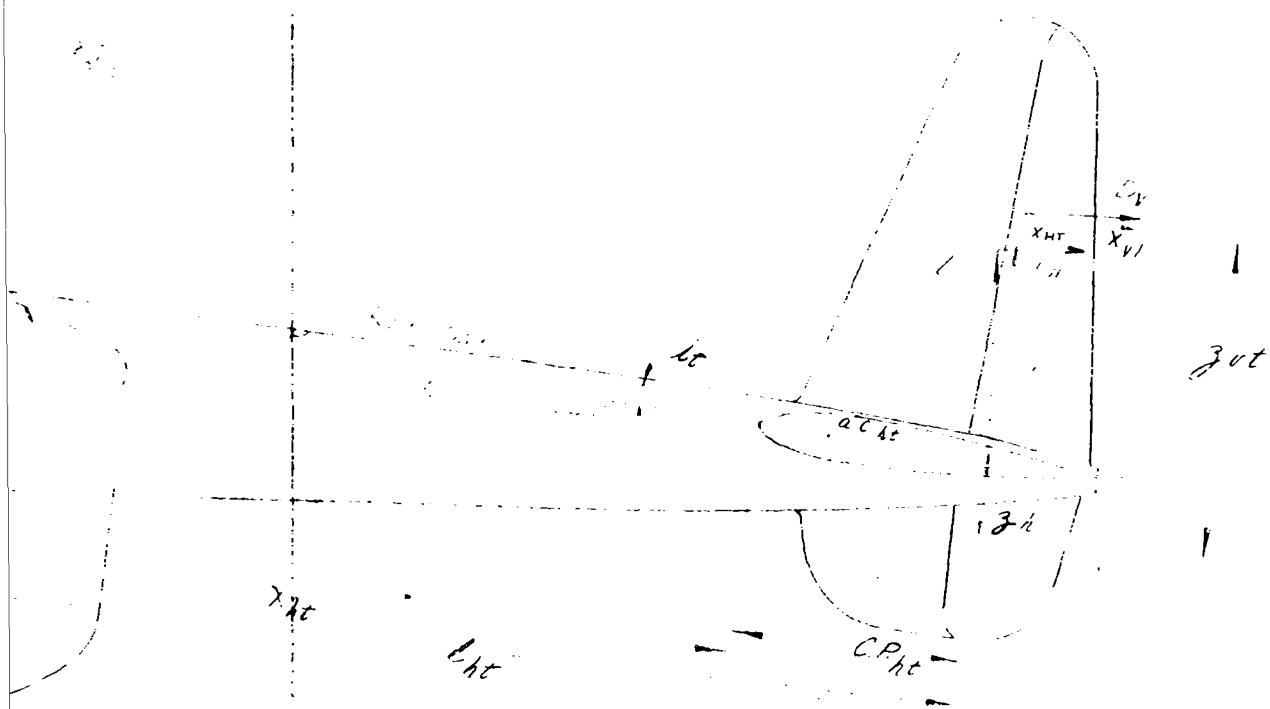
FIG. 1
FIG. 6

Z+
M+
-X

z^+
CG — l_+

[illegible] AS DIRECTED UP [illegible] ARE POSITIVE IN Z [illegible] X
UP MOMENTS ARE POSITIVE

ALL DISTANCES MEASURED [illegible] THE REST OF THE C.G. ARE POSITIVE IN SIGN ~ z AND l



ALL DISTANCES MEASURED FROM LEADING EDGE OF AIRFOIL OR BODIES ARE CONSIDERED POSITIVE WHEN AFT OF LEADING EDGE —
CP, X AND ac.

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REPORT NO. R107-014 | FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION | PAGES | PAGE 44

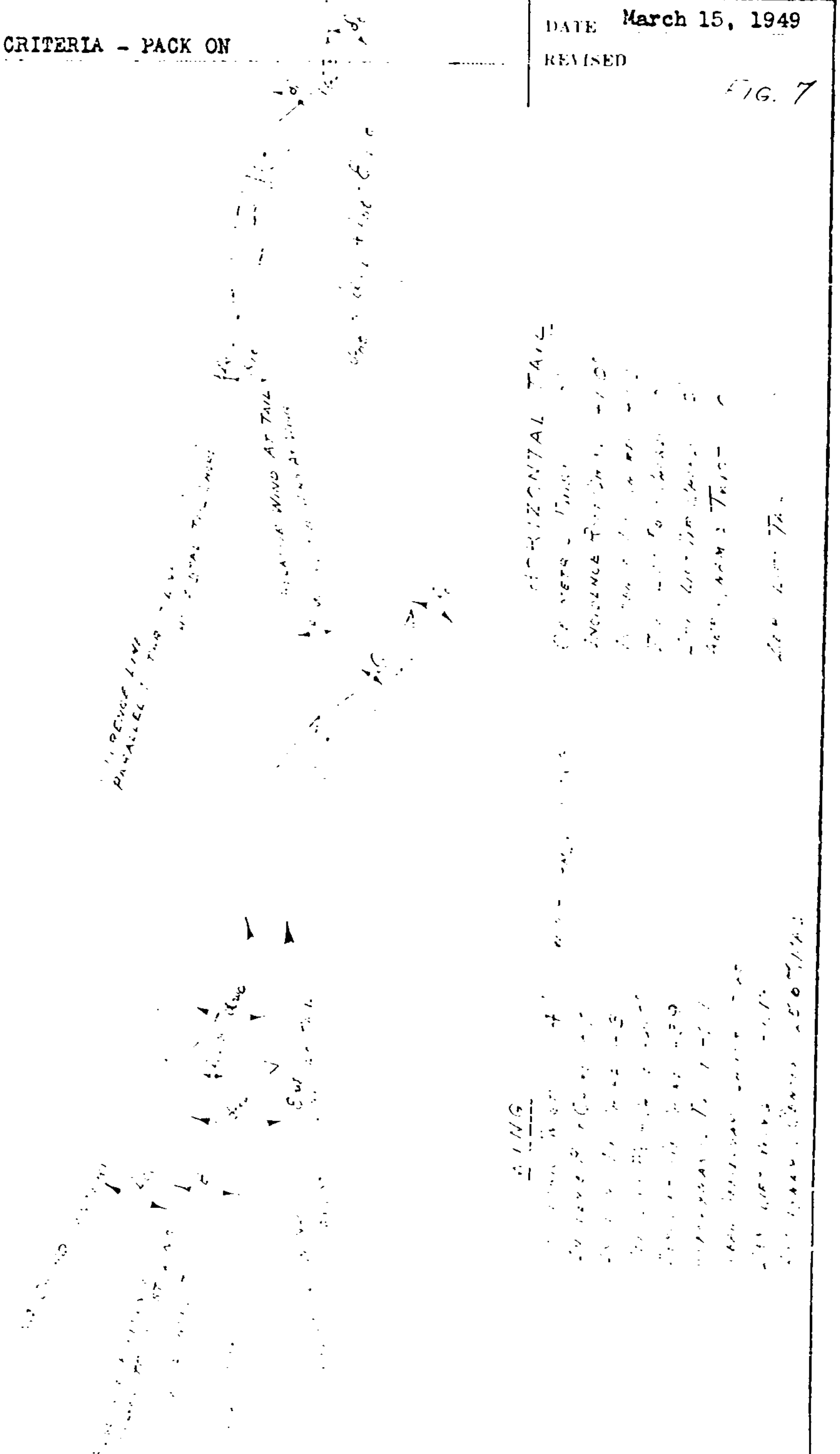
MODEL XC-120 | PREPARED BY | CHECKED BY | APPROVED BY

DATE March 15, 1949

REVISED

Subject:— BASIC FLIGHT CRITERIA - PACK ON

FIG. 7



PART II - A-4

4 Summary of Equations for Complete Airplane

A summary of all of the equations for the complete airplane in symmetrical flight is presented in the following table.

TABLE

SUMMARY OF EQUATIONS FOR COMPLETE AIRPLANE
IN PITCH - PACK ON

Total Z Force

$$C_{Z_A} = C_{Z_W} + C_{Z_F}\frac{P_f}{S_w} + 2C_{Z_B}\frac{P_b}{S_w} + C_{Z_{HT}}\frac{q_{ht}\,S_{ht}}{q\,S_w} + 2C_{Z_G} + 2\frac{Z_P}{q\,S_w}$$

$$+ 2\frac{\Delta Z_{WI}}{q\,S_w}$$

Total X Force

$$C_{X_A} = C_{X_W} + C_{X_F}\frac{F_f}{S_w} + 2\,C_{X_B}\frac{F_b}{S_w} + C_{X_{HT}}\frac{q_{ht}S_{ht}}{q\,S_w} + 2C_{X_{VT}}\frac{q_{vt}\,S_{vt}}{q\,S_w}$$

$$+ 2C_{X_G} + \frac{2\,X_P}{q\,S_w} + \frac{2\,\Delta X_{WI}}{q\,S_w}$$

Total Moment	Contribution Of:
$C_{M_{A\,c.g.}} = C_{M_{W_{A.C.}}} + C_{Z_W}\left(\frac{x_w}{MAC_w} - \frac{A.C._w}{MAC_w}\right) + C_{X_W}\frac{z_w}{MAC_w} +$	Wing
$C_{M_{F.25L_f}}\frac{P_f\,L_f}{S_w\,MAC_w} + C_{Z_F}\frac{P_f}{S_w}\left(\frac{x_f}{MAC_w} - \frac{.25L_f}{MAC_w}\right) C_{X_F}\frac{F_f}{S_w}\frac{z_f}{MAC_w}$	Fuselage
$+ 2\left[C_{M_{B.25L_b}}\frac{P_b\,L_b}{S_w\,MAC_w} + C_{Z_B}\frac{P_b}{S_w}\left(\frac{x_b}{MAC_w} - \frac{.25L_b}{MAC_w}\right) + C_{X_B}\frac{F_b}{S_w}\frac{z_b}{MAC_w}\right.$	Booms
$+ C_{M_{HTO\,a.c.}}\frac{q_{ht}\,S_{ht}\,c_{ht}}{q\,S_w\,MAC_w} + C_{Z_{HT}}\frac{q_{ht}S_{ht}}{q\,S_w}\left(\frac{x_{ht}}{MAC_w} - \frac{A.C._{ht}}{MAC_w}\right) + C_{X_{HT}}\frac{q_{ht}\,S_{ht}}{q\,S_w}\frac{z_{ht}}{MAC_w} +$	Horizontal Tail
$2\,C_{X_{VT}}\frac{q_{vt}\,S_{vt}}{q\,S_w}\frac{z_{vt}}{MAC_w}$	Vertical Tails

TABLE

SUMMARY OF EQUATIONS FOR COMPLETE AIRPLANE - PACK ON

(Cont.)

Total Moment	Contribution of
$+ 2 C_{X_G} \frac{z_g}{MAC_w} - 2 C_{Z_G} \frac{l_g}{MAC_w}$	Gear
$+ \frac{2}{q S_w MAC_w} \left(- l_p Z_p + z_p X_p \right) +$	Propeller
$2 \Delta C_{M_{WI}} + 2 \left[\Delta C_{Z_{WI}} \left(\frac{x_{wi}}{MAC_w} - \frac{A.C._i}{MAC_w} \right) + \Delta C_{X_{WI}} \frac{z_{wi}}{MAC_w} \right]$	Slipstream

PART II - B

B. Pitching Moments of Component Parts

This section covers the determination of the contributions of each of the component parts of the airplane to the total pitching moment about the various locations of center of gravity.

The contribution of each part is found by the equations developed in Part II-A.

The aerodynamic characteristics of each component part were obtained from reference (1).

The final pitching moment coefficients of the component parts are based on wing area, wing mean aerodynamic chord and free stream dynamic pressure.

PART II - B - 1

1. Wing Unflapped - Power Off

$$C_{M_W} = C_{M_{W_{A.C.}}} + C_{Z_W}\ (A) + C_{X_W}\ (B)$$

where

$$A = \left(\frac{x_W}{MAC_W} - \frac{A.C._W}{MAC_W} \right)$$

$$B = \left(\frac{z_W}{MAC_W} \right)$$

$C_{M_{W_{A.C.}}}$, C_{Z_W} & C_{X_W} are from Figure 4 of Reference (1)

C. G. Location

1.	$A_1 = -.056$	$B_1 = .1598$
2.	$A_2 = .044$	$B_2 = .1598$
3.	$A_3 = -.056$	$B_3 = .3440$
4.	$A_4 = .044$	$B_4 = .3440$
5.	$A_5 = -.018$	$B_5 = .1924$

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REPORT NO. R107-014 FAIRCHILD AIRCRAFT DIVISION
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PAGES | PAGE 50

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

Subject: BASIC FLIGHT CRITERIA - PACK ON

DATE March 15, 1949

REVISED

PART II - B - 1

PART II - B - 1

WING UNFLAPPED - POWER OFF

1	2	3	4	5	6	7	8	9	10	11
					C.G. 1			C.G. 2		
α_{TL}	$C_{M_{W2C}}$	C_{Z_W}	C_{X_W}	$C_{Z_W}\cdot A_1$	$C_{X_W}\cdot B_1$	C_{M_W}	$C_{Z_W}\cdot A_2$	$C_{X_W}\cdot B_2$	C_{M_W}	$C_{Z_W}\cdot A_3$
-12	-.0630	-.32550	-.05165	.01823	-.00708	-.05385	-.01432	-.00708	-.08640	.01823
-8	-.0630	.00396	.00193	-.00022	.00111	-.06211	.00018	.00111	-.06171	-.00022
-4	-.0630	.34620	.03645	-.01950	.00582	-.07668	.01532	.00582	-.04186	-.01950
0	-.0630	.68510	.02703	-.03836	.00431	-.09705	.03014	.00431	-.02855	-.03836
4	-.0630	1.02270	-.01675	-.05727	-.00300	-.12327	.04500	-.00300	-.02100	-.05727
8	-.0630	1.36160	-.09150	-.07626	-.01462	-.15348	.05992	-.01462	-.01770	-.07626

12	13	14	15	16	17	18	19	20	21	22
C.G. 3			C.G. 4			C.G. 5				
$C_{X_W}\cdot B_3$	C_{M_W}	$C_{Z_W}\cdot A_4$	$C_{X_W}\cdot B_4$	C_{M_W}	$C_{Z_W}\cdot A_5$	$C_{X_W}\cdot B_5$	C_{M_W}			
-.[illegible]156	-.06433	-.01432	-.01956	-.09688	.00566	-.01094	-.06808			
.00238	-.06084	.00018	.00238	-.06049	-.00172	.00133	-.06174			
.01254	-.06776	.01532	.01254	-.03514	-.00627	.00701	-.06226			
.00721	-.09207	.03014	.00929	-.02357	-.01233	.00517	-.07014			
-.00645	-.12670	.04500	-.00645	-.02445	-.01841	-.00361	-.08502			
-.03148	-.17074	.05992	-.03148	-.03456	-.02451	-.01760	-.10511			

PART II - B - 2

2. Wing - Flapped - Power Off $\delta_f = 40°$

$$C_{M_{WF}} = C_{M_{WF_{AC}}} + C_{Z_{WF}}(A) + C_{X_{WF}}(B)$$

where

$$A = \left(\frac{x_W}{MAC_W} - \frac{A.C._W}{MAC_W}\right)$$

$$B = \left(\frac{z_W}{MAC_W}\right)$$

$C_{M_{WF_{AC}}}$, $C_{Z_{WF}}$, $C_{X_{WF}}$ are from Figure 8 of Reference (1).

C.G. Location

1. $A_1 = -.056$ $B_1 = .1598$
2. $A_2 = .044$ $B_2 = .1598$
3. $A_3 = -.056$ $B_3 = .3440$
4. $A_4 = .044$ $B_4 = .3440$
6. $A_6 = -.034$ $B_6 = .2192$

PART II - B - 2

WING FLAPPED - POWER OFF $\delta_f = 40°$

				C G 1			C G 2			
α	$C_{[illegible]}$	$C_{Z_{WF}}$	$C_{X_{WF}}$	$C_{Z_{WF}} \cdot A_1$	$C_{X_{WF}} \cdot B_1$	$C_{M_{WF}}$	$C_{Z_{WF}} \cdot A_2$	$C_{X_{WF}} \cdot B_2$	$C_{M_{WF}}$	$C_{Z_{WF}} \cdot A_3$
[illegible]	[illegible]	[illegible]	[illegible]	-.0013	.0062	-.1436	.0002	.0062	-.1433	-.0003
[illegible]	[illegible]	[illegible]	.0632	-.0191	.0143	-.1545	.0155	.0143	-.1204	-.0171
-2	[illegible]	[illegible]	[illegible]	-.0351	.0166	-.1712	.0247	.0166	-.1072	-.0341
0	[illegible]	[illegible]	.0069	-.0573	.0129	-.1941	.0450	.0127	-.0916	-.0573
2	[illegible]	[illegible]	.0116	-.0764	.0122	-.2229	.0600	.0032	-.0865	-.0762
6	[illegible]	[illegible]	-.0665	-.0754	-.0109	-.2560	.0750	-.0109	-.0865	-.0754

C G 3		C G 4			C G 6		
$C_{X_{WF}} \cdot B_3$	$C_{M_{WF}}$	$C_{Z_{WF}} \cdot A_4$	$C_{X_{WF}} \cdot B_4$	$C_{M_{WF}}$	$C_{Z_{WF}} \cdot A_6$	$C_{X_{WF}} \cdot B_6$	$C_{M_{WF}}$
.0122	[illegible]	.0012	.0292	-.1362	-.0002	.0085	-.1414
.0157	-.1551	.0155	.0317	[illegible]	-.0116	.0136	-.1417
.0297	-.1621	.0211	.0357	-.0541	-.0231	.0226	-.1502
[illegible]	-.1792	.0453	.0276	-.0769	-.0348	.0177	-.1668
.0166	-.2173	.0600	.0066	-.0629	-.0464	.0043	-.1116
-.0232	-.2357	.0750	.0236	-.0983	-.0579	-.0150	-.2226

PART II - B-3

3. FUSELAGE

$$C_{M_f} = C_{M_{f.25L_f}} \text{ (A)} + C_{Z_f} \text{ (B)} + C_{X_f} \text{ (C)}$$

where

$$A = \frac{P_f L_f}{S_w \, MAC_w}$$

$$B = \left[\frac{P_f}{S_w} \left(\frac{X_f}{MAC_w} - \frac{.25L_f}{MAC_w} \right) \right]$$

$$C = \frac{F_f}{S_w} \; \frac{z_f}{MAC_w}$$

$C_{M_{f.25L_f}}$ from figure 19 of reference (1)

C_{Z_f} & C_{Z_X} from figure 18 of reference (1)

C. G. Location

1	$A_1 = 1.596$	$B_1 = .4013$	$C_1 = -.0194$
2	$A_2 = 1.596$	$B_2 = .4413$	$C_2 = -.0194$
3	$A_3 = 1.596$	$B_3 = .4013$	$C_3 = .0029$
4	$A_4 = 1.596$	$B_4 = .4413$	$C_4 = .0029$
5	$A_5 = 1.596$	$B_5 = .4165$	$C_5 = -.0155$
6	$A_6 = 1.596$	$B_6 = .4101$	$C_6 = -.0122$

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PART II - B - 3

FUSELAGE

1	2	3	4	5 (C.G. 1)	6 (C.G. 1)	7 (C.G. 1)	8 (C.G. 1)	9 (C.G. 2)	10 (C.G. 2)	11 (C.G. 2)
α_{FL}	$C_{M_{F\,25\%}}$	C_{Z_F}	C_{X_F}	$C_{M_{F\,25\%}} \cdot A_1$	$C_{Z_F} \cdot B_1$	$C_{X_F} \cdot C_1$	C_{M_F}	$C_{M_{F\,25\%}} \cdot A_2$	$C_{Z_F} \cdot B_2$	$C_{X_F} \cdot C_2$
-12	-.1126	-.0461	.0766	-.03128	-.02616	-.00147	-.07035	-.03128	-.02447	-.00147
-8	-.1148	-.0519	.0827	-.02362	-.02063	-.00150	-.04025	-.02362	-.02295	-.00180
-4	-.0076	-.0189	.0776	-.01245	-.00756	-.00173	-.02176	-.01245	-.01534	-.00173
0	0	0	.0791	0	0	-.00172	-.00172	0	0	-.00172
4	.0078	.0164	.0796	.01245	.00758	-.00173	[illegible]	.01245	.01534	-.00173
8	.0148	.0519	.0927	.02362	.02063	-.00150	.04265	.02362	.02290	-.00165

12	13 (C.G. 3)	14 (C.G. 3)	15 (C.G. 3)	16 (C.G. 3)	17 (C.G. 4)	18 (C.G. 4)	19 (C.G. 4)	20 (C.G. 4)	21 (C.G. 5)	22 (C.G. 5)
C_{M_F}	$C_{M_{F\,25\%}} \cdot A_3$	$C_{Z_F} \cdot B_3$	$C_{X_F} \cdot C_3$	C_{M_F}	$C_{M_{F\,25\%}} \cdot A_4$	$C_{Z_F} \cdot B_4$	$C_{X_F} \cdot C_4$	C_{M_F}	$C_{M_{F\,25\%}} \cdot A_5$	$C_{Z_F} \cdot B_5$
-.07474	.03128	-.03816	.00022	-.06922	-.03128	-.04197	.00022	-.07303	-.03128	-.03161
-.04832	.02362	-.02063	.00027	-.04418	-.02362	-.02290	[illegible]	-.04620	-.02362	-.02162
-.02272	-.01245	-.00758	.00029	-.01974	-.01245	-.01534	.00029	-.02650	-.01245	-.00797
-.00192	0	0	.00024	.00027	0	0	.00029	.00021	0	0
.01866	.01245	.00758	.00024	.02032	.01245	.01534	.00027	.01216	.01245	.00267
.04472	.02362	.02063	.00027	.04472	.02362	.02290	.00027	.04577	.02362	.02162

REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION		PAGES PAGE 55
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY
Subject:— BASIC FLIGHT CRITERIA - PACK ON			DATE March 15, 1949 REVISED

PART II - B - 3

FUSELAGE

23	24	25	26	27	28
C.G. 5		C.G. 6			
C_{X_F} C_5	C_{M_F}	$C_{M_{F_{c/4}}}$ A_6	C_{Z_F} B_6	C_{X_F} C_6	C_{M_F}
-.00042	-.07131	-.03126	-.03900	-.00093	-.07121
-.00051	-.04575	-.02362	-.02126	-.00113	-.04603
-.00055	-.02087	-.01245	-.00775	-.00122	-.02142
-.00055	-.00055	0	0	-.00121	-.00121
-.00055	.01977	.01245	.00775	-.00122	.01898
-.00051	.04473	.02362	.02126	-.00113	.04377

PART II - B - 4

4. Booms

$$C_{M_B} = 2\left[C_{M_{B.25L_b}}(A) + C_{Z_B}(B) + C_{X_B}(C)\right]$$

where:

$$A = \frac{P_b\,L_b}{S_W MAC_W}$$

$$B = \frac{P_b}{S_W}\left(\frac{X_b}{MAC_W} - \frac{.25L_b}{MAC_W}\right)$$

$$C = \left(\frac{F_b}{S_W}\quad\frac{Z_b}{MAC_W}\right)$$

$C_{M_{B.25L_b}}$ from Figure 29, Reference 1.

C_{Z_B} and C_{X_B} from Figure 28, Reference 1.

C. G. Locations

1	$A_1 = 1.0495$	$B_1 = -.0492$	$C_1 = -.0015$
2	$A_2 = 1.0495$	$B_2 = -.0276$	$C_2 = -.0015$
3	$A_3 = 1.0495$	$B_3 = -.0492$	$C_3 = .0037$
4	$A_4 = 1.0495$	$B_4 = -.0276$	$C_4 = .0037$
5	$A_5 = 1.0495$	$B_5 = -.0410$	$C_5 = -.00056$
6	$A_6 = 1.0495$	$B_6 = -.0445$	$C_6 = .00017$

REPORT NO. R107-014 — FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION — RESTRICTED — PAGE 57

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

DATE March 15, 1949

REVISED

Subject:— BASIC FLIGHT CRITERIA - PACK ON

PART II - B - 4

BOOMS (2)

1	2	3	4	5	6	7	8	9	10	11
				CG 1				CG 2		
α	$C_{m_{B\,.256}}$	C_{Z_B}	C_{X_B}	$C_{m_{B\,.256}} \times A_1$	$C_{Z_B} \cdot B_1$	$C_{X_B} C_1$	C_{m_B}	$C_{m_{B\,.256}} \times A_2$	$C_{Z_B} B_2$	$C_{X_B} C_2$
-12	-[illegible]65	-.0347	.1790	-.00682	.00171	-.000268	-.0106	-.00682	.000957	-.000268
-8	-[illegible]25	-.0165	.1890	-.00262	.000812	-.000284	-.00418	-.00262	.000455	-.000284
-4	.0030	-.0042	.1897	.00315	.000207	-.000284	.00615	.00315	.000116	-.000284
0	.0115	0	.1890	.01210	0	-.000284	.02363	.01210	0	-.000284
4	.0255	.0042	.1897	.02470	-.000207	-.000284	.04842	.02470	-.000116	-.000284
[illegible]	.0370	.0165	.1890	.03880	-.000812	-.000284	.07541	.03880	-.000455	-.000284

12	13	14	15	16	17	18	19	20	21	22
	CG 3			CG 4					CG 5	
C_{m_B}	$C_{m_{B\,.256}} \times A_3$	$C_{Z_B} \cdot B_3$	$C_{X_B} \cdot C_3$	C_{m_B}	$C_{m_{B\,.256}} \times A_4$	$C_{Z_B} B_4$	$C_{X_B} C_4$	C_{m_B}	$C_{m_{B\,.256}} \times A_5$	$C_{Z_B} B_5$
-.012260	-.00682	.00171	.00066	-.00890	-.00682	.000957	.00066	-.01041	-.00682	.00142
-.004898	-.00262	.000812	.00070	-.00222	-.00262	.000455	.00070	-.00293	-.00262	.000676
.005764	.00315	.000207	.00070	.00812	.00315	.000116	.00070	.007932	.00315	.000172
.02363	.012100	0	.00070	.0256	.01210	0	.00070	.02560	.01210	0
.04860	.02470	-.000207	.00070	.05038	.02470	-.000116	.00070	.05057	.02470	-.000172
.07612	.03880	-.000812	.00070	.07738	.03880	-.000455	.00070	.07809	.03880	-.000676

BASIC FLIGHT CRITERIA - PACK ON

PART II - B - 4

BOOMS (2)

23	24	25	26	27	28
C6 LOC 5			C6	6	
$C_{x_B}\,C_5$	C_{M_B}	$C_{M_B .254} \times A_6$	$C_{Z_B}\,B_L$	$C_{x_B}\,C_L$	C_{M_B}
-.000100	-.01100	-.00682	.00154	.0000304	-.01050
-.000106	-.00410	-.00262	.000734	.0000322	-.00371
-.000106	.006432	.00315	.000187	.0000322	.00674
-.000106	.02400	.01210	0	.0000322	.02426
-.000106	.04884	.02470	-.000187	.0000322	.0491
-.000106	.07604	.03880	-.000734	.0000322	.07620

PART II - B-5

5. VERTICAL TAILS

$$C_{N_{VT}} = 2 \times C_{X_{VT}} \times \frac{q_{vt}}{q} \times \frac{S_{vt}}{S_W} \times \frac{z_{vt}}{MAC_W}$$

It is assumed that $C_{D_{VT}}$ does not vary with angle of attack so that $C_{X_{VT}} = C_{D_{VT}}$ and $C_{L_{VT}} = C_{Z_{VT}} = 0$

$$C_{X_{VT}} = C_{D_{VT}} = .0081 \quad \text{For } \alpha_{vt} = 0^\circ, \quad \delta_r = 0^\circ$$

from page 234, reference (1)

RESTRICTED

REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION		PAGES	PAGE 60
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY	
Subject:-- BASIC FLIGHT CRITERIA - PACK ON			DATE March 15, 1949	REVISED

PART II - B - 5

VERTICAL TAILS (2)

1	2	3	4	5	6	7	8
CC	D_{VT}	$2C_{XVT}$	$\bar{y}_{VC}/b$	S_{VT}/S_W	$\bar{z}_{VT}/MAC_W$	$2C'_{YVT}(S_{VT}/S_W)$	C_{MVT}
1	[illegible]	.01644	1.0	.0690	.2891	.00114	.00033
2					.2891		.00033
3					.4133		.00054
4					.4133		.00054
5					.3217		.00037
6					.3485		.00040

RESTRICTED

REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION		PAGES	PAGE 61
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY	
Subject:— BASIC FLIGHT CRITERIA - PACK ON			DATE March 15, 1949 REVISED	

PART II - B-6

6. LANDING GEAR

$$C_{M_G} = 2\,C_{X_G}\,\frac{z_g}{MAC_w} - 2C_{Z_G}\,\frac{l_g}{MAC_w}$$

$$A = \frac{z_g}{MAC_w}$$

$$B = \frac{l_g}{MAC_w}$$

C_{X_G} & C_{Z_G} from figure 47 reference (1)

C. G. Location

1	$A_1 = -.5165$	$B_1 = .187$
2	$A_2 = -.5165$	$B_2 = .0873$
3	$A_3 = -.3322$	$B_3 = .187$
4	$A_4 = -.3322$	$B_4 = .0873$
6	$A_6 = -.4575$	$B_6 = .165$

FA-80C-23

PART II - B - 6

LANDING GEAR

1	2	3	4	5	6	7	8	9	10	11
					CG Loc 1			CG Loc 2		
$\alpha_{T.L.}$	$C_{X_{LG}}$	$C_{Z_{LG}}$	$2C_{X_{LG}}$	$2C_{Z_{LG}}$	④ · A_1	⑤ · B_1	C_{MG} ⑥ - ⑦	④ · A_2	⑤ · B_2	C_{MG} ⑨ - ⑩
-12	.0321	.00681	.0642	.0136	-.0332	.0025	-.0357	-.0332	.0012	-.0344
-8	.0325	.00456	.0650	.0091	-.0336	.0017	-.0353	-.0336	.0008	-.0344
-4	.0327	.00229	.0654	.0046	-.0338	.0009	-.0347	-.0338	.0004	-.0342
0	.0328	0	.0656	0	-.0339	0	-.0339	-.0339	0	-.0339
4	.0327	.00229	.0654	.0046	-.0338	.0009	-.0347	-.0338	.0004	-.0342
8	.0325	.00456	.0650	.0091	-.0336	.0017	-.0353	-.0336	.0008	-.0344

12	13	14	15	16	17	18	19	20
CG Loc 3			CG Loc 4			CG Loc 6		
④ · A_3	⑤ · B_3	C_{MG} ⑫ - ⑬	④ · A_4	⑤ · B_4	C_{MG} ⑮ - ⑯	④ · A_6	⑤ · B_6	C_{MG} ㉑ - ㉒
-.0214	+.0025	-.0239	-.0214	.0012	-.0226	-.0294	.0022	-.0316
-.0216	.0017	-.0233	-.0216	.0008	-.0224	-.0297	.0015	-.0312
-.0217	.0009	-.0226	-.0217	.0004	-.0221	-.0298	.0008	-.0307
-.0218	0	-.0218	-.0218	0	-.0218	-.0300	0	-.0300
-.0217	.0009	-.0226	-.0217	.0004	-.0221	-.0299	.0008	-.0307
-.0216	.0017	-.0233	-.0216	.0008	-.0224	-.0297	.0015	-.0312

PART II - B - 7

7. Propellers (2) - Military Power

$$C_{M_p} = \frac{2}{q S_w MAC_w} \left(-l_p Z_p + z_p X_p \right)$$

$$C_{Z_p} = \frac{Z_p}{q\ F_p}$$

C_{Z_p} from figure 60, reference (1)

$$Z_p = C_{Z_p}\ q\ F_p$$

$$X_p = -T$$

T is from figure 58 reference (1)

Results of calculations are presented in figures 8 through 13

RESTRICTED

REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION	PAGES	PAGE 64
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY
Subject:— BASIC FLIGHT CRITERIA - PACK ON			DATE March 15, 1949 REVISED

PART II - B - 7

PROPELLERS T_C = .639 $V\sqrt{\sigma}$ = 100 MPH   q = 25.58

1	2	3	4	5	6	7	8	9	10	11
						CG. 1				
α_{FL}	ϵ_p	T	Z_p	X_p	$\frac{z}{q S_w MAC_w}$	$-\ell_p Z_p$	$z_p X_p$	$-\ell_p Z_p + z_p X_p$	C_{M_p}	$-\ell_p Z_p$
-12	-.0541	7400	-246.6	-7400	3.21×10^{-7}	-43032	66600	23568	.00757	-47175
-8	-.0372		-169.6			-29595		31005	.01188	-32444
-4	-.0183		-83.4			-14553		52047	.01671	-15954
0	0		0			0		66600	.02138	0
4	.0183		83.4			14553		81153	.02605	15954
8	.0372		169.6			29595		96195	.03088	32444

12	13	14	15	16	17	18	19	20	21	22
CG. 2			CG. 3				CG. 4			
$z_p X_p$	$-\ell_p Z_p + z_p X_p$	C_{M_p}	$-\ell_p Z_p$	$z_p X_p$	$-\ell_p Z_p + z_p X_p$	C_{M_p}	$-\ell_p Z_p$	$z_p X_p$	$-\ell_p Z_p + z_p X_p$	C_{M_p}
66600	19,445	.00624	-43032	-162800	-205832	-.06608	47175	-162800	-209975	-.06741
	34,156	.01097	-29595		-192395	-.06177	-32444		-195244	-.06268
	50,646	.01626	-14553		-177353	-.05694	-15954		-178754	-.05739
	66,600	.02138	0		-162800	-.05227	0		-162800	-.05227
	82,554	.02650	14553		-148247	-.04759	15945		-146846	-.04714
	99,044	.03180	29595		-133205	-.04276	32444		-130356	-.04185

Subject:- BASIC FLIGHT CRITERIA - PACK ON

PART II - B - 7

PROPELLERS

$T_C = .639$ $V\sqrt{\sigma} = 100$ MPH $q = 25.58$

23	24	25	26	27	28	29	30
CG Loc. 5				CG Loc. 6			
$-l_p Z_p$	$\mathfrak{z}_p X_p$	$-l_p Z_p + \mathfrak{z}_p X_p$	C_{M_p}	$-l_p Z_p$	$\mathfrak{z}_p X_p$	$-l_p Z_p + \mathfrak{z}_p X_p$	C_{M_p}
-44610	26048	-18562	-.00596	-43944	-7400	-51344	-.01648
-30681		4633	-.00149	-30223		-37623	-.01208
-15087		10961	.00352	-14862		-22262	-.00715
0		26048	.00836	0		-7400	-.00238
15087		41135	.01321	14862		7462	.00240
30681		56729	.01821	30223		22823	.00733

$T_C = .198$ $V\sqrt{\sigma} = 160$ MPH $q = 65.48$

1	2	3	4	5	6	7	8	9	10	11
					CG Loc. 1					CG 2
$\delta_{T.L.}$	C_{Z_p}	T	Z_p	X_p	$\frac{Z}{q S_w MAC_w}$	$-l_p Z_p$	$\mathfrak{z}_p X_p$	$-l_p Z_p + \mathfrak{z}_p X_p$	C_{M_p}	$-l_p Z_p$
-12	-.0493	5900	-575.3	-5900	1.2541×10^{-8}	-100390	53100	-47290	-.00593	-110055
-8	-.0335		-390.9			-68212		-15178	-.00190	-74779
-4	-.0170		-198.4			-34621		18479	.00232	-37954
0	0		0			0		53100	.00666	0
4	.0170		198.4			34621		87721	.01100	37954
8	.0335		390.9			68212		121312	.01521	79779

REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION	PAGES	PAGE 66
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY

Subject:— BASIC FLIGHT CRITERIA - PACK ON

DATE March 15, 1949
REVISED

PART II - B - 7

PROPELLERS

$T_c = 198$ $V\sqrt{\sigma} = 160$ MPH $\delta = 65.48$

12	13	14	15	16	17	18	19	20	21	22
CG Loc #2			CG Loc #3				CG Loc #4			
$3_p x_p$	$-l_p z_p + 3_p x_p$	C_{m_p}	$-l_p z_p$	$3_p x_p$	$-l_p z_p + 3_p x_p$	C_{m_p}	$-l_p z_p$	$3_p x_p$	$-l_p z_p + 3_p x_p$	C_{m_p}
53100	-56955	-.00714	-100390	-129800	-230190	-.02887	-110055	-129800	-239855	-.03008
	-21679	-.00272	-68212		-198012	-.02483	-74779		-204579	-.02566
	15146	.00190	-34621		-164421	-.02062	-37954		-167754	-.02104
	53100	.00666	0		-129800	-.01628	0		-129800	-.01628
	91054	.01142	34621		-95179	-.01194	37954		-91846	-.01152
	127879	.01605	68212		-61588	-.00772	74779		-55021	-.00690

23	24	25	26	27	28	29	30
CG Loc 5				CG Loc 6			
$-l_p z_p$	$3_p x_p$	$-l_p z_p + 3_p x_p$	C_{m_p}	$-l_p z_p$	$3_p x_p$	$-l_p z_p + 3_p x_p$	C_{m_p}
-104072	20768	-83304	-.01045	-102518	-5900	-108418	-.01360
-70714		-49946	-.00626	-69658		-75558	-.00948
-35891		-15123	-.00190	-35355		-41155	-.00516
0		20768	.00260	0		-5900	-.00074
35891		56659	.00711	35355		29455	.00369
70714		91482	.01147	69658		63758	.00800

REPORT NO. R107-014 FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION | PAGE 67

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

DATE March 15, 1949

REVISED

Subject: BASIC FLIGHT CRITERIA - PACK ON

PART II - B - 7

PROPELLERS

$T_c = .132$ $\qquad V\sqrt{\sigma} = 185$ MPH $\qquad \delta = 87.54$

1	2	3	4	5	6	7	8	9	10	11
							CG Loc 1			CG 2
α_{TL}	C_{Z_p}	T	Z_p	X_p	$\frac{2}{\rho S_w MAC_w}$	$-l_p Z_p$	$j_p X_p$	$-l_p Z_p + j_p X_p$	C_{m_p}	$-l_p Z_p$
-12	-.0505	5270	-787.8	-5270	9.381×10^{-8}	-137314	47430	-89884	-.00843	-150706
-8	-.0350		-546.0			-95168		-47738	-.00448	-104450
-4	-.0175		-273.0			-47584		-154	-.00001	-52225
0	0		0			0		47430	.00445	0
4	.0175		273.0			47584		95014	.00891	52225
8	.0350		546.0			95168		142598	.01338	104450

12	13	14	15	16	17	18	19	20	21	22
CG Loc 2			CG Loc 3				CG Loc 4			
$j_p X_p$	$-l_p Z_p + j_p X_p$	C_{m_p}	$-l_p Z_p$	$j_p X_p$	$-l_p Z_p + j_p X_p$	C_{m_p}	$-l_p Z_p$	$j_p X_p$	$-l_p Z_p + j_p X_p$	C_{m_p}
-47430	-103276	-.00969	-137314	-115940	-253254	-.02376	-150706	-115940	-266648	-.02501
	-57020	-.00535	-95168		-211108	-.01980	-104450		-220390	-.02067
	-4795	-.00045	-47584		-163524	-.01534	-52225		-168165	-.01578
	47430	.00445	0		-115940	-.01088	0		-115940	-.01088
	99655	.00935	47584		-68356	-.00641	52225		-63715	-.00598
	151880	.01425	95168		-20772	-.00195	104450		-11490	-.00108

Subject: BASIC FLIGHT CRITERIA - PACK ON

PART II - B - 7

PROPELLERS

T_c = .132 $V\sqrt{\sigma}$ = 185 MPH δ = 87.54

23	24	25	26	27	28	29	30
CG Loc #5				C.G. Loc. #6			
$-l_pZ_p$	$\mathcal{J}_pX_p$	$-l_pZ_p + \mathcal{J}_pX_p$	C_{M_p}	$-l_pZ_p$	$\mathcal{J}_pX_p$	$-l_pZ_p + \mathcal{J}_pX_p$	C_{M_p}
-142513	18550	-123963	-.01163	-140386	-5270	-145656	-.01366
-98771		-80221	-.00753	-97297		-102567	-.00962
-49386		-30836	-.00289	-48649		-53919	-.00506
0		18550	.00174	0		-5270	-.00049
49386		67936	.00637	48649		43379	.00407
98771		117321	.01101	97297		92027	.00863

T_c = .056 $V\sqrt{\sigma}$ = 250 MPH δ = 160

1	2	3	4	5	6	7	8	9	10	11
						CG Loc #1				
$\alpha_{T.L.}$	C_{Z_p}	T	Z_p	X_p	$\frac{Z}{\delta\, S_w\, MAC_w}$	$-l_pZ_p$	$\mathcal{J}_pX_p$	$-l_pZ_p + \mathcal{J}_pX_p$	C_{M_p}	$-l_pZ_p$
-12	-.0560	4140	-1595.7	-4140	2136 x 10^{-8}	-278450	37260	-241190	-.01239	-305257
-8	-.0381		-1085.6			-189437		-152177	-.00782	-207675
-4	-.0187		-532.8			-92956		-55696	-.00286	-101925
0	0		0			0		37260	.00191	0
4	.0187		532.8			92956		130216	.00669	101925
8	.0381		1085.6			189437		226697	.01164	207675

REPORT NO R107-014 | FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION | PAGE 69

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

DATE March 15, 1949

REVISED

Subject:— BASIC FLIGHT CRITERIA - PACK ON

RESTRICTED

PART II - B - 7

PROPELLERS

$\bar{T} = .056$ $\qquad V\sqrt{\sigma} = 250$ $\qquad \delta = 160$

12	13	14	15	16	17	18	19	20	21	22
CG Loc # 2			CG Loc 3				CG Loc 4			
$3_p X_p$	$-l_p Z_p + 3_p X_p$	C_{mp}	$-l_p Z_p$	$3_p X_p$	$-l_p Z_p + 3_p X_p$	C_{mp}	$-l_p Z_p$	$3_p X_p$	$-l_p Z_p + 3_p X_p$	C_{mp}
37260	-267997	-.01376	-278450	-91080	-369530	-.01898	-305257	-91080	-396337	-.02036
	-170415	-.00875	-189437		-280517	-.01441	-207675		-298755	-.01534
	-64665	-.00382	-92956		-184036	-.00945	-101925		-193005	-.00991
	37260	.00191	0		-91080	-.00468	0		-91080	-.00468
	139185	.00715	92956		1876	.00010	101925		10845	.00056
	244935	.01258	189437		98357	.00505	207675		116595	.00599

23	24	25	26	27	28	29	30
CG Loc 5				CG Loc 6			
$-l_p Z_p$	$3_p X_p$	$-l_p Z_p + 3_p X_p$	C_{mp}	$-l_p Z_p$	$3_p X_p$	$-l_p Z_p + 3_p X_p$	C_{mp}
-288662	14573	-274089	-.01408	-284354	-4140	-288494	-.01482
-198385		-181812	-.00934	-193454		-197594	-.01015
-96384		-81811	-.00420	-94945		-99085	-.00509
0		14573	.00075	0		-4140	-.00021
96384		110957	.00570	94945		90805	.00466
196385		210958	.01083	193454		189314	.00972

RESTRICTED

REPORT NO. R107-014 — FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION — PAGES / PAGE 70

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

DATE March 15, 1949

REVISED

Subject: BASIC FLIGHT CRITERIA - PACK ON

PART II - B - 7

PROPELLERS

$T_C = .028$ $V_{TG} = 313$ MPH $q = 250$

1	2	3	4	5	6	7	8	9	10	11
							CG Loc. 1			
$\alpha_{T.L.}$	C_{Z_p}	T	Z_p	X_p	$\frac{z}{q S_w MAC_w}$	$-l_p Z_p$	$z_p X_p$	$-l_p Z_p + z_p X_p$	C_{m_p}	$-l_p Z_p$
-12	-.0634	3240	-2831	-3240	3.77×10^{-8}	-494010	29160	-464850	-.01523	-541570
-8	-.0422		-1885			-328933		-299773	-.00982	-360601
-4	-.0211		-942			-164379		-135219	-.00443	-180205
0	0		0			0		29160	.00076	0
4	.0211		942			164379		193539	.00634	180205
8	.0422		1885			328933		358093	.01173	360601

12	13	14	15	16	17	18	19	20	21	22
CG Loc 2			CG Loc 3				CG Loc 4			
$z_p X_p$	$-l_p Z_p + z_p X_p$	C_{m_p}	$-l_p Z_p$	$z_p X_p$	$-l_p Z_p + z_p X_p$	C_{m_p}	$-l_p Z_p$	$z_p X_p$	$-l_p Z_p + z_p X_p$	C_{m_p}
29160	-512410	-.01679	-494010	-71280	-565290	-.01852	-541570	-71280	-612850	-.02008
	-332441	-.01066	-328933		-400213	-.01311	-360601		-431881	-.01415
	-151045	-.00495	-164379		-235658	-.00772	-180205		-251485	-.00824
	29160	.00096	0		-71280	-.00234	0		-71280	-.00234
	209365	.00686	164379		93099	.00305	180205		108925	.00357
	387761	.01277	328933		257653	.00844	360601		289321	.00948

Subject – BASIC FLIGHT CRITERIA – PACK ON

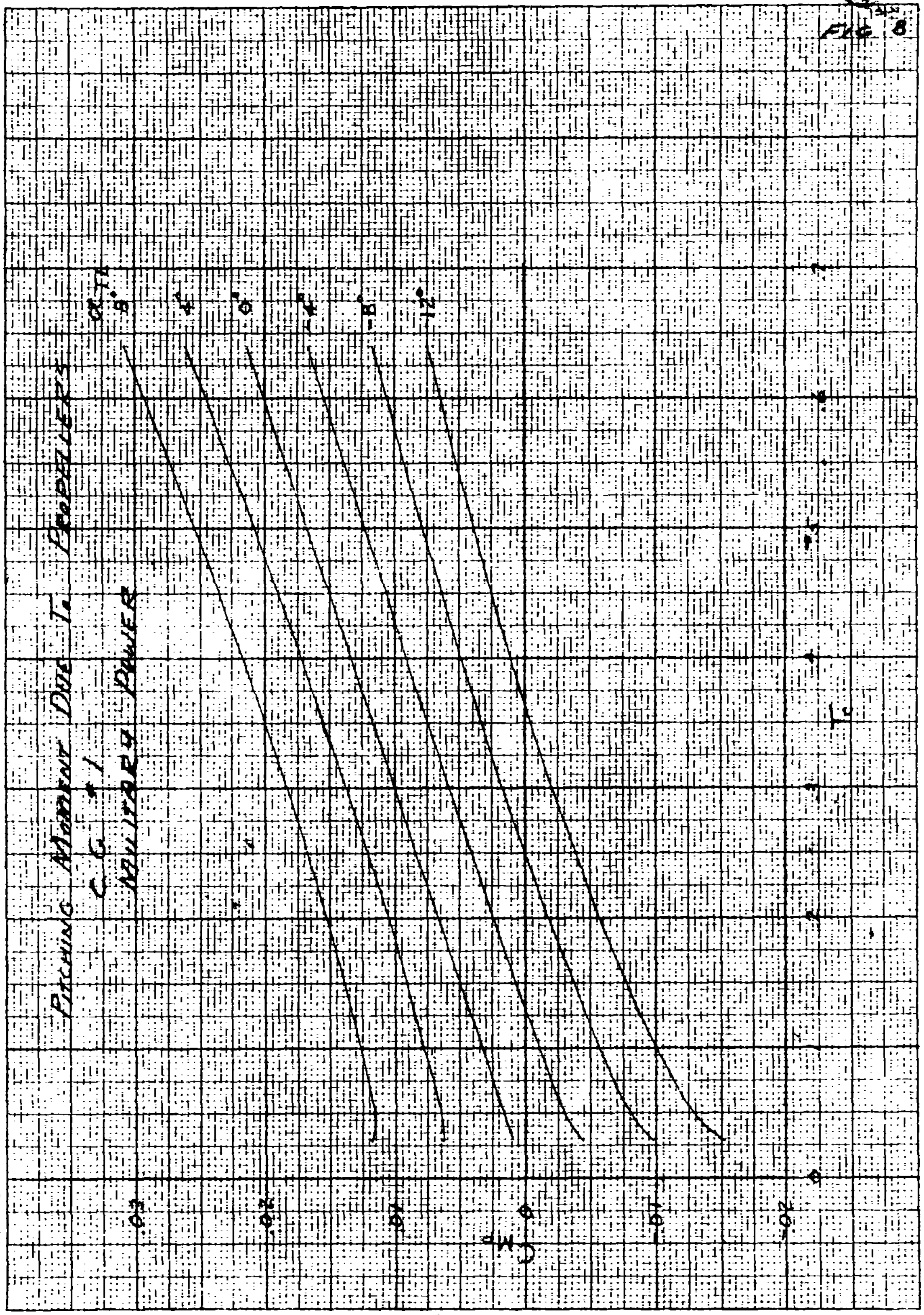
PART II – B – 7

PROPELLERS

$T_c = .028$ $V\sqrt{\sigma} = 313$ MPH $\delta = 250$

23	24	25	26	27	28	29	30
	CG Loc 5				CG Loc 6		
$-l_p z_p$	$\delta_p x_p$	$-l_p z_p + \delta_p x_p$	C_{M_p}	$-l_p z_p$	$\delta_p x_p$	$-l_p z_p + \delta_p x_p$	C_{M_p}
512128	11405	-500723	-.01641	-504484	-3240	-507724	-.01664
-340997		-329592	-.01080	-335907		-339147	-.01111
-170408		-159003	-.00521	-167864		-171104	-.00561
0		11405	.00037	0		-3240	-.00011
170408		181813	.00596	167864		164624	.00539
340997		352402	.01155	335907		332667	.01090
512128		523128	.01714	504484		501244	.01642

Fig 8


Pitching Moment Due To Propellers
C.G. #1
Military Power
α T.L.
8°
4°
0°
-4°
-8°
-12°
.03
.02
.01
0
-.01
-.02
C_{M_P}
T_c

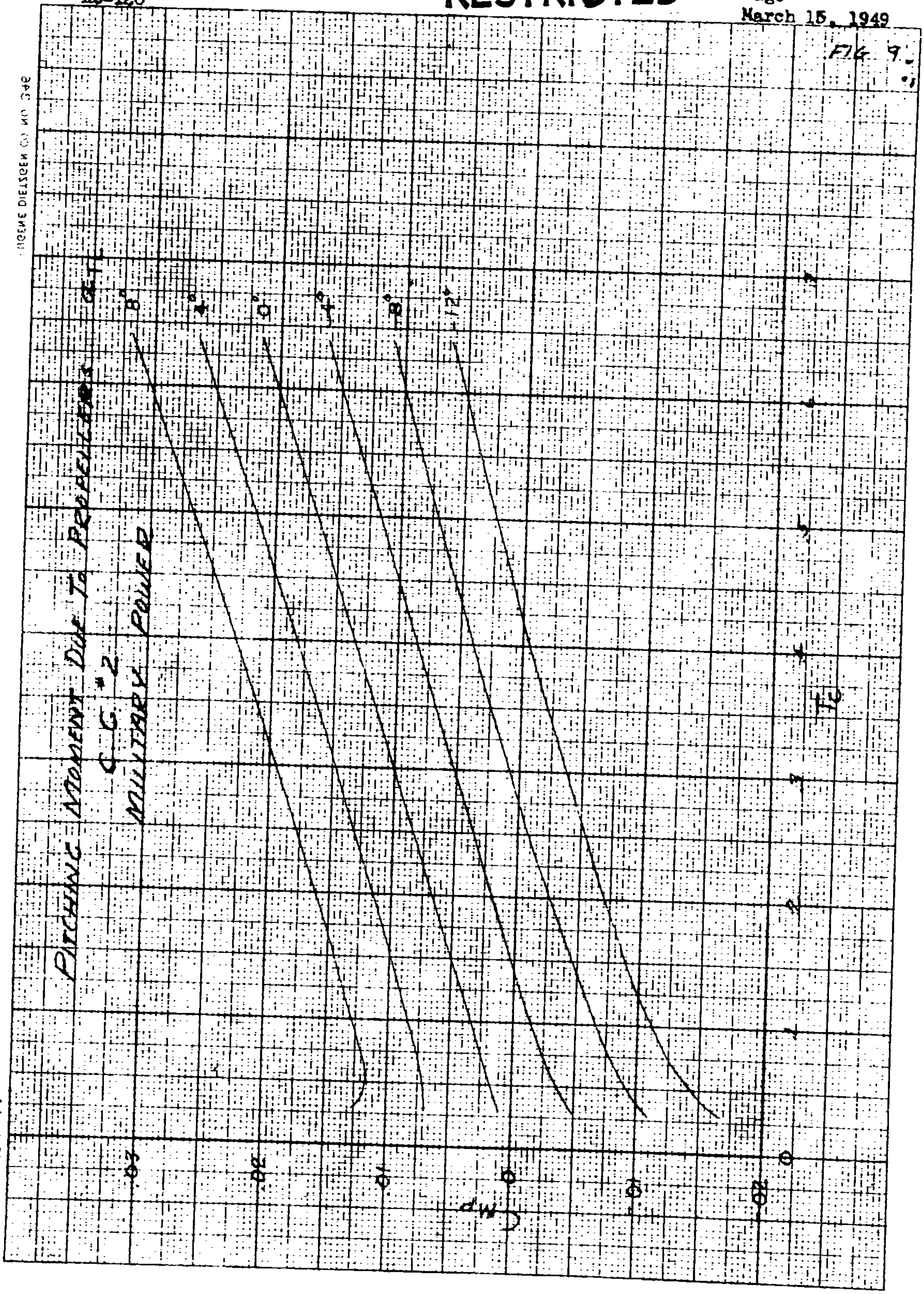


FIG. 9.
PITCHING MOMENT DUE TO PROPELLERS
C.G. #2
MILITARY POWER
8°
4°
0°
-4°
-8°
-12°
.03
.02
.01
0
-.01
-.02
C_{m_p}
T_c
7
6
5
4
3
2
1
0

FIG. 10

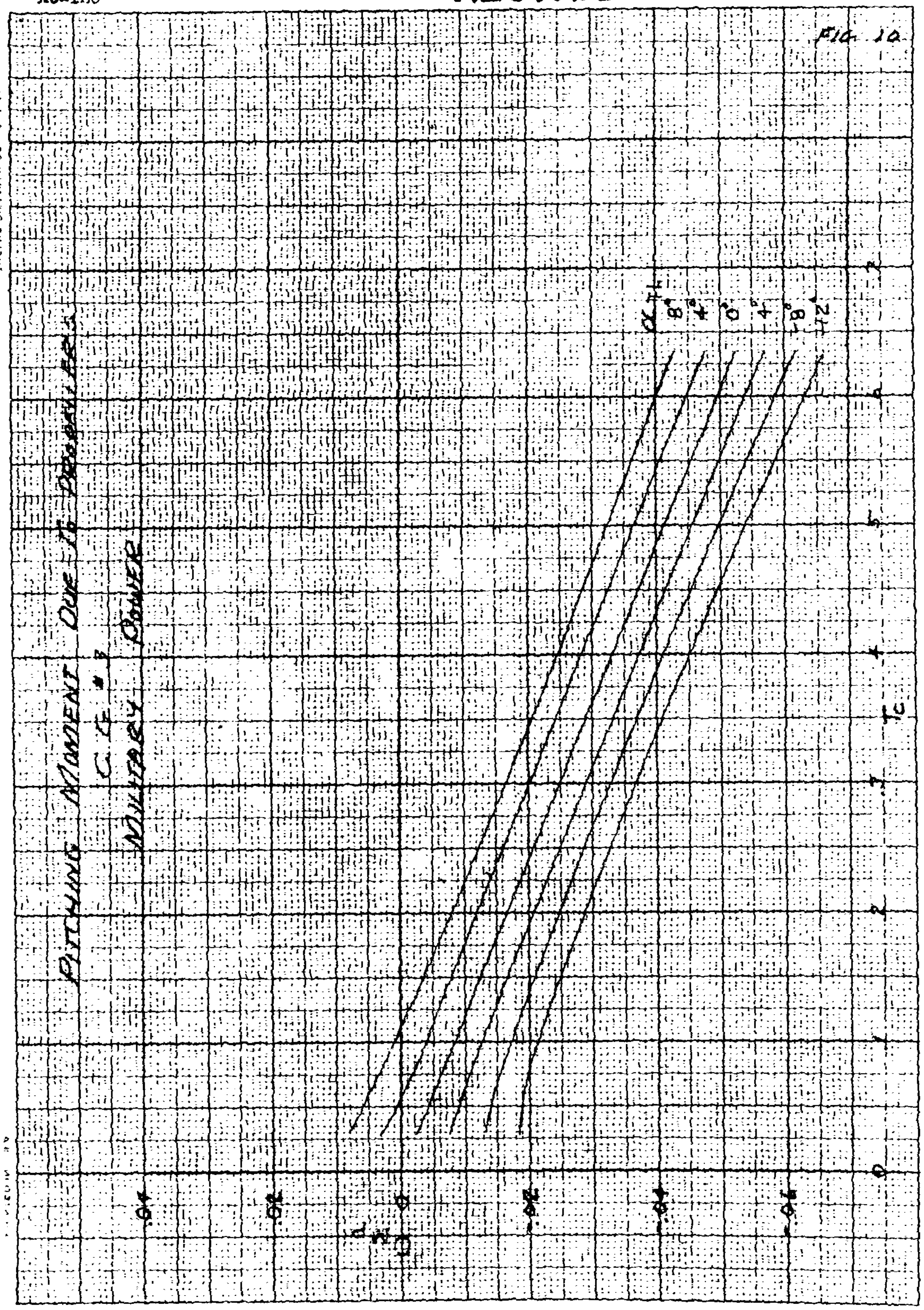

PITCHING MOMENT DUE TO PROPELLERS
MILITARY POWER
α
8°
4°
0°
-4°
-8°
-12°
T_c
0
1
2
3
4
5
6
7
C_{m_P}
.04
.02
0
.02
.04
.06

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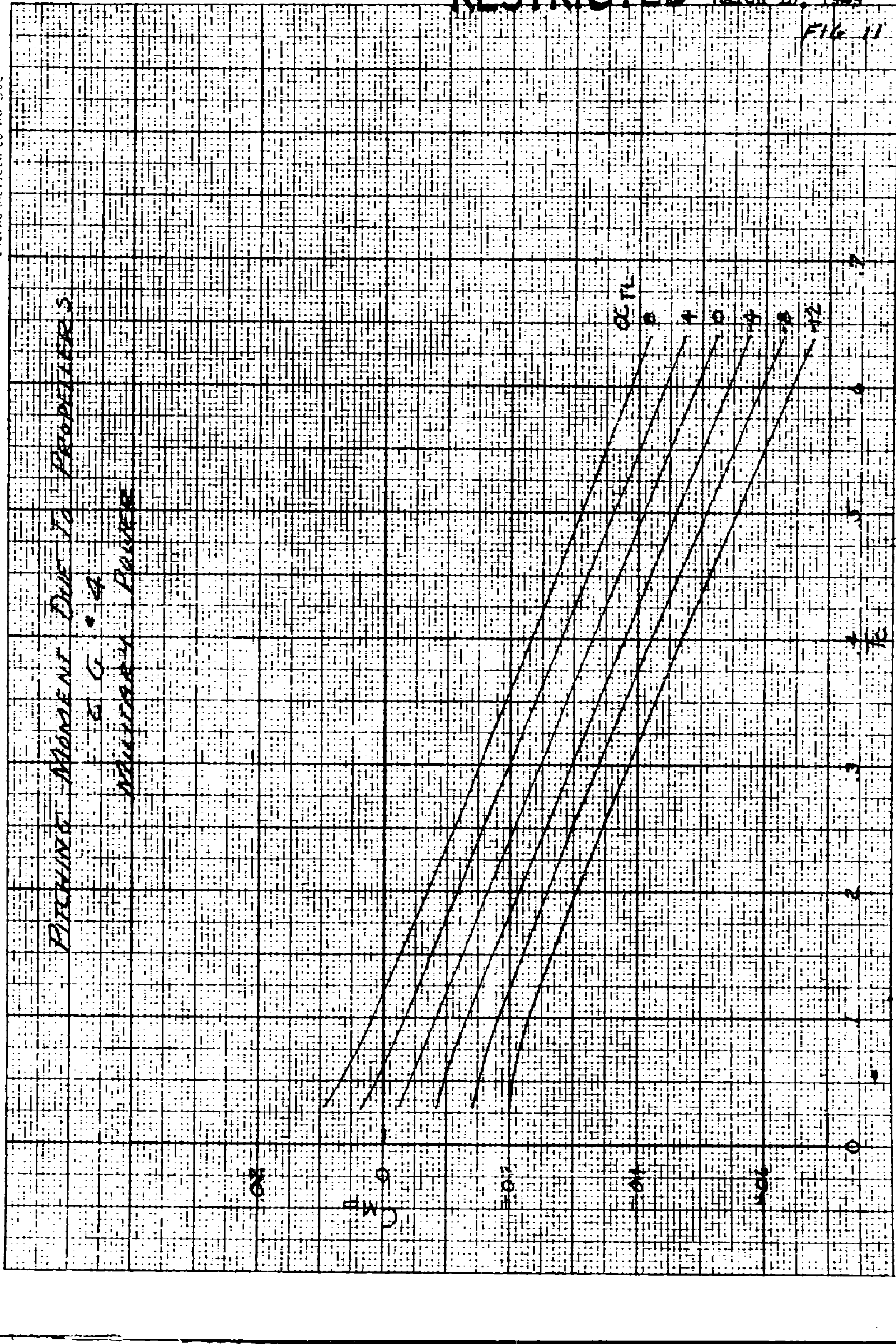
FIG 11



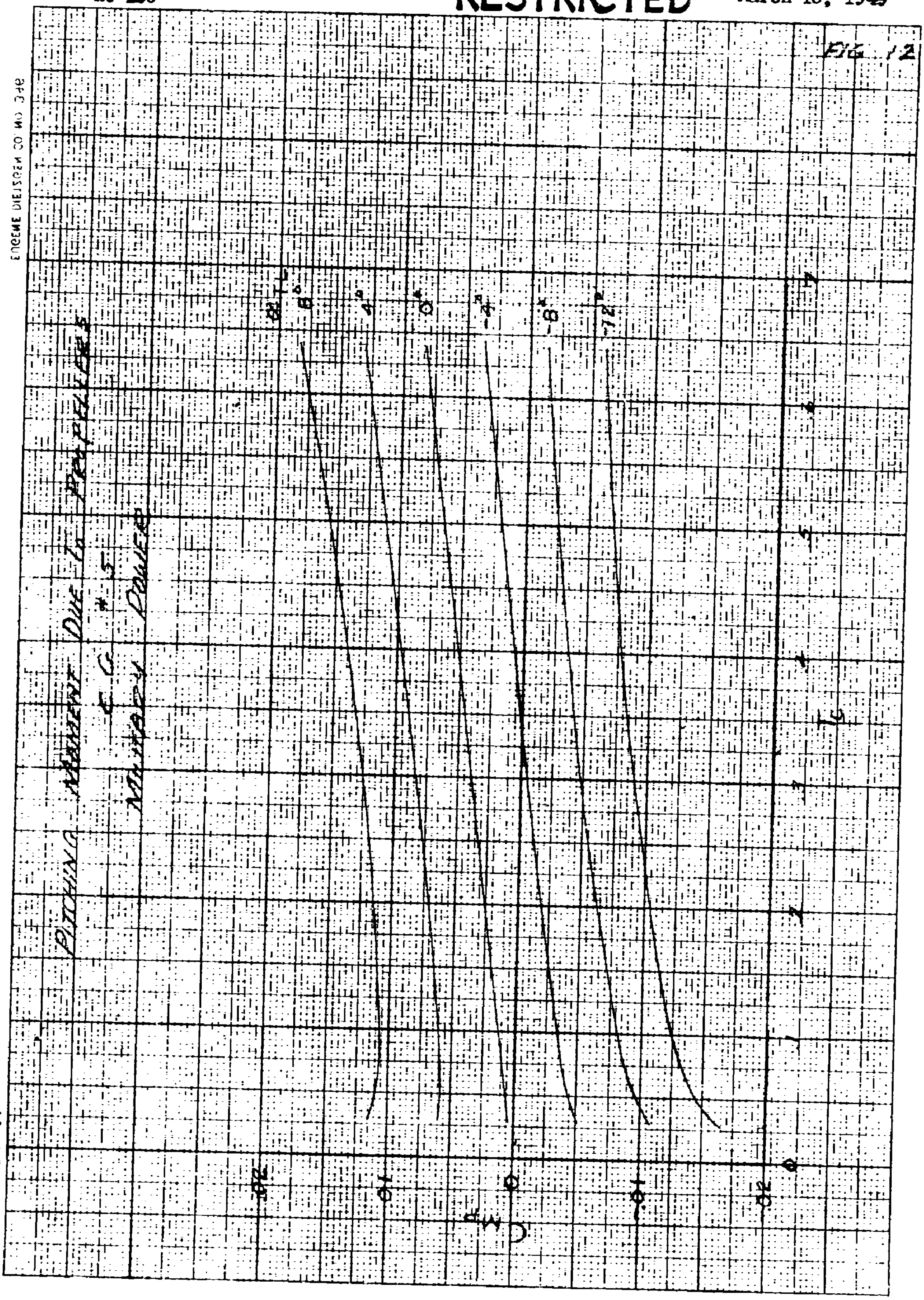

PITCHING MOMENT DUE TO PROPELLERS
C.G. # 5
MILITARY POWER
C_{M_P}
C_L

FIG. 12

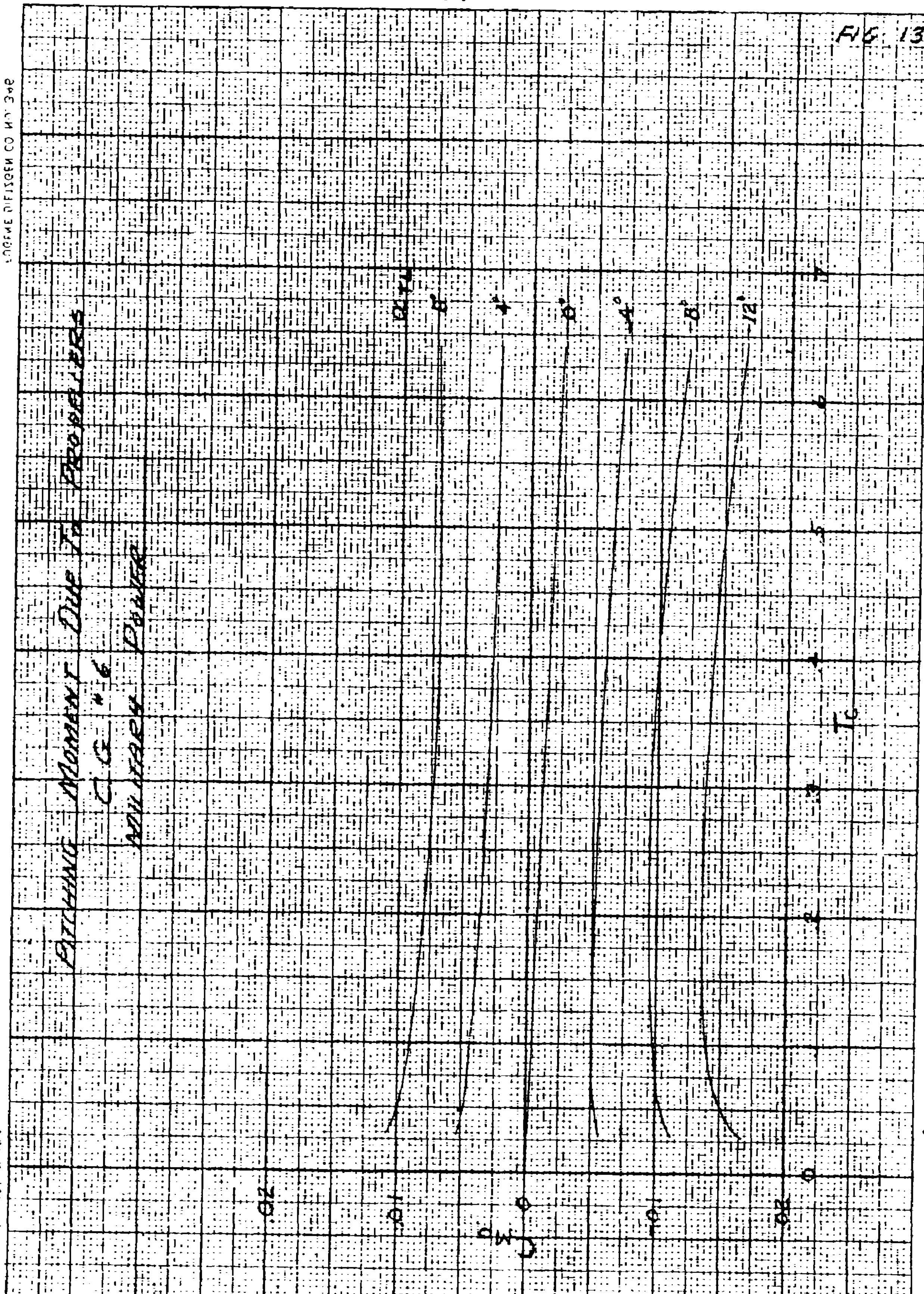


FIG. 13
Pitching Moment Due To Propellers
C.G. 6
Military Power
T_c
C_{m_p}
.02
.01
0
-.01
-.02
0
1
2
3
4
5
6
7

PART II - B - 8

8a. SLIPSTREAM EFFECTS - WING UNFLAPPED

$$C_{M_S} = 2\Delta C_{M_{WI}} + 2\left[C_{Z_{WI}} \quad (A) + \Delta C_{X_{WI}} \quad (B)\right]$$

$\Delta C_{M_{WI}}$ is considered negligible for the unflapped wing

so that

$$C_{M_S} = 2\left[\Delta C_{Z_{WI}} \quad (A) + \Delta C_{X_{WI}} \quad (B)\right]$$

$$A = \left(\frac{x_{wi}}{MAC_w} - \frac{A.C._{wi}}{MAC_w}\right)$$

$$B = \left(\frac{z_{wi}}{MAC_w}\right)$$

$\Delta C_{Z_{WI}}$ from figure 51 reference (1)

$\Delta C_{X_{WI}}$ from figure 50 reference (1)

C. G. Location

1	A_1 = .0148	B_1 = .0824
2	A_2 = .1147	B_2 = .0824
3	A_3 = .0148	B_3 = .2666
4	A_4 = .1147	B_4 = .2666
5	A_5 = .0537	B_5 = .1150
6	A_6 = .0365	B_6 = .1418

Results of calculations are presented in figures 14 through 18.

Subject:— BASIC FLIGHT CRITERIA – PACK ON

PART II – B – 8a

SLIPSTREAM EFFECTS — WING UNFLAPPED

T_c = .639 $V\sqrt{\sigma}$ = 100 MPH

1	2	3	4	5	6	7	8	9	10	11
			C.G. 1			C.G. 2			C.G. 3	
α_{TL}	$\Delta C_{Z_{WI}}$	$\Delta C_{X_{WI}}$	$\Delta C_{Z_{WI}} \cdot A_1$	$\Delta C_{X_{WI}} \cdot B_1$	C_{M_S}	$\Delta C_{Z_{WI}} \cdot A_2$	$\Delta C_{X_{WI}} \cdot B_2$	C_{M_S}	$\Delta C_{Z_{WI}} \cdot A_3$	$\Delta C_{X_{WI}} \cdot B_3$
-2	-.0026	.00525	-.000039	.000432	.000784	-.000293	.000432	.000266	-.000057	.001400
0	.0185	.00334	.000271	.000275	.001072	.002100	.000275	.004750	.000271	.000590
4	.0323	.00227	.000541	.000187	.001540	.004500	.000187	.009378	.000581	.000610
[illegible]	.0603	.00227	.000891	.000240	.002262	.006700	.000240	.014250	.000891	.000776
[illegible]	.0825	.00452	.001220	.000372	.003187	.009450	.000372	.017649	.001220	.001204
[illegible]	.1053	.00765	.001560	.000690	.004350	.012070	.000650	.025700	.001560	.002040

12	13	14	15	16	17	18	19	20	21	
	C.G. 4			C.G. 5			C.G. 6			
C_{M_S}	$\Delta C_{Z_{WI}} \cdot A_4$	$\Delta C_{X_{WI}} \cdot B_4$	C_{M_S}	$\Delta C_{Z_{WI}} \cdot A_5$	$\Delta C_{X_{WI}} \cdot B_5$	C_{M_S}	$\Delta C_{Z_{WI}} \cdot A_6$	$\Delta C_{X_{WI}} \cdot B_6$	C_{M_S}	
.002722	-.000278	.001400	.001204	-.000137	.001603	.000932	-.000595	.000745	.001300	
.002942	.002100	.000590	.005760	.000765	.000384	.002698	.000668	.000473	.002282	
.002362	.004500	.000610	.010220	.002026	.000263	.004666	.001432	.000321	.003572	
.003319	.006900	.000778	.015356	.003110	.000372	.007012	.002200	.000414	.005228	
.004558	.009450	.001204	.021306	.004350	.000520	.009740	.003010	.000640	.007900	
.007200	[illegible]	.002040	.026220	.005552	.000860	.012864	.003645	.001089	.009658	

REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION		PAGES	PAGE 80
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY	
Subject:— BASIC FLIGHT CRITERIA - PACK ON			DATE March 15, 1949	REVISED

PART II - B - 8a

SLIPSTREAM EFFECTS — WING UNFLAPPED

$T_c = .198 \qquad V\sqrt{\sigma} = 160$ MPH

1	2	3	4	5	6	7	8	9	10	11
			C.G. 1			C.G. 2			C.G. 3	
$\alpha_{T.L.}$	$\Delta C_{Z_{W1}}$	$\Delta C_{X_{W1}}$	$\Delta C_{Z_{W1}} \cdot A_1$	$\Delta C_{X_{W1}} \cdot B_1$	C_{m_1}	$\Delta C_{Z_{W1}} A_2$	$\Delta C_{X_{W1}} B_2$	C_{m_2}	$\Delta C_{Z_{W1}} A_3$	$\Delta C_{X_{W1}} B_3$
-12	-.0047	.00067	-.000070	.000170	.000210	-.000590	.000120	-.000120	-.000075	.000557
-8	.0097	.00076	.000120	.000079	.000296	.000590	.000072	.001238	.000170	.000256
-4	.0172	.00044	.001116	.000036	.000492	.001690	.000030	.003332	.000215	.000117
0	.0248	.00095	.001752	.000071	.000558	.002130	.000077	.005007	.000352	.000248
4	.0322	.00221	.002498	.000190	.001326	.003565	.000190	.008110	.000495	.000615
8	.0436	.00412	.001569	.000360	.002056	.005020	.000360	.010800	.000645	.000720

12	13	14	15	16	17	18	19	20	21
	C.G. 4			C.G. 5			C.G. 6		
C_{m_3}	$\Delta C_{Z_{W1}} A_4$	$\Delta C_{X_{W1}} B_4$	C_{m_4}	$\Delta C_{Z_{W1}} A_5$	$\Delta C_{X_{W1}} B_5$	C_{m_5}	$\Delta C_{Z_{W1}} A_6$	$\Delta C_{X_{W1}} B_6$	C_{m_6}
.000112	-.000540	.000551	.000022	-.000245	.000238	-.000120	-.000111	.000234	.000346
.000402	.000540	.000216	.001592	.000249	.000110	.000716	.000171	.000136	.000614
.000447	.001630	.000114	.003495	.000742	.000151	.001000	.000577	.000127	.001162
.001211	.002150	.000295	.005156	.001255	.000117	.003224	[illegible]	.000132	.002602
.002230	.002605	.001615	.008060	.001726	.001268	.004083	.000235	.000327	.003114
.002956	.005020	.001230	.012500	.002307	.000531	.005676	.001599	.000655	.004506

PART II - B - 8a

SLIPSTREAM EFFECTS — NO FLAPS

T_c = .132 $V\sqrt{\sigma}$ = 185 MPH

1	2	3	4	5	6	7	8	9	10	11
			CG Loc #1			CG Loc #2			CG Loc #3	
α_{TL}	$\Delta C_{Z_{W1}}$	$\Delta C_{X_{W1}}$	$\Delta C_{Z_{W1}} \cdot A_1$	$\Delta C_{X_{W1}} \cdot B_1$	C_{M_S}	$\Delta C_{Z_{W1}} \cdot A_2$	$\Delta C_{X_{W1}} \cdot B_2$	C_{M_S}	$\Delta C_{Z_{W1}} \cdot A_3$	$\Delta C_{X_{W1}} \cdot B_3$
-12	-.0037	.00170	-.000547	.000140	.000171	-.000434	.000140	-.000588	-.001059	.000452
-8	.0030	.00061	.000494	.000050	.000189	.000329	.000050	.000186	.000099	.000162
-4	.0100	.00018	.000198	.000014	.000324	.001147	.000014	.002232	.000198	.000047
0	.0115	.00065	.000249	.000155	.000594	.001692	.000053	.003870	.000249	.000123
4	.0230	.00196	.000949	.000161	.001020	.00272	.000161	.005162	.000391	.000522
8	.0312	.00420	.000962	.000346	.001616	.00258	.000296	.007852	.000462	.00121

12	13	14	15	16	17	18	19	20	21
	CG Loc #4			CG Loc #5			CG Loc #6		
C_{M_S}	$\Delta C_{Z_{W1}} \cdot A_4$	$\Delta C_{X_{W1}} \cdot B_4$	C_{M_S}	$\Delta C_{Z_{W1}} \cdot A_5$	$\Delta C_{X_{W1}} \cdot B_5$	C_{M_S}	$\Delta C_{Z_{W1}} \cdot A_6$	$\Delta C_{X_{W1}} \cdot B_6$	C_{M_S}
.000[illegible]77	-.000434	.000452	.00064	-.000195	.000195	0	-.000135	.000241	.00121
.000912	.000324	.000162	.00101	.000158	.000170	.00046	.000109	.000056	.00139
.000896	.00114[illegible]	.000047	.00229	.000527	.000021	.00110	.000365	.000026	.00076
.00[illegible]	.001692	.000123	.00413	.000869	.000075	.00163	[illegible]	.000092	.00139
.001142	.00272	.000522	.00696	.001242	.000225	.00232	.000[illegible]	.000279	.00228
.003349	.00[illegible]	.000121	.00930	.001143	.000453	.00424	.001123	.000595	.003468

RESTRICTED

REPORT NO. R137-014	FAIRCHILD AIRCRAFT DIVISION of FAIRCHILD ENGINE & AIRPLANE CORPORATION		PAGES	PAGE 82
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY	
			DATE March 15, 1949	
SUBJECT BASIC FLIGHT CRITERIA - PACK ON			REVISED	

PART II - B - 8a

SLIPSTREAM EFFECTS — NO FLAPS

T_c = .036 VTC = 250 MPH

1	2	3	4	5	6	7	8	9	10	11
			C.G. 1			C.G. 2			C.G. 3	
[illegible]	$\Delta C_{Z_{WI}}$	$\Delta C_{X_{WI}}$	$\Delta C_{Z_{WI}} \cdot A_1$	$\Delta C_{X_{WI}} \cdot B_1$	C_{M_1}	$\Delta C_{Z_{WI}} \cdot A_2$	$\Delta C_{X_{WI}} \cdot B_2$	C_{M_2}	$\Delta C_{Z_{WI}} \cdot A_3$	$\Delta C_{X_{WI}} \cdot B_3$
[illegible]	[illegible]	[illegible]	.000276	[illegible]	[illegible]	-.000229	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	.000128	[illegible]	.000104	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	-.000065	.00011	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	.000266	[illegible]	[illegible]	.00111	[illegible]	[illegible]
[illegible]	[illegible]	.000171	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	.00262	[illegible]	[illegible]
[illegible]	[illegible]	.000315	[illegible]	.000309	.001045	[illegible]	.000309	.00394	[illegible]	[illegible]

12	13	14	15	16	17	18	19	20	21	
	C.G. 4			C.G. 5			C.G. 6			
C_{M}	$\Delta C_{Z_{WI}} \cdot A_4$	$\Delta C_{X_{WI}} \cdot B_4$	C_{M_4}	$\Delta C_{Z_{WI}} \cdot A_5$	$\Delta C_{X_{WI}} \cdot B_5$	C_{M_5}	$\Delta C_{Z_{WI}} \cdot A_6$	$\Delta C_{X_{WI}} \cdot B_6$	C_{M_6}	
[illegible]	-.000220	[illegible]	.00019	-.000105	.000146	[illegible]	-.000093	.000123	[illegible]	
[illegible]	[illegible]	[illegible]	.00041	.000053	[illegible]	.000156	[illegible]	[illegible]	[illegible]	
[illegible]	[illegible]	[illegible]	.00070	.000216	[illegible]	.000416	[illegible]	[illegible]	[illegible]	
[illegible]	[illegible]	[illegible]	.00183	[illegible]	[illegible]	.000534	[illegible]	.000247	[illegible]	
[illegible]	.001216	[illegible]	.00324	.000556	.000174	.001464	.000347	.000219	.001402	
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	.000432	[illegible]	[illegible]	.000532	.002139	

Subject: BASIC FLIGHT CRITERIA - PACK ON

PART II - B - 8a

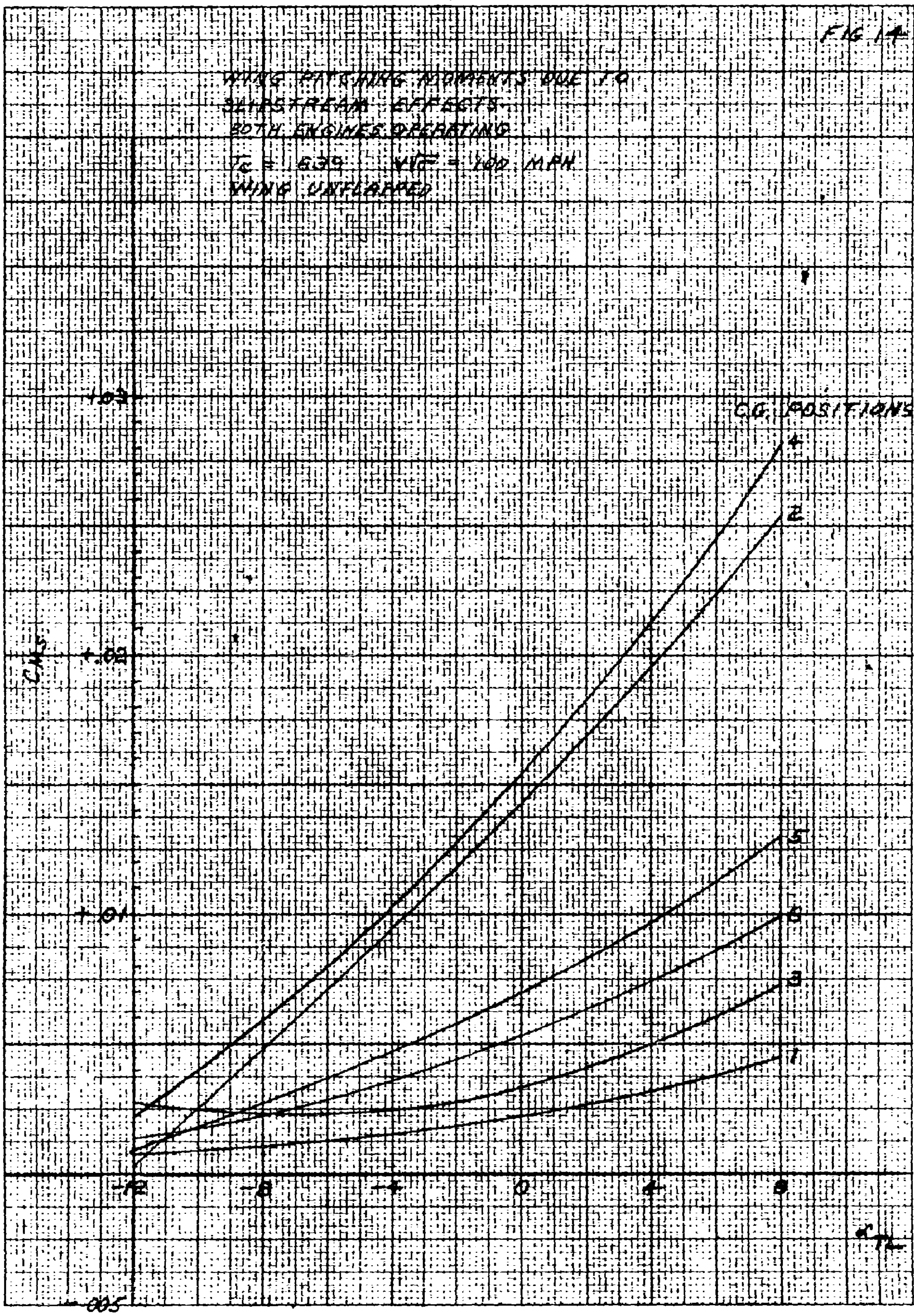
SLIPSTREAM EFFECTS — NO FLAPS

T_c = .028 $V\sqrt{\sigma}$ = 313 MPH

1	2	3	4	5	6	7	8	9	10	11
			C.G. 1			C.G. 2			C.G. 3	
i_T	$\Delta C_{Z_{W1}}$	$\Delta C_{X_{W1}}$	$\Delta C_{Z_{W1}} \cdot A_1$	$\Delta C_{X_{W1}} \cdot B_1$	C_{M_S}	$\Delta C_{Z_{W1}} \cdot A_2$	$\Delta C_{X_{W1}} \cdot B_2$	C_{M_S}	$\Delta C_{Z_{W1}} \cdot A_3$	$\Delta C_{X_{W1}} \cdot B_3$
-2	[illegible]	.00117	-.000015	.000088	.000126	-.000115	.000063	-.000054	-.000015	.000265
[illegible]	.00113	.00011	.000014	.000009	.000026	.000014	.000070	.000278	.000014	.000229
[illegible]	[illegible]	-.00022	.000036	-.000010	.000024	.000229	-.000013	.000432	.000030	-.000045
[illegible]	.00057	.00024	.000057	.000024	.000155	.000345	.000026	.000896	.000056	.000064
4	.00115	.00146	.000076	.000115	.000344	.000668	.000115	.001446	.000078	.000322
6	.00179	[illegible]	.000117	.000268	.000310	.000960	.000288	.003368	.000117	.000936

12	13	14	15	16	17	18	19	20	21
		C.G. 4			C.G. 5			C.G. 6	
C_{M_S}	$\Delta C_{Z_{W1}} \cdot A_4$	$\Delta C_{X_{W1}} \cdot B_4$	C_{M_S}	$\Delta C_{Z_{W1}} \cdot A_5$	$\Delta C_{X_{W1}} \cdot B_5$	C_{M_S}	$\Delta C_{Z_{W1}} \cdot A_6$	$\Delta C_{X_{W1}} \cdot B_6$	C_{M_S}
.000549	-.000115	.000485	+.000342	-.000063	.000123	+.000190	-.000136	.000232	+.000232
.000066	.000034	[illegible]	.000126	.000016	.000013	[illegible]	[illegible]	.000016	.000029
-.000026	.000229	.000045	.000328	.000144	-.000018	.000196	[illegible]	-.000023	+.000100
.000238	.000125	.000014	.000277	.000136	.000026	.000476	.000122	.000044	.000338
.000900	.000168	.000472	[illegible]	.000356	.000161	.000842	.000123	.000126	.000742
.002094	.000266	[illegible]	.003612	.000417	.000402	.001630	.000255	.000496	.001568

FIG 14


WING PITCHING MOMENTS DUE TO
SLIPSTREAM EFFECTS
BOTH ENGINES OPERATING
T_c = 639 V_{IC} = 100 MPH
WING UNFLAPPED
C.G. POSITIONS
4
2
5
6
3
1
C_{M_S}
+.03
+.02
+.01
-.005
-12
-8
-4
0
4
8
C_{TL}

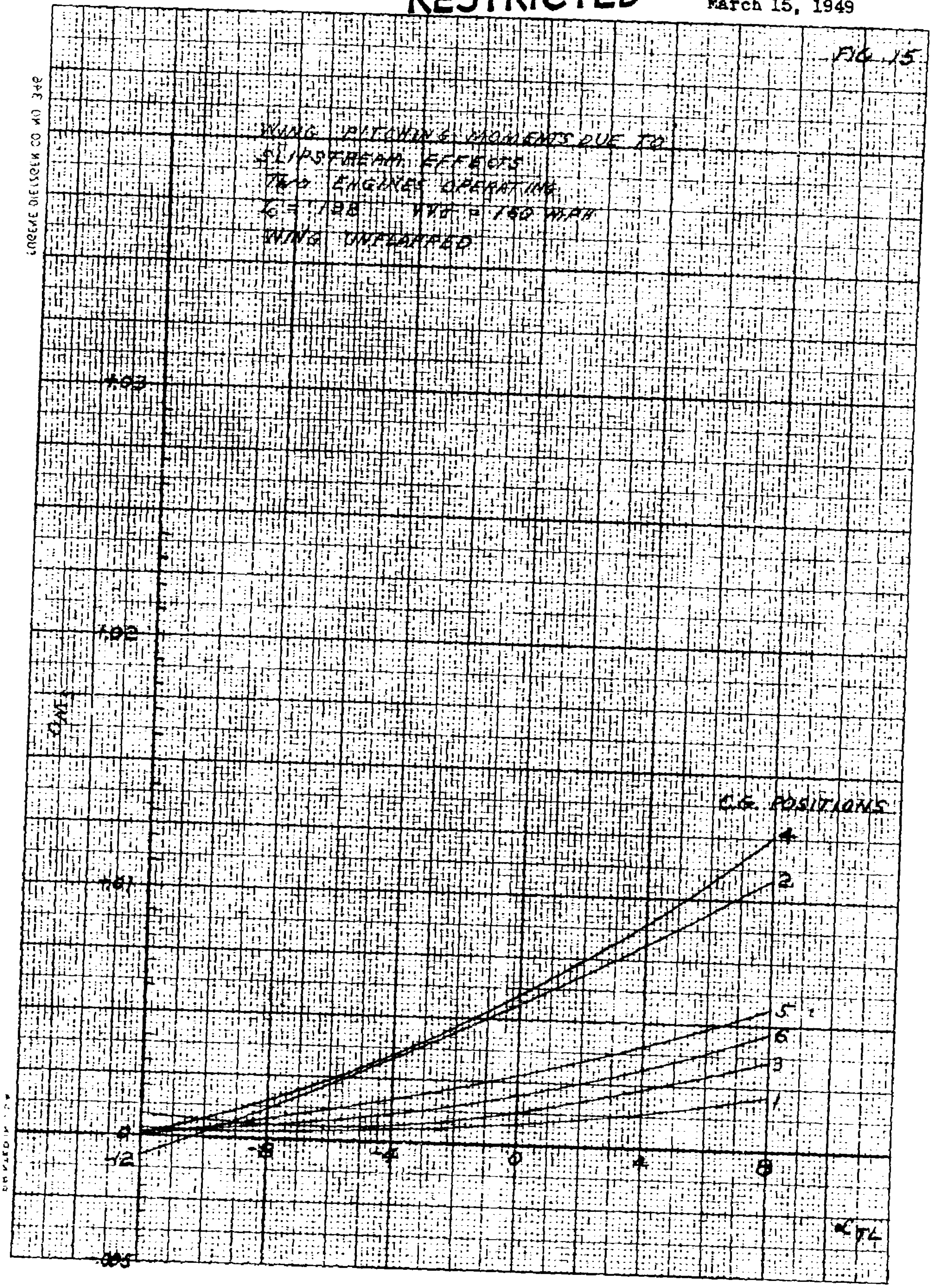


FIG. 15
WING PITCHING MOMENTS DUE TO
SLIPSTREAM EFFECTS
TWO ENGINES OPERATING
C = .198 VVD = 160 MPH
WING UNFLAPPED
.03
.02
.01
0
-.005
C.G. POSITIONS
4
2
5
6
3
1
-12
-8
-4
0
4
8
C_{TL}

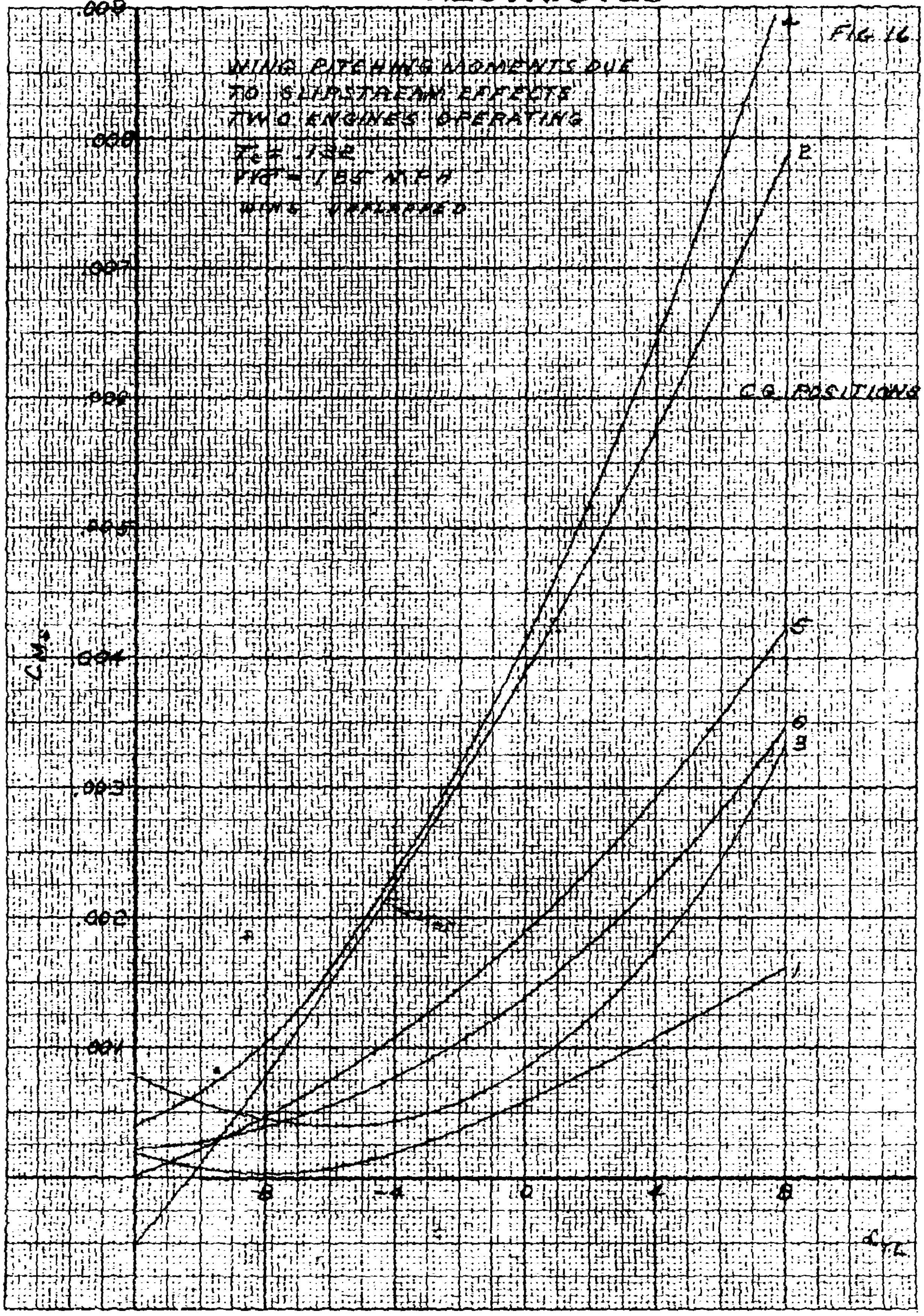


FIG. 16
WING PITCHING MOMENTS DUE
TO SLIPSTREAM EFFECTS
TWO ENGINES OPERATING
.009
.008
.007
.006
.005
.004
.003
.002
.001
CG POSITIONS
1
2
3
4
5
6
-8
-4
0
4
8
C_{M_s}
α_{FL}

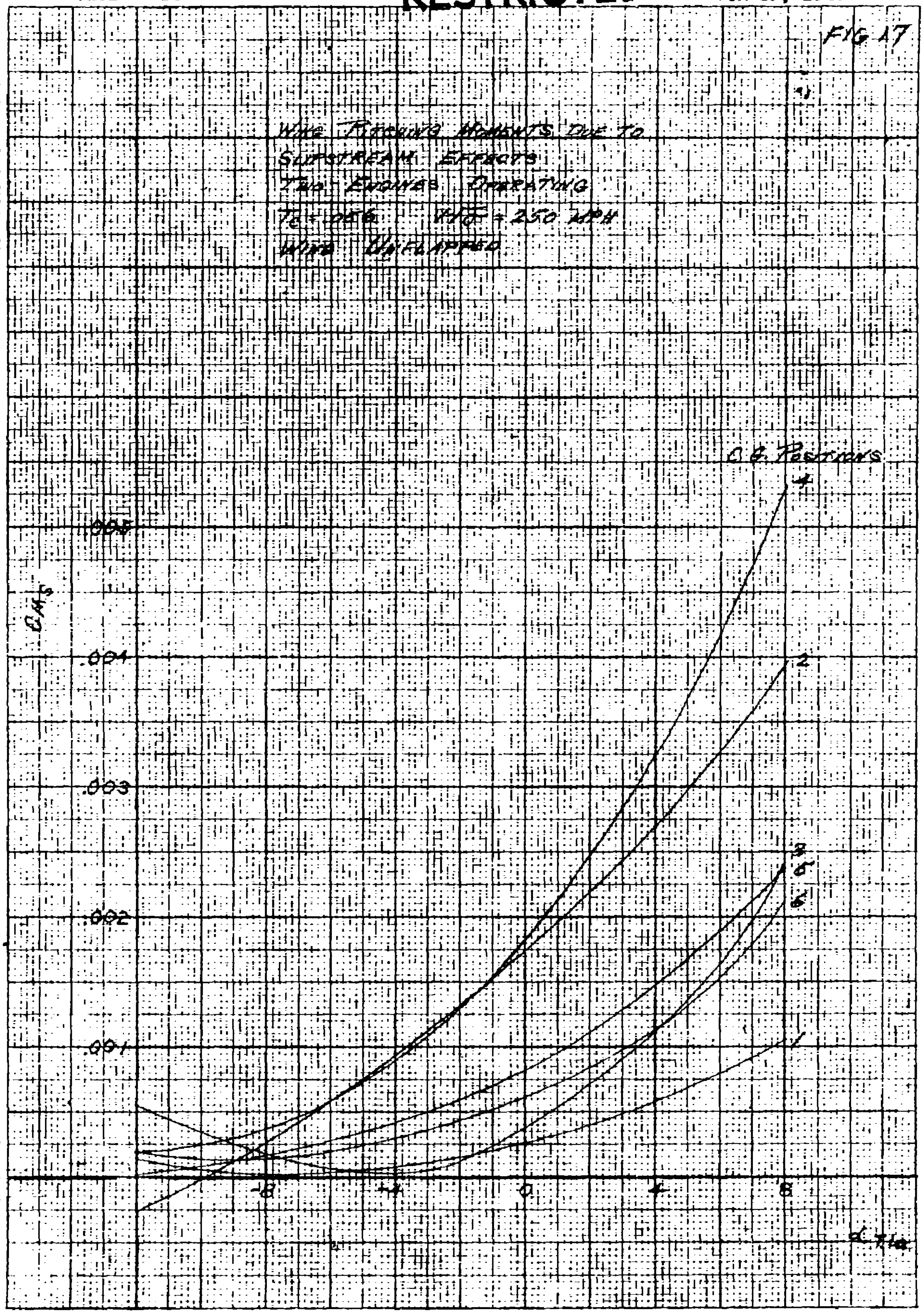
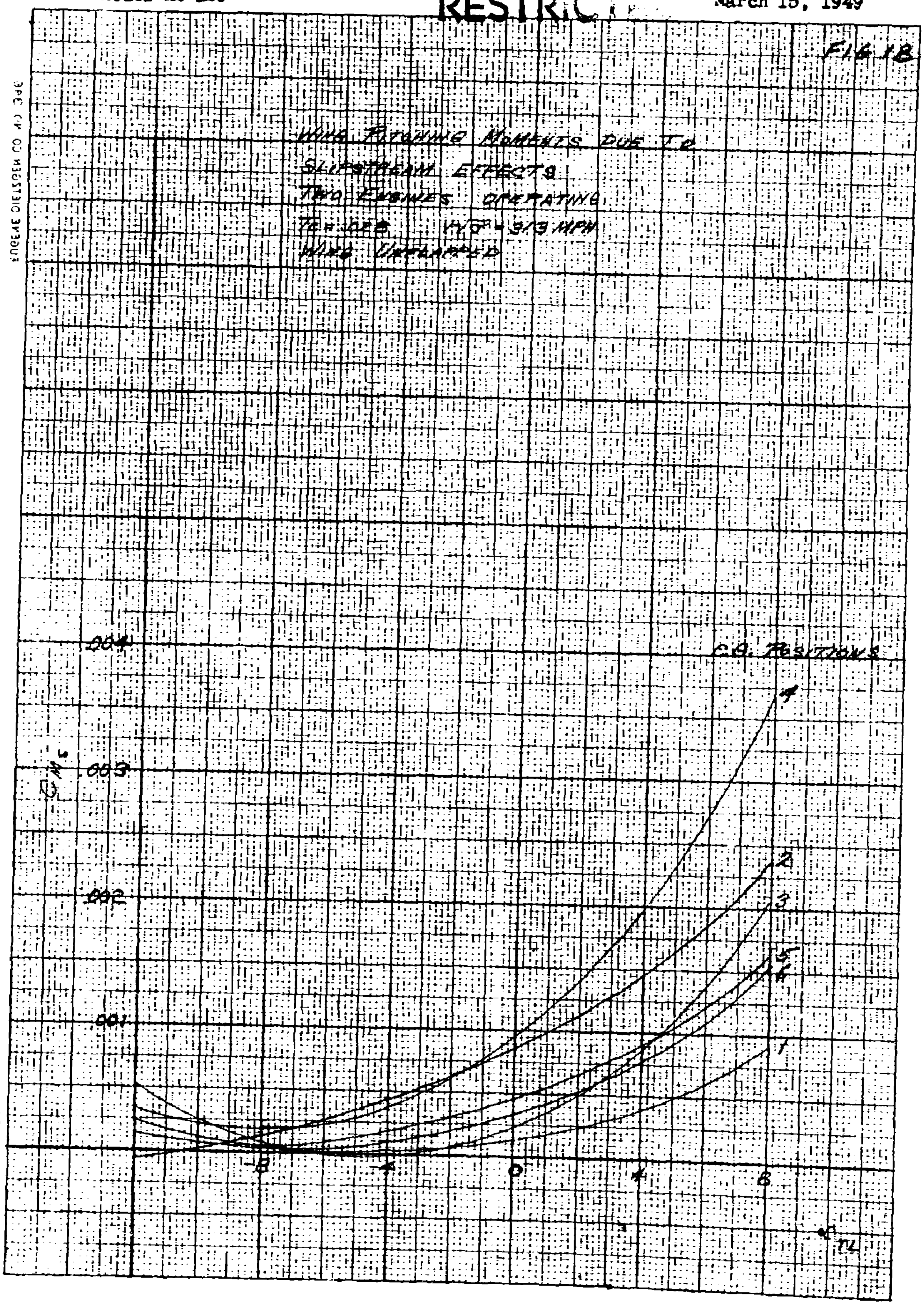


FIG 17
WING PITCHING MOMENTS DUE TO
SLIPSTREAM EFFECTS
TWO ENGINES OPERATING
WING UNFLAPPED
C.G. POSITIONS
4
2
3
5
6
1
.005
.004
.003
.002
.001
-8
-4
0
4
8

FIG 18


WING PITCHING MOMENTS DUE TO
SLIPSTREAM EFFECTS
TWO ENGINES OPERATING
WING UNFLAPPED
.004
.003
.002
.001
C.G. POSITIONS
4
2
3
5
6
1
-8
-4
0
4
8

PART II - B-8

8b. SLIPSTREAM EFFECTS - WING FLAPPED

$$C_{M_S} = 2\Delta C_{M_{WFI}} + 2\left[\Delta C_{Z_{WFI}}\ (A) + \Delta C_{X_{WFI}}\ (B)\right]$$

where

$$A = \left(\frac{x_{wi}}{MAC_w} - \frac{A.C._{wi}}{MAC_w}\right)$$

$$B = \left(\frac{z_{wi}}{MAC_w}\right)$$

$\Delta C_{M_{WFI}}$ from figure 52 reference (1)

$\Delta C_{Z_{WFI}}$ from figure 56 reference (1)

$\Delta C_{X_{WFI}}$ from figure 55 reference (1)

C. G. Location

1	$A_1 = .0148$	$B_1 = .0824$
2	$A_2 = .1147$	$B_2 = .0824$
3	$A_3 = .0148$	$B_3 = .2666$
4	$A_4 = .1147$	$B_4 = .2666$
5	$A_5 = .0527$	$B_5 = .1160$
6	$A_6 = .0365$	$B_6 = .1418$

Results of calculations are presented in figures 19 through 23.

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REPORT NO. R107-014 FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION

PAGES | PAGE 90

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

DATE March 15, 1949

REVISED

Subject: Basic Flight Criteria - Pack On

PART II - B - 8b

SLIPSTREAM EFFECTS — WING FLAPPED

$T_c = .639$ $\sqrt{\sigma} V = 100$ MPH

1	2	3	4	5	6	7	8	9	10	11
					CG 1			CG 2		
$\alpha_{T.L}$	$\Delta C_{M_{WFI}}$	$2\Delta C_{M_{WFI}}$	$\Delta C_{Z_{WFI}}$	$\Delta C_{X_{WFI}}$	④ $\cdot A_1$	⑤ $\cdot B_1$	C_{M_S}	④ $\cdot A_2$	⑤ $\cdot B_2$	C_{M_S}
-12	-.0825	-.1650	.0700	.0104	.0010	.0009	-.1612	.0080	.0009	-.1472
-8	-.0858	1116	.0902	.0104	.0013	.0009	-.1672	.0103	.0009	-.1472
-4	-.0885	-.1770	.1318	.0097	.0020	.0008	-.1714	.0151	.0008	-.1452
0	-.0900	-.1800	.1740	.0080	.0026	.0007	-.1734	.0200	.0007	-.1386
4	-.0908	-.1816	.2150	.0061	.0032	.0005	-.1742	.0247	.0005	-.1312
8	-.0911	-.1822	.2575	.0048	.0038	.0004	-.1738	.0295	.0004	-.1224

12	13	14	15	16	17	18	19	20
CG 3			CG 4			CG 6		
④ $\cdot A_3$	⑤ $\cdot B_3$	C_{M_S}	④ $\cdot A_4$	⑤ $\cdot B_4$	C_{M_S}	④ $\cdot A_6$	⑤ $\cdot B_6$	C_{M_S}
.0010	.0028	-.1574	.0080	.0028	-.1434	.0026	.0015	-.1568
.0013	.0028	-.1634	.0103	.0028	-.1454	.0033	.0015	-.1620
.0020	.0026	-.1678	.0151	.0026	-.1416	.0048	.0014	-.1646
.0026	.0021	-.1706	.0200	.0021	-.1358	.0064	.0011	-.1650
.0032	.0016	-.1720	.0247	.0016	-.1290	.0079	.0009	-.1640
.0038	.0013	-.1720	.0295	.0013	-.1206	.0094	.0007	-.1620

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DATE March 15, 1949

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Subject:— BASIC FLIGHT CRITERIA - PACK ON

PART II - B - 8b

SLIPSTREAM EFFECTS — WING FLAPPED

$T_c = .40$

1	2	3	4	5	6	7	8	9	10	11
					CG 1			CG 2		
$\alpha_{T.L.}$	$\Delta C_{m_{WFI}}$	$2\Delta C_{m_{WFI}}$	$\Delta C_{Z_{WFI}}$	$\Delta C_{X_{WFI}}$	(4)·A_1	(5)·B_1	C_{m_S}	(4)·A_2	(5)·B_2	C_{m_S}
-12	-.0531	-.1062	.0456	.0062	.0007	.0005	-.1038	.0052	.0005	-.0948
-8	-.0553	-.1106	.0610	.0062	.0009	.0005	-.1078	.0070	.0005	-.0956
-4	-.0570	-.1140	.0968	.0058	.0013	.0005	-.1104	.0104	.0005	-.0922
0	-.0581	-.1162	.1205	.0047	.0018	.0004	-.1118	.0138	.0004	-.0878
4	-.0585	-.1170	.1505	.0036	.0022	.0003	-.1120	.0173	.0003	-.0818
8	-.0593	-.1186	.1511	.0030	.0027	.0002	-.1128	.0208	.0002	-.0766

12	13	14	15	16	17	18	19	20
CG 3			CG 4			CG 6		
(4)·A_3	(5)·B_3	C_{m_S}	(4)·A_4	(5)·B_4	C_{m_S}	(4)·A_6	(5)·B_6	C_{m_S}
.0007	.0017	-.1014	.0052	.0017	-.0424	.0017	.0009	-.1010
.0009	.0017	-.1054	.0070	.0017	-.0432	.0032	.0009	-.1044
.0013	.0015	-.1084	.0104	.0015	-.0902	.0033	.0008	-.1058
.0018	.0013	-.1100	.0138	.0013	-.0860	.0044	.0007	-.1060
.0022	.0010	-.1106	.0173	.0010	-.0804	.0055	.0005	-.1050
.0027	.0008	-.1110	.0208	.0008	-.0754	.0066	.0004	-.1046

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REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION	PAGES	PAGE 92
MODEL XC-120	PREPARED BY / CHECKED BY	APPROVED BY	

Subject:— BASIC FLIGHT CRITERIA - PACK ON

DATE March 15, 1949

REVISED

PART II - B - 8b

SLIPSTREAM EFFECTS - WING FLAPPED

$T_c = .198$ $V\sqrt{\sigma} = 160$ MPH

1	2	3	4	5	6	7	8	9	10	11
$\alpha_{T.L}$	$\Delta C_{m_{WFT}}$	$2\Delta C_{m_{WFT}}$	$\Delta C_{Z_{WFT}}$	$\Delta C_{X_{WFT}}$	④·A_1	⑤·B_1	C_{m_S}	④·A_2	⑤·B_2	C_{m_S}
-12	-.0266	-.0532	.0235	.0025	.0003	.0002	-.0522	.0027	.0002	-.0474
-8	-.0277	-.0554	.0325	.0025	.0005	.0002	-.0540	.0037	.0002	-.0476
-4	-.0285	-.0570	.0445	.0024	.0007	.0002	-.0552	.0057	.0002	-.0452
0	-.0293	-.0586	.0675	.0022	.0010	.0002	-.0562	.0077	.0002	-.0425
4	-.0295	-.0590	.0850	.0020	.0013	.0002	-.0560	.0097	.0002	-.0392
8	-.0300	-.0600	.1015	.0027	.0015	.0002	-.0566	.0116	.0002	-.0364

12	13	14	15	16	17	18	19	20
CG 3			CG 4			CG 6		
④·A_3	⑤·B_3	C_{m_S}	④·A_4	⑤·B_4	C_{m_S}	④·A_6	⑤·B_6	C_{m_S}
.0003	.0007	-.0512	.0027	.0007	-.0464	.0009	.0004	-.0506
.0005	.0007	-.0530	.0037	.0007	-.0466	.0012	.0004	-.0522
.0007	.0006	-.0544	.0057	.0006	-.0444	.0018	.0003	-.0528
.0010	.0006	-.0554	.0077	.0006	-.0420	.0025	.0003	-.0530
.0013	.0005	-.0554	.0097	.0005	-.0386	.0031	.0003	-.0522
.0015	.0007	-.0556	.0116	.0007	-.0354	.0037	.0004	-.0518

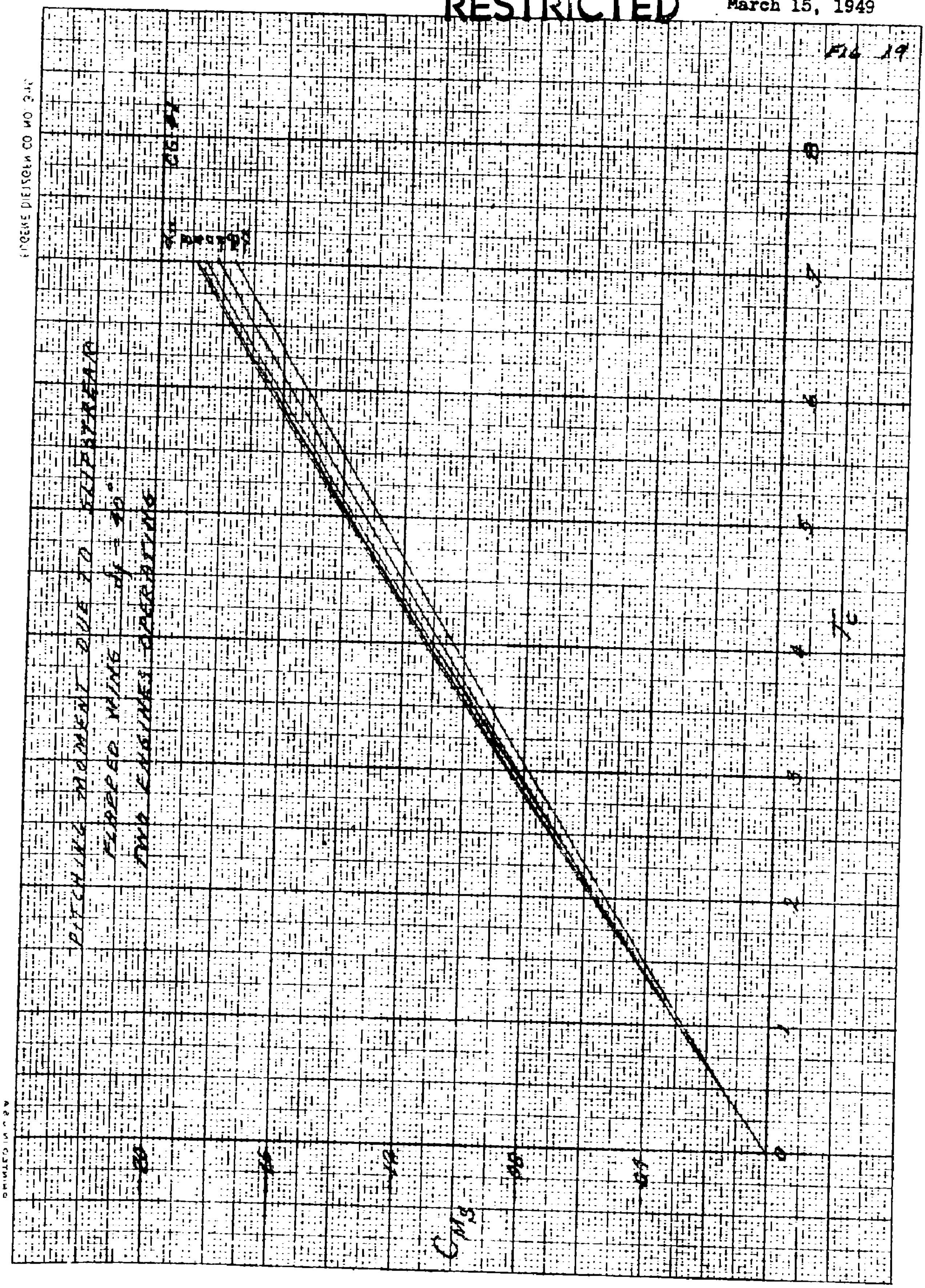


Fig 19
PITCHING MOMENT DUE TO SLIPSTREAM
FLAPPED WING δf = 40°
TWO ENGINES OPERATING
Tc
0
1
2
3
4
5
6
7
8
Cms

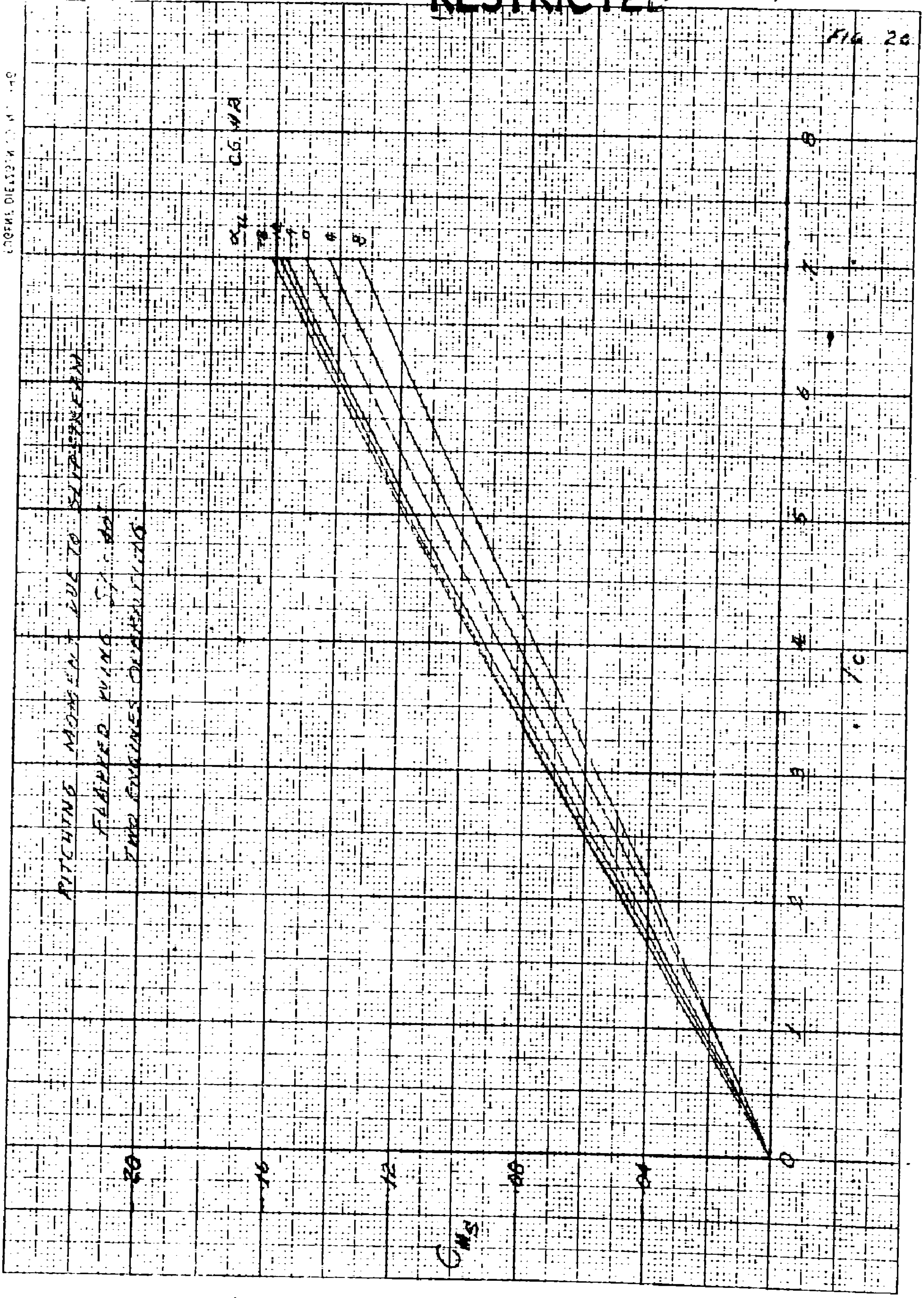


FIG. 2c
PITCHING MOMENT DUE TO SLIPSTREAM
FLAPPED WING
TWO ENGINES OPERATING
CG WA
α_w
T_c
C_{M_S}
.20
.16
.12
.08
.04
0
1
2
3
4
5
6
7
8

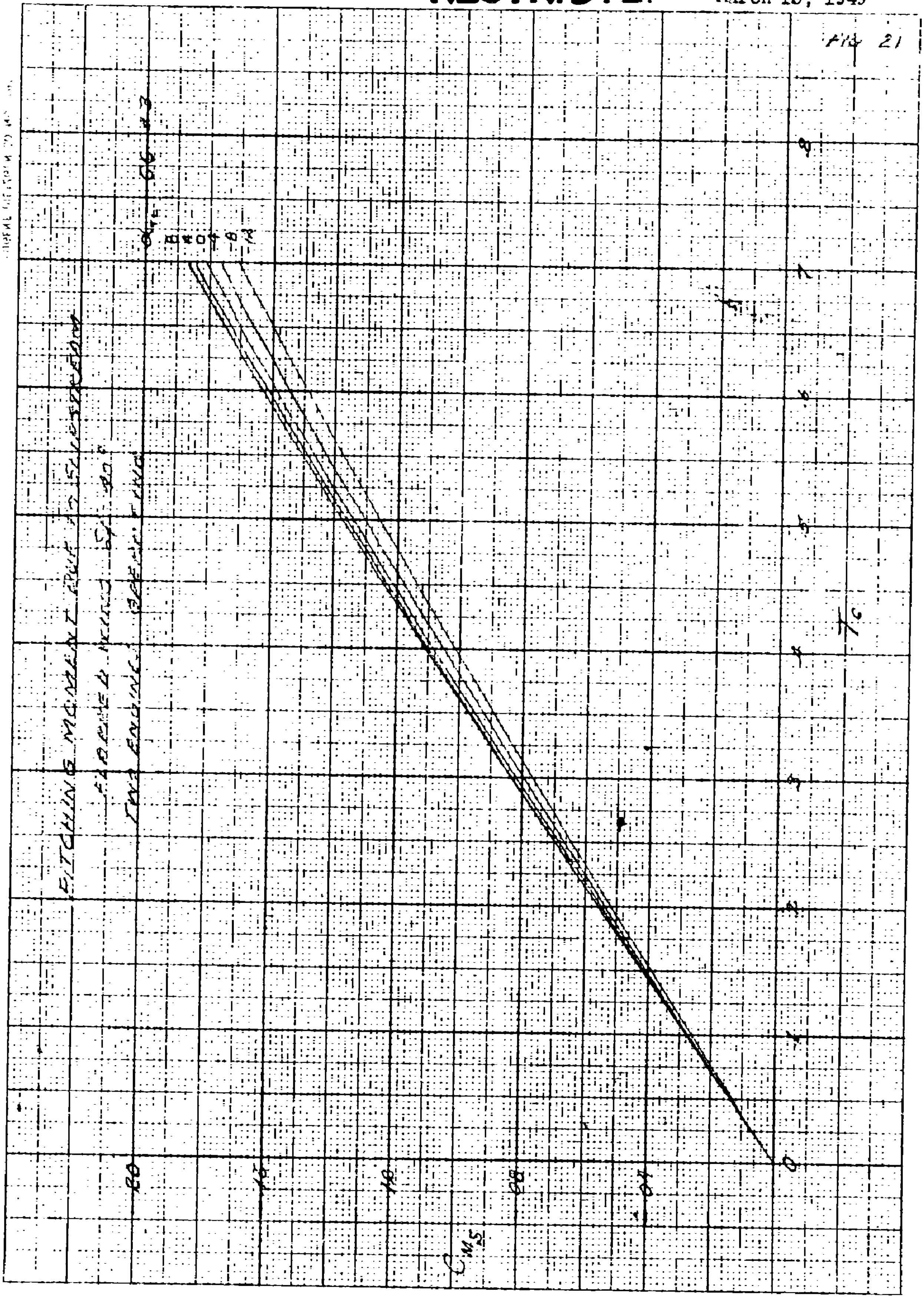


FIG 21
0
.1
.2
.3
.4
.5
.6
.7
.8
T'c
.04
.08
.12
.16
.20
C_{M_S}

FIG 22

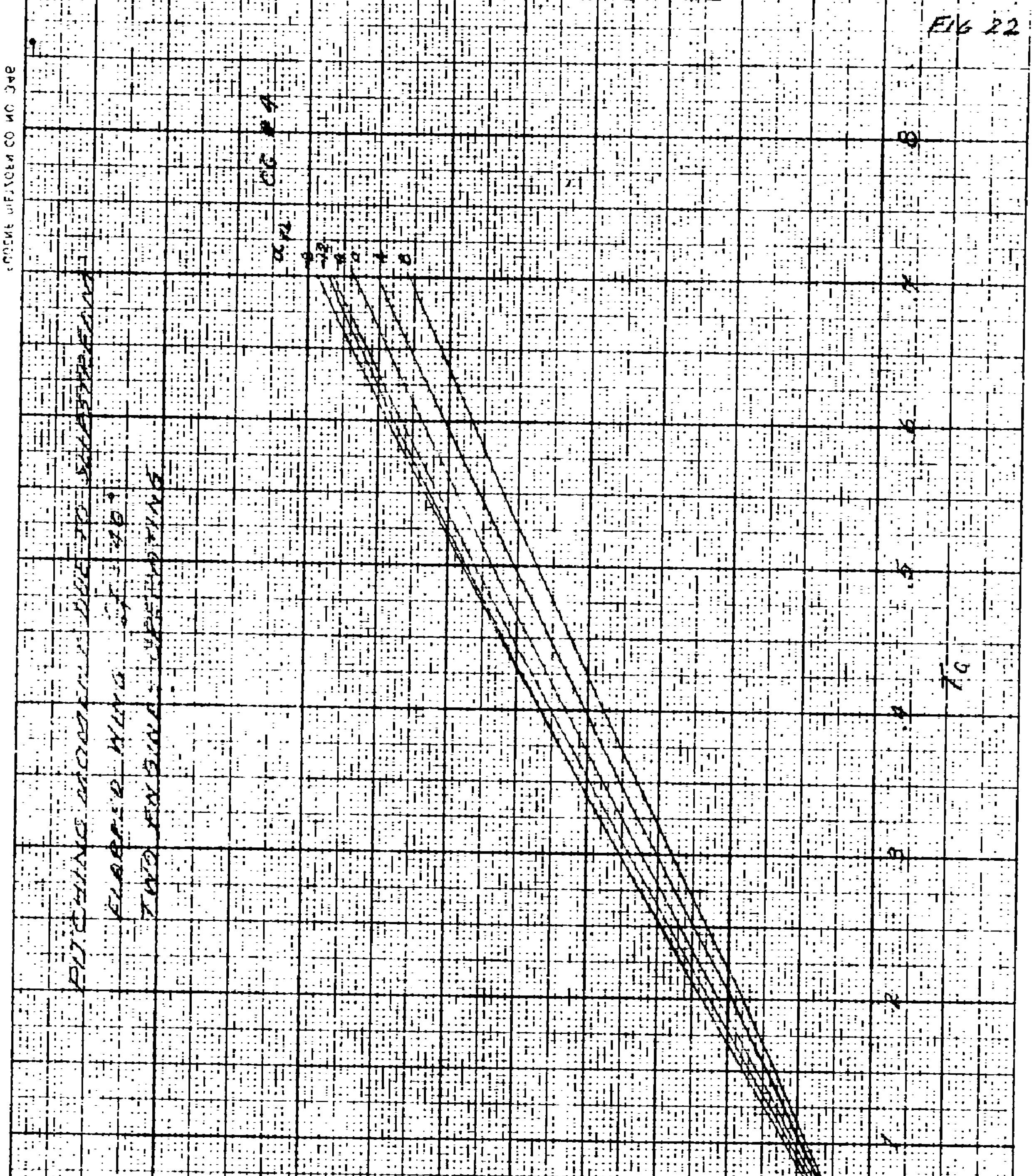
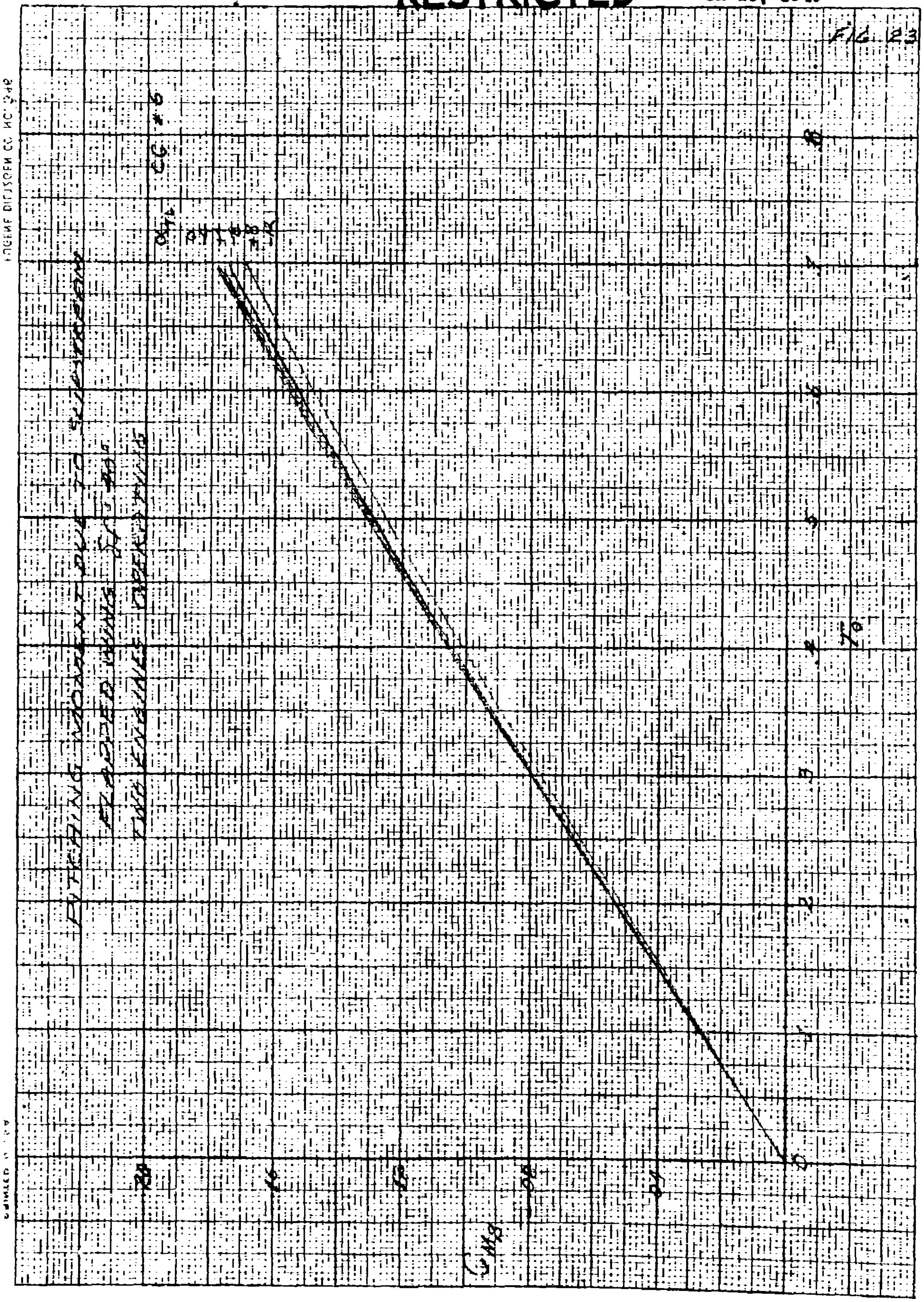

PITCHING MOMENT DUE TO SLIPSTREAM
FLAPPED WING
TWO ENGINES OPERATING
CG #4
α_{FL}
0
1
2
3
4
5
6
7
8
T_C
.04
.08
.12
.16
.20
C_{M_S}

FIG. 23


PITCHING MOMENT DUE TO SLIPSTREAM
FLAPPED WING δf = 30°
TWO ENGINES OPERATING
CG # 6
0
1
2
3
4
5
6
7
8
TC
.04
.08
.12
.16
.20
C_{M_S}

PART II - C

C. PITCHING MOMENTS OF TAILLESS AIRPLANE

This section covers the summation of the pitching moments of the various component parts to obtain the pitching moment of the tailless airplane.

The characteristics of the tailless airplane will be determined for all of the c.g. locations discussed under loading conditions, for both power-off and power-on, for the flaps and gear up, and for the special conditions with both flaps and gear down.

The pitching moment coefficients of the tailless airplane as shown in this section are all based on wing area, mean aerodynamic chord of the wing, and free stream dynamic pressure.

Subject: BASIC FLIGHT CRITERIA - PACK ON

PART II - C - 1a FLAPS AND GEAR UP - POWER OFF

PART II - C - 1a FLAPS AND GEAR UP - POWER OFF

1a. $C_{M_{A-T}} = C_{M_W} + C_{M_F} + C_{M_B} + C_{M_{VT}}$

CG #1

$\alpha_{T.L.}$	-12	-8	-4	0	4	8
C_{M_W}	-.0539	-.06210	-.0767	-.0771	-.1253	-.1559
C_{M_F}	-.0709	-.04630	-.0220	-.0019	.0181	.0427
$C_{M_{B+INT}}$	-.0108	-.00418	.00615	.02363	.04842	.07541
$C_{M_{VT}}$	.0005	.0003	.0003	.0003	.0003	.0003
$C_{M_{A-T}}$	-.1353	-.1126	-.0922	-.0751	-.0565	-.0555

CG 2

$\alpha_{T.L.}$	-12	-8	-4	0	4	8
C_{M_W}	-.0864	-.0617	-.0419	-.0286	-.0210	-.0177
C_{M_F}	-.0747	-.0483	-.0227	-.0019	.0189	.0447
$C_{M_{B+INT}}$	-.0123	-.0049	.0040	.0236	.0486	.0761
$C_{M_{VT}}$	.0003	.0003	.0003	.0003	.0003	.0003
$C_{M_{A-T}}$	-.1731	-.1146	-.0583	-.0066	.0468	.1034

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DATE March 15, 1949 REVISED

Subject:— BASIC FLIGHT CRITERIA - PACK ON

PART II - C - 1a FLAPS AND GEAR UP - POWER OFF

PART II - C - 1a FLAPS AND GEAR UP - POWER OFF

$$C_{M_{A-T}} = C_{M_W} + C_{M_F} + C_{M_B} + C_{M_{VT}}$$

CG 3

$\alpha_{T.L}$	-12	-8	-4	0	4	8
C_{M_W}	-.0643	-.0608	-.0700	-.0921	-.1267	-.1707
C_{M_F}	-.0692	-.0442	-.0127	.0003	.0203	.0947
$C_{M_{B+INT}}$	-.0089	-.0022	.0081	.0256	.0504	.0174
$C_{M_{VT}}$	.0005	.0005	.0005	.0005	.0005	.0005
$C_{M_{A-T}}$	-.1419	-.1067	-.0811	-.0657	-.0555	-.0481

CG 4

$\alpha_{T.L}$	-12	-8	-4	0	4	8
C_{M_W}	-.0969	-.0604	-.0351	-.0236	-.0245	-.0346
C_{M_F}	-.0730	-.0463	-.0205	.0003	.0211	.0468
$C_{M_{B+INT}}$	-.0104	-.0029	.0079	.0256	.0506	.0781
$C_{M_{VT}}$	.0005	.0005	.0005	.0005	.0005	.0005
$C_{M_{A-T}}$	-.1798	-.1091	-.0472	.0028	.0477	.0908

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REPORT NO. R107-014 | FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION | PAGES | PAGE 101

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

Subject:— BASIC FLIGHT CRITERIA - PACK ON

DATE March 15, 1949

REVISED

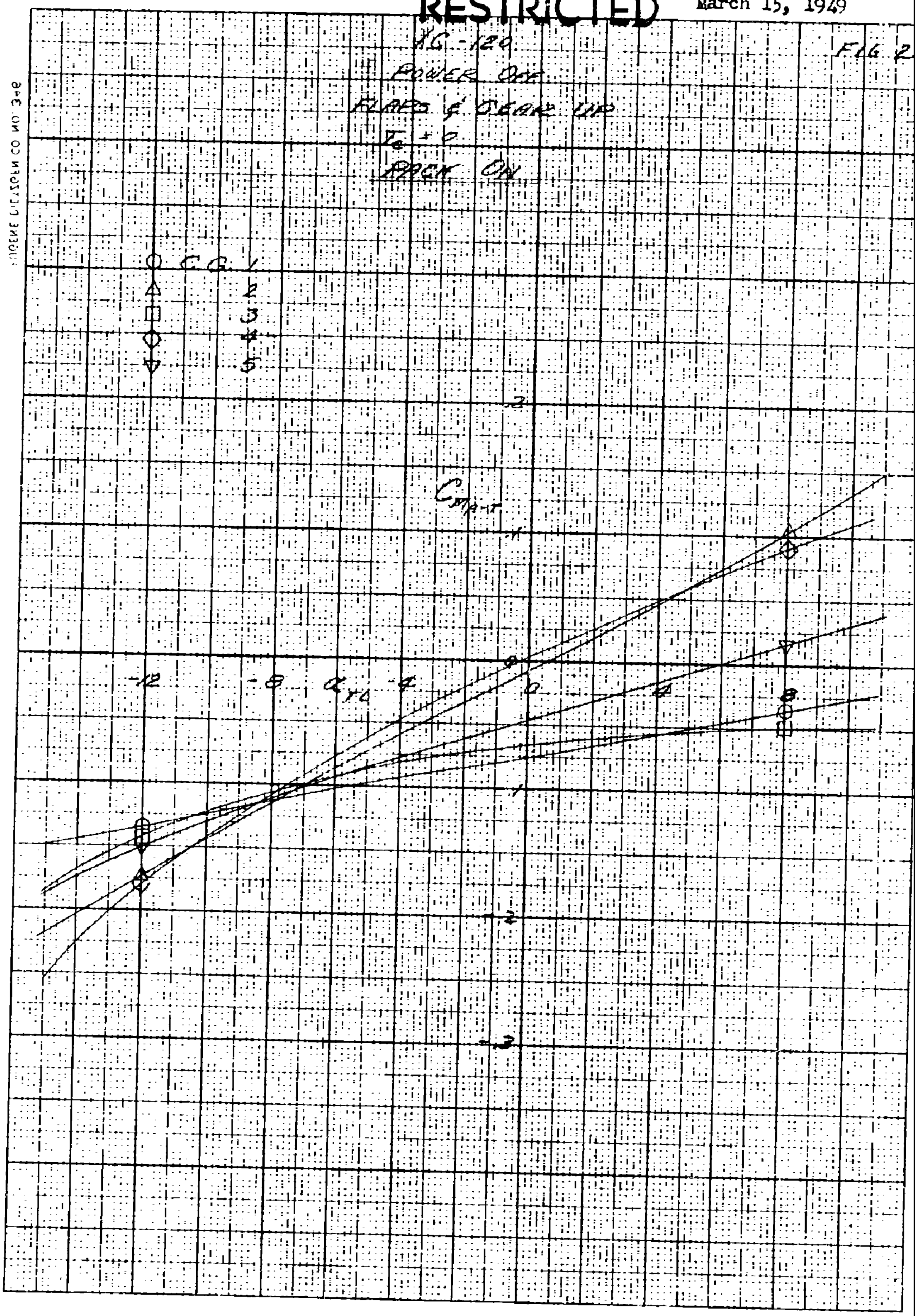
PART II - C - 1a FLAPS AND GEAR UP - POWER OFF

PART II - C - 1a FLAPS AND GEAR UP - POWER OFF

$$C_{M_{A-T}} = C_{M_W} + C_{M_F} + C_{M_B} + C_{M_{VT}}$$

CG 5

$\alpha_{T.L.}$	-12	-8	-4	0	4	8
C_{M_W}	-.0681	-.0617	-.0623	-.0701	-.0950	-.1051
C_{M_F}	-.0713	-.0458	-.0209	-.0006	.0178	.0447
$C_{M_B+INT.}$	-.0110	-.0041	.0064	.0240	.0488	.0760
$C_{M_{VT}}$	.0004	.0004	.0004	.0004	.0004	.0004
$C_{M_{A-T}}$	-.1500	-.1112	-.0764	-.0463	-.0160	.0160


XC-120
POWER OFF
FLAPS & GEAR UP
δ_e + 0
RACK ON
C.G. 1
2
3
4
5
$C_{m_{A-T}}$
.3
.1
0
-.1
-.2
-.3
-12
-8
α_{TL}
-4
0
4
8
FIG 2

PART II - C -1b. FLAPS AND GEAR UP - MILITARY POWER

PART II - C -1b FLAPS AND GEAR UP - MILITARY POWER

$$C_{M_{A-T}} = C_{M_W} + C_{M_F} + C_{M_B} + C_{M_{VT}} + C_{M_P} + C_{M_S}$$

$T_c = .639$
$V\sigma^{1/2} = 100$ mph.

	-12	[illegible]	-4	[illegible]	4	[illegible]
$C_{M_{A-T}}$ Power off	-.1353	.1126	-.0[illegible]	-.0151	-.05[illegible]	[illegible]
C_{M_P}	.0016	.0117	.0167	.0214	.0261	.0[illegible]
C_{M_S}	.0008	.0011	.0015	.0023	.00[illegible]	.0044
$C_{M_{A-T}}$ Power On	-.1264	-.0946	-.0740	-.0514	-.0212	-.0002

C.G. Loc #2

$\alpha_{W.L.}$	-12	-8	-4	0	4	8
$C_{M_{A-T}}$ Power off	-.1131	-.1146	-.0583	-.0066	.0468	.1034
C_{M_P}	.0062	.0110	.0163	.0214	.0265	.0318
C_{M_S}	.0003	.0048	.0094	.0143	.0196	.0254
$C_{M_{A-T}}$ Power On	-.1666	-.0988	-.0326	.0291	.0929	.1606

Subject: BASIC FLIGHT CRITERIA - PACK

PART II - C -1b. FLAPS AND GEAR UP - MILITARY POWER

PART II - C - 1b FLAPS AND GEAR UP - MILITARY POWER

Tc = .639
$V\sigma^{1/2}$ = 100 mph.

CG Loc #3

$\alpha_{T.L}$	-12	-8	-4	0	4	8
$C_{M_{A-T}}$ Power off	-.1419	-.1067	-.0811	-.0657	-.0555	-.0481
C_{m_p}	-.0661	-.0618	-.0569	-.0523	-.0476	-.0428
C_{m_s}	.0027	.0023	.0024	.0033	.0049	.0072
$C_{M_{A-T}}$ Power On	-.2053	-.1662	-.1356	-.1147	-.0982	-.0837

CG Loc #4

$\alpha_{T.L}$	-12	-8	-4	0	4	8
$C_{M_{A-T}}$ Power off	-.1798	-.1091	-.0472	.0028	.0477	.0908
C_{m_p}	-.0674	-.0627	-.0574	-.0523	-.0471	-.0419
C_{m_s}	.0022	.0067	.0102	.0154	.0242	.0282
$C_{M_{A-T}}$ Power On	-.2450	-.1651	-.0944	-.0341	.0219	.0771

PART II - C -1b FLAPS AND GEAR UP - MILITARY POWER

PART II - C -1b

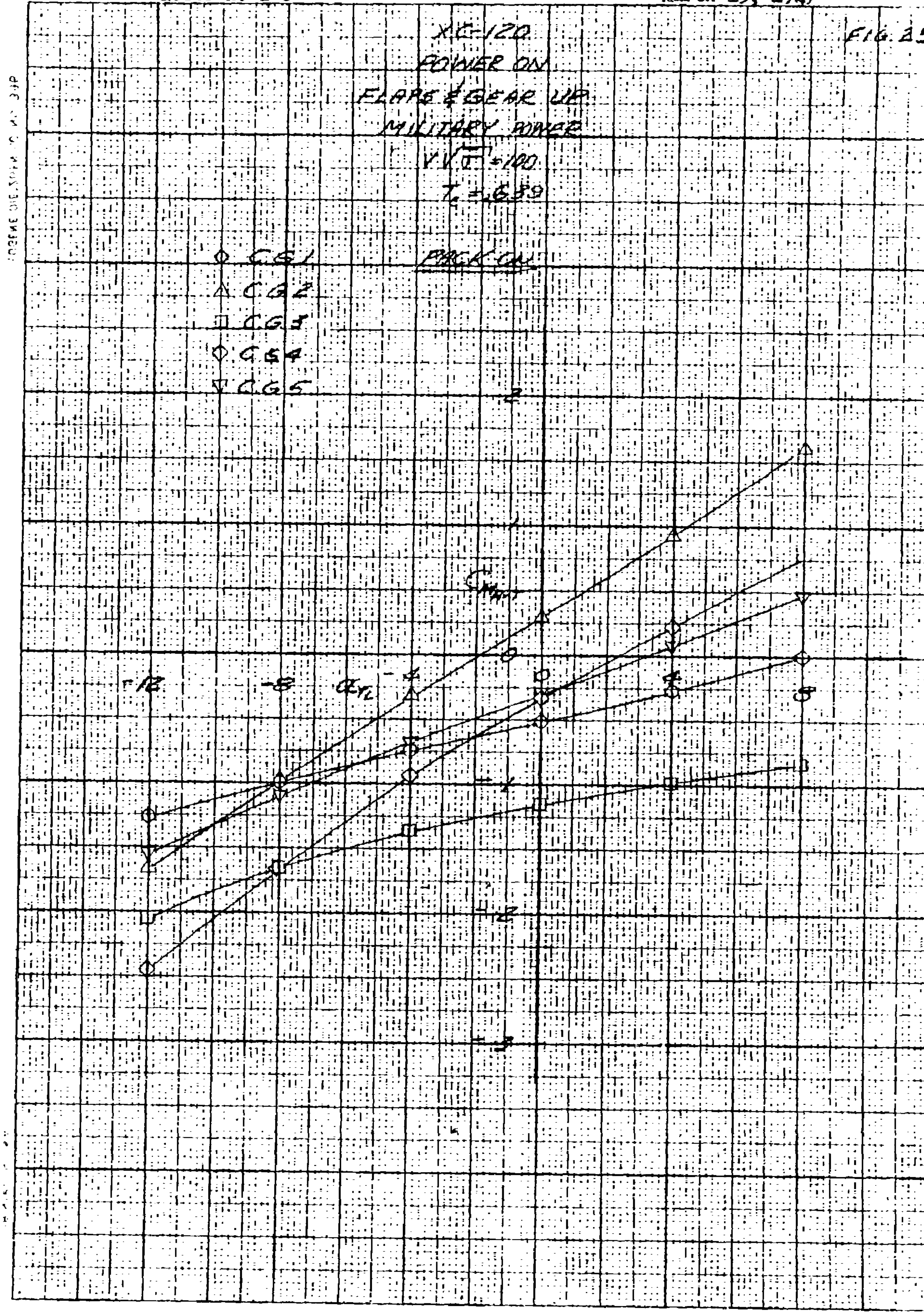
FLAPS AND GEAR UP - MILITARY POWER

T_c = .639

$V\sigma^{1/2}$ = 100 mph.

CG Loc #5

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{0T}}$	-.1500	-.1112	-.0764	-.0463	-.0160	.0160
C_{M_P}	-.0060	-.0015	.0035	.0089	.0132	.0182
C_{M_S}	.0009	.0027	.0047	.0070	.0097	.0129
$C_{M_{AT}}$ Power on	-.1551	-.1100	-.0692	-.0309	.0067	.0471


XC-120
POWER ON
FLAPS & GEAR UP
MILITARY POWER
V√σ = 100
T_c = .639
PACK ON
◇ C.G.1
△ C.G.2
□ C.G.3
◇ C.G.4
▽ C.G.5
FIG. 25
.2
.1
-.1
-.2
-.3
-12
-8
-4
0
4
8

RESTRICTED

REPORT NO. R107-014 | FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION | PAGE 107

MODEL XC-120 | PREPARED BY | CHECKED BY | APPROVED BY | DATE March 15, 1949 | REVISED

Subject: BASIC FLIGHT CRITERIA - PACK ON

PART II - C - 1b

FLAPS AND GEAR UP - MILITARY POWER

PART II - C - 1b FLAPS AND GEAR UP - MILITARY POWER

$T_c = .198$

$V\sigma^{1/2} = 160$ mph.

CG #1

$\alpha_{T.L}$	12°	-8°	-4°	0	4°	8°
Cm_{A-T} Power off	-.1353	-.1126	-.0922	-.0751	-.0565	-.0355
Cm_p	-.0057	-.0019	.0023	.0067	.0110	.0152
Cm_s	.0002	.0003	.0005	.0009	.0014	.0021
Cm_{A-T} Power on	-.1410	-.1142	-.0894	-.0675	-.0441	-.0182

CG #2

$\alpha_{T.L}$	-12°	-8°	-4°	0	4°	8°
Cm_{A-T} Power off	.1731	-.1146	-.0583	-.0066	.0468	.1034
Cm_p	-.0071	-.0027	.0019	.0067	.0114	.0161
Cm_s	-.0007	.0012	.0033	.0056	.0081	.0108
Cm_{A-T} Power on	-.1809	-.1161	-.0531	.0057	.0663	.1305

PART II - C - 1b. FLAPS AND GEAR UP - MILITARY POWER

T_c = .198

$V\sigma^{1/2}$ = 160 mph.

CG # 3

$\alpha_{T.L.}$	-12°	-8°	-4°	0	4°	8°
$C_{M_{A-T}}$ Power off	-.1419	-.1067	-.0811	-.0657	-.0555	-.0481
C_{mp}	-.0289	-.0248	-.0206	-.0163	-.0119	-.0077
C_{ms}	.0010	[illegible]	.0007	.0012	.0022	.0038
$C_{M_{A-T}}$ Power On	-.1698	-.1300	-.1010	-.0808	.0652	-.0520

CG # 4

$\alpha_{T.L.}$	-12°	-8°	-4°	0	4°	8°
$C_{M_{A-T}}$ Power off	-.1798	-.1091	-.0472	.0028	.0477	.0908
C_{mp}	-.0301	-.0257	-.0210	-.0163	-.0115	-.0064
C_{ms}	0	.0016	.0035	.0060	.0090	.0135
$C_{M_{A-T}}$ Power On	-.2044	-.1332	-.0647	-.0075	.0452	.0464

PART II - C -1.b. FLAPS AND GEAR UP - MILITARY POWER

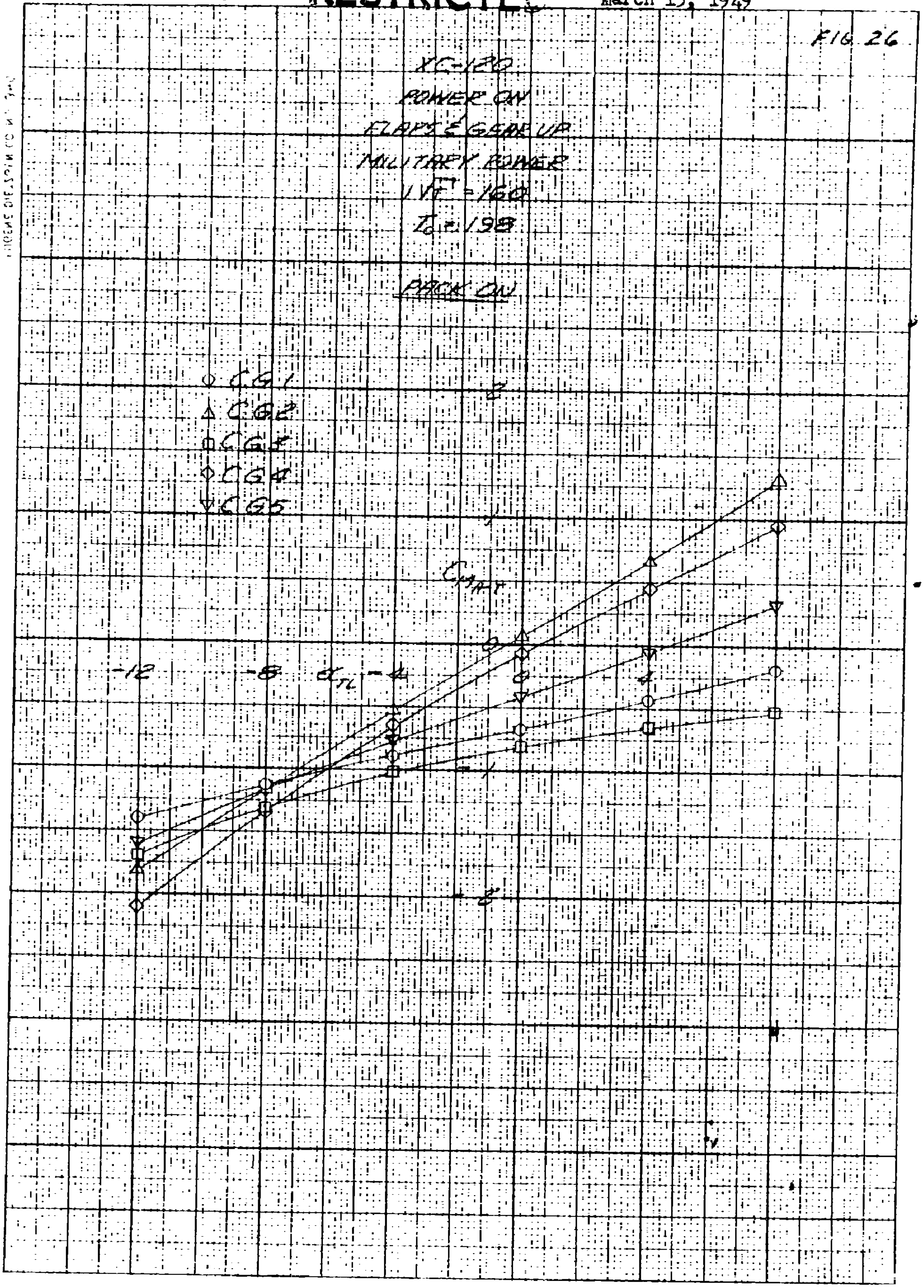
T_c = .198

$V\sigma^{\frac{1}{2}}$ = 160 mph.

CG $\neq$ 5

$\alpha_{T.L}$	-12°	-8°	-4°	0	4°	8°
$C_{M_{A-T}}$ POWER OFF	-.1500	-.1112	-.0784	-.0465	-.0160	.0160
C_{mp}	-.0105	-.0063	-.0019	.0026	.0071	.0115
C_{ms}	0	.0007	.0016	.0027	.0041	.0057
$C_{M_{A-T}}$ POWER ON.	-.1605	-.1160	-.0767	-.0410	-.0048	.0332

FIG 26


XC-120
POWER ON
FLAPS & GEAR UP
MILITARY POWER
IVF = 160
T_c = 198
PACK ON
CG 1
CG 2
CG 3
CG 4
CG 5
$C_{M_{AT}}$
α_{TL}
-12
-8
-4
0
4
.2
.1
-.1
-.2

RESTRICTED

REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION	PAGES	PAGE 111
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY
Subject:— BASIC FLIGHT CRITERIA - PACK ON			DATE March 15, 1949 REVISED

PART II - C - 1b. FLAPS AND GEAR UP - MILITARY POWER

T_c = .132

$V\sigma^{1/2}$ = 185 mph.

CG Loc. 1

α_{TL}	$-12°$	$-8°$	$-4°$	0	$4°$	$8°$
$C_{M_{a-T}}$ Power Off	-.1353	-.1126	-.0922	-.0751	-.0565	-.0355
C_{mp}	-.0024	-.0045	0	.0045	.0087	.0134
C_{M_s}	.0002	0	.0003	.0006	.0010	.0016
$C_{M_{a-T}}$ Power On	-.1435	-.1121	-.0919	-.0700	-.0466	[illegible]

CG Loc. 2

α_{TL}	$-12°$	$-8°$	$-4°$	0	$4°$	$8°$
$C_{M_{a-T}}$ Power Off	-.1731	-.1146	-.0583	-.0066	.0463	.1034
C_{mp}	-.0097	-.0059	-.0005	.0045	.0094	.0143
C_{M_s}	-.0006	.0008	.0023	.0039	.0058	.0079
$C_{M_{a-T}}$ Power On	-.1834	-.1192	-.0565	.0018	.0620	.1256

PART II - C -1b. FLAPS AND GEAR UP - MILITARY POWER

PART II - C - 1b.

FLAPS AND GEAR UP - MILITARY POWER

$T_c = .132$

$V\sigma^{1/2} = 185$ mph.

C.G. Location #3

$\alpha_{T.L.}$	-12°	-8°	-4°	0	4°	8°
$C_{m_{A-T}}$ Power off	-.1411	-.1067	-.0811	-.0657	-.0555	-.0481
C_{mp}	-.0238	-.0193	-.0153	-.0139	-.0064	-.0070
C_{ms}	.0006	.0004	.0005	.0008	.0017	.0033
$C_{m_{A-T}}$ Power On	-.1643	-.1261	-.0959	-.0258	-.0602	.0408

C.G. Location #4

$\alpha_{T.L.}$	-12°	-8°	-4°	0	4°	8°
$C_{m_{A-T}}$ Power off	-.1798	-.1091	-.0472	.0028	.0477	.0908
C_{mp}	-.0250	-.0207	-.0158	-.0109	-.0060	-.0011
C_{ms}	0	.0010	.0024	.0041	.0065	.0073
$C_{m_{A-T}}$	-.2048	-.1288	-.0606	-.0040	.0482	.0990

PART II -- C - 1b. FLAPS AND GEAR UP - MILITARY POWER

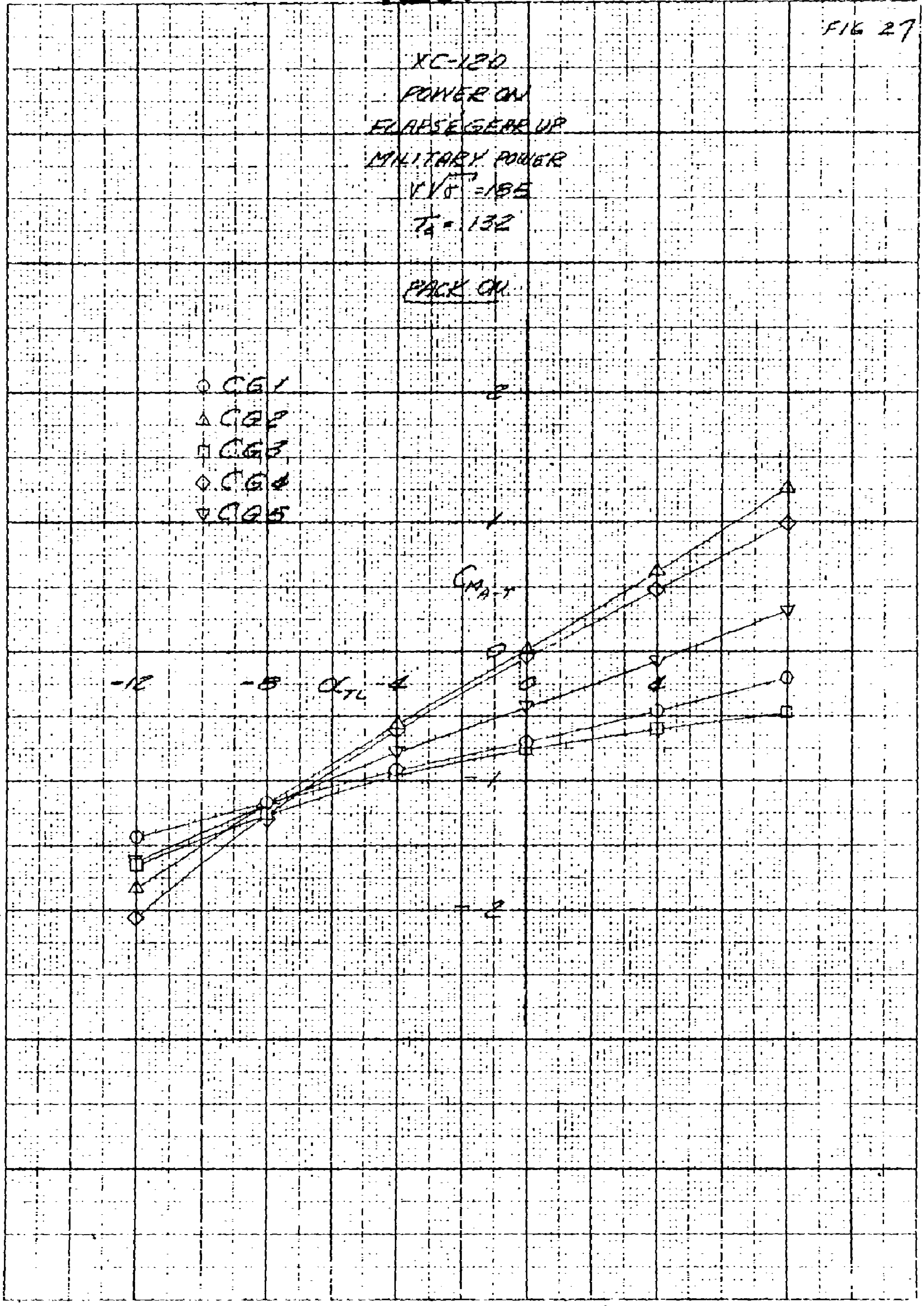
T_c = .132

$V_{0}^{1/2}$ = 185 mph.

CG LOCATION #5

$\alpha_{T.L}$	-12°	-8°	-4°	0	4°	8°
Cm_{A-T} POWER OFF	-.1500	-.1112	-.0769	-.0463	-.0160	.0160
Cm_p	-.0116	-.0075	-.0024	.0017	.0064	.0110
Cm_s	0	.0005	.0011	.0019	.0029	.0042
Cm_{A-T} POWER ON	-.1616	-.1182	-.0782	-.0427	-.0067	.0312

FIG 27


XC-120
POWER ON
FLAPS & GEAR UP
MILITARY POWER
$V\sqrt{\sigma}$ =185
T_c = .132
PACK ON
CG 1
CG 2
CG 3
CG 4
CG 5
.2
.1
$C_{M_{A-T}}$
-12
-8
α_{TL}
-4
4
8
-.1
-.2

PART II - C - 1b. FLAPS AND GEAR UP - MILITARY POWER

PART II - C - 1b'
FLAPS AND GEAR UP - MILITARY POWER

$T_c = .056$
$V\sigma^{1/2} = 250$ mph.

CG LOCATION #1

$\alpha_{T.L}$	-12°	-8°	-4°	0	4°	8°
$C_{m_{A-T}}$ Power off	-.1353	-.1126	-.0922	-.0751	-.0565	-.0395
C_{m_p}	-.0124	-.0078	-.0029	.0019	.0067	.0116
C_{m_s}	.0001	.0001	.0001	.0003	.0006	[illegible]
$C_{m_{A-T}}$ Power on	-.1476	-.1203	-.0950	-.0729	-.0492	[illegible]

CG LOCATION #2

$\alpha_{T.L}$	-12°	[illegible]	-4°	0	4°	5
$C_{m_{A-T}}$ Power off	-.1731	-.1146	-.0583	-.0066	.0466	.1034
C_{m_p}	-.0138	-.0085	-.0053	.0019	.0072	.0126
C_{m_s}	-.0005	.0003	.0009	.0017	.0027	.0039
$C_{m_{A-T}}$ Power on	-.1372	-.1231	.0617	-.0020	.0567	.1197

RESTRICTED

Subject: BASIC FLIGHT CRITERIA - PACK ON

PART II - C - 1b. FLAPS AND GEAR UP - MILITARY POWER

PART II - C - 1b. FLAPS AND GEAR UP - MILITARY POWER

$T_c = .056$

$V\sigma^{1/2} = 250$ mph.

CG #3

$\alpha_{T.L}$	-12°	-8°	-4°	0	4°	8°
$C_{M_{A-T}}$ Power Off	-.1419	-.1667	-.0811	-.0657	-.0555	-.0451
C_{m_p}	-.0170	-.0144	-.0095	-.0047	.0001	.0051
C_{m_s}	.0006	.0002	.0001	.0004	.0011	[illegible]
$C_{M_{A-T}}$ Power On	-.1603	-.1209	-.0905	-.0700	-.0543	-.0466

CG #4

$\alpha_{T.L}$	-12°	-8°	-4°	0	4°	8°
$C_{M_{A-T}}$ Power Off	-.1298	-.1091	-.0472	.0028	.0477	.0908
C_{m_p}	-.0204	-.0153	-.0099	-.0047	.0006	.0060
C_{m_s}	.0002	.0004	.0007	.0018	.0032	[illegible]
$C_{M_{A-T}}$ Power On	-.2000	-.1240	-.0562	-.0001	.0515	.1021

PART II - C - 1b. FLAPS AND GEAR UP - MILITARY POWER

PART II - C - 1b. FLAPS AND GEAR UP - MILITARY POWER

T_c = .056

= 250 mph.

CG. LOCATION 5

$\alpha_{T.L}$	-12°	-8°	-4°	0	4°	8°
$C_{M_{A-T}}$ Power Off	-.1500	-.1112	-.0764	-.0463	-.0168	+.0160
C_{m_P}	-.0141	-.0093	-.0042	.0008	.0057	.0108
C_{m_S}	-.0001	-.0002	.0004	.0006	.0015	.0024
$C_{M_{A-T}}$ Pearson.	-.1640	.1203	-.0802	-.0447	[illegible]	.0272

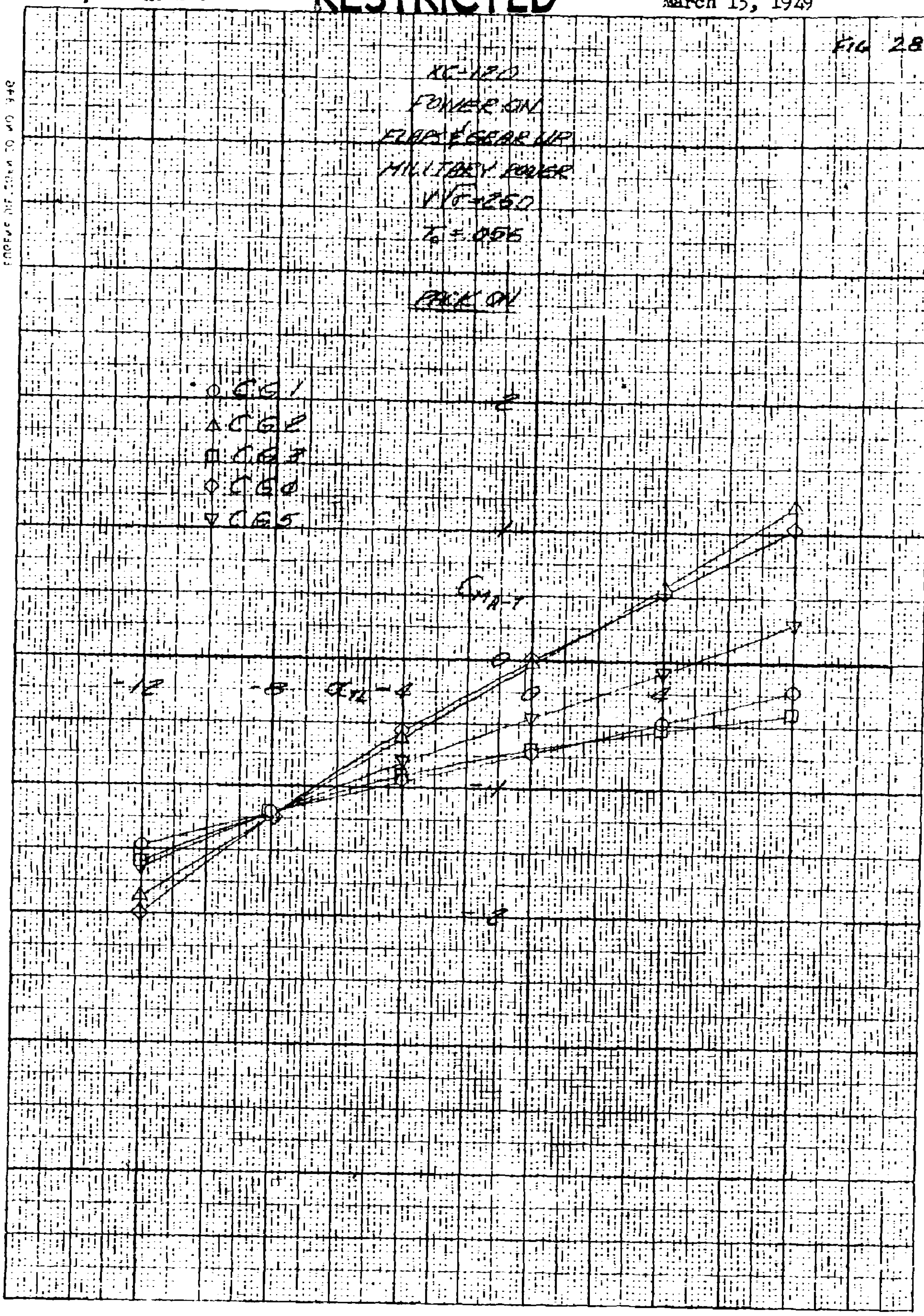


FIG 28
XC-120
POWER ON
FLAPS & GEAR UP
MILITARY POWER
V/√σ = 250
Tc = .055
PACK ON
CG 1
CG 2
CG 3
CG 4
CG 5
$C_{m_{A-T}}$
α_{TL}
-12
-8
-4
0
4
8
4
-4
-8

PART II - C -1b. FLAPS AND GEAR UP - MILITARY POWER

PART II - C - 1b. FLAPS AND GEAR UP - MILITARY POWER

$T_c = .028$
$V\sigma^{1/2} = 313$ mph.

C.G. 1

α_{TL}	-12	-8	-4	0	4	8
Power off C_{MA-T}	-.1353	-.1126	-.0922	-.0751	-.0565	-.0355
C_{MP}	-.0152	-.0098	-.0044	.0010	.0063	.0117
C_{MS}	.0001	0	0	.0002	.0004	.0009
Power on C_{MA-T}	-.1504	-.1224	-.0966	-.0739	-.0498	-.0229

C.G. 2

α_{TL}	-12	-8	-4	0	4	8
Power off C_{MA-T}	-.1731	-.1146	-.0583	-.0066	.0468	.1034
C_{MP}	-.0168	-.0109	-.0050	.0010	.0069	.0128
C_{MS}	-.0001	.0002	.0004	.0009	.0014	.0024
Power on C_{MA-T}	-.1900	-.1253	-.0629	-.0047	.0551	.1186

PART II - C - 1b. FLAPS AND GEAR UP - MILITARY POWER

PART II - C - 1b.

FLAPS AND GEAR UP - MILITARY POWER

$T_c = .028$

$V\sigma^{1/2} = 313$ mph.

C.G. 3						
α_{TL}	-12	-8	-4	0	4	8
POWER OFF $C_{M_{A\text{-}T}}$	-.1419	-.1061	-.0811	-.0657	-.0556	-.0481
C_{M_P}	-.0185	-.0131	-.0077	-.0023	.0031	.0089
C_{M_S}	.0005	.0001	0	.0002	.0009	.0021
POWER ON $C_{M_{A\text{-}T}}$	-.1599	-.1197	-.0888	-.0678	-.0515	-.0376

C.G. 4						
α_{TL}	-12	-8	-4	0	4	8
POWER OFF $C_{M_{A\text{-}T}}$	-.1798	-.1051	-.0472	.0028	.0477	.0902
C_{M_P}	-.0201	-.0142	-.0082	-.0023	.0036	.0095
C_{M_S}	.0003	.0001	.0004	.0010	.0020	.0037
POWER ON $C_{M_{A\text{-}T}}$	-.1996	-.1232	-.0550	.0015	.0533	.1140

PART II - C - 1b. FLAPS AND GEAR UP - MILITARY POWER

PART II - C - 1b.

FLAPS AND GEAR UP - MILITARY POWER

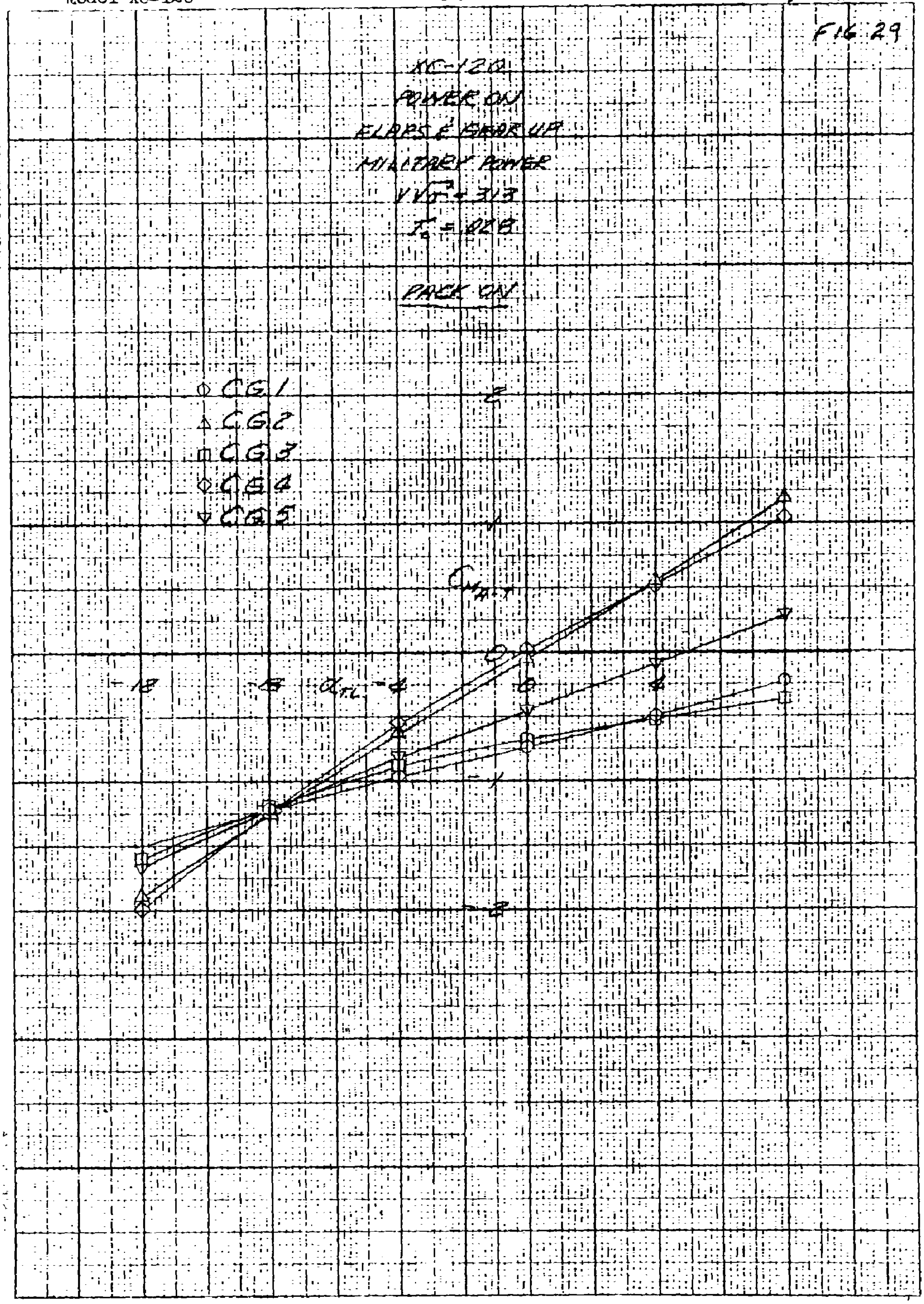
T_c = .028

$V\sigma^{1/2}$ = 313 mph.

C.G. 5

α_{TL}	-12	-8	-4	0	4	8
Power off $C_{M_{\Delta T}}$	-.1500	-.1112	-.0764	-.0463	-.0160	.0160
C_{M_P}	-.1164	-.0108	-.0552	.0004	.0060	.0116
C_{M_S}	.0001	.0001	.0002	.0004	.0009	.0016
Power on $C_{M_{\Delta T}}$	-.1663	-.1219	-.0814	-.0455	-.0091	.0292

FIG. 29


XC-120
POWER ON
FLAPS & GEAR UP
MILITARY POWER
$V\sqrt{\sigma} = 313$
$T_c = .028$
PACK ON
CG 1
CG 2
CG 3
CG 4
CG 5
$C_{M_{A\cdot T}}$
α_{TL}
-12
-8
-4
4
8
.2
.1
-.1
-.2

PART II - C - 2a

2a. FLAPS AND GEAR DOWN - POWER OFF

PART II - C -2a. FLAPS AND GEAR DOWN - POWER OFF

$$C_{M_{A-T}} = C_{M_{WF}} + C_{M_F} + C_{M_B} + C_{M_{VT}} + C\,M_{LG}.$$

C.G. 1

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{W+}}$	[illegible]	[illegible]	-.1712	-.1941	-.2229	-.2560
C_{M_F}	[illegible]	-.1463	-.0220	-.0019	.0181	.0427
$C_{M_{B+INT}}$	-.0108	.0416	+.00615	+.0236	+.0484	.0754
$C_{M_{VT}}$	.0013	.0003	.0003	.0003	.0003	.0003
$C_{M_{LG}}$	-.0354	-.0353	-.0347	-.0339	-.0347	-.0353
$C_{M_{A-T}}$	-.2609	-.2400	-.2214	-.2060	-.1908	-.1729

C.G. 2

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{WF}}$	-.1498	[illegible]	[illegible]	[illegible]	-.0866	[illegible]
C_{M_F}	-.0747	-.0463	-.0227	[illegible]	.0189	.0447
$C_{M_{B+INT}}$	[illegible]	-.0049	+.0060	+.0236	+.0486	+.0761
$C_{M_{VT}}$	[illegible]	[illegible]	.0003	[illegible]	.0003	.0003
$C_{M_{LG}}$	-.0344	-.1349	-.0342	-.0339	-.0342	-.0349
$C_{M_{A-T}}$	-.2643	-.2077	-.1538	-.1037	-.0529	.0011

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REPORT NO. R107-014 | FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION | PAGE 124

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

DATE March 15, 1949

Subject:-- BASIC FLIGHT CRITERIA - PACK ON

REVISED

PART II - C -2a. FLAPS AND GEAR DOWN - POWER OFF

PART II - C - 2a. FLAPS AND GEAR DOWN - POWER OFF

C.G. 3

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{WF}}$	-.1370	-.1881	-.1521	-.1792	-.2193	-.2667
C_{M_F}	-.0692	-.0442	-.0197	.0003	.0203	.0447
$C_{M_{B+INT}}$	-.0189	-.0022	.0061	+.0256	+.0504	.0774
$C_{M_{VT}}$	.0005	.0005	.0005	.0005	.0005	.0005
$C_{M_{LG}}$	-.0239	.0233	-.0226	-.0218	-.0226	-.0233
$C_{M_{A.T}}$	-.2395	-.2073	-.1858	-.1746	-.1707	-.1694

C.G. 4

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{WF}}$	-.1362	-.1040	-.0841	-.0769	-.0629	-.0983
C_{M_F}	-.0730	-.0463	-.0215	.0003	.0211	.0468
$C_{M_{B+INT}}$	-.0104	-.0029	+.0079	+.0256	+.0506	.0781
$C_{M_{VT}}$	.0005	.0005	.0005	.0005	.0005	.0005
$C_{M_{LG}}$	-.0226	-.0224	-.0221	-.0218	-.0221	-.0224
$C_{M_{A.T}}$	-.2417	-.1751	-.1183	-.0723	-.0328	.0047

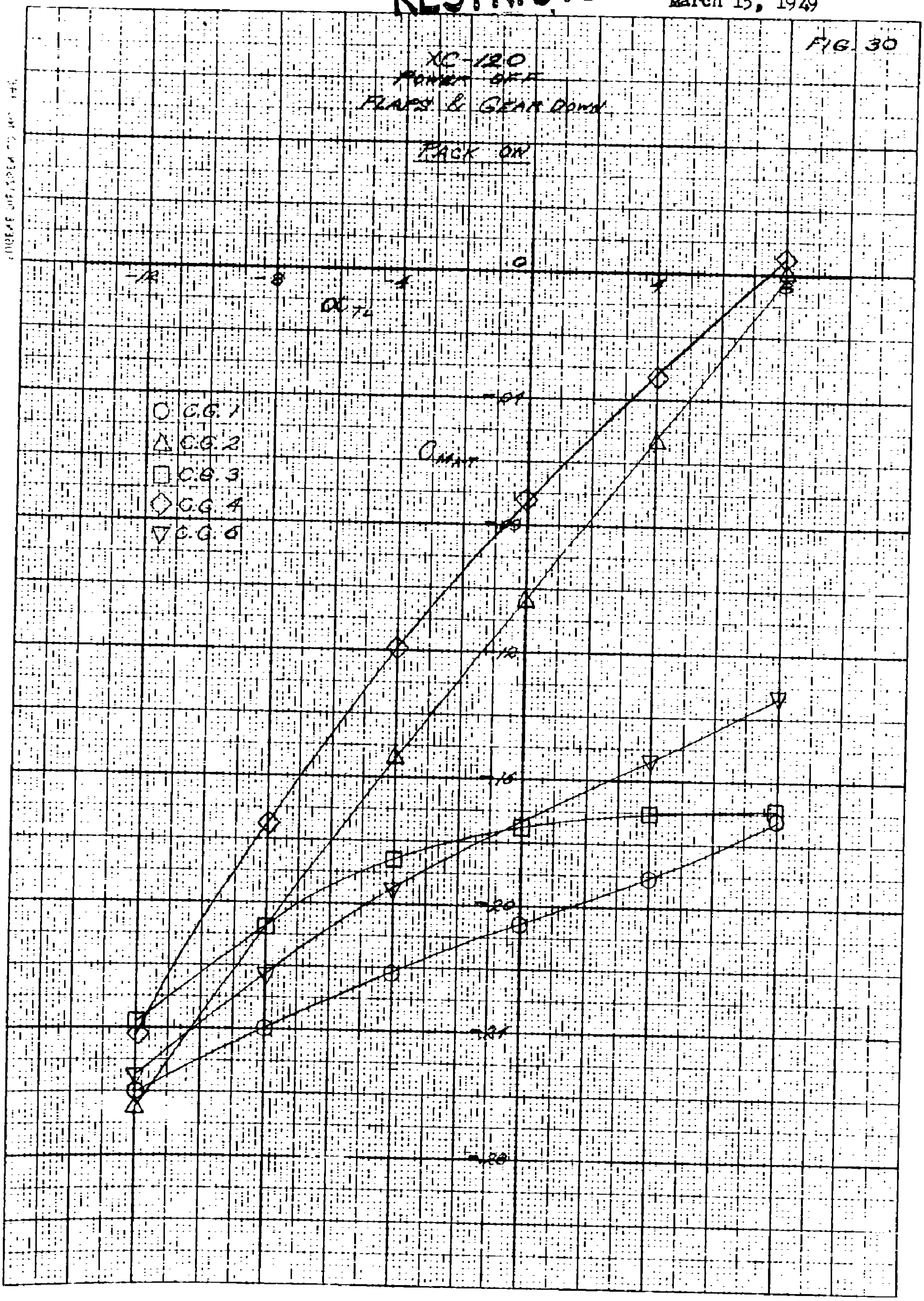
PART II - C - 2a. FLAPS AND GEAR DOWN - POWER OFF

PART II - C - 2a. FLAPS AND GEAR DOWN - POWER OFF

C.G. 6

α_{TL}	-12	-8	-4	0	4	8
$C_{m_{WF}}$	-.1419	-.1411	-.1502	-.1668	-.1918	-.2226
$C_{m_{H}}$	-.0712	-.0460	-.0214	-.0012	.0190	.0438
$C_{m_{AIENT}}$	-.0110	-.0041	+.0064	+.0290	+.0488	+.0760
$C_{m_{VT}}$	.0004	.0004	.0004	.0004	.0004	.0004
$C_{m_{LG}}$	-.0316	-.0312	-.0307	-.0300	-.0307	-.0312
$C_{m_{A.T}}$	-.2548	-.2226	-.1955	-.1736	-.1543	-.1336

FIG. 30


XC-120
POWER OFF
FLAPS & GEAR DOWN
PACK ON
-12
-8
-4
0
4
8
α_{T4}
○ C.G. 1
△ C.G. 2
□ C.G. 3
◇ C.G. 4
▽ C.G. 6
δ_{MAT}
-4
-8
-12
-16
-20
-24
-28

PART II - C - 2b. FLAPS AND GEAR DOWN - MILITARY POWER

T_c = .639 $V\sigma^{1/2}$ = 100 mph.

C.G. 1

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{A-T}}$	[illegible]	-.2400	-.2214	-.2060	-.1908	-.1729
C_{M_P}	.0076	.0119	.0167	.0214	.0261	.0309
C_{M_S}	-.1612	-.1672	-.1714	-.1734	-.1742	-.1758
$C_{M_{A-T}}$	-.4145	-.3953	-.3761	-.3580	-.3389	-.3158

C.G. 2

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{A-T}}$	-.2643	-.2074	-.1538	-.1037	-.0529	.0011
C_{M_P}	.0062	.0110	.0163	.0214	.0265	.0318
C_{M_S}	-.1472	-.1492	-.1452	[illegible]	-.1312	-.1224
$C_{M_{A-T}}$	-.4053	-.3459	-.2627	-.2209	-.1576	-.0895

PART II - C - 2b. FLAPS AND GEAR DOWN - MILITARY POWER

T_c = .639 $V\sigma^{1/2}$ = 100 mph.

C.G. 3

α_{TL}	-12	-8	-4	0	4	8
Power off C_{MAT}	-.2385	-.2073	-.1658	-.1746	-.1701	-.1694
C_{MP}	-.0661	-.0618	-.0569	-.0523	-.0476	-.0428
C_{MS}	-.1594	-.1634	-.1678	-.1706	-.1720	-.1720
Power on C_{MAT}	-.4620	-.4325	-.4105	-.3975	-.3903	-.3642

C.G. 4

α_{TL}	-12	-8	-4	0	4	8
Power off C_{MAT}	-.2417	-.1751	-.1183	-.0723	-.0326	.0047
C_{MP}	-.0674	-.0627	-.0574	-.0523	-.0471	-.0419
C_{MS}	-.1434	-.1454	-.1416	-.1358	-.1290	-.1206
Power on C_{MA-T}	-.4525	-.3832	-.3173	-.2604	-.2069	-.1662

Subject:— BASIC FLIGHT CRITERIA - PACK ON

PART II - C - 2b. FLAPS AND GEAR DOWN - MILITARY POWER

PART II-C-2b

FLAPS AND GEAR DOWN - MILITARY POWER

T_c = .639 $V\sigma^{1/2}$ = 100 mph.

C.G. 6

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{AT}}^{Power\ off}$	-.2548	-.2226	-.1955	-.1736	-.1543	-.1336
C_{M_P}	-.0165	-.0121	-.0072	-.0024	.0024	.0073
C_{M_S}	.1568	-.1620	-.1646	-.1650	-.1640	-.1620
$C_{M_{A-T}}^{Power\ on}$	-.4281	-.3967	-.3673	-.3410	-.3159	-.2663

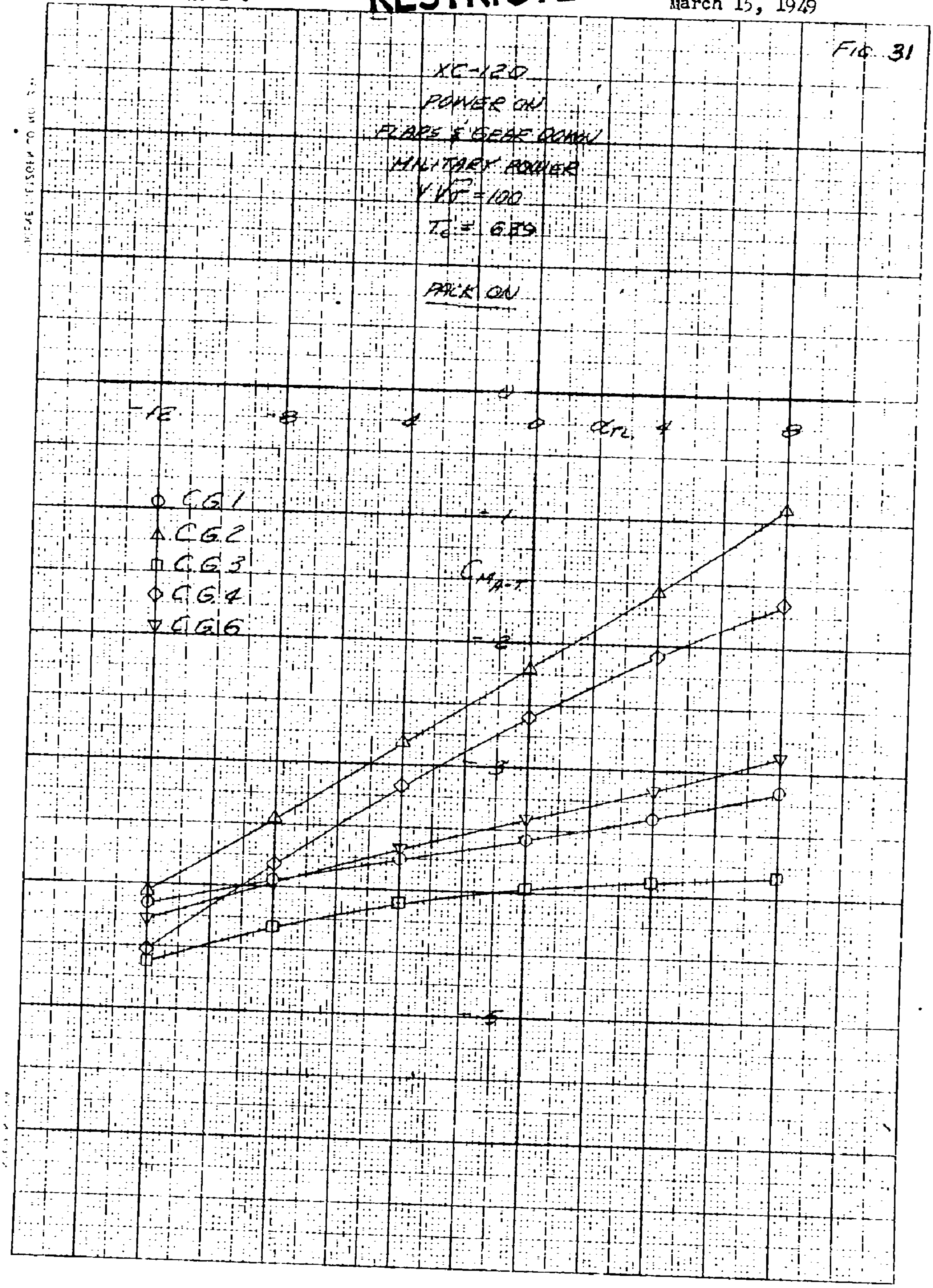

XC-120
POWER ON
FLAPS & GEAR DOWN
MILITARY POWER
V√σ = 100
PACK ON
-12
-8
-4
0
αTL
4
8
CG 1
CG 2
CG 3
CG 4
CG 6
-.1
C_{M A-T}
-.2
-.3
-.5

FIG. 31

PART II - C - 2b. FLAPS AND GEAR DOWN - MILITARY POWER

PART II - C - 2b. FLAPS AND GEAR DOWN - MILITARY POWER

T_c = .198 $V\sigma^{1/2}$ = 160 mph.

C. G. 1

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{A-T}}^{\text{POWER OFF}}$	-.2609	-.2400	-.2214	-.2060	-.1908	-.1729
C_{HP}	-.0059	-.0019	.0023	.0067	.0110	.0152
C_{HS}	-.0522	-.0540	-.0552	-.0562	-.0560	-.0566
$C_{M_{A-T}}^{\text{POWER ON}}$	-.3190	-.2959	-.2743	-.2555	-.2358	-.2143

C. G. 2

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{A-T}}^{\text{POWER OFF}}$	-.2143	-.2077	-.1538	-.1037	-.0529	.0011
C_{HP}	-.0071	-.0027	.0019	.0067	.0114	.0161
C_{HS}	-.0474	-.0476	-.0452	-.0428	-.0392	-.0364
$C_{M_{A-T}}^{\text{POWER ON}}$	-.3188	-.2580	-.1971	-.1398	-.0807	-.0192

Subject:-- BASIC FLIGHT CRITERIA - PACK ON

PART II - C-2b. FLAPS AND GEAR DOWN - MILITARY POWER

PART II - C - 2b. FLAPS AND GEAR DOWN - MILITARY POWER

$T_c = .198$ $V\sigma^{1/2} = 160$ mph.

C.G. 3

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{A-T}}$ power off	-.2385	-.2073	-.1856	-.1746	-.1707	-.1694
C_{M_P}	-.0299	-.0248	-.0206	-.0163	-.0119	-.0077
C_{M_S}	-.0512	-.0530	-.0544	-.0554	-.0554	-.0556
$C_{M_{A-T}}$ power on	-.3186	-.2851	-.2608	-.2463	-.2360	-.2327

C.G. 4

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{A-T}}$ power off	-.2417	-.1751	-.1183	-.0723	-.0328	.0047
C_{M_P}	-.0301	-.0257	-.0210	-.0163	-.0115	-.0069
C_{M_S}	-.0464	-.0466	-.0444	-.0420	-.0386	-.0354
$C_{M_{A-T}}$ power on	-.3182	-.2474	-.1837	-.1306	-.0629	-.0376

Subject: BASIC FLIGHT CRITERIA - PACK ON

PART II - C - 2b. FLAPS AND GEAR DOWN - MILITARY POWER

PART II - C - 2b. FLAPS AND GEAR DOWN - MILITARY POWER

$T_c = .198$ $V\sigma^{1/2} = 160$ mph.

C.G. 6

α_{TL}	-12	-8	-4	0	4	8
$C_{M_{A-T}}$ power off	-.2546	-.2226	-.1955	-.1736	-.1543	-.1336
C_{M_P}	-.0136	-.0095	-.0052	-.0007	.0037	.0080
C_{M_S}	-.0516	-.0522	-.0528	-.0530	-.0522	-.0516
$C_{M_{A-T}}$ power on	-.[illegible]200	-.2643	-.2535	-.2273	-.2028	-.1779

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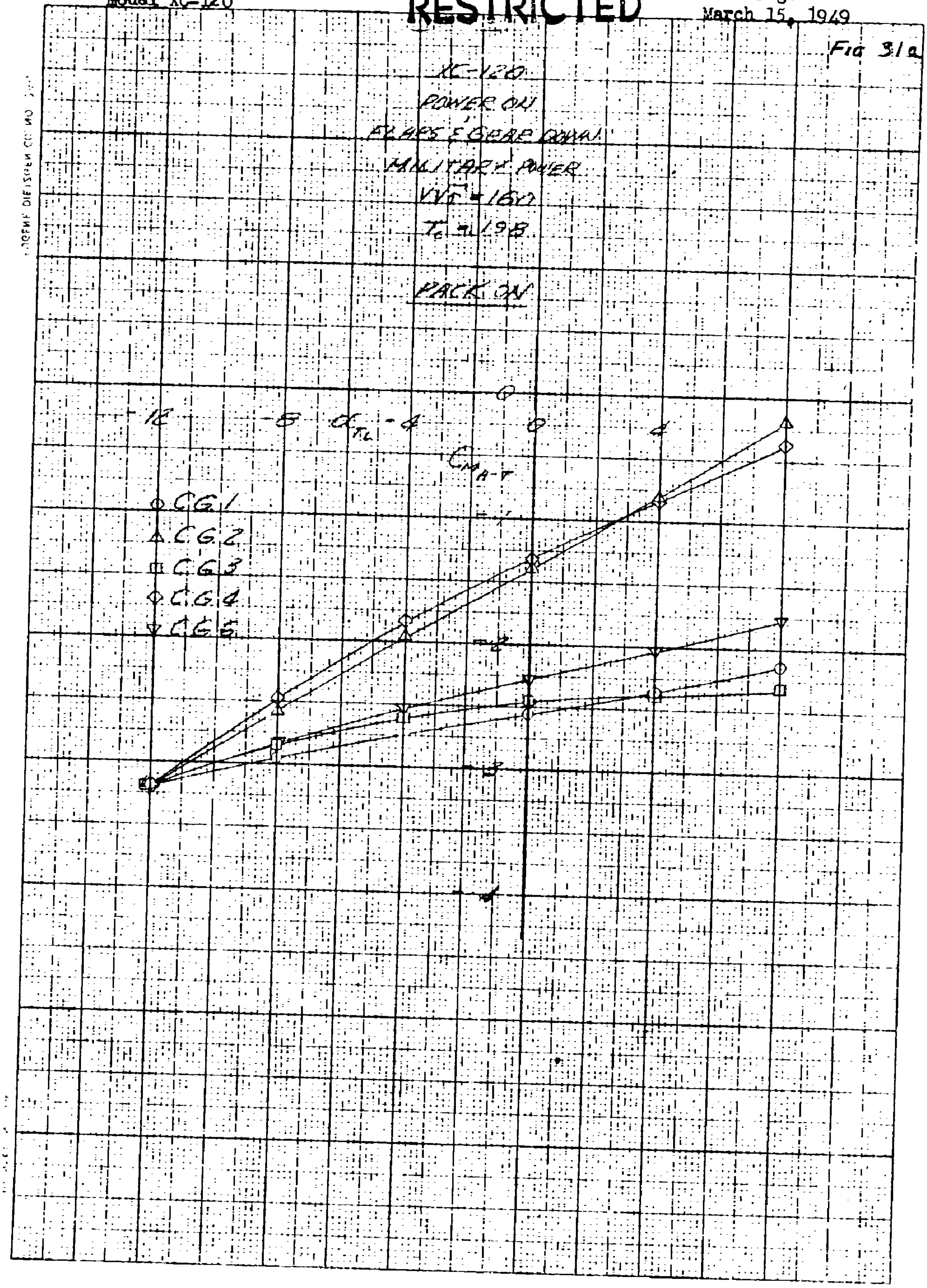
Fig 31a



PART II-D BALANCING TAIL LIFT

Since many of the required design conditions are for symmetrical flight without pitching, it is necessary to determine the characteristics of the complete airplane for this condition. For this condition $C_{M_A} = 0$. Since the only control which is variable for symmetric flight is the horizontal tail, this is the only balancing element; and in order to determine the horizontal tail load required to balance the airplane, it is necessary to know the characteristics of the complete airplane less the horizontal tail.

In order to simplify these calculations, several justifiable assumptions are made. First, that the contribution of the X_{HT} force to the total tail moment is negligible and second, that the motion of the tail center of pressure is a negligibly small fraction of the total tail length.

Then to balance the airplane

$$C_{M_A} = C_{M_{A-T}} + C_{M_{HT}} = 0$$

$$C_{M_{A-T}} = -C_{M_{HT}}$$

let $C_{Z_{HT}} = \eta_{ht} \; C_{L_{HT}}$

so that

$$-C_{M_{HT}} = \eta_{ht} \; C_{L_{HT}} \; \frac{S_{ht}}{S_w} \; \frac{l_{ht}}{MAC_w} \; \frac{q_{ht}}{q}$$

where

$\eta_{ht} = .85$ lift efficiency of the horizontal tail

$$\frac{S_{ht}}{S_w} = \frac{346.2}{1447.25} = .2392$$

PART II - D

$$K = \frac{S_{ht}}{S_W} \frac{l_{ht}}{MAC_W} \frac{q_{ht}}{q}$$

$$K = .2392 \times \frac{l_{ht}}{168.28} \times \frac{q_{ht}}{q}$$

$$K = .00142\ l_{ht} \frac{q_{ht}}{q}$$

The effective tail lift is then

$$\eta_{ht}\ C_{L_{HT}} = \frac{C_{M_{A-T}}}{K} \quad \text{based on} \quad q_{ht} \quad S_{ht}$$

l_{ht} is distance from C.G. to elevator hinge line.

PART II - D - 1a.

1a. FLAPS AND GEAR UP - POWER OFF

$\frac{q_{ht}}{q} = 1.00$ page 57 reference (1)

C.G. Location	l_{ht}	K
1	621.1	.882
2	604.2	.858
3	621.1	.882
4	604.2	.858
5	614.8	.873
6	617.4	.877

a_{ht} from figure 35 of reference (1)

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REPORT NO. R107-014 | FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION | PAGES | PAGE 138

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

Subject:— BASIC FLIGHT CRITERIA - PACK ON

DATE March 15, 1949

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PART II - D - 1 BALANCING TAIL LIFT

PART II - D - 1 BALANCING TAIL LIFT

FLAPS AND GEAR UP - POWER OFF

		C.G ①			CG ②			CG ③		
α_{TL}	α_{ht}	$C_{M_{A-T}}$	K	$\eta_{ht} C_{L_{HT}}$	$C_{M_{A-T}}$	K	$\eta_{ht} C_{L_{HT}}$	$C_{M_{A-T}}$	K	$\eta_{ht} C_{L_{HT}}$
-12	-9.40	-.1353	.882	-.153	-.1731	.858	-.202	-.1419	.882	-.161
-8	-7.05	-.1126		-.128	-.1146		-.134	-.1067		-.121
-4	-4.70	-.0922		-.105	-.0583		-.068	-.0811		-.092
0	-2.35	-.0751		-.085	-.0066		-.008	-.0657		-.074
4	0	-.0565		-.064	.0468		+.055	-.0555		-.063
8	2.35	-.0355		-.040	.1034		+.121	-.0481		-.055

C.G. ④			C.G ⑤		
$C_{M_{A-T}}$	K	$\eta_{ht} C_{L_{HT}}$	$C_{M_{A-T}}$	K	$\eta_{ht} C_{L_{HT}}$
-.1798	.858	-.210	-.1500	.873	-.172
-.1091		-.127	-.1112		-.127
-.0472		-.055	-.0764		-.088
.0028		+.003	-.0463		-.053
.0477		+.056	-.0160		-.018
.0908		+.106	.0160		+.018

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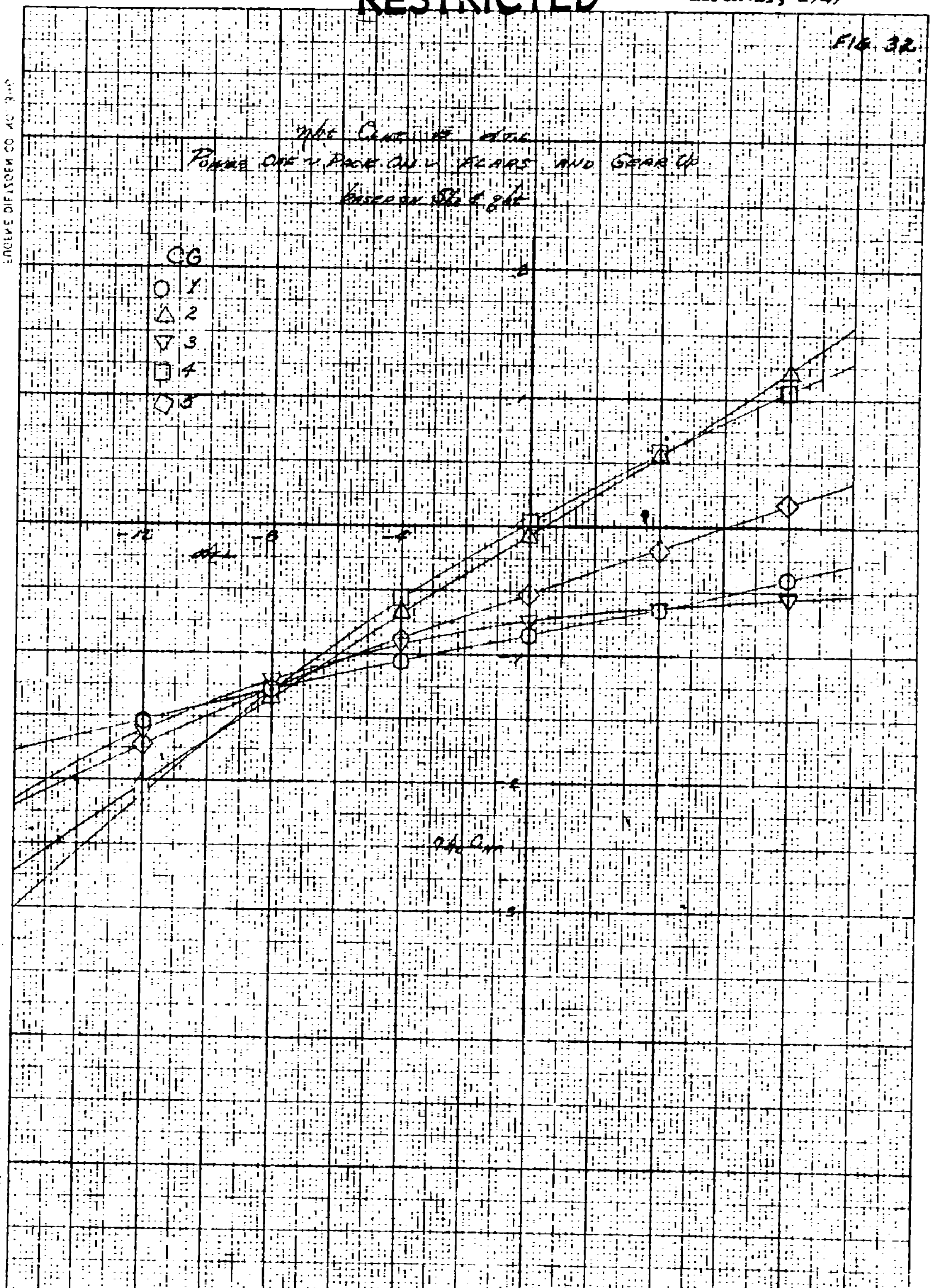
FIG. 32
CG
1
2
3
4
5
-12
-8
-4

PART II - D - 1b.

1-b. FLAPS AND GEAR UP - POWER ON

For flaps and gear up, power on $\frac{q_{ht}}{q}$ and ϵ will vary with power. Then K power on = K power off $\left(\frac{q_{ht}}{q}\right)$ a_{ht} is obtained from figure 63 reference (1).

$\frac{q_{ht}}{q}$ is obtained from figure 62 reference (1).

Curve of balancing tail lift coefficients are shown following each tabulation.

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REPORT NO. R107-014 | FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION | PAGES | PAGE 141

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

Subject: BASIC FLIGHT CRITERIA - PACK ON

DATE March 15, 1949

REVISED

PART II - D - 1-b. FLAPS AND GEAR UP - POWER ON

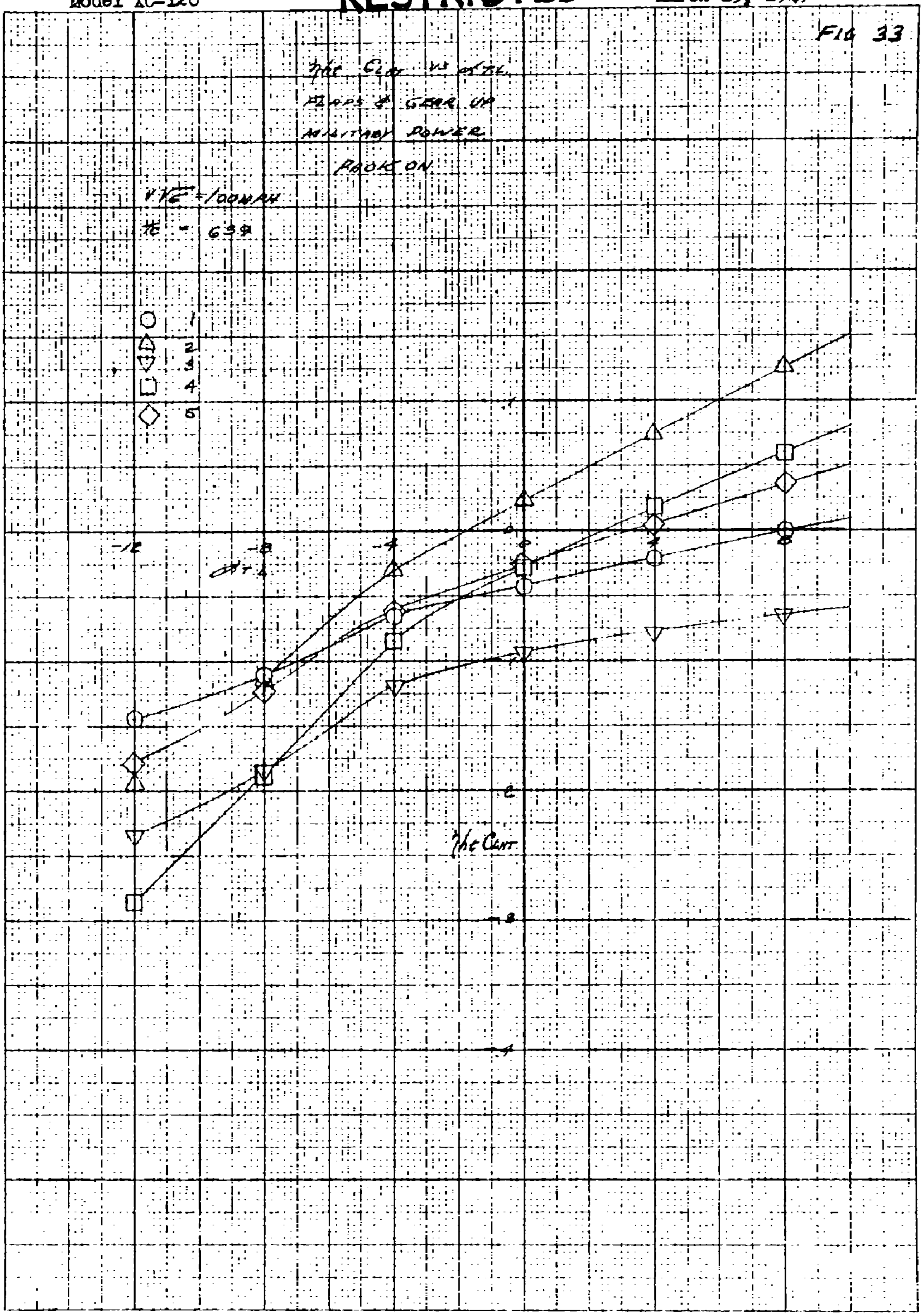
PART II - D - 1b. FLAPS AND GEAR UP - POWER ON

$V\sigma^{\frac{1}{2}} = 100 - T_c = .639$

			C.G. 1			2		
α_{TL}	α_{ht}	$\frac{q_{ht}}{q}$	$C_{M_{A-T}}$	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{L_{HT}}$	$C_{M_{A-T}}$	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{L_{HT}}$
-12	-9.40	1.000	-.1269	.882	-.144	-.1666	.858	-.194
-8	-7.05	1.029	-.0996	.908	-.110	-.0988	.883	-.112
-4	-4.10	1.300	-.0740	1.147	-.065	-.0326	1.115	-.029
0	-2.57	1.404	-.0514	1.233	-.042	.0291	1.205	+.024
4	-1.26	1.452	-.0272	1.281	-.021	.0929	1.246	+.075
8	-.10	1.490	-.0002	1.314	0	.1606	1.278	+.126

3			4			5		
$C_{M_{A-T}}$	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{L_{HT}}$	$C_{M_{A-T}}$	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{L_{HT}}$	$C_{M_{A-T}}$	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{L_{HT}}$
-.2053	.882	-.233	-.2450	.858	-.286	-.1551	.873	-.178
-.1662	.908	-.183	-.1651	.883	-.187	-.1100	.898	-.122
-.1356	1.147	-.118	-.0944	1.115	-.085	-.0682	1.135	-.060
-.1147	1.238	-.093	-.0341	1.205	-.028	-.0309	1.226	-.025
-.0982	1.281	-.072	.0219	1.246	+.018	.0069	1.265	+.005
-.0837	1.314	-.064	.0771	1.278	+.060	.0471	1.301	+.036

FIG 33



PART II - D - 1b. FLAPS AND GEAR UP - POWER ON

PART II - D - 1b. FLAPS AND GEAR UP - POWER ON

$V\sigma^{\frac{1}{2}} = 160 \quad T_c = .198$

			C.G ①			②		
$\alpha_{T.L.}$	α_{ht}	$\frac{q_{ht}}{q}$	$C_{M_{A-T}}$	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{L_{HT}}$	$C_{M_{A-T}}$	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{L_{HT}}$
-12	-9.40	1.000	-.1410	.882	-.160	-.1809	.858	-.211
-8	-7.05	1.000	-.1142	.882	-.129	-.1161	.858	-.135
-4	-4.70	1.058	-.0894	.933	-.096	-.0531	.908	-.058
0	-2.48	1.148	-.0675	1.013	-.067	.0057	.985	+.006
4	-.75	1.178	-.0441	1.039	-.042	.0663	1.011	+.066
8	.88	1.194	-.0182	1.053	-.017	.1303	1.024	+.127

③			④			⑤		
$C_{M_{A-T}}$	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{L_{HT}}$	$C_{M_{A-T}}$	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{L_{HT}}$	$C_{M_{A-T}}$	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{L_{HT}}$
-.1698	.882	-.193	-.2099	.858	-.245	-.1605	.873	-.184
-.1308	.882	-.148	-.1332	.858	-.155	-.1168	.873	-.134
-.1010	.933	-.108	-.0647	.908	-.071	-.0767	.924	-.083
-.0808	1.013	-.080	-.0075	.985	-.008	-.0410	1.002	-.041
-.0652	1.039	-.063	-.0452	1.011	+.045	-.0048	1.028	-.005
-.0520	1.053	-.049	.0964	1.024	+.094	.0332	1.042	+.032

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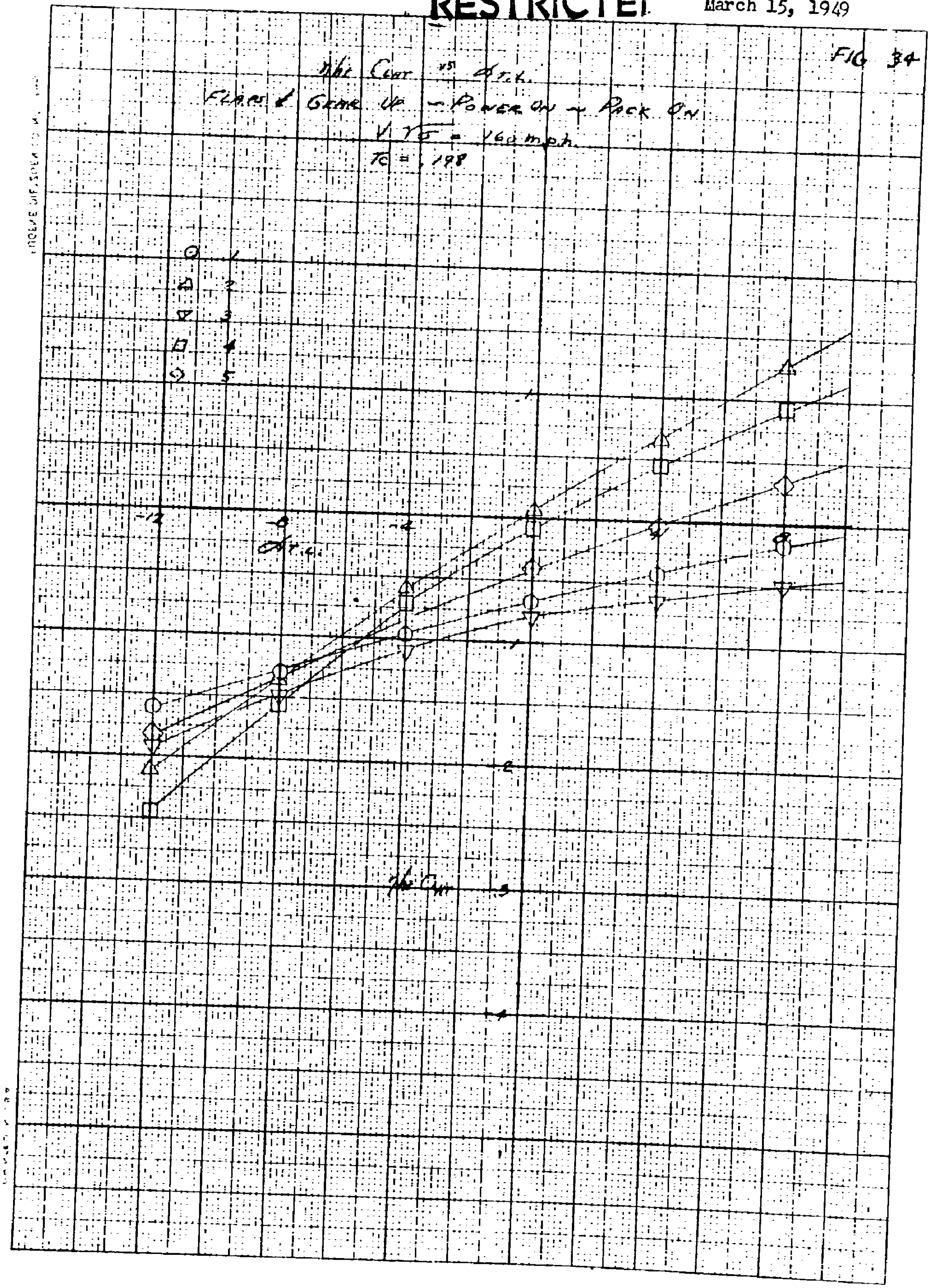


PART II - D - 1b. FLAPS AND GEAR UP - POWER ON

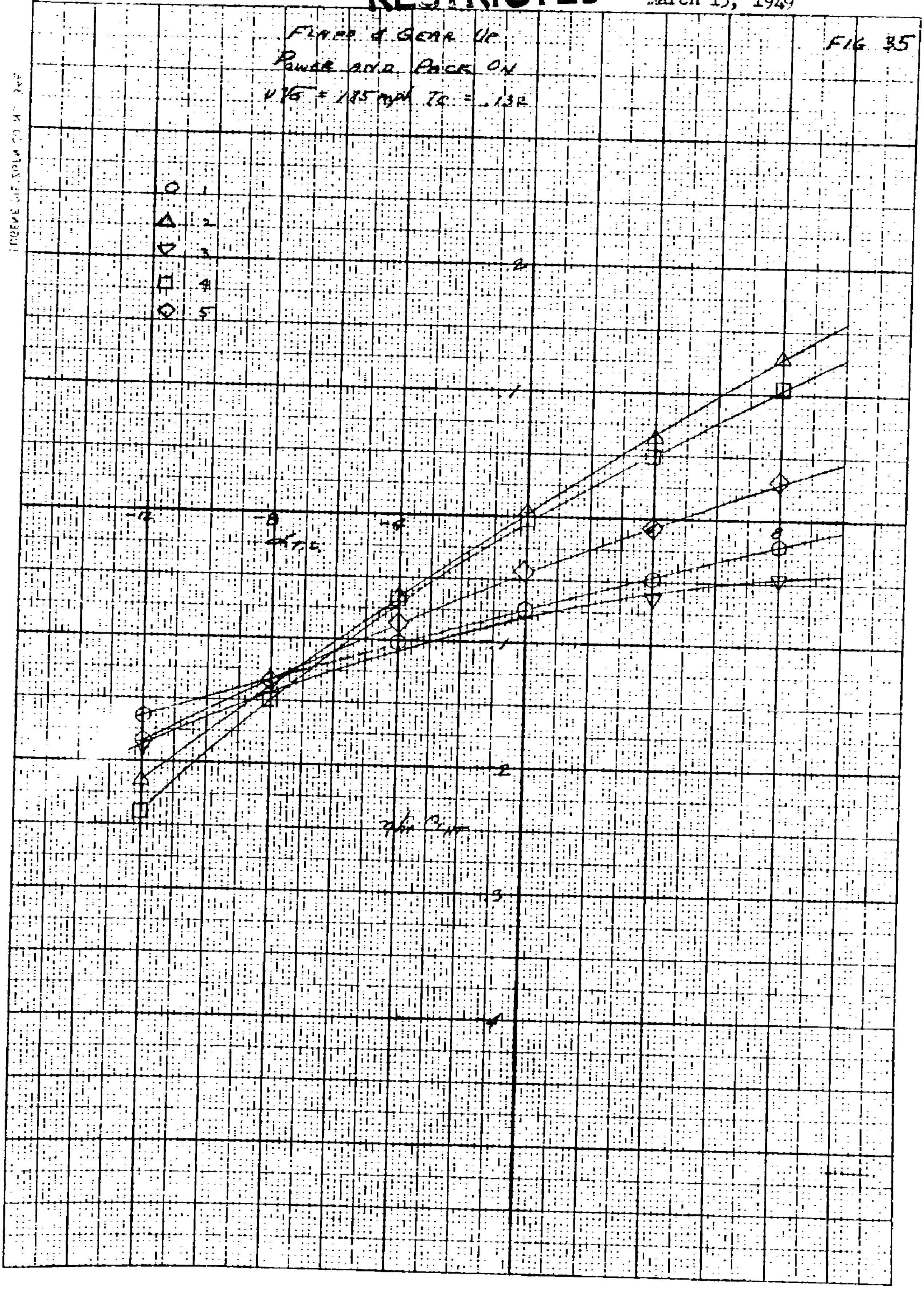
PART II - D - 1-b. FLAPS AND GEAR UP - POWER ON

$V_{\sigma}^{\frac{1}{2}} = 185 \qquad T_c = .132$

			C.G. #1			#2		
α_{TL}	α_{ht}	$\frac{q_{ht}}{q}$	$C_{M_{A-T}}$	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{L_{HT}}$	$C_{M_{A-T}}$	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{L_{HT}}$
-12	-9.40	1.000	-.1435	.882	-.163	-.1834	.858	-.214
-8	-7.05	1.000	-.1171	.882	-.133	-.1142	.858	-.139
-4	-4.70	1.027	-.0917	.906	-.101	-.0565	.881	-.064
0	-2.47	1.100	-.0700	.970	-.072	.0018	.944	+.002
4	-.63	1.123	-.0466	.990	-.047	.0626	.964	+.064
8	1.08	1.135	-.0205	1.001	-.020	.1256	.974	+.129

#3			#4			#5		
$C_{M_{A-T}}$	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{L_{HT}}$	$C_{M_{A-T}}$	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{L_{HT}}$	$C_{M_{A-T}}$	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{L_{HT}}$
-.1649	.882	-.187	-.2048	.858	-.239	-.1616	.873	-.185
-.1261	.882	-.143	-.1288	.858	-.150	-.1182	.873	-.135
-.0959	.906	-.106	-.0606	.881	-.069	-.0782	.897	-.087
-.0758	.970	-.075	-.0040	.944	-.004	-.0427	.960	-.044
-.0602	.940	-.061	+.0482	.964	+.050	-.0067	.980	-.007
-.0468	1.001	-.047	+.0990	.974	+.102	.0312	.991	+.031

FIG 35


Flaps & Gear Up
Power and Pack On
$V_{T/O}$ = 185 mph T_C = .138
O 1
△ 2
▽ 3
□ 4
◇ 5
-12
-8
-4
8
2
1
1
2
3
4

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REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION		PAGES	PAGE 147
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY	

Subject:— BASIC FLIGHT CRITERIA - PACK ON

DATE March 15, 1949

REVISED

PART II - D - 1b. FLAPS AND GEAR UP - POWER ON

PART II - D - 1b. FLAPS AND GEAR UP - POWER ON

$V\sigma^{\frac{1}{2}} = 250 \qquad T_c = .056$

			C.G. #1			CG #2		
α_{TL}	α_{ht}	$\frac{q_{ht}}{q}$	C_{MA-T}	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{LHT}$	C_{MA-T}	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{LHT}$
-12	-9.40	1.000	-.1476	.882	-.167	-.1572	.855	-.218
-8	-7.05	1.000	-.1203	.882	-.136	-.1231	.855	-.143
-4	-4.70	1.006	-.0450	.887	-.107	-.0637	.863	-.074
0	-2.44	1.042	-.0729	.919	-.079	-.1030	.894	-.003
4	-.45	1.055	-.0492	.931	-.053	.0567	.905	+.063
8	1.43	1.060	-.0229	.935	-.024	.1199	.909	+.132

CG #3			CG #4			CG #5		
C_{MA-T}	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{LHT}$	C_{MA-T}	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{LHT}$	C_{MA-T}	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{LHT}$
-.1603	.852	-.182	-.2000	.858	-.233	-.1640	.873	-.188
-.1209	.882	-.137	-.1240	.858	-.145	-.1203	.873	-.138
-.0905	.887	-.102	-.0562	.863	-.065	-.0802	.878	-.091
-.0700	.919	-.070	-.0001	.894	0	-.0447	.910	-.049
-.0543	.931	-.058	.0515	.905	+.057	-.0098	.921	-.010
-.0406	.935	-.043	.1021	.909	+.112	.0292	.925	+.032

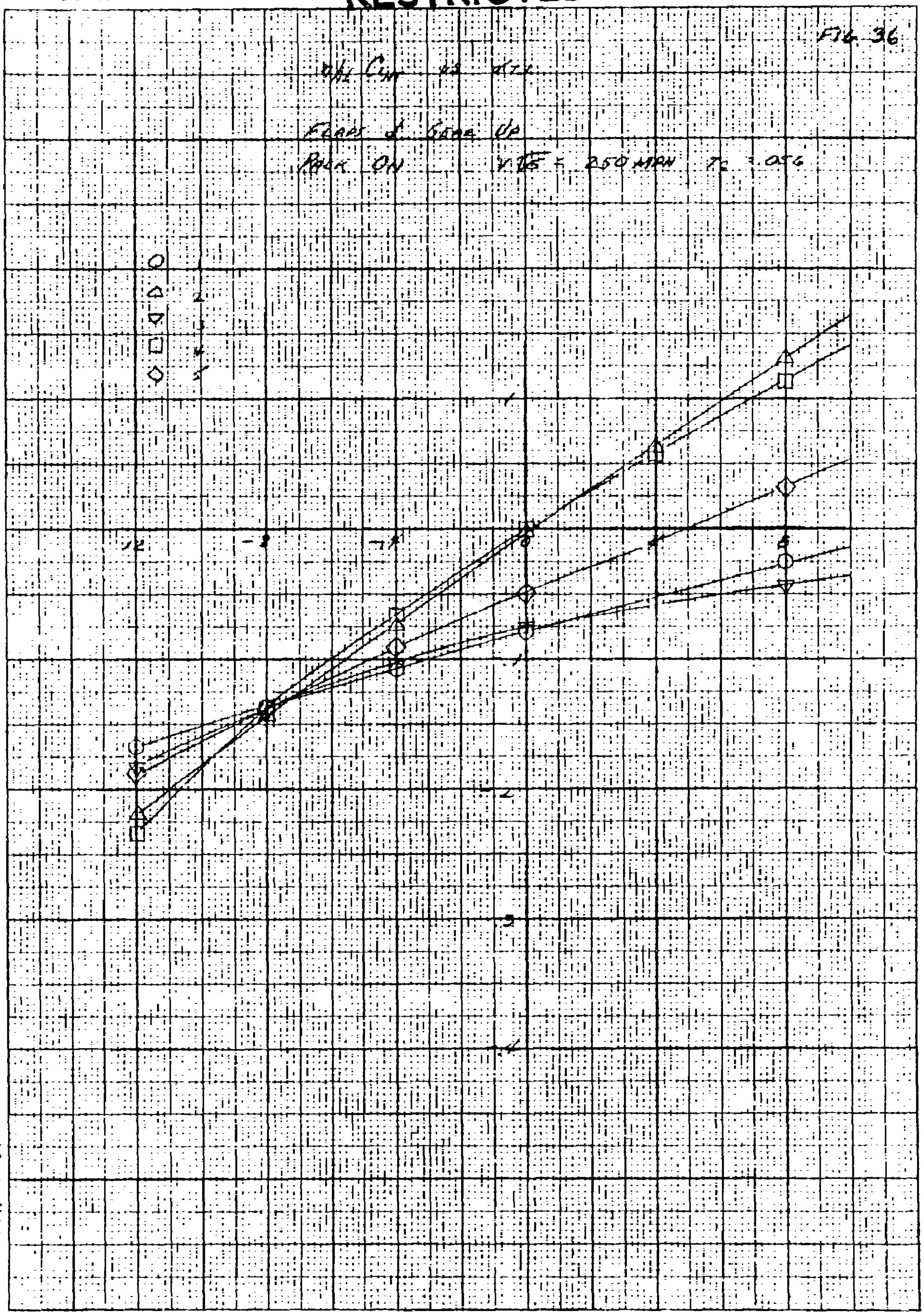
FIG 36
Flaps & Gear Up
Pack On
250 MPH
.056
1
2
3
4
5
-12
-8
-4
0
4
8
-1
-2
-3
-4

REPORT NO. R107-014 — FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION — RESTRICTED — PAGE 149

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

DATE March 15, 1949

REVISED

Subject:— BASIC FLIGHT CRITERIA - PACK ON

PART II - D - 1b. FLAPS AND GEAR UP - POWER ON

PART II - D - 1b. FLAPS AND GEAR UP - POWER ON

V_{T} = 313 T_c = .028

			CG 1			2		
α_{TL}	α_{ht}	$\frac{q_{ht}}{q}$	$C_{M_{A-T}}$	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{L_{HT}}$	$C_{M_{A-T}}$	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{L_{HT}}$
-12	-9.40	1.000	-.1504	.882	-.171	-.1900	.858	-.221
-8	-7.05	1.000	-.1224	.882	-.139	-.1253	.858	-.146
-4	-4.70	1.000	-.0966	.882	-.110	-.0629	.858	-.073
0	-2.40	1.021	-.0739	.901	-.052	-.0047	.876	-.005
4	-.27	1.025	-.0498	.907	-.055	.0551	.882	+.062
8	1.67	1.030	-.0229	[illegible]	-.025	.1186	.884	+.134

3			4			5		
$C_{M_{A-T}}$	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{L_{HT}}$	$C_{M_{A-T}}$	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{L_{HT}}$	$C_{M_{A-T}}$	$K\frac{q_{ht}}{q}$	$\eta_{ht}C_{L_{HT}}$
-.1599	.882	-.181	-.1996	.858	-.233	-.1663	.873	-.190
-.1197	.882	-.136	-.1232	.858	-.144	-.1219	.873	-.140
-.0888	.882	-.101	-.0550	.858	-.064	-.0814	.873	-.093
-.0678	.901	-.075	.0015	.876	+.002	-.0455	.891	-.051
-.0515	.907	-.057	.0533	.882	+.060	-.0091	.897	-.010
-.0376	.908	-.041	.1040	.884	+.118	+.0292	.899	+.032

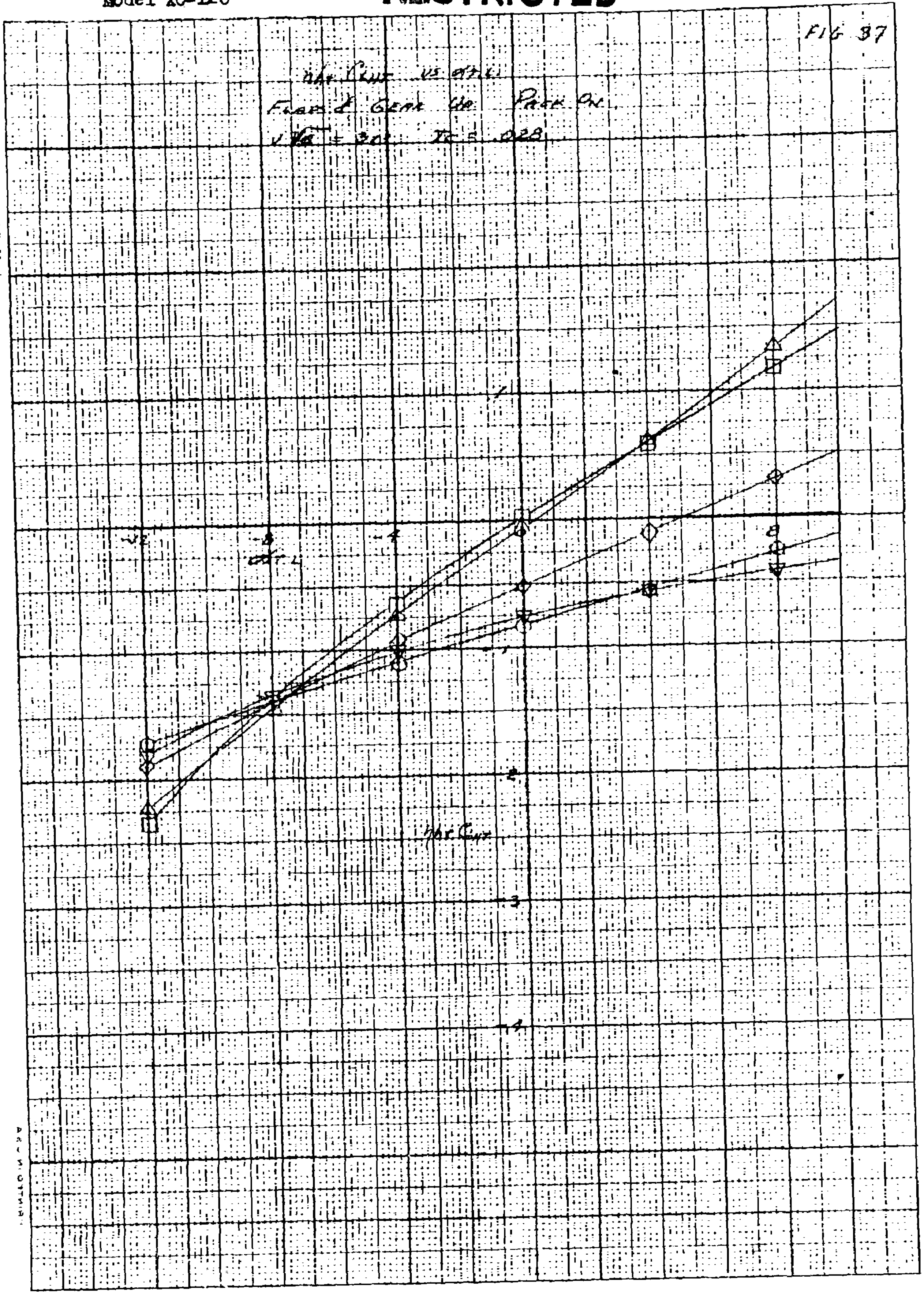
FIG 37
Flaps & Gear Up
-12
-8
-4
8
1
2
3
4

PART II - D - 2a.

2a. FLAPS AND GEAR DOWN - POWER OFF

For flaps and gear down, power off

$$\frac{q_{ht}}{q} = 1.00 \qquad \text{page 57 reference (1)}$$

Therefore the values of K will be the same as shown in Part II-D-1a.

α_{ht} is from figure 36 reference (1)

Curves of balancing tail lift coefficient are shown following each tabulation

REPORT NO. R107-014 | FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION | PAGES | PAGE 152

RESTRICTED

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

DATE March 15, 1949

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Subject:— BASIC FLIGHT CRITERIA - PACK ON

PART II - D - 2a. FLAPS AND GEAR DOWN - POWER OFF

PART II - D - 2a. FLAPS AND GEAR DOWN - POWER OFF

		C.G. #1			C.G. #2			C.G. #3		
$\alpha_{T.L.}$	α_{ht}	C_{MA-T}	K	$\eta_{ht}C_{LHT}$	C_{MA-T}	K	$\eta_{ht}C_{LHT}$	C_{MA-T}	K	$\eta_{ht}C_{LHT}$
-12	-12.57	-.2609	.862	-.296	-.2643	.858	-.308	-.2385	.852	-.270
-8	-10.18	-.2400		-.272	-.2077		-.242	-.2073		-.235
-4	-7.88	-.2214		-.251	-.1538		-.179	-.1858		-.211
0	-5.68	-.2060		-.234	-.1037		-.121	-.1746		-.198
4	-3.62	-.1908		-.216	-.0529		-.062	-.1707		-.194
8	-1.76	-.1729		-.176	.0011		+.001	-.1694		-.192

C.G. #4			C.G. #6		
C_{MA-T}	K	$\eta_{ht}C_{LHT}$	C_{MA-T}	K	$\eta_{ht}C_{LHT}$
-.2417	.857	-.282	-.2548	.877	-.291
-.1751		-.204	-.2226		-.254
-.1163		-.135	-.1955		-.223
-.0723		-.084	-.1736		-.198
-.0328		-.035	-.1543		-.176
.0047		+.005	-.1336		-.152

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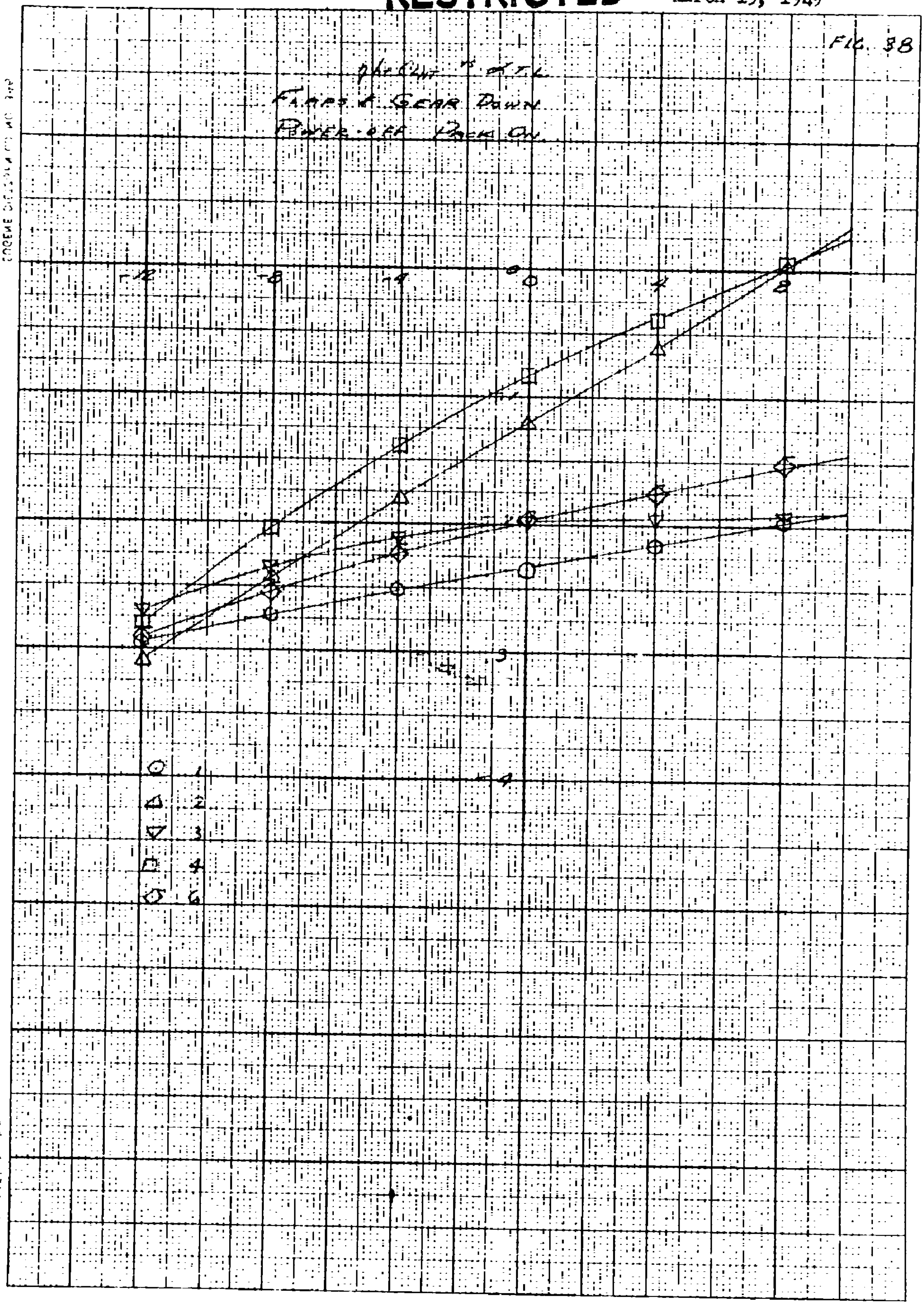

FIG. 38

PART II - D - 2b.

2b. FLAPS AND GEAR DOWN - POWER ON

For flaps and gear down, power on,

$$\frac{q_{ht}}{q} = 1.00 \quad \text{page 101 reference (1).}$$

Therefore the values of K will be the same as shown in Part II - D - 1b.

α_{ht} is from figure 64 reference (1).

Curves of balancing tail lift coefficient are shown following each tabulation.

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REPORT NO. R107-014

FAIRCHILD AIRCRAFT DIVISION
OF FAIRCHILD ENGINE & AIRPLANE CORPORATION

PAGES | PAGE 155

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

DATE March 15, 1949

REVISED

Subject - BASIC FLIGHT CRITERIA - PACK ON

PART II - D - 2b. FLAPS AND GEAR DOWN - POWER ON

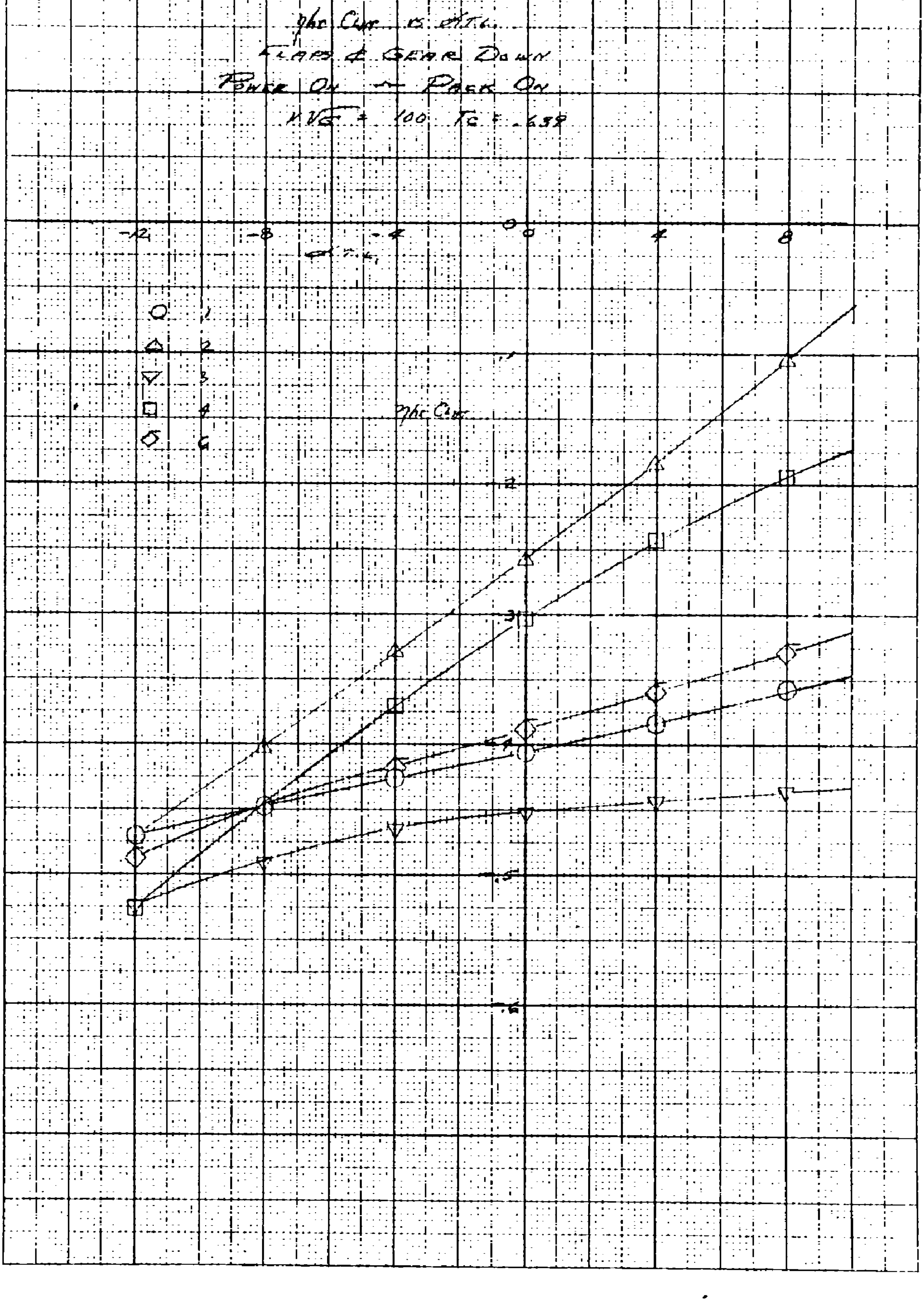
PART II - D - 2b. FLAPS AND GEAR DOWN - POWER ON

$V_c = 100 \quad T_c = .639$

		C.G. 1			2			3		
δ_{TL}	α_{ht}	$C_{m_{A-T}}$	K	$\eta_{ht} C_{L_{HT}}$	$C_{m_{A-T}}$	K	$\eta_{ht} C_{L_{HT}}$	$C_{m_{A-T}}$	K	$\eta_{ht} C_{L_{HT}}$
-12	-12.87	-.4145	.882	-.470	-.4053	.858	-.472	-.4620	.882	-.524
-8	-10.77	-.3953		-.448	-.3459		-.403	-.4325		-.490
-4	-8.73	-.3761		-.426	-.2827		-.329	-.4105		-.465
0	-6.76	-.3580		-.406	-.2209		-.257	-.3975		-.451
4	-4.92	-.3389		-.384	-.1576		-.184	-.3903		-.443
8	-3.33	-.3158		-.358	-.0595		-.104	-.3842		-.436

4			6		
$C_{m_{A-T}}$	K	$\eta_{ht} C_{L_{HT}}$	$C_{m_{A-T}}$	K	$\eta_{ht} C_{L_{HT}}$
-.4525	.858	-.527	-.4281	.877	-.485
-.3832		-.447	-.3967		-.452
-.3173		-.370	-.3673		-.418
-.1604		-.303	-.3410		-.389
-.2089		-.243	-.3159		-.360
-.1662		-.194	-.2553		-.329

FIG 39

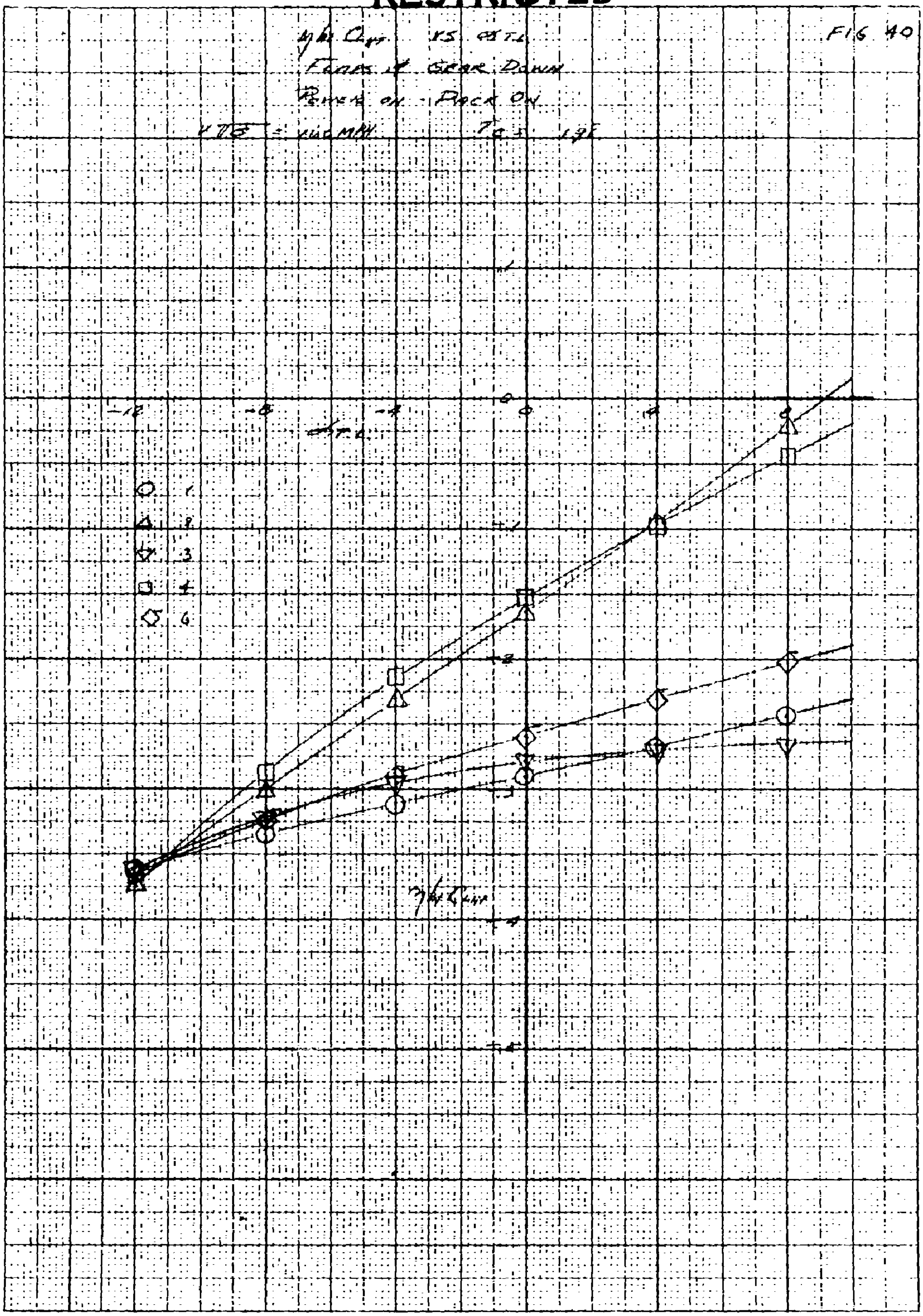

Cm vs α T.L.
FLAPS & GEAR DOWN
POWER ON ~ PACK ON
V = 100 Tc = .639
-12
-8
-4
0
4
8
α T.L.
1
2
3
4
6
.1
.2
.3
.4
.5
.6
Cm

PART II - D - 2b. FLAPS AND GEAR DOWN - POWER ON

V_T^2 = 160 T_c = .198

		C.G. 1			C.G. 2			C.G. 3		
$\alpha_{T.L.}$	α_{ht}	$C_{M_{A-T}}$	K	$\eta_{ht} C_{L_{HT}}$	$C_{M_{A-T}}$	K	$\eta_{ht} C_{L_{HT}}$	$C_{M_{A-T}}$	K	$\eta_{ht} C_{L_{HT}}$
-12	-12.66	-.3190	.882	-.362	-.3188	.858	-.372	-.3186	.882	-.361
-8	-10.44	-.2959		-.335	-.2780		-.301	-.2851		-.323
-4	-8.24	-.2743		-.311	-.1971		-.230	-.2608		-.296
0	-6.13	-.2555		-.290	-.1348		-.163	-.2463		-.279
4	-4.13	-.358		-.267	-.0567		-.094	-.2386		-.270
8	-2.37	-.2143		-.242	-.0192		-.022	-.2347		-.264

C.G. 4			C.G. 6		
$C_{M_{A-T}}$	K	$\eta_{ht} C_{L_{HT}}$	$C_{M_{A-T}}$	K	$\eta_{ht} C_{L_{HT}}$
-.3182	.858	-.371	-.3200	.877	-.365
-.2470		-.288	-.2543		-.324
-.1837		-.214	-.2535		-.289
-.1306		-.152	-.2273		-.259
-.0629		-.097	-.2028		-.231
-.0376		-.044	-.1774		-.202



PART II-E

E. COMPLETE AIRPLANE NORMAL FORCE COEFFICIENTS

This section covers the determination of the normal force coefficients of the complete airplane.

As the horizontal tail normal force coefficient will vary with center of gravity and/or power conditions, the normal force coefficients of the tailless airplane are first determined and the normal force coefficients of the horizontal tail then added to obtain the normal force coefficient of the entire airplane for a given configuration, center of gravity location, and/or power condition.

The aerodynamic force coefficients of the component parts as obtained from reference (1) are all converted to wing area and free stream dynamic pressure before summing to obtain total normal force coefficients.

PART II - E - 1

1-a. FLAPS ANG GEAR UP - POWER OFF

The normal force coefficients of the complete airplane for flaps and gear up, power off, are as follows.

$$C_{Z_A} = C_{Z_W} + \underline{C}_{Z_F} + \underline{C}_{Z_B} + \underline{C}_{Z_{VT}} + \underline{C}_{Z_{HT}}$$

where the force coefficients are all based on wing area and free stream dynamic pressure.

As the values of C_{Z_W} for the unflapped wing as obtained from Part II-A-1-o reference (1) are all based on wing area no corrections are necessary.

As the values of C_{Z_F} for the fuselage power off as obtained from Part II-A-5-e are based on fuselage planform area, they are corrected as follows.

$$\underline{C}_{Z_F} = C_{Z_F} \frac{P_f}{S_W} = C_{Z_F} \frac{579}{1447.25} = .4001\ C_{Z_F}$$

α_{TL}	C_{Z_F}	$\underline{C}_{Z_F}$
-12	-.0951	-.0380
- 8	-.0519	-.0208
- 4	-.0189	-.0076
0	0	0
4	.0189	.0076
8	.0519	.0208

PART II - E - 1

The values of C_{Z_B} for the boom power off as obtained from Part II-A-6-e reference (1) are for one boom only and are based on the planform area of one boom. These values are corrected as follows:

$$\underline{C_{Z_B}} = 2 \left(C_{Z_B} \frac{P_b}{S_W} \right) = 2 \; C_{Z_B} \frac{312.5}{1447.25} = .4319 \; C_{Z_B}$$

α TL	C_{Z_B}	$\underline{C_{Z_B}}$
-12	-.0347	-.0150
- 8	-.0166	-.0072
- 4	-.0042	-.0018
0	0	0
4	.0042	.0018
8	.0166	.0072

For the vertical tail $\underline{C_{Z_{VT}}} = 0$ as $C_{Z_{VT}}$

$= 0$ from Part II-A-9-e reference (1).

The values of $\underline{C_{Z_{HT}}}$ will of course vary with center of gravity locations and/or power conditions as previously noted.

so that

$$\underline{C_{Z_{HT}}} = \eta_{ht} \; C_{L_{HT}} \; \frac{S_{ht}}{S_W} \; \frac{q_{ht}}{q}$$

where $\eta_{ht} \; C_{L_{HT}}$ is based on S_{ht} and q_{ht} and is obtained from Part II-D .

$$\eta_{ht} \; C_{L_{HT}} = C_{M_{A-T}} \; \frac{S_W}{S_{ht}} \; \frac{MAC_W}{l_{ht}} \; \frac{q}{q_{ht}}$$

substituting

$$\underline{C_{Z_{HT}}} = C_{M_{A-T}} \; \frac{MAC_W}{l_{ht}}$$

PART II - E - 1a. FLAPS AND GEAR UP - POWER OFF

PART II - E - 1a. FLAPS AND GEAR UP - POWER OFF

						C.G. ①			C.C. ②	
α_{TL}	C_{Z_W}	C_{Z_F}	C_{Z_B}	$C_{Z_{A-T}}$	$C_{M_{A-T}}$	$\frac{MAC_W}{Z_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_W}{Z_{HT}}$
-12	-.325	-.0380	-.0150	-.3780	-.1353	.2709	-.0362	-.4147	-.1731	.2785
-8	.005	-.0206	-.0072	-.0230	-.1126		-.0305	-.0535	-.1146	
-4	.342	-.0026	-.0018	.3376	-.0922		-.0250	+.3126	-.0583	
0	.655	0	0	.6550	-.0751		-.0203	+.6647	-.0066	
4	1.123	.0026	.0018	1.0324	-.0565		-.0153	+1.0171	.0468	
8	1.363	.0203	.0072	1.3910	-.0303		-.0096	+1.3814	.1034	

		C.G. ③				C.G. ④			
$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_W}{Z_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_W}{Z_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}
-.0482	-.4262	-.1419	.2709	-.0384	-.4164	-.1798	.2785	-.0500	-.4280
-.0319	-.0549	-.1067		-.0289	-.0519	-.1091		-.0304	-.0534
-.0162	+.3214	-.0511		-.0220	+.3156	.0472		-.0131	+.3245
-.0018	+.6832	-.0657		-.0176	+.6672	.0028		+.0008	+.6858
+.0130	+1.0454	-.0555		-.0150	+1.0174	.0477		+.0133	+1.0457
+.0288	+1.4198	-.0451		-.0130	+1.3780	.0908		+.0253	+1.4163

Subject: BASIC FLIGHT CRITERIA - PACK ON

PART II - E - 1a. FLAPS AND GEAR UP - POWER OFF

PART II - E - 1-a. FLAPS AND GEAR UP - POWER OFF

C.G. 5

C_{MA-T}	$\frac{MAC_w}{Z_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}
-.1500	.2737	-.0411	-.4191
-.1112		-.0304	-.0534
-.0764		-.0209	+.3167
-.0463		-.0127	+.6723
-.0160		-.0044	+1.0280
.0160		+.0044	+1.3954

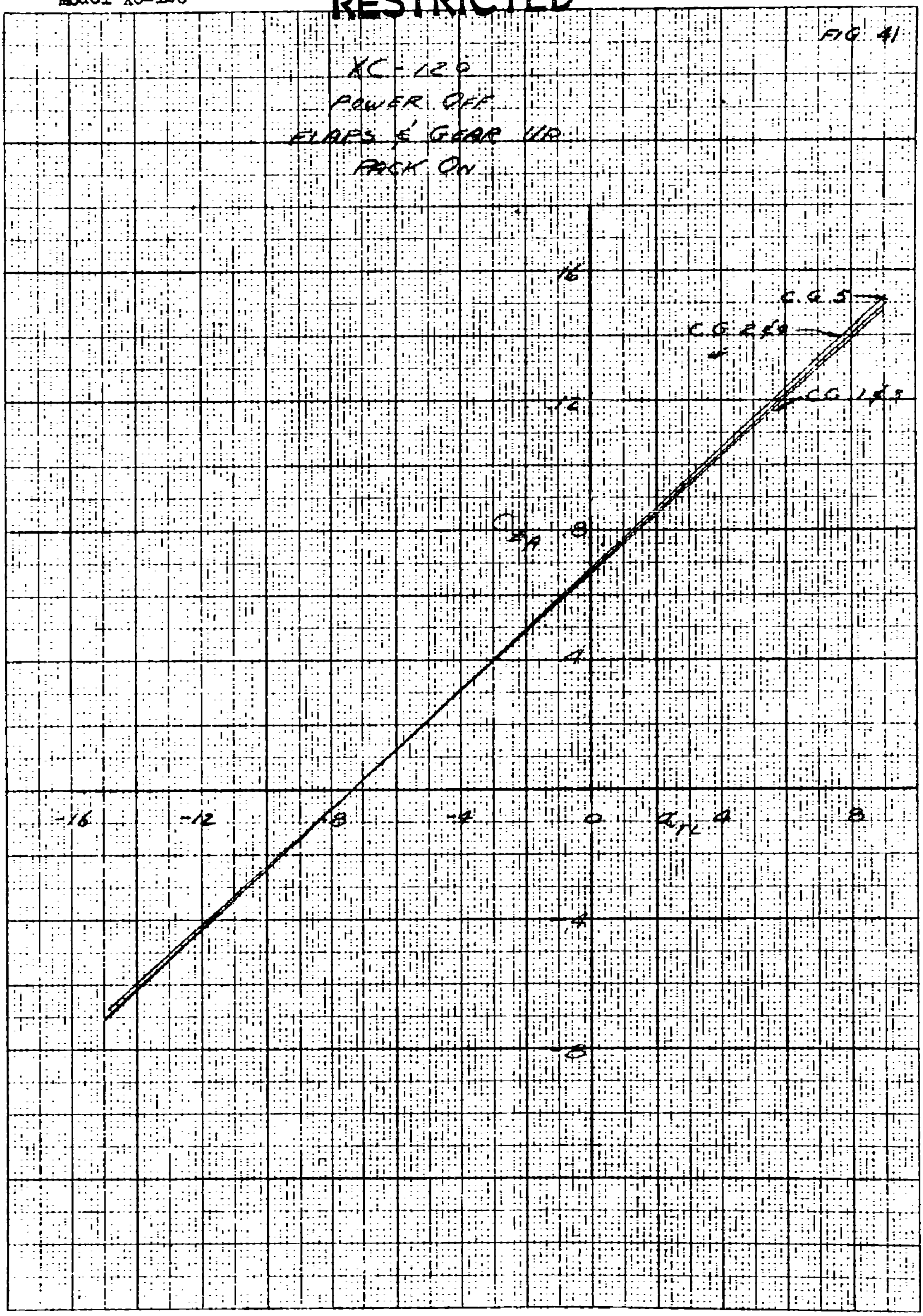


FIG. 41
XC-120
POWER OFF
FLAPS & GEAR UP
PACK ON
C.G. 5
C.G. 28
C.G. 18
16
12
8
4
-4
-8
-16
-12
-8
-4
0
4
8

PART II - E - 1

1b. FLAPS AND GEAR UP - POWER ON

The normal force coefficients of the complete airplane for flaps and gear up, power on, are as follows:

$$C_{Z_A} = C_{Z_{A-T}}{}_{\text{Power Off}} + 2\,\Delta C_{Z_{WI}} + 2\,\underline{C}_{Z_P} + \underline{C}_{Z_{HT}}$$

Where $C_{Z_{A-T}}{}_{\text{Power Off}}$ is the same as determined in Part II-E-1a

The values of $\Delta C_{Z_{WI}}$ due to the propeller slipstream effects on the portion of the wing immersed in the propeller slipstream as obtained from Part II-B-1e reference (1) are for one immersed section only and must be doubled to take care of two engine operation. The values are based on total wing area and therefore need no other correction.

The values of $\underline{C}_{Z_P}$ from Part II-B-8 of reference (1) are also doubled to take care of two engine operation and are determined as follows:

$$2\,\underline{C}_{Z_P} = 2\,C_{Z_P}\,\frac{F_P}{S_W}$$

where $C_{Z_P} = \dfrac{Z_P}{q\,F_P}$ Figure 60 reference (1) for α_{TL}

The values of $\underline{C}_{Z_{HT}}$ are determined the same as in Part II-E-1a.

PART II - E - 1b POWER ON - FLAPS AND GEAR UP

PART II - E - 1b POWER ON - FLAPS AND GEAR UP

$V\sqrt{\sigma}$ = 100 mph.

T_c = .639

						C.G. 1				C.G. 2
α_{TL}	$C_{Z_{A-T}}$ POWER OFF	$2\Delta C_{Z_{WI}}$	C_{Y_P}	$\angle C_{Z_P}$	$C_{Z_{A-T}}$ POWER ON	$C_{M_{A-T}}$	$\frac{MAC_w}{l_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$
-12	-.3780	-.0052	-.0541	-.0134	-.3966	-.1269	.2709	-.0344	-.4310	-.1666
-8	-.0230	.0366	-.0372	-.0092	.0044	-.0996		-.0270	-.0226	-.0985
-4	.3371	.0786	-.0183	-.0045	.4112	-.0740		-.0200	+.3912	-.0326
0	.6650	.1206	0	0	.8056	-.0514		-.0139	+.7917	.0291
4	1.0324	.1650	.0183	.0045	1.2019	-.0272		-.0074	1.1945	.0929
8	1.3910	.2106	.0372	.0092	1.6108	-.0002		-.0001	1.6107	.1606

C.G. 2				C.G. 3				C.G. 4		
$\frac{MAC_w}{l_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_w}{l_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_w}{l_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}
.2785	-.0464	-.4431	-.2053	.2709	-.0556	-.4522	-.2410	.2785	-.0682	-.4648
	-.0275	-.0231	-.1662		-.0450	-.0460	-.1651		-.0460	-.0416
	-.0091	+.4021	-.1356		-.0367	+.3745	-.0941		-.0263	+.3849
	+.0081	+.8137	-.1147		-.0311	+.7745	-.0341		-.0095	+.7961
	+.0259	1.2278	-.0982		-.0266	1.1753	.0219		+.0061	1.2080
	+.0447	1.6555	-.0837		-.0227	+1.5881	.0721		+.0215	1.6223

FA-800-23

RESTRICTED

REPORT NO. R107-014 — FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION — PAGE 167

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

DATE March 15, 1949

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Subject:— BASIC FLIGHT CRITERIA - PACK ON

PART II - E - 1b POWER ON - FLAPS AND GEAR UP

PART II - E - 1b POWER ON - FLAPS AND GEAR UP

$v\sqrt{\sigma}$ = 100 mph

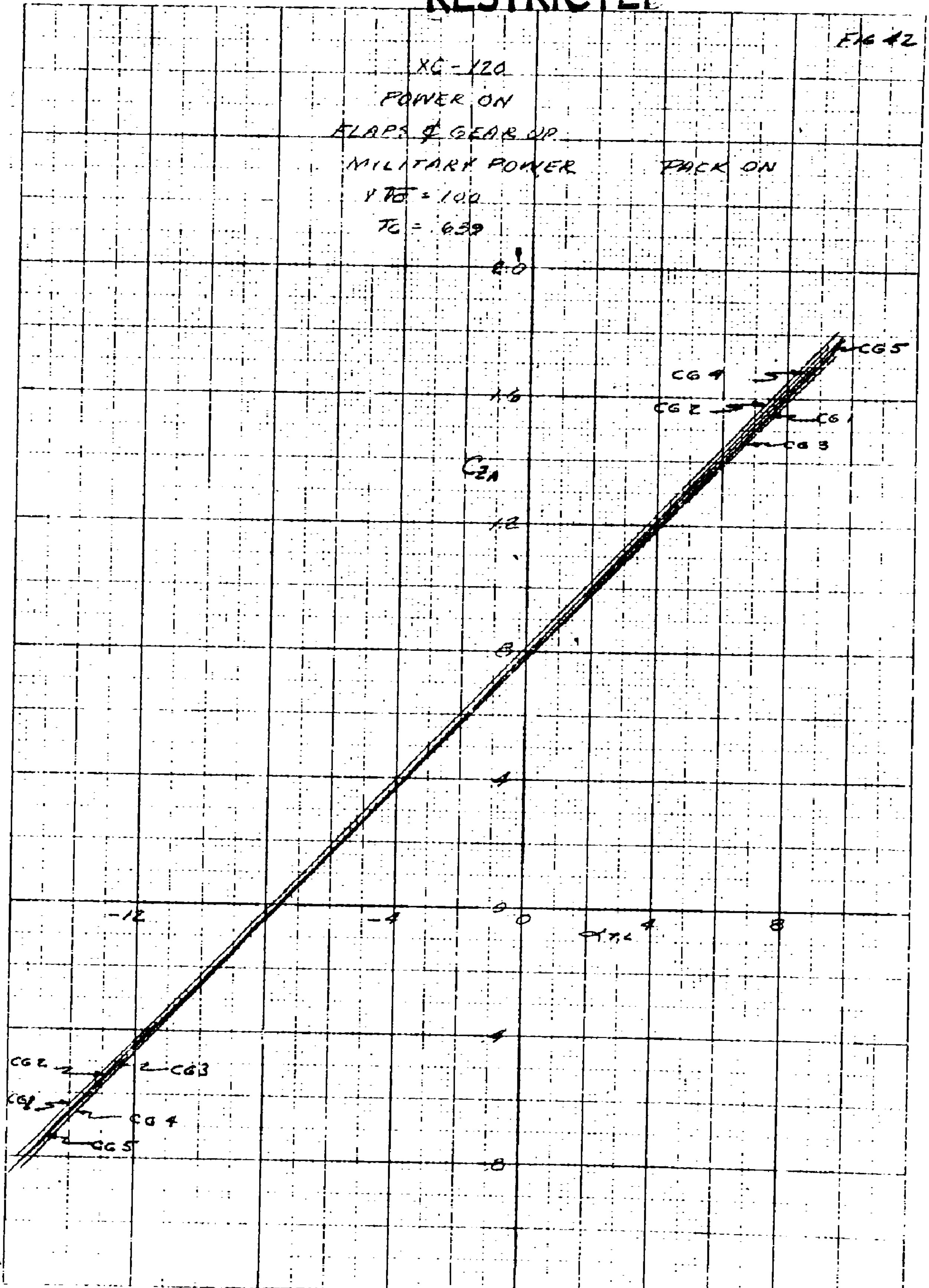
T_c = .639

C. G. 5

$C_{m_{A-T}}$	$\frac{MAC_W}{\ell_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}
-.1551	.2737	-.0425	-.4391
-.1100		-.0301	-.0257
-.0682		-.0157	+.3925
-.0309		-.0085	+.7971
.0069		+.0019	1.2038
.0471		+.0124	1.6237

RESTRICTED

FIG 42



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

Subject:— BASIC FLIGHT CRITERIA - PACK ON | DATE March 15, 19 | REVISED

PART II - E - 1b POWER ON - FLAPS AND GEAR UP

PART II - E - 1b POWER ON - FLAPS AND GEAR UP

$V\sqrt{\sigma}$ = 160 mph

T_c = .198

							C.G. 1			C.G. 2
α_{T_L}	$C_{Z_{A-T}}$ POWER OFF	$2\Delta C_{Z_{WI}}$	C_{Y_P}	$2C_{Z_P}$	$C_{Z_{A-T}}$ POWER ON	$C_{M_{A-T}}$	$\frac{MAC_W}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$
-12	-.3780	-.0094	-.0493	-.0122	-.3996	-.1410	.2709	-.0382	-.4378	-.1809
8	-.0230	.0094	-.0335	-.0083	-.0.19	-.1142		-.0309	-.0528	-.1161
-4	.3371	.0284	-.0170	-.0042	.3613	-.0894		-.0242	+.3371	-.0531
0	.6850	.0476	0	0	.7326	-.0675		-.0183	+.7143	.0057
4	1.0324	.0674	.0170	.0042	1.1040	-.0441		-.0119	+1.0921	.0663
8	1.3910	.0876	.0335	.0083	1.4869	-.0182		-.0049	+1.4820	.1303

C.G. 2				C.G. 3						
$\frac{MAC_W}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_W}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_W}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}
.2785	-.0504	-.4500	-.1698	.2709	-.0460	-.4456	-.2099	.2785	-.0585	-.4581
	-.0323	-.0542	-.1308		-.0354	-.0573	-.1332		-.0371	-.0590
	-.0148	+.3465	-.1010		-.0274	+.3339	-.0647		-.0180	+.3433
	+.0016	+.7342	-.0808		-.0219	+.7107	-.0075		-.0021	+.7305
	+.0185	+1.1225	-.0652		-.0177	+1.0863	.0452		+.0126	+1.1166
	+.0363	+1.5232	-.0520		-.0141	+1.4728	.0964		+.0268	+1.5197

FA-806-25

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MODEL XC-120 PREPARED BY CHECKED BY APPROVED BY

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Subject: BASIC FLIGHT CRITERIA - PACK ON

PART II - E -1b POWER ON - FLAPS AND GEAR UP

PART II - E - 1b POWER ON - FLAPS AND GEAR UP

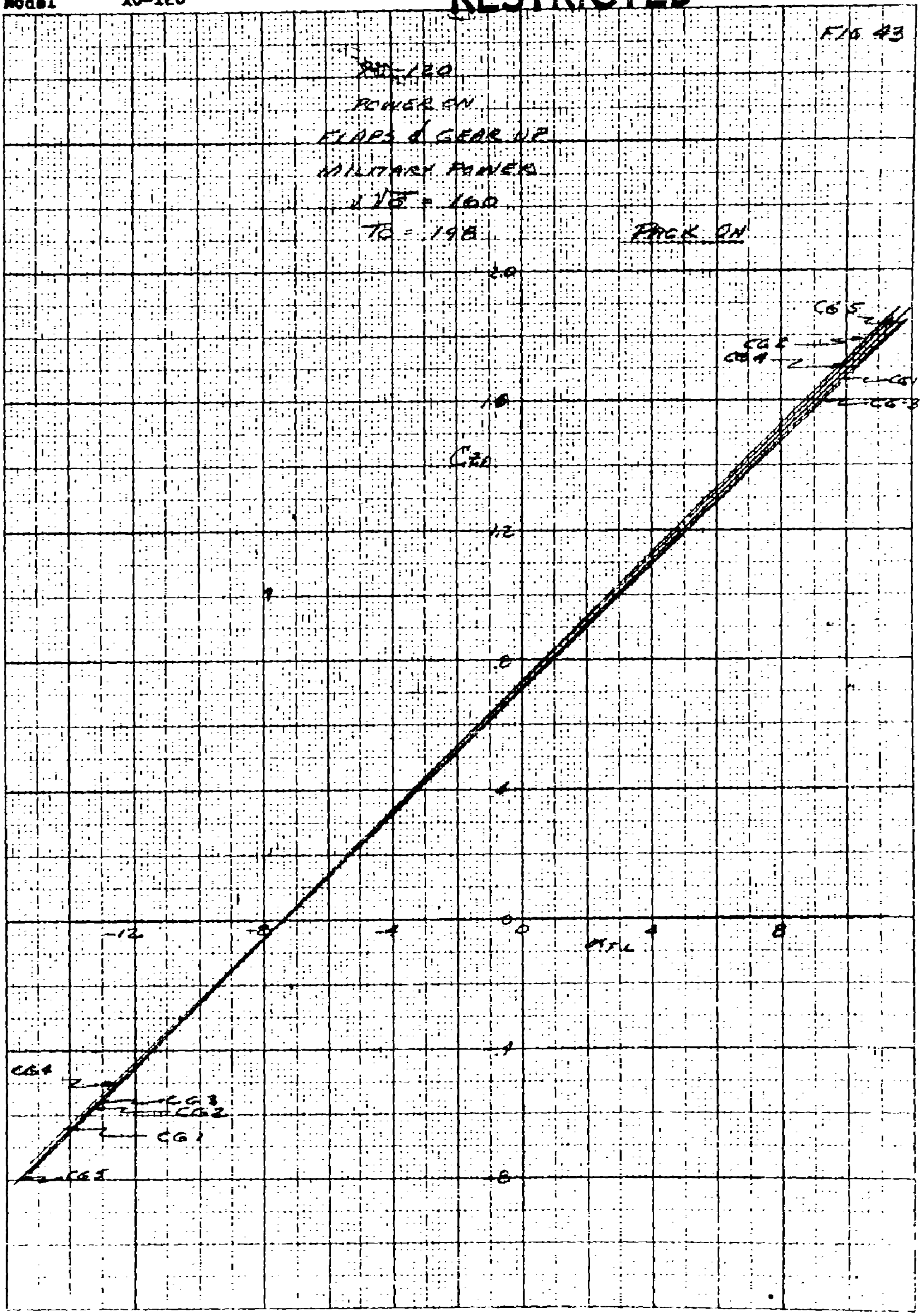
$V\sqrt{\sigma}$ = 160 mph

T_c = .198

c.G.S

$C_{M_{A-T}}$	$\frac{MAC_w}{\ell_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}
-.1605	.2737	-.0439	-.4435
-.1162		-.0326	-.0539
-.0762		-.0210	+.3403
-.0410		-.0112	+.7214
-.0048		-.0013	+1.1027
.0332		+.0091	+1.4960

FIG 43


XC-120
POWER ON
FLAPS & GEAR UP
MILITARY POWER
$V\sqrt{\sigma}$ = 160
TC = .198
PACK ON
CG 5
CG 2
CG 4
CG 1
CG 3
C_{ZL}
2.0
1.6
1.2
.8
.4
0
-.4
-.8
-12
-8
-4
4
8
M_{TL}
CG 4
CG 3
CG 2
CG 1
CG 5

PART II - E - 1b. POWER ON - FLAPS AND GEAR UP

PART II - E - 1b POWER ON - FLAPS AND GEAR UP

$V\sqrt{\sigma}$ = 185 mph

T_c = .132

						C.G. (1)				C.G. (2)
α_{TL}	$C_{Z_{A-T}}$ POWER OFF	$2\Delta C_{Z_{WT}}$	C_{Y_P}	$2C_{Z_F}$	$C_{Z_{A-T}}$	$C_{M_{A-T}}$	$\frac{MAC_w}{\ell_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$
-12	-.3780	-.0074	-.0505	-.0125	-.3479	-.1435	.2709	-.0389	-.4368	-.1834
-8	-.0230	.0060	-.0350	-.0086	-.0256	-.1171		-.0317	-.0573	-.1100
-4	.3371	.0200	-.0175	-.0043	.3528	-.0919		-.0249	+.3279	-.0565
0	.6850	.0330	0	0	.7180	-.0700		-.0190	+.6990	.0018
4	1.0324	.0472	.0175	.0043	1.0839	-.0466		-.0126	+1.0713	.0620
8	1.3910	.0624	.0350	.0086	1.4620	-.0265		-.0056	+1.4564	.1256

C.G. (2)				C.G. (3)				C.G. (4)		
$\frac{MAC_w}{\ell_{ht}}$	$C_{Z_{HT}}$	C_{Z_R}	$C_{M_{A-T}}$	$\frac{MAC_w}{\ell_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_w}{\ell_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}
.2785	-.0511	-.4490	-.1649	.2709	-.0447	-.4426	-.2045	.2785	-.0570	-.4549
	-.0306	-.0562	-.1261		-.0342	-.0598	-.1288		-.0359	-.0615
	-.0157	+.3371	-.0959		-.0260	+.3268	-.0606		-.0169	+.3359
	+.0005	+.7185	-.0758		-.0205	+.6975	-.0040		-.0011	+.7169
	+.0173	+1.1012	-.0602		-.0163	+1.0676	.0482		+.0134	+1.0773
	+.0350	+1.4470	-.0468		-.0127	+1.4433	.0990		+.0276	+1.4896

Subject: BASIC FLIGHT CRITERIA - PACK ON

PART II - E - 1b POWER ON - FLAPS AND GEAR UP

PART II - E - 1b POWER ON - FLAPS AND GEAR UP

$V\sqrt{\sigma}$ = 185 mph.

T_c = .132

C.G. 5

$C_{m_{A-T}}$	$\frac{MAC_w}{\ell_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}
-.1616	.2737	-.0442	-.4421
-.1182		-.0324	-.0580
-.0782		-.0211	+.3314
-.0427		-.0117	+.7063
-.0061		-.0018	+1.0821
.0312		+.0055	+1.4705

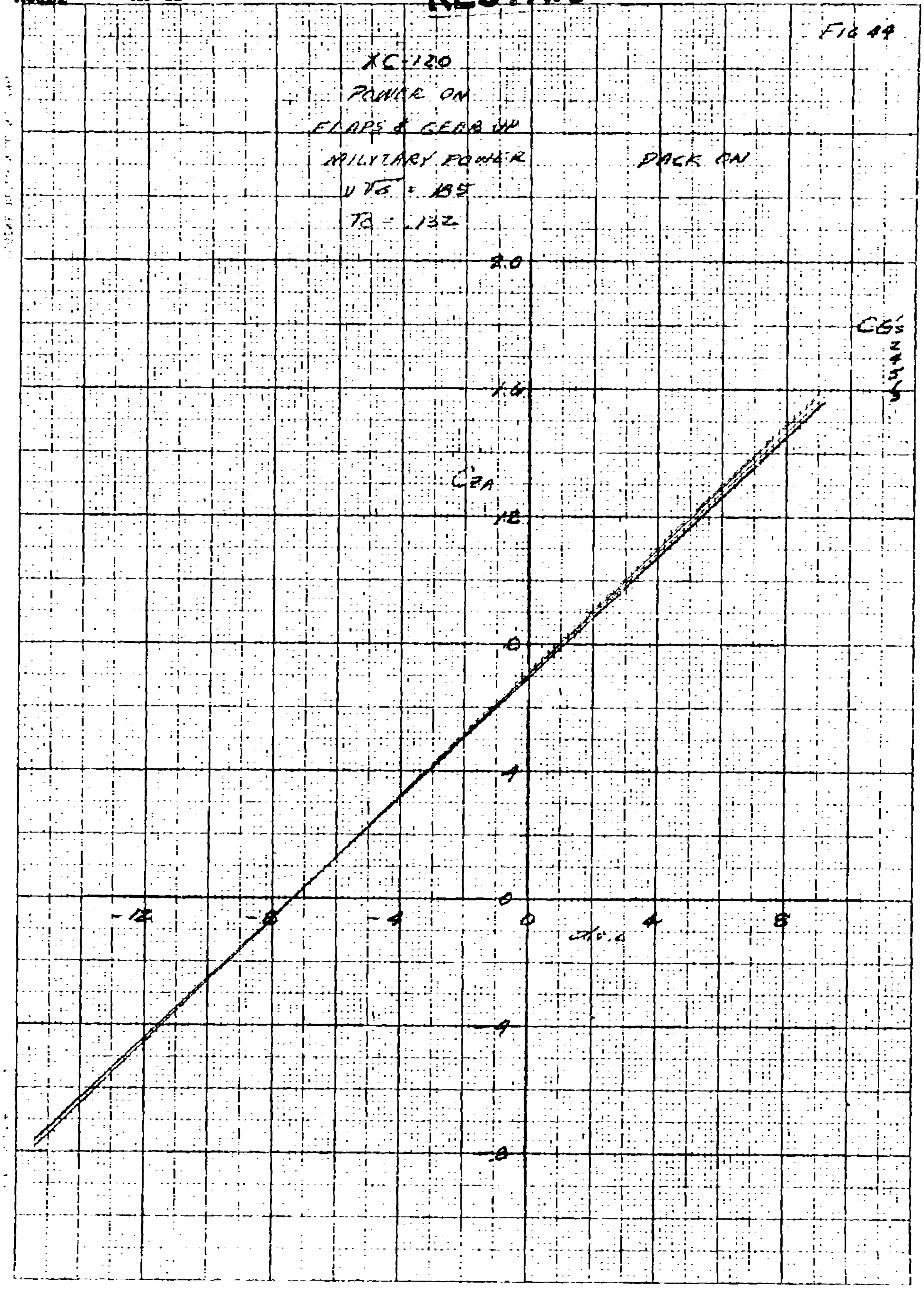


FIG 44
XC-120
POWER ON
FLAPS & GEAR UP
MILITARY POWER
PACK ON
$V\sqrt{\sigma}$ = 185
T_C = .132
CG's
2.0
1.6
C'_{ZA}
1.2
.8
.4
0
-.4
-.8
-12
-8
-4
0
4
8

PART II - E - 1b POWER ON - FLAPS AND GEAR UP

PART II - E - 1b POWER ON - FLAPS AND GEAR UP

$V\sqrt{\sigma}$ = 250 mph.

T_c = .056

							C.G. 1			C.G. 2
α_{TL}	$C_{Z_{A-T}}$ POWER OFF	$2\Delta C_{Z_{WI}}$	$C_{Y_{1'}}$	$2C_{Z_F}$	$C_{Z_{A-T}}$ POWER ON	$C_{M_{A-T}}$	$\frac{MAC_w}{\ell_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$
-12	-.3780	-.0040	-.0560	-.0138	-.3958	-.1476	.2709	-.0400	-.4358	-.1877
-8	-.0230	.0020	-.0381	-.0094	-.0364	-.1203		-.0326	-.0630	-.1231
-4	.3371	.0082	-.0187	-.0046	.3407	-.0956		-.0217	+.3156	-.0607
0	.6850	.0144	0	0	.6994	-.0729		-.0197	+.6797	-.0030
4	1.0324	.0212	.0187	.0046	1.0552	-.0492		-.0133	+1.0449	.0527
8	1.3910	.0290	.0381	.0094	1.4274	-.0229		-.0062	1.4232	.1199

C.G. 2				C.G. 3				C.G. 4		
$\frac{MAC_w}{\ell_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_w}{\ell_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_w}{\ell_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}
.2785	-.0521	-.4472	-.1603	.2709	-.0434	-.4392	-.2000	.2785	-.0557	-.4515
	-.0343	-.0647	-.1207		-.0328	-.0632	-.1240		-.0345	-.0649
	-.0169	+.3238	-.0965		-.0245	+.3162	-.0562		-.0157	+.3256
	-.0008	+.6986	-.0700		-.0190	+.6804	-.0001		0	+.6494
	+.0158	+1.0740	-.0543		-.0147	+1.0435	.0515		+.0143	+1.0725
	+.0334	+1.4628	-.0406		-.0110	+1.4184	.1021		+.0284	+1.4578

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REPORT NO. R107-014 — FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION — PAGE 176

MODEL: XC-120 | PREPARED BY | CHECKED BY | APPROVED BY

DATE: March 15, 1949

REVISED

Subject:-- BASIC FLIGHT CRITERIA - PACK ON

PART II - E - 1b POWER ON - FLAPS AND GEAR UP

PART II - E - 1b POWER ON - FLAPS AND GEAR UP

$V\sqrt{\sigma}$ = 250 mph.

T_c = .056

C. G. (5)

$C_{m_{A-T}}$	$\frac{MAC_w}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}
-.1640	.2737	-.0449	-.4407
-.1263		-.0329	-.0633
-.0862		-.0220	+.3187
-.0447		-.0122	+.6572
.0085		-.0024	+1.0558
.0292		+.0080	+1.4374

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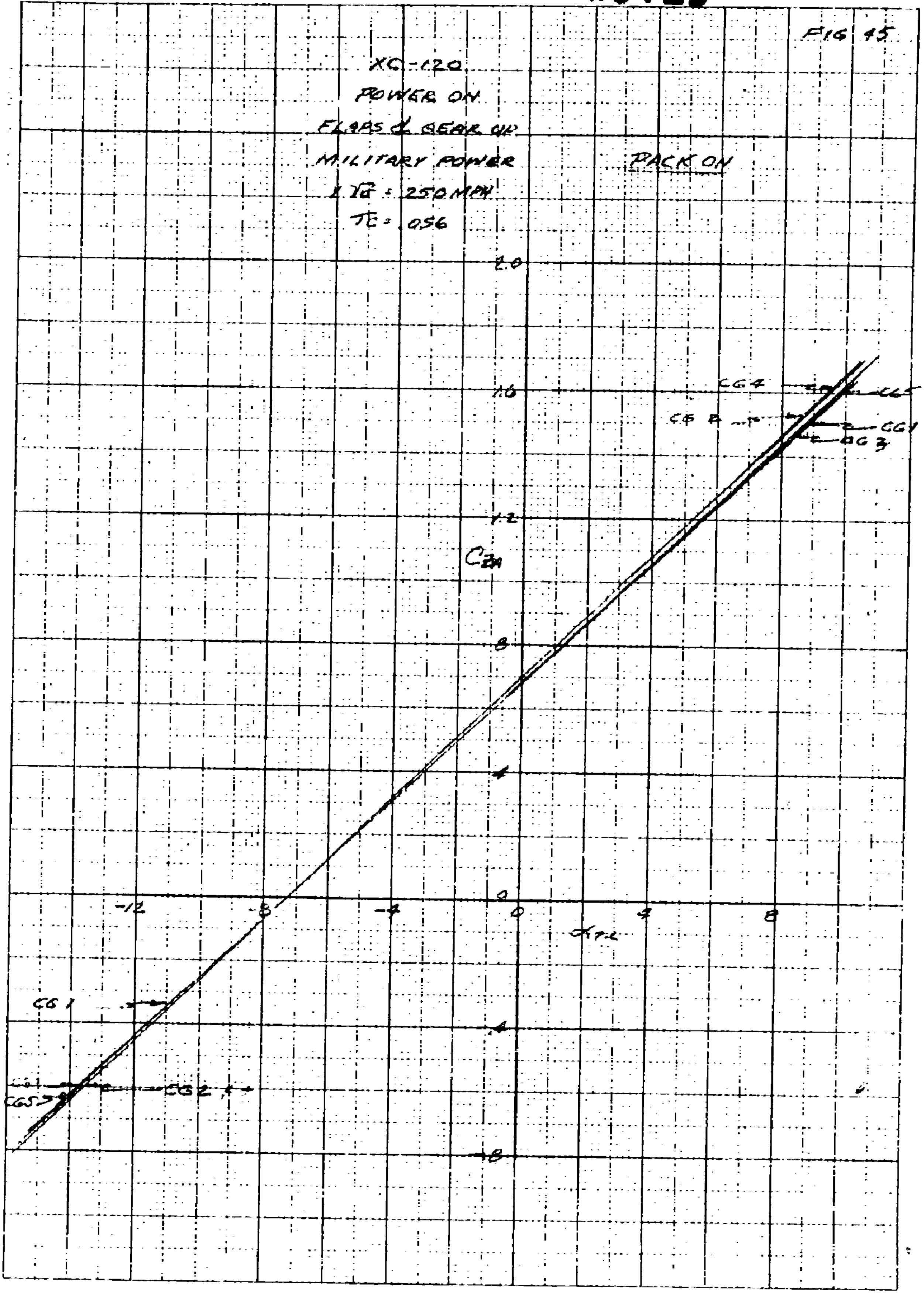
FIG 45
XC-120
POWER ON
FLAPS & GEAR UP
MILITARY POWER
PACK ON
250 MPH
.056
2.0
1.6
1.2
.8
.4
0
-.4
-.8
-12
-8
-4
0
4
8
CG 4
CG 5
CG 2
CG 1
CG 3
CG 1
CG 2
CG 5

Subject - BASIC FLIGHT CRITERIA - PACK ON

PART II - E - 1b POWER ON - FLAPS AND GEAR UP

PART II - E - 1b POWER ON - FLAPS AND GEAR UP

$V\sqrt{\sigma}$ = 313 mph.

T_c = .028

α_{TL}	$C_{Z_{A-T}}$ POWER OFF	$2\Delta C_{Z_{WL}}$	C_{Y_P}	$2C_{Z_P}$	$C_{Z_{A-T}}$ POWER ON	$C_{M_{A-T}}$	C.G. ① $\frac{MAC_W}{\ell_{Ar}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$
-12	-.3760	-.0020	-.0674	-.0157	-.3457	-.1504	.2709	-.0407	-.4364	-.1900
-8	-.0220	.0006	-.0422	-.0104	-.0328	-.1224		-.0332	-.0660	-.1253
-4	.3371	.0040	-.0211	-.0052	.3359	-.0966		-.0262	+.3097	-.0629
0	.6850	.0074	0	0	.6921	-.0139		-.0200	+.6724	-.0047
4	1.0324	.0106	.0211	.0052	1.0482	-.0495		-.0135	+1.0347	.0551
8	1.3410	.0158	.0422	.0104	1.4172	-.0229		-.0062	+1.4110	.1186

C.G. ② $\frac{MAC_W}{\ell_{Ar}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	C.G. ③ $\frac{MAC_W}{\ell_{Ar}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	C.G. ④ $\frac{MAC_W}{\ell_{Ar}}$	$C_{Z_{HT}}$	C_{Z_A}
.2785	-.0529	-.4486	-.1599	.2709	-.0433	-.4390	-.1496	.2785	-.0556	-.4513
	-.0349	-.0677	-.1197		-.0324	-.0652	-.1232		-.0343	-.0671
	-.0175	+.3184	-.0888		-.0241	+.3118	-.0550		-.0153	+.3206
	-.0013	+.6911	-.0678		-.0181	+.6740	.0015		+.0004	+.6928
	+.0153	+1.0635	-.0515		-.0140	+1.0342	.0533		+.0148	+1.0690
	+.0330	+1.4502	-.0376		-.0102	+1.4070	.1040		+.0290	+1.4562

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REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION		PAGE
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY
			DATE March 15,
Subject BASIC FLIGHT CRITERIA - PACK ON			REVISED

PART II - E - 1b POWER ON - FLAPS AND GEAR UP

PART II - E - 1b POWER ON - FLAPS AND GEAR UP

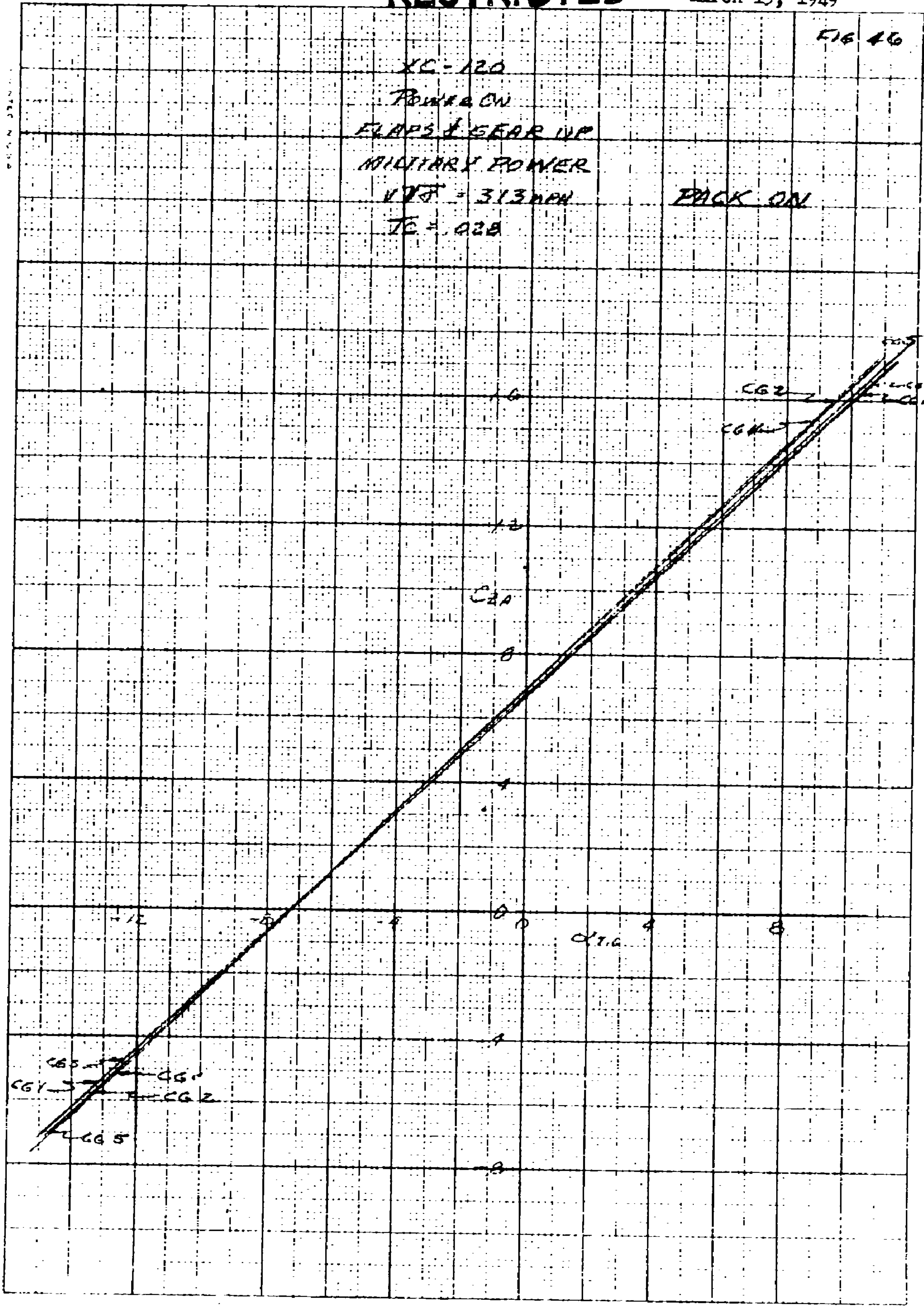
$V\sqrt{\sigma} = 313$ mph

$T_c = .028$

C.G. 5

$C_{m_{A-T}}$	$\frac{MAC_w}{\ell_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}
-.1662	.2737	-.0433	-.9412
-.1219		-.0334	-.0666
-.0814		-.0223	+.3136
-.0455		-.0125	+1.0799
.0041		-.0021	+1.4057
.0292		+.0080	+1.4252

FIG 46


XC-120
POWER ON
FLAPS & GEAR UP
MILITARY POWER
V_T = 313 MPH
T_C = .028
PACK ON
CG 5
CG 2
CG 8
CG 1
CG 4
1.6
1.2
C_{L_A}
.8
.4
0
-12
-8
-4
4
8
$\alpha_{T.C}$
-.4
-.8
CG 5
CG 1
CG 4
CG 2
CG 5

PART II - E - 2a

2a. FLAPS AND GEAR DOWN - POWER OFF

The complete airplane normal force coefficients for flaps and gear down - power off, are as follows:

$$C_{Z_A} = C_{Z_{WF}} + \underline{C_{Z_F}} + \underline{C_{Z_B}} + \underline{C_{Z_{VT}}} + \underline{C_{Z_G}} + \underline{C_{Z_{HT}}}$$

Where the aerodynamic force coefficients of the component parts are all based on wing area and free stream dynamic pressure.

As the values of $C_{Z_{WF}}$ for the flapped wing, power off as obtained from Part II-A-2-1 reference (1), are based on wing area, no corrections are necessary.

The values of $\underline{C_{Z_F}}$ and $\underline{C_{Z_B}}$ and $\underline{C_{Z_{VT}}}$ are the same as shown in Part II-E-1.

The values of C_{Z_G} for the landing gear power off as obtained from Part II-A-11-c, reference (1), are based on wing area so that:

$$\underline{C_{Z_G}} = 2\, C_{Z_G}$$

α_{TL}	C_{Z_G}	$\underline{C_{Z_G}}$
-12	-.00681	-.0136
- 8	-.00456	-.0091
- 4	-.00229	-.0045
0	0	0
4	.00229	.0045
8	.00456	.0091

Values of $\underline{C_{Z_{HT}}}$ are determined as outlined in Part II-E-1.

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PART II - E - 2a. FLAPS AND GEAR DOWN - POWER OFF

PART II - E - 2a POWER OFF - FLAPS AND GEAR DOWN

C.G. 1

α	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	MAC$_w$ / ℓ_{HT}	$C_{Z_{HT}}$	C_{Z_H}
-12	[illegible]	-.0331	-.0150	-.0036	-.0026	[illegible]	.2709	-.0701	-.1333
-8	.3420	-.0208	.0072	-.0091	.3057	-.2100		-.0650	+.2397
-4	[illegible]	-.0016	[illegible]	[illegible]	[illegible]	[illegible]		-.0600	+.6121
0	1.0040	0	0	[illegible]	[illegible]	-.2050		[illegible]	+.9082
4	1.312	.0020	[illegible]	[illegible]	[illegible]	-.1985		-.0317	[illegible]
8	1.704	.0268	.0072	.0091	[illegible]	.1924		[illegible]	[illegible]

C.G. 2				C.G. 3				C.G. 4			
$C_{m_{A-T}}$	MAC$_w$ / ℓ_{H}	$C_{Z_{HT}}$	C_{Z_H}	$C_{m_{A-T}}$	MAC$_w$ / ℓ_{HT}	$C_{Z_{HT}}$	C_{Z_H}	$C_{m_{A-T}}$	MAC$_w$ / ℓ_{HT}	$C_{Z_{HT}}$	C_{Z_H}
-.2613	.2785	[illegible]	-.1367	[illegible]	.2709	[illegible]	-.1212	.2477	.2765	-.0673	-.1299
-.2077		[illegible]	+.2071	.2073		[illegible]	[illegible]	-.1751		-.0488	+.2561
-.1538		.0428	+.6215	-.1878		[illegible]	+.6218	-.1183		-.0224	+.6392
[illegible]		-.0289	+.9951	-.1746		[illegible]	+.9767	-.0722		-.0261	+1.0039
-.0539		-.0147	+1.3712	-.1703		-.0401	+1.3597	.0328		-.0091	+1.3768
.0011		[illegible]	+1.7414	-.1694		-.0357	+1.6452	.0047		+.0013	+1.7424

Subject: BASIC FLIGHT CRITERIA - PACK ON

PART II - E - 2a FLAPS AND GEAR DOWN - POWER OFF

PART II - E - 2a FLAPS AND GEAR DOWN - POWER OFF

C. G. 6

C_{MA-T}	$\frac{MAC_w}{\ell_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}
-.2548	.2726	-.0695	-.1371
-.2226		-.0607	+.2442
-.1953		-.0533	+.6180
-.1776		-.0473	+.9767
-.1543		-.0421	+1.3436
-.1336		-.0366	+1.7047

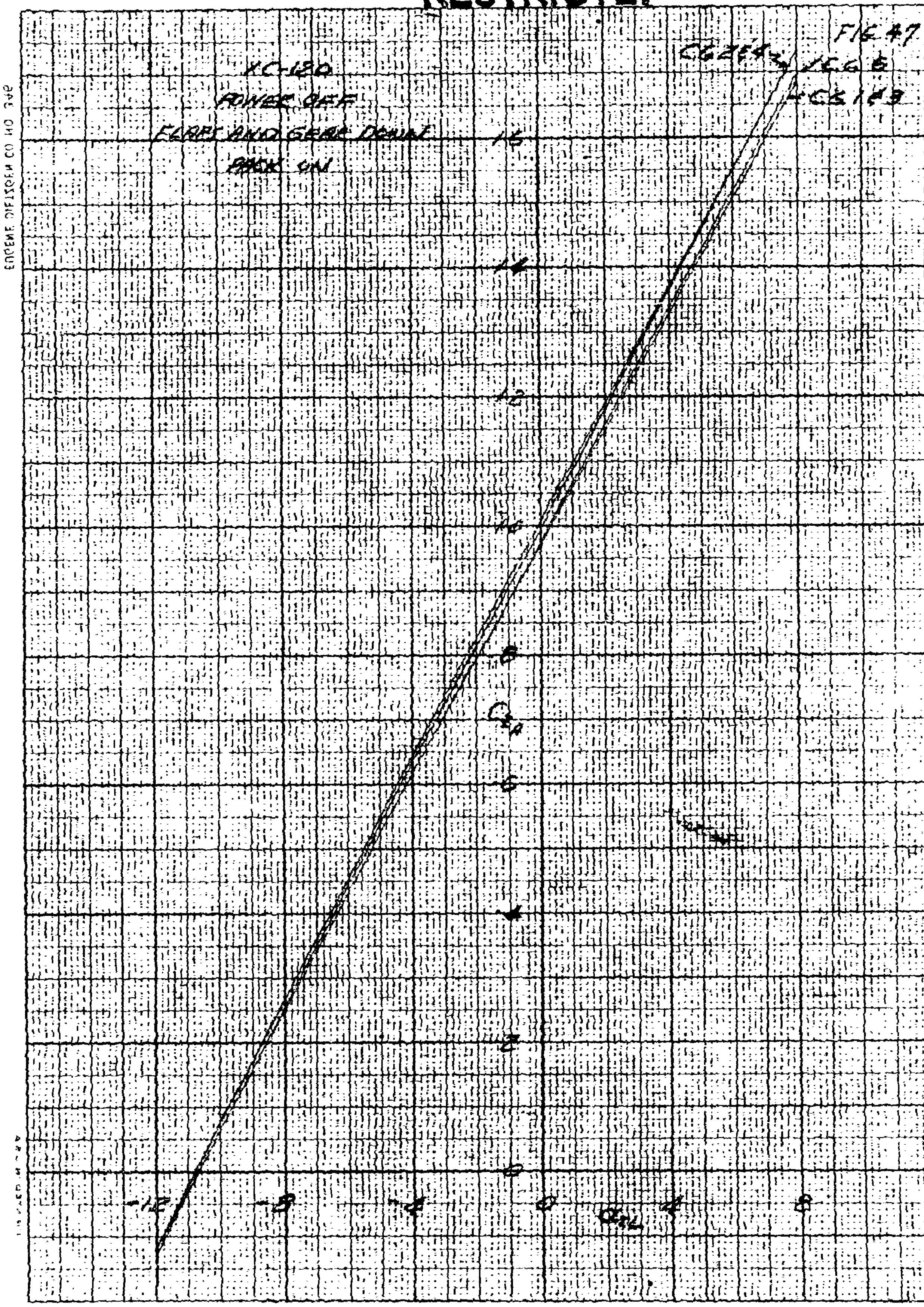
FIG 47
XC-120
POWER OFF
FLAPS AND GEAR DOWN
PACK ON
16
14
12
10
8
6
4
2
0
-12
-8
-4
0
4
8

PART II - E - 2

2b. FLAPS AND GEAR DOWN - POWER ON

The normal force coefficients for the complete airplane for flaps and gear down, power on, are as follows:

$$C_{Z_A} = C_{Z_{A-T_{Power\ Off}}} + 2\,\Delta C_{Z_{WFl}} + 2\,C_{Z_P} + C_{Z_{HT}}$$

Where $C_{Z_{A-T_{Power\ Off}}}$ is the same as determined in Part II-E-2a.

The values of $\Delta C_{Z_{WFl}}$ due to the propeller slipstream effects on the portion of the flapped wing immersed in the propeller slipstream as obtained from Part II-B-2-e reference (1) are for one immersed section only and must be doubled to take care of two engine operation. The values are based on total wing area and therefore need no additional corrections.

The values of C_{Z_P} are the same as those determined in Part II-E-1a for the same power conditions.

The values of $C_{Z_{HT}}$ are determined as outlined in Part II-E-1.

PART II - E -2b FLAPS AND GEAR DOWN - POWER ON

PART II - E - 2b FLAPS AND GEAR DOWN - POWER ON

$V\sqrt{\sigma}$ = 100 mph

T_c = .639

							C.G. 1			C.G. 2
α_{TL}	$C_{Z_{A-T}}$ POWER OFF	$2\Delta C_{Z_{WI}}$	C_{Y_P}	$2C_{Z_P}$	$C_{Z_{A-T}}$ POWER ON	$C_{M_{A-T}}$	$\frac{MAC_W}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$
-12	.0026	.1020	-.0541	-.0134	.0260	-.4145	.2709	-.1123	-.0863	-.4053
-8	.3049	.1804	-.0322	-.0092	.4761	-.3953		-.1071	+.3670	-.3459
-4	.6221	.2636	-.0183	-.0045	.9312	-.3761		-.1019	+.8243	-.2627
0	1.0246	.3410	0	0	1.3744	-.3580		-.0970	+1.2750	-.2269
4	1.3559	.4300	.0183	.0045	1.8201	-.3389		-.0918	+1.7256	-.1576
8	1.7411	.5152	.0372	.0092	2.2653	-.3158		-.0872	+2.1792	-.0895

C.G. 2				C.G. 3				C.G. 4		
$\frac{MAC_W}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_W}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_W}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}
.2785	-.1129	-.0869	-.4620	.2709	-.1252	-.0992	-.4525	.2795	-.1260	-.1000
	-.0963	+.3798	-.4325		-.1172	+.3589	-.3632		-.1067	+.3694
	-.0787	+.8525	-.4105		.1112	+.8210	-.3173		-.0884	+.8428
	-.0615	+1.3105	-.3975		-.1077	+1.2643	-.2604		-.0725	+1.2995
	-.0439	+1.7765	-.3903		-.1057	+1.7147	-.2069		-.0582	+1.7622
	-.0249	+2.2404	-.3842		-.1041	+2.1612	-.1662		-.0463	+2.1612

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REPORT NO. R107-014 FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION PAGE 187

MODEL XC-120 PREPARED BY CHECKED BY APPROVED BY

Subject: BASIC FLIGHT CRITERIA DATE March 15, 1949 REVISED

PART II - E - 2b FLAPS AND GEAR DOWN - POWER ON

PART II - E - 2b POWER ON - FLAPS AND GEAR DOWN

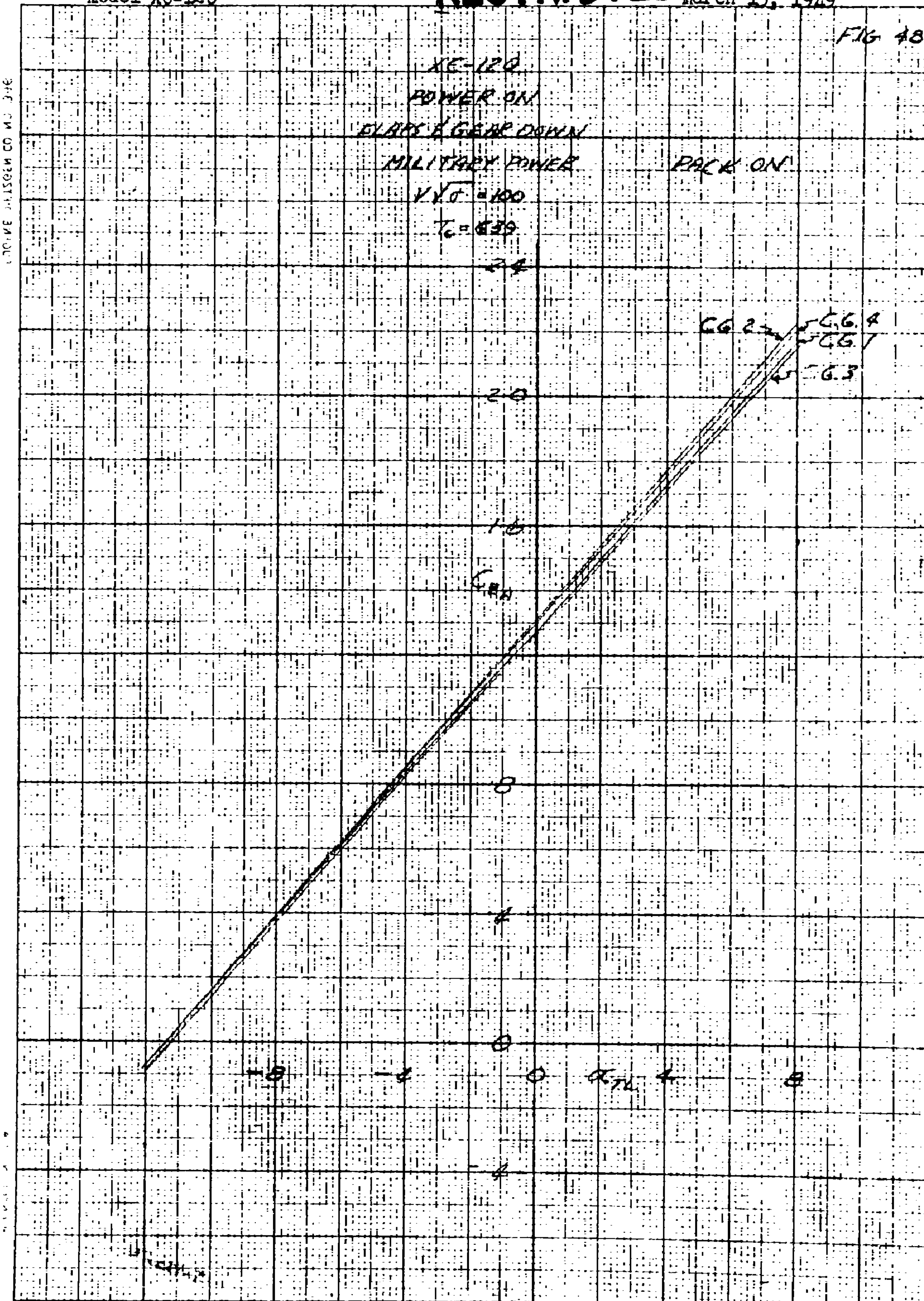
$V\sqrt{\sigma}$ = 100 mph

T_c = .639

C.G. 6

$C_{m_{A-T}}$	$\frac{MAC_w}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}
-.4281	.2726	-.1167	-.0907
.3467		-.1081	+.3686
-.3673		-.1001	+.8311
-.3410		-.0930	+1.2790
.3159		-.0861	+1.7343
-.2883		-.0786	+2.1867

FIG 48


XC-120
POWER ON
FLAPS & GEAR DOWN
MILITARY POWER
PACK ON
$V\sqrt{\sigma} = 100$
$T_c = .639$
CG 2
CG 4
CG 1
CG 3
2.4
2.0
1.6
1.2
.8
.4
0
-.4
C_{R_N}
-8
-4
0
4
8
α_{TL}

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REPORT NO. R107-014 — FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION — PAGE 189

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

DATE March 15, 1949

REVISED

Subject:— BASIC FLIGHT CRITERIA - PACK ON

PART II - E - 2b POWER ON - FLAPS AND GEAR DOWN

PART II - E - 2b POWER ON - FLAPS AND GEAR DOWN

$V\sqrt{\sigma}$ = 160 mph

T_c = .198

							C.G. 1			C.G. 2
α_{TL}	$C_{Z_{H-T}}$ POWER OFF	$2\Delta C_{ZW_1}$	C_{Y_P}	$2C_{X_P}$	$C_{Z_{H-T}}$ POWER ON	$C_{M_{A-T}}$	$\frac{MAC_w}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$
-12	-.0626	.0470	-.0493	-.0122	-.0278	-.3190	.2709	-.0864	-.1142	-.3188
-8	.3049	.0652	-.0335	-.0083	.3016	-.2959		-.0802	+.2814	-.2580
-4	.6721	.0990	-.0170	-.0042	.7069	-.2743		-.0743	+.6926	-.1971
0	1.0240	.1350	0	0	1.1590	-.2555		-.0692	+1.0898	-.1398
4	1.3859	.1700	.0170	.0042	1.5601	-.2358		-.0639	+1.4962	-.0807
8	1.7411	.2030	.0335	.0053	1.9124	-.2143		-.0551	+1.8943	-.0192

C.G. 2				C.G. 3				C.G. 4		
$\frac{MAC_w}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_w}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}	$C_{M_{A-T}}$	$\frac{MAC_w}{l_{HT}}$	$C_{Z_{HT}}$	C_{Z_A}
.2735	-.0888	-.1166	-.3186	.2709	-.0563	.1141	-.3132	.2755	-.0886	-.1164
	-.0719	+.2597	-.2551		-.0772	+.2844	-.2474		-.0669	+.4927
	-.0549	+.7120	-.2605		-.0707	+.6962	-.1537		-.0512	+.7157
	-.0359	+1.1201	.2463		-.0667	+1.0923	-.1300		-.0364	+1.1226
	-.0725	+1.5376	.2380		-.0645	+1.4956	-.0827		-.0231	+1.5370
	-.0053	+1.9472	-.2327		-.0630	+1.8894	-.0376		-.0105	+1.9419

PART II - E - 2b POWER ON - FLAPS AND GEAR DOWN

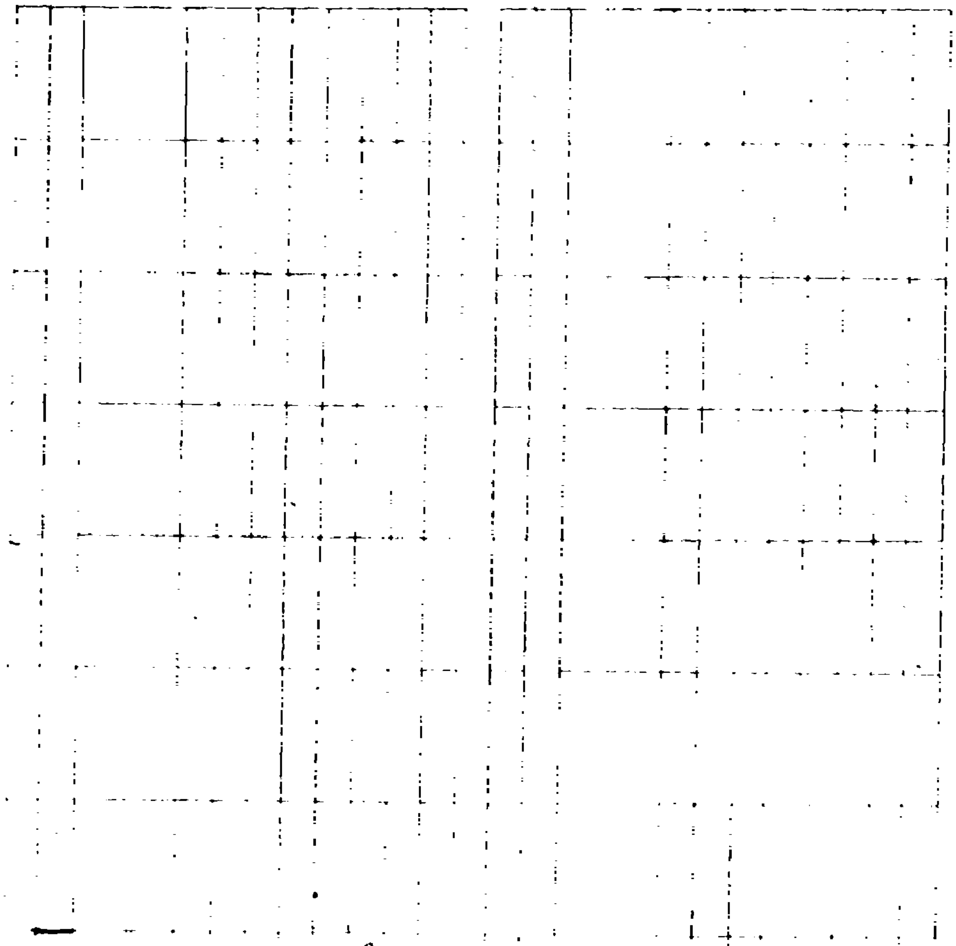
PART II - E - 2b POWER ON - FLAPS AND GEAR DOWN

$V\sqrt{\sigma}$ = 160 mph

T_c = .198

C. G. 6

$C_{M_{A-T}}$	$\frac{MAC_w}{\ell_{ht}}$	$C_{Z_{HT}}$	C_{Z_A}
-.3200	.2726	-.0872	-.1150
-.2643		-.0775	+.2841
-.2535		-.0691	+.6778
-.2273		-.0622	+1.0970
-.2028		-.0553	+1.5048
-.1774		-.0484	+1.70[illegible]



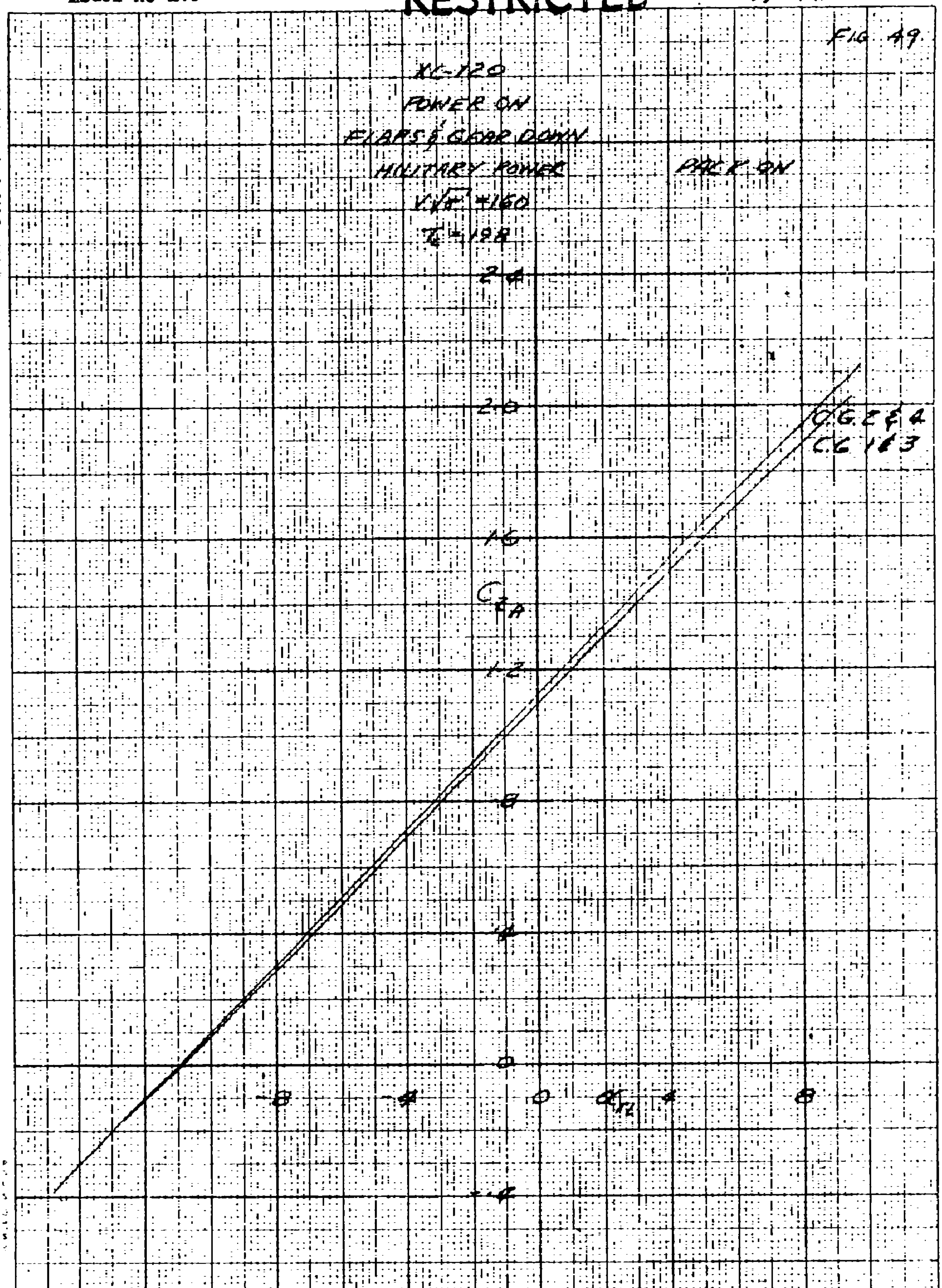


FIG 49
XC-120
POWER ON
FLAPS & GEAR DOWN
MILITARY POWER
PACK ON
2.4
2.0
C.G. 2 & 4
C.G. 1 & 3
1.6
1.2
.8
.4
0
-.4
-8
-4
0
4
8

PART II - F

F. COMPLETE AIRPLANE POLAR DIAGRAM

It is the purpose of this section to determine the polar diagram of the complete balanced airplane.

The total airplane lift and drag coefficients were determined by the addition of the lift and drag coefficients of each of the component parts. Lift and drag coefficients were obtained from reference (1).

Figure 50 presents the polar diagram for the complete balanced airplane for all center of gravity locations, flaps and gear up, power off.

PART II - F AIRPLANE LIFT COEFFICIENT - POWER OFF - FLAPS AND GEAR UP

PART II - F AIRPLANE LIFT COEFFICIENT - POWER OFF, FLAPS AND GEAR UP

$C_{L_{HT}} = \eta_{ht} \frac{S_{HT}}{S_W} \frac{q_{ht}}{q} = \eta_{ht} C_{L_{HT}} \frac{346.2}{1447.25} \times 1.00$

								CG ①		
$\alpha_{T.L.}$	C_{L_W}	C_{L_F}	C_{L_E}	C_{L_B} (ONE)	$2C_{L_B}$	$C_{L_{VT}}$	$C_{L_{A-T}}$	$\eta_{ht} C_{L_{HT}}$	$C_{L_{HT}}$	C_{L_A}
-12	-.3260	-.0282	-.0353	-.0290	-.0125	0	-.3738	-.153	-.0366	-.4104
-8	.0110	-.0475	-.0190	-.0130	-.0056	0	-.0136	-.128	-.0306	-.0442
-4	.3471	-.0161	-.0067	-.0025	-.0011	0	.3399	-.105	-.0251	+.3148
0	.6846	0	0	0	0	0	.6846	-.085	-.0203	+.6643
4	1.0210	.0167	.0067	.0025	.0011	0	1.0288	[illegible]	-.0153	1.0135
8	1.3580	.0475	.0190	.0130	.0056	0	1.3826	-.040	-.0096	1.3730

CG ②			CG ③			CG ④			CG ⑤		
$\eta_{ht} C_{L_{HT}}$	$C_{L_{HT}}$	C_{L_A}	$\eta_{ht} C_{L_{HT}}$	$C_{L_{HT}}$	C_{L_A}	$\eta_{ht} C_{L_{HT}}$	$C_{L_{HT}}$	C_{L_A}	$\eta_{ht} C_{L_{HT}}$	$C_{L_{HT}}$	C_{L_A}
-.202	-.0483	-.4221	-.161	-.0385	-.4123	-.210	-.0502	-.4240	-.172	-.0411	-.4149
-.134	-.0321	-.0457	-.121	-.0289	-.0425	-.127	-.0304	-.0440	-.127	-.0304	-.0440
-.068	-.0163	.3236	-.092	-.0220	.3179	-.055	-.0132	.3267	-.088	-.0210	.3189
-.008	-.0019	.6827	-.074	-.0177	.6669	.003	.0007	.6853	-.053	-.0127	.6719
.055	.0132	1.0420	-.063	-.0151	1.0137	.056	.0134	1.0422	-.018	-.0043	1.0245
.121	.0289	1.4115	-.055	-.0132	1.3694	.106	.0259	1.4080	.018	.0043	1.3869

PART II - F AIRPLANE DRAG COEFFICIENTS - POWER OFF
FLAPS AND GEAR UP

PART II - F AIRPLANE DRAG COEFFICIENTS - POWER OFF, FLAPS AND GEAR UP

α_{TL}	α_{ht}	C_{D_W}	C_{D_F}	C_{D_E}	C_{D_B} (ONE)	$2C_{D_B}$	$C_{D_{VT}}$	$2C_{D_{VT}}$	$C_{D_{A-T}}$	δ_e
-12	-4.40	.0120	.1403	.0169	.230	.0130	.0061	.000114	.0420	10.95
-8	-7.05	.0070	.1156	.0139	.205	.0116			.0326	7.66
-4	-4.70	.0120	.1037	.0125	.191	.0108			.0354	4.30
0	-2.35	.0270	.0991	.0120	.189	.0107			.0498	2.68
4	0	.0525	.1037	.0125	.191	.0108			.0759	-2.60
8	2.35	.0990	.1156	.0139	.205	.0116			.1246	-5.94

CG ①				CG ②				CG ③		
$C_{D_{HT}}$	$C_{D_{HT}}$	C_{D_A}	δ_e	$C_{D_{HT}}$	$C_{D_{HT}}$	C_{D_A}	δ_e	$C_{D_{HT}}$	$C_{D_{HT}}$	C_{D_A}
.0132	.0032	.0452	9.00	.0140	.0033	.0453	10.7	.0130	.0031	.0451
.0117	.0029	.0355	7.42	.0111	.0029	.0355	7.97	.0113	.0027	.0353
.0100	.0024	.0378	5.80	.0100	.0024	.0378	4.83	.0100	.0024	.0378
.0092	.0022	.0520	5.80	.0105	.0025	.0523	3.13	.0090	.0021	.0519
.0095	.0024	.0783	[illegible]23	.0092	.0022	.0781	-.33	.0082	.0020	.0779
.0103	.0025	.1271	[illegible]61	.0101	.0024	.1270	-6.55	.0110	.0026	.1272

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REPORT No. R107-014 FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION PAGES PAGE 195

MODEL XC-120 PREPARED BY CHECKED BY APPROVED BY

DATE March 15, 1949

Subject:— BASIC FLIGHT CRITERIA - PACK ON REVISED

PART II - F AIRPLANE DRAG COEFFICIENTS - POWER OFF FLAPS AND GEAR UP

PART II - F AIRPLANE DRAG COEFFICIENTS - POWER OFF, FLAPS AND GEAR UP

	CG ④				CG ⑤			
$\alpha_{T.L}$	δe	C_{DHT}	C_{DHT}	C_{DA}	δe	C_{DHT}	C_{DHT}	C_{DA}
-12	8.65	.0140	.0033	.0453	10.2	.0132	.0032	.0452
-8	7.73	.0117	.0029	.0355	7.72	.0117	.0028	.0354
-4	6.35	.0102	.0024	.0378	8.33	.0110	.0026	.0380
0	6.26	.0090	.0021	.0519	3.98	.0090	.0021	.0519
4	4.51	.0105	.0025	.0784	1.50	.0087	.0021	.0780
8	0	.0070	.0021	.1267	-3.57	.0092	.0022	.1266

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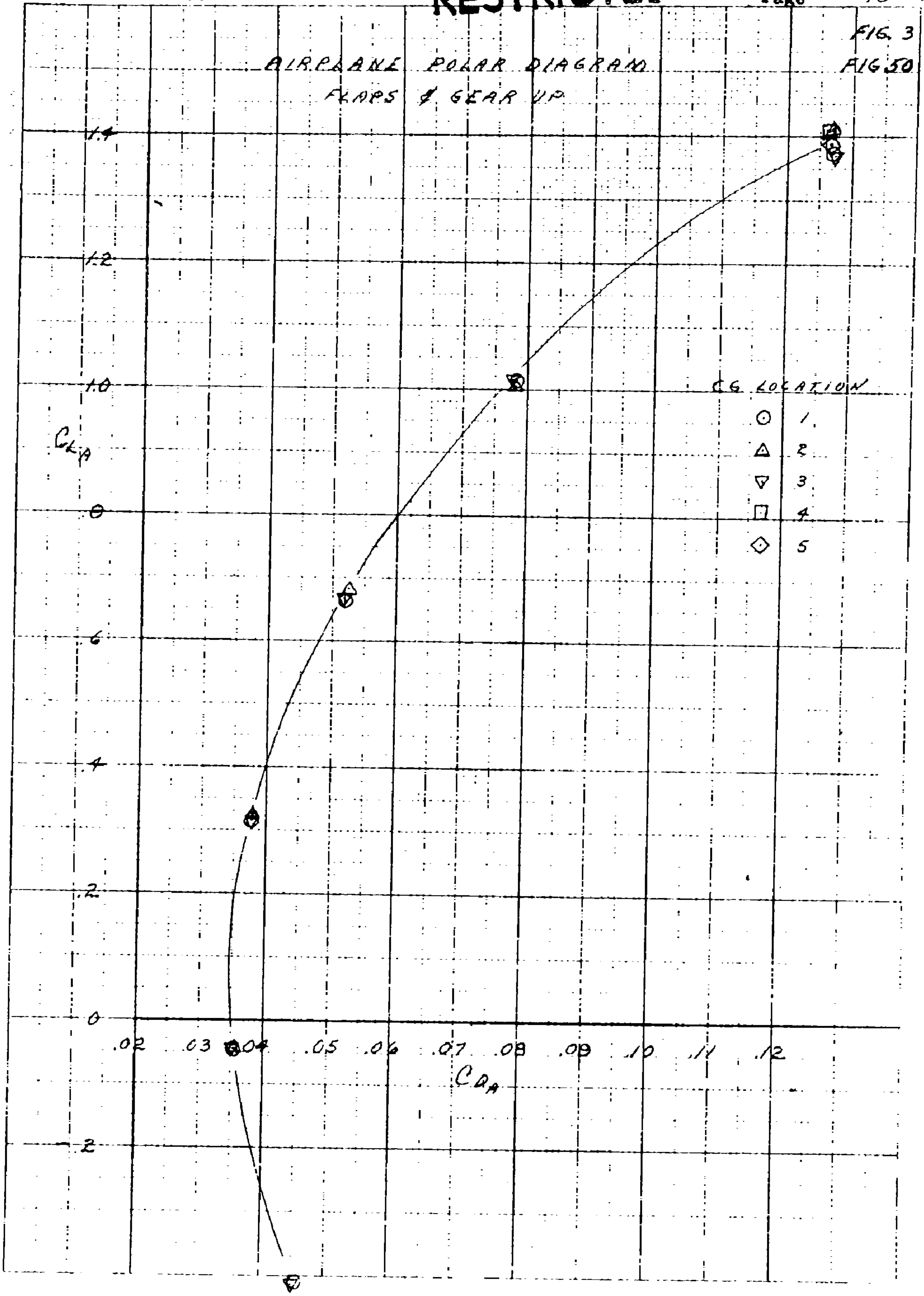
FIG. 3
FIG 50



PART II - G

G. V-n DIAGRAMS

1. LIMIT SPEEDS AND LOAD FACTORS

V-n diagrams for the subject airplane will be constructed in accordance with the flight conditions required by reference (2) using the procedure of reference (2).

DATA FOR CONSTRUCTION OF V-n DIAGRAM

Type of Diagram	B
Gross Wts. Considered	Maximum Alternate Gross Weights 74,000 lbs. Maximum Take-Off Gross Weight 74,000 lbs. Design Gross Weight 64,000 lbs. Maximum landing Gross Weight 60,000 lbs. Lightweight Alternate Condition 48,000 lbs. Minimum Flying Gross Weight 42,136 lbs.
Load Factors Symmetrical Assymmetrical	At 74,000 lbs. +2 and - 1 At all other weights, + 3 and -1.5 All weights at $V_R = 0.8\ V_H$, + 2 and - 1 with $\frac{P\ B_W}{2V} = .08$ at $V_D = 1.25\ V_H$, +2 with $\frac{P\ B_W}{2V} = .015$ Since flaps down is intended for landing only, no V-n Diagram is required for flapped conditions.

PART II-0-1

Altitudes Considered		Sea Level	18,000 Ft. (Highest M.N.)
V_H True	MPH	250	270
V_R True	MPH	200	216
V_D True	MPH	313	338
V_T True	MPH	502	522

Converting the above speeds at sea level and 18,000 feet to indicated air speed, it is immediately apparent that only sea level conditions will be required to be investigated. Therefore all future reference to V-n diagrams will refer to sea level conditions only and no V-n diagrams are necessary for 18,000 feet altitude.

Terminal Velocity for the preceding table was calculated by the method outlined in reference (2)

$$V_T = .6818 \; M_T \, (a)$$

where a = speed of sound at altitude of V-n diagram, in ft/sec.

M_{CR} = critical M.N. of wing root airfoil section; from reference (11) for 2418 at $C_L = 0 = .605$

C_{D_F} = equivalent airplane parasite drag coefficient at zero lift = .040 from performance calculations in reference (10).

S_W = 1447.25 sq. ft.

$\frac{P}{P_O} = \frac{A^2 \sigma}{A_O}$ for sea level = 1.0

18,000 feet = 0.4992

PART II - G - 1 (Continued)

Then the constant for entering figure 4 of reference (2)

- 2 B

$$K = \frac{W/S_W}{1481 \frac{P}{P_0} C_{D_f} M^2_{CR}} = \frac{W}{(1447.25)(1481)(.040)(.366)\left(\frac{P}{P_0}\right)}$$

$$= \left[\frac{W}{31,600}\right] \quad \text{at sea level}$$

$$= \left[\frac{W}{15,780}\right] \quad \text{at 18,000 feet}$$

GROSS WEIGHT	SEA LEVEL					18,000 FEET				
	K	M_T/M_{CR}	M_T	a ft/sec	V_T mph	K	M_T/M_{CR}	M_T	a	V_T mph
74,000	2.34	1.095	.660	1116.19	502	4.70	1.215	.732	1044.32	522
64,000	2.03	1.073	.647		493	4.06	1.187	.720		513

In order to simplify the V-n diagrams and the calculations, select the maximum terminal velocities of 502 and 522 mph as applicable to all weights.

PART II - G - 2a

2. MAXIMUM LIFT LINES

Maximum lift lines for the V-n diagrams are determined in accordance with reference (2)

a. FLAPS AND GEAR UP - POWER OFF

$$n = \frac{C_{Z_A\,MAX}\; q}{W/S_W}$$

Gross Weight	W/S.
74,000 lbs.	51.10
64,000 lbs.	44.25
42,136 lbs.	29.11

α T.L for $+ C_{L_W\,MAX}$ = 8.55° α_{W_C} = 15.55°

α T.L. for $- C_{L_W\,MAX}$ = -14.88° α_{W_C} = -7.88°

C.G. Location	$+ C_{L\,MAX}$	$- C_{L\,MAX}$	$d\,C_{Z_A}/d\alpha_{T.L.}$
1	1.420	-.6645	.0892
2	1.475	-.698	.0920
3	1.420	-.6645	.0892
4	1.475	-.698	.0920
5	1.45	-.690	.0910

In order to simplify the construction of V-n diagrams, the maximum positive and negative values of C_{Z_A} and the maximum value of $dC_{Z_A}/d\alpha_{T.L.}$ will be considered as applicable to all C.G. locations.

PART II - G - 2b

Maximum n's for C.G. Location 1,2,3,4, and 5

POWER OFF

V	q	$(C_{Z_{MAX}})(q)$	n	n	n
75	14.40	21.3	.417	.482	.620
		-10.05	-.196	-.228	-.345
100	25.56	37.80	.733	.855	1.10
		-17.83	-.348	-.404	-.611
150	57.50	85.20	1.66	1.93	2.48
		-40.10	-.803	-.910	-1.375
200	102.10	151.00	2.95	3.420	4.40
		-71.40	-1.393	-1.615	-2.45
250	159.0	235.50	4.60	5.320	6.85
		-111.00	-2.17	-2.51	-3.80

b. FLAPS AND GEAR UP - POWER ON

The "Power On" maximum lift lines are complicated by the effects of Power on the normal force coefficient and must be considered at various speeds.

$V\sqrt{\sigma}$ = 100 mph, T_c = .639

C.G. Location	$+C_{Z_A}$ max	$-C_{Z_A}$ max	$dC_{Z_A}/d\alpha_{T\ L}$
1	1.65	-.721	.1021
2	1.705	-.748	.1049
3	1.639	-.750	.1020
4	1.681	-.773	.1048
5	1.670	-.750	.1031

$V\sqrt{\sigma}$ = 160, T_c = .198

C.G. Location	$+C_{Z_A}$ max	$-C_{Z_A}$ max	$dC_{Z_A}/d\alpha_{T\ L}$
1	1.532	-.713	.0958
2	1.580	-.728	.0987
3	1.521	-.730	.0959
4	1.562	-.740	.0986
5	1.549	-.730	.0969

PART II - G- 2b

C.G. Location	$+C_{Z_A\ max}$	$-C_{Z_A\ max}$	$dC_{Z_A}/d_{\alpha\ TL}$
$V\sqrt{\sigma}$ = 185 T_c = .132			
1	1.500	-.719	.0947
2	1.540	-.739	.0973
3	1.498	-.719	.0946
4	1.535	-.739	.0972
5	1.515	-.729	.0956
$V\sqrt{\sigma}$ = 250 T_c = .056			
1	1.472	-.710	.0930
2	1.512	-.730	.0956
3	1.465	-.709	.0929
4	1.508	-.728	.0955
5	1.481	-.718	.0939

In order to simplify construction of the V-n diagram, the maximum positive and negative values of C_{Z_A} and $d\,C_{Z_A}/d_{\alpha\ TL}$ will be considered as applicable to all C.G. locations for any given speed.

$$n = \frac{C_{Z_A\ max}\ q}{W/S_W}$$

Maximum n's for C.G. Location 1,2,3,4, and 5

POWER ON

$V\sqrt{\sigma}$ mph	q	$(C_{Z_A max})$ q	n 74,000 lbs.	n 64,000 lbs.	n 42,136 lbs.
100	25.6	43.6 -19.8	.854 -.387	.987 -.447	1.495 -.679
160	65.5	103.6 -45.5	2.022 -.948	2.340 -1.095	3.55 -1.661
185	87.5	134.5 -63.0	2.623 -1.231	3.041 -1.423	4.61 -2.16

PART II - G - 3a

3. GUST LOAD FACTORS

Gust load factors are determined in accordance with reference (2).

(a) FLAPS AND GEAR UP - POWER OFF

$$n = 1 \pm \frac{5.0\ K \dfrac{d\ C_{Z_A}}{d\ \alpha_{TL}} V\sqrt{\sigma}}{W/S_W}$$

Gross Weight	W/S	K	5.0 K/W/S
74,000 lbs.	51.10	.599	.0585
64,000 lbs.	44.25	.589	.0665
60,000 lbs.	41.50	.581	.0700
48,000 lbs.	33.20	.567	.0853
42,136 lbs.	29.11	.559	.0959

In order to simplify construction of V-n diagrams, the maximum value of $d\ C_{Z_A}/d\alpha_{TL}$ shall be considered applicable to all C.G. locations so:

$$d\ C_{Z_A}/d\ \alpha_{TL} = .0920$$

Gross Weight	n
74,000 lbs.	$1 \pm .0585 \times .0920 \times V\sqrt{\sigma}$
64,000 lbs.	$1 \pm .0665 \times .0920 \times V\sqrt{\sigma}$
42,136 lbs.	$1 \pm .0959 \times .0920 \times V\sqrt{\sigma}$

PART II - G - 3a.

n - GUST LOADS - POWER OFF

SPEED $V\sqrt{\sigma}$ mph	n 74,000 lbs.	n 64,000 lbs.	n 42,136 lbs.
100	1.540 .460	1.615 .385	1.888 .112
200	2.080 -.080	2.230 -.230	2.775 -.775
250	2.350 -.350	2.530 -.530	3.220 -1.220

As the positive gust loads are only critical for two of the gross weight conditions, namely 74,000 lbs. (at $V\sqrt{\sigma}$ = 200 and 250 n $>$ 2.0) and 42,136 lbs. (at $V\sqrt{\sigma}$ = 250 n $>$ 3.0), gust loads for these gross weights only are shown on the V-n diagram.

Negative gust loads are not critical.

PART II - G - 3b.

b. FLAPS AND GEAR UP - POWER ON

Again for simplification, the maximum value of $dC_{Z_A}/d\alpha_{TL}$ as determined in Part II-G-2b shall be considered as as applicable to all C.G. locations for any given speed.

$$n = 1 \pm \frac{5.0\,K\,\frac{dC_{Z_A}}{d\alpha_{TL}}\,V\sqrt{\sigma}}{W/S_W}$$

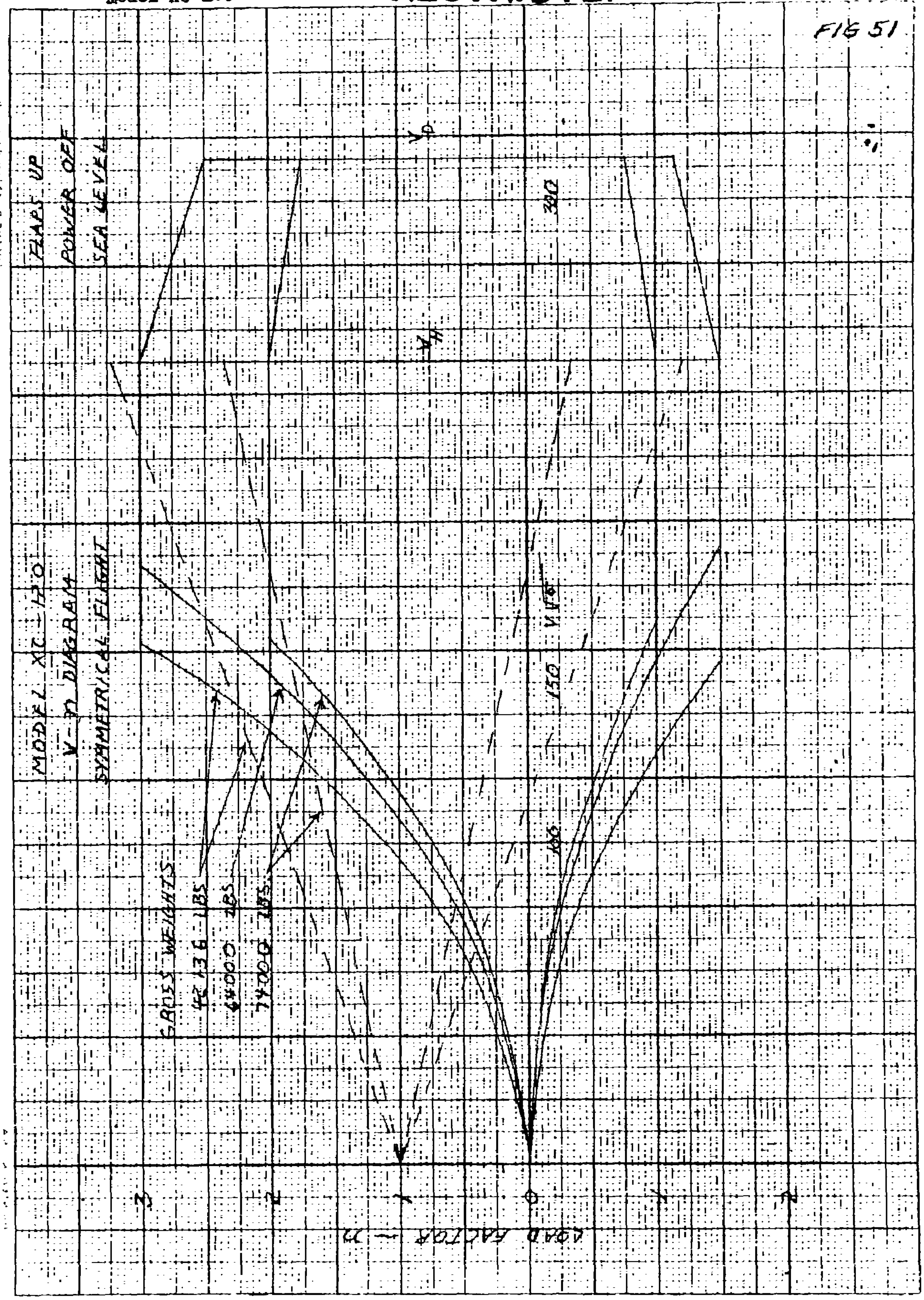
PART II - G - 3

Gross Weight	n
74,000 lbs.	$1 \pm .0585 \times \frac{d\,C_{Z_A}}{d\,\alpha_{TL}} \times V\sqrt{\sigma}$
64,000 lbs.	$1 \pm .0665 \times \frac{d\,C_{Z_A}}{d\,\alpha_{TL}} \times V\sqrt{\sigma}$
42,136 lbs.	$1 \pm .0959 \times \frac{d\,C_{Z_A}}{d\,\alpha_{TL}} \times V\sqrt{\sigma}$

n GUST LOADS - POWER ON

$V\sqrt{\sigma}$ mph	$\frac{d\,C_{Z_A}}{d\,\alpha_{TL}}$	74,000 lbs.	64,000 lbs.	42,136 lbs.
100	.1049	1.614 .386	1.697 .303	2.006 -.006
160	.0987	1.924 .076	2.050 -.050	2.515 -.515
185	.0973	2.052 -.052	2.198 -.198	2.727 -.727
250	.0956	2.400 -.400	2.568 -.568	3.295 -1.28

FIG 51


FLAPS UP
POWER OFF
SEA LEVEL
MODEL XC-120
V-n DIAGRAM
SYMMETRICAL FLIGHT
GROSS WEIGHTS
64000 LBS
74000 LBS
300
150
100
LOAD FACTOR — n
3
2
1
0
1
2

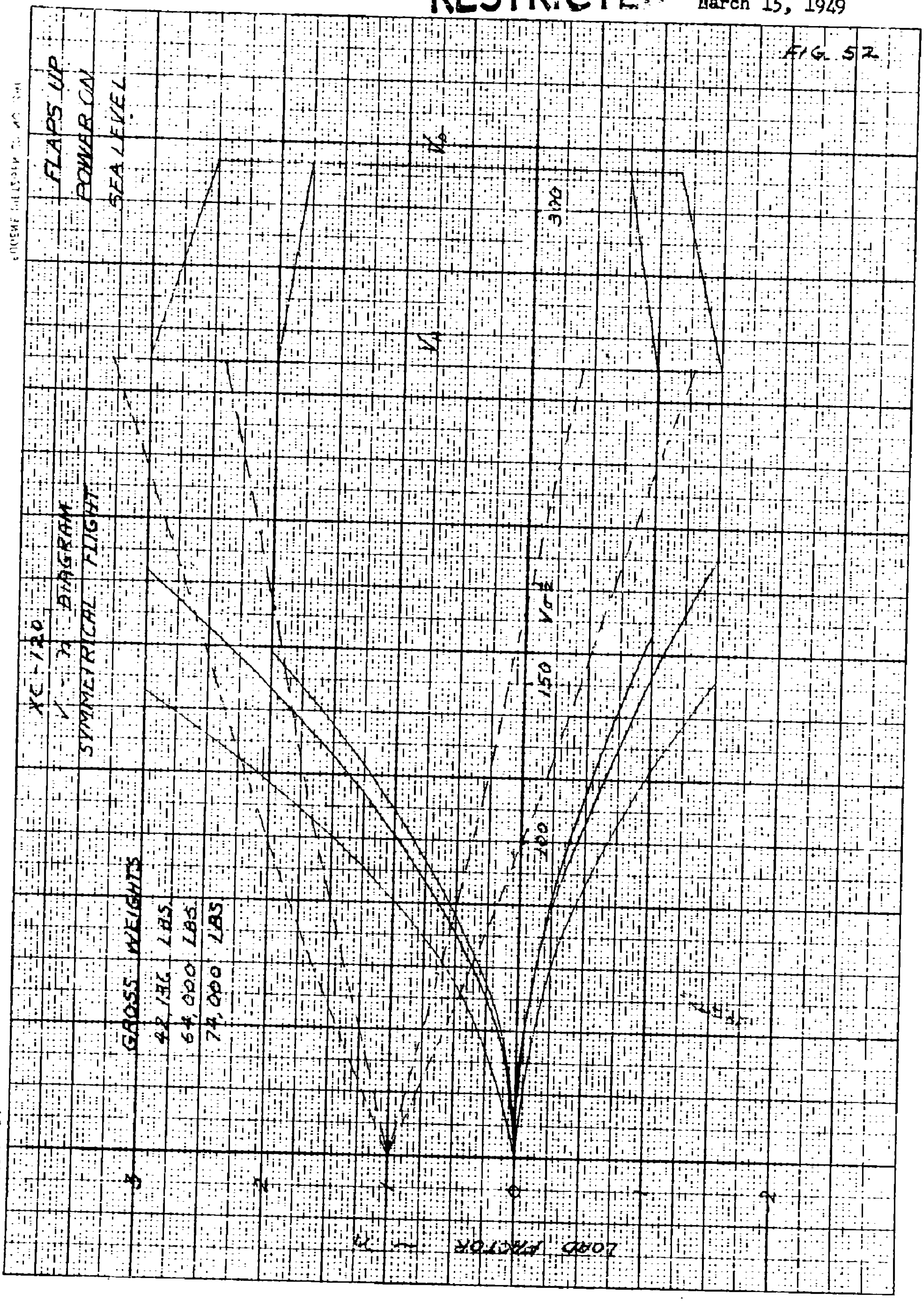


FIG. 52
XC-120
n - V DIAGRAM
SYMMETRICAL FLIGHT
FLAPS UP
POWER ON
SEA LEVEL
GROSS WEIGHTS
42,136 LBS.
64,000 LBS.
74,000 LBS.
LOAD FACTOR n
100
150
300

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PART II - G - 3c.

c. FLAPS AND GEAR DOWN - POWER OFF

As the limiting speed for the flapped wing is 160 mph; V-n diagrams will show gust loads up to that speed.

$$n = 1 \pm \frac{5.0\ K \frac{d\ C_{Z_A}}{d\ \alpha_{TL}}\ V\sqrt{\sigma}}{W/S_W}$$

Gross Weight	W/S_W	K	$5.0\ K/(W/S_W)$
74,000 lbs.	51.10	.599	.0585
64,000 lbs.	44.25	.589	.0665
42,136 lbs.	29.11	.559	.0959

Values of $d\ C_{Z_A}/d\ \alpha_{TL}$ are now determined.

C.G. Location	$\frac{d\ C_{Z_A}}{d\ \alpha_{TL}}$
1	.0913
2	.0938
3	.0910
4	.0936
5	.0918

Again the maximum value of $d\ C_{Z_A}/d\ \alpha_{TL}$ will be used.

$$d\ C_{Z_A}/d\ \alpha_{TL} = .0938$$

Gross Weight	n
74,000 lbs.	$1 \pm .0585 \times .0938 \times V\sqrt{\sigma}$
64,000 lbs.	$1 \pm .0665 \times .0938 \times V\sqrt{\sigma}$
42,136 lbs.	$1 \pm .0959 \times .0938 \times V\sqrt{\sigma}$

PART II - G - 3c.

n - GUST LOADS - POWER OFF

SPEED

$V\sqrt{\sigma}$ mph	n 74,000 lbs.	n 64,000 lbs.	n 43,136 lbs.
100	1.548 .452	1.623 .377	1.90 .10
160	1.978 .122	2.00 0	2.44 -.44

PART II - G - 3d

d. FLAPS AND GEAR DOWN - POWER ON

$$n = 1 \pm \frac{5.0 \quad K \quad \frac{d\,C_{Z_A}}{d\,\alpha_{TL}} \quad V\sqrt{\sigma}}{W/S_w}$$

$V\sqrt{\sigma}$ = 100 mph T_c = .639

C.G. Location	$\frac{d\,C_{Z_A}}{d\,\alpha_{TL}}$
1	.1110
2	.1160
3	.1130
4	.1160
6	.1140

$V\sqrt{\sigma}$ = 160 mph T_c = .198

1	.1004
2	.1034
3	.1000
4	.1030
6	.1010

Maximum values of $dC_{Z_A}/d\,\alpha_{TL}$ shall be considered as applicable to all C.G. Locations for any given speed.

Gross Weight	n
74,000	$1 \pm (.0585) \times \frac{dC_{Z_A}}{d\alpha_{TL}} \times V\sqrt{\sigma}$
64,000	$1 \pm (.0665) \times \frac{dC_{Z_A}}{d\alpha_{TL}} \times V\sqrt{\sigma}$
42,136	$1 \pm (.0959) \times \frac{dC_{Z_A}}{d\alpha_{TL}} \times V\sqrt{\sigma}$

n - GUST LOADS - POWER ON

$V\sqrt{\sigma}$	42,136 lbs.	64,000 lbs.	74,000 lbs.
100	2.11 -.11	1.774 .226	1.68 .32
160	2.585 -.585	2.11 -.11	1.97 .03

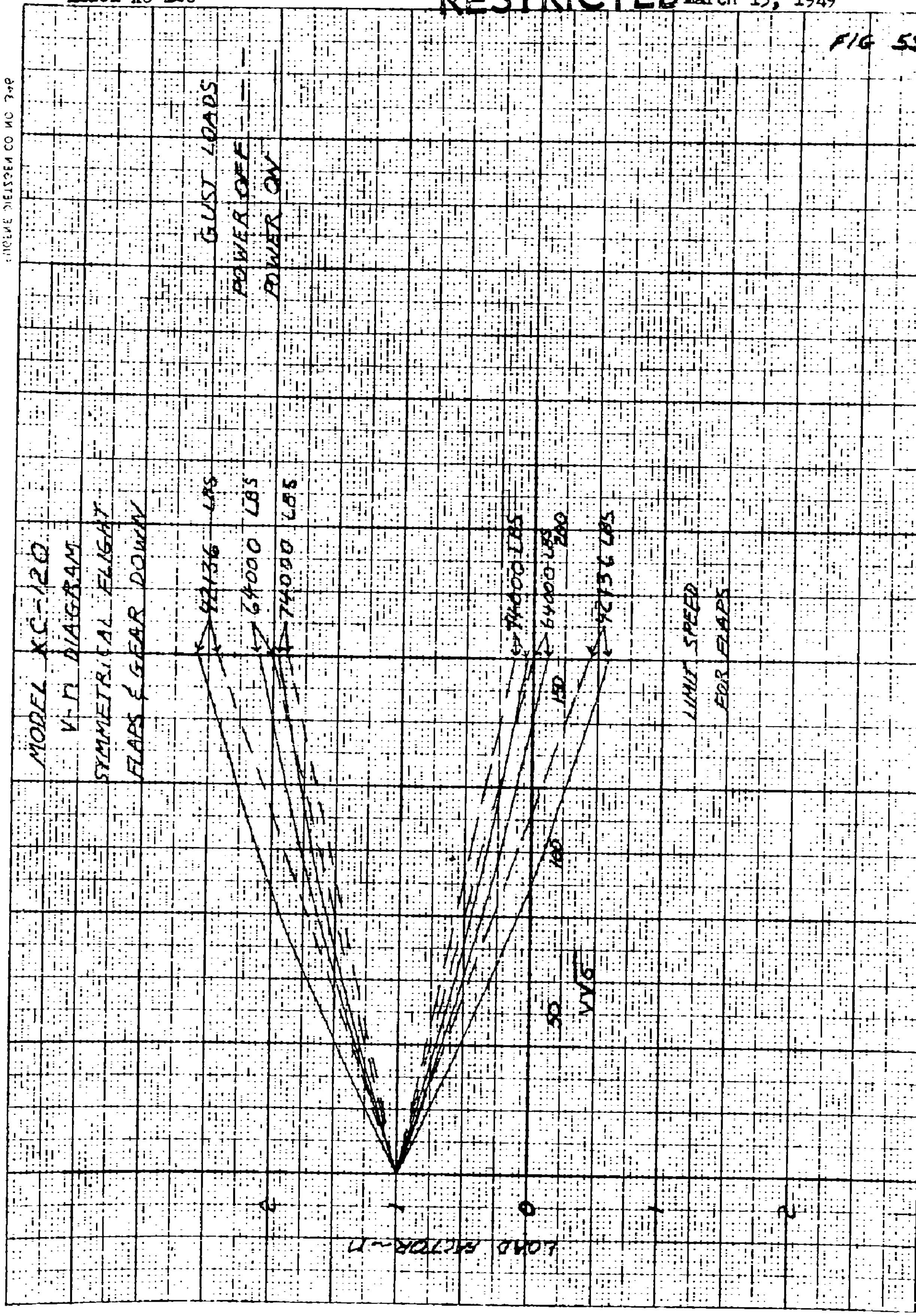
MODEL XC-120
V-n DIAGRAM
SYMMETRICAL FLIGHT
FLAPS & GEAR DOWN
GUST LOADS
POWER OFF
POWER ON
42136 LBS
64000 LBS
74000 LBS
74000 LBS
64000 LBS
42136 LBS
LIMIT SPEED
FOR FLAPS
50
100
150
200
LOAD FACTOR ~ n
0
1
2

FIG 53

PART II - H

H. COMPLETE AIRPLANE COMPONENT LOADS - BALANCED CONDITION

The loads on the component parts of the airplane have been determined for the balanced condition for all six points on the V-n diagram for the gross weight, load factor, and C.G. location that gives the maximum load on each of the component parts. For design gross weight this corresponds to the C.G. position which gives the maximum α_{TL} in combination with the maximum balancing tail load for positive and negative low angles of attack. For positive and negative high angle of attack it corresponds to the C.G. location which gives the maximum dynamic pressure, which is obtained with minimum values of CZ_A, angle of attack being fixed.

The C.G. locations so selected yield the most critical stress conditions for the design gross weight. In addition, the maximum alternate and minimum flying weight gust conditions at $PLAA_{VH}$ are checked for possibly critical stress conditions.

All conditions noted above are checked for power off and power on and the calculations are presented in the following tables.

All coefficients are based on wing area, mean aerodynamic chord and free stream dynamic pressure.

Part II - H -1a.

1a. FLAPS AND GEAR UP - POWER OFF

The tables immediately following present the calculations of the maximum loads on each of the component parts for flaps and gear up, power off.

RESTRICTED

REPORT NO R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION	PAGES	PAGE 213
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY
Subject BASIC FLIGHT CRITERIA - PACK ON			DATE March 15, 1949 REVISED

ANALYSIS OF AIRPLANE COMPONENT LOADS
SYMMETRICAL FLIGHT WITHOUT PITCHING ACCELERATIONS
Condition FLAPS & GEAR UP — POWER OFF

V-n Diagram	PHAA	NHAA	PLAA-V_H	PLAA-V_D	NLAA-V_H	NLAA-V_D
C. G. Location	3	1	1	1	4	2
Gross Weight lbs.	64,000	64,000	64,000	64,000	64,000	64,000
AIRPLANE						
$n Z_A$	3.00	-1.50	3.00	2.51	-1.50	-1.14
W/S_W, #/sq. ft.	44.25	44.25	44.25	44.25	44.25	44.25
$n_A W/S_W$	132.75	-66.375	132.75	111.07	-66.375	-50.445
C_{Z_A}	1.4196	-.6556	.8291	.4433	-.4148	-.2013
$q = n_A(W/S_W)/C_{Z_A}$, #/sq. ft.	93.51	101.2	160	250	160	250
$q/(W/S_W)$	2.1132	2.2870	36158	5.6497	3.6158	5.6497
$V\sigma^{1/2} = \sqrt{391\, q}$, mph	191.2	198.9	250	313	250	313
α T.L., degrees	8.55	-14.88	1.85	-2.43	-11.92	-9.56
C_{X_A}	-.07404	-.10040	.03796	.06778	-.02700	.01564
$n X_A = C_{X_A}\, q/(W/S_W)$	-.1565	-.2296	.1373	.3829	-.0976	.0884
C_{L_A}	1.41599	-.67454	.82461	.45282	-.41556	-.19167
C_{D_A}	.13698	.06188	.06320	.04411	.04621	.03863
WING						
C_{L_W}	1.4010	-.5670	.8403	.4799	-.3191	-.1204
C_{D_W}	.1073	.0215	.0372	.0166	.0120	.0075
C_{Z_W}	1.4014	-.5425	.8411	.4788	-.3147	-.1199
C_{X_W}	-.1021	-.1248	.0101	.0370	-.0543	-.0126
$n Z_W = C_{Z_W}\, q/(W/S_W)$	2.9614	-1.2407	3.0412	2.7051	-1.1379	-.6774
$n X_W = C_{X_W}\, q/(W/S_W)$	-.2155	-.2854	.0365	.2090	-.1963	-.0712
$Z_W = n Z_W\, W$, lbs.	189,530	-79,405	194,637	173,126	-72,872	-43,354
$X_W = n X_W\, W$, lbs.	-13,792	-18,266	2336	13,376	-12,574	-4557
C_{M_W} (a.c.)	-.0630	-.0630	-.0630	-.0630	-.0630	-.0630
$M_W = q\, S_W\, MAC_W\, C_{M_W}$, ft. lbs.	-119,530	-129,358	-204,528	-319,575	-204,528	-319,575
ΔC_{M_A} due to wing	-.1765	-.0525	-.10847	-.08389	-.09552	-.07028

RESTRICTED

REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION		PAGES	PAGE 214
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY	
Subject: BASIC FLIGHT CRITERIA - PACK ON			DATE March 15, 1949	REVISED

V-n Diagram	PHAA	NHAA	PLAA-V_N	PLAA-V_D	NLAA-V_N	NLAA-V_D
C. G. Location	3	1	1	1	4	2
Gross Weight lbs.	69,000	69,000	64,000	59,000	64,000	64,000
α TL degrees	8.55	-14.88	1.85	-2.43	-11.92	-9.56
FUSELAGE						
C_{L_F}	.0212	-.0485	.0023	-.0036	-.0348	-.0250
C_{D_F}	.0144	.0207	.0122	.0123	.0169	.0150
C_{Z_F}	.0231	-.0524	.0027	-.0039	-.0375	-.0270
C_{X_F}	.0110	.0065	.0120	.0121	.0093	.0106
$\eta\, Z_F = C_{Z_F}\, q/(W/S_W)$	.0488	-.1198	.0098	-.0220	-.1356	-.1525
$\eta\, X_F = C_{X_F}\, q/(W/S_W)$	.0232	.0149	.0434	.0684	.0336	.0599
$Z_F = \eta\, Z_F\, W$, lbs.	3126	-7670	627	-1408	-8683	-9760
$X_F = \eta\, X_F\, W$, lbs.	1485	954	2778	4378	2150	3834
C_{M_F} (.25 L_F)	.0251	-.0308	.00589	-.00774	-.03111	-.02734
$M_F = q\, S_W\, MAC_W\, C_{M_F}$, ft. lbs.	47,622	-63,242	19,121	-39,261	-100,996	-138,682
$\Delta C_{M_A}'$ due to fuselage	.0485	-.0844	.00667	-.01359	-.07225	-.05882
BOOM (ONE)						
C_{L_B}	.00322	-.00913	.00011	-.00017	-.0061	-.00406
C_{D_B}	.00574	.00733	.0053	.0053	.0064	.0060
C_{Z_B}	.0041	-.0108	.00026	-.00039	-.0074	-.0050
C_{X_B}	.0053	.0045	.0053	.0053	.00506	.00526
$\eta\, Z_B = C_{Z_B}\, q/(W/S_W)$	.00845	-.0247	.00094	-.0022	-.0268	-.0282
$\eta\, X_B = C_{X_B}\, q/(W/S_W)$	.0112	.0103	.0192	.0299	.0183	.0297
$Z_B = \eta\, Z_B\, W$, lbs.	555	-1581	602	-141	-1715	-1805
$X_B = \eta\, X_B\, W$, lbs.	717	659	1229	1914	1171	1901
C_{M_B} (.25 L_B)	.0406	-.0092	.0175	.0061	-.0067	-.0044
$M_B = q\, S_W\, MAC_W\, C_{M_B}$, ft. lbs.	77,033	-18,890	56,812	30,942	-21,751	-22,319
ΔC_{M_A} due to one boom	.0404	-.0069	.01663	.00691	-.00509	-.00404

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REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION		PAGES	PAGE 215
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY	
Subject BASIC FLIGHT CRITERIA - PACK ON			DATE March 15, 1949 REVISED	

V-n Diagram	PHAA	NHAA	PLAA-V_N	PLAA-V_D	NLAA-V_N	NLAA-V_D
C. G. Location	3	1	1	1	4	2
Gross Weight lbs.	64,000	64,000	64,000	64,000	64,000	64,000
α_{TL} degrees	8.55	-14.88	1.85	-2.43	-11.92	-9.56
VERTICAL TAIL (ONE)						
$C_{L_{VT}}$	0	0	0	0	0	0
$C_{D_{VT}}$	.00057	.00057	.00057	.00057	.00057	.00057
$C_{Z_{VT}}$	0	0	0	0	0	0
$C_{X_{VT}}$	.00057	.00057	.00057	.00057	.00057	.00057
$\eta X_{VT} = C_{X_{VT}} q/(W/S_W)$	.0012	.0013	.0021	.0032	.0021	.0032
$X_{VT} = \eta X_{VT} W$, lbs.	76.8	83.2	134.4	204.8	134.4	204.8
ΔC_{M_A} due to one tail	.00027	.000165	.000165	.000165	.00027	.000165
HORIZONTAL TAIL						
α_{ht}	2.68	-11.08	-1.27	-3.82	-9.38	-7.99
q_{ht}/q	1.00	1.00	1.00	1.00	1.00	1.00
δ_e	-7.05	13.34	-0.77	3.05	8.75	8.13
$\eta_{ht} C_{L_{HT}}$	-.01265	-.04078	-.01821	-.02314	-.04946	-.03815
$C_{D_{HT}}$	.00266	.00388	.00206	.00347	.00337	.00299
$C_{Z_{HT}}$	-.01265	-.04078	-.01821	-.02314	-.04946	-.03815
$C_{X_{HT}}$	.00266	.00388	.00206	.00347	.00337	.00299
$\eta Z_{HT} = C_{Z_{HT}} q/(W/S_W)$	-.02673	-.09326	-.06584	-.13073	-.17878	-.2155
$\eta X_{HT} = C_{X_{HT}} q/(W/S_W)$	.00562	.00887	.00745	.01960	.01219	.01689
$Z_{HT} = \eta Z_{HT} W$, lbs.	-1712	-5975	-4213	-8365	-11,442	-13,789
$X_{HT} = \eta X_{HT} W$, lbs.	360	568	477	1254	780	1081
$C_{M_{HT}}$ (a. c.)	.00862	-.01630	.00094	-.00373	-.01069	-.00993
$M_{HT} = q S_W MAC_W C_{M_{HT}}$, ft lbs	16,355	-33,470	3052	-18,920	-39,704	-59,370
SUMMATION						
Z_A, lbs.	192,010	-96,212	191,171	163,071	-96,427	-70,513
X_A, lbs.	-10,359	-15,260	8,317	23,296	-7,034	4,570
$C_{M_{A-T}}$	-.0467	-.1504	-.06715	-.08533	-.17741	-.13684

ANALYSIS OF AIRPLANE COMPONENT LOADS
SYMMETRICAL FLIGHT WITHOUT PITCHING ACCELERATIONS
Condition FLAPS & GEAR UP ~ POWER OFF GUST CONDITIONS

V-n Diagram	PLAA - V_H	PLAA - V_H				
C. G. Location	3	5				
Gross Weight lbs.	74,000	42,136				
AIRPLANE						
$n\ Z_A$	2.35	3.22				
W/S_W, #/sq. ft.	51.13	29.12				
$n\ A\ W/S_W$	120.156	93.734				
C_{Z_A}	.7510	.5858				
$q = nA(W/S_W)/C_{Z_A}$, #/sq. ft.	160	160				
$q/(W/S_W)$	3.129	5.496				
$V\sigma' = 1391\ q$, mph	250	250				
α T.L., degrees	.98	-.975				
C_{X_A}	.04500	.05843				
$n\ X_A = C_{X_A}\ q/(W/S_W)$	.14081	.32108				
C_{L_A}	.75099	.58659				
C_{D_A}	.05821	.04821				
WING						
C_{L_W}	.76706	.60245				
C_{D_W}	.0323	.0221				
C_{Z_W}	.76747	.60197				
C_{X_W}	.01915	.03237				
$n\ Z_W = C_{Z_W}\ q/(W/S_W)$	2.4016	3.3081				
$n\ X_W = C_{X_W}\ q/(W/S_W)$	.05991	.17789				
$Z_W = n\ Z_W\ W$, lbs.	177,715	139,391				
$X_W = n\ X_W\ W$, lbs.	4,434	7,496				
C_{M_W} (a.c.)	-.0630	-.0630				
$M_W = q\ S_W\ MAC_W\ C_{M_W}$, ft. lbs.	-204,528	-204,528				
ΔC_{M_A} due to wing	-.09937	-.06759				

RESTRICTED

REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION		PAGES	PAGE 217
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY	
Subject:— BASIC FLIGHT CRITERIA - PACK ON			DATE March 15, 1949 REVISED	

V-n Diagram	PLAA - V_H	PLAA - V_H				
C. G. Location	3	5				
Gross Weight lbs.	74,000	42,136				
α TL degrees	.96	-.975				
FUSELAGE						
C_{L_F}	.00100	-.00098				
C_{D_F}	.01209	.01208				
C_{Z_F}	.00124	-.00120				
C_{X_F}	.01201	.01201				
$\eta Z_F = C_{Z_F}\, q/(W/S_W)$	.00388	-.00660				
$\eta X_F = C_{X_F}\, q/(W/S_W)$	.03759	.06598				
$Z_F = \eta Z_F W$, lbs.	287	-278				
$X_F = \eta X_F W$, lbs.	2782	2780				
C_{M_F} (.25 L_F)	.00306	-.00303				
$M_F = q\, S_W\, MAC_W\, C_{M_F}$, ft. lbs.	9950	-9846				
ΔC_{M_A} due to fuselage	.004459	-.00582				
BOOM (ONE)						
C_{L_B}	.00004	-.00004				
C_{D_B}	.00531	+.00531				
C_{Z_B}	.00009	-.00009				
C_{X_B}	.00532	.00532				
$\eta Z_B = C_{Z_B}\, q/(W/S_W)$	.00028	-.00047				
$\eta X_B = C_{X_B}\, q/(W/S_W)$	.01665	.02924				
$Z_B = \eta Z_B W$, lbs.	21	-20				
$X_B = \eta X_B W$, lbs.	1232	1232				
C_{M_B} (.25 L_B)	.01481	.00956				
$M_B = q\, S_W\, MAC_W\, C_{M_B}$, ft. lbs.	48,081	31,031				
ΔC_{M_A} due to one boom	.01548	.00946				

RESTRICTED

REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION	PAGES / PAGE 218
MODEL XC-120 / PREPARED BY	CHECKED BY	APPROVED BY
Subject: BASIC FLIGHT CRITERIA - PACK ON		DATE March 15, 1949 / REVISED

V-n Diagram	PLAA-V_H	PLAA-V_H				
C. G. Location	3	5				
Gross Weight lbs.	74,000	42,136				
α_{TL} degrees	.98	-.975				
VERTICAL TAIL (ONE)						
$C_{L_{VT}}$	0	0				
$C_{D_{VT}}$	.00057	.00057				
$C_{Z_{VT}}$	0	0				
$C_{X_{VT}}$	.00057	.00057				
$\eta X_{VT} = C_{X_{VT}} q/(W/S_w)$	.00178	.00313				
$X_{VT} = \eta X_{VT}$ W, lbs.	132	132				
ΔC_{M_A} due to one tail	.00027	.000185				
HORIZONTAL TAIL						
α_{ht}	-1.80	-2.55				
q_{ht}/q	1.00	1.00				
δ_e	.376	2.877				
$\eta_{ht} C_{L_{HT}}$	-.01715	-.01481				
$C_{D_{HT}}$	.00206	.00227				
$C_{Z_{HT}}$	-.01715	-.01481				
$C_{X_{HT}}$	.00206	.00227				
$\eta Z_{HT} = C_{Z_{HT}} q/(W/S_w)$	-.05366	-.08140				
$\eta X_{HT} = C_{X_{HT}} q/(W/S_w)$	.00645	.01248				
$Z_{HT} = \eta Z_{HT}$ W, lbs.	-3971	-3430				
$X_{HT} = \eta X_{HT}$ W, lbs.	477	526				
$C_{M_{HT}}$ (a. c.)	-.00046	-.00352				
$M_{HT} = q S_w MAC_w C_{M_{HT}}$, ft lbs	-1492	-11,414				
SUMMATION						
Z_A, lbs.	174,073	135,961				
X_A, lbs.	10,421	13,530				
$C_{M_{A-T}}$	-.06329	-.05411				

PART II - H -1b.

1b. FLAPS AND GEAR UP - POWER ON

The following tables present the calculations necessary to show the effects of power on the loads of the component parts.

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

DATE March 15, 1949
REVISED

Subject:— BASIC FLIGHT CRITERIA – PACK ON

ANALYSIS OF AIRPLANE COMPONENT LOADS
SYMMETRICAL FLIGHT WITHOUT PITCHING ACCELERATIONS
Condition FLAPS & GEAR UP ~ POWER ON

V-n Diagram	PHAA	NHAA	PLAA · V_H	PLAA · V_0	NLAA · V_H	NLAA · V_0
C. G. Location	3	1	1	1	4	2
Gross Weight lbs.	64 000	64 000	64 000	64 000	64000	64000
AIRPLANE						
$n\,Z_A$	3.00	-1.50	3.00	2.51	-1.50	-1.14
W/S_W, #/sq. ft.	44.25	44.25	44.25	44.25	44.25	44.25
$n\,A\,W/S_W$	132.75	-66.375	132.75	111.07	-66.380	-50.445
C_{Z_A}	1.48588	-.69208	.8296	.4443	-.4149	-.2018
$q = nA(W/S_W)/C_{Z_A}$, #/sq. ft.	89.28	95.86	160	250	160	250
$q/(W/S_W)$	2.0176	2.1663	3.616	5.650	3.616	5.650
$V\sigma^{1/2} = \sqrt{391\,q}$, mph	186.9	193.6	250	313	250	313
α T.L., degrees	8.55	-14.88	1.60	-2.51	-11.65	-9.45
C_{X_A}	-.14166	-.15996	.0039	.04509	.01339	-.00269
$n\,X_A = C_{X_A}\,q/(W/S_W)$	-.2858	-.3465	.014102	.25476	.04842	-.015198
C_{L_A}	1.4933	-.6893	.8243	.4470	-.4044	-.2001
C_{D_A}	.0838	.0175	.02928	.0213	.0859	.0244
WING						
C_{L_W}	1.4010	-.5670	.8193	.4732	-.2964	-.1111
C_{D_W}	1073	.0215	.0354	.0162	.0111	.0075
C_{Z_W}	1.4014	-.5425	.8200	.4723	-.2927	-.1108
C_{X_W}	-.1021	-.1248	.0126	.0370	-.0489	-.0109
$n\,Z_W = C_{Z_W}\,q/(W/S_W)$	2.8274	-1.1752	2.961	2.668	-1.058	-.626
$n\,X_W = C_{X_W}\,q/(W/S_W)$	-.2060	-.2703	.0456	.20905	-.1769	-.0616
$Z_W = n\,Z_W\,W$, lbs.	181,075	-75,263	189,800	170,776	-67,778	-40,089
$X_W = n\,X_W\,W$, lbs.	-13,192	-17,314	2920	13,390	-11,323	-3940
C_{M_W} (a.c.)	-.0630	-.0630	-.0630	-.0630	-.0630	-.0630
$M_W = q\,S_W\,MAC_W\,C_{M_W}$, ft. lbs.	-114,126	-122538	-204,528	-319,575	-204,528	-319,575
ΔC_{M_A} due to wing	-.1765	-.0525	-.1069	-.0655	-.0927	-.06961

REPORT NO. R167-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION		PAGE 221
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY
Subject:— BASIC FLIGHT CRITERIA — PACK ON			DATE March 15, 1949 REVISED

V-n Diagram	PHAA	NHAA	PLAA V_H	PLAA V_D	NLAA V_H	NLAA V_D
C. G. Location	3	1	1	1	4	2
Gross Weight lbs.	64 000	64 000	64 000	64 000	64 000	64 000
α TL degrees	8.55	-14.88	1.60	-2.51	-11.65	-9.45
FUSELAGE						
C_{L_F}	.0212	-.0485	.0019	-.00356	-.0336	-.0246
C_{D_F}	.0144	.0207	.01404	.0123	.01666	.0149
C_{Z_F}	.0231	-.0524	.0022	-.0041	-.0362	-.0264
C_{X_F}	.0110	.0065	.0120	.0121	.00951	.0166
$\eta Z_F = C_{Z_F}\, q/(W/S_W)$	.0466	-.1135	.00796	-.0232	-.1309	-.1492
$\eta X_F = C_{X_F}\, q/(W/S_W)$	.0222	.0141	.0434	.0684	.0344	.0599
$Z_F = \eta Z_F W$, lbs.	2985	-7270	510	-1483	-8380	-9550
$X_F = \eta X_F W$, lbs.	1421	902	2775	4375	2201	3835
C_{M_F} (.25 L_F)	.0251	-.0308	.0051	-.00798	-.0308	-.02695
$M_F = q\, S_W\, MAC_W\, C_{M_F}$, ft. lbs.	45,469	-59,907	16 561	-40,451	-100,014	-136,738
ΔC_{M_A} due to fuselage	.0485	-.0844	.00538	-.01403	-.07048	-.05775
BOOM (ONE)						
C_{L_B}	.00322	.00913	.0001	-.0002	-.0059	-.00395
C_{D_B}	.00574	.00733	.0053	.0053	.0064	.0060
C_{Z_B}	.0040	-.0108	.0002	-.0004	-.0071	-.0049
C_{X_B}	.0053	.0045	.0052	.0053	.0051	.0053
$\eta Z_B = C_{Z_B}\, q/(W/S_W)$	.0081	-.0234	.0007	-.0023	-.0257	-.0277
$\eta X_B = C_{X_B}\, q/(W/S_W)$	.0107	.0097	.01860	.0299	.0184	.0299
$Z_B = \eta Z_B W$, lbs.	517	-1498	46	-145	-1643	-1771
$X_B = \eta X_B W$, lbs.	685	624	1205	1917	1180	1917
C_{M_B} (.25 L_B)	.0406	-.0092	.0168	.00598	-.0064	-.0042
$M_B = q\, S_W\, MAC_W\, C_{M_B}$, ft. lbs.	73,548	-17,894	54,554	30,313	-20 782	-21 310
ΔC_{M_A} due to one boom	.0404	-.00697	.01648	.00579	-.00482	-.00385

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

DATE March 15, 1949

REVISED

Subject: BASIC FLIGHT CRITERIA - PACK ON

V-n Diagram	PHAA	NHAA	PLAA V_H	FLAA V_o	NLAA V_H	NLAA V_o
C. G. Location	3	1	1	1	4	2
Gross Weight, lbs.	64 000	64 000	64 000	64 000	64 000	64 000
α TL degrees	8.55	14.88	1.60	-2.51	-11.65	
WING SECTION IMMERSED IN SLIPSTREAM (ONE)						
$\Delta C_{L_{WI}}$	.0278	-.0070	.0087	.0028	-.0017	-.0002
$\Delta C_{D_{WI}}$	.0094	.0048	.0009	-.00025	.0015	.00075
$\Delta C_{Z_{WI}}$	.0325	-.0068	.0086	.0026	-.0015	-.0001
$\Delta C_{X_{WI}}$	.0046	.00235	.0007	-.0001	.00111	.0004
$\lambda Z_{WI} = \Delta C_{Z_{WI}} q/(W/S_w)$	.0656	-.0147	.0311	.0147	-.0054	-.0006
$\lambda X_{WI} = \Delta C_{X_{WI}} q/(W/S_w)$	.0093	.0051	.0025	-.0006	.0040	.0023
$\Delta Z_{WI} = \lambda \Delta Z_{WI}$ W, lbs.	4199	-943	1990	946	-347	-36
$\Delta X_{WI} = \lambda \Delta X_{WI}$ W, lbs.	594	326	162	-36	257	145
$\Delta C_{M_{WI}}$ (a.c.)	---	---	---	---	---	---
$M_{WI} = q\ S_w\ MAC_w\ C_{M_{WI}}$, ft. lbs.	---	---	---	---	---	---
ΔC_{M_A} due to one wing section immersed in slipstream	.00171	.00009	.00018	.00003	.00012	.00002
PROPELLER (ONE)						
C_{L_P}	.01052	-.01698	.00142	-.0020	-.0029	-.0075
C_{D_P}	-.0394	-.0333	-.0179	-.0089	.0189	-.0078
C_{Z_P}	.00455	-.00787	.00092	-.00161	.00666	-.00609
C_{X_P}	-.0405	-.0365	-.0179	-.00896	.0179	-.00896
$\lambda Z_P = C_{Z_P}\ q/(W/S_w)$	.0092	-.0170	.0033	-.0091	-.0241	-.0344
$\lambda X_P = C_{X_P}\ q/(W/S_w)$	-.0817	-.0791	-.0648	-.0506	.0648	-.0506
$Z_P = \lambda Z_P$ W, lbs.	588	-1092	213	-582	-1542	-2203
$X_P = \lambda X_P$ W, lbs.	-5233	-5064	-4140	-3240	+4140	-3240
ΔC_{M_A} due to one propeller	-.00057	-.00621	.00191	-.00119	-.00992	-.00645

RESTRICTED

REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION	PAGES	PAGE 223
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY
Subject: BASIC FLIGHT CRITERIA - PACK ON			DATE March 15, 1949 REVISED

V-n Diagram	PHAA	NHAA	PLAA V_H	PLAA V_D	NLAA V_H	NLAA V_D
C. G. Location	3	1	1	1	4	2
Gross Weight lbs.	64000	64000	64000	64000	64000	64000
α_{TL} degrees	8.55	-14.88	1.60	-2.51	-11.65	-9.45
VERTICAL TAIL (ONE)						
$C_{L_{VT}}$	0	0	0	0	0	0
$C_{D_{VT}}$	.00057	.00057	.00057	.00057	.00057	.00057
$C_{Z_{VT}}$	0	0	0	0	0	0
$C_{X_{VT}}$	.00057	.00057	.00057	.00057	.00057	.00057
$\eta X_{VT} = C_{X_{VT}} q/(W/S_W)$	.0011	.0012	.0021	.0032	.0021	.0032
$X_{VT} = \eta X_{VT} W$, lbs.	74	79	132	206	132	206
ΔC_{M_A} due to one tail	.00027	.000165	.000165	.000165	.00027	.000165
HORIZONTAL TAIL						
α_{ht}	1.30	-11.14	-1.63	-3.74	-9.22	-7.44
q_{ht}/q	1.11	1.000	1.049	1.0125	1.000	1.000
δ_e	-4.22	12.88	.168	2.84	7.78	7.54
$\eta_{ht} C_{L_{HT}}$	-.01202	-.04410	-.01735	-.02382	-.05344	-.0411
$C_{D_{HT}}$	.0095	.0165	.0021	.00237	.00342	.00299
$C_{Z_{HT}}$	-.01202	-.04410	-.01735	-.02382	-.05344	-.0411
$C_{X_{HT}}$	.0095	.0165	.00216	.00237	.00342	.00299
$\eta Z_{HT} = C_{Z_{HT}} q/(W/S_W)$	-.0243	-.0955	-.06273	-.1347	-.1932	-.2321
$\eta X_{HT} = C_{X_{HT}} q/(W/S_W)$	.0192	.0357	.00702	.0134	.01237	.01640
$Z_{HT} = \eta Z_{HT} W$, lbs.	-1553	-6118	-4018	-8620	-12375	-14873
$X_{HT} = \eta X_{HT} W$, lbs.	1229	2285	501	658	792	1082
$C_{M_{HT}}$ (a. c.)	.00572	-.01574	-.000215	-.00351	-.00951	-.00921
$M_{HT} = q S_W MAC_W C_{M_{HT}}$, ft lbs	10362	-30,615	-698	-17,792	-37,058	-46,729
SUMMATION						
Z_A, lbs.	193115	-95,717	190,798	161,099	-95,597	-72,531
X_A, lbs.	-16,302	-22,197	914	16,317	3,088	-967
C_{MA-T}	-.04438	-.16275	-.06405	-.08794	-.19188	-.14759

FAS 500 23

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REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION	PAGES	PAGE 224
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY

Subject:- BASIC FLIGHT CRITERIA - PACK ON

DATE March 15, 1949

REVISED

ANALYSIS OF AIRPLANE COMPONENT LOADS
SYMMETRICAL FLIGHT WITHOUT PITCHING ACCELERATIONS
Condition FLAPS + GEAR UP - POWER ON
GUST CONDITIONS

V-n Diagram	PLAA - V_H	PLAA - V_H				
C. G. Location	3	5				
Gross Weight lbs.	74,000	42,136				
AIRPLANE						
ηZ_A	2.400	3.295				
W/S_W, #/sq. ft.	51.13	29.12				
$\eta_A W/S_W$	122.61	95.95				
C_{Z_A}	.7685	.5998				
$q = \eta_A (W/S_W)/C_{Z_A}$, #/sq. ft.	160	160				
$q/(W/S_W)$	3.129	5.496				
$V\sigma^{1/2} = \sqrt{391\,q}$, mph	250	250				
α T.L., degrees	.982	-.85				
C_{X_A}	.01045	.02265				
$\eta X_A = C_{X_A}\, q/(W/S_W)$	.03270	.12470				
C_{L_A}	.76817	.61129				
C_{D_A}	.02383	.01314				
WING						
C_{L_W}	.7675	.6130				
C_{D_W}	.0323	.0229				
C_{Z_W}	.76764	.61265				
C_{X_W}	.01914	.03199				
$\eta Z_W = C_{Z_W}\, q/(W/S_W)$	2.4019	3.3671				
$\eta X_W = C_{X_W}\, q/(W/S_W)$	.0599	.1758				
$Z_W = \eta Z_W\, W$, lbs.	177,741	141,876				
$X_W = \eta X_W\, W$, lbs.	4433	7408				
C_{M_W} (a.c.)	-.0630	-.0630				
$M_W = q\, S_W\, MAC_W\, C_{M_W}$, ft. lbs.	-204,528	-204,528				
ΔC_{M_A} due to wing	-.09941	-.06789				

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REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION	PAGES	PAGE 225
MODEL XC-120	PREPARED BY / CHECKED BY	APPROVED BY	
Subject:– BASIC FLIGHT CRITERIA – PACK ON		DATE March 15, 1949 REVISED	

V-n Diagram	PLAA-V_H	PLAA-V_H				
C. G. Location	3	5				
Gross Weight lbs.	74,000	42,136				
α TL degrees	.98	-.85				
FUSELAGE						
C_{L_F}	.00096	.00088				
C_{D_F}	.0121	.0121				
C_{Z_F}	.00120	-.00116				
C_{X_F}	.0120	.0120				
$\eta Z_F = C_{Z_F}\, q/(W/S_W)$	.003755	-.006375				
$\eta X_F = C_{X_F}\, q/(W/S_W)$	.037548	.065952				
$Z_F = \eta Z_F\, W$, lbs.	278	-269				
$X_F = \eta X_F\, W$, lbs.	2778	2779				
C_{M_F} (.25 L_F)	.00303	-.00271				
$M_F = q\, S_W\, MAC_W\, C_{M_F}$, ft. lbs.	9837	-8798				
ΔC_{M_A} due to fuselage	.00451	-.00545				
BOOM (ONE)						
C_{L_B}	.000043	-.000043				
C_{D_B}	.00531	.00531				
C_{Z_B}	.000086	-.000065				
C_{X_B}	.00533	.00533				
$\eta Z_B = C_{Z_B}\, q/(W/S_W)$	.000269	-.000357				
$\eta X_B = C_{X_B}\, q/(W/S_W)$	.016678	.029294				
$Z_B = \eta Z_B\, W$, lbs.	20	-15				
$X_B = \eta X_B\, W$, lbs.	1234	1234				
C_{M_B} (.25 L_B)	.01479	.00986				
$M_B = q\, S_W\, MAC_W\, C_{M_B}$, ft. lbs.	48,015	32,010				
ΔC_{M_A} due to one boom	.01547	.00375				

RESTRICTED

REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION		PAGES	PAGE 226
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY	
Subject:— BASIC FLIGHT CRITERIA - PACK ON			DATE March 15, 1949	REVISED

V-n Diagram	PLAA - V_H	PLAA V_H				
C. G. Location	3	5				
Gross Weight, lbs.	74,000	42,136				
α TL degrees	.98	-.85				
WING SECTION IMMERSED IN SLIPSTREAM (ONE)						
$\Delta C_{L_{WI}}$	.0081	.0068				
$\Delta C_{D_{WI}}$	.0006	-.0001				
$\Delta C_{Z_{WI}}$	.0081	.0065				
$\Delta C_{X_{WI}}$	.00053	.00015				
$\lambda \Delta Z_{WI} = \Delta C_{Z_{WI}} q/(W/S_W)$	.025345	.035724				
$\lambda \Delta X_{WI} = \Delta C_{X_{WI}} q/(W/S_W)$	.001658	.000824				
$\Delta Z_{WI} = \lambda \Delta Z_{WI}$ W, lbs.	1876	1505				
$\Delta X_{WI} = \lambda \Delta X_{WI}$ W, lbs.	123	35				
$\Delta C_{M_{WI}}$ (a.c.)	—	—				
$\Delta M_{WI} = q\ S_W\ MAC_W\ C_{M_{WI}}$, ft. lbs.	—	—				
ΔC_{M_A} due to one wing section immersed in slipstream	.000262	.000359				
PROPELLER (ONE)						
C_{L_P}	.00090	-.00077				
C_{D_P}	-.01789	-.01789				
C_{Z_P}	.00059	-.00050				
C_{X_P}	-.0179	-.0179				
$\lambda Z_P = C_{Z_P}\ q/(W/S_W)$	.001846	-.002748				
$\lambda X_P = C_{X_P}\ q/(W/S_W)$	-.056009	-.098378				
$Z_P = \lambda Z_P$ W, lbs.	137	-116				
$X_P = \lambda X_P$ W, lbs.	-4145	-4145				
ΔC_{M_A} due to one propeller	-.001728	-.000162				

RESTRICTED

REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION	PAGES	PAGE 227
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY
Subject: BASIC FLIGHT CRITERIA - PACK ON			DATE March 15, 1949 REVISED

V-n Diagram	PLAA-V_H	PLAA-V_H				
C. G. Location	3	5				
Gross Weight lbs.	74,000	42,136				
α_{TL} degrees	.98	-.85				
VERTICAL TAIL (ONE)						
$C_{L_{VT}}$	0	0				
$C_{D_{VT}}$	.00057	.00057				
$C_{Z_{VT}}$	0	0				
$C_{X_{VT}}$	.00057	.00057				
$\eta X_{VT} = C_{X_{VT}}\, q/(W/S_W)$	.001784	.00313				
$X_{VT} = \eta X_{VT}\, W$, lbs.	132	132				
ΔC_{M_A} due to one tail	.00027	.000185				
HORIZONTAL TAIL						
α_{ht}	-1.35	-2.67				
q_{ht}/q	1.048	1.038				
δ_e	.65	2.87				
$\eta_{ht}\, C_{L_{HT}}$	-.01798	-.01456				
$C_{D_{HT}}$	.00225	.00236				
$C_{Z_{HT}}$	-.01798	-.01456				
$C_{X_{HT}}$	.00225	.00236				
$\eta Z_{HT} = C_{Z_{HT}}\, q/(W/S_W)$	-.05626	-.0800				
$\eta X_{HT} = C_{X_{HT}}\, q/(W/S_W)$	.007040	.012971				
$Z_{HT} = \eta Z_{HT}\, W$, lbs.	-4163	-3370				
$X_{HT} = \eta X_{HT}\, W$, lbs.	521	547				
$C_{M_{HT}}$ (a. c.)	-.00083	-.00354				
$M_{HT} = q\, S_W\, MAC_W\, C_{M_{HT}}$, ft lbs	-2695	-11,493				
SUMMATION						
Z_A, lbs.	177,922	140,985				
X_A, lbs.	2420	5246				
$C_{M_{A-T}}$	-.066352	-.053076				

PART II - H - 2

2. FLAPS AND GEAR DOWN - POWER OFF AND ON

The limit speed for flaps and gear down is 160 mph so that the loads on the component parts of the airplane for balance condition are less critical than for the flaps and gear up conditions.

PART II - I - 1

I. WING LOADS

1. FLAPS AND GEAR UP

a. Balanced Conditions

The wing loads as determined in Part II-H-1 have been analyzed by the Structures Section and the following conditions selected as being possibly critical. A summary of the loads for these critical conditions is presented in Part I-C-1a.

Condition	Possibly Critical for
PHAA Speed 191.2 mph Power Off C.G. Location 3 Gross Weight 64,000 lbs.	Upper surface in compression. Lower surface in tension, outer panel. Front spar, outer panel. Rear spar, lower cap, center section. Outer panel to center section splice, front spar.
$PLAA_{VH}$ Speed 250 mph Power Off C.G. Location 1 Gross Weight 64,000 lbs.	Upper surface in compression Outer panel to center section splice, front and rear spars. Lower surface in tension, center section. Front spar, lower cap, center section.
$PLAA_{VD}$ Speed 313 mph Power Off C.G. Location 1 Gross Weight 64,000 lbs.	Rear spar, outer panel

PART II - I - 1

Condition		Possibly critical for
NHAA		Lower surface in compression. Upper surface, outer panel to center section splice.
Speed	198.9 mph	
Power	Off	
C.G. Location		
Gross Weight	64,000 lbs.	
NLAA, D		Outer panel skin splices.
Speed	313 mph	
Power	Off	
C.G. Location	2	
Gross Weight	64,000 lbs.	
PLAAVH gust		Local attachments of weight items.
Speed	250 mph	
Power	On	
C.G. Location	5	
Gross Weight	42,130 lbs.	
PHAA		Special leading edge conditions
Speed		
Power	Off	
C.G. Location		
Gross Weight	64,000 lbs.	
NLAA		Special leading edge conditions.
Speed		
Power	Off	
C.G. Location		
Gross Weight	64,000 lbs.	

PART II - I - 1

b. SPECIAL BALANCE CONDITIONS

It is required by reference (2) that wing ribs and leading edge shall be investigated for pressure distribution corresponding to

(1) Positive lift coefficient giving most forward center of pressure. — This corresponds to PHAA conditions considered in Part II-H-1.

(2) Lift coefficient giving positive limit load factor at V_D or V_H if higher. -- This corresponds to [illegible] conditions considered in Part II-H-1.

(3) Distribution giving maximum negative load on the forward 20% of the wing. -- This corresponds to PHAA and/or NHAA conditions considered in Part II-H-1.

Therefore the special balance conditions of reference (2) are all covered by the critical balance conditions of Part II-H-1.

c. SPECIAL LEADING EDGE CONDITION

It is requested by reference (2) that the wing leading edge be designed for loads that exist at most forward C.P. and for maximum negative loadings on forward 20%, modified for momentary increases in accelerated flight. Examination of accelerated flight conditions indicate that an increase of 25% in lift coefficient is the maximum ever to be encountered. Therefore it is assumed that the maximum lift coefficient would be increased by 25% and the other coefficients extrapolated to match this value. Conditions are computed for a constant load factor at reduced speed and the necessary aerodynamic coefficients are tabulated in the table immediately following.

Note that this load should only be applied over the leading edge of the wing back to the front spar.

The effect on the airplane force coefficients due to this arbitrary 25% increase in wing lift coefficient is considered to be directly attributable to the increase in wing force coefficients -- no other component parts being considered.

RESTRICTED

REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION	PAGES	PAGE 232
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY
Subject:- BASIC FLIGHT CRITERIA - PACK ON			DATE March 15, 1949
			REVISED

ANALYSIS OF AIRPLANE COMPONENT LOADS
SYMMETRICAL FLIGHT WITHOUT PITCHING ACCELERATIONS

Condition SPECIAL LEADING EDGE CONDITION
FLAPS & GEAR UP - POWER OFF

V-n Diagram	FHAA	NHAA				
C. G. Location						
Gross Weight lbs.	64000	64,000				
AIRPLANE						
ηZ_A	3.0	-1.5				
W/S_W, #/sq. ft.	44.25	44.25				
$\eta_A W/S_W$	132.75	-66.375				
C_{Z_A}	1.790	-.832	EXTRAPOLATED			
$q = \eta_A(W/S_W)/C_{Z_A}$, #/sq. ft.	74.1	79.7				
$q/(W/S_W)$						
$V\sigma^{1/2} = \sqrt{391\,q}$, mph	170.2	176.8				
α T.L., degrees	12.69	-16.55	$\alpha_{T.L.} =$	$\frac{C_{LW}}{.0842}$	$= -8.13°$	
C_{X_A}						
$\eta X_A = C_{X_A}\,q/(W/S_W)$						
C_{L_A}						
C_{D_A}						
WING						
C_{L_W}	1.751	-.709	$C_{LW_{max}} \times 1.25$			
C_{D_W}	.210	.0298	EXTRAPOLATED			
C_{Z_W}	1.7561	-.6885	$C_{LW} \cos \alpha_{T.L.} + C_{DW} \sin \alpha_{T.L.}$			
C_{X_W}	-.1790	-.1736	$C_{DW} \cos \alpha_{T.L.} - C_{LW} \sin \alpha_{T.L.}$			
$\eta Z_W = C_{Z_W}\,q/(W/S_W)$	2.840	-1.154				
$\eta X_W = C_{X_W}\,q/(W/S_W)$	-.300	-.2806				
$Z_W = \eta Z_W\,W$, lbs.	181,800	-73,900				
$X_W = \eta X_W\,W$, lbs.	-19,200	-17,980				
C_{M_W} (a.c.)	-.0630	-.0630				
$M_W = q\,S_W\,MAC_W\,C_{M_W}$, ft. lbs.	-94,800	-102,000				
ΔC_{M_A} due to wing,						

PART II - I - 2

2. FLAPS AND GEAR DOWN

As noted in Part II-H-2 flapped conditions are not critical for wing loads. Actual flap loads are considered in Part II-L.

PART II - J

J. AILERON LOADS

Aileron loads for symmetrical flight are for neutral ailerons only. These loads are those existing from the pressure distributions for the critical wing conditions of Part II-I over the aileron area.

K. AILERON TAB LOADS

Aileron trim and balance tab loads are those considered under Part IV Assymmetrical Flight - Roll.

PART II - L

L. FLAP LOADS

It is required by reference (2) 6B that flap limit loading be determined from the formula:

$$W \ \#/\text{Sq.Ft.} = .002558 \left(\Delta C_{L\ \text{flap only}} \right) \times V^2 \times 6$$

where ($\Delta C_{L\ \text{Flap Only}}$) is flap lift coefficient in fully deflected position with the gust condition at 160 mph indicated airspeed (Flap Limit Speed).

The maximum flap load is obtained when the flapped wing lift coefficient and the dynamic pressure are at maximum values for the conditions investigated.

As the maximum speed for flaps deflected is 160 mph, the maximum flapped wing lift coefficient with power-on, the dynamic pressure in the slipstream of the propeller is increased, which also tends to increase the load on the flaps as a portion of each of the flaps is in the slipstream of the propeller.

Therefore, the maximum flap load will be obtained under the following conditions.

POWER ON

$V\sqrt{\sigma} = 160$ mph G.W. = 74,000 lbs. $n = 1.97$ $T_c = .198$

$$C_{Z_A} = \frac{n \times W/S}{q} = \frac{1.97 \times 51.1}{65.5}$$

$$= 1.538$$

α T.L. = 4.38 C.G. = 3

$C_{L_{WF}} = 1.39$

$\Delta C_{L_{WF_T}} = .085$ from figure 53 reference (1) due to the change in effective angle of attack and dynamic pressure of that portion of the wing immersed in the slipstream of one propeller and is considered to be effective over the wing section immersed in the slipstream.

PART II - L

To convert $\triangle C_{LWF_T}$ to section value, it is multiplied by the ratio of the wing area to the area of the wing in the slipstream of one propeller.

$$\triangle c_{l_s} = \frac{S_W \ \triangle C_{LWF_I}}{S_{W_i}}$$

$$\triangle c_{l_s} = \frac{1447.25 \times .085}{232.7} = .528$$

The total section lift coefficient in the slipstream is then

$$c_l = c_{l_a} + c_{l_{bf}} + c_{l_{bt}} + \triangle c_{l_s}$$

where $c_{l_a} = c_{l_{a1}} \times C_{LWF}$

Spanwise values of c_l are shown in the following table.

The pressure distribution outside of the slipstream is:

$$P = \frac{p}{q} = p_l - p_n = P_o + P_{a\delta} \, c_{nf} + P_{al} \, (c_n - c_{nf})$$

It is assumed that $c_l = c_n$ and $c_{nf} = c_{lf}$

The $P_{a\delta}$ distribution was obtained from figure 25 of reference (11) extrapolated for $\delta f = 40^{\circ}$ and plotted in figure (54) as a chordwise distribution of $P_{a\delta}$ c_{nf}. The P_o and P_{al} distributions are from reference (7) replotted in figure 55.

PART II - L

The increase in dynamic pressure in the slipstream is equal to:

$$\Delta q = \frac{8}{\pi} q \, T_c = 33.00$$

$$T_c = .198 \text{ for } 160 \text{ mph}$$

$$\frac{\Delta q}{q} = \frac{33}{65.5} = .505$$

$$\frac{q_s}{q} = \frac{q + \Delta q}{q} = 1.505$$

$$q = \frac{(160)^2}{391} = 65.5 \text{ lbs/Sq.Ft.}$$

$$q_s = 1.505 \times 65.5 = 98.5 \text{ lbs/Sq.Ft.}$$

In the slipstream the section additional lift coefficient, c_{l_a} is then

$$c_{l_a} = c_l - 1.505 \, c_{l_f}$$

and the pressure distribution in the slipstream is:

$$P_s = \frac{p}{q} = p_l - p_n = 1.505 \ (P_o + P_{a_\delta} \ c_{n_f}) + P_{a_l} \ (c_n - 1.505 \ c_{n_f})$$

From the chordwise incremental pressure distribution curves figures 54 and 55. The load on the flap due to each incremental pressure distribution is determined by integrating the pressure distribution over the chord of the flap.

FIG 54

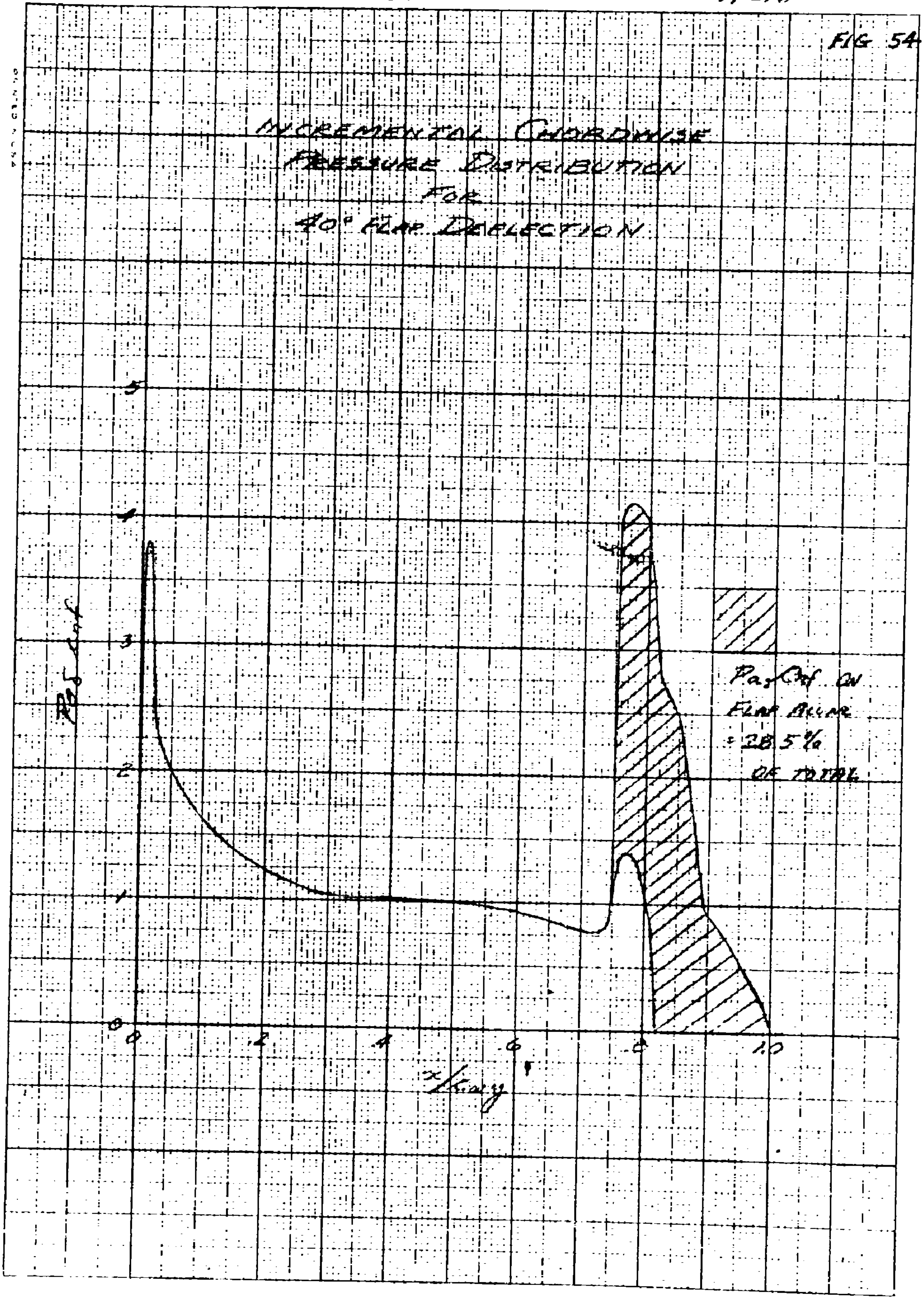
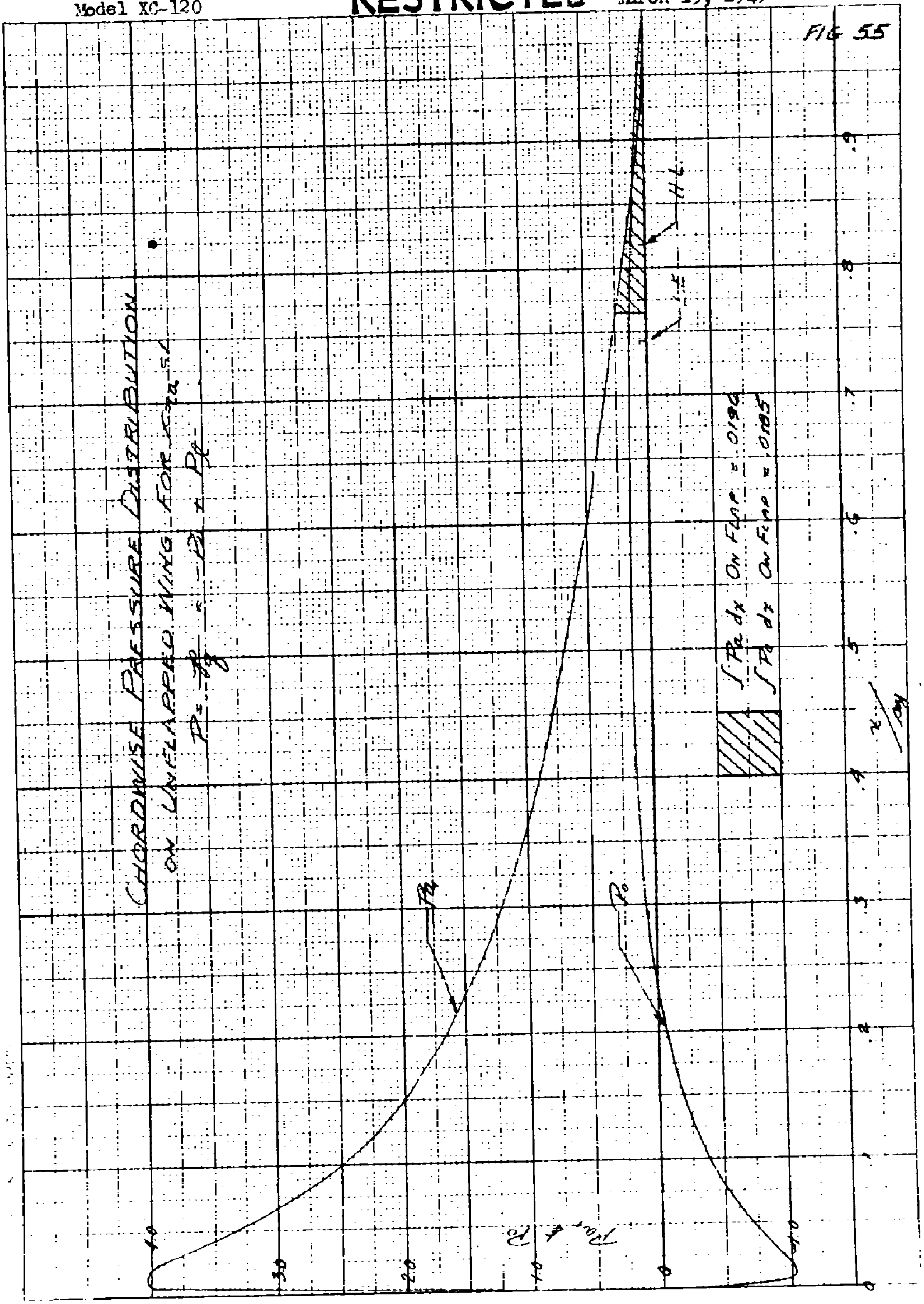

INCREMENTAL CHORDWISE
PRESSURE DISTRIBUTION
FOR
40° FLAP DEFLECTION
5
4
3
2
1
0
0
.2
.4
.6
.8
1.0
Pa, Cnf ON
FLAP BLUR
= 38.5%
OF TOTAL

FIG 55


CHORDWISE PRESSURE DISTRIBUTION
ON UNFLAPPED WING
H.L.
∫Pa dx On Flap = .0196
∫Pa dx On Flap = .0185
Po
4.0
3.0
2.0
1.0
0
-1.0
.1
.2
.3
.4
.5
.6
.7
.8
.9

CALCULATION OF FLAP LOA

$\Delta c l_s = \dfrac{\Delta C_{L_{WFI}} S_W}{S_{W_T}}$ δ_f FLAPS = 40° $V\sqrt{\sigma}$ = 160 MPH

1	$2y/B_w$	0	.10	.1144	.1289	.1642	.200	.2139	.2804
2	y_w	0	5.464	6250	7.043	8.971	10.427	11.686	15.320
3	c_w		17.91	16 82	16 69	16 37	16.05	15.92	
4	ξ correction	1	1	1	1.505	1.505	1.505	1.505	1.505
5	$c_{l_{a_1}}$	.959	.985	.990	.992	1.001	1.010	1.013	1.030
6	$c_{l_a} = 1.39\, c_{l_{a_1}}$	1.33	1.37	1.375	1.380	1.391	1.402	1.410	1.430
7	$c_{l_{b_f}} + c_{l_{b_t}}$	-1.38	0	.235	.460	.520	.470	.300	-.153
8	Δc_{l_p}	0	0	0	.528	.528	.528	.528	.528
9	(6) + (7) + (8)	1.192	1.37	1.610	1.840 / 2.368	2.439	2.400	2.238	1.855
10	(4) · $c_{l_f} = c_{l_f}$	0	0	1.383	1.383 / 2.080	2.080	2.080	2.080	0
11	(9) − (10)	1.192	1.37	.227	.457 / .288	.359	320	.158	1.855
12	(11)/(4)	1.192	1.37	.227	.1912	.2382	2122	1.050	.1200
	FLAP SECTION ONLY IN SLIPSTREAM								
13	$[\int \bar{P}_a\, dx] \times$ (11)	0	0	—	.00564	.007040	.00626	.0031	0
14	1.505(.0185 + .285 c_{n_f})				.62100	.6210	.62100	.6210	
15	$\dfrac{L}{q b_f c_{w\,ON\,FLAP}}$ = (13) + (14)				.62664	.62804	.62726	.6241	
16	$\dfrac{L}{S_f}$ = (15) $\dfrac{q c_w}{c_f}$				160	160	160.2	159.1	
	OUT OF SLIPSTREAM								
17	.0196 × (11)			.00445	.00896				
18	(.0185 + .285 c_{n_f})			41250	.41250				
19	$\dfrac{L}{q b_f c_{w\,ON\,FLAP}}$ (17) + (18)			.41695	.42146				
20	$\dfrac{L}{S_f}$ = (19) $\dfrac{q c_w}{c_f}$			106.2	107.5				

FLAP LOADS

O MPH POWER ON

39	.2804	.300	.3469	.4000	.4050	.4717	.4764	.50	.60	.70
686	15.320	16.391	18.953	21.854	22.127	22.493	26.028	27.318		
92			14.62	14.25	14.20	14.13	13.49	13.25		
05	1.505	1.505	1.505	1.505	1.505	1	1	1	1	1
13	1.030	1.036	1.038	1.041	1.041	1.042	1.050	1.051	1.043	1.027
10	1.430	1.439	1.441	1.449	1.449	1.450	1.460	1.461	1.450	1.426
00	-.153	-.142	.280	.500	.493	.502	.235	.006	-.177	-.197
28	.528	.528	.528	.528	.528	0	0	0	0	0
38	1.825	1.825	2.299	2.477	2.470 / 1.942	1.952	1.695	1.467	1.273	1.229
80	0	0	2.080	2.080	2.080 / 1.383	1.383	1.383	0	0	0
78	1.825	1.825	.169	.397	.39 / .559	.569	.312	1.467	1.273	1.229
50	.1200	.1211	.1121	.264	.260	.569	.312	1.467	1.273	1.229
31	0	0	.00331	.00779	.00765					
10			.62100	.62100	.62100					
241			.62931	.62879	.62865					
9.1			159.1	160.2	160.5					
					.01094	.01112	.006115			
					.41250	.41250	.41250			
					.42344	.42362	.418615			
					108.0	108	106.5			

PART II - L

P over the flap span outside of the slipstream based on c_W is:

$$P = .0185 + .285\ c_{nf} + .0196\ (c_n - c_{nf})$$

and the load on the flap is:

$$L = \frac{p\,q\,c_W}{c_f}$$

Integrating, we obtain

ΔC_L on flap outside slipstream equals

$$\frac{3.901 \times 107.5 \times 3.667 + .793 \times 107.0}{q\,(3.901 \times 3.667 + .793 \times 4.1985)}$$

$$\frac{1537.782 + 356.2}{q\,(14.304 + 3.329} = \frac{1893.98}{65.5 \times 17.637}$$

$$\Delta C_L = 1.6402$$

L outside of slipstream

$$W\ \#/\text{sq.ft.} = .002558\ (C_L \text{ flap only}) \times V^2 \times 6$$

$$= .002558\ (1.6402) \times (160)^2$$

$$L = 107.39 \quad \text{lbs/sq.ft.} \quad \text{outside of slipstream}$$

PART II - L

Then P_S over the flap span in the slipstream, based on c_W is

$$P_S = 1.505\ (.0185 + .285\ c_{nf}) + .0196\ (cn - 1.505\ c_{nf})$$

and the load on the flap is

$$L = P_S\ q\ \frac{c_W}{cf}$$

Integrating we obtain

$\triangle C_L$ on flap in the slipstream

$$= \frac{4.643 \times 160 \times 4.1983 + 3.174 \times 159.8 \times 3.6667}{qs\ (4.643 \times 4.1983 + 3.174 \times 3.667}$$

$$\frac{3118.833 + 1859.7693}{q\ (19.4927 + 11.639} = \frac{4978.6023}{98.5\ (31.1317)}$$

$$C_L = 1.6235$$

L in slip stream

$$\#/\text{sq.ft.} = \left[.002558 \times 1.6235 \times (160^2\right]$$

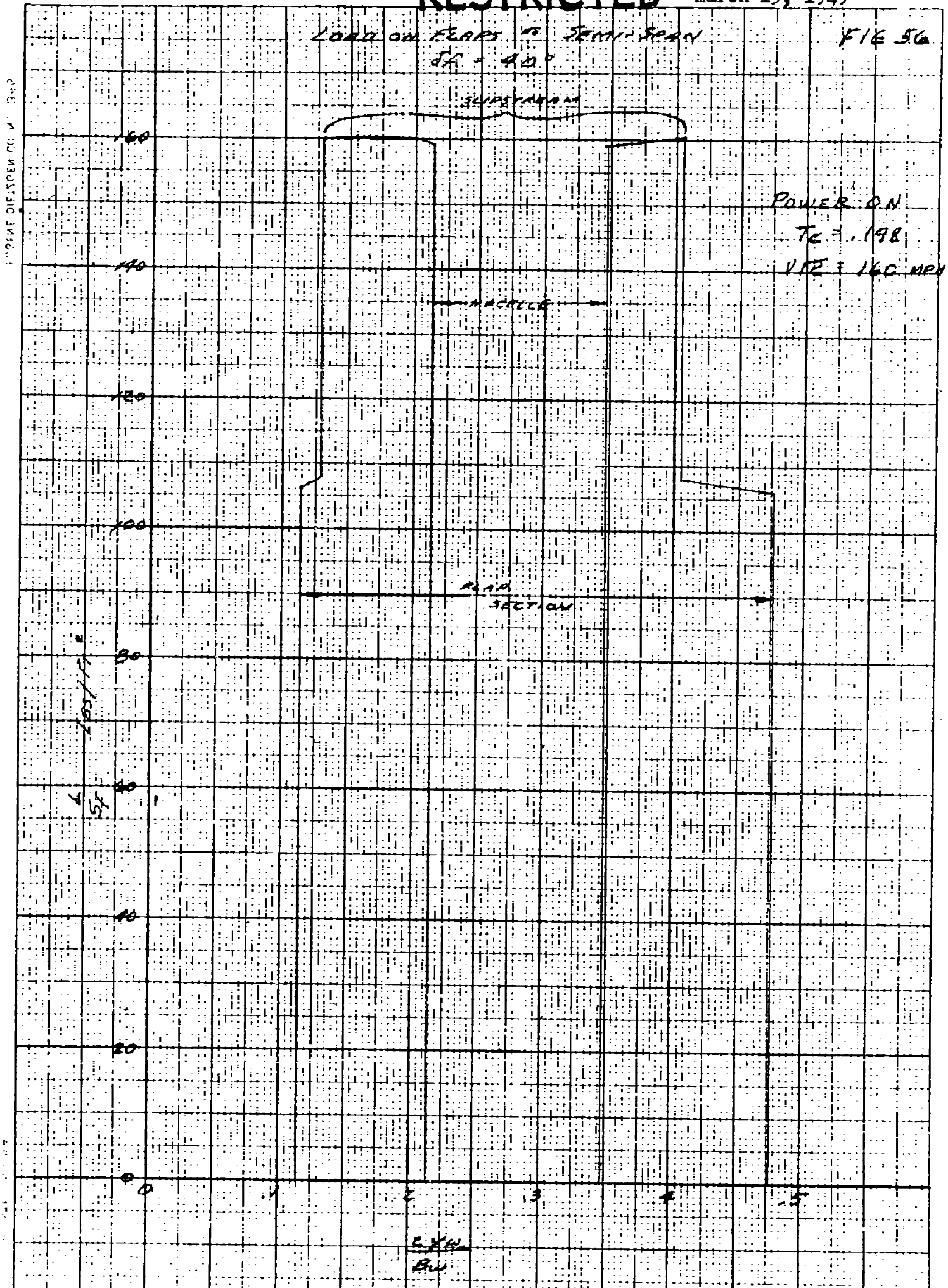
$$L = 159.6\ \text{lbs/Sq.Ft. in slipstream}$$

The Weighted Average Load on the Flaps is:

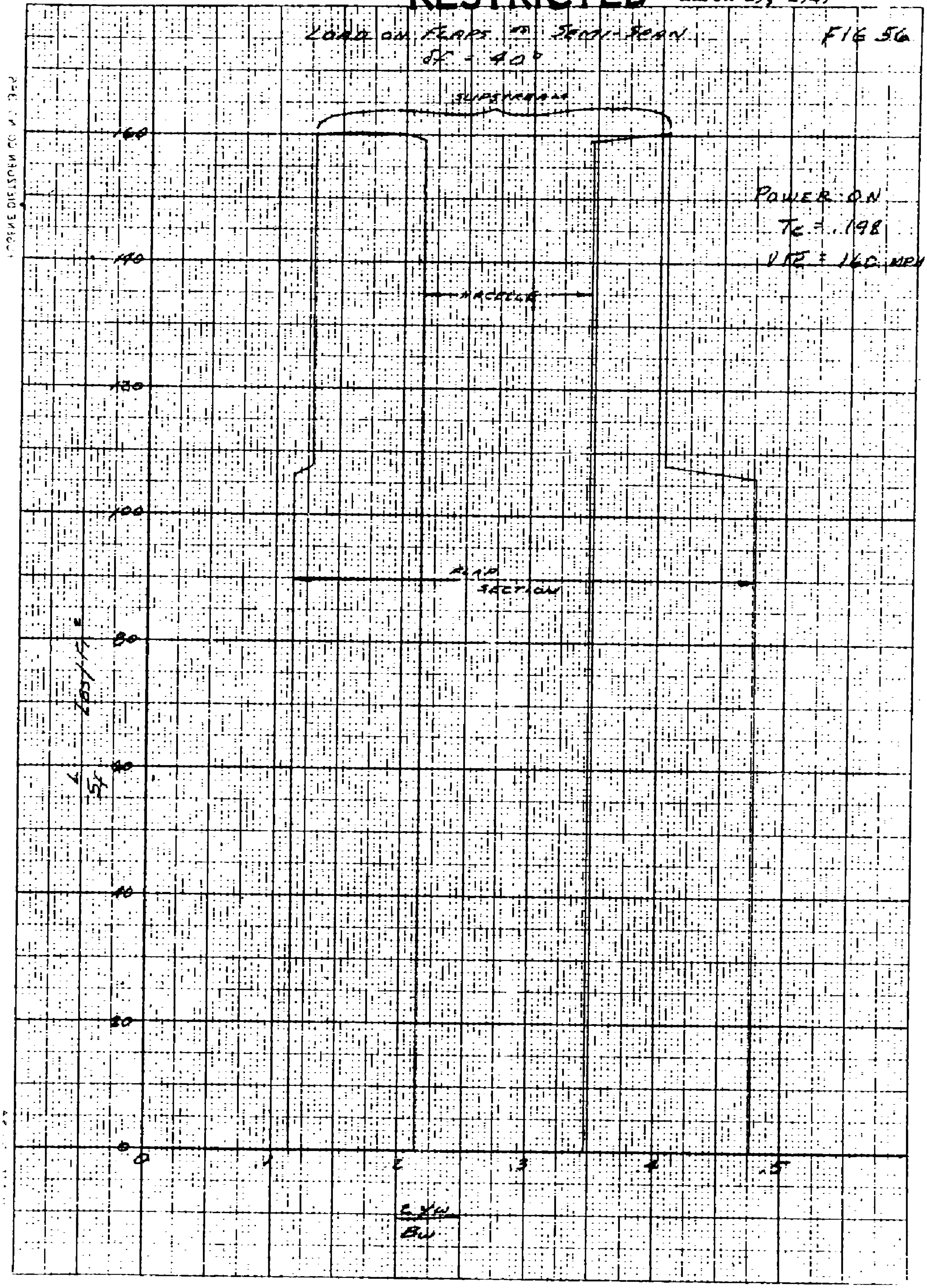
$$= \frac{107.39 \times 17.633 + 159.6 \times 31.126}{17.633 + 31.126}$$

$$= \underline{140.72\ \text{lbs./Sq.Ft.}}$$

Figure 56 shows the spanwise variation of maximum flap loading density for 160 mph, power on, flaps 40°.


LOAD ON FLAPS vs SEMI-SPAN
$\delta_f = 40°$
FIG 56
SLIPSTREAM
POWER ON
$T_c = .198$
$V_{IE} = 140$ MPH
NACELLE
FLAP SECTION
160
140
120
100
80
60
40
20
0
0
.1
.2
.3
.4
.5
$\frac{2Y_W}{B_W}$

RESTRICTED


LOAD ON FLAPS IN SEMI-SPAN
δF = 40°
FIG 56
SLIPSTREAM
POWER ON
Tc = .198
VTE = 140 MPH
NACELLE
FLAP SECTION
160
140
120
100
80
60
40
20
0
0
.1
.2
.3
.4
.5
L/δF LBS/FT
2Xw/Bw

PART II-M

M. Horizontal Tail Loads

1. Balance Conditions

Balancing horizontal tail loads were determined for all six points of the V-n diagram for all C. G. locations and gross weights, flaps up and flaps down.

The maximum balance load was determined to be -14,873 lbs. for the following condition $\eta = -1.14$ gross weight 64,000 lbs.

Flaps and gear up power on,

NLAA, V_D; and is shown in conjunction with the loads on the other component parts in Part II-H-1.

PART II-M-2

2. Gust Loads - Horizontal Tail

Gust loads on the horizontal tail are determined as specified in reference (2)-6B for minimum (42, 136 lbs.), normal (64,000 lbs.) and maximum (74,000 lbs.) gross weights.

The increments of horizontal tail load from a 50 ft. per second gust (50K sharp edged gust) on the horizontal tail at level-flight, high-speed are computed and added to the balancing tail load for unaccelerated flight at level-flight, high-speed with the stabilizers and elevators acting as a single unit. The effect at wing downwash, angular acceleration, etc. are considered.

a. Increment Gust Loads

Reference (2)-6b (Sections E-2d(1) and E-2d(2)) gives equations for airplanes with single vertical tails and for twin vertical tails as end plates, neither of which is entirely applicable in the case of the XC-120, which has twin vertical tails but a horizontal tail with tips extending beyond. The equations described above are identical except for the last term which is found to be equal to the lift curve slope of the tail. This being the case it is considered more accurate to insert the lift curve slope of the XC-120 horizontal tail as previously determined in reference (1) Part II-A-7-b,- as the last term in the above equations. This will result in the following equation which will be used:

$$\Delta L_{HT} = \Delta F_H = \frac{57.3}{V_H} U \times S_H \times q \left(1 - \frac{52 \times a_w \times K_P \times K_T \times K_S}{A}\right) \times \left(C_{L\alpha_{ht}}\right)$$

$C_{L\alpha_{ht}}$ being equal to $\left(\dfrac{\pi A_H \times a_{oH}}{(\pi A_H) + (57.3 \times a_{oH} \times r)}\right)$

PART II-M-2a (Cont.)

where

$\Delta F_H = \Delta L_T$ Increment of the horizontal tail load due to a gust ~ in. lbs.

V_H = True level flight high speed = 250 mph = 366.7 ft./sec.

U = $\pm 50\ K/\sigma^{\frac{1}{2}}$ K = Gust alleviation factor figure 5, reference (2)-2B

S_H = S_{ht} = Horizontal tail surface area = 346.2 sq. ft.

q = $\sigma V_{mph}^2/391$ dynamic pressure = 160 lbs./sq. ft.

a_w = .0842 = Wing lift curve slope

K_P = Downwash correction factor for horizontal tail position = .561 (from figure 1, reference (2)-6B where:

l_{ht} = 47.5 ft. and y = .38c

K_T = Downwash correction factor for wing taper = 1.51
Taper ratio = .5 figure 2, reference (2)-6B

K_S = Downwash correction factor for horizontal tail span = .959 figure 3, reference (2)-6B

A = Aspect Ratio of Wing = 8.25

$\eta_{ht} C_{L\alpha_{ht}}$ = .0450 reference (1) Part II-A-7b effective lift curve slope of horizontal tail

Weight lbs.	S_w sq.ft.	W/S_w #/ft.^{2}	K	U	$\frac{57.3\ S_H q}{V_H}$	$\left(1 - \frac{52\ a_w K_P K_T K_S}{A}\right)$	$\eta_{ht} C_{L\alpha_{ht}}$	$\Delta F_H = \Delta L_T$ lbs.
42136	1447.25	29.18	.556	±27.88	8650	.569	.0450	±6140
64000	1447.25	44.22	.589	±29.45	8650	.569	.0450	±6510
74000	1447.25	51.13	.600	±30.00	8650	.569	.0450	±6640

PART II-M-2b

b. Balance Loads

The balancing tail loads for unaccelerated flight are obtained from Part II-D for 250 mph, power off and power on, presented in figures 57 and 58, respectively.

For unaccelerated level flight at 250 mph

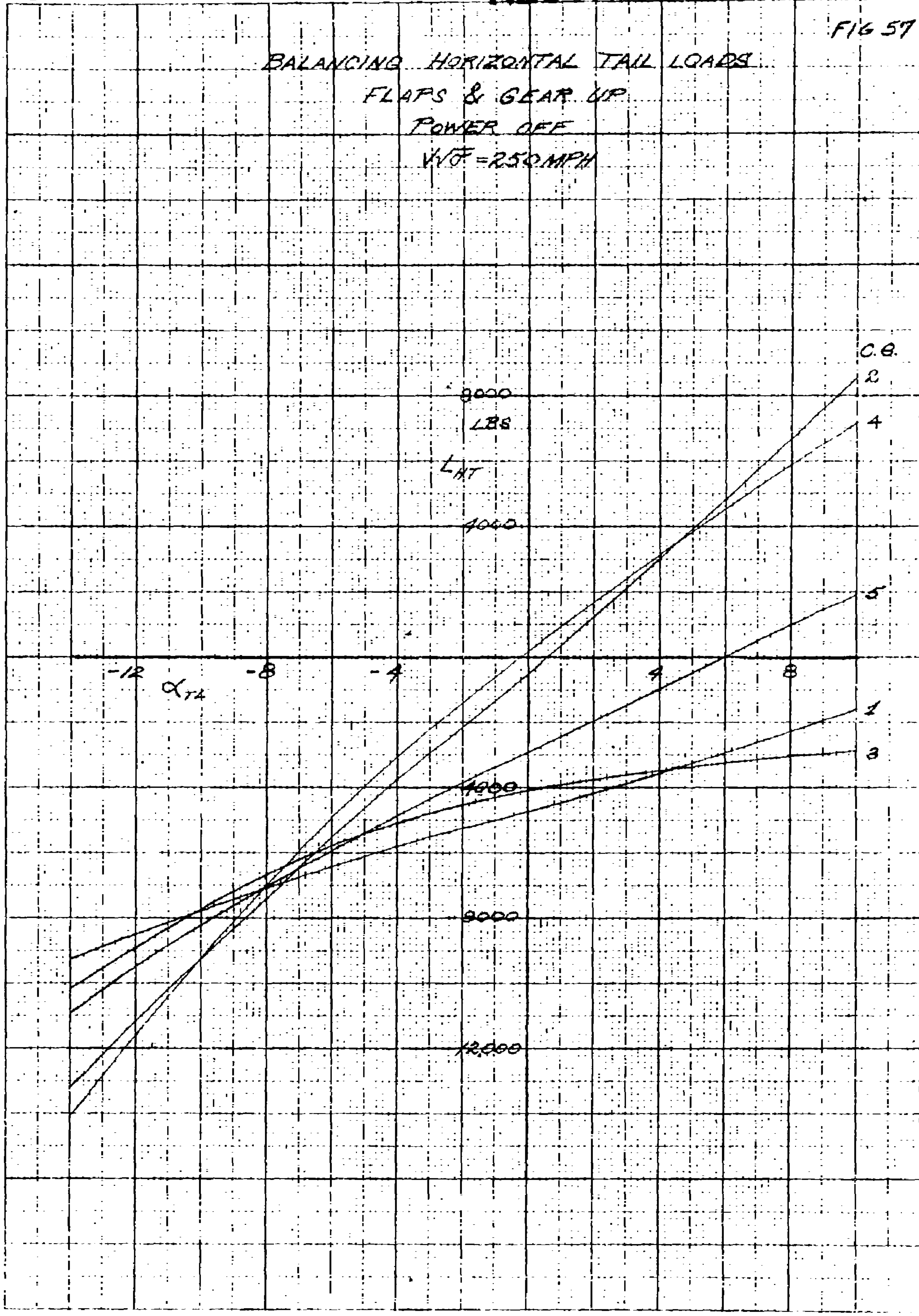
$$C_L = \frac{.391 \times G.W.}{\sigma \times S_w \times V_{mph}^2}$$

For

42136 lbs.	C_L	= .1820
64000 lbs.	C_L	= .2762
74000 lbs.	C_L	= .320

α_{TL} is obtained from figures 59 which is a plot of the balanced airplane lift coefficient versus angle of attack of the thrust line as determined in Part II-F for power off and figure 60 for power on at 250 mph.

FIG 57


BALANCING HORIZONTAL TAIL LOADS
FLAPS & GEAR UP
POWER OFF
$V\sqrt{\sigma}$ = 250 MPH
C.G.
2
4
5
1
3
8000
LBS
L_{HT}
4000
4000
8000
12,000
-12
-8
-4
4
8
α_{TA}

Report No. B107-014
Model XC-120

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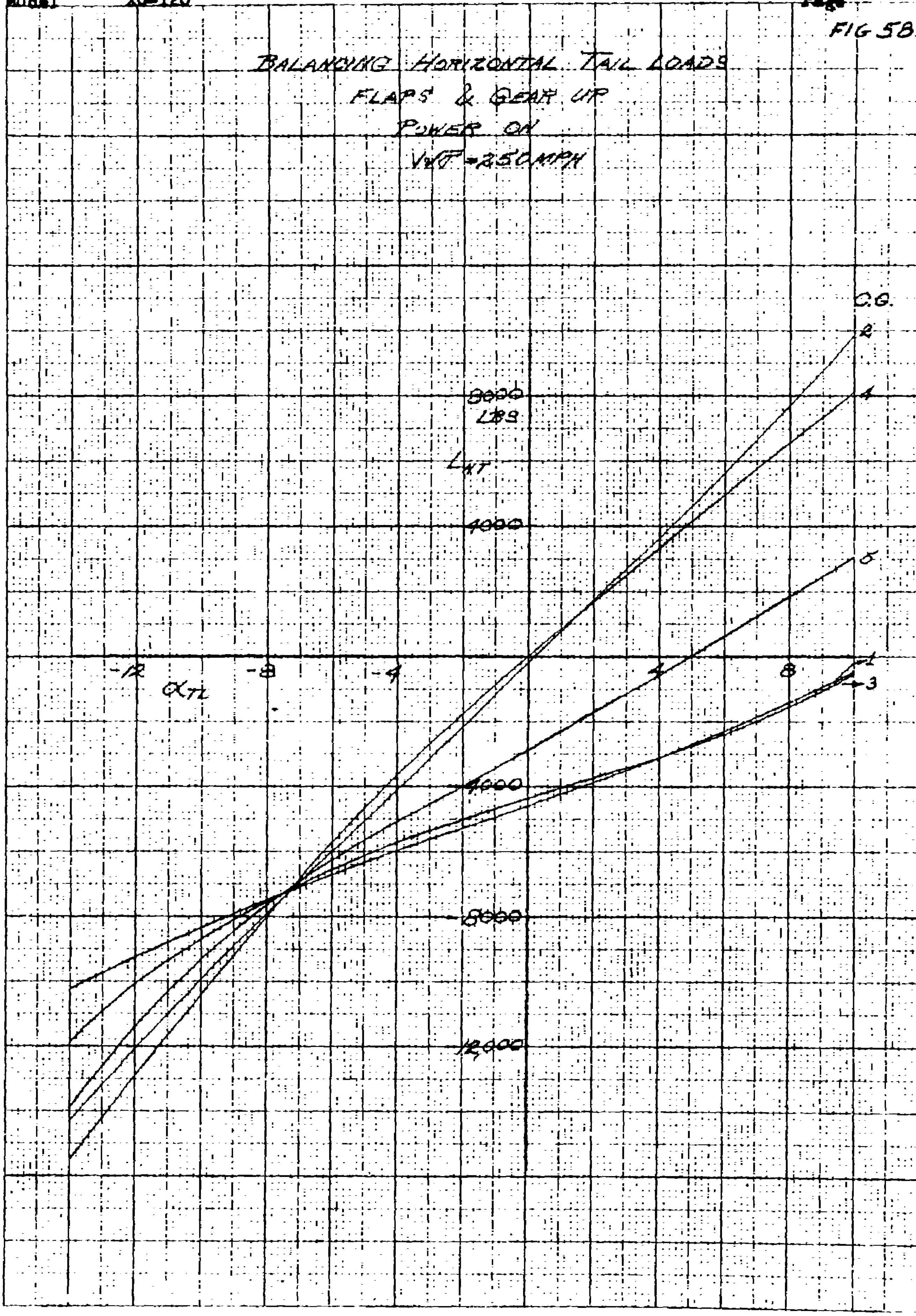
FIG 58



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REPORT NO. R107-014 | FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION | PAGE 250

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

DATE 15 March 1949

REVISED

Subject: BASIC FLIGHT CRITERIA - PACK ON

PART II-M-2b

BALANCING HORIZONTAL TAIL LOADS
FLAPS & GEAR UP - POWER OFF
$V\sqrt{\sigma} = 250$ mph

	C.G. 1		C.G. 2		C.G. 3		C.G. 4		C.G. 5	
α_{TL}	$\eta_t C_{L_{HT}}$	L_{HT}	$\eta_t C_{L_{HT}}$	L_{HT}	$\eta_t C_{L_{HT}}$	L_{HT}	$\eta_t C_{L_{HT}}$	L_{HT}	$\eta_t C_{L_{HT}}$	L_{HT}
-12	-.153	-8470	-.202	-11,190	-.161	-8920	-.210	-11,630	-.172	-9530
-8	-.128	-7090	-.134	-7420	-.121	-6700	-.127	-7030	-.127	-7050
-4	-.105	-5820	-.066	-3710	-.092	-5100	-.055	-3050	-.088	-4870
0	-.055	-4710	-.008	-440	-.074	-4100	+.003	+170	-.053	-2940
4	-.064	-3540	+.055	+3050	-.063	-3490	+.056	+3100	-.018	-997
8	-.040	-2220	+.121	+6700	-.055	-3050	+.106	+5870	+.018	+997

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REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION	PAGES	PAGE 251
MODEL XO-120	PREPARED BY	CHECKED BY	APPROVED BY
			DATE 15 March 1949
Subject:— BASIC FLIGHT CRITERIA - PACK ON			REVISED

PART II-M-2b

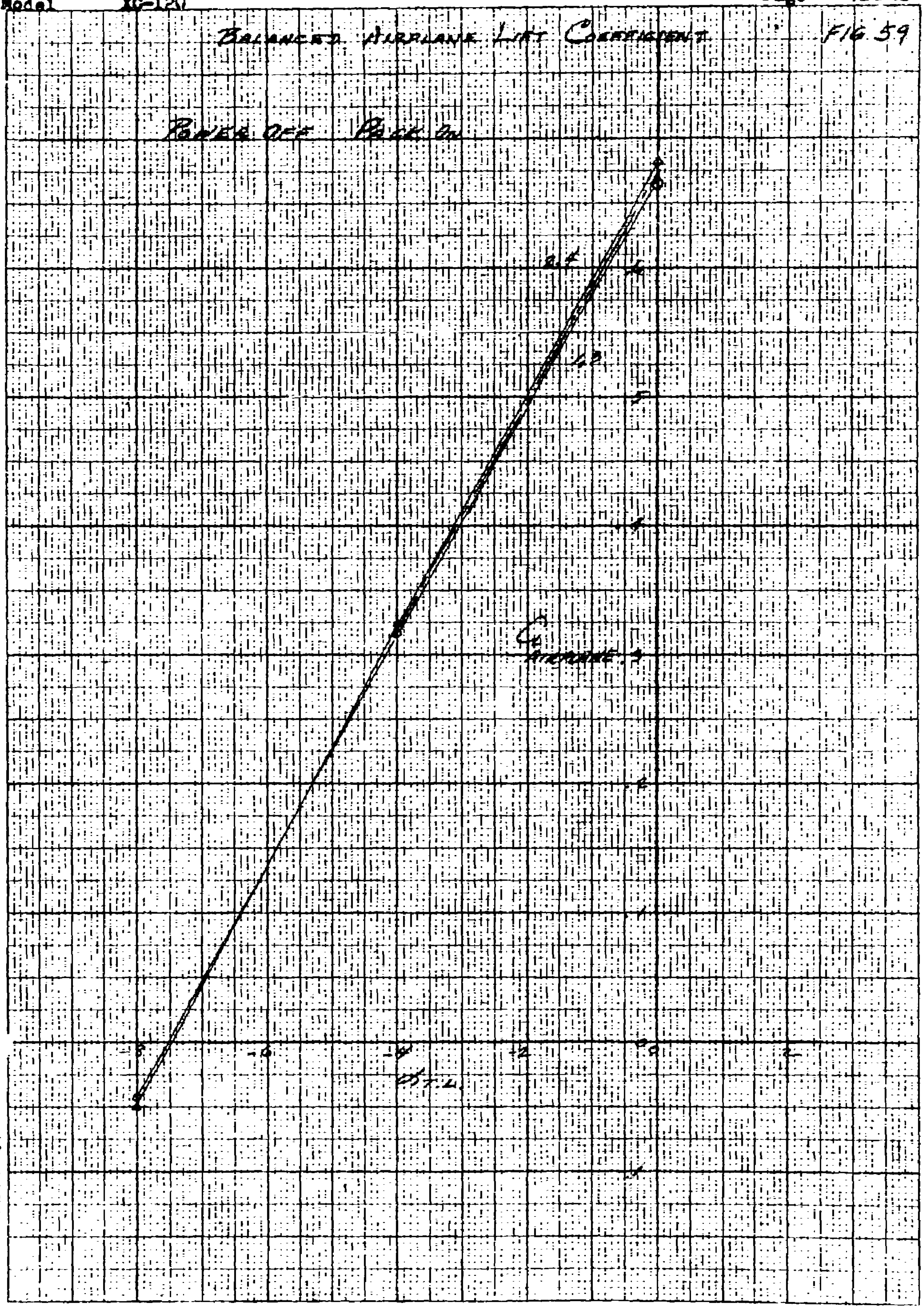
BALANCING HORIZONTAL TAIL LOADS
FLAPS & GEAR UP - POWER ON

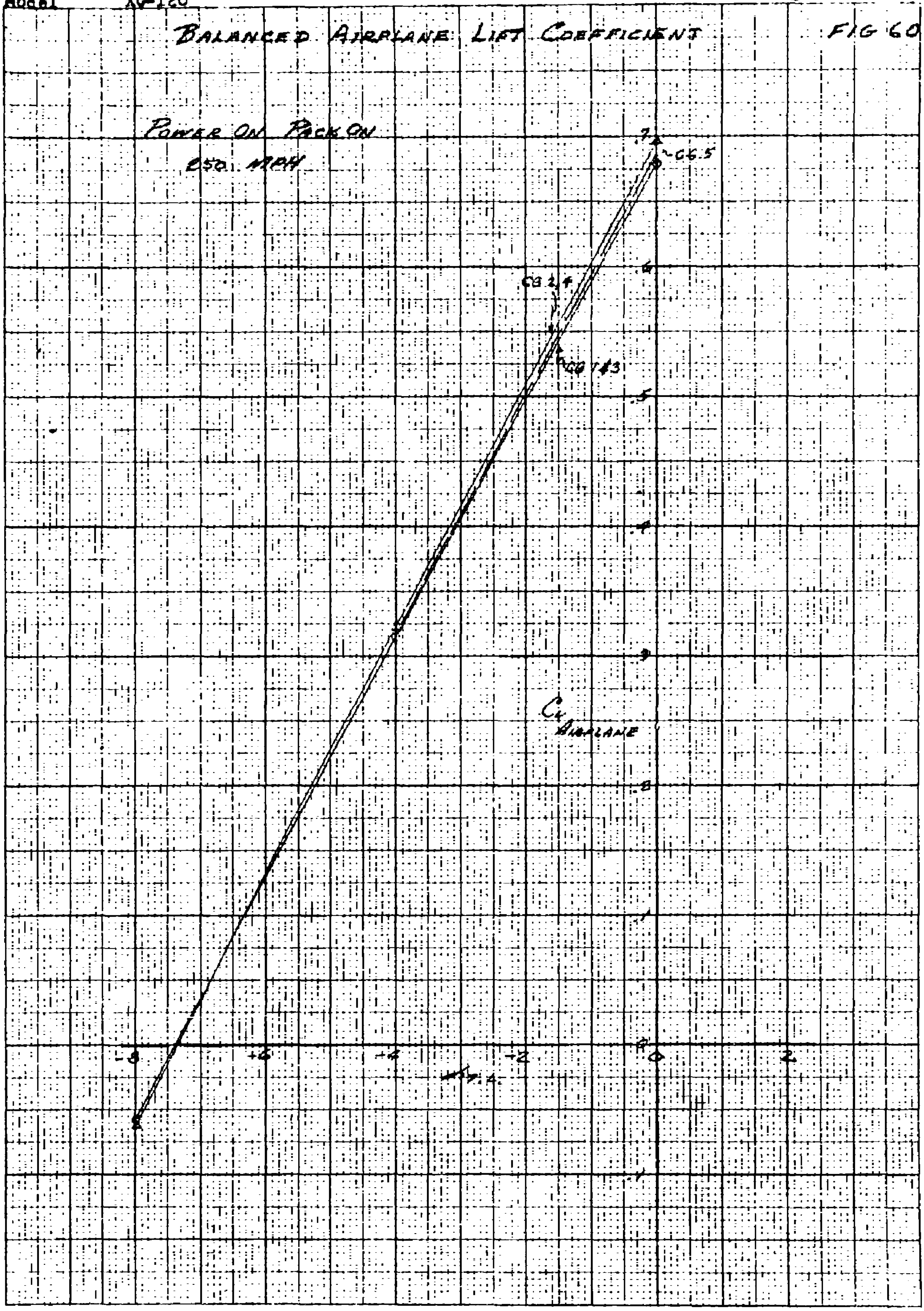
$V\sqrt{\sigma}$ = 250 mph

T_c = .056

		C.G. 1		C.G. 2		C.G. 3		C.G. 4		C.G. 5
α_{TL}	q_{HT}/q	$\eta_{HT} C_{L_{HT}}$	L_{HT}	$\eta_{HT} C_{L_{HT}}$	L_{HT}	$\eta_{HT} C_{L_{HT}}$	L_{HT}	$\eta_{HT} C_{L_{HT}}$	L_{HT}	$\eta_{HT} C_{L_{HT}}$
-12	1.000	-.167	-9250	-.218	-12,060	-.182	-10,080	-.233	-12,910	-.188
-8	1.000	-.136	-7530	-.143	-7,920	-.137	-7,590	-.145	-8030	-.138
-4	1.006	-.107	-5960	-.074	-4120	-.112	-5,660	-.065	-3620	-.091
0	1.042	-.079	-4560	-.003	-170	-.076	-4,390	-.000	-0	-.049
4	1.155	-.053	-3100	+.063	+3680	-.058	-3,390	+.057	+3330	-.010
8	1.160	-.024	-1410	+.132	+7750	-.043	-2,520	+.112	+6580	+.032

L_{HT}
-10,410
-7,640
-5,070
-2,830
-580
+1,880


BALANCED AIRPLANE LIFT COEFFICIENT
FIG 59
POWER OFF BOOM ON
C_L AIRPLANE


BALANCED AIRPLANE LIFT COEFFICIENT
FIG 60
POWER ON PACK ON
250 MPH
CG 6.5
CG 24
CG 743
C_L AIRPLANE
.7
.6
.5
.4
.3
.2
.1
0
-.1
-8
-6
-4
-2
0
2
$i_{T.A.}$

Subject:— BASIC FLIGHT CRITERIA - PACK ON

PART II-M-2b

AIRPLANE LIFT COEFFICIENT — POWER ON

FLAPS & GEAR UP — 250 MPH

						C.G. 1			C.G. 2	
α_{TL}	$C_{LA.T}$ POWER OFF	ΔC_{LWT} TWO	C_{LP} TWO	$C_{LA.T}$ POWER ON	$\frac{q_{ht}}{q}$	$\eta_{ht} C_{LHT}$	C_{LHT}	C_{LA}	$\eta_{ht} C_{LHT}$	C_{LHT}
-12	-.3738	-.0034	-.0215	-.3981	1.000	-.167	-.0399	-.4386	-.218	-.0521
-8	-.0156	.0020	-.0144	-.0260	1.000	-.136	-.0325	-.0565	-.143	-.0342
-4	.3323	.0080	-.0071	.3480	1.006	-.107	-.0257	+.3223	-.074	-.0178
0	.6946	.0140	0	.6986	1.042	-.079	-.0197	+.6189	-.003	-.0007
4	1.0288	.0200	.0071	1.0559	1.055	-.053	-.0139	+1.0425	+.063	+.0159
8	1.3626	.0274	.0144	1.4244	1.060	-.024	-.0061	+1.4183	+.132	+.0335

	C.G. 3			C.G. 4			C.G. 5		
C_{LA}	$\eta_{ht} C_{LHT}$	C_{LHT}	C_{LA}	$\eta_{ht} C_{LHT}$	C_{LHT}	C_{LA}	$\eta_{ht} C_{LHT}$	C_{LHT}	C_{LA}
-.4508	-.192	-.0485	-.4422	-.233	-.0557	-.4544	-.188	-.0450	-.4437
-.0602	-.191	-.0328	-.0588	-.145	-.0347	-.0607	-.138	-.0330	-.0590
+.3302	-.102	-.0245	+.3235	-.065	-.0156	+.3324	-.091	-.0219	+.3261
+.6979	-.076	-.0182	+.6797	-.000	-.0000	+.6986	-.049	-.0122	+.6864
+1.0718	-.058	-.0146	+1.0413	+.057	+.0144	+1.0703	-.010	-.0025	+1.0534
+1.4579	-.043	-.0109	+1.4135	+.112	+.0284	+1.4528	+.032	+.0061	+1.4325

PART II-M-2b

PROPELLER LIFT COEFFICIENT

$$C_{L_P} = 2(C_{Z_P} \cos \alpha_{TL} - C_{X_P} \sin \alpha_{TL})$$

$$V\sqrt{\sigma} = 250 \text{ mph}$$

1 EACH

α_{TL}	$\cos\alpha$	$\sin\alpha$	C_{Z_P} ONE	C_{X_P} ONE	$C_{Z_P}\cos\alpha$	$C_{X_P}\sin\alpha$	C_{L_P} (ONE)	$\frac{C_{L_P}}{(ONE)}$	$\frac{C_{L_P}}{TWO}$
-12	.9781	-.2079	-.0560	-.1459	-.0548	.03033	-.0871	-.01074	-.02148
8	.9903	-.1392	-.0381	-.1459	-.0380	.0203	-.0563	-.00718	-.01436
-4	.9976	-.0698	-.0187	-.1459	-.01868	.01017	-.02885	-.00356	-.00712
0	1.000	0	0	-.1459	0	0	0	0	0
4	.9976	.0698	.0187	-.1459	.01868	-.01017	.02885	.00356	.00712
8	.9903	.1392	.0381	-.1459	.0380	-.0203	.0563	.00718	.01436

PART II-M-2-c

c. Total Horizontal Tail Loads with Gust

Power Off

Gross Weight 42,136 lbs.

C. G.	C_{L_A}	α_{TL}	Balancing Load	Gust Load	Maximum Total Load
5	.1820	-5.5°	-5700	± 6140	-11840

Gross Weight 64,000 lbs.

C. G.	C_{L_A}	α_{TL}	Balancing Load	Gust Load	Maximum Total Load
1		-4.45	-6000		-12510
2		-4.5	-4200		-10710
3	.2762	-4.45	-5300	± 6510	-11810
4		-4.5	-3550		-10060
5		-4.47	-5200		-11710

Gross Weight 74,000 lbs.

C. G.	C_{L_A}	α_{TL}	Balancing Load	Gust Load	Maximum Total Load
1		-3.95	-5820		-12460
2		-4.0	-3770		-10410
3	.320	-3.95	-5100	±6640	-11740
4		-4.0	-3050		- 9690
5		-3.97	-4870		-11510

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PART II-M-2c (Cont.)

Total Horizontal Tail Loads with Gust

Power On

Gross Weight 42,136 lbs.

C. G.	C_{L_A}	α_{TL}	Balancing Load	Gust Load	Total Load
5	.1820	-5.5°	-5950	± 6140	-12090

Gross Weight 64,000 lbs.

C. G.	C_{L_A}	α_{TL}	Balancing Load	Gust Load	Total Load
1		-4.45	-6150		-12660
2		-4.5	-4600		-11110
3	.2762	-4.45	-5900	± 6510	-12410
4		-4.5	-4150		-10660
5		-4.47	-5350		-11860

Gross Weight 74,000 lbs.

C. G.	C_{L_A}	α_{TL}	Balancing Load	Gust Load	Total Load
1		-3.95	-5960		-12600
2		-4.00	-4120		-10760
3	.320	-3.95	-5680	± 6640	-12320
4		-4.00	-3620		-10260
5		-3.97	-5070		-11710

The maximum horizontal load with gust (balancing tail load plus gust load) at 250 mph is -12,660 lbs. for a gross weight of 64,000 lbs. - power on.

PART II-M-2c

The elevator deflection for balance for the above critical condition is obtained as follows:

$$\delta_e = \frac{\eta_{ht}\ C_{L_{HT}} - .0450\,\alpha_{ht}}{.0246}$$

for C. G. location (1) $\alpha_{TL} = -4.45°$

balancing $\eta_{ht}\ C_{L_{HT}} = -.1110$ figure 36

$\alpha_{ht} = -5.0°$ figure 63 reference (1)

$$\delta_e = \frac{-.1110 + .2250}{.0246} = 4.63°$$

PART II-M-3

3. Maneuvering Loads

The maximum maneuvering horizontal tail load for design gross weight of 64,000 lbs. at the maneuvering speed ($V\sqrt{\sigma}$ = 184 mph) and at the maneuvering load factor (3.0 n) was determined in reference (8) and is 11,131 lbs. for the upper center of gravity location at 35% MAC_w.

The angle of attack of the horizontal tail and the elevator deflection for the above condition are obtained by adding the incremental angle of attack of the horizontal tail and the incremental elevator deflection obtained in maneuvering to the angle of attack of the horizontal tail and elevator deflection for the initial balanced condition.

The variation of the balanced airplane lift coefficient with angle of attack of the thrust line for $V\sqrt{\sigma}$ = 184 mph flaps and gear up, power on is determined in the following table for various center of gravity locations.

PART II-M-3

BALANCED AIRPLANE LIFT COEFFICIENT

FLAPS & GEAR UP - 184 MPH - POWER ON

T_c = .132

		C.G. 1				C.G. 2			C.G. 3	
α_{TL}	$\frac{S_{HC}}{S_W} \cdot \frac{\ell_{HC}}{\ell}$	$\eta_{HC} C_{L_{HT}}$	$\underline{C_{L_{HT}}}$	$C_{L_{A-T}}$ POWER ON	C_{L_A}	$\eta_{HC} C_{L_{HT}}$	$\underline{C_{L_{HT}}}$	C_{L_A}	$\eta_{HC} C_{L_{HT}}$	$\underline{C_{L_{HT}}}$
12	.2222	-.163	-.0390	-.4098	-.4488	-.214	-.0512	-.4610	-.187	-.0427
-8	.2392	-.133	-.0318	-.0288	-.0606	-.139	-.0332	-.0620	-.143	-.0342
-4	.2457	-.121	-.0245	.3475	.3227	-.064	-.0157	.3318	-.106	-.0266
0	.2631	-.072	-.0143	.7142	.6953	.022	.0035	.7141	-.076	-.0205
4	.2626	-.047	-.0126	1.0804	1.0678	.064	.0112	1.0976	-.061	-.0164
8	.2715	-.020	-.0054	1.4562	1.4508	.129	.0350	1.4912	-.047	-.0128

$C_{L_{A-T}}$ POWER ON	C_{L_A}
-.4098	-.4545
-.0288	-.0630
.3475	.3215
.7142	.6937
1.0804	1.0640
1.4562	1.4434

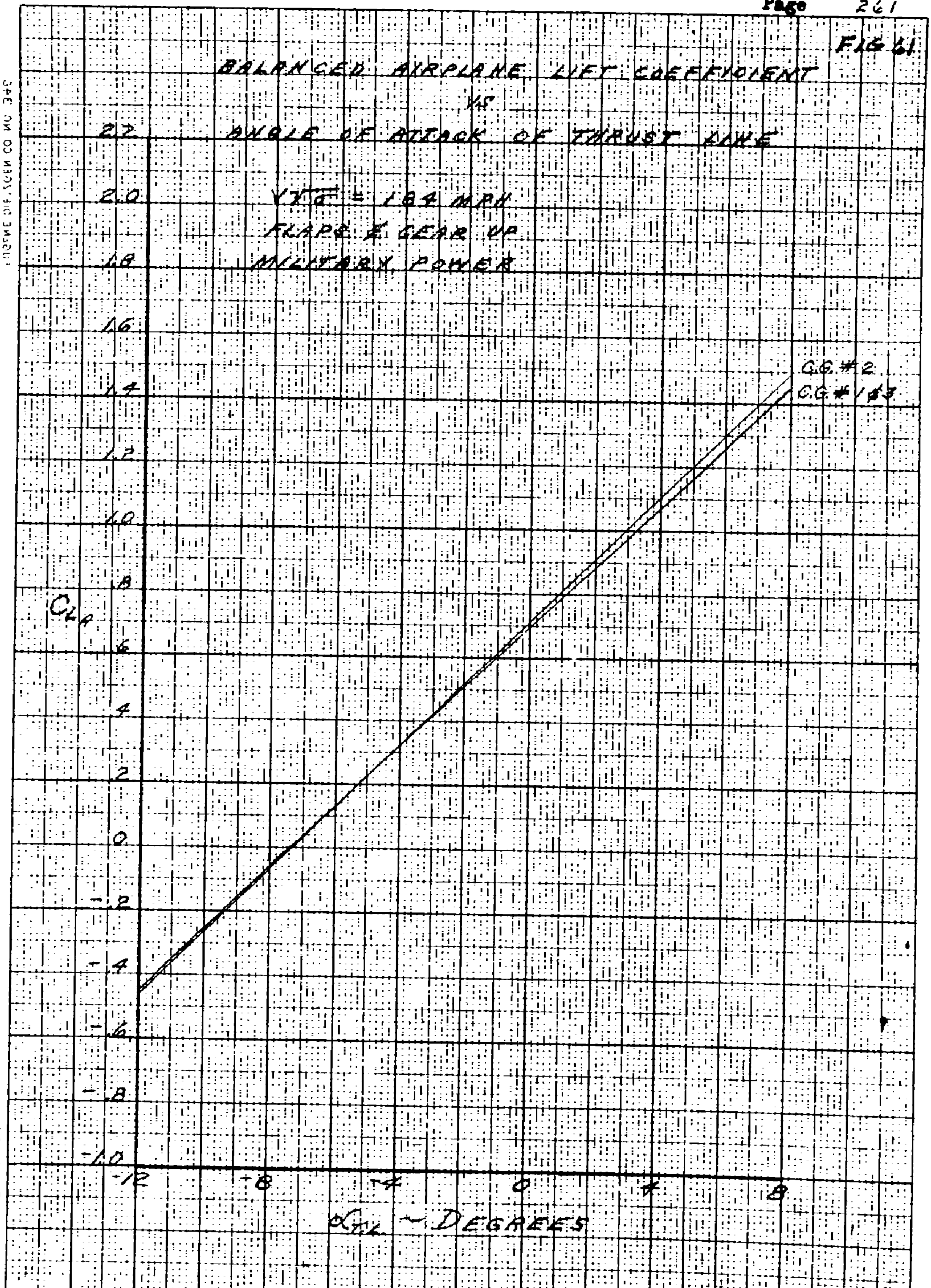


FIG 61
BALANCED AIRPLANE LIFT COEFFICIENT
VS
ANGLE OF ATTACK OF THRUST LINE
V√σ = 184 MPH
FLAPS & GEAR UP
MILITARY POWER
C.G. #2
C.G. #1 & 3
C_{LA}
2.2
2.0
1.8
1.6
1.4
1.2
1.0
.8
.6
.4
.2
0
-.2
-.4
-.6
-.8
-1.0
-12
-8
-4
0
4
8
α_{TL} ~ DEGREES

PART II-M-3

<u>For the Initial Balance Condition - Power On</u>

$V\sqrt{\sigma}$ = 184 mph   Gross Weight = 64,000 lbs.

$$C_{L_A} = .5072$$

C. G. Location	α_{TL}	α_{ht}	$\eta_{ht}\, C_{L_{HT}}$	δ_e
1	-2.0	-3.47	-.0890	2.73
2	-2.1	-3.53	-.0325	5.14
35% MAC_w Upper	-2.1*	-3.53	-.00425*	6.28
3	-2.0	-3.47	-.0950	2.49

α_{ht} is from figure 63 reference (1)

$\eta_{ht}\, C_{L_{HT}}$ is from figure 35

$$\delta_e = \frac{\eta_{ht}\, C_{L_{HT}} - .0450\,\alpha_{ht}}{.0246}$$

The incremental values of α_{ht} and δ_e obtained in the maneuvering condition at the peak load point are obtained from table XVII of reference (8)

C. G. Location	Δ_{ht}	$\Delta\delta_e$
35% MAC_w Upper	7.72	0

The total values for the critical condition are then

C. G. Location	α_{ht}	δ_e
35% MAC_w Upper	4.19	6.28

* Values for 35% MAC_w are extrapolated values

PART II-M-4

4. Summary of Design Horizontal Tail Loads

Condition	NLAA V_D n = -1.14 Power Off	NLAA V_D n = -1.14 Power On	Gust 50 fps Power On	Maneuvering n = 3.0 Power On
Gross Weight	64,000	64,000	64,000	64,000
$V\sqrt{\sigma}$,mph	313	313	250	184
C. G. Location	2	2	1	35% MAC_w Upper
Load, lbs.	-13,789	-14,873	-12,660	11,131
α_{ht} deg.	-7.99	-7.94	-7.61	4.19
δ_e deg.	8.13	7.54	4.63	6.28
q_{ht}/q	1.00	1.00	1.00	1.081

Pressure distributions for the above conditions are presented in reference (7).

PART II - N

N. ELEVATOR TAB LOADS

This section covers the determination of the elevator tab loads. The maximum possible tab loads are determined as being those obtained with maximum possible tab deflection at V_D.

Summary of Design Tab Loads.

Elevator trim tab = 83.5 lbs./sq.ft.

Elevator spring tab = 73.5 lbs./sq.ft.

PART II - N

1. ELEVATOR TRIM TAB

As the maximum tab load will occur at maximum surface deflection at V_D (313 mph) the trim tab load will be calculated for this condition. This is a conservative procedure.

The elevator trim tab hinge moment is determined as follows:

$$H_{e_{tt}} = \frac{d\,CH_{e_{tt}}}{d\,\delta_{e_{tt}}}\,\delta_{e_{tt}} \times q \times \frac{q_{ht}}{q} \times S_{e_{tt}} \times c_{e_{tt}}$$

$S_{e_{tt}}$ = 5.44 Sq.Ft. $c_{e_{tt}}$ = .5 Ft.

$\delta_{e_{tt}}$ maximum = 25° Page 15 Reference (1)

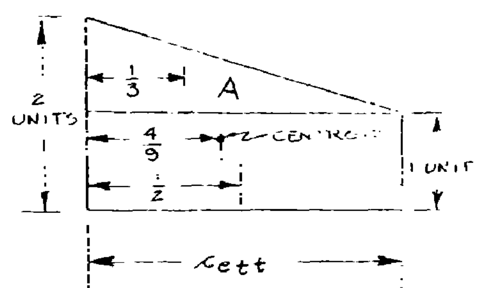
q = 248 $\frac{q_{ht}}{q}$ = 1.00

$$H_{e_{tt}} = -.006 \times 25 \times 248 \times 1.00 \times 5.44 \times .5$$

$$H_{e_{tt}} = -101 \text{ ft. lbs.}$$

Using the following load distribution as specified in reference 1803 the tab load is determined.

Centroid of Loading:



$-y\,N_t = H_t$ $N_t = \frac{H_t}{-.222}$ H_t = Tab hinge moment Ft./lbs.

$N_t = \frac{H_t}{-y}$ $N_t = -4.5\,H_t$ N_t = Normal Force on Tab lbs.

$N_t = \frac{H_t}{-4/9 \; .5'}$ y = Force Arm

A = Area

PART II - N

Therefore

$$Ne_{tt} = -4.5 \times He_{tt} \quad -4.5 \times -101.$$

$$Ne_{tt} = 455 \text{ lbs.}$$

$$\frac{455 \text{ lbs.}}{5.44 \text{ Sq.Ft.}} = 83.6 \text{ lbs./Sq.Ft. of Tab Area}$$

2. ELEVATOR SPRING TAB

The maximum load on the elevator spring tab (occurring with maximum deflection at V_D) is the same as that determined in Part II-N-2 of reference (5) and is 73.5 lbs./sq.ft. of tab area.

PART II-O

O. Fuselage, Booms, Cowling, Doors, and Miscellaneous

1. Fuselage and Booms

Total loads amd moments on fuselage and booms are presented for all conditions investigated in Part II-H. The fuselage and booms are not critical from local airloads but from weight distribution and airloads of attached component parts.

2. Cowling

Engine cowl loads are in accordance with the pressure distribution of reference (2)-5B.

3. Doors

Local door loads were obtained from flight measurements on the C-82 and C-119 airplanes, except for the front nose doors which are presented immediately following.

4. Front Cargo Doors

The front cargo doors of the XC-120 pack are of the clamshell type and open out from the center line of the airplane, providing nose loading provisions similar to the rear loading provisions incorporated in the C-82, C-119B, and XC-120 airplanes. The nose doors are particularly important since failure of these doors in flight may mean loss of the airplane. For this reason pressure distributions over the doors were obtained from 1/32 scale model tests in the Wright Field Five Foot Wind Tunnel; and it is the purpose of this section to present the final door loads as obtained from the integrated pressure distributions.

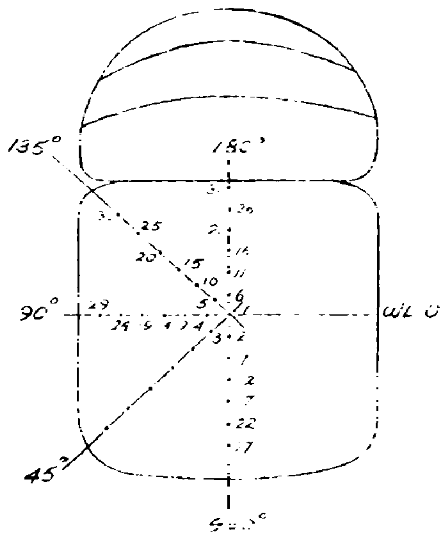
a. Method of Analysis

Source of Data - Pressure distributions obtained during additional tests on 1/32 scale XC-120 model in Wright Field Five Foot Wind Tunnel. Preliminary data.

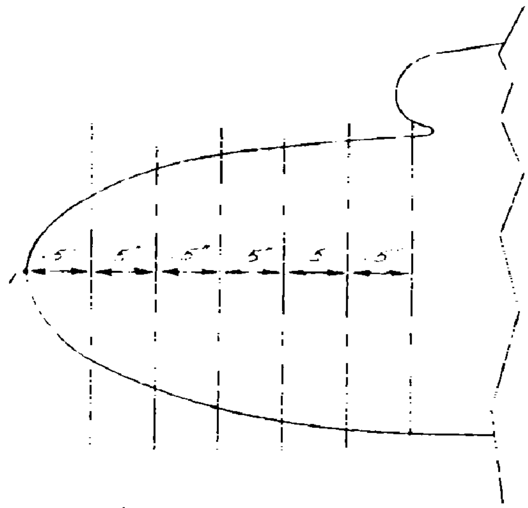
Model - Full Scale Relationship - Figure 62 presents a sketch of the model showing location of pressure tubes and tabular comparison of model and full scale stations.

FIG. 62

MODEL - FULL SCALE RELATIONSHIP



MODEL PRESSURE TUBE LOCATIONS



MODEL SCALE

MODEL STATION	FULL SCALE STATION
0	-18
5	-2
10	14
15	30
20	46
25	62
30	78

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REPORT NO. R107-014 FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION PAGES PAGE 269

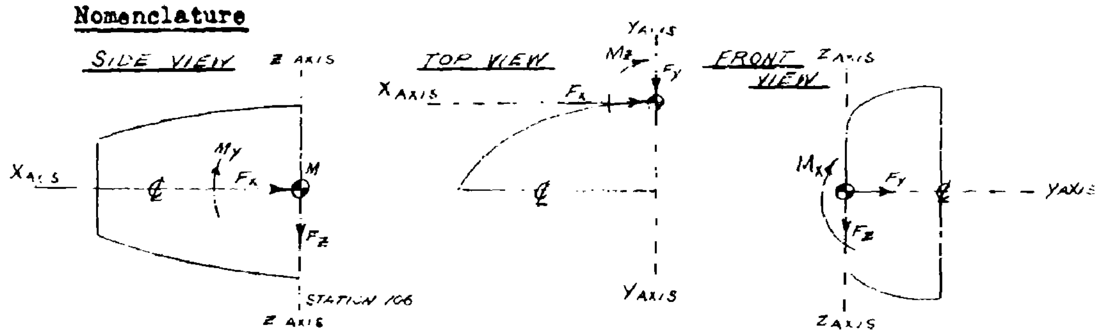
MODEL XC-120 PREPARED BY CHECKED BY APPROVED BY

DATE 15 March 1949

Subject: BASIC FLIGHT CRITERIA - PACK ON REVISED

PART II-0-4a

Nomenclature



Moment Reference Point: Station 106
Water Line 0
Butt Line 66.7

All Forces and Moments Positive as Indicated Above

$\frac{\Delta P}{q}$	- non-dimensional pressure coefficient	
F_Y	- total side force, lbs.	positive towards door
F_Z	- total vertical force, lbs.	positive down
F_X	- total aft force, lbs.	positive aft
M_Y	- moment about Y axis, in lbs.	positive clockwise
M_Z	- moment about Z axis, in lbs.	positive clockwise
M_X	- moment about X axis, in lbs.	positive clockwise
f_Y	- section side force, lbs.	positive towards door
f_Z	- section vertical force, lbs.	positive down
f_X	- section aft force, lbs.	positive aft
x	- distance along X axis, in.	positive aft station - 18
z	- distance along Z axis, in.	positive above water line 0
y	- distance along Y axis, in.	positive outboard butt line 0
θ	- angle of line of pressure orifices on wind tunnel model measured positive clockwise from butt line zero below water line zero (figure 62).	

Subscripts

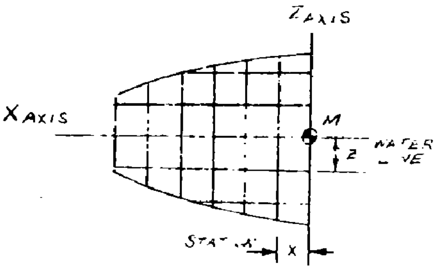
B. L. - butt line
S - station
W. L. - water line

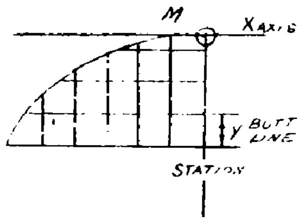
PART II-0-4a (Cont.)

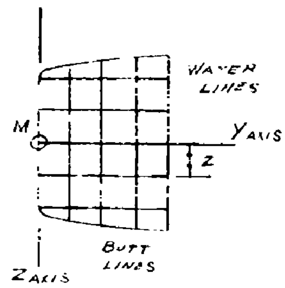
Procedure For Calculation Of Component Loads

Assume $\frac{\Delta p}{q}$ Inside = Static

Side **Top** **Front**

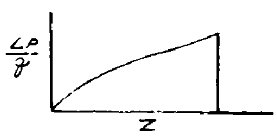






Side (Y) Force
+ in.

1. integrate $\frac{\Delta p}{q}$ vs. z for various stations



2. plot integrated values versus station and integrate again for final load

Lift (Z) Force
+ Down

1. integrate $\frac{\Delta p}{q}$ vs y for various stations
2. plot integrated values versus station and integrate again for final load

Drag (X) Force
+ Aft

1. integrate $\frac{\Delta p}{q}$ vs z for various B. L.
2. plot integrated values versus B. L. and integrate again for final loads

From the above analysis the following formulae may be derived:

PART II-0-4a

Drag = X Forces

$$f_X = \left(\int \frac{\Delta p}{q} dy\right)_{WL} q = \left(\int \frac{\Delta p}{q} dz\right)_{BL} q$$

$$F_X = \int\left(\int \frac{\Delta p}{q} dy\right)_{WL} q \, dz = \int\left(\int \frac{\Delta p}{q} dz\right)_{BL} q \, dy$$

Lift = Z Forces

$$f_Z = \left(\int \frac{\Delta p}{q} dy\right)_{S} q = \left(\int \frac{\Delta p}{q} dx\right)_{BL} q$$

$$F_Z = \int\left(\int \frac{\Delta p}{q} dy\right)_{S} q \, dx = \int\left(\int \frac{\Delta p}{q} dx\right)_{BL} q \, dy$$

Side Forces = Y Forces

$$f_Y = \left(\int \frac{\Delta p}{q} dz\right)_{S} q = \left(\int \frac{\Delta p}{q} dx\right)_{WL} q$$

$$F_Y = \int\left(\int \frac{\Delta p}{q} dz\right)_{S} q \, dx = \int\left(\int \frac{\Delta p}{q} dx\right)_{WL} q \, dz$$

The calculations were made using the following formulae:

$$F_X = \int\left(\int \frac{\Delta p}{q} dz\right)_{BL} q \, dy = \int f_{X_{BL}} \, dy$$

$$F_Z = \int\left(\int \frac{\Delta p}{q} dy\right)_{S} q \, dx = \int f_{Z_S} \, dx$$

$$F_Y = \int\left(\int \frac{\Delta p}{q} dz\right)_{S} q \, dx = \int f_{Y_S} \, dx$$

The final results of calculations are presented in Part II-0-4b. Figures 72 through 81 present the section loads for use in determining local skin loads and figures 63 through 71 present final total loads.

PART II-C-4a

Procedure For Calculation of Moments

$$M_Z = \int\left(\int\frac{\Delta p}{q}\,dz\right)_{BL} q\,y\,dy + \int\left(\int\frac{\Delta p}{q}\,dz\right)_{S} q\,x\,dx$$

$$M_Y = \int\left(\int\frac{\Delta p}{q}\,dy\right)_{WL} q\,z\,dz + \int\left(\int\frac{\Delta p}{q}\,dy\right)_{S} q\,x\,dx$$

$$M_X = \int\left(\int\frac{\Delta p}{q}\,dx\right)_{WL} q\,z\,dz + \int\left(\int\frac{\Delta p}{q}\,dx\right)_{BL} q\,y\,dy$$

$$q\left(\frac{\Delta p}{q}\,dz\right)_{BL} = f_{X\,BL}$$

$$q\left(\frac{\Delta p}{q}\,dz\right)_{S} = f_{Y_S}$$

$$q\left(\frac{\Delta p}{q}\,dy\right)_{WL} = f_{X_{WL}}$$

$$q\left(\frac{\Delta p}{q}\,dy\right)_{S} = f_{Z_S}$$

$$q\left(\frac{\Delta p}{q}\,d_x\right)_{WL} = f_{y_{WL}}$$

$$q\left(\frac{\Delta p}{q}\,d_x\right)_{BL} = f_{Z_{BL}}$$

$$M_Z = \int f_{X_{BL}}\,y\,dy + \int f_{Y_S}\,x\,dx$$

$$M_Y = \int f_{X_{WL}}\,z\,dz + \int f_{X_S}\,x\,dx$$

$$M_X = \int f_{Y_{WL}}\,z\,dz + \int f_{Z_{BL}}\,y\,dy$$

DATE 15 March 1949

PART II-O-4a (Cont.)

Now the following section loads have been previously evaluated in determining total loads

$$f_{X_{BL}}$$

$$f_{Y_S}$$

$$f_{Z_S}$$

Now since all of the required terms were not previously evaluated, find equivalent espressions in the evaluated data:

$$f_{X_{WL}} \; z \, dz = \left(\int \frac{\Delta p}{q} \, dy\right)_{WL} q \, z \, dz$$

Can be approximated by estimating C.P. at each Butt Line for $f_{X_{BL}} = \left(\int \frac{\Delta p}{q} \, dz\right)_{BL} q$ multiplying $f_{X_{BL}}$ by distance from Y axis to C.P. and plot vs. B. L. (Y) and integrate

$$f_{Y_{WL}} \; z \, dz = \left(\int \frac{\Delta p}{q} \, dx\right)_{WL} q \, z \, dz$$

Approximate by estimating C.P. at each station for $f_{Y_S} = \left(\int \frac{\Delta p}{q} \, dz\right)_S q$ multiply f_{Y_S} by distance from X axis to C.P. and plot versus Station (X) and integrate

$$f_{Z_{BL}} \; y \, dy = \left(\int \frac{\Delta p}{q} \, dx\right)_{BL} q \, y \, dy$$

Approximate by estimating C. P. at each station for

$f_{Z_S} = \left(\int \frac{\Delta p}{q} \, dy\right)_S q$ multiply f_{Z_S} by distance from X axis to C. P. and plot versus Station (X) and integrate.

Calculations were made for all the desired moments and results are presented in figures 67 through 71 of Part II-O-4b.

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REPORT NO. R107-014 FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION | PAGES | PAGE 274

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

DATE 15 March 1949

Subject: BASIC FLIGHT CRITERIA - PACK ON

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PART II-O-4a

Data Required

From the wind tunnel data and geometric data the following plots were constructed to aid in integrated final loads:

Title	Stations	θ	α_{TL}, deg.	ψ, deg.	Water Line	Butt Line
$\frac{\Delta p}{q}$ vs θ	-2,14,30, 46,62, & 78		0° and 8° for each station	-4° to -16° for each plot	-	-
y vs station (projected water lines)	Top Bottom	- -	- -	- -	0 to 50 -10 to -50	
z vs. station (projected butt lines)	Top Bottom	- -	- -	- -	- -	7 to 57 7 to 57
y vs. z (fuselage ordinates)	-2,14,30, 46,62, & 78	-	-	-	-	-
$\frac{\Delta p}{q}$ vs. station		0°,45° 90°,135° 180°	0° and 8° for each value of θ	-4° to +16° for each plot	-	-
$\frac{\Delta p}{q}$ vs z	-	-	0° and 8° for each butt line	-4° to +16° for each plot	-	0,20,25, 30,35,40, 45,50
	-2,14,30, 46, & 62	-	0° and 8° for each station	-4° to +16° for each plot	-	-
$\frac{\Delta p}{q}$ vs y	-2,14,30, 46, & 62	-	0° and 8° for each station	-4° to +16° for each plot	-	-

These supplementary figures were used to determine section loads, but are not presented in this report since the final section and total loads present the complete picture.

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PART II-C-4b

b. Nose - Door Load and Moment Coefficients

The accompanying figures present the final aerodynamic load and moment coefficients on the right hand nose door of the XC-120 airplane for various angles of yaw and angles of attack. The figures show the ratio of the load to q, the dynamic pressure. The load in pounds is obtained by multiplying by q. Similarly, the moments in inch pounds are obtained by multiplying the value obtained from the figures by q.

The loads are assumed to act through the point; station 106, waterline 0, butt line 66.7. The moments are given about a system of axis through this point. The loads are the same for any moment center and the moments may be transferred to any standard procedure.

The distribution of the loads in longitudinal (x) direction, lateral (y) direction and vertical (z) direction are also shown. The load at any point is indicated by a lower case f with a subscript indicating the direction. The total loads are indicated by a capital F.

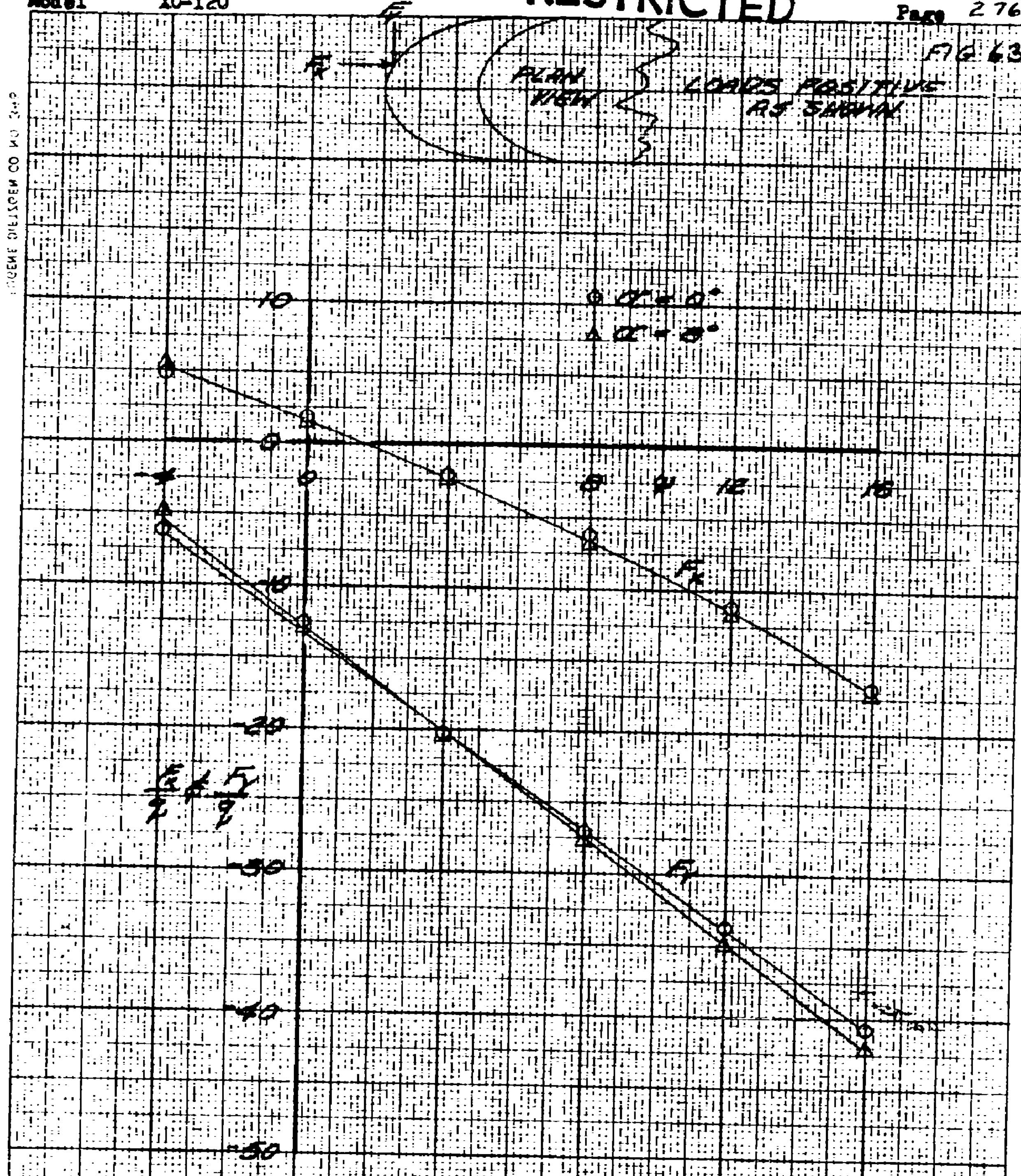
It is noted that there are two figures for some of the coefficients. In each case the first figure was obtained exactly from integration of wind tunnel pressure data and the second figure presents extrapolations beyond the scope of the actual test data.

f or F - force in pounds

q - free stream dynamic pressure in pounds per square foot

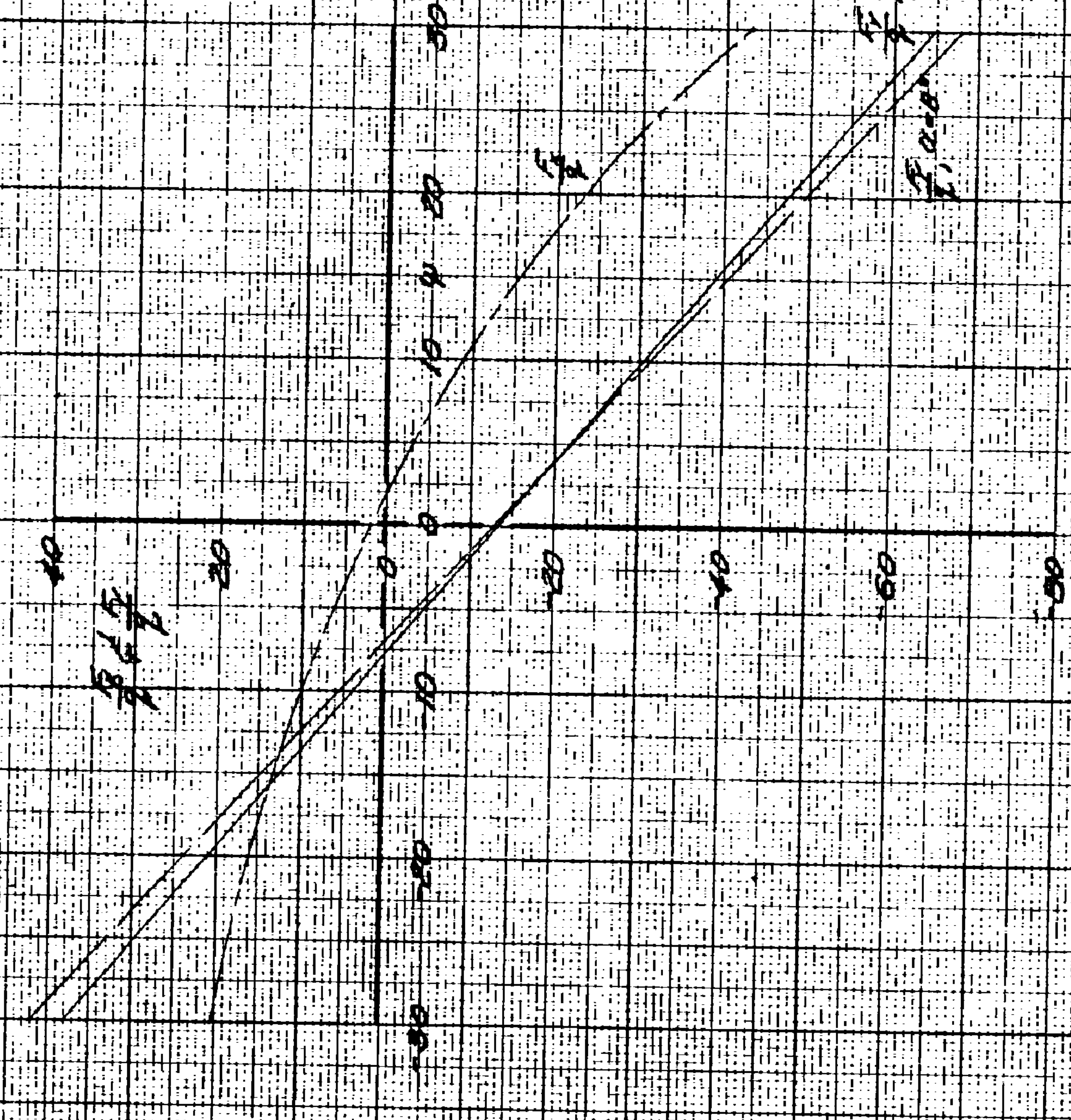
M - moment in inch pounds

FIG 63



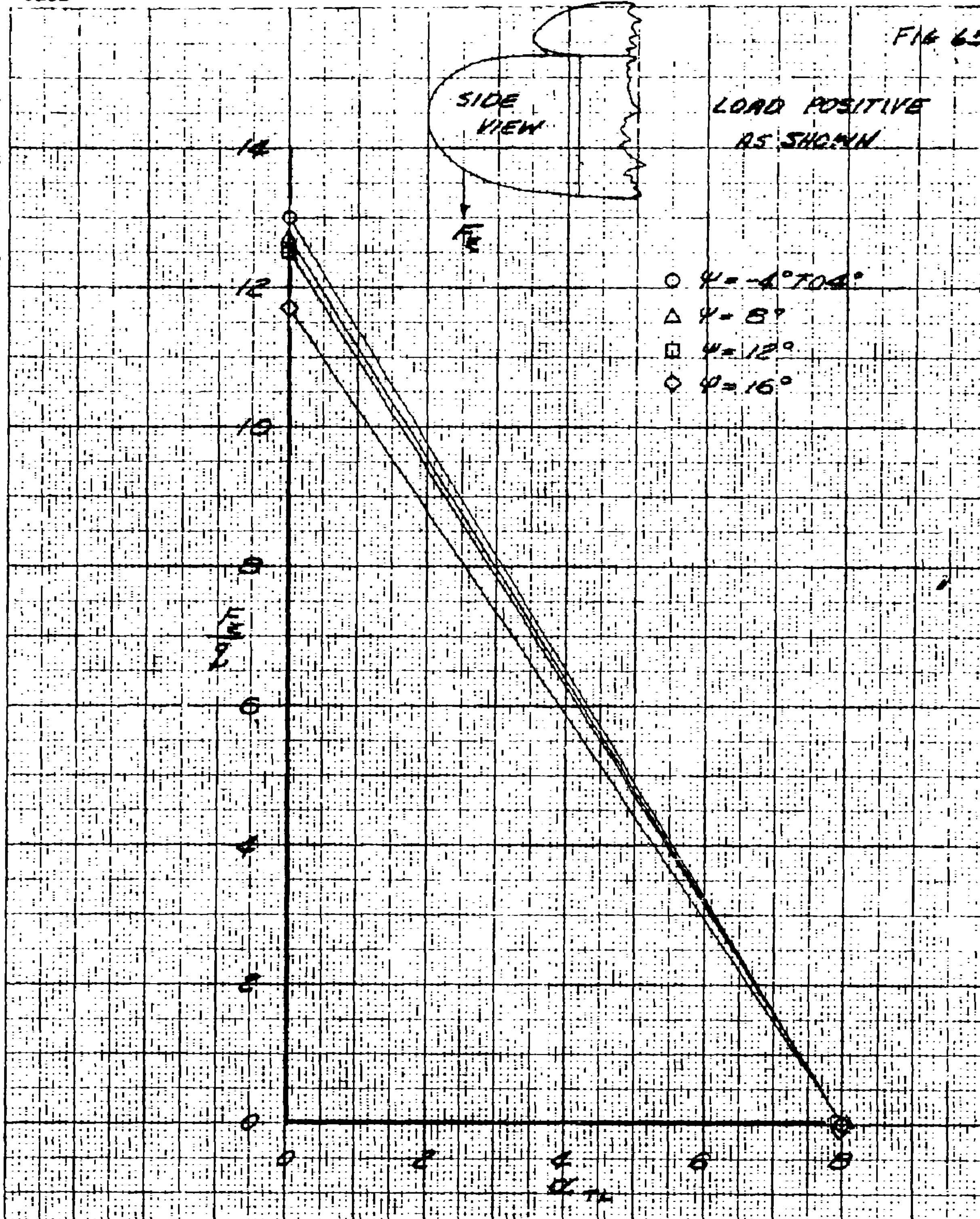
XC-120
Nose Door Total Drag & Side Loads vs ψ
(Right Door Only)
Wind Tunnel Data

FIG 64



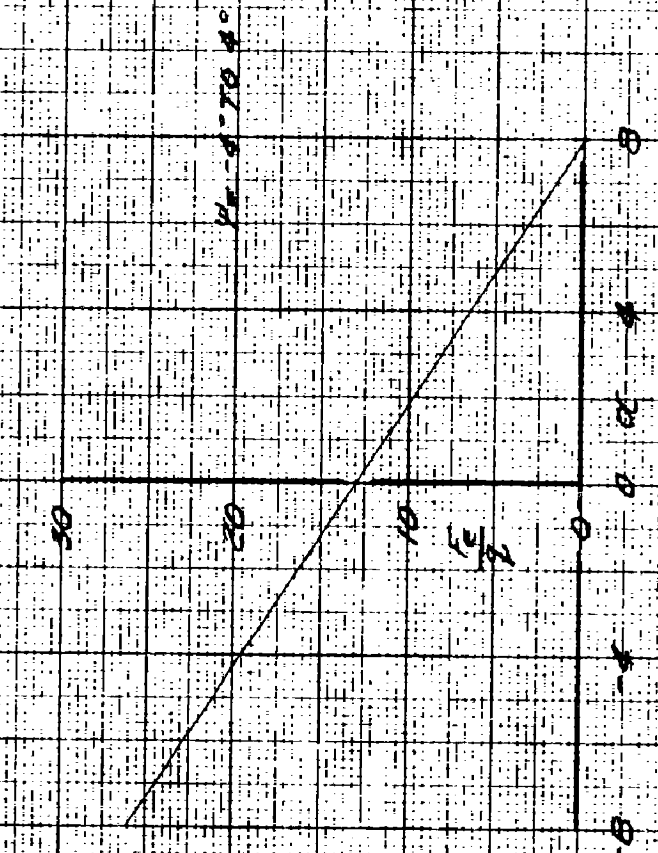
XC-120
Nose Door Total Drag And Side Loads vs ψ
(Right Door Only)
Extrapolated Wind Tunnel Data

FIG 65



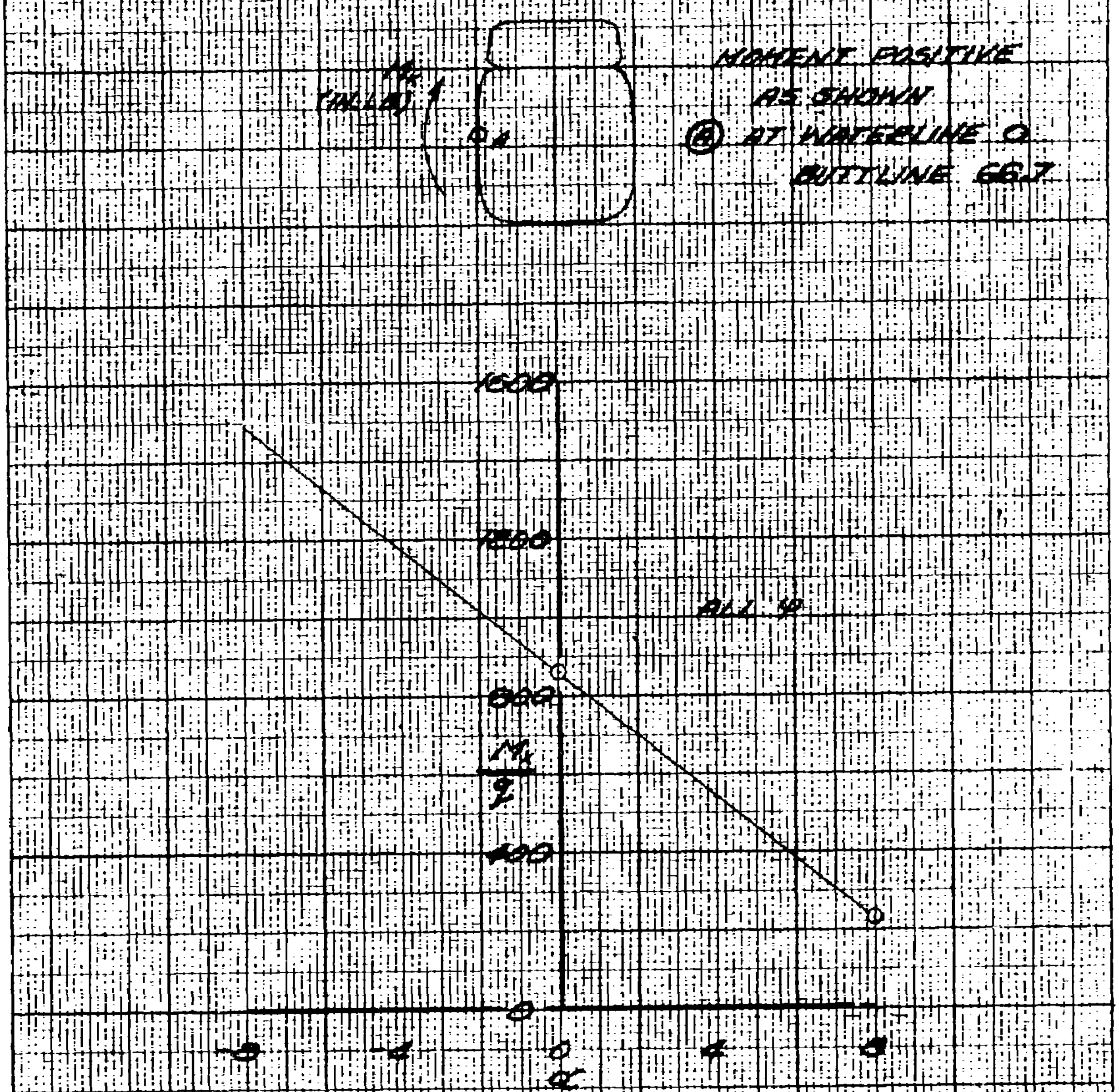
XC-120
NOSE DOOR TOTAL LIFT LOADS VS α_{TL}
(RIGHT DOOR ONLY)
WIND TUNNEL DATA

FIG 66



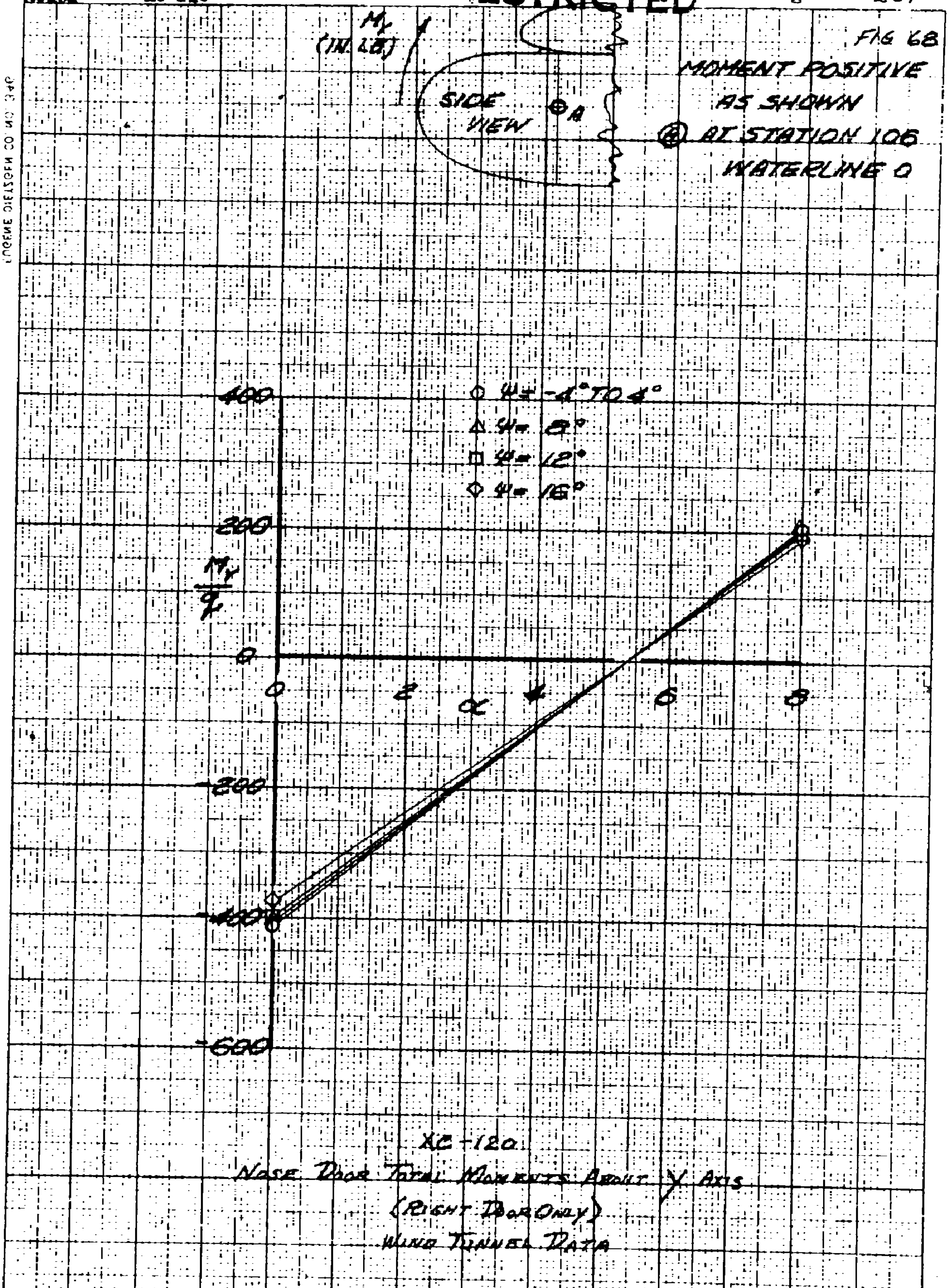
XC-120
NOSE DOOR TOTAL LIFT LOAD vs α
(RIGHT DOOR ONLY)
EXTRAPOLATED WIND TUNNEL DATA

FIG 67



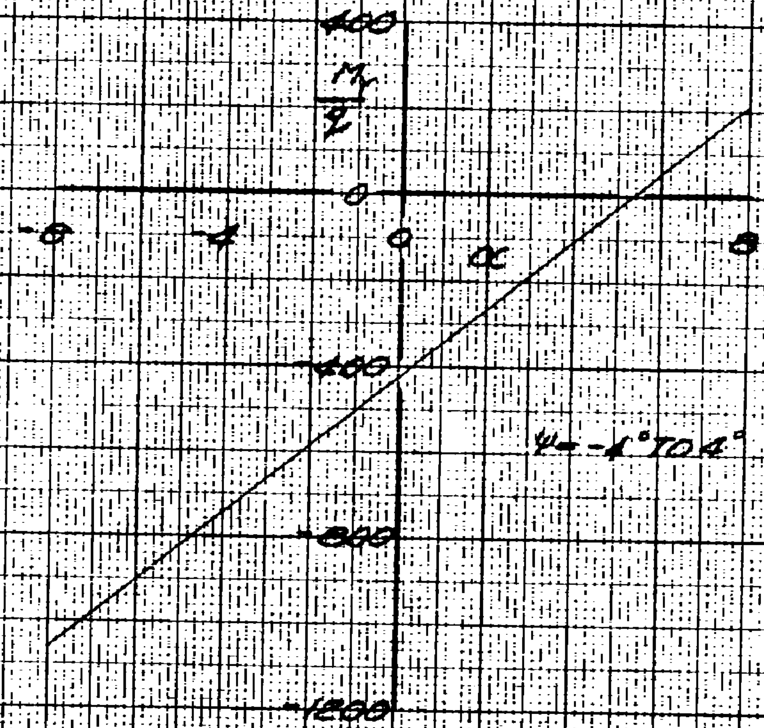
XC-120
Nose Door Total Moments About X Axis
(Right Door Only)
Extrapolated Wind Tunnel Data

FIG 68



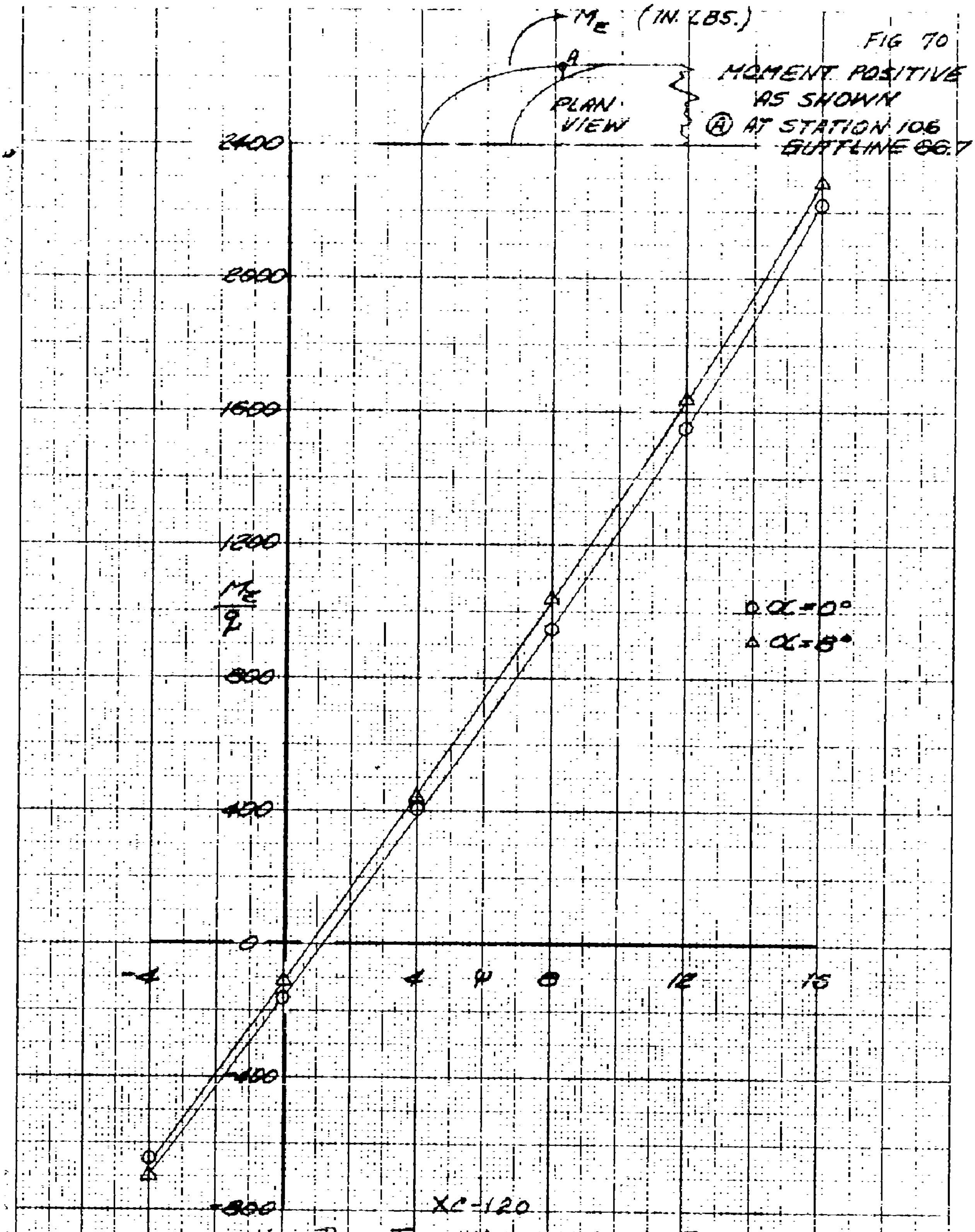
XC-120
Nose Door Total Moments About Y Axis
(Right Door Only)
Wind Tunnel Data

FIG 69



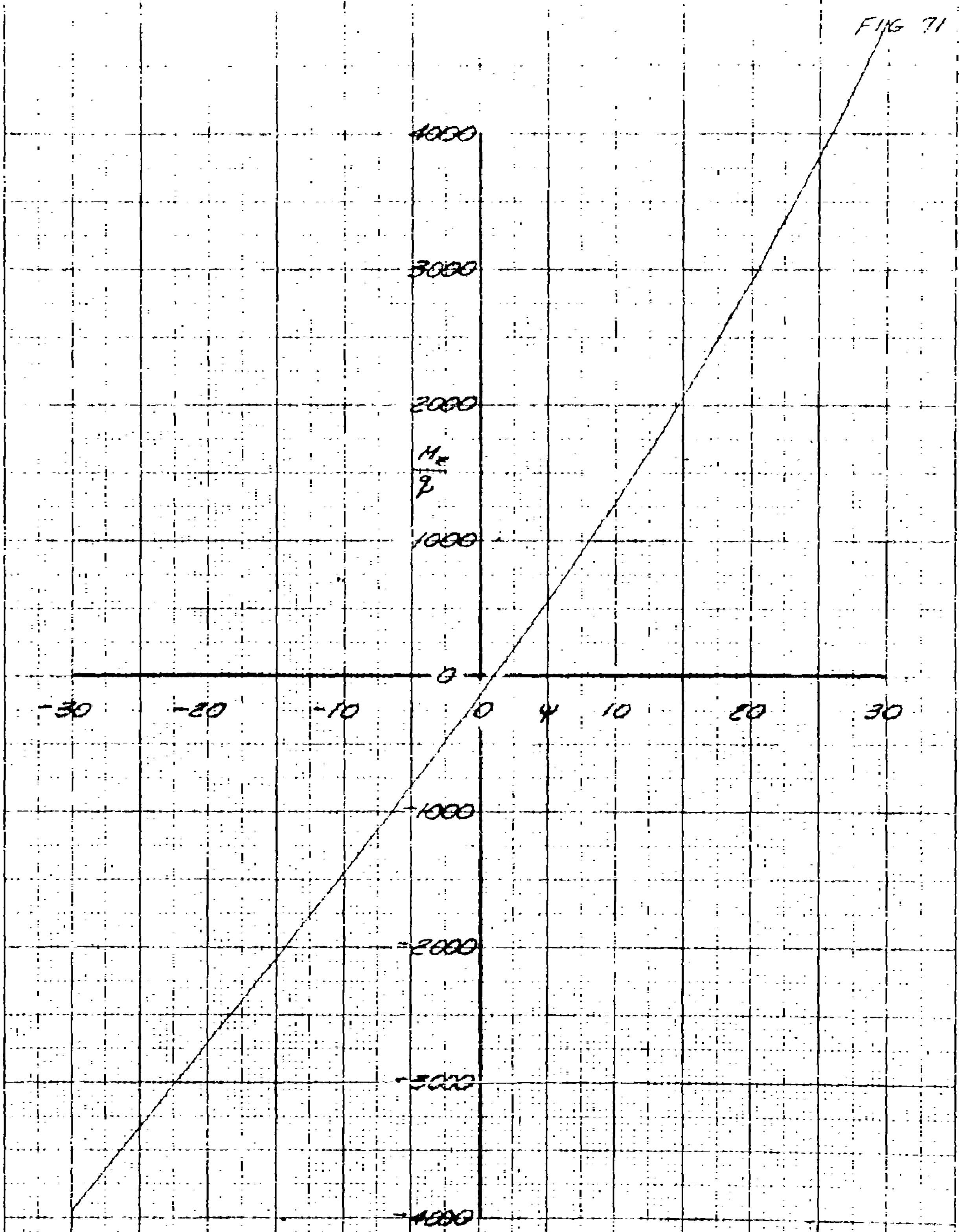
XC-120
NOSE DOOR TOTAL MOMENTS ABOUT Y AXIS
(RIGHT DOOR ONLY)
EXTRAPOLATED WIND TUNNEL DATA

FIG 70



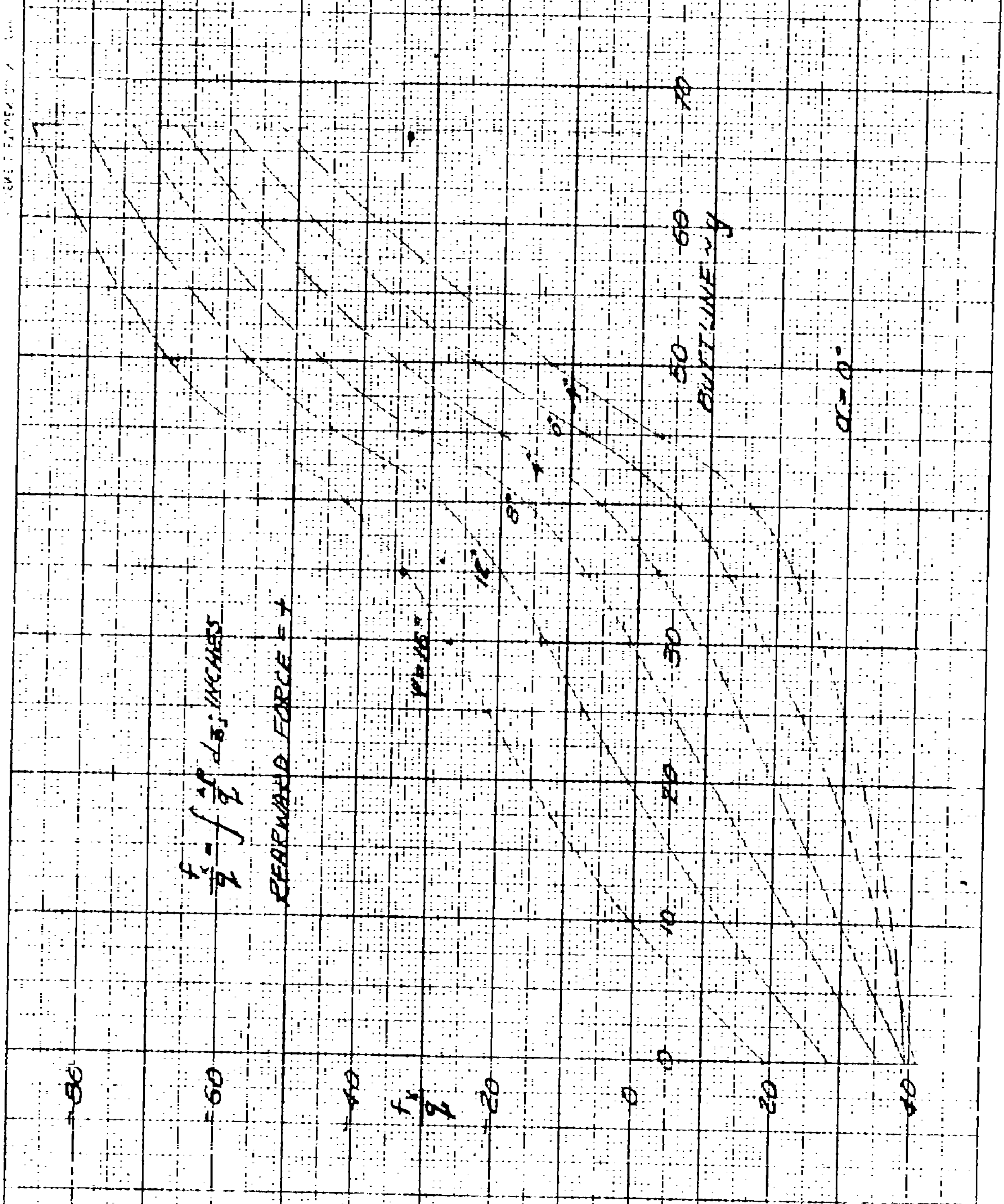
XC-120
NOSE DOOR TOTAL MOMENTS ABOUT Z AXIS
(RIGHT DOOR ONLY)
WIND TUNNEL DATA

FIG 71



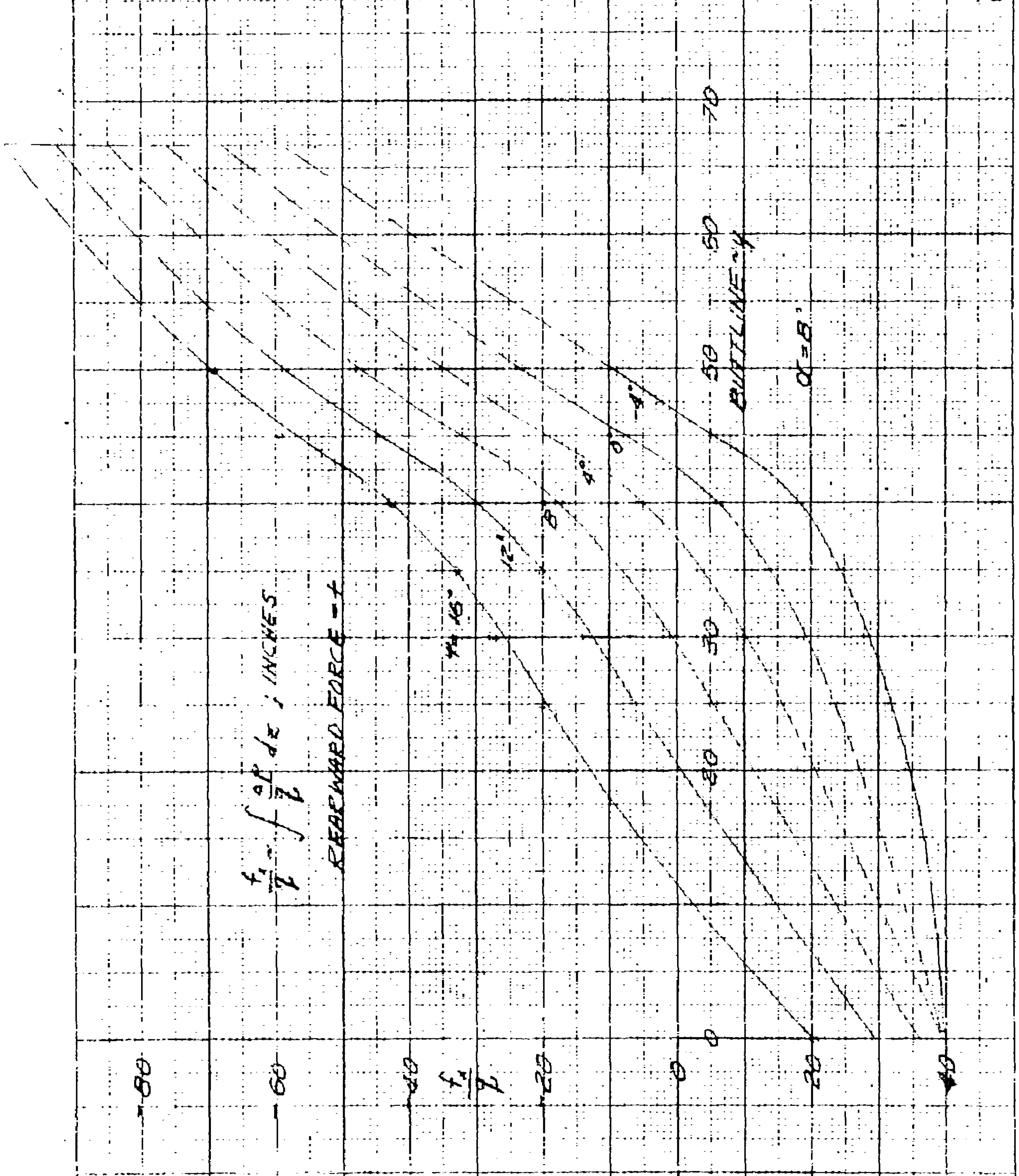
XC-120
NOSE DOOR TOTAL MOMENTS ABOUT Z AXIS
(RIGHT DOOR ONLY)
EXTRAPOLATED WIND TUNNEL DATA

FIG 72



XC-120
Section Drag Loads vs Buttline
(Right Door Only)

FIG 73



XC-120
Section Drag Loads vs Buttline
(Wing Drag Only)

FIG 74

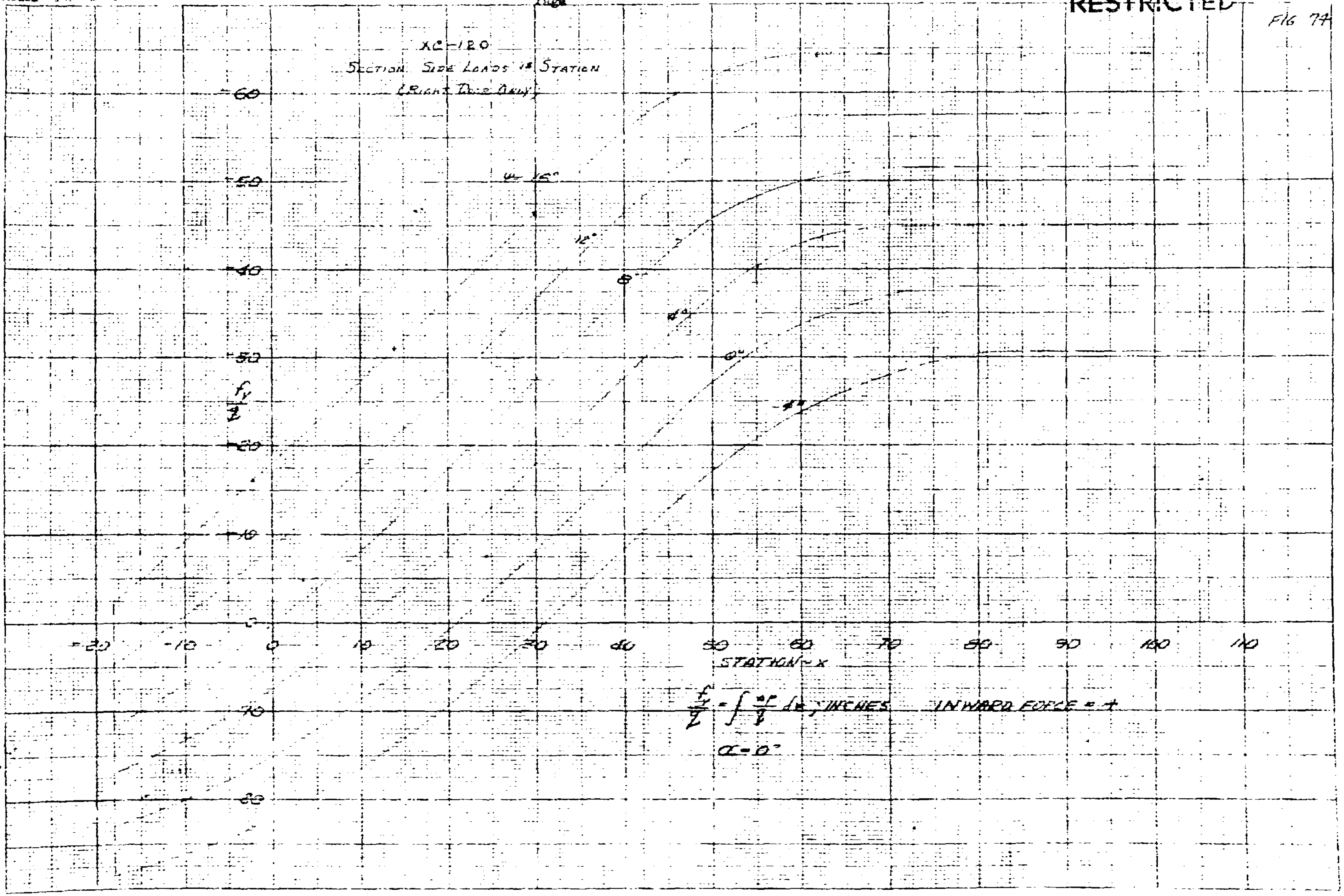

XC-120
SECTION SIDE LOADS vs STATION
(RIGHT TAIL BODY)
ψ = 16°
12°
8°
4°
0°
-4°
60
50
40
30
20
10
0
-10
-20
$\frac{f_Y}{q}$
-20
-10
0
10
20
30
40
50
60
70
80
90
100
110
STATION - x
$\frac{f_Y}{q} = \int \frac{\Delta P}{q}\, dz$, INCHES INWARD FORCE = +
$\alpha = 0°$

FIG 75

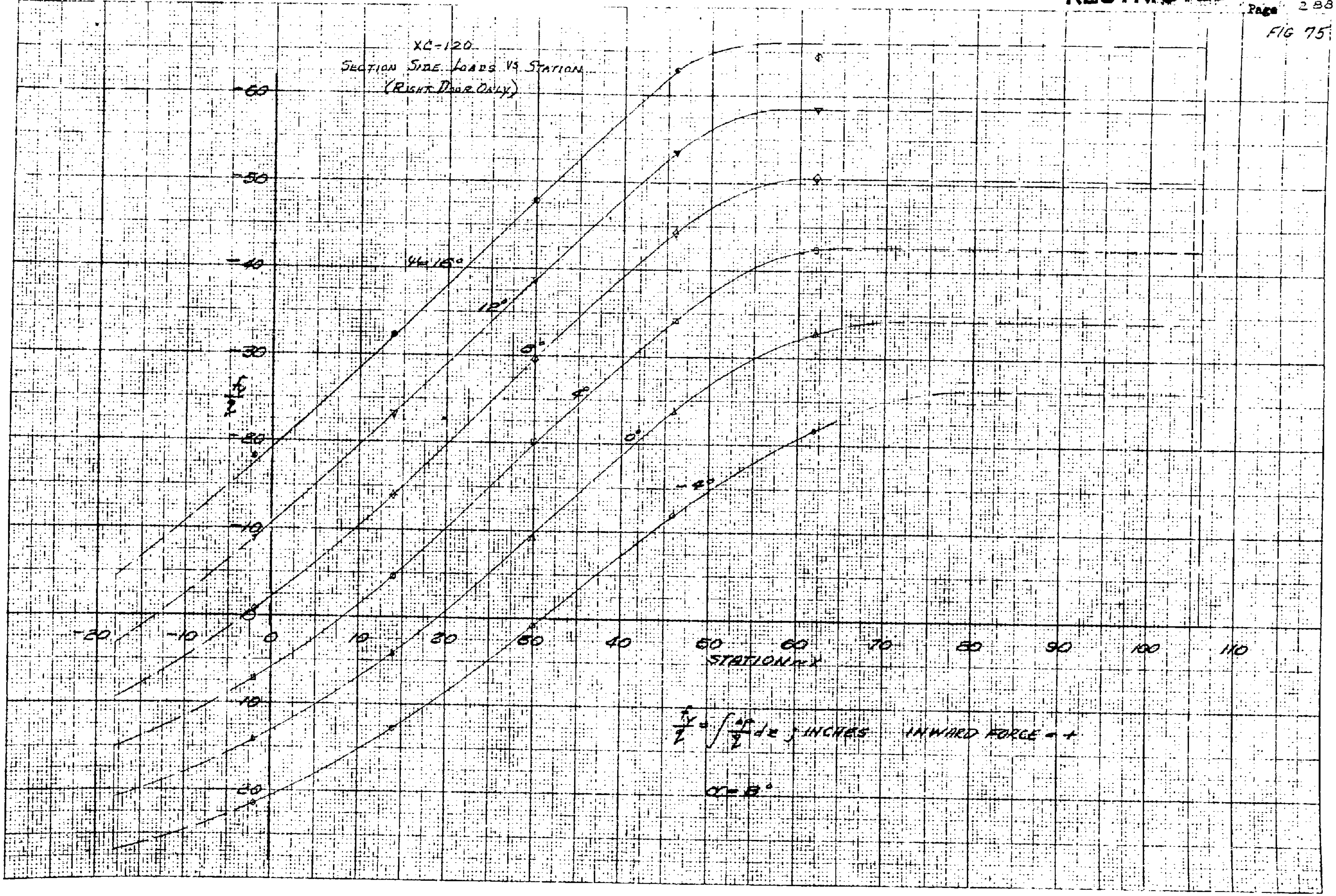
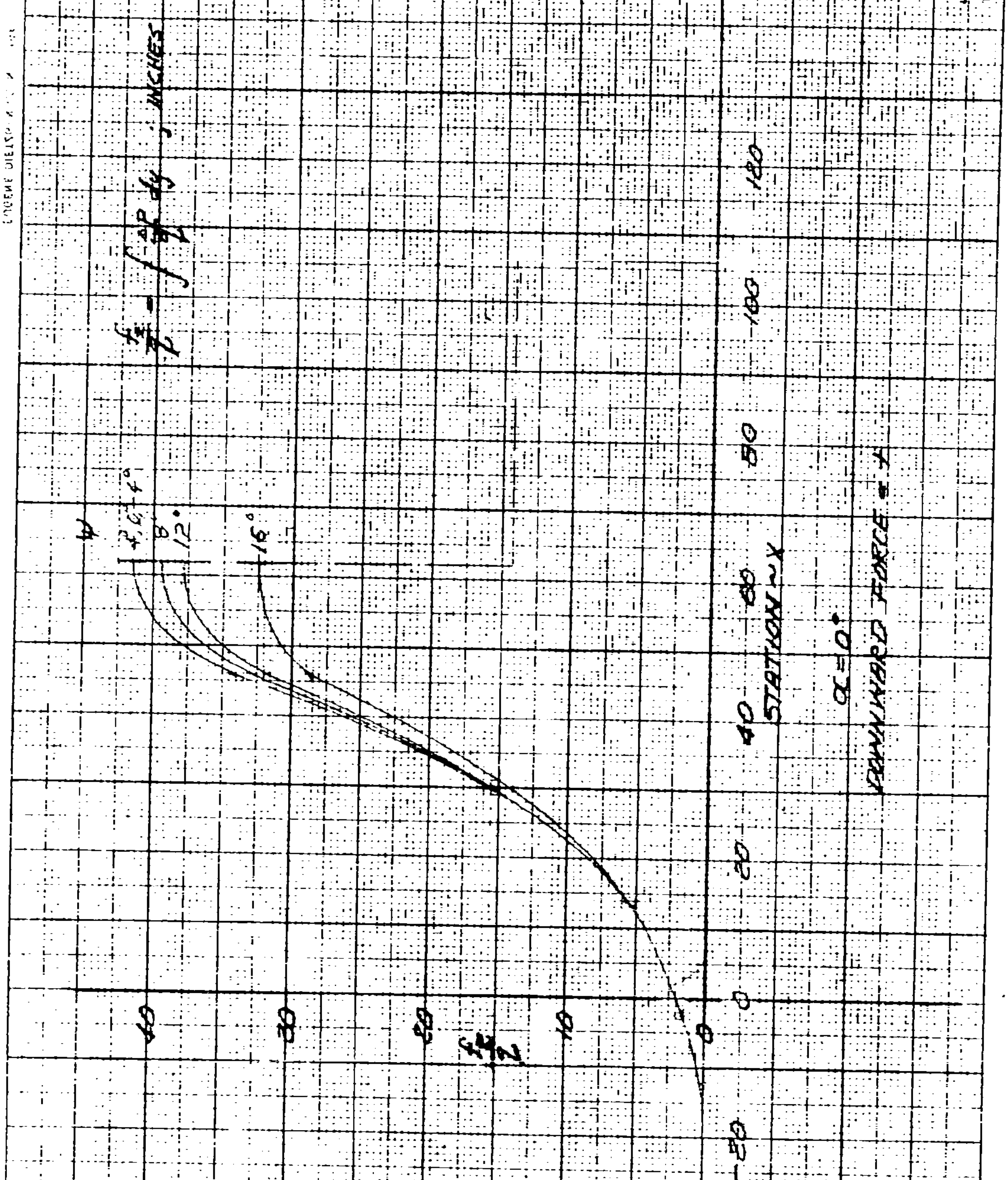
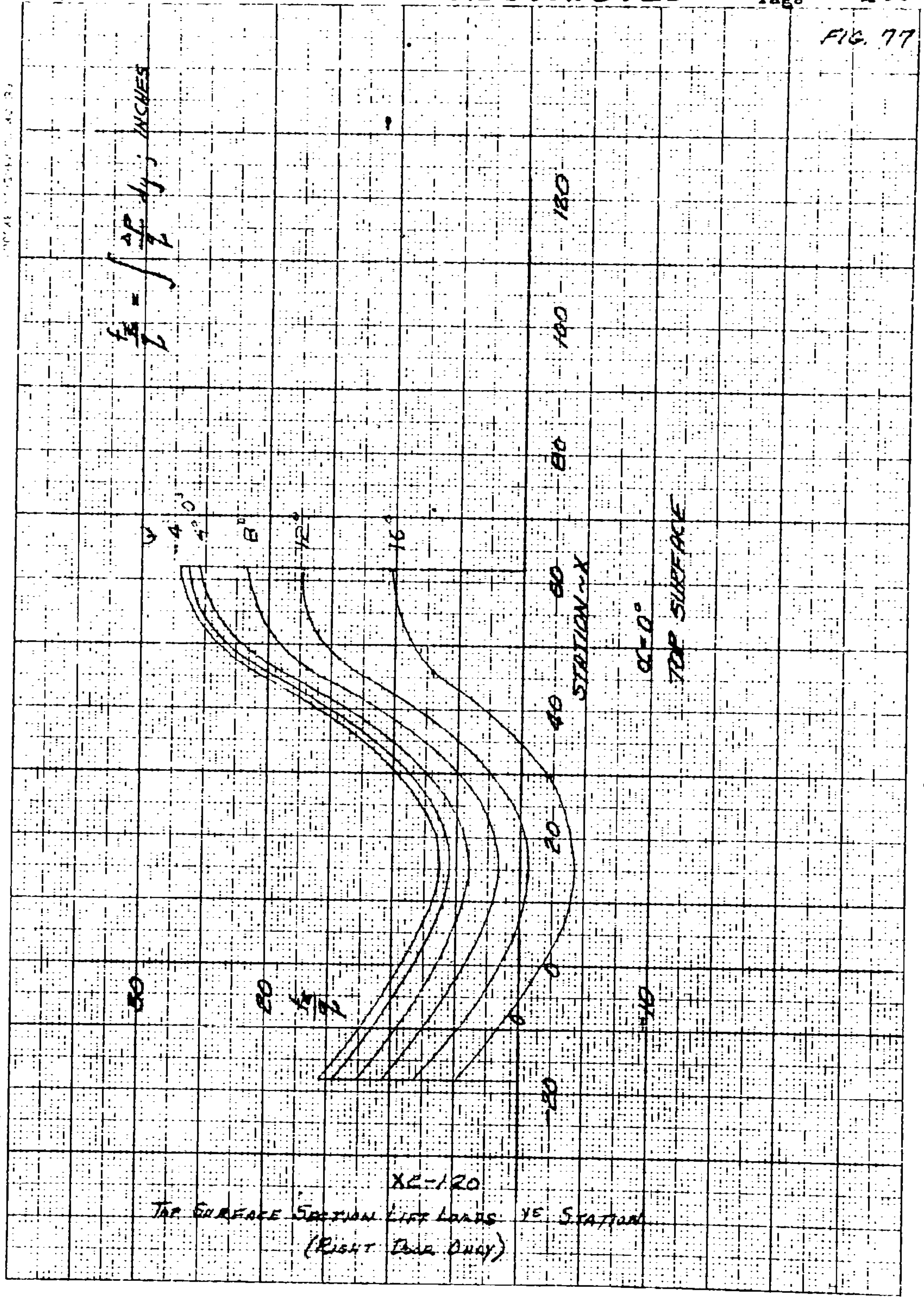

XC-120
SECTION SIDE LOADS VS STATION
(RIGHT DOOR ONLY)
ψ=16°
12°
8°
4°
0°
-4°
-60
-50
-40
-30
-20
-10
0
10
20
-20
-10
0
10
20
30
40
50
60
70
80
90
100
110
STATION ~ X
$\frac{F_Y}{Z} = \int \frac{\Delta P}{Z} dz$; INCHES INWARD FORCE = +
α = 8°

FIG. 76



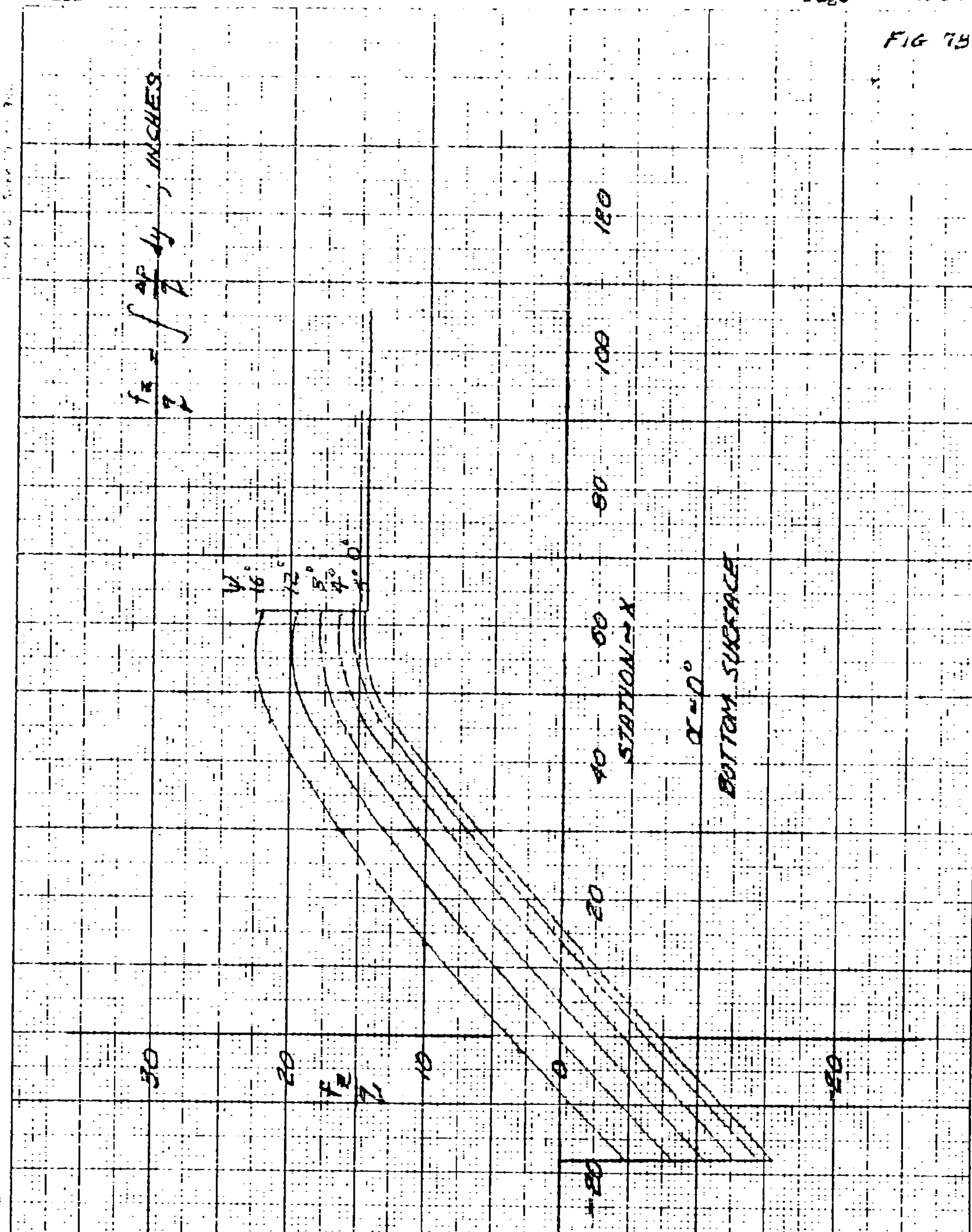
XC-120
SECTION LIFT LOADS VS STATION
(RIGHT WING ONLY)
SUM OF TOP & BOTTOM SURFACES

FIG. 77

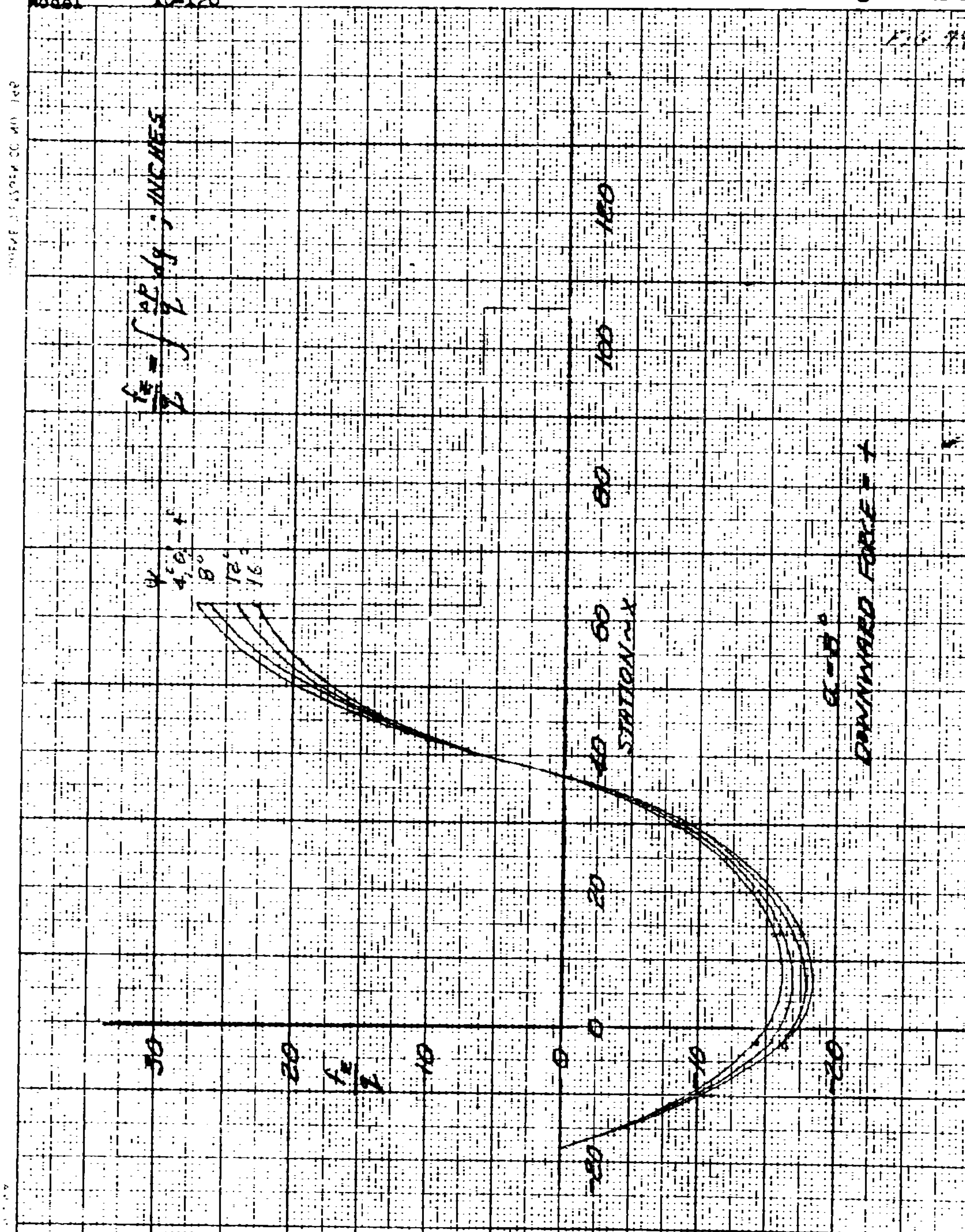


XC-120
TOP SURFACE SECTION LIFT LOADS VS STATION
(RIGHT TAIL DUCT)

FIG 7B

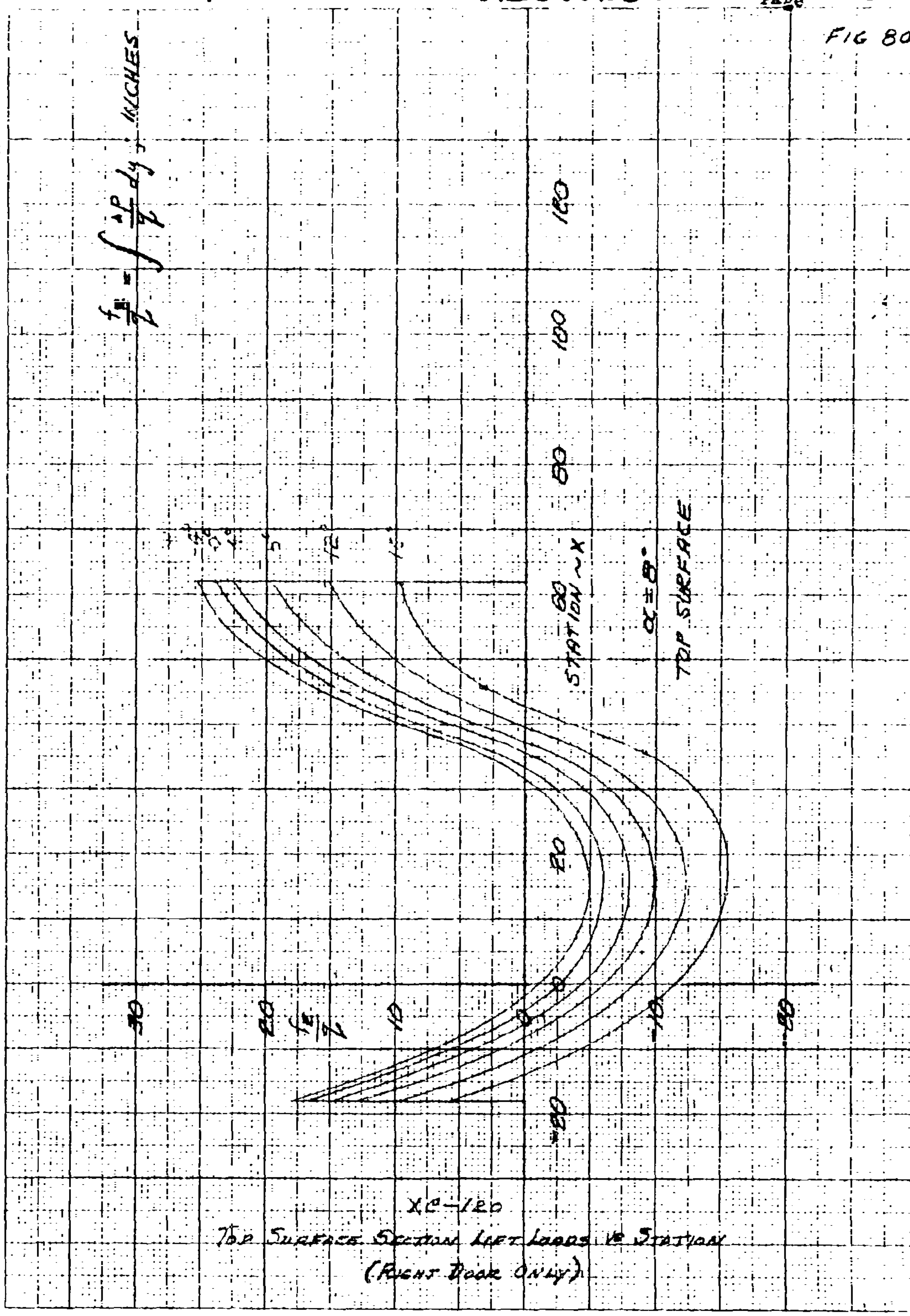


XC-120
Bottom Surface Section Lift Loads vs Station
(Right Door Only)



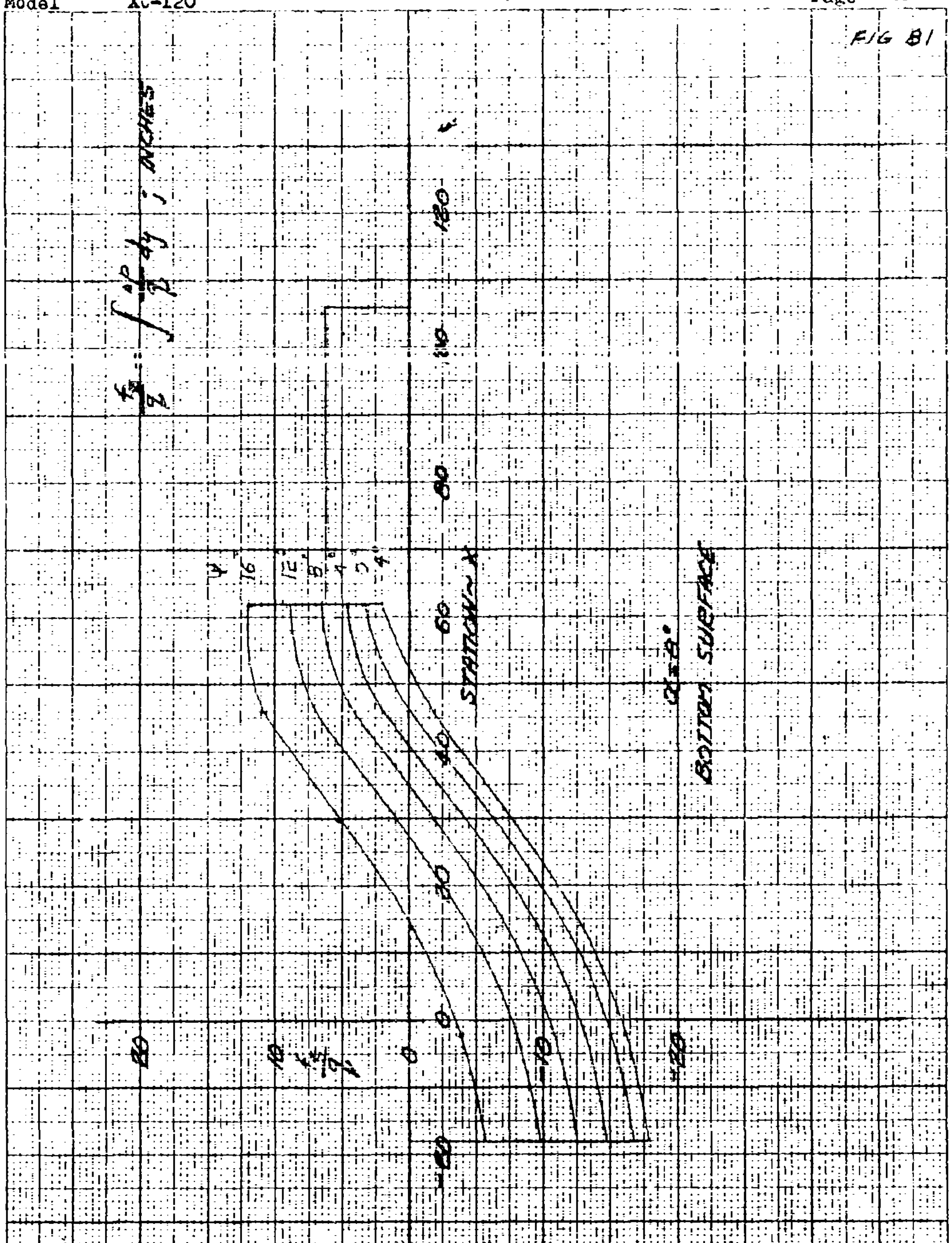
XC-120
Section Lift Loads vs Station
(Right Door Only)
Sum of Top & Bottom Surface

FIG 80



XC-120
TOP SURFACE SECTION LIFT LOADS VS STATION
(RIGHT DOOR ONLY)

FIG 81



XC-120
Bottom Surface Station Lift Loads vs Station
(Right Door Only)

PART II-0-4c

c. Nose Door Loads and Moments

Points on V-n Diagram (Yaw = 0)

The primary variables of the V-n diagram are angle of attack and speed. Yaw is assumed to be zero throughout. An inspection of the nose door load and moment coefficients (figures 63 to 71) indicates that only F_Z, M_Y, and M_X vary with α . Since they all have their largest values at negative angles of attack, the critical loading condition from the V-n diagram will be negative load factors at high speeds. A gross weight of 64,000 lbs. will be critical since it requires the largest values of angle of attack to attain the specified value of load factor. The loads and moments are determined in the table immediately following for three speeds at maximum negative load factor for a weight of 64,000 lbs. It is apparent that the greatest loads are attained at V_D. The directions of the loads and moments are shown on the figures presenting the coefficients.

PART II-0-4c

CRITICAL LOADS FROM V-N DIAGRAM

POWER OFF - 64,000 LBS.

V mph	q #/ft.^{2}	n	C_{Z_A}	α TL deg	F_Z/q	M_X/q	M_Y/q	F_Z lbs.	$M_X \times 10^{-3}$ in. lbs.	$M_Y \times 10^{-3}$ in. lbs.
190	92.4	-1.5	-.719	-15°	37.1	2040	-1620	3430	188.5	-149.6
250	160.	-1.5	-.415	-11.9°	32.5	1800	-1370	5200	288.	-219.
313	251.	-1.14	-.201	-9.5°	28.7	1600	-1180	7200	401	-296.

V mph	ψ deg.	F_X/q	F_Y/q	M_Z/q	F_X lbs.	F_Y lbs.	$M_Z \times 10^{-3}$ in. lbs.
313	0°	2	-13.6	-220	502	-3410	-55.2

PART II-0-4c

Angles of Yaw

The door loads and moments will be determined for two conditions; 10 degrees of yaw at V_D and 20 degrees of yaw at low speed. It is apparent that for those loads and moments which are primarily a function of yaw, the highest speed at which 20 degrees of yaw is likely to be attained will give the largest values. For those loads and moments dependent primarily on angle of attack, the highest speed will also result in the largest values, since the coefficients increase with decreasing α and the value of q will be greater.

Air Force Specification 1815-B requires that the directional stability be such that the adverse yaw due to aileron deflection in abrupt, rudder-fixed rolls is less than one degree per five percent of total aileron deflection at $1.4\ V_{S_{PA}}$. This would be equivalent to 20° at approximately 130 miles per hour. In order to insure meeting this requirement with sufficient margin of safety the speed will be assumed to be 150 mi./hr.

The loads and moments corresponding to the given conditions are determined in the following table. Their directions are shown on the figures showing the coefficients. It is clear that 10° of yaw at V_D is the most critical yaw condition.

PART II-C-4c

CRITICAL LOADS DUE TO YAW
POWER-OFF - 64,000 LBS.

V mph	q #/ft.^{2}	C_{Z_A}	α TL deg.	ψ deg.	$\frac{F_X}{q}$	$\frac{F_Y}{q}$	$\frac{F_Z}{q}$	$\frac{M_X}{q}$	$\frac{M_Y}{q}$	$\frac{M_Z}{q}$
150	57.5	.770	1.1°	20°	-23.3	-47.5	11.2	770	-340	2920
313	251	.176	-5.5°	10°	-9.0	-30.0	22.0	1290	-860	1270
				-10°	10.0	3.0	22.0	1290	-860	-1450

V mph	ψ deg.	F_X lbs.	F_Y lbs.	F_Z lbs.	M_X in. lbs	M_Y in. lbs	M_Z in lbs
150	20°	-1340	-2730	644	44250	-19550	168000
313	10°	-2260	-7530	5525	324000	-216000	319000
	-10°	2510	753	5525	324000	-216000	364000

PART II-0-4c

Loads Throughout the Speed Range

The loads throughout the speed range are not computed since they will be in every instance less than those for the previous conditions. This is so because the loads are greatest (at zero yaw) for the largest negative angles of attack and highest speed. The most critical such combination occurs on the V-n diagram.

Special Load for Latch Design

In order to insure that the latches, which tie the right and left nose doors together, are overstrength the following arbitrary loading is assumed.

Side Load - Apply 1 q load at 313 mph (V_D) uniformly over door.

Summary - Critical Nose Door Loads

(Right Door Only)

Condition	V mph	α_{TL} deg.	ψ deg.	F_X lbs.	F_Y lbs.	F_Z lbs.	M_X 10^{-3} in lbs.	M_Y 10^{-3} in lbs.	M_Z 10^{-3} in lbs.
NLAA (V_D)	313	-9.5	0	502	-3,410	7,200	401	-296	-55.2
Level Flt. (Yaw)	313	-5.5°	+10	-2,260	-7,530	5,525	324	-216	319
			-10	2,510	753	5,525	324	-216	-364
Special Latch Load	Apply 1 q Load at 313 MPH(V_D) Uniformly Over Door								

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REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION		PAGES	PAGE 300
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY	
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PART III-A

ASSYMETRICAL FLIGHT ~ YAW

A. General

This section covers the method used to determine the yawing moments of the component parts of the airplane both for two engine operation and for single engine operation.

Summary of Aerodynamic Coefficients of Component Parts

The following Aerodynamic Coefficients are used in this analysis.

Component Part	Cross Force Coefficient	Drag Coefficient	Yawing Moment Coefficient
Wing	$C_{C_W} = \dfrac{C_W}{q\ S_W}$	$C_{D_W} = \dfrac{D_W}{q\ S_W}$	$C_{N_{W_{AC}}} = \dfrac{N_W}{q\ S_W\ B_W}$
Fuselage	$C_{C_F} = \dfrac{C_F}{q\ S_f}$	$C_{D_F} = \dfrac{D_F}{q\ F_f}$	$C_{N_{F.25\ L_f}} = \dfrac{N_F}{q\ S_f\ L_f}$
Boom (One)	$C_{C_B} = \dfrac{C_B}{q\ S_b}$	$C_{D_B} = \dfrac{D_B}{q\ F_b}$	$C_{N_{B.25\ L_b}} = \dfrac{N_B}{q\ S_b\ L_b}$
Tail	$C_{C_T} = \dfrac{C_T}{q_t\ S_t}$	$C_{D_T} = \dfrac{D_T}{q_t\ S_t}$	$C_{N_{T_{AC}}} = \dfrac{N_T}{q_t\ S_t\ B_t}$

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REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION		PAGES	PAGE 54301
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY	
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FIG. 2
FIG. 82

PART III-A-1

1. <u>Summation of Yawing Moments About Any C. G. Location Two Engine Operation</u>

Figure 82 presents a general picture of the airplane in yawed flight with the pertinent forces and arms noted. Taking moments about the c.g. with two engines operating.

$$N_W = -l_w \; Y_W$$

$$N_F = -l_f \; Y_F$$

$$N_B = (-l_b \; Y_B)^2$$

N_{HT} = Neglect forces - negligible

$$N_{VT} = (-l_{vt} \; Y_{VT})^2$$

N_S = Slipstream effects on wing neglected in yaw

$$N_P = (-l_p \; Y_p)^2$$

Now considering the contribution of each component part to the moment about the c.g. and refer the final moments to wing area, wing span, and free flight velocity.

<u>Wing</u> Consider two possible c.g. positions

$$N_{W_{c.g.\,1}} = -l_{w1} \; Y_W$$

$$N_{W_{c.g.\,2}} = -l_{w2} \; Y_W$$

$$C_{N_{W_{c.g.\,1}}} = \frac{l_{w1}}{B_w} \; C_{Y_W}$$

$$C_{N_{W_{c.g.\,2}}} - C_{N_{W_{c.g.\,1}}} = C_{Y_W}\left(-\frac{l_{w2}}{B_w} + \frac{l_{w1}}{B_w}\right)$$

PART III-A-1 (Cont.)

Now from figure 82 we see $l_w = Cp_w - X_w$

$$C_{N_{W_{c.g.\ 2}}} = C_{N_{W_{c.g.\ 1}}} + C_{Y_W}\left(\frac{Cp_w}{B_w} - \frac{X_{w_1}}{B_w} - \frac{Cp_w}{B_w} + \frac{X_{w_2}}{B_w}\right)$$

$$C_{N_{W_{c.g.\ 2}}} = C_{N_{W_{c.g.\ 1}}} + C_{Y_W}\left(\frac{X_{w_2}}{B_w} - \frac{X_{w_1}}{B_w}\right)$$

Now assume c.g. position 1 to be at AC_w of wing.

$$X_{w_1} = AC_w$$

$$C_{N_{W_1}} = C_{N_{W_{AC}}}$$

$$C_{N_{W_{c.g.}}} = C_{N_{W_{AC}}} + C_{Y_W}\left(\frac{X_w}{B_w} - \frac{AC_w}{B_w}\right)$$

Fuselage

Consider two possible c.g. positions

$$N_{F_1} = -l_{f_1} Y_F$$

$$N_{F_2} = -l_{f_2} Y_F$$

Transferring to coefficient form based on wing area, etc.

$$C_{N_{F_{c.g.\ 1}}} = -C_{Y_F}\frac{l_{f_1}}{B_w}\frac{S_f}{S_w}$$

$$C_{N_{F_{c.g.\ 2}}} = -C_{Y_F}\frac{l_{f_2}}{B_w}\frac{S_f}{S_w}$$

PART III-A-1 (Cont.)

Now from figure 82 $l_f = C_{P_f} - X_f$

$$C_{N_{F\,c.g.\,2}} - C_{N_{F\,c.g.\,1}} = C_{Y_F} \frac{S_f}{S_w} \frac{l_{f_2}}{B_w} + C_{Y_F} \frac{S_f}{S_w} \frac{l_{f_1}}{B_w}$$

$$C_{N_{F\,c.g.\,2}} = C_{N_{F\,c.g.\,1}} + C_{Y_F} \frac{S_f}{S_w} \left(\frac{C_{P_f}}{B_w} - \frac{X_{f_1}}{B_w} - \frac{C_{P_f}}{B_w} - \frac{X_{f_2}}{B_w} \right)$$

$$C_{N_{F\,c.g.\,2}} = C_{N_{F\,c.g.\,1}} + C_{Y_F} \frac{S_f}{S_w} \left(\frac{X_{f_2}}{B_w} - \frac{X_{f_1}}{B_w} \right)$$

Now assume c.g. position 1 to be at $.25\ L_f$

$$C_{N_{F\,c.g.\,1}} = C_{N_{F\,.25L_f}} \frac{S_f\ L_f}{S_w\ B_w}$$

$$X_{f_1} = .25L_f$$

$$C_{N_{F\,c.g.}} = C_{N_{F\,.25L_f}} \frac{S_f}{S_w} \frac{L_f}{B_w} + C_{Y_F} \frac{S_f}{S_w} \left(\frac{X_f}{B_w} - \frac{.25\ L_f}{B_w} \right)$$

Boom

The development of the transfer formulas for the boom is the same as for the fuselage.

$$C_{N_{B\,c.g.}} = 2 \left[C_{N_{B\,.25L_b}} \frac{S_b\ L_b}{S_w\ B_w} + C_{Y_B} \frac{S_b}{S_w} \left(\frac{X_b}{B_w} - \frac{.25\ L_b}{B_w} \right) \right]$$

Vertical Tail

Now considering the vertical tail the situation is somewhat different because of the effects of change in dynamic pressure and of sidewash.

PART III-A-1 (Cont.)

Vertical Tail (Cont.)

$$N_{VT_{c.g.\,1}} = 2(-l_{vt_1}\ Y_{VT})$$

$$N_{VT_{c.g.\,2}} = 2(-l_{vt_2}\ Y_{VT})$$

Now transferring to coefficient form based on wing area

$$C_{N_{VT_{c.g.\,1}}} = -2\ C_{Y_{VT}}\ \frac{q_{vt}\ S_{vt}\ l_{vt_1}}{q\ S_w\ B_w}$$

$$C_{N_{VT_{c.g.\,2}}} = -2\ C_{Y_{VT}}\ \frac{q_{vt}\ S_{vt}\ l_{vt_2}}{q\ S_w\ B_w}$$

$$C_{N_{VT_{c.g.\,2}}} - C_{N_{VT_{c.g.\,1}}} = -2\ C_{Y_{VT}}\ \frac{q_{vt}}{q}\ \frac{S_{vt}}{S_w}\left(\frac{l_{vt_2}}{B_w} - \frac{l_{vt_1}}{B_w}\right)$$

Now since $l_{vt} = CP_{vt} - X_{vt}$

$$C_{N_{VT_{c.g.\,2}}} = C_{N_{VT_{c.g.\,1}}} + 2\ C_{Y_{VT}}\ \frac{q_{vt}}{q}\ \frac{S_{vt}}{S_w}\left(\frac{CP_{vt}}{B_w} - \frac{X_{vt_1}}{B_w} - \frac{CP_{vt}}{B_w} - \frac{X_{vt_2}}{B_w}\right)$$

$$C_{N_{VT_{c.g.\,2}}} = C_{N_{VT_{c.g.\,1}}} + 2\ C_{Y_{VT}}\ \frac{q_{vt}}{q}\ \frac{S_{vt}}{S_w}\left(\frac{X_{vt_2}}{B_w} - \frac{X_{vt_1}}{B_w}\right)$$

Now assuming first c.g. position at AC_{vt} of the vertical tail

$$C_{N_{VT_{c.g.}}} = 2\ C_{N_{VT_{AC}}}\ \frac{q_{vt}}{q}\ \frac{S_{vt}}{S_w}\ \frac{B_{vt}}{B_w}$$

$$X_{vt_1} = AC_{vt}$$

$$C_{N_{VT_{c.g.}}} = 2\left[C_{N_{VT_{AC}}}\ \frac{q_{vt}\ S_{vt}\ B_{vt}}{q\ S_w\ B_w} + C_{Y_{VT}}\ \frac{q_{vt}}{q}\ \frac{S_{vt}}{S_w}\left(\frac{X_{vt}}{B_w} - \frac{AC_{vt}}{B_w}\right)\right]$$

PART III-A-1

Propeller Effects

$$N_{P_{c.g.}} = -2\, l_p\, Y_P$$

$$C_{N_{P_{c.g.}}} = -2\, Y_P\, \frac{l_p}{q\, S_w\, B_w}$$

So:

$$C_{N_{P_{c.g.}}} = -\, Y_P\, \frac{2\, l_p}{q\, S_w\, B_w}$$

Angular Relationship

In order to properly sum the component parts of the airplane it is necessary to know the relationship of each part to the thrust line (fuselage reference line) since all of the characteristic terms are parallel and perpendicular to this reference line.

$$\psi_w = \psi_f = \psi_b = \psi_p = \psi_{ht}$$

$$\psi_{vt} = \psi_w - \sigma_{P_{ave}} - \frac{d\sigma}{d\psi}\, \psi$$

where σ is angle of sidewash

PART III-A-2

2. SUMMARY OF EQUATIONS FOR COMPLETE AIRPLANE

TWO ENGINE OPERATION

TOTAL MOMENT	CONTRIBUTION OF
$C_{N_{A_{c.g.}}} = C_{N_{W_{AC}}} + C_{Y_W}\left(\frac{x_w}{B_w} - \frac{AC_w}{B_w}\right) +$	Wing
$C_{N_{F_{.25L_f}}} \frac{S_f L_f}{S_w B_w} + C_{Y_F} \frac{S_f}{S_w}\left(\frac{x_f}{B_w} - \frac{.25\,L_f}{B_w}\right) +$	Fuselage
$2\left[C_{N_{B_{.25L_b}}} \frac{S_b L_b}{S_w B_w} + C_{Y_B} \frac{S_b}{S_w}\left(\frac{x_b}{B_w} - \frac{.25L_b}{B_w}\right)\right] +$	Booms
$2\left[C_{N_{VT_{AC}}} \frac{q_{vt}}{q}\frac{S_{vt}}{S_w}\frac{B_{vt}}{B_w} + C_{Y_{VT}} \frac{q_{vt}}{q}\frac{S_{vt}}{S_w}\left(\frac{x_{vt}}{B_w} - \frac{AC_{vt}}{B_w}\right)\right] +$	Vertical Tail
$-\frac{2\,l_p}{q\,S_w\,b_w}\,Y_p$	Propellers

Note: (1) That "X" forces on all components do not cause moments for two engine flight.

(2) Effect of horizontal tail and slipstream have been checked and found negligible and are omitted above.

ANGULAR RELATIONSHIPS

$$\psi_w = \psi_f = \psi_b = \psi_p = \psi_{ht}$$

$$\psi_{vt} = \psi_w - \sigma_{P_{ave}} - \frac{d\sigma}{d\psi}\,\psi$$

PART III-A-3

3. <u>SUMMATION OF YAWING MOMENTS ABOUT ANY C. G. LOCATION - SINGLE ENGINE OPERATION</u>

Figure 52 presents a general picture of the airplane in yawed flight with the pertinent forces and arms noted. Taking moments about the c.g. with left engine dead

$$N_W = -l_w Y_W$$

$$N_F = -l_f Y_F$$

$$N_B = (-l_b Y_B)^2 + y_b \Delta X_{\text{right cowl flaps open}}$$

N_{HT} = neglet forces - negligible

$$N_{VT} = -\underset{\text{left}}{l_{vt} Y_{VT}} - \underset{\text{right}}{l_{vt} Y_{VT}} + \underset{\text{left}}{y_b X_{VT}} + \underset{\text{right}}{y_b X_{VT}}$$

Note:

y_b is negative to left and positive to right

$$N_S = \underset{\text{right}}{y_b \Delta X_S}$$

$$N_P = -\underset{\text{right}}{l_p Y_p} + \underset{\text{right}}{y_b X_p} + \underset{\text{left-feathered propeller}}{y_b \Delta X_P}$$

<u>Wing</u>

Equations worked out for two engine operation

<u>Fuselage</u>

Equations worked out for two engine operation

<u>Booms</u>

Basic equation worked out for two engine operation except for increment of one engine cowl flaps open

$$C_{NB_{\text{c.g.}}} = 2\left[C_{NB_{.25L_b}} \frac{S_b L_b}{S_w B_w} + C_{Y_B} \frac{S_b}{S_w}\left(\frac{X_b}{B_w} - \frac{.25L_b}{B_w}\right)\right] + \underset{\text{Right cowl flaps open}}{\Delta C_{X_B}} \frac{y_b F_b}{B_w S_w}$$

PART III-A-3 (Cont.)

Vertical Tail

$$N_{VT_{c.g.\ 1}} = -\,l_{vt_1} \underset{\text{left}}{Y_{VT}} - l_{vt} \underset{\text{right}}{Y_{VT}} + y_b \underset{\text{left}}{X_{VT}} + y_b \underset{\text{right}}{X_{VT}}$$

$$N_{VT_{c.g.\ 2}} = -\,l_{vt_2} \underset{\text{left}}{Y_{VT}} - l_{vt_2} \underset{\text{right}}{Y_{VT}} + y_b \underset{\text{left}}{X_{VT}} + y_b \underset{\text{right}}{X_{VT}}$$

$$C_{N_{VT_{c.g.\ 1}}} = -\underset{\text{left}}{C_{Y_{VT}}} \frac{q_{vt}\, S_{vt}\, l_{vt_1}}{q\, S_w\, B_w} - \underset{\text{right}}{C_{Y_{VT}}} \frac{q_{vt}\, S_{vt}\, l_{vt}}{q\, S_w\, B_w}$$

$$+ \underset{\text{left}}{C_{X_{VT}}} \frac{q_{vt}\, S_{vt}\, y_b}{q\, S_w\, B_w} + \underset{\text{right}}{C_{X_{VT}}} \frac{q_{vt}\, S_{vt}\, y_b}{q\, S_w\, B_w}$$

Combining

$$C_{N_{VT_{c.g.\ 2}}} - C_{N_{VT_{c.g.\ 1}}} = \underset{\text{left}}{C_{Y_{VT}}} \frac{q_{vt}\, S_{vt}\, l_{vt2}}{q\, S_w\, B_w} + \underset{\text{left}}{C_{Y_{VT}}} \frac{q_{vt}\, S_{vt}\, l_{vt1}}{q\, S_w\, B_w} +$$

$$- \underset{\text{right}}{C_{Y_{VT}}} \frac{q_{vt}\, S_{vt}\, l_{vt2}}{q\, S_w\, B_w} + \underset{\text{right}}{C_{Y_{VT}}} \frac{q_{vt}\, S_{vt}\, l_{vt1}}{q\, S_w\, B_w}$$

$$C_{N_{VT_{c.g.\ 2}}} - C_{N_{VT_{c.g.\ 1}}} = \underset{\text{left}}{C_{Y_{VT}}} \frac{S_{vt}}{q\, S_w\, B_w} \left(- \underset{\text{left}}{q_{vt}}\, l_{vt_2} + \underset{\text{left}}{q_{vt}}\, l_{vt_1} \right) +$$

$$\underset{\text{right}}{C_{Y_{VT}}} \frac{S_{vt}}{q\, S_w\, B_w} \left(- q_{vt}\, \underset{\text{right}}{l_{vt_2}} + \underset{\text{right}}{q_{vt}}\, l_{vt_1} \right)$$

Now -

$$l_{vt} = C.\,P._{vt} - X_{vt}$$

$$C_{N_{VT_{c.g.\ 2}}} - C_{N_{VT_{c.g.\ 1}}} = \frac{S_{vt}}{q\, S_w\, B_w} \Big[C_{Y_{VT}} \underset{\text{left}}{q_{vt}} (x_{vt_2} - x_{vt_1}) +$$

$$C_{Y_{VT}} \underset{\text{right}}{q_{vt}} (x_{vt_2} - x_{vt_1}) \Big]$$

PART III-A-3 (Cont.)

Vertical Tail (Cont.)

Assume c.g. position 1 at line between vertical tail at AC_{vt} of vertical tail

$$C_{N_{VT}} = C_{N_{VT_{AC}}\,\text{left}} \frac{q_{vt}\,S_{vt}\,B_{vt}}{q\;S_w\;B_w} + C_{N_{VT_{AC}}\,\text{right}} \frac{q_{vt}\,S_{vt}\,B_{vt}}{q\;S_w\;B_w} +$$

$$C_{X_{VT}\,\text{left}} \frac{q_{vt}\,S_{vt}\,y_B}{q\;S_w\;B_w} + C_{X_{VT}\,\text{right}} \frac{q_{vt}\,S_{vt}\,y_b}{q\;S_w\;B_w}$$

$$X_{vt_1} = AC_{vt}$$

So:
$$C_{N_{VT_{c.g.}}} = \frac{S_{vt}\,B_{vt}}{S_w\;B_w}\left(\frac{q_{vt}}{q}_{\text{left}} C_{N_{VT_{AC}}} + \frac{q_{vt}}{q}_{\text{right}} C_{N_{VT_{AC}}}\right) +$$

$$\frac{S_{vt}}{S_w}\left[C_{X_{VT}\,\text{left}} \frac{q_{vt}}{q}\frac{y_b}{B_w} + C_{X_{VT}\,\text{right}} \frac{q_{vt}\,y_b}{q\;B_w} + \right.$$

$$\left. C_{Y_{VT}\,\text{left}} \frac{q_{vt}}{q}\left(\frac{X_{vt}}{B_w} - \frac{AC_{vt}}{B_w}\right) + C_{Y_{VT}\,\text{right}} \frac{q_{vt}}{q}\left(\frac{X_{vt}}{B_w} - \frac{AC_{vt}}{B_w}\right)\right]$$

Slipstream

$$N_S = y_b\,\Delta X_{S\,\text{right}}$$

$$C_{N_{S_{c.g.}}} = \Delta X_{S\,\text{right}} \frac{y_b}{q\;S_w\;B_w}$$

Propeller

$$N_P = -l_p\,Y_P + y_{b\,(\text{right})} + y_b\,\Delta X_P \;(\text{left-feathered prop})$$

$$C_{N_{P_{c.g.}}} = -Y_P\frac{l_p}{q\;S_w\;B_w} + X_{P\,(\text{right})}\frac{y_b}{q\;S_w\;B_w} + \Delta X_P \frac{y_b}{q\;S_w\;B_w}\;(\text{left-feathered prop})$$

PART III-A-3 (Cont.)

Angular Relationships

$$\psi_w = \psi_f = \psi_b = \psi_p = \psi_{ht}$$

$$\psi_{vt_{right}} = \psi_w - \sigma_{p_{ave\ right}} - \frac{d\sigma}{d\psi}\psi$$

$$\psi_{vt_{left}} = \psi_w - \sigma_{p_{ave\ left}} - \frac{d\sigma}{d\psi}\psi$$

PART III-A-4

4. SUMMARY OF EQUATIONS FOR COMPLETE AIRPLANE

SINGLE ENGINE OPERATION

Total Moment	Contribution Of
$C_{NA_{c.g.}} = C_{N_{W_{AC}}} + C_{Y_W}\left(\frac{X_w}{B_w} - \frac{AC_w}{B_w}\right) +$	Wing
$C_{N_{F.25L_f}} \frac{S_f L_f}{S_w B_w} + C_{Y_F} \frac{S_f}{S_w}\left(\frac{X_f}{B_w} - \frac{.25L_f}{B_w}\right) +$	Fuselage
$2\left[C_{N_{B.25L_b}} \frac{S_b L_b}{S_w B_w} + C_{Y_B} \frac{S_b}{S_w}\left(\frac{X_b}{B_w} - \frac{.25 L_b}{B_w}\right)\right] + \underset{\text{cowl flaps}}{\Delta C_{X_B}} \frac{y_b F_b}{B_w S_w} +$	Boom
$\frac{S_{vt} B_{vt}}{S_w B_w}\left(\frac{q_{vt}}{q} C_{N_{VT_{AC}}}{}_{\text{left}} + \frac{q_{vt}}{q} C_{N_{VT_{AC}}}{}_{\text{right}}\right) +$ $\frac{S_{vt}}{S_w}\left[C_{X_{VT}}{}_{\text{left}} \frac{q_{vt}}{q} \frac{y_b}{B_w} + C_{X_{VT}}{}_{\text{right}} \frac{q_{vt}}{q} \frac{y_b}{B_w} + C_{Y_{VT}}{}_{\text{left}} \frac{q_{vt}}{q}\left(\frac{X_{vt}}{B_w} - \frac{AC_{vt}}{B_w}\right)\right.$ $\left. + C_{Y_{VT}}{}_{\text{right}} \frac{q_{vt}}{q}\left(\frac{X_{vt}}{B_w} - \frac{AC_{vt}}{B_w}\right)\right] +$	Vertical Tail
$\Delta X_{S_{\text{right}}} \frac{y_b}{q S_w B_w} +$	Slipstream
$- Y_P \frac{l_p}{q S_w B_w} + X_{P_{\text{right}}} \frac{y_b}{q S_w B_w} + \Delta X_{P_{\text{left feathered}}} \frac{y_b}{q S_w B_w}$	Propellers

Angular Relationships

$$\psi_w = \psi_f = \psi_b = \psi_p = \psi_{ht}$$

$$\psi_{vt_{\text{right}}} = \psi_w - \sigma_{\text{Pave right}} - \frac{d\sigma}{d\psi}\psi$$

$$\psi_{vt_{\text{left}}} = \psi_w - \sigma_{\text{Pave left}} - \frac{d\sigma}{d\psi}\psi$$

PART III-B

B. Yawing Moments of Component Parts

This section covers the determination of the yawing moments of the component parts of the airplane. As the vertical tail loads need to be determined for only two yaw conditions with flaps and gear up-namely:

Zero yaw with single engine operation and 5° yaw at V_D ($V\sqrt{\sigma}$ = 313 mph), the yawing moments of the component parts were determined for these conditions.

Final results for design criteria conditions may be found on the following pages or figures.

Summary - Final Data - Yawing Moments

Component Part	Condition: Zero Yaw Single Engine	Condition: 5° Yaw at V_D Two Engines
Wing Unflapped	Page 314	Page 314
Wing Flapped	Page 316	Page 316
Fuselage	Page 317	Page 317
Booms	Page 319	Page 319
Horizontal Tail	Page 321	Page 321
Landing Gear	Page 322	Page 322
Propellers	Figure 83	Page 323
Slipstream Effects	Figure 84	Page 328
Vertical Tail	Figures 85, 86, 87	Figure 88

PART III-B-1

1. Wing Unflapped

$$C_{N_W} = C_{N_{W_{AC}}} + C_{Y_W}\left(\frac{x_W}{B_W} - \frac{AC_W}{B_W}\right)$$

for both single engine and two engine operation. Slipstream effects are covered in Part III-B-8

Zero Yaw - Single Engine Operation

For zero yaw $C_{N_W} = 0$ for all C. G. locations.

5° Yaw — V_D $V\sqrt{\sigma} = 313$ mph

$$\frac{A.C._W}{B_W} = \frac{.256 \times 168.28}{109.27 \times 12} = .0329$$

C. G. Position	① x_W	② x_W/B_W	③ $\left(\frac{x_W}{B_W} - \frac{A.C._W}{B_W}\right)$
1	33.66	.0257	-.0072
2	50.48	.0385	+.0056
3	33.66	.0257	-.0072
4	50.48	.0385	+.0056
5	40.01	.0306	-.0023
6	37.32	.0285	-.0044

For design gross weight of 64,000 lbs.

$$C_{L_W} = \frac{64000}{.00256 \times 1447.24\ (313)^2} = .177$$

At this speed the airplane is flying at an $\alpha_{TL} = -6^\circ$. Figure 2 reference (1)

Assuming that C_{D_W} in yaw $= C_{D_W}$ in pitch at $\alpha_{TL} = -6^\circ$

Then C_{D_W} in yaw $= .008$ from figure 3 reference (1)

$$C_{Y_W} = (C_{D_W}, \text{ in yaw}) \sin\psi$$

PART III-B-1

1. Wing Unflapped (Cont.)

$$C_{Y_W} = .008\ (.0872)$$

$$= .000699$$

$$C_{N_{W_{A.C.}}} = .00008 \text{ from figure 5 reference (1)}$$

C. G. Position	$C_{Y_W}\left(\frac{x_W}{B_W} - \frac{AC_W}{B_W}\right)$	C_{N_W}
1	-.00001	.00007
2	negligible	.00008
3	-.00001	.00007
4	negligible	.00008
5	negligible	.00008
6	negligible	.00008

PART III-B-2

2. **Wing Flapped**

The yawing moment of the flapped wing is not required by the design criteria requirements.

PART III-B-3

3. <u>Fuselage</u>

$$C_{N_F} = C_{N_{F.25\,L_f}} \frac{S_f}{S_w} \frac{L_f}{B_w} + C_{Y_F} \frac{S_f}{S_w} \left(\frac{x_f}{B_w} - \frac{.25\,L_f}{B_w} \right)$$

for both single engine and two engine operation.

<u>Zero Yaw - Single Engine Operation</u>

For zero yaw $C_{N_F} = 0$ for all C.G. Locations

<u>5° Yaw - V_D $V\sqrt{\sigma} = 313$ mph</u>

$$C_{N_{F.25\,L_f}} = C_{M_{F.25\,L_f}} \times \frac{P_f}{S_f}$$ page 145 reference (1)

From figure 19, reference (1)

$$C_{M_{F.25\,L_f}} = C_{M_{F.25\,L_f}} \times \frac{P_f}{S_f}$$ page 146 reference (1)

From figure 19, reference (1)

$$C_{M_{F.25\,L_f}} = .00968 \qquad \alpha_{TL} = 5^\circ$$

$$C_{N_{F.25\,L_f}} = .00968 \times \frac{579}{801} = .00697 \text{ for } \psi = 5^\circ$$

$$\frac{S_f}{S_w} = \frac{801}{1447.25} = .553$$

$$\frac{L_f}{B_w} = \frac{55.94}{109.27} = .512$$

$$C_{Y_F} = C_{C_F} \cos\psi + C_{D_F\,yaw} \frac{F_f}{S_f} \sin\psi$$

From page 146 and figures 16 and 17 of reference (1)

PART III-B-3

Fuselage (Cont.)

$C_{C_F} = .0234$ for $\psi = 5°$

$C_{D_{F_{yaw}}} = .1054$ for $\psi = 5°$

Then

$$C_{Y_F} = (.0234)(.9962)+(.1054)\left(\frac{175.3}{801}\right)(.0872)$$

$$= .02531$$

$$C_{N_F} = (.00697)(.553)(.512)+(.02531)(.553)\left(\frac{X_f}{B_w} - \frac{.25\ L_e}{B_w}\right)$$

$$= .00198 - .014\left(\frac{X_f}{B_w} - .128\right)$$

C. G. Location	X_f/B_w	C_{N_F}
1	.256	.00377
2	.268	.00394
3	.256	.00377
4	.268	.00394
5	.260	.00383
6	.258	.00380

PART II-B-4

4. Booms

Zero Yaw ~ Single Engine Operation

$$C_{N_B} = 2\left[C_{N_{B.25 L_b}} \frac{S_b L_b}{S_w B_w} + C_{Y_B} \frac{S_b}{S_w}\left(\frac{x_b}{B_w} - \frac{.25 L_b}{B_w}\right) + \Delta C_{X_B \text{ cowl flaps}} \frac{y_b}{B_w} \frac{F_b}{S_w}\right.$$

For zero yaw this equation reduces to

$$C_{N_B} = \Delta C_{D_{B \text{ cowl flaps}}} \frac{y_b}{B_w}$$

where ΔC_{D_B} is the drag coefficient of the open cowl flaps on the operating engine and is based on q S_w cowl flaps on the non operating engine being closed.

$$\Delta C_{D_{\text{cowl flaps}}} = .0037 \text{ for one engine}$$

reference (10)

Then for all C. G. locations

$$C_{N_B} = .0037 \times \frac{14.59}{109.27} = .00048$$

5° Yaw ~ V_D $V\sigma^{\frac{1}{2}}$ = 313 mph

$$C_{N_B} = 2\left[C_{N_{B.25 L_b}} \frac{S_b L_b}{S_w B_w} + C_{Y_B} \frac{S_b}{S_w}\left(\frac{x_b}{B_w} - \frac{.25 L_b}{B_w}\right)\right]$$

$C_{N_{B.25L_b}}$ = .0056

From figure 30 reference 1

$$C_{Y_B} = C_{C_B} \cos\psi + C_{D_{B \text{ yaw}}} \frac{F_b}{S_b} \sin\psi$$

$$C_{D_{B \text{ yaw}}} = C_{D_{B \text{ pitch}}} \text{ for corresponding } \alpha_{TL}$$

From figure 27 reference 1

$$C_{D_B} = .1338$$

PART III-B-4 (Cont.)

4. Booms (Cont.)

From figure 31, reference 1

$$C_{O_B} = .0055$$

$$C_{Y_B} = (.0055)(.9962) - (.1938)\left(\frac{40.76}{363.2}\right)(.0872)$$

$$= .00739$$

① C.G. Position	② X_b	③ $\frac{X_b}{B_w}$	④ ③ $- \frac{.25\, L_b}{B_w}$	⑤ $C_{Y_B} \frac{S_b}{S_w} \times$ ④	⑥ $C_{N_B .25\, L_b} \frac{S_b\, L_b}{S_w\, B_w} +$ ⑤	⑦ $C_{N_B} = 2 \times$ ⑥
1	166.1	.1266	-.0300	-.0000556	.00146	.00292
2	182.9	.1493	-.0073	-.0000135	.00150	.00300
3	166.1	.1266	-.0300	-.0000556	.00146	.00292
4	182.9	.1493	-.0073	-.0000135	.00150	.00300
5	172.5	.1316	-.0250	-.0000464	.00147	.00294
6	169.8	.129	-.0276	-.0000512	.00147	.00294

PART III-B-5

5. Horizontal Tail

It is assumed that the yawing moment of the horizontal tail due to yaw is negligible.

PART III-B-6

6. Landing Gear

The yawing moments of the landing gear down are not required by flight criteria requirements.

PART III-B-7

7. Propeller

For single engine operation, zero yaw, the yawing moments of the propellers will be identical with those of the C-119B airplane. Complete calculations may be found in report number R110-008 pages 395 and 396. A curve of propeller yawing moment versus air speed follows in figure 83.

5° Yaw $\sim V_D$ $\quad V\sigma^{\frac{1}{2}} = 313$ mph

$$C_{N_p} = -2 \frac{l_p}{B_w} \frac{F_p}{S_w} C_{Y_p}$$

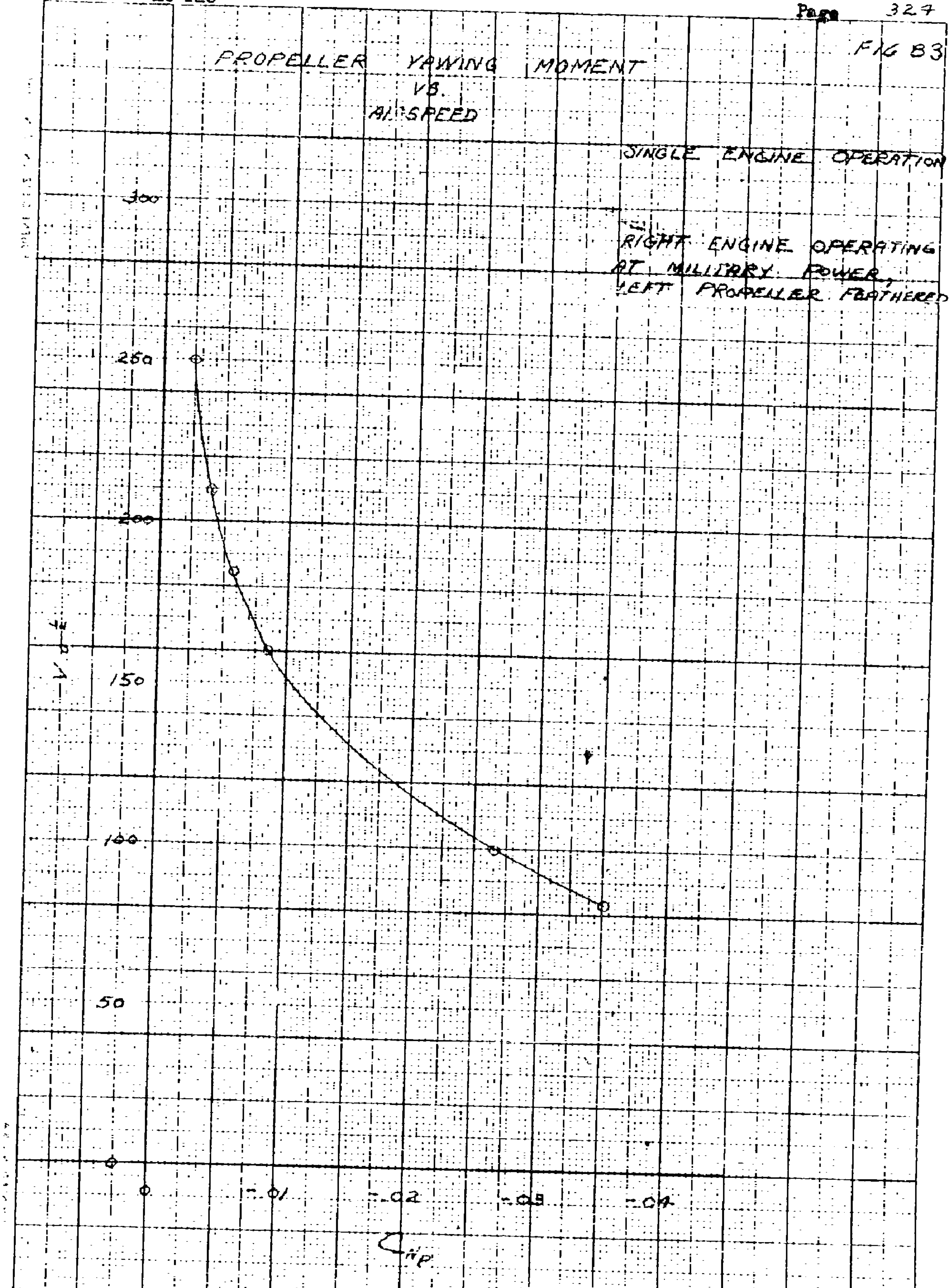
$C_{Y_p} = .0265$ figure 59 reference (1)

$$C_{N_p} = -2 \times \frac{178.2}{109.27 \times 12 \times 1447.25} \times .0265\ l_p$$

$$C_{N_p} = -.00000497\ l_p$$

C. G. Position	l_p	C_{N_p}
1	-174.5	.00087
2	-191.3	.00095
3	-174.5	.00087
4	-191.3	.00095
5	-180.9	.00090
6	-178.2	.00089

FIG 83


PROPELLER YAWING MOMENT
VS.
AIRSPEED
SINGLE ENGINE OPERATION
RIGHT ENGINE OPERATING
AT MILITARY POWER,
LEFT PROPELLER FEATHERED
300
250
200
150
100
50
-.01
-.02
-.03
-.04

PART III-B-8

8. <u>Slipstream Effects</u>

The yawing moments due to slipstream effects are negligible for two engine operation, but for single engine operation may be appreciable as they vary with speed, airplane angle of attack, and thrust.

<u>Single Engine Operation</u>

$$C_{N_S} = \Delta X_S \frac{y_b}{q\, S_w\, B_w}$$

a. <u>Wing Unflapped</u>

<u>Zero Yaw - Single Engine Operation</u>

For the unflapped wing, zero yaw, single engine operation, yawing moments due to slipstream effects are determined at various angles of attack for a range of speeds as required by design criteria.

For zero yaw $\Delta X_S = \Delta D_S$

$$\Delta C_{D_{WI}} = \frac{\Delta D_S}{q\, S_w} \quad \text{figure 118 reference 1}$$

so that for all C. G. Locations

$$C_{N_S} = \Delta C_{D_{WI}} \frac{y_b}{B_w} \text{ and } \frac{y_b}{B_w} = \frac{14.59}{109.27} = .1332$$

PART III-B-8a

$V\sqrt{\sigma}$ = 100 T_C = .639 q = 25.58

α_{TL}	-12	-8	-4	0	4	8
$\Delta C_{D_{WI}}$	.00545	.0009	-.0004	.0027	.0101	.0225
C_{N_S}	.00075	.00012	-.000053	.000358	.00133	.00299

$V\sqrt{\sigma}$ = 160 T_C = .198 q = 65.5

α_{TL}	-12	-8	-4	0	4	8
$\Delta C_{D_{WI}}$	.0031	.00021	-.0007	.00095	.0048	.0106
C_{N_S}	.000413	.000028	-.000093	.000126	.000638	.00141

$V\sqrt{\sigma}$ = 185 T_C = .132 q = 87.5

α_{TL}	-12	-8	-4	0	4	8
$\Delta C_{D_{WI}}$	.00241	.0001	-.0007	.0006	.0037	.0086
C_{N_S}	.00032	.0000133	-.000093	.0008	.000493	.000114

$V\sqrt{\sigma}$ = 250 T_C = .056 q = 159.8

α_{TL}	-12	-8	-4	0	4	8
$\Delta C_{D_{WI}}$	.0019	.00005	-.0007	.0001	.0025	.0058
C_{N_S}	.000252	.0000067	-.000093	.000013	.00033	.00077

Figure 84 on the following page shows the variation of yawing moment due to slipstream effects with angle of attack of the thrust line for various speeds.

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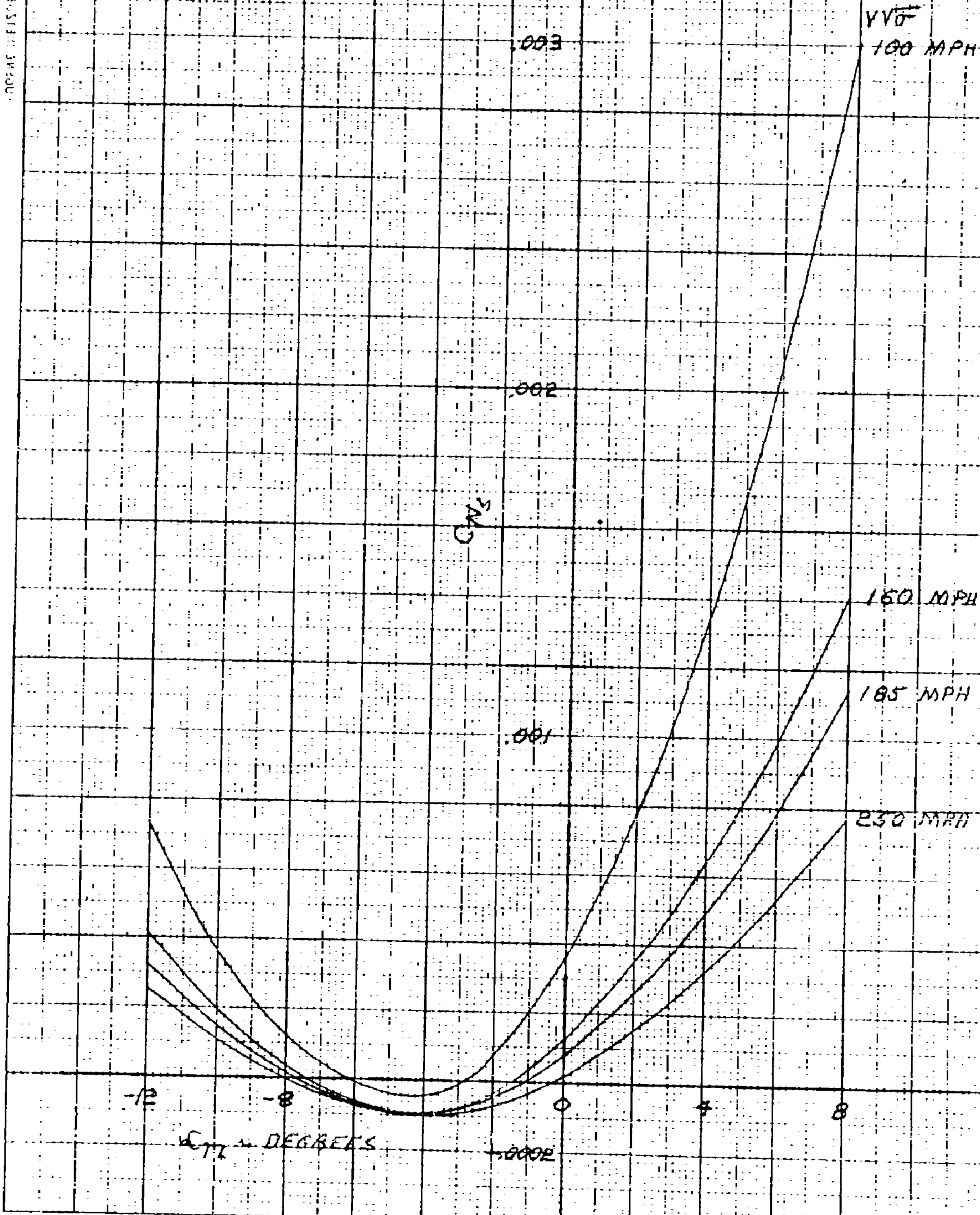
FIG 84

YAWING MOMENT COEFFICIENTS DUE TO SLIPSTREAM, LEFT ENGINE INOPERATIVE AND PROPELLER FEATHERED, RIGHT ENGINE MILITARY POWER, ZERO YAW WING UNFLAPPED



PART III-B-8a

8a. Two Engine Operation

For two engine operation the yawing moment due to slipstream is zero.

PART III-B-8b

8b. Wing Flapped

The yawing moment due to slipstream effects on the flapped wing are not required by the flight criteria requirements.

For two engine operation the yawing moment would be zero.

PART III-B-9a

9. Vertical Tail

This section covers the determination of the yawing moments of the vertical tail for zero yaw, single engine operation; and for 5° yaw, two engine operation; for various degrees of rudder deflection.

a. Single Engine Operation

For single engine operation the dynamic pressure at the vertical tail in the slipstream of the operating engine will be difference from that at the vertical tail behind the non-operating engine. As a result the X and Y forces of the vertical tails will differ for the same rudder deflection. It is assumed that the motion of the vertical tail center of pressure is a negligible small portion of the total tail length and is located at the rudder hinge line so that with l_{vt} equal to distance from the C. G. location to the center of pressure of the vertical tail

then

$$C_{N_{VT}} = \frac{S_{vt}}{S_w}\left\{\left[C_{X_{VT}}\frac{q_{vt}}{q_L}\frac{y_b}{B_w} + C_{X_{VT}}\frac{q_{vt}}{q_R}\frac{y_b}{B_w}\right] + \left[C_{Y_{VT}}\frac{q_{vt}}{q_L} - \frac{l_{vt}}{B_w} + C_{Y_{VT}}\frac{q_{vt}}{q_R} - \frac{l_{vt}}{B_w}\right]\right\}$$

Where subscripts L and R denote left and right vertical surfaces

$$C_{Y_{VT}} = \eta_{vt}\ C_{C_{VT}} = .75\ (.0560\,\alpha_{vt} + .0269\,\delta_r)$$

from Part II-9-b, reference (1)

PART III-B-9a (Cont.)

For Zero Yaw

$$C_{Y_{VT}} = .75(.0269\,\delta_e)$$

As the q_{vt}/q ratio is smaller at low gross weight for any given speed and since the lowest q_{vt}/q will yield the maximum rudder deflections to maintain zero yaw, the q_{vt}/q ratio for a gross weight of 40,451. lbs. as shown in figure 65 reference (1) is used in determining the yawing moments of the vertical tail for given rudder deflections.

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REPORT NO. R107-014	FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION		PAGES	PAGE 332
MODEL XC-120	PREPARED BY	CHECKED BY	APPROVED BY	

Subject: BASIC FLIGHT CRITERIA - PACK ON

DATE 15 March 1949

REVISED

PART III-B-9a

YAWING MOMENT OF VERTICAL TAIL
SINGLE ENGINE OPERATION
$\delta_r = 0^\circ$

For All C. G. Locations

1	2	3	4	5	6	7	8	9	10	11
$V\sqrt{\sigma}$	$\frac{S_{vt}}{S_w}$	$C_{x_{VT}}$	$\frac{Y_c}{B_w}$ (−L, +R)	$\left(\frac{q_{vt}}{q}\right)_L$	$\left(\frac{q_{vt}}{q}\right)_R$	$C_{x_{VT}}\frac{Y_c}{B_w}\left(\frac{q_{vt}}{q}\right)_L$ −L	$C_{x_{VT}}\frac{Y_c}{B_w}\left(\frac{q_{vt}}{q}\right)_R$ +R	⑦+⑧	$C_{Y_{VT}}$	$-\frac{l_{vt}}{B_w}$
100	.0687	.0082	.133	1.00	1.49	−.00109	+.00157	+.00048	0	−.474
160					1.103		.00120	+.00011		
185					1.06		.00115	+.00006		
250					1.022		.00111	+.00002		

12	13	14	15	16
⑩×⑪×⑮	⑩×⑪×⑥	⑫+⑬	⑨+⑭	$C_{N_{VT}}$ = ②×⑮
0	0	0	+.00048	+.000033
			+.00011	—
			+.00006	—
			+.00002	—

Subject: BASIC FLIGHT CRITERIA - PACK ON

PART III-B-9a

YAWING MOMENT OF VERTICAL TAIL
SINGLE ENGINE OPERATION
$\delta_r = -5°$

1	2	3	4	5	6	7	8	9	10	11
$V\sqrt{\sigma}$	$\frac{S_{vt}}{S_w}$	$C_{X_{VT}}$	$\frac{Y_b}{B_w}$ $-L$ $+R$	$\left(\frac{q_{vt}}{q}\right)_L$	$\left(\frac{q_{vt}}{q}\right)_R$	$C_{X_{VT}}\frac{Y_b}{B_w}\left(\frac{q_{vt}}{q}\right)_L$	$C_{X_{VT}}\frac{Y_b}{B_w}\left(\frac{q_{vt}}{q}\right)_R$	(7) + (8)	$C_{Y_{VT}}$	$-\frac{l_{vt}}{B_w}$
100	.0687	.0121	.133	1.00	1.49	-.00161	.00232	.00071	-.1010	-.474
160	↓	↓	↓	↓	1.109	↓	.00177	.00016	↓	↓
165					1.06		.00171	.00010		
250					1.023		.00165	.00003		

12	13	14	15	16	17	18	19	20
C.G. LOCATIONS 1+3					C.G. LOCATIONS 2+4		C.G. LOCATION 6	
(10)×(11)×(3)	(10)×(11)×(6)	(12)+(13)	(9)+(14)	$C_{N_{VT}} =$ (2)×(16)	$-\frac{l_{vt}}{B_w}$	$C_{N_{VT}}$	$-\frac{l_{vt}}{B_w}$	$C_{N_{VT}}$
+.0476	.0686	.1162	.11691	.00800	-.461	.00780	-.469	.00790
↓	.0525	.1001	.10026	.00688	↓	.00670	↓	.00678
	.0505	.0981	.0982	.00675		.00658		.00668
	.0488	.0964	.0643	.00662		.00646		.00655

REPORT NO. R107-014 FAIRCHILD AIRCRAFT DIVISION OF FAIRCHILD ENGINE & AIRPLANE CORPORATION PAGES PAGE 334
MODEL XC-120 PREPARED BY CHECKED BY APPROVED BY
DATE 15 March 1949
Subject: BASIC FLIGHT CRITERIA - PACK ON REVISED

PART III-B-9a

YAWING MOMENT OF VERTICAL TAIL
SINGLE ENGINE OPERATION
$\delta_r = -10°$

1	2	3	4	5	6	7	8	9	10	11
$V\sqrt{\sigma}$	$\frac{S_{VT}}{S_W}$	$C_{X_{VT}}$	$\frac{Y_b}{B_W}$ (−L +R)	$\left(\frac{q_{VT}}{q}\right)_L$	$\left(\frac{q_{VT}}{q}\right)_R$	$C_{X_{VT}}\cdot\frac{Y_b}{B_W}\left(\frac{q_{VT}}{q}\right)_L$	$C_{X_{VT}}\cdot\frac{Y_b}{B_W}\left(\frac{q_{VT}}{q}\right)_R$	⑦+⑧	$C_{Y_{VT}}$	$-\frac{l_{VT}}{B_W}$
[illegible]	.0081	.1248	.133	1.000	1.[illegible]	−.0033	.00470	.0014	−.2020	−.474
[illegible]	↓	↓	↓	↓	1.103	↓	.00352	.00022	↓	↓
[illegible]					1.0[illegible]		.00338	.00006		
[illegible]					1.023		.00331	.00001		

12	13	14	15	16	17	18	19	20
	C.G. LOCATIONS		1 + 3		C.G. LOCATIONS	2 + 4	C.G. LOCATIONS	5
①×②×⑤	⑩×⑪×⑥	⑫+⑬	⑨+⑭	$C_{N_{VT}}$ ②×⑮	$-\frac{l_{VT}}{B_W}$	$C_{N_{VT}}$	$-\frac{l_{VT}}{B_W}$	$C_{N_{VT}}$
+.0955	+.1361	+.2316	.2330	.0160	−.461	.0156	−.469	.0158
↓	+.1018	+.1973	.19752	.0136	↓	.0132	↓	.0134
	+.0978	+.1933	.19338	.0133		.01295		.0132
	+.0970	+.1925	.19254	.0132		.01285		.01305

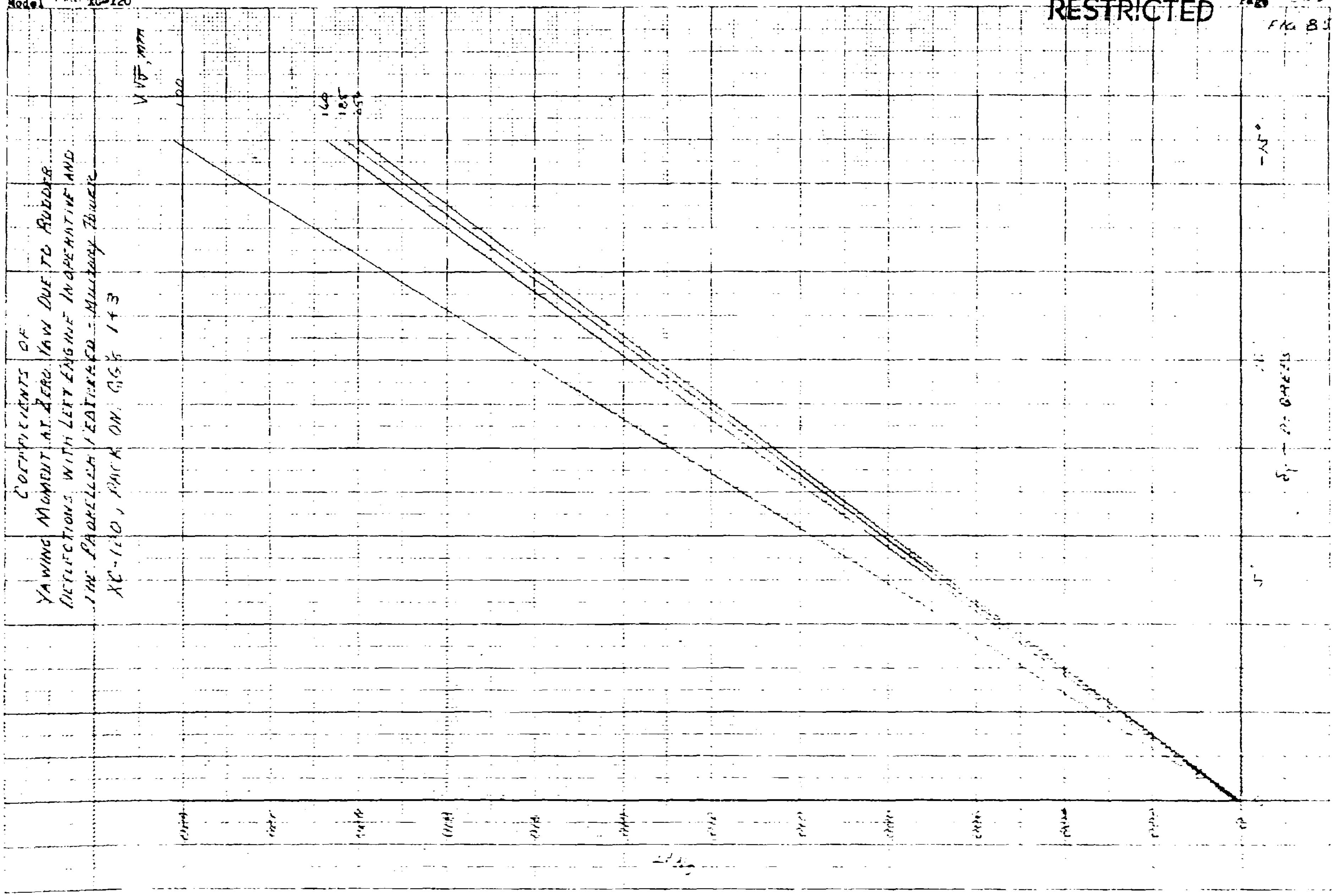
Subject:— BASIC FLIGHT CRITERIA - PACK ON

PART III-B-9a

YAWING MOMENT OF VERTICAL TAIL
SINGLE ENGINE OPERATION
$\delta_r = -15^\circ$

1	2	3	4	5	6	7	8	9	10	11
$V\sqrt{\sigma}$	$\frac{S_{vt}}{S_w}$	$C_{x_{VT}}$	$\frac{y_b}{B_w}$ (−L +R)	$\left(\frac{q_{VT}}{q}\right)_L$	$\left(\frac{q_{VT}}{q}\right)_R$	$C_{x_{VT}} \cdot \frac{y_b}{B_w}\left(\frac{q_{VT}}{q}\right)_L$	$C_{x_{VT}} \cdot \frac{y_b}{B_w}\left(\frac{q_{VT}}{q}\right)_R$	(7) + (8)	$C_{Y_{VT}}$	$-\frac{l_{vt}}{B_w}$
100	.0667	.0456	.133	1.00	1.44	−.00606	.00872	.00264	−.3030	−.474
120	↓	↓	↓	↓	1.163	↓	.00668	.00060	↓	↓
145	↓	↓	↓	↓	1.66	↓	.00643	.00035	↓	↓
250	↓	↓	↓	↓	1.623	↓	.00620	.00012	↓	↓

12	13	14	15	16	17	18	19	20
	C.G. LOCATIONS 1 + 3				C.G. LOCATIONS 2 + 4		C.G. LOCATION 5	
(10)×(11)×(5)	(10)×(11)×(6)	(12)+(13)	(9)+(14)	$C_{N_{VT}}$ (2)×(15)	$-\frac{l_{vt}}{B_w}$	$C_{N_{VT}}$	$-\frac{l_{vt}}{B_w}$	$C_{N_{VT}}$
+.1435	.2060	.3495	.35214	.0242	−.461	.0236	−.469	.0240
↓	.1580	.3015	.3021	.0202	↓	.0201	↓	.0205
↓	.1520	.2955	.29585	.0203	↓	.0198	↓	.0201
↓	.1410	.2905	.29062	.0199	↓	.0194	↓	.0197


COEFFICIENTS OF
YAWING MOMENT AT ZERO YAW DUE TO RUDDER
DEFLECTIONS WITH LEFT ENGINE INOPERATIVE AND
THE PROPELLER FEATHERED - MILITARY POWER
XC-120, PACK ON. CG 14.3
V√σ, mph
100
160
185
250
-30°
δ_r — DEGREES

COEFFICIENTS OF
YAWING MOMENT AT ZERO YAW DUE TO RUDDER
DEFLECTIONS WITH LEFT ENGINE INOPERATIVE AND
THE PROPELLER FEATHERED + MILITARY POWER

XC-120, PACK ON, C.G.'S 24.4

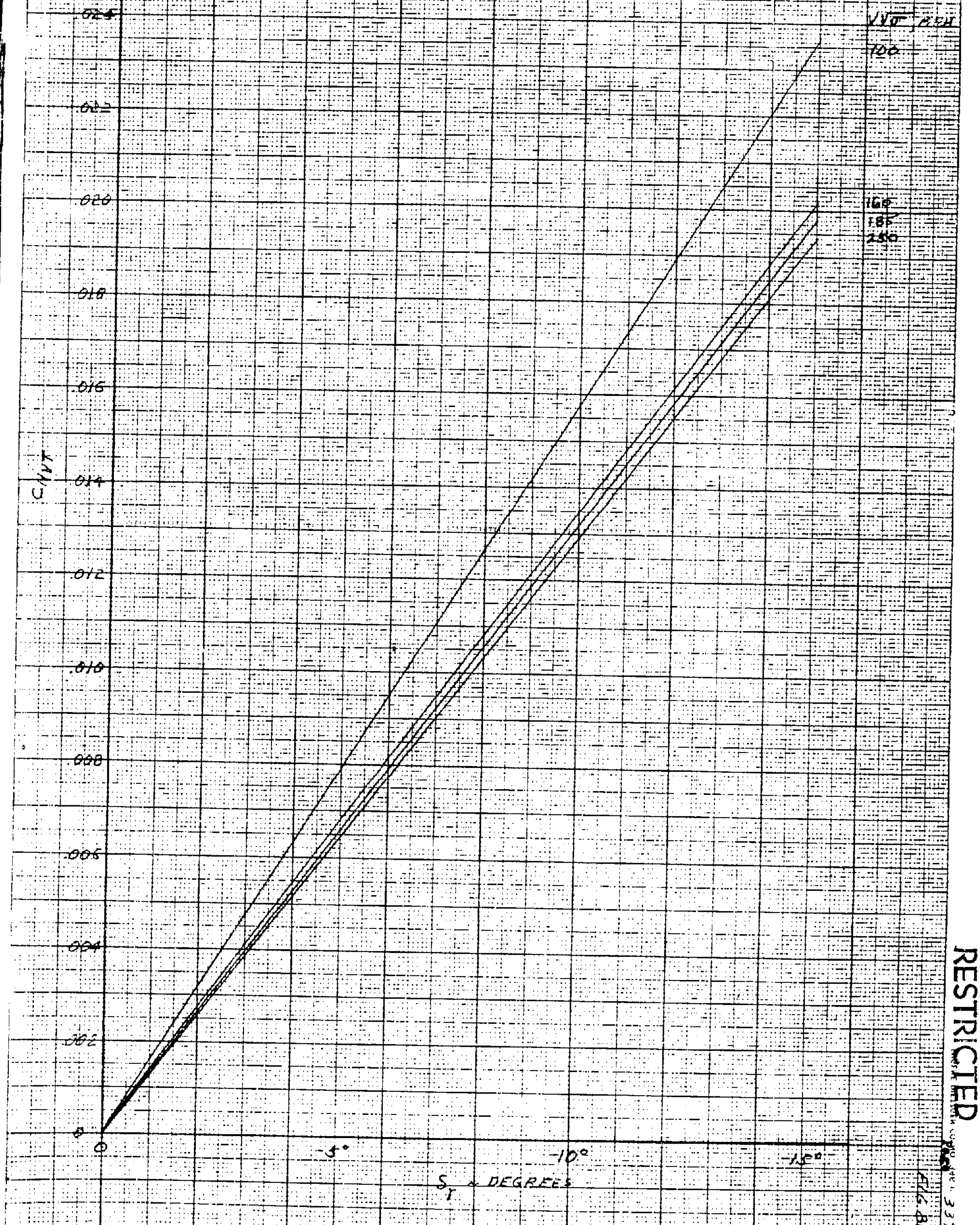


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FIG. 86

Report No. E107-014
Model XC-120

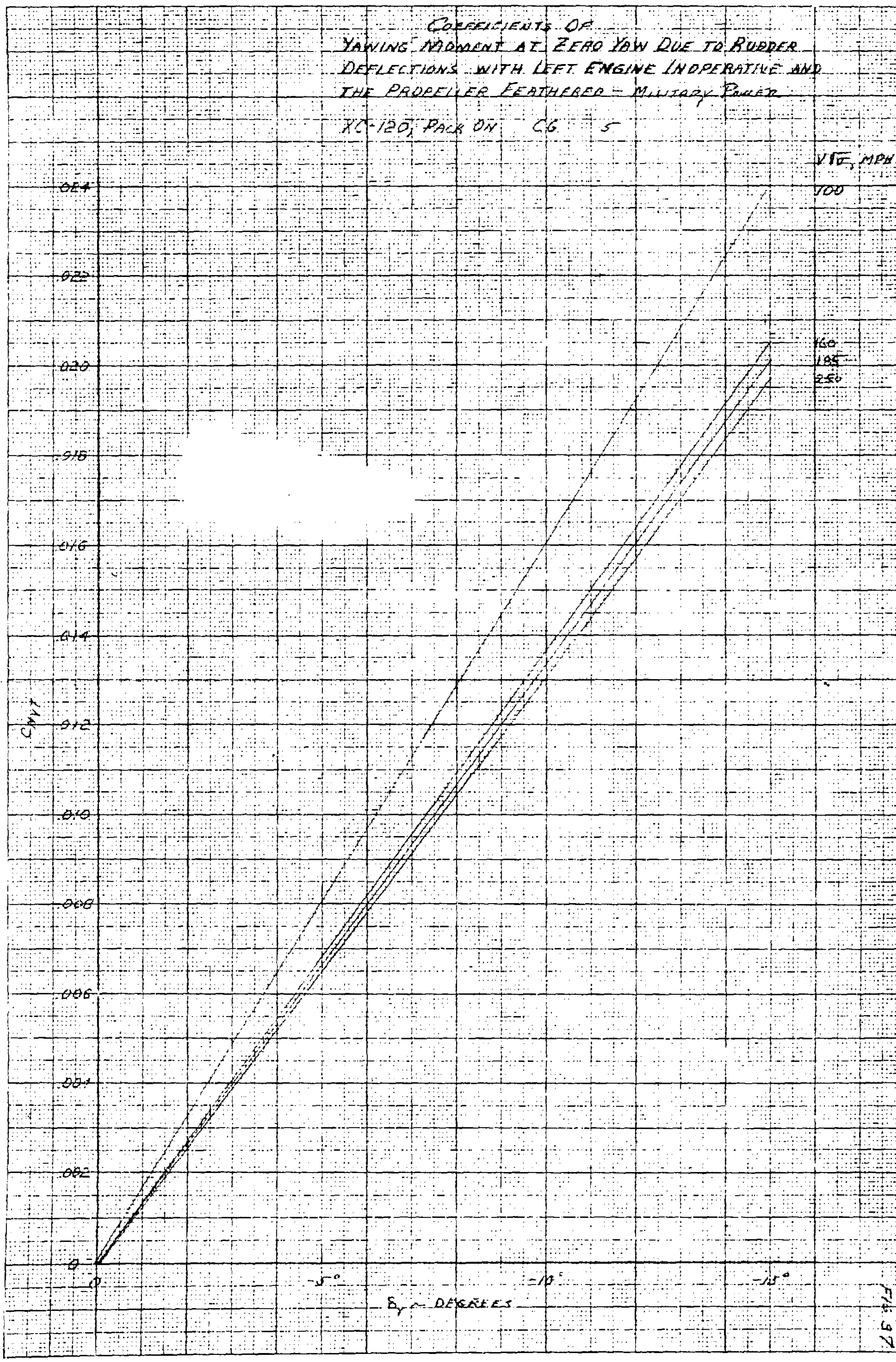

FIG. 97

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PART III-B-9b

9b. Two Engine Operation

$$C_{N_{VT}} = 2\ C_{Y_{VT}} \frac{q_{vt}}{q} \frac{S_{vt}}{S_w} - \frac{l_{vt}}{B_w}$$

5° Yaw at V_D $\quad V\sqrt{\sigma} = 313$ mph

$$\alpha_{vt} = \psi - \sigma_{p_{ave}} - \psi \frac{d\sigma}{d\psi}$$

$\frac{d\sigma}{d\psi} = .107$ Page 241 reference (1)

$\sigma_{p_{ave}} = .131$ Page 336 reference (1)

$$\alpha_{vt} = 5 - .131 - (5 \times .107) = 4.334^\circ$$

δ_r	0°	-5	-10°	-15°
$C_{Y_{VT}}$	.1821	.0814	-.0196	-.1201

$C_{Y_{VT}} = \eta_{vt}\, C_{C_{VT}} = .75(.056\alpha_{vt} + .0269\ \delta_r)$ from Part II-A-9 reference (1)

$\frac{q_{vt}}{q} = 1.0063$ reference (1) Part II-B-11

$$\frac{S_{vt}}{S_w} \frac{B_{vt}}{B_w} = \frac{99.6}{1447.25} \quad \frac{16.67}{109.27} = .0105$$

$$\frac{q_{vt}}{q} \left(\frac{S_{vt}}{S_w} \cdot \frac{B_{vt}}{B_w}\right) = 1.0063 \times .0105 = .0106$$

$$\frac{q_{vt}}{q} \quad \frac{S_{vt}}{S_w} = 1.0063 \times \frac{99.6}{1447.25} = .06925$$

PART III-B-9b (Cont.)

For C. G. Location 1 and 3

$$-\frac{l_{vt}}{B_w} = \frac{621.1}{109.27 \times 12} = -.4737$$

$$\delta_r = 0^\circ$$

$$C_{N_{VT}} = 2 \times .1821 \times .06925 \times -.4737 = -.01194$$

$$\delta_r = -5^\circ$$

$$C_{N_{VT}} = 2 \times .0814 \times .06925 \times -.4737 = -.00534$$

$$\delta_r = -10^\circ$$

$$C_{N_{VT}} = 2 \times -.0196 \times .06925 \times -.4737 = +.001285$$

$$\delta_r = -15^\circ$$

$$C_{N_{VT}} = 2 \times -.1201 \times .06925 \times -.4737 = +.00787$$

For C. G. Locations 2 and 4

$$-\frac{l_{vt}}{B_w} = \frac{604.2}{109.27 \times 12} = -.4608$$

$$\delta_r = 0^\circ$$

$$C_{N_{VT}} = 2 \times .1821 \times .06925 \times -.4608 = -.01161$$

$$\delta_r = -5^\circ$$

$$C_{N_{VT}} = 2 \times .0814 \times .06925 \times -.4608 = -.00519$$

$$\delta_r = -10^\circ$$

$$C_{N_{VT}} = 2 \times -.0196 \times .06925 \times -.4608 = .00125$$

PART III-B-9b (Cont.)

$\delta_r = -15^{\circ}$

$C_{N_{VT}} = 2 \times -.1201 \times .06925 \times -.4608 = .00766$

$-\frac{l_{vt}}{B_w} = \frac{614.8}{109.27 \times 12} = -.4689$

$\delta_r = 0^{\circ}$

$C_{N_{VT}} = 2 \times .1821 \times .06925 \times -.4689 = -.01182$

$\delta_r = -5^{\circ}$

$C_{N_{VT}} = 2 \times .0814 \times .06925 \times -.4689 = -.00529$

$\delta_r = -10^{\circ}$

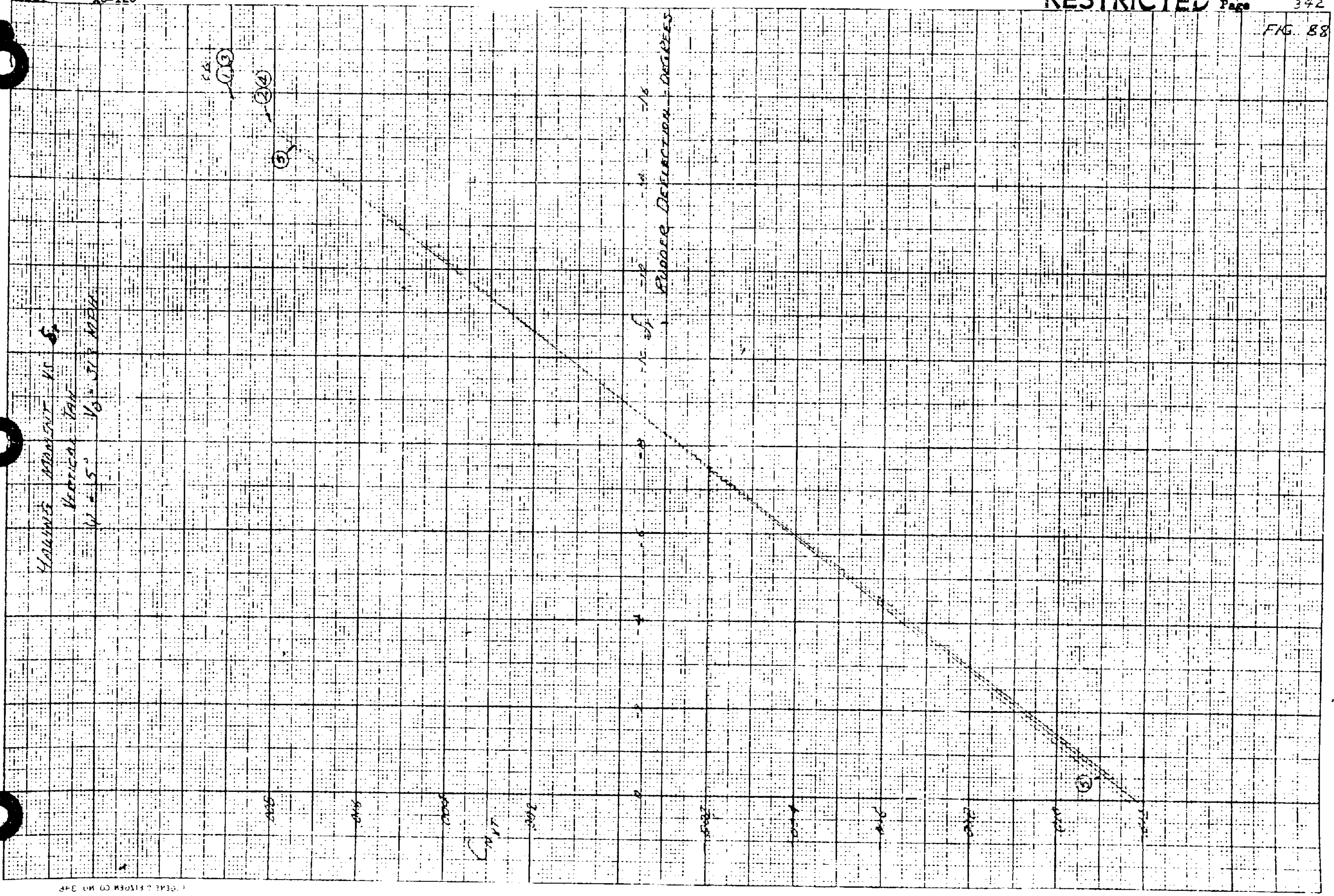
$C_{N_{VT}} = 2 \times -.0196 \times .06925 \times .4689 = .001272$

$\delta_r = -15^{\circ}$

$C_{N_{VT}} = 2 \times -.1201 \times .06925 \times .4689 = .00779$

Figure 88 on the following page shows the variation of yawing moment due to the vertical tail versus rudder deflections for 5 degrees yaw at 313 mph (V_D).

FIG. 88


Yawing Moment vs δr
Vertical Tail
ψ = 5° V0 = 313 MPH
Rudder Deflection - Degrees

PART III-C-1

C. **Yawing Moment of the Tailless Airplane**

This section covers the summation of the yawing moments of the component parts to determine the yawing moment of the tailless airplane.

1. **Zero Yaw Single Engine Operation**

$$C_{N_{A-T}} = C_{N_W} + C_{N_F} + C_{N_B} + C_{N_S} + C_{N_P}$$

Using minimum gross weight and military power which gives the most critical conditions, the yawing moment of the tailless airplane is determined for a range of speeds.

$V\sqrt{\sigma}$	100	160	185	250
C_L	1.140	.445	.370	.182
α_{TL}	5.45°	-2.75°	-3.7°	-5.9°
C_{N_W}	0	0	0	0
C_{N_F}	0	0	0	0
C_{N_B} cowl flaps	.00048	.00048	.00048	.00048
C_{N_S}	.00187	-.00007	-.00010	-.00008
C_{N_P}	-.02652	-.00842	-.00568	-.00244
$C_{N_{A-T}}$	-.02417	-.00801	-.00530	-.00204

$$C_L = \frac{391 \times 42136}{\sigma \times 1447.24\, V^2} \quad (\sigma = 1 \text{ at sea level})$$

α_{TL} is from figure 2, page 21, reference (1)

PART III-C-2

2. 5° Yaw V_D $V\sqrt{\sigma}$ = 313 mph

$$C_{N_{A-T}} = \left[C_{N_W} + C_{N_F} + C_{N_B} + C_{N_P}\right] F_I$$

C. G. Position	C_{N_W}	C_{N_F}	C_{N_B}	C_{N_P}	$\left(C_{N_{A-T}} / F_I\right)$
1	.00007	.00377	.00292	.00087	.00763
2	.00008	.00394	.00300	.00095	.00787
3	.00007	.00377	.00292	.00087	.00763
4	.00008	.00394	.00300	.00095	.00797
5	.00008	.00383	.00294	.00090	.00775
6	.00008	.00380	.00294	.00089	.00771

From reference (5) pages 440 and 441 we shall use an overall interference factor of .65 for yawed conditions.

Considering interference effects: F_I = .65

C. G. Position	$\left(C_{N_{A-T}} / FI\right)$	$C_{N_{A-T}}$
1	.00763	.00497
2	.00797	.00518
3	.00763	.00497
4	.00797	.00518
5	.00775	.00503
6	.00771	.00501

PART III-D

D. Vertical Tail Loads — Tabs Neutral

This section covers the determination of the vertical tail loads as required by the design criteria requirements.

The vertical tail loads are determined for the following conditions, flaps and gear up.

(a) A side gust of 50 fps at $V\sqrt{\sigma} = 250$ mph with initial $\alpha_{vt} = 0$

(b) Zero yaw with one engine dead and the other engine delivering take-off power for speeds up to $V\sqrt{\sigma} = 250$ mph

(c) Zero yaw with one engine dead and the other engine delivering military power at V_{max} ($V\sqrt{\sigma} = 190$ mph) for this condition plus a side gust of 30 fps.

(d) Five degrees yaw with resulting rudder deflection at V_D ($V\sqrt{\sigma} = 313$ mph)

(e) Five degrees yaw with zero rudder deflection at V_D ($V\sqrt{\sigma} =$ 313 mph).

A summary of all loads is presented in a table following the calculations.

All loads given herein are for one tail only. A comparison of the aerodynamic characteristics of the XC-120 vertical tail as determined in reference (1) with the aerodynamic characteristics of the modified vertical tail (upper portion only) of the C-119B as determined in reference (12) indicates that approximately 84% of the total loads shown are carried by the upper portion of the tail.

PART III-D

Vertical Tail Load for a Side Gust of 50 ft. per Second at $V\sqrt{\sigma}$ = 250 mph with Initial α_{vt} = 0°

$$V\sqrt{\sigma} = 250 \text{ mph} \qquad q = 159.8$$

$$\text{initial } \alpha_{vt} = 0^\circ$$

$$\frac{q_{vt}}{q} = 1.024 \qquad \text{figure 65 reference (1)}$$

$$\Delta\alpha_{vt} = \frac{U}{V} = \frac{50}{250 \times 1.467} = .1364 = 7.8^\circ$$

$$\alpha_{vt} = 0 - 7.8 = 7.8^\circ \qquad \delta_r = 0^\circ$$

$$\eta_{vt}\, C_{\alpha_{VT}} = .75(.056\,\alpha_{vt} + .0269\,\delta_r)$$

$$\eta_{vt}\, C_{\alpha_{VT}} = .328$$

$$L_{VT} = \eta_{vt}\, C_{\alpha_{VT}} \times q \times \frac{q_{vt}}{q} \times S_{vt}$$

$$L_{VT} = .328 \times 159.5 \times 1.024 \times 99.6$$

$$L_{VT} = 5346 \text{ lbs.} \qquad \text{(each vertical tail)}$$

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REPORT NO. E107-014

FAIRCHILD AIRCRAFT DIVISION
OF FAIRCHILD ENGINE & AIRPLANE CORPORATION

PAGES | PAGE 347

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

DATE 15 March 1949

REVISED

Subject:- BASIC FLIGHT CRITERIA - PACK ON

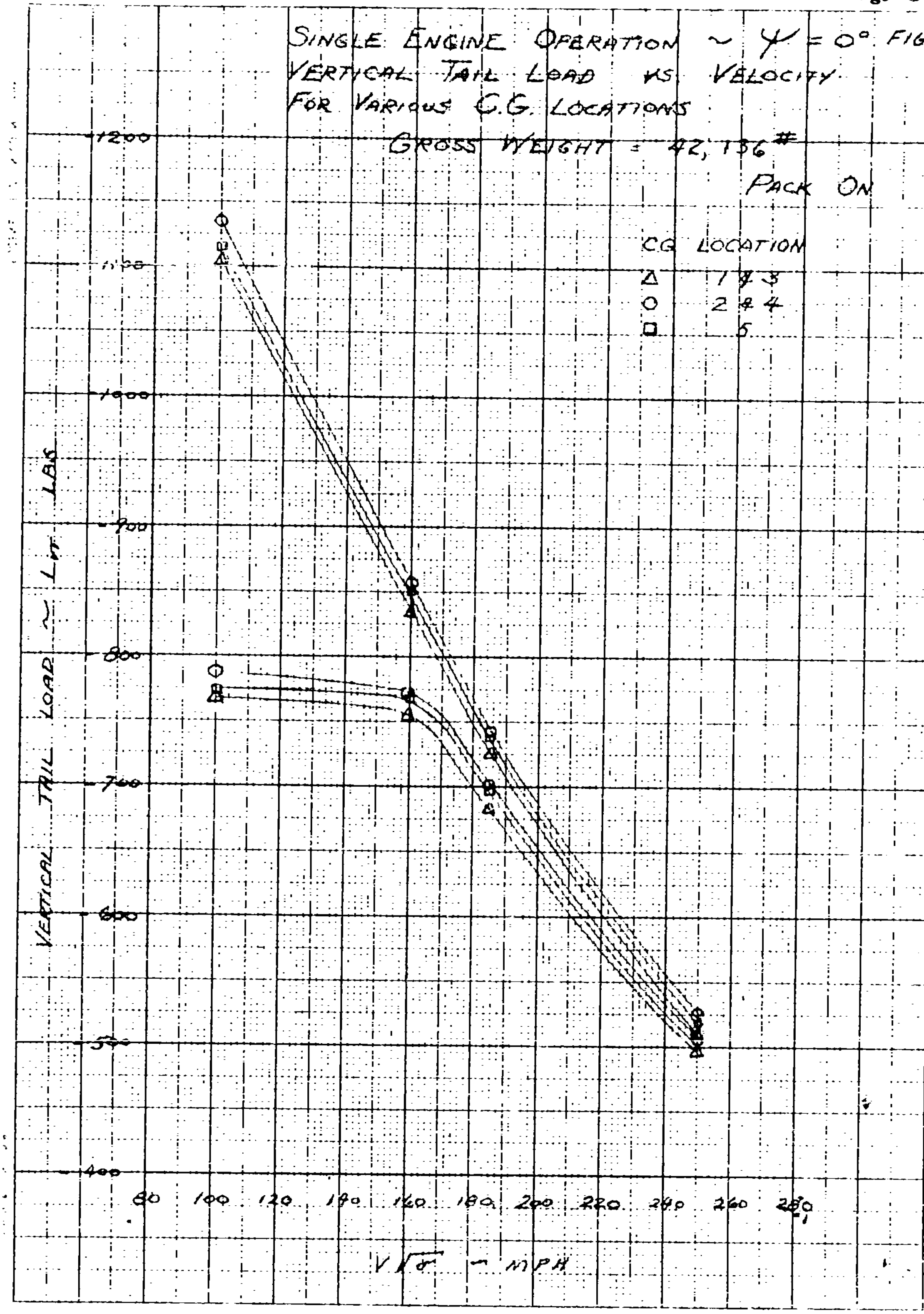
PART III-D

VERTICAL TAIL LOADS FOR ZERO YAW WITH ONE ENGINE DEAD AND THE OTHER ENGINE DELIVERING TAKE-OFF (MILITARY) POWER FOR VARIOUS SPEEDS

C.G	1&3	2&4	5	1&3	2&4	5	1&3	2&4	5	1&3	2&4	5
$V\sigma^{1/2}$	100			160			185			250		
q #/□'	25.6			65.5			87.5			160		
$C_{N_{A-T}}$	-.02417			-.00801			-.00530			-.00204		
$C_{N_{VT}}$	.02417			.00801			.00530			.00204		
δ_r °	-14.95	-15.35	-15.1	-5.76	-5.9	-5.89	-3.90	-4.0	-3.97	-1.55	-1.6	-1.58
α_{vt} ° *	0	0	0	0	0	0	0	0	0	0	0	0
α_{vt} °	0	0	0	0	0	0	0	0	0	0	0	0
$\eta_{vt}C_{C_{VT}}$ *	-.301	-.309	-.304	-.116	-.119	-.1187	-.0785	-.0805	-.080	-.0313	-.0323	-.0318
$\eta_{vt}C_{C_{VT}}$	-.301	-.309	-.304	-.116	-.119	-.1187	-.0785	-.0805	-.080	-.0313	-.0323	-.0318
q_{vt}/q *	1.44	1.44	1.44	1.103	1.103	1.103	1.06	1.06	1.06	1.023	1.023	1.023
q_{vt}/q	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00
L_{VT} *	-1105	-1135	-1115	-835	-856	-855	-725	-743	-740	-510	-526	-518
L_{VT}	-768	-788	-774	-755	-773	-772	-684	-702	-699	-498	-513	-506

* Indicates tail in slipstream of operating engine

$$\eta_{vt} C_{C_{VT}} = .75(.056\,\alpha_{vt} + .0269\,\delta_r)$$


SINGLE ENGINE OPERATION ~ ψ = 0° FIG 89
VERTICAL TAIL LOAD VS. VELOCITY
FOR VARIOUS C.G. LOCATIONS
GROSS WEIGHT = 42,136#
PACK ON
CG LOCATION
△ 1 & 3
○ 2 & 4
□ 5
1200
1100
1000
900
800
700
600
500
400
80
100
120
140
160
180
200
220
240
260
280
V√σ – MPH
VERTICAL TAIL LOAD ~ LVT LBS

PART III-D

Vertical Tail Loads with One Engine Dead and the Other Engine Delivering Military Power at $V\sqrt{\sigma} \cong 190$ mph Plus a Side Gust of 30 Ft. per Second at Zero Yaw

$$\Delta\alpha_{vt} \text{ due to gust} = \frac{\pm 30}{190 \times 1.467} = \pm .1075 = \pm 6.12$$

C. G. Location	① and ③		② and ④		⑤	
$C_{N_{A-T}}$	-.00495		-.00495		-.00495	
C_{N_T}	-.00495		-.00495		-.00495	
δr	-3.67		-3.8		-3.75	
q	92.5		92.5		92.5	
q_{vt}/q *	1.055		1.055		1.055	
q_{vt}/q	1.00		1.00		1.00	
Initial α_{vt} *	0		0		0	
Initial α_{vt}	0		0		0	
Gust	+	-	+	-	+	-
Effective α_{vt} *	+6.1	-6.1	+6.1	-6.1	+6.1	-6.1
Effective $_{vt}$	+6.1	-6.1	+6.1	-6.1	+6.1	-6.1
$\eta_{vt}\ C_{C_{VT}}$ *	+.182	-.331	+.180	-.333	+.181	-.332
$\eta_{vt}\ C_{C_{VT}}$	+.182	-.331	+.180	-.333	+.181	-.332
L_{vt}, lbs.*	+1770	-3220	+1750	-3240	+1760	-3235
L_{vt}, lbs.	+1676	-3045	+1660	-3070	+1670	-3065

* Indicates vertical tail in slipstream of operating engine

PART III-D

Vertical Tail Loads for 5 Degrees Yaw at $V\sigma^{\frac{1}{2}}$ = 313 mph

C. G. Location	1 and 3	2 and 4	5
$C_{N_{A-T}}$	.00497	.00518	.00503
$C_{N_{VT}}$	-.00497	-.00518	-.00503
q #/ft.^{2}	250	250	250
q_{vt}/q	1.016	1.016	1.016
δ_r	-5.3	-5.0	-5.18
α_{vt}	4.334	4.334	4.334
$\eta_{vt}\ C_{C_{VT}}$	.0748	.0810	.0775
L_{VT}, lbs.	1880	2040	1950

Vertical Tail Loads with Zero Rudder Deflection at 5 Degrees Yaw At $V\sigma^{\frac{1}{2}}$ = 313 mph

$C_{N_{A-T}}$	.00497	.00518	.00503
C_{N_T}	-.01194	-.01161	-.01182
C_{N_A}	-.00697	-.00643	-.00679
q	250	250	250
q_{vt}/q	1.016	1.016	1.016
δ_r	0	0	0
α_{vt}	4.334	4.334	4.334
$\eta_{vt}\ C_{C_{VT}}$	.1820	.1820	.1820
L_{VT}, lbs.	4580	4580	4580

Subject: BASIC FLIGHT CRITERIA - PACK ON

PART III-D

TABULATION OF VERTICAL TAIL LOADS

CONDITION	ψ	C.G.	ENG. OPER.	$V\sigma^{\frac{1}{2}}$	q #/□	$\frac{q_{vc}}{q}$ *	$\frac{q_{vt}}{q}$	δ_r	α_{vc} DEG. *	α_{vt} DEG.	L_{VT} LBS. *	L_{VT} LBS.
Side gust 50 fps at 250 mph with initial $\alpha_{vt}=0$	0	ALL	R&L	250	160	1.024		0	7.8		5346	
Zero yaw, one engine dead, other delivering Military power	0	1-3	R	100	25.6	1.44	1.00	-14.95	0	0	-1105	-768
	0	2-4	R	100	25.6	1.44	1.00	-15.35	0	0	-1135	-788
	0	5	R	100	25.6	1.44	1.00	-15.1	0	0	-1115	-774
	0	1-3	R	160	65.5	1.103	1.00	-5.76	0	0	-835	-755
	0	2-4	R	160	65.5	1.103	1.00	-5.9	0	0	-856	-773
	0	5	R	160	65.5	1.103	1.00	-5.89	0	0	-855	-772
	0	1-3	R	185	87.5	1.06	1.00	-7.90	0	0	-725	-684
	0	2-4	R	185	87.5	1.06	1.00	-4.0	0	0	-743	-702
	0	5	R	185	87.5	1.06	1.00	-3.47	0	0	-740	-699
	0	1-3	R	250	160	1.023	1.00	-1.55	0	0	-510	-498
	0	2-4	R	250	160	1.023	1.00	-1.6	0	0	-526	-573
	0	5	R	250	160	1.023	1.00	-1.58	0	0	-518	-506
One eng dead, other delivering Military Power; + gust 30 fps	0	1-3	R	190	92.5	1.055	1.00	-3.67	+6.1	+6.1	+1770	+1676
	0	2-4	R	190	92.5	1.055	1.00	-3.80	+6.1	+6.1	+1750	+1660
	0	5	R	190	92.5	1.055	1.00	-3.75	+6.1	+6.1	+1760	+1670
- gust 30 fps	0	1-3	R	190	92.5	1.055	1.00	-3.67	-6.1	-6.1	-3220	-3085
	0	2-4	R	190	92.5	1.055	1.00	-3.80	-6.1	-6.1	-3240	-3070
	0	5	R	190	92.5	1.055	1.00	-3.75	-6.1	-6.1	-3235	-3065
5° yaw with δ_r = As Noted 313 mph.	5	1-3	R&L	313	250	1.016		-5.3	4.334	4.334	+1880	
	5	2-4	R&L	313	250	1.016		-5.0	4.334	4.334	+2040	
	5	5	R&L	313	250	1.016		-5.18	4.334	4.334	+1950	
5° yaw with $\delta_r = 0$ 313 mph	5	ALL	R&L	313	250	1.016		0	4.334	4.334	4582	

* Indicates tail in slipstream of propeller

PART III-E

E. Rudder Tab Loads

Rudder tab loads were determined for maximum tab deflections at V_D (313 mph).

For both rudder trim tab and rudder spring tab the loads as determined are maximum loads. For either tab the loads did not exceed the minimum specified design load of reference (2).

It is therefore advisable to design to the minimum specification value of 50 lbs. per sq. ft. for each tab.

PART III-E-1

1. Rudder Trim Tab

Using the same load distribution for the rudder trim tab as for the elevator trim tab (loading as specified by reference (2)) the rudder trim tab loads are determined for maximum deflection at V_D (313 mph).

The rudder trim tab hinge moment is determined as follows:

$$H_{rtt} = \frac{d\,C_{H_{rtt}}}{d\,\delta_{rtt}}\ \delta_{rtt} \times q\ \frac{q_{vt}}{q} \times S_{rtt} \times C_{rtt}$$

$S_{rtt} = 3.01$ sq. ft. $\qquad c_{rtt} = .673$ ft.

$\delta_{rtt\ maximum} = 15^\circ \pm 2^\circ$ page 17 reference (1)

$\frac{d\,C_{H_{rtt}}}{d\,\delta_{rtt}} = -.0043$ per degree page 246 reference (1)

$q = 248 \qquad \frac{q_{vt}}{q} = 1.015$ figure 65 reference (1)

$$H_{rtt} = -.0043 \times 17^\circ \times 248.0 \times 1.015 \times 3.01 \times .673$$

$$H_{rtt} = -37.5 \text{ ft. lbs.}$$

$$N_{rtt} = \frac{-H_{rtt}}{-(4/q \times .673)} = -3.34 \times H_{rtt}$$

$$N_{rtt} = -3.34 \times -37.5$$
$$= 125 \text{ lbs.}$$

$$\frac{N_{rtt}}{S_{rtt}} = \frac{125}{3.01} = 41.5 \text{ lbs. per sq. ft. at trim tab area}$$

PART III-E-2

2. Rudder Spring Tab

The load on the rudder spring tab is determined for the same condition of maximum deflection and V_D (313 mph).

The rudder spring tab hinge moment is determined as follows:

$$H_{r_{st}} = \frac{d\,C_{H_{r_{st}}}}{d\,\delta_{r_{st}}}\ \delta_{r_{st}} \times q \times \frac{q_{vt}}{q} \times S_{r_{st}} \times C_{r_{st}}$$

$S_{r_{st}}$ = 1.66 sq. ft. $\quad C_{r_{st}}$ = .673 ft.

$\delta_{r_{st}\ maximum}$ = 17.5° ± 2° page 17 reference (1)

$\frac{d\,C_{H_{r_{st}}}}{d\,\delta_{r_{st}}}$ = -.0043 per degree page 246 reference (1)

q = 248.0 $\quad \frac{q_{vt}}{q}$ = 1.015 figure 65 reference (1)

$H_{r_{st}}$ = -.0043 x 19.5 x 248.0 x 1.015 x 1.66 x .673

$H_{r_{st}}$ = -13.42 ft. lbs.

$N_{r_{st}}$ = -3.34 x -13.42

$N_{r_{st}}$ = 45 lbs.

$\frac{N_{r_{st}}}{S_{r_{st}}} = \frac{45}{1.66}$ = <u>27.1 lbs. per sq. ft. of spring tab area</u>

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PART IV -

ASSYMETRICAL FLIGHT - ROLL

The rolling moments, rates of roll and all wing and aileron loads will be the same as those obtained in reference (5) Part IV.

This is a consequence of the similarity of the XC-120 and C-119B wing and aileron configurations. In addition, the critical speeds and associated rates of roll are the same for the two airplanes. The differences in configuration of the two airplanes will not affect any of the lateral stability derivatives nor the aileron power. Only the value of the angle of attack of the wing for a given airplane lift coefficient will change, due primarily to the change in wing incidence and airplane configuration. This change is only

$$\Delta \alpha_c \cong .10^\circ$$

which would affect only the aileron loads, and because of its small value, it may be neglected.

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AD-B803 064

FAIRCHILD ENGINE AND AIRPLANE CORP., FAIRCHILD AIRCRAFT DIV., HAGERSTOWN, MD. (ENGINEERING REPORT NO. R107-014)

BASIC FLIGHT CRITERIA - PACK ON - MODEL XC-120 (M-107)

A. ATHANASIAN; J. POOLE; A. GRENIS AND OTHERS 15 MARCH'49
360PP. GRAPHS, DRWGS

EO 10501 dd 5 NOV 1953

STRUCTURES (7) LOADS, CRITICAL
LOADS AND CRITERIA (1) AIRPLANES - AERODYNAMICS LOADING
C-120 Aircraft

P1/3,5

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* Transport Aircraft

DEFENSE TECHNICAL INFORMATION CENTER

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11. AD NUMBER (If Known): AD-B803064

12. (TITLE, REPORT NUMBER, AUTHOR(S)), ETC

Basic Flight Criteria - Pack On - Model XC-120 (M-107), 360 pages

SECTION III REQUESTER JUSTIFICATION

13. REQUESTER JUSTIFICATION (Explain need in detail)...

A paper copy of this unclassified document, currently releasable to DOD only, is being requested for administrative use in a preliminary project study. NASA-DFRC engineers are looking at several possible concepts for future Unmanned Aerial Vehicle applications. This historical document may help determine a future NASA REVCON (Revolutionary Concept) proposal to be engineered at this Center.

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